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EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 3: DESIGN/COST TRADEOFF STUDIES

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EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 3: DESIGN/COST TRADEOFF STUDIES

Prepared For

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABBREVIATIONS AND ACRONYMS

ACM	Attitude Control Module
ACS	Attitude Control Subsystem
AGE	Aerospace Ground Equipment
AOP	Advanced On-Board Processor
ARC	Absolute Radiometric Calibration
ATS-F	Applications Technology Satellite F
BER	Bit Error Rate
B/L	Baseline
BRM	Branch and Mark Return Location
BST	Boresighted Star Tracker
BPSK	Biphase Shift Keying
CC	Control Center
CCIR	Comite Consultalif International Radio
C&DH	Communications & Data Handling
CDP	Central Data Processing
CERS	Cost Estimating Relationships
CIPS	Conical Image Plane Scanner
CMD	Command
CMD/TLM	Command/Telemetry
CPF	Central Processing Facility
CRT	Cathode Ray Tube
CSC	Computer Sciences Corporation
C&W	Caution & Warning
DCS	Data Collection System
DDT&E	Design Development Test and Evaluation
DHG	Data Handling Group
DMA	Direct Memory Access
DMS	Data Management System(s)
DNTD	Descending Node Time-Of-Day
DOD	Department of Defense
DOMSAT	Domestic Satellite
DOS	Data Operating Supervision

DPS	Data Processing System
EBR	Electron Beam Recorders
ELMS	Earth Limb Measurements Satellite
EOS	Earth Observatory Satellite
EPS	Electrical Power Subsystem
ETR	Eastern Test Range
EVA	Extra-Vehicular Activity
FEP	Front End Processor
FHT	Fixed Head Tracker
FMEA	Failure Mode Effects Analysis
FOM	Figure-of-Merit
FSK	Frequency Shift Keying
FSS	Flight Support System
GAC	Grumman Aerospace Corporation
GC	Ground Controller
GCP	Ground Control Points
GN_2	Gaseous Nitrogen
GLS	Ground Logistics System
GPS	Ground Processing System
GSE	Ground Support Equipment
HRPI	High Resolution Pointable Imager
IDA	International Data Acquisition
IMPATT	Impact-Avalance and Transit Time
IMS	Information Management System
I/O	Input Output
IVA	Intra Vehicular Activity
JBS	Japanese Broadcast Satellite
LBR	Laser Beam Recorder
LCC	Life Cycle Cost
LCGS	Low Cost Ground System
LIPS	Linear Image Plane Scanner
LM	Lunar Module
LOPS	Linear Object Plane Scanner
LRM	Land Resource Management
LSA	Limited Space Charge Accumulation
L/V	Launch Vehicle
MBPS	Megabits Per Second

MEM	Module Exchange Mechanism
MEM	Multiplexer/Encoder Module
MIB	Minimum Impulse Bit
MISCON	Mission Control
MLI	Multi Layer Insulation
MMD	Mean Mission Duration
MMH	Monomethyl Hydrazine
MOCC	Mission Operations Control Center
MOMS	Multi-Megabit-Operational Multiplexer System
MSS	Multi Spectral Scanner
MUS	Magnetic Unloading System
MUX	Multiplexer
NASCOM	NASA Communications
N_2H_4	Hydrazine
N_2O_4	Nitrogen Tetroxide
N/R	Non-Recurring (cost)
NTTF	NASA Test & Training Facility
NUS	No Upper Stage
OAO	Orbiting Astronomical Observatory
OAS	Orbit Adjust Subsystem
OBC	On Board Computer
OBP	On-Board Processor (used on OAO)
OBDC	On-Board Data Compaction
OCC	Operations Control Center
OMS	Orbit Maneuvering Subsystem
OPS	Operations
OTS	Orbit Transfer Subsystem
OWS	Orbital Workshop (Skylab)
OSR	Optical Solar Reflector Coating
PCM	Pulse Code Modulation
PCU	Power Control Unit
PDSS	Precision Digital Sun Sensor
PFD	Power Flux Density
PGST	Precision Gimballed Star Tracker
P/L	Payload

PMMR	Passive Multichannel Microwave Radiometer
POC	Project Operations Controller
PRN	Pseudo Random Noise
PROD	Production
PROM	Program Read Only Memory
PRU	Power Regulation Unit
PSK	Phase Shift Keying
PSM	Power Supply Module
QPSK	Quadraphase Shift Keying
RCS	Reaction Control Subsystem
REL	Reliability
RF	Radio Frequency
RGA	Rate Gyro Assembly
ROM	Read Only Memory
R&QA	Reliability & Quality Assurance
RS	Resupply System
RTS	Remote Tracking Site
SAMS	Shuttle Attached Manipulator System
SAMSO	Space and Missile Systems Organization
SAR	Synthetic Aperture Radar
S/C	Spacecraft
SCO	Sub-Carrier Oscillation
SCPS	Support Computer Program Systems
SEASAT	Sea Satellite
SEOS	Synchronous "EOS"
SM	Subsystem Module
SMBC	Shared Memory Bus Control
SMBI	Shared Memory Bus Interface
SMS	Synchronous Meterological Satellite
SNR	Signal-to-Noise Ratio
S/O	Shut-off
SOW	Statement of Work
SRM	Solid Rocket Motor
SSR	Scanning Spectral Radiometer
STAB	Space Transportation & Budget (Computer Program)

STE	Special Test Equipment
STDN	Space Tracking Data Network
TDRS	Tracking & Data Relay Satellite
TEA	Transferred Electron Amplifier
TEO	Transferred Electron Oscillator
TIIID	Titan IIID
T&IS	Test & Integration Station
TM	Thematic Mapper
TRAPATT	Trapped-Plasma-Avalance Triggered Transit
SMM	Solar Maximum Mission
TT&C	Telemetry, Tracking and Command
TULIP	Tug Life Cycle Cost Program (Computer Program)
TWTA	Traveling Wave Tube Amplifier
U/R	Unit Recurring (cost)
WBS	Work Breakdown Structure
WBVTR	Wide Band Video Tape Recorder
WTR	Western Test Range
VCHP	Variable Conductance Heat Pipe

INTRODUCTION

1- INTRODUCTION

The key issues in the EOS program that are subject to configuration study and tradeoff are:

- Design, cost, and cost benefits of a standardized, modular basic spacecraft having flexibility to accommodate a broad range of missions.
- Orbit, Spacecraft, Instrument and Data Management System approach leading to an operational Land and Water Resources Management capability at the lowest total program cost.
- Technical and programmatic relationship between operational and R&D segments of the EOS program.
- Data Management System configuration that strikes the best balance between initial configuration and subsequent growth to accommodate new missions and new technology.
- Approach to Shuttle utilization to enhance system effectiveness and reduce cost.
- Management approach to be used for a low cost EOS program.

The configuration studies and trades reported in this volume deal with these issues and have led to some interesting conclusions. The issue of a combined operational and R&D EOS program has been explored to a considerable degree. Further examination is needed however, since this is a relatively new issue in the study.

We consider cost and spacecraft weight to be key design variables throughout our study. We have portrayed all design options in terms of these parameters wherever possible.

We have performed detailed costing for the basic spacecraft and for the EOS-A and A' program. All costs will be subject to possible re-targetting with NASA as part of our design-to-cost approach. We do, however, recommend herein what we feel are reasonable target costs for the recurring basic spacecraft and for the EOS-A and A' total program.

KEY STUDY
CONCLUSIONS

2 - KEY STUDY CONCLUSIONS

The key conclusions resulting from our design/cost trades conducted during the first three months of study are given below. A few of these conclusions, so noted, are subject to further evaluation during the remainder of the system definition study and will be modified if necessary in the final report.

2.1 MISSION MODEL

The Goddard EOS mission model, shown in Fig. 2.1-1, has been evaluated in terms of accommodation by a basic spacecraft, data management system design concepts, and operational considerations.

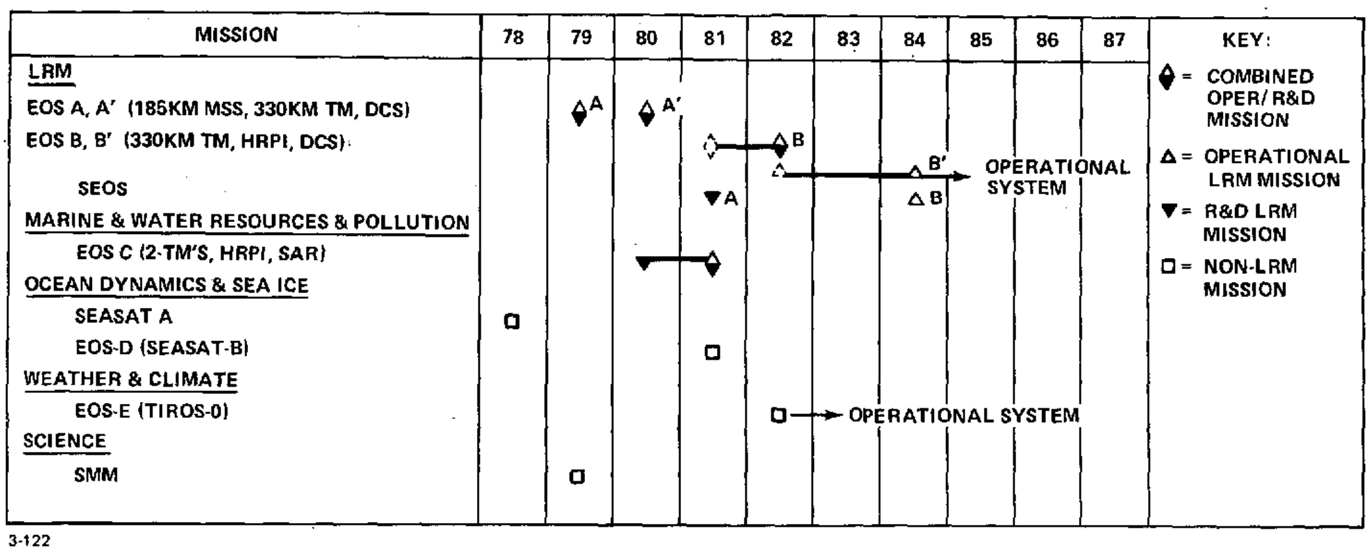


Fig. 2.1-1 EOS Mission Model

The mission model is felt to be a good typical representation of an EOS "family" of remote sensing missions extending into the 1980's.

It is interesting and challenging in that it embodies: a) a broad range of instruments, b) a range of launch vehicle requirements including transition to Space Shuttle, and c) combined operational/R&D missions. Our chief conclusions regarding this mission model are:

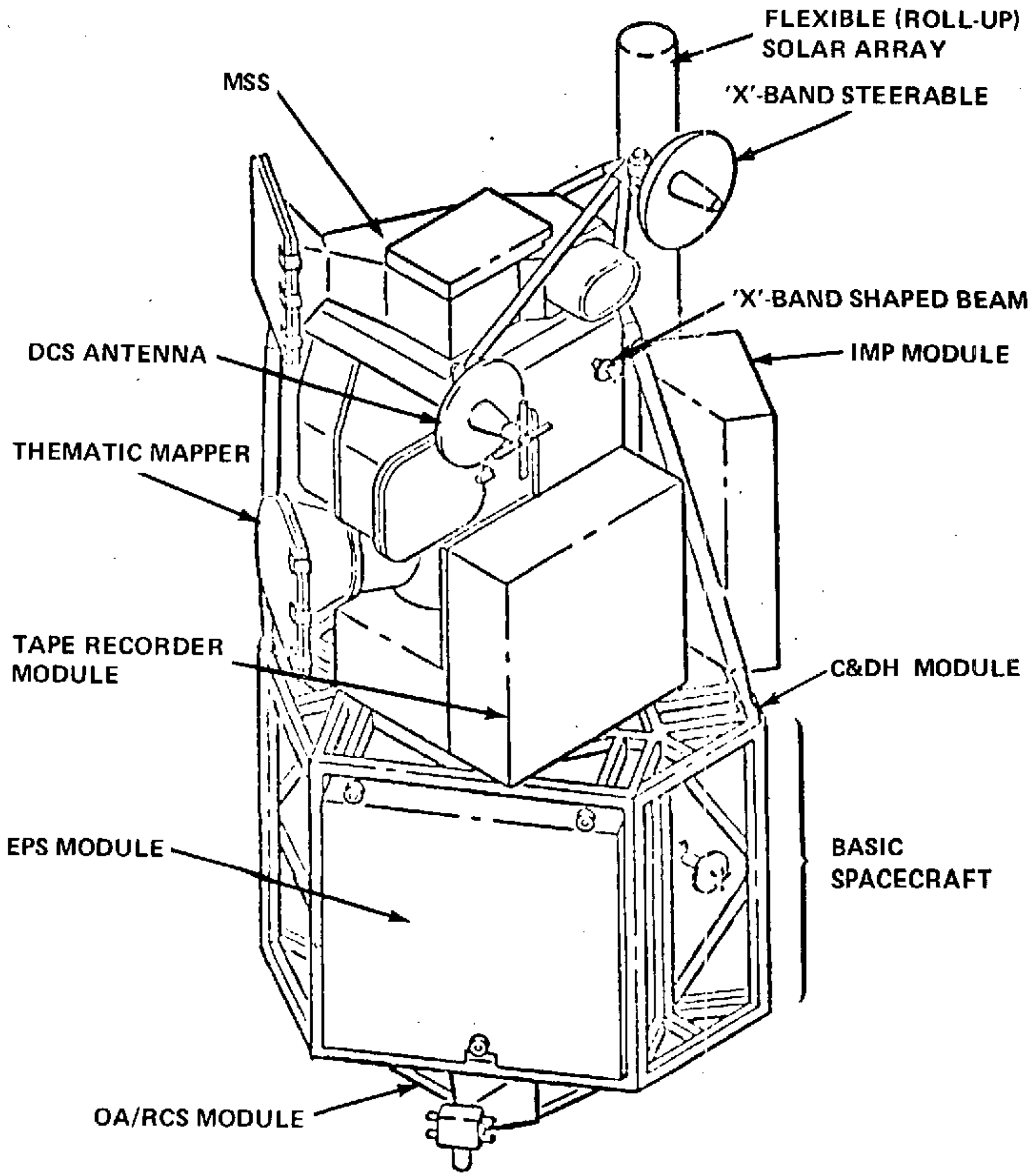
- A single basic spacecraft can accommodate the range of instruments (missions) and launch vehicles represented by the model
- A significant cost saving over dedicated spacecraft is anticipated by employing a standard basic spacecraft for the range of missions
- Evolution of the Data Management System is a major driver on the mission model for land and water resources missions requiring ground data processing of wide band imagery. Slippage of EOS-B and -B' as shown may be necessary to allow a more moderate build-up and demonstration of DMS capability.
- Deployment of two or more satellites in orbit for a typical land resources mission is more costly than using a single satellite. There may, however, be operational advantages to using multiple satellites. Further study is needed to identify and evaluate these advantages.

2.2 BASIC SPACECRAFT

The basic spacecraft configuration, shown in Fig. 2.2-1 with EOS-A mission peculiars, embodies three major features.

- The basic spacecraft can be used for a wide variety of missions with large payload capability. These include:
  - Earth Pointing
  - Stellar
  - Inertial
  - Solar Pointing
  - Geosynchronous
- The design involves no technical development issues. Design features include:
  - A structural configuration, including shuttle resupply, which provides simple, straight forward and low cost design, analysis, and manufacturing concepts
  - Subsystems configurations that utilize well proven hardware or concepts demonstrated on other satellite programs. The data bus command and telemetry unit is the only new development. The major controllers, sensors, actuators, and components used in the ACS, Power, and Communications. Data Handling modules are existing or modifications of "off the shelf" equipment being used on other programs
  - A spacecraft thermal control approach that utilizes well proven, easily analyzed, low cost techniques and which we feel is not a development issue
- The design is adaptable to a wide variety of launch vehicles. We have defined the spacecraft design requirements for the viable launch and retrieval systems. These include Delta 2910 and 3910, Titan III C and shuttle deploy and retrieve. None of these present an environment outside the space craft capability. Interface adapters are simple and inexpensive. The use of Atlas was not investigated in detail but could represent a viable launch system.

In summary, the basic spacecraft defined in this report, represents a low cost, low risk vehicle with flexibility to capture a wide variety of satellite missions using existing and projected launch systems, and making full utilization of the Space Shuttle's deploy, retrieve and resupply/repair capability.



BASIC SPACECRAFT CHARACTERISTICS

• WEIGHT:	1361 LB
• SIZE:	7 FT DIAM X 5 FT
• LAUNCH VEHICLE:	DELTA, TITAN, ATLAS OR SHUTTLE
• ORBIT:	ANY; THERMAL MODIFICATIONS NEEDED
• PAYLOAD CAPABILITY:	1000 LB (DELTA CONFIG) 4000 LB (TITAN CONFIG)
• MODULAR SUBSYSTEMS:	ACS, EPS, CDH, OAS/RCS
• ATTITUDE CONTROL:	0.01 DEG POINTING 10^{-6} DEG/SEC STABILITY
• ELECTRICAL POWER:	1200 WATTS (ORBIT AVG) 3500 WATTS (PEAK)
• DESIGN LIFE:	TWO-YR OPERATION 5-YR SURVIVAL FOUR-YR OPERATION WITH ADDED REDUNDANCY
• IN-ORBIT RETRIEVAL/RESUPPLY:	OPTIONAL HARD POINTS FOR RETRIEVAL (27 LB) OPTIONAL LATCH MECHANISM FOR RESUPPLY (52 LB)
• ORBIT ADJUST/TRANSFER	OPTIONAL MODULE

Fig. 2.2-1 EOS-A and A'

2.3 EOS-A, A' PROGRAM COST

The total program cost for the EOS-A and -A' program is estimated at $161.1M. This cost includes:

- Observatory design, development and qualification
- Two flight spacecraft including instruments, and component-level spares
- Two years of flight and ground operations for each spacecraft with a one-year overlap
- Launch vehicles (Delta 2910) and launch costs
- Project Control Center design, build and operations costs
- R&D and operational, flight and co-located Ground Data Management System design, build, integration and operations
- Network modifications
- Low-cost management approaches, including moderate simplification of test, documentation, and controls, use of a System Integrator Team for project management, and a design to cost approach.

A Design to Cost target for the A and A' program of $150M* is recommended as a reasonable goal. The delta between the identified program cost of $161.1M* and recommend target of $150M, 11.1M, applies only to the Observatory and Ground Data Management elements of the EOS program since the Instrument and Launch cost which represent $60.25M were, by NASA definition, considered as fixed costs. It is recommended that the design to cost targets for the Observatory and Ground Data Management systems be treated as a design requirement and incorporated into the basic specifications.

Program costs were also analyzed in terms of those program costs attributed to the operational mission performed by the MSS vs those program costs incurred by the R&D elements. These costs are shown on Table 2.3-1. Table 2.3-2 presents a representative distribution of costs incurred versus fiscal year, assuming a program start of mid-calendar '76, launch of EOS-A in 4/79 and EOS-A' in 4/80, and two years of on-orbit operations for each observatory.

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*1974 constant dollars, in millions

Table 2.3-1, EOS-A and A' Program Costs

	NONRECURRING	RECURRING	TOTAL
o FIXED COSTS – INSTRUMENTS			$43.0M
– TM (2)	($13.0M)	($14.0M)	
– MSS (2)	($ 1.0M)	($12.0M)	
– DCS (2)	($ 2.0M)	($ 1.0M)	
– LAUNCH COSTS (2)	($ 0.250M)	($17.0M)	17.25M
o OPERATIONAL SYS. COSTS			($20.71M)
– MSS IMP (2)	($ 1.01M)	($ 4.58M)	
– GND DMS	($11.68M)	($ 3.44M)	
– NETWORK	(–)	(UNKNOWN)	
o R & D SYS COSTS			($32.06M)
– TM IMP	($ 4.40M)	($ 2.82M)	
– GND DMS	($11.91M)	($ 8.88M)	
– NETWORK	($ 2.73M)	($ 1.32M)	
o SPACECRAFT			($38.46M)
– BASIC SPACECRAFT (2)	($18.32M)	($12.57M)	
– M.P. SPACECRAFT (2)	($2.67M)	($ 3.28M)	
– SPARES & LOGISTICS	($0.41M)	($ 1.21M)	($ 9.63)
o MISSION OPS	($ 4.73M)	($ 4.90M)	($ 9.63M)
			TOTAL ($161.11M)

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Table 2.3-2 EOS-A and A' Program Funding Summary

	FY'77	FY'78	FY'79	FY'80	FY'81	FY'82	TOTAL
DATA MGT. SYSTEM	$ 6.3	$14.9	$ 8.5	$ 4.2	$ 3.9	$2.1	$39.9
INSTRUMENTS	6.9	20.3	14.6	1.2	–	–	43.0
FLIGHT OPERATIONS	.3	1.0	4.3	1.9	1.2	.9	9.6
LAUNCH SYSTEM	.1	1.9	10.6	4.7			17.3
SPACECRAFT PROJECT	10.3	16.8	17.9	5.7	.4	.2	51.3
TOTAL PROGRAM	$23.9	$54.9	$55.9	$17.7	$ 5.5	$3.2	$161.1

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2.4 BASIC SPACECRAFT COST

The total recurring cost for the basic spacecraft of \$6.3M*, shown on Table 2.4-1, represents the cost required to produce a basic spacecraft including solar array and RCS, ready for integration of the mission peculiars. It includes:

1) All manufacturing engineering and management manpower required to,

- Build and wire the basic spacecraft and module structure
- Procure and integrate the subsystem components into the modules
- Integrated and the basic system software
- Functionally and environmentally acceptance test the subsystem modules
- Integrate the subsystem modules in to the basic spacecraft and functionally verify the spacecraft performance.

2) Procurement costs of all the hardware including total arrays and RCS for the EOS A or A' basic spacecraft.

Table 2.4-1 Basic Spacecraft

	NONRECURRING COST, M	RECURRING COST, M
PROG. MGMT.	\$ 1.58	.424
SYS ENG. & INT.	.80	.400
R & QA	.72	.320
I & T	.29	.240
DEV TEST	2.40	
GSE	2.31	
STRUCTURE, ADAPTER, ETC.	1.80	.599
EPS	1.11	.780
SOLAR ARRAY & DRIVE	.66	.755
CDH	2.93	1.138
ACS	2.37	1.160
RCS	.57	.471
O/B SOFTWARE	.80	
	\$18.32M	\$6.285M

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A design to cost target of \$5.5M* is recommended for the EOS A OR A' basic spacecraft. Based on this study, it is also concluded that potential basic spacecraft recurring costs of \$5.0M could be targeted for a multiple buy (5 or more) basic spacecraft procurement.

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*1974 constant dollars, in millions

2.5 ORBITAL/LAUNCH VEHICLE TRADE STUDY

The recommended orbit for the EOS mission should be sun synchronous with an altitude between 365 and 385 n mi. This range of altitudes has acceptable orbit decay, swath sideslip and ground station coverage (Table 2.5-1). The 365-n mi orbit was evaluated for orbit decay and was found to be operationally acceptable; the node sideslip at the equator was ± 2.1 n mi in 30 days (assuming a 1979 nominal atmosphere). Ground coverage from Sioux Falls of orbits within the recommended range yields complete CONUS coverage (with a 2° horizon mask).

Table 2.5-1 Orbital/Launch Vehicle Trade Study

ATMOSPHERIC DRAG EFFECTS: (MID-1979 LAUNCH)

ORBIT	ATMOSPHERE	30 DAY SIDESLIP	60 DAY SIDESLIP	90 DAY SIDESLIP
366 N.M.	NOMINAL	± 2.1 N.M.	± 17.3 N.M.	± 38.8 N.M.
366 N.M.	NOMINAL + 2 σ	± 13.0 N.M.	± 52.0 N.M.	± 116.0 N.M.

SWATH/REVISIT RELATION:

ORBIT	NO. OF SAT'S	REVISIT TIME	SWATH (37 KM OVERLAP)
366 N.M.	1	17 DAY	185 KM
382 N.M.	1	9 DAY	330 KM
366 N.M.	2	9 DAY	185 KM

WIDE BAND COMM COVERAGE:

- FULL CONUS COVERAGE FROM SIOUX FALLS USING 2° MAX. HORIZON MASK
- FOR 5° MAX. HORIZON MASK, LOSE SOUTH TIP OF FLORIDA

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A promising orbit for the EOS is 366 n mi when using an instrument with a 100-n mi swath width. This orbit has a 17-day repeat cycle and a 14-n mi swath overlap. The adjacent swath overlap occurs in three days.

Initial deployment of the EOS class of satellites can be accomplished using four types of conventional launch vehicles. The Delta 2910, Delta 3910, and Titan III B (SSB) are used to deliver EOS satellites which have sun-synchronous and polar orbits. The Titan III C7 is used to deploy the EOS-F to its geosynchronous equational mission orbit. The range of orbits is also compatible with Shuttle capability.

2.6 RECOMMENDED THEMATIC MAPPER/DATA PROCESSOR DESIGN

Our recommended TM/DP design features are summarized as follows:

- 330 km swath (680-km orbit alt)
- 30-meter resolution output over 185 Mbs link
- MSS-compatible output for MSS backup
- Pseudo-HRPI output for local users
- Linear, object plane scanning
- Digital TM-DP interface
- Cooled solid state detectors
- 90% commonality with an HRPI.

To achieve a repeat cycle of 8 days, a TM swath width of 330 km is recommended. This appears to be a significant performance/cost improvement since the cost differential over a 185-km/17-day repeat instrument is less than 10%.

The wide-swath system can still provide 30-meter performance over the wideband link by using both of the two quadrature channels. This performance employs a 27-meter pixel size which is a submultiple of the MSS pixel of 54 x 81 meters. Thus, it is easy for the on board processor to generate an MSS simulation output which is indistinguishable from the normal MSS signal.

For follow-on mission planning, it is also possible to obtain from the on-board processor a HRPI-like signal (at 30 meters resolution) consisting of a selectable 40-km swath from the full 330-km swath.

A preferred design has been used in our configurations which consists of a telescope with its axis aligned to the flight vector and a scanning mirror providing an object plane scan. Such a design is believed to be the simplest, lightest and capable of meeting all performance requirements. This design also employs cooled solid state array sensors in the focal plane which are capable of providing a considerably higher S/N performance than in the instrument point designs in order to meet the stringent radiometric accuracy requirements of the users.

The output of the instruments has been specified as digital in order to simplify as much as possible the engineering effort required at the interface to avoid noise and distortion of the signals.

A TM design has evolved which could share almost 90% of its parts and design effort with a later HRPI if this long term goal is emphasized at the start of the TM design effort.

During the design study, three areas of significant cost savings were recognized with regard to the instruments. They are listed in Table 2.6-1.

Table 2.6-1 TM Cost Saving Features

DESIGN	COST SAVING
COOLED SOLID STATE DETECTORS	$0.25 MILLION (R)
LINEAR SCANNING	$36/SCENE*
HRPI COMMONALITY	$8 MILLION (NR)

*100% INCREASE FOR NEAREST NEIGHBOR GEOM. CORR. ALGORITHM, 40% INCREASE FOR BI-LINEAR, 18% FOR CUBIC

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In the design of the instrument, a significant cost saving can be accomplished by switching from photomultipler tube detectors to solid state detectors (with a significant improvement in performance if two-dimensional sensor arrays are used). This saving is partially due to the obsolescence of photomultipler tubes, resulting in increasing costs as volume shrinks.

By employing a 27 x 27-meter pixel size, and linear scanning, a saving of about $36 per picture can be achieved over that of the earlier point design. This saving amounts to more than $2 million per year in data processing costs for just the continental US imagery. It could be much greater if significant international usage is involved.

As mentioned earlier, if provided for in the original TM procurement, it is possible to acquire a HRPI at a later date exhibiting a high degree of commonality with the TM. This could result in a significant one time savings in design, system integration and program management.

2.7 DATA MANAGEMENT SYSTEM

Table 2.7-1 shows a breakdown of the costs of the Data Management System for Missions A and A'. Included in the costs is a Control Data Processing Facility (CDAF) sized for 20 TM scenes per day. The costs are separated into the nonrecurring (initial investment) and recurring (annual O&M, data processing expendables, etc.). The largest cost components of the Data Management System are due to the CDPF. Within the CDPF non-recurring costs, the basic drivers are the data processing equipment and computer software. The other major non-recurring categories have been calculated on the basis of rule-of-thumb percentages of the hardware and software costs. The annual costs are driven by the daily rates at which data is copied in various media and formats for distribution to users. These costs are based on a rather modest production rate, with the assumption that any intensive copying and distribution will occur in a later phase and/or a separate facility.

Table 2.7-1 Mission A and A' DMS Cost Breakdown

	NONRECURRING, M	RECURRING, M
CENTRAL DATA PROCESSING		
FACILITIES	$ .07	$ .05
INT & TEST	1.210	–
DATA PROCESSING EQUIPMENT	6.206	–
COMPUTER SOFTWARE	1.729	–
DOCUMENTATION	.931	–
ENGINEERING & MANAGEMENT	1.678	–
OPERATIONS	–	1.60
EXPENDABLES		.90
LOGISTICS		0.60
NETWORK MODIFICATIONS	2.73	–
NETWORK OPERATIONS	–	0.19
NETWORK EXPENDABLES	–	0.25
TOTALS	$ 14.544	$ 3.59

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The costs are based on a processing facility configured with flexible processing modules such as mini-computers. A flexible system during the early R&D stages of the EOS permits economical development and change of the processing algorithms and other CDPF functions. Once these have been finalized and accepted by the user community, expansion of the CDPF throughput capability can be accomplished using high speed special processors. Thus, the CDPF philosophy for an economical transition from early R&D with limited throughput to a processing system with higher throughput should be:

- Initial configuration which utilizes flexible processing modules
- Expanded capability accomplished by adding high speed special processors.

2.8 INSTRUMENT DATA COMPACTION

Table 2.8-1 indicates data compaction options for the TM and HRPI. These options permit the sending of TM and/or HRPI data, in modified form, to the Local User Station as direct transmission from the EOS. Compaction is required since the Local User link has an upper bound data rate of 20 Mbps. This corresponds to a rate slightly above the 16 Mbps of the existing MSS. Three compaction approaches are presented:

- Band Selection - The sending of one or more of the spectral or IR bands to the LCGS at rates up to 20 Mbps with full, or reduced resolution
- Resolution Reduction - The combination of adjacent pixel data of the instruments into a larger pixel size so that the resulting bit rate is not greater than 20 Mbps. This can also be for all bands or only selected combinations of bands, depending on resolution reduction
- Partial coverage - The sending of a partial scene, or partial swath, to the Local User at full or reduced resolution in either selected, or all, bands.

Table 2.8-1 TM and HRPI Compaction Options

	NUMBER BANDS	RESOLUTION	% SWATH	DATA RATE (MBPS)
330 KM TM (WIDE SWATH)	ALL	FULL	100	130
	ONE	FULL	100	18.5
	FOUR	HALF	100	18.4
	ALL	FOURTH	100	17.3
	TWO	HALF	100	
	ALL	HALF	56	18.2
	THREE	FULL	34	18.5
HRPI	ALL	FULL	100	130
	ONE	FULL	56	18.2
	TWO	HALF	100	16.2
TM + HRPI	ALL TM	FOURTH	100	16.2
	ONE HRPI	HALF	100	

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The compaction options shown in Table 2.8-1 are based on a 330-km (wide swath) TM and a 48-km HRPI. The compaction options for a 185-km TM swath are similar. With a basic data rate of 20 Mbps to the LBS, a narrower TM swath will permit a different combination of bands, resolution, or partial swath to be transmitted. Partial coverage requires a buffer to smooth the burst-type data gathering into a continuous output. With a 185-km TM, a total of 1.17 megabits are transmitted during each scan so that, for even the largest swath, a minimum of 305 kilobits of storage are required. For a 20% swath, for example, 970 kilobits are required.

2.9 ANTENNA SYSTEM FOR EOS TO LOW COST GROUND STATION COMMUNICATION

Two types of EOS antenna systems were investigated for transmitting instrument data to the low cost ground station (Table 2.9-1). These were:

- Alternative 1 - Steerable Antenna System
- Alternative 2 - Fixed Antenna System.

Table 2.9-1 EOS To LCGS Communications Alternatives

SYSTEM IMPACT / LINK OPTIONS	LCGS ALTERNATIVES: ALTERNATIVE 1 STEERABLE NARROW BEAM ANTENNA	ALTERNATIVE 2 FIXED WIDE BEAM ANTENNA	IMPACT EVALUATION
• WEIGHT, LB			FAVORS STEERABLE BEAM ANTENNA
ANTENNA	20	4	
P.A.	8	10	
TOTAL	*28*	*14*	
• SIZE			
ANTENNA	$1.5\ FT^3$ ($2500\ IN^3$)	$10\ IN^3$	FAVORS FIXED BEAM ANTENNA
P.A.	$310\ IN^3$	$330\ IN^3$	
TOTAL	*$2810\ IN^3$*	*$340\ IN^3$*	
• POWER (PRIME WATTS)			FAVORS STEERABLE BEAM ANTENNA
ANTENNA	10	0	
P.A.	14	172	
TOTAL	*24*	*172*	
• COST ($)			FAVORS STEERABLE BEAM ANTENNA
S/C ANTENNA	170K(2)	20K(3)	
S/C P.A.	60K(2)	100K(1)	
TOTAL	*230K*	*120K*	
• RISK	MODERATE (STEERABLE ANTENNA, BUT REDUNDANCY)	MODERATE (SINGLE THREAD 50W TWTA)	EQUAL

NOTES: (1) RECURRING COST ONLY: TUBE IS DEVELOPED.
(2) RECURRING COST ONLY: PRIMARY LINK PICKS UP NON-RECURRING COST.
(3) INCLUDES NON-RECURRING AND RECURRING.

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These alternatives are tabulated for the spacecraft since the ground station parameters are not affected. Significant characteristics for each alternative are:

- Steerable Antenna - volume of 1.5 cubic feet. Dish size is approximately one foot in diameter. Uses a small (14-watt) TWT power amplifier. Antenna gain is large.

- Fixed Antenna - small, lightweight unit. Can be of the horn or cavity-backed spiral type configuration to cover a 70° cone or a 500-km radius ground coverage. Nominal gain is 7 dB. Uses a relatively large (50 watts) TWT power amplifier requiring high (172 watts) prime power.

The fixed antenna approach is favored by size and cost considerations, while the steerable is favored by weight and power. The risk factors are judged to be about equal. Other considerations are operational: The steerable will allow service over a wider area (±50 degrees or more), which may be important for some users on some passes. However, a disadvantage is that the steerable needs to be steered toward the Local User to be served.

The nominal performance (S/N margin) is the same in both cases. However, in practice the steerable system may be operable at higher EIRP, allowing better margin or smaller ground stations.

On balance, there is no overwhelming advantage to either approach.

2.10 LOCAL USER SYSTEM/LOW COST GROUND STATION

A systems viewpoint was taken with respect to a wide family of Local User systems which includes the low-cost ground station concept. Centralized as well as local operations are necessary to assure system viability, and these operations have been considered.

The basic cost conclusions (Table 2.10-1) are that minimum (basic) capability LCGS's can be provided for an equipment (hardware) cost, in quantities of 10 or more, of $125K, and that the enhanced processor and display subsystems, increasing the hardware cost to about $300K in quantity, should provide as much local processing and analysis capabilities as most local area analysis specialists would need.

Table 2.10-1 Low-Cost Ground Station Costs

HARDWARE	CAPABILITIES	COST 10TH UNIT
1 – MINICOMPUTER 1 – DISK 2 – MAGNET TAPE 1 – CRT/KEYBOARD 1 – B&W DISPLAY 1 – DATA REPRODUCER	DISPLAY B&W IMAGES DATA PROCESSING (SLOW) IMAGE ANALYSIS (VERY SLOW) HARDCOPY (W/CAMERA)	$130K
ALL ABOVE PLUS: 1 – 2ND MINICOMPUTER 1 – LINE PRINTER 1 – COLOR DISPLAY 1 – HARDWARE X/:	DISPLAY B&W & COLOR DATA PROCESSING (MODERATE SPEED) IMAGE ANALYSIS (INTERACTIVE) HARDCOPY (W/CAMERA & PRINTER)	$223K
ALL ABOVE PLUS: 1 – 2ND DISK 2 – 3RD & 4TH MAGNETIC TAPE 1 – B&W & COLOR IMAGE RECORDER 1 – 2ND COLOR DISPLAY	DISPLAY B&W & 2 COLOR DATA PROCESSING (REASONABLE SPEED) IMAGE ANALYSIS (MODERATE SPEED) HARD COPY (PRINTER & PHOTO)	$300K

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In arriving at these design concepts, the following tradeoffs were considered:

- Three cost targets: $130K, $220K, and $300K for recurring (quantity 10 or more) hardware costs for LCGS LUS's that includes about $70K for the RF/IF and data handling/recording subsystems
- A single family of equipment
- RF/IF and data handling/recording subsystems common for all LCGS models
- Processor and display subsystem with modular software, expandable to meet a variety of user applications needs.

2.11 COST IMPACT OF TM SCAN TECHNIQUE, DATA LOAD, AND PROCESSING ALGORITHM

The trend of annual processing costs is a function of the number of scenes of TM data which are processed each day, scan technique, and processing algorithm. The scene load of primary concern ranges from 20 per day (approximately 4×10^{10} bits/day) to 400 per day (8×10^{11} bits/day). Over this range, costs increase linearly with scene load (Figure 2.11-1).

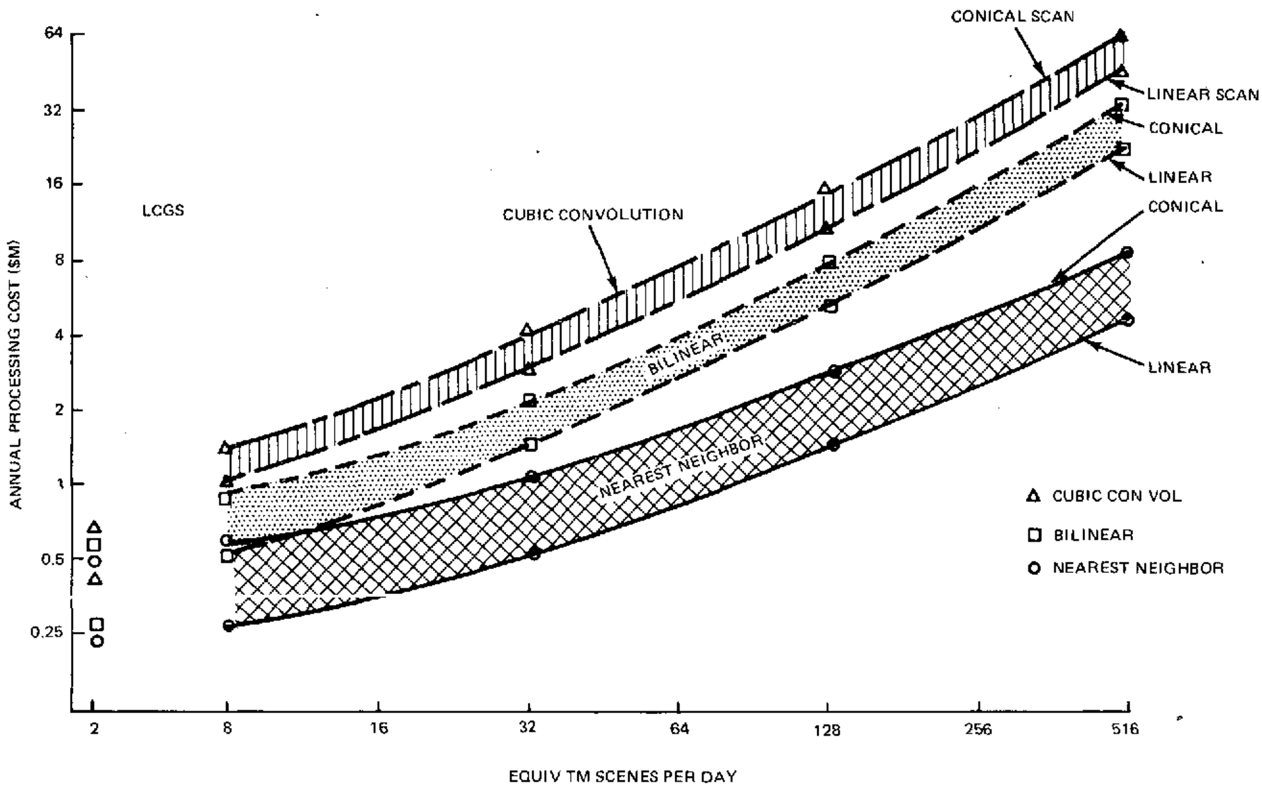

Fig. 2.11-1 Annual Processing Costs

A second trend shown is the strong dependence of processing cost on the two-dimensional interpolation algorithm used during Level II (III) processing (i.e., during resampling/interpolation of the original image data). As processing moves from the simplest algorithm, nearest neighbor (NN) interpolation, to bilinear interpolation (BI), costs increase almost three-to-one. If algorithm complexity is increased still further to "cubic convolution" (approximation to two-dimensional sin(X)/X interpolation) costs increase again by more than two-to-one compared to BI.

Finally, approximate differences between the processing costs for the linear and conical scan data are shown. This difference is due to a fixed increase in the number of machine instructions per pixel which are necessary to compute the coordinates of each output pixel when the original data is resampled. This coordinate computation is relatively simple for the linear scanner (can be performed recursively with only a few instructions) but becomes more complicated with the conical scan data. The impact of the conical scanner decreases as the interpolation algorithm becomes more complex, since the fixed number of additional instructions for coordinate computations is added to a much larger number of instructions per pixel required for interpolation.

At the far left of the figure, the approximate region of cost/throughput where the Low Cost Ground Station might operate is also depicted.

2.12 COST EFFECTIVENESS OF THE TDRSS

The cost effectiveness of the TDRSS for instrument data transmission to the ground was evaluated against (a) direct transmission (DT) to Regional Ground Stations and Primary Ground Stations and (b) the use of wide band video tape recorders (WBVTR) for the recording of data and playback when in contact with a STDN site (Table 2.12-1). This study indicated that the TDRSS was a cost effective means for data transmission for EOS provided the total rental cost of the TDRSS for a single-access user is not charged to the EOS. Costs could vary from no cost (if the network supplies the TDRSS to the EOS program) to $25 million per year if total cost must be borne. Under a bandwidth-time usage formula (i.e., the program pays for use time only), the TDRSS can still be considered cost effective.

Table 2.12-1 TDRSS System Cost Breakdown

OPTION	EARTH TERMINAL	SPACECRAFT COSTS	$M/YEAR DATA PROCESSING & HANDLING COSTS	TOTAL COST (COST IMPACT TO EOS)**	
1. D.T. WITH SIX REGIONAL STATIONS	$6M	—	$4.2M	$10.2M	(0)
2. WBVTR (2TR's)	–––	$2M	$4.2M	$ 6.2M	($2M)
3. TDRSS	$25M (BW PRICING)* $2.5M (BT PRICING)	$3.0M	$4.2M	$32.2M 7.7M	($3M) ($1M)
4. HYBRID 6 LCGS & WBVTR (1 TR)	$0.6M	$1M	$0.4M	$ 2.0M	($1M)

* TDRSS – PRORATED COSTS BASED ON BANDWIDTH (BW) PROPORTION USED BY EOS ($25M) OR BANDWIDTH TIME PRODUCT ($2.5M)

** EOS COST IMPACT INCLUDES ONLY SPACECRAFT EQUIPMENT COSTS

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In addition to cost, TDRSS use offers certain other advantages:

- The WBVTR (two required without TDRSS) would not be used. This saves significant spacecraft weight, power, and cost.
- International data acquisition is enhanced since a significantly larger area of the world can be scanned for data transmission. Using coverage of all land area as an example:

Configuration	% All Land
TDRSS	90%
WBVTR (2)	61%
WBVTR (1)	46%
Primary + Regional Stations	53%

2.13 LOW COST MANAGEMENT APPROACH

It is recommended that the EOS program and the follow-on earth observation mission programs be conducted in a Design-to-Cost environment, which insure program requirements are met within allocated budgets.

Design-to-Cost program acquisition offers specific advantages to insuring that essential program requirements are controlled within allocated budgets. It requires innovative designs and functional concepts, and establishes the mechanism by which cost visibility is provided both to designer and management. The net result is a lower risk program and will maintain a total program cost within prescribed limits by designing to established cost goals and trading performance against cost for selected program requirements. This approach will reduce the cost of the EOS-A and -A' development by approximately $11M.

We recommend a centralized program manager, designated as the System Integrator. He is responsible to the NASA/Goddard EOS Project Manager, and is the system contractor for the Basic Spacecraft, Control Center and Mission Controls, Mission Peculiar Spacecraft, Central Data Processing Facility and Low Cost Ground Station. In addition to the above responsibilities, the System Integrator is responsible for assessing the performance of the Instrument and System GFE contractors, including cost, schedule and technical performance.

We envision a System Integrator functioning with a flexible working team, which will include personnel from NASA/Goddard, user groups, GFE contractors, and the Instrument contractors.

The team concept differs from normal management approaches in that it establishes a working group with the most knowledgeable personnel from each of the participants in the EOS program, with his responsibilities defined to avoid duplication of effort.

The working team will reduce documentation requirements and response times since the various program groups will be intimately involved in program assessment and modification as active team members. The team mix varies as program focus varies through the program phases, and the System Integrator responsibility may very well be assigned to other contractors for follow-on earth observation missions.

A low-cost test program without high risk includes system and component environmental acceptance testing at the module level, the basic spacecraft structure and modules qualified for follow-on as well as the basic mission, and separate component qualification testing.

Cost savings expected from the above approaches are summarized in Table 2.13-1.

Table 2.13-1 Potential Cost Savings

MANAGEMENT APPROACH	POTENTIAL COST SAVING (EOS A AND A')
• DESIGN TO TARGET COST FOR BASIC SPACECRAFT AND INITIAL DMS	11.0M
• SYSTEM INTEGRATION TEAM CONCEPT	1.0M
• SIMPLIFIED CONTROLS AND DOCUMENTATION	1.25M
• SIMPLIFIED TEST	1.8M
• GFE INSTRUMENTS	12.4M
• DIRECT PROCUREMENT-OPERATIONS DATA PROCESSING	3.2M
	TOTAL 30.65M

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2.14 PROGRAM PLAN

The recommended program plan for the EOS-A and -A' is shown in Fig. 2.14-1. The key elements of the recommended plan are;

- Program start in mid-CY'76 with the launch of EOS-A 34 months from program start
- EOS-A and -A' launched one year apart to provide the most cost effective utilization of personnel, GSE and facilities while meeting EOS mission objectives
- Design development and qualification completed prior to the start of the fabrication of flight hardware eliminating costly rework should design deficiencies be found during development testing

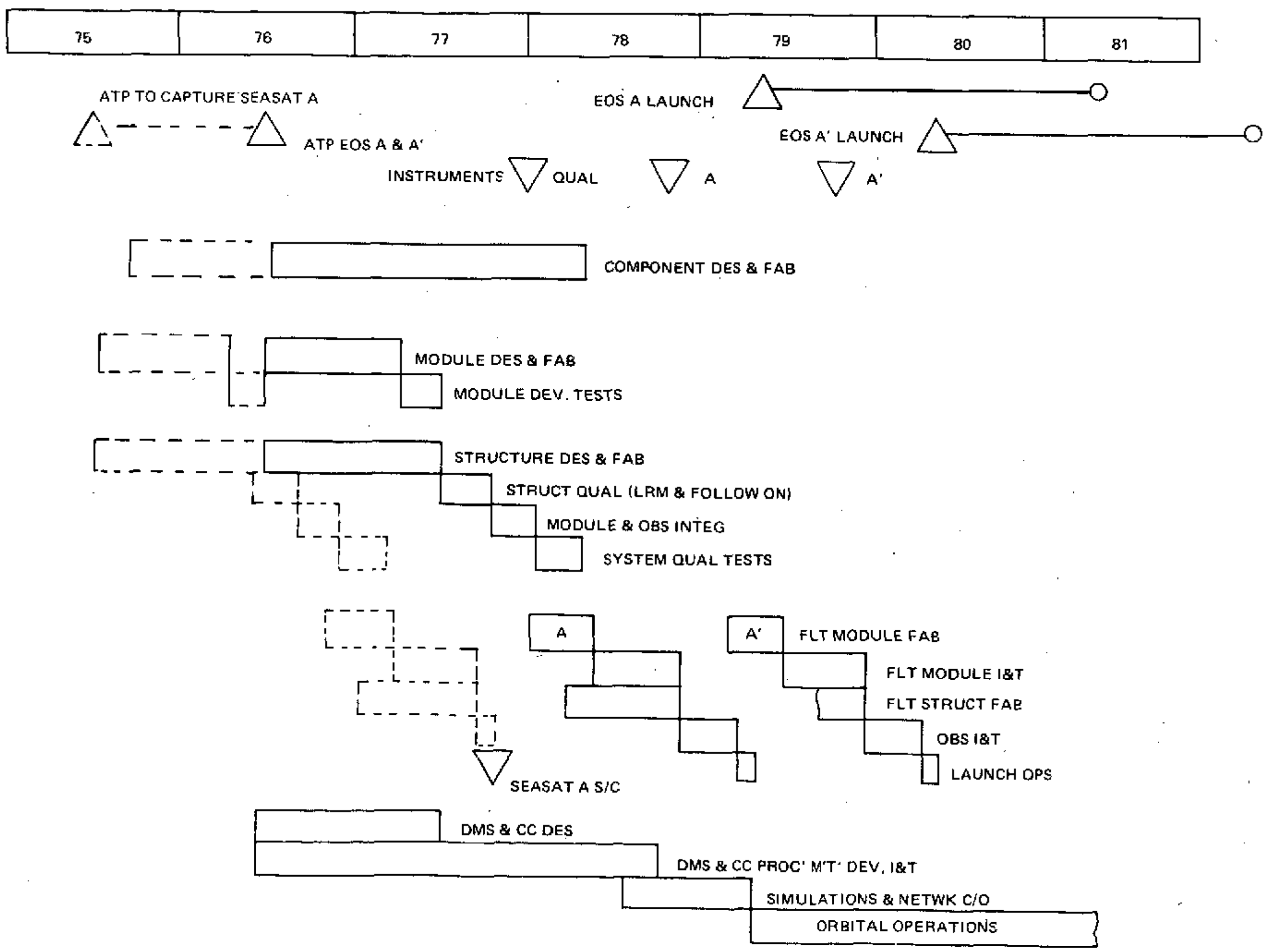

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Fig 2.14-1 EOS Program Schedule Summary

- Early structural qualification tests with component mass representations to define component environments prior to the start of component environments prior to the start of component qualification tests
- Consolidation of all flight hardware environmental tests at the module level.

Inherent in the recommend program plan is a subplan which can be used to provide an acceptance tested basic spacecraft independent of a particular mission. This is illustrated by the schedule option shown on Fig. 2.14-1, which provides a basic spacecraft which meets the SEASAT "A" program requirements for a 1978 launch.

STUDY SUMMARY

3 - STUDY SUMMARY

3.1 SYSTEMS CONCEPT

This EOS study addresses the requirements of both the R&D user, represented by the NASA-JSC Earth Resources Working Group, and the operational users (i.e., The Department of Interior (DOI), the Department of Commerce, the Weather Bureau etc.). The GSFC mission model shown in Figure 3.1-1 depicts the overall scenario of the operational and R&D missions that may be encompassed in the EOS program. EOS A & A' missions for example, are designed to be primarily operational missions combined with R&D missions which eventually evolve into operational missions. We have shown an optional slippage of missions B & B' in the mission model for the following reasons:

a) EOS A & A' have a design life of at least two years, particularly since our concept of the Thematic Mapper (TM) design provides full backup to the Multi Spectral Scanner (MSS)

b) It allows a more practical buildup of the Data Management System (DMS)

c) It minimizes the possibility of requiring Mission Operations Control Center (MOCC) support for more than two missions at the same time.

One of the most critical aspects of this effort is the design of a highly efficient and flexible DMS. This system must be capable of handling the increased data rate dictated by the increased quality of the R&D instruments and capable of delivering data promptly (i.e. 24-48 hr. turnaround) to the user at minimal cost.

The EOS Systems Integration Diagram, Figure 3.1-2, indicates our overview of an integrated system. Of major significance in this system is the design of an efficient and flexible DMS. The DMS must be capable of handling the increased data rates and loads required by the increased quality (higher resolution) of the R&D instruments and also process the daily volume of data promptly (within a 16-hour working day) at reasonable costs.

Requirements for the DMS dictated designs that would be small scale prototypes of an operational system, capable of processing the R&D instrument data that prove out the data processing designs, and also capable of being economically expanded to develop a large scale operational DMS. The R&D DMS development and operations precede the time phased requirements of the operational systems. Therefore our approach and DMS designs provide the practical advantages of implementing pre-operational (small scale) data

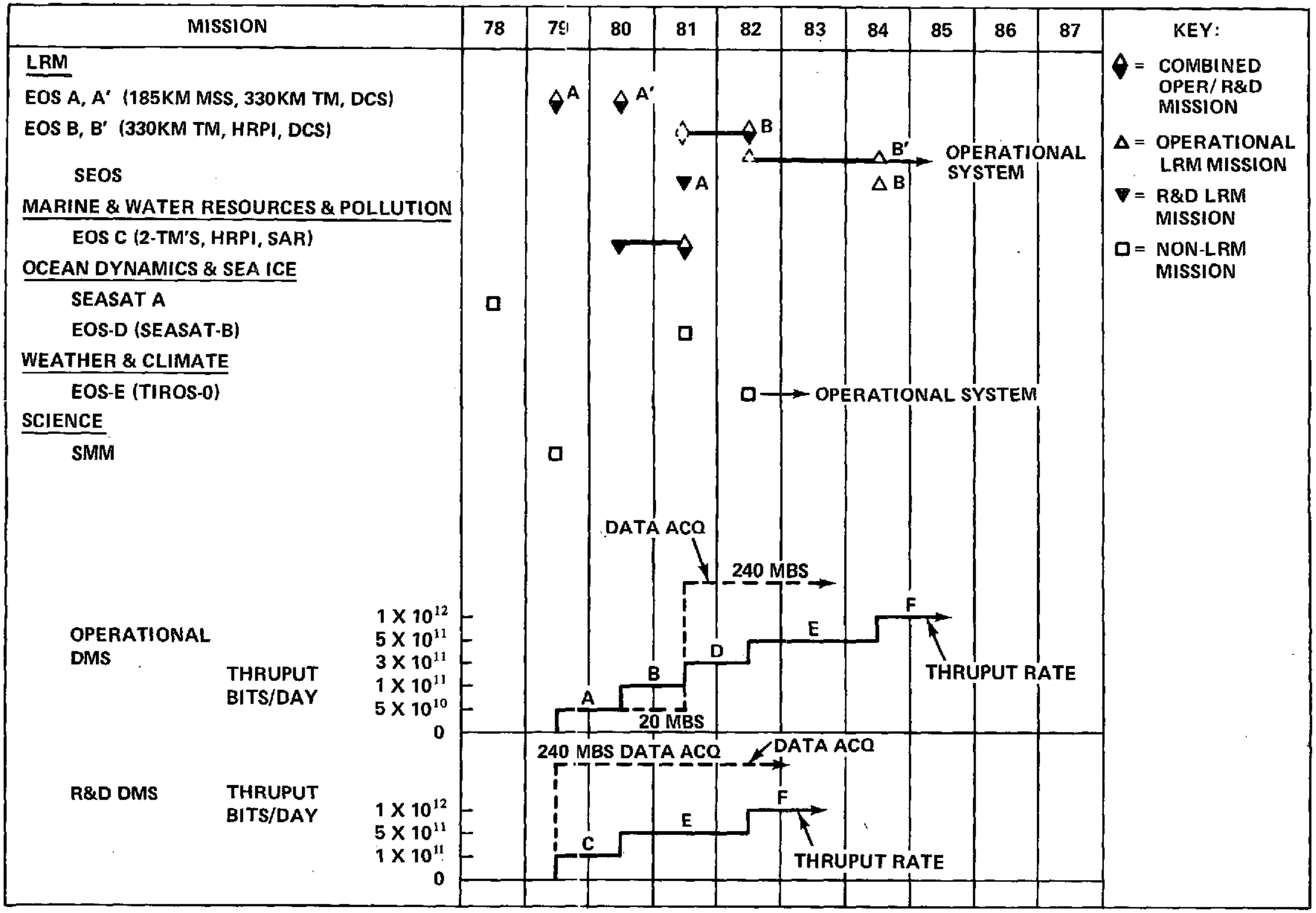

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Fig. 3.1-1 EOS Program Mission Schedule

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processing while permitting R&D verification as enhanced capability is added to the operational DMS.

The system objectives are achieved with two types of data acquisition and processing configurations:

- A primary or high-data-rate configuration made up of Primary Ground Stations (PGSs) and a Central Data Processing Facility (CDPF)
- Secondary or Local User Systems (LUSs) composed of low cost receiving, recording, and processing and display subsystems (Low Cost Ground Station)

Major conceptual advances were accomplished for both configurations. Those for the primary configuration include modular high data acquisition recording and modularized CDPF processors which use minicomputer (later, special purpose hardware processors) that can be simply expanded to develop large scale operational DMS processors.

Similar significant advances for the LUS designs include modular processor and display subsystems utilizing low cost minicomputers. This design permits the data analyst to configure his terminal from a family of hardware, in accordance with his particular needs and available funds. Therefore the capability and the necessary flexibility for future growth is assured.

In the Mission Operations Control Center (MOCC) area, we have interfaced heavily with the operations personnel to assure that our design concepts are viable. Some of the significant features of our MOCC design that led to system flexibility and cost effectiveness were:

a) MOCC basic hardware designed to be independent of mission peculiars

b) Heavy use of interactive CRTs

c) A MOCC modular design which facilitates system expansion to support expanded payload

d) Grouped minicomputer configurations with shared memory

e) Ease of MOCC maintenance

f) On-line edit of contact messages

Our studies have indicated that the existing STDN PGS, with minimum modifications for X-Band reception and CDPF interfacing, can support the EOS missions. These studies have also shown that a PGS located at Sioux Falls could handle all the CONUS Operational Communication requirements and, as a low total cost option, all R&D requirements.

The system design drivers, of course, were the instrument packages. Our design approach was to examine all the available configurations for the Thematic Mapper (TM) and the High Resolution Pointable Imager (HRPI) and define a unit with the maximum composite capability for both instruments. Then a physical configuration was developed for that unit and the required data interfaces identified. The results of this design approach was:

a) A TM utility approximately three-times greater than the baseline system

b) A 3 to 1 cost advantage over the baseline system as a result of the greater utility rate

c) A TM and data processor that can emulate both the MSS and a Quasi-HRPI while meeting the Local User requirements

d) A HRPI that is 90% common to the TM design and hardware

In designing the basic spacecraft, particular attention was given to providing multi-mission capability with minimum cost. This was accomplished through maximum use of space proven designs while allowing sufficient flexibility to utilize improved follow-on capability, e.g. Shuttle.

Our basic spacecraft structure, Fig. 3.1-3, is a delta-frame designed to accept modular subsystems, and to be launched on a Delta, Titan or Shuttle vehicle, with minimum design impact. A special feature of our spacecraft is the resupply mechanisms which use a basic latch and roller concept. This allows commonality in the various resupply latch systems that results in a lower development and manufacturing cost. The basic latch system, which is lighter and less complex than previous concepts, provides three point support for each refurbishable unit. Since it utilizes a single active latch operator, it eliminates synchronization difficulties associated with multiple active latch operators. Rolling contact, rotating joints with dry-film lubricant and rotational redundancy at each joint were designs used throughout to provide high reliability.

The Basic Spacecraft, the Communications and Data Handling (C&DH) subsystem module and our concept of its configuration and interfaces are shown in Fig. 3.1-4. Where applicable and possible, we used developed hardware in the data handling group that was selected from other operational programs. This helped to reduce costs. A cost and configuration summary is shown in Table 3.1-1. A significant design feature of the Data Handling group was the elimination of the on-board tape recorder. Our studies indicated that by adding OBC software for the purposes of compressing housekeeping data we could affect a cost savings by eliminating the tape recorder. The Communications

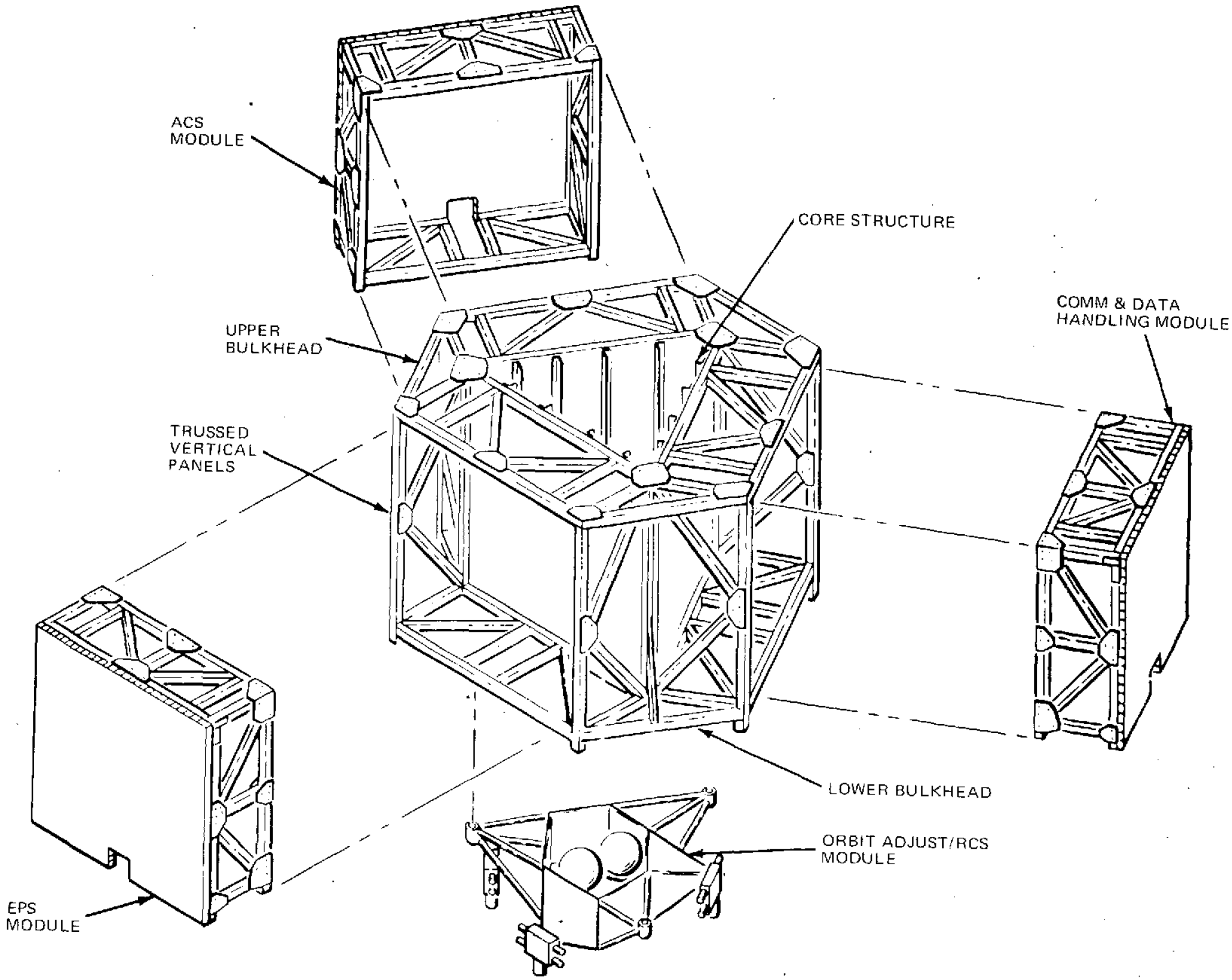

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Fig. 3.1-3 EOS Delta Basic Spacecraft

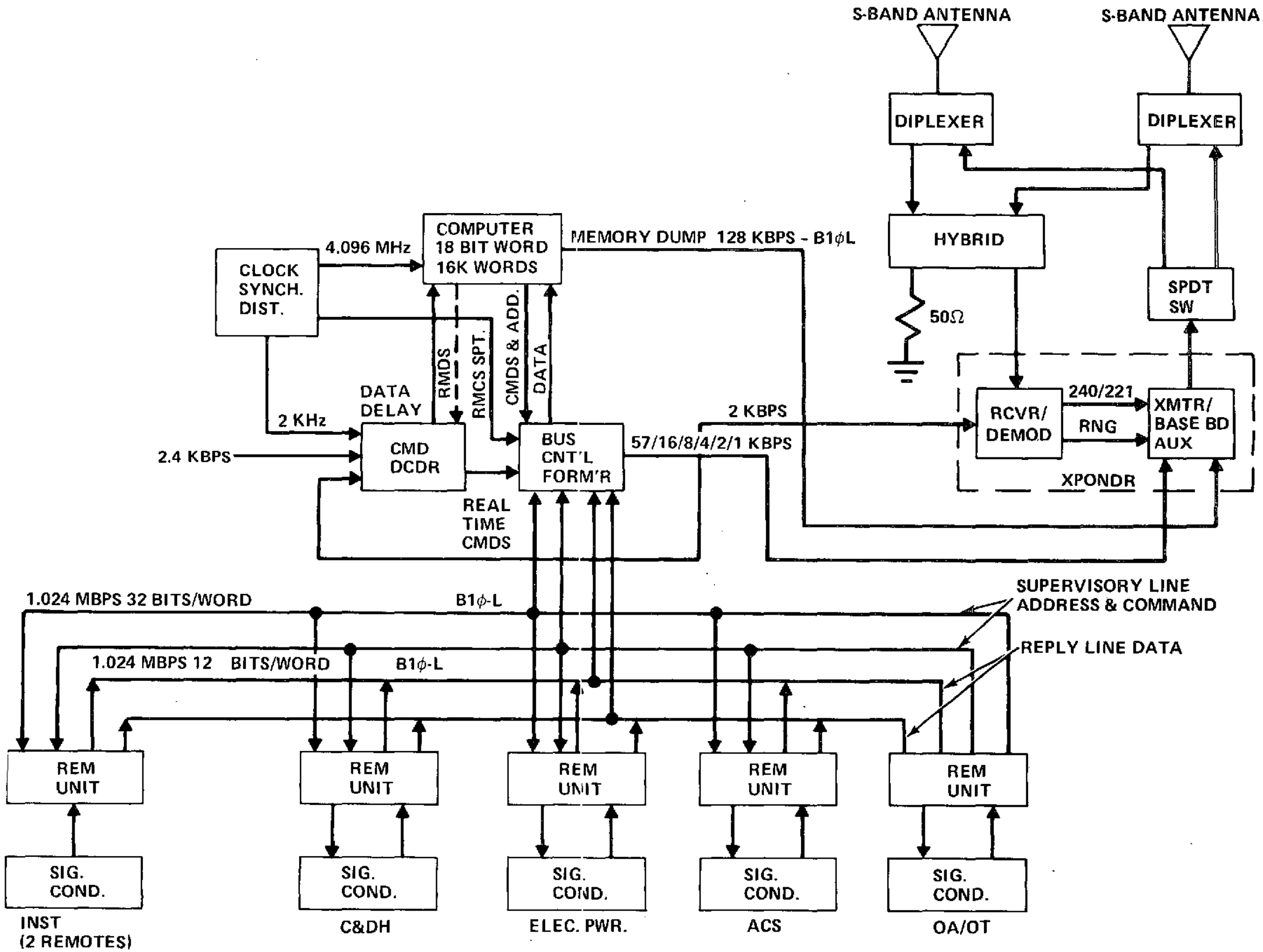

Fig. 3.1-4 Communications and Data Handling, Subsystem (No Redundancy)

Table 3.1-1 Comm Data Handling Summary

COMPONENT	MFG	STATUS	RECURRING COST (1ST UNIT)	WT
TRANSPONDER ASSY o TRANSPONDER o DIPLEXERS o HYBRID o COAX SW.	MOTOROLA	EXIST/ MOD	107	7.8
ANTENNA	GE	EXIST	16	7.4
COMPUTER	AOP	EXIST	110	20
CONT. FORMATTER	HARRIS	NEW	17	3
REMOTE UNIT	HARRIS	NEW	10	1.5
CMD DECODER	HARRIS	MOD	92	12
SIG COND.	GAC	NEW	45	12
CLOCK	GULTON	EXIST	18.75	4
WIRE, CONN ETC,	SEN		14.06	39
STRUCTURE				73
MODULE PRO. TOTAL			429.81K	180
PROCURE, BUILD, TEST ETC.			708.2 K	
TOTAL			1.13801M	180

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group of the C&DH subsystem module provides telemetry, tracking and command capability with the STDN and the Shuttle Orbiter. Communications with the TDRSS and Sioux Falls are available options. Seven communication configurations were derived and compared. Our selection for the basic communications group uses a single S-Band transponder unit with an integrated hybrid, a coaxial switch and two diplexers in conjunction with two broad band S-Band shaped beam antennas.

Our studies of the Reaction Control and Orbit Adjust requirements resulted in an integrated Reaction Control/Orbit Adjust subsystem module. 1.0 lb. and 0.1 lb. reaction control thrusters and the 5.0 lb. OAS thrusters are fed from a common hydrazine fuel supply. The subsystem operates in a blow-down mode. Combining the two subsystems resulted in a weight saving of approximately 35 lb and in a safer, more reliable subsystem module with a simpler installation. This was accomplished at a minor cost penalty of $28K. The flexibility of the module's structural design allows the mounting of two additional propellant tanks (i.e. 23 lb of propellant) to accommodate vehicle growth. The schematic is shown in Fig. 3.1-5 and a cost and configuration summary is shown in Fig. 3.1-6.

Our Electrical Power Subsystem (EPS) was designed with the emphasis placed on providing a configuration that combines high power handling capability with the use of existing equipment. This configuration features the following significant design concepts:

a) It can support spacecraft loads that vary from 400 to 1500 watts

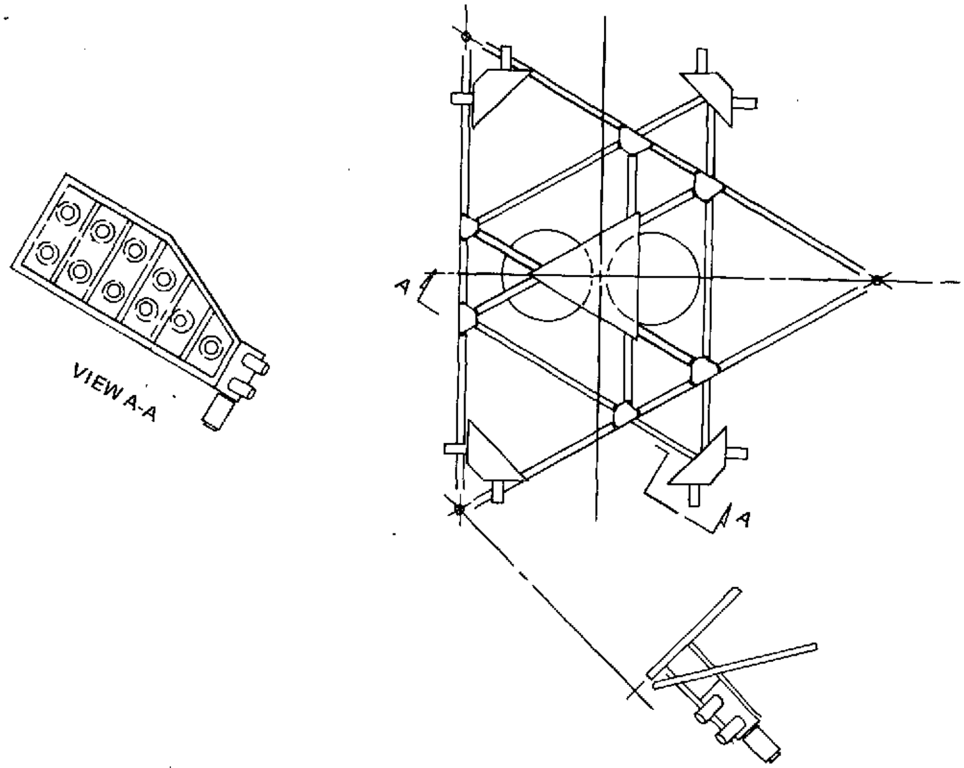

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Fig. 3.1-5 Hydrazine Reaction Control Subsystem (No Redundancy)

Table 3.1-2 RCS Module Summary

COMPONENT	MFG	STATUS	RECURRING COST (1ST UNIT)	WT
FILTER	APM	EXIST	2	1
SOLENOID LATCH (2)	CARLTON	EXIST	10.5	1.4
FILL VALUE, N_2H_4	STERER	EXIST	.6	1
GN_2 FILL	STERER	EXIST	1.2	1
TANK	PSI	EXIST	13	6.9
.1# THRUSTER (8)	TRW	EXIST	42.65	2.0
1# THRUSTER (8)	TRW	EXIST	88.3	3.6
SIG COND	GAC	NEW	34	6.0
REMOTE UNIT	HARRIS	NEW	10	1.5
WIRE THERM, ETC			12.14	12
STRUCTURE				25
			$214.3K	62.9
BUILD TEST, PRO			256 K	
			571 M	62.9#

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b) It has a weight savings of approx. 100-lb over the GSFC demonstration power module

c) It has a cost saving of over $150K.

The EPS mission peculiar options include:

a) 40 to 120 amp hours of stored energy

b) A "direct-energy-transfer" capability of the solar array that can be varied to supply all, part, or none of the spacecraft load

c) Capability ot "max power track" part or all of the solar array output. A block diagram and cost and configuration for the EPS module are shown in Fig. 3.1-6 and Table 3.1-3.

The approach used in designing the Attitude Control Subsystem (ACS) was to encompass the LRM requirements in addition to the applicable follow-on missions. Several ACS configurations were established from which one was selected by trading-off of performance and cost effectiveness. The significant features of the design selected were:

a) 0.01° pointing accuracy and 1×10^{-6} deg/sec stability will support all EOS missions. The SEOS may require the addition of an easily integrated gimballed star tracker for roll control.

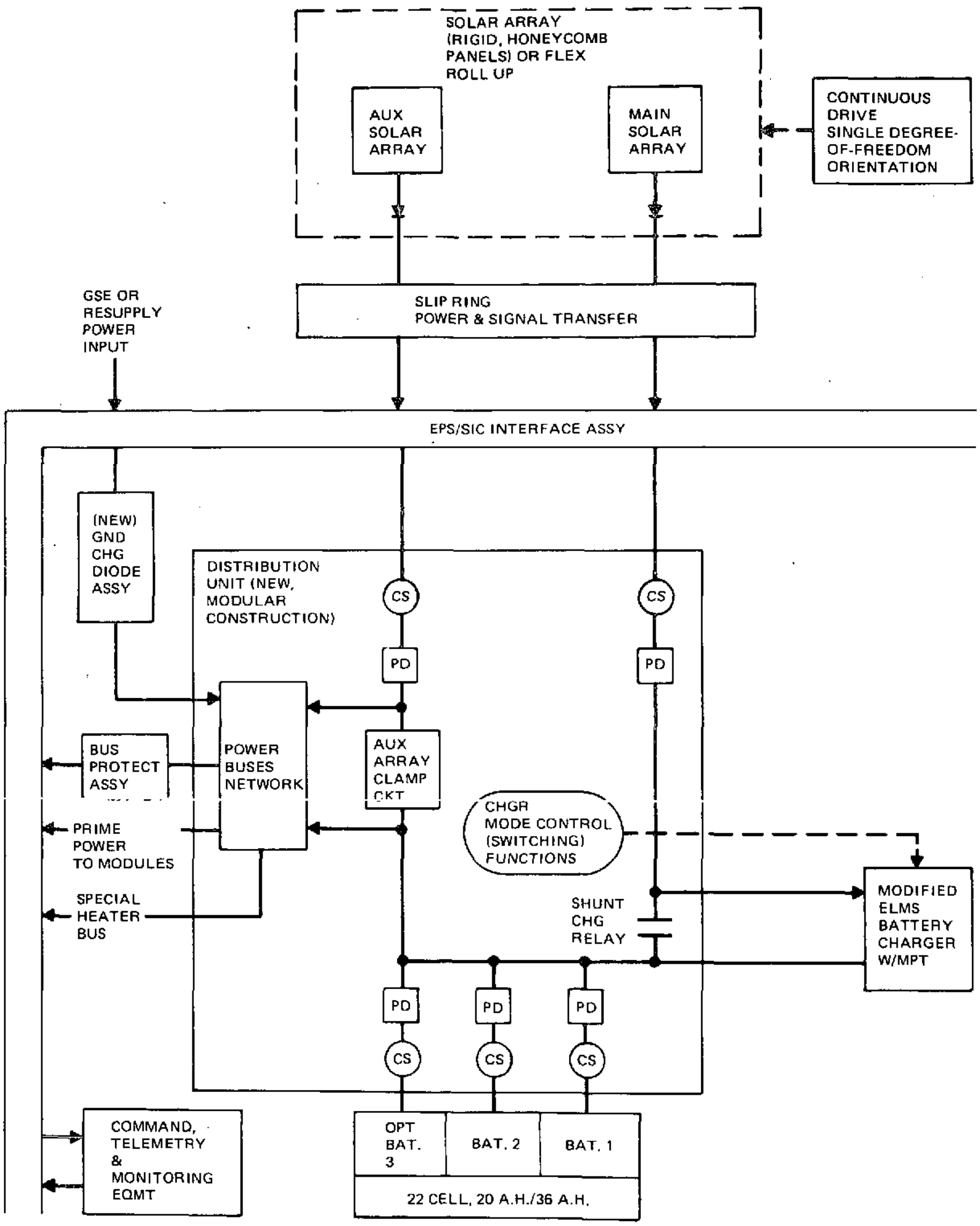
b) All sensors and actuators are existing or modifications of existing equipment

c) Use of the on-board computer eliminates hardware cost changes as missions and requirements change

d) Backup vehicle safe mode eliminates high cost of redundancy for vehicle survival.

A block diagram of the ACS and the cost and configuration is shown in Table 3.1-4 and Fig. 3.1-7. The general approach used in establishing the EOS thermal design was to use passive control techniques wherever possible. The instruments and modules are thermally isolated from the structure and the structure was wrapped with thermal insulation. Variable conductance heat pipes and split heat sinks are employed for battery thermal control. Our design minimizes costs by using state of the art techniques and materials and by reducing test time.

While the EOS design has been centered around the well defined requirements spelled out for the Land Resources Management and Oceanology missions, sufficient flexibility has been provided in the systems design to support Ocean Dynamics/Sea Ice and Weather/Climate missions with a progressive R&D/Operational capability.



3-234 **Fig. 3.1-6 Electrical Power Subsystem (No Redundancy)**

Table 3.1-3 EPS Module Summary

COMPONENT	MFG	STATUS	RECURRING COST (1ST UNIT)	WT
BATT CHARGER	GULTON	EXIST	55	27
BATTERY	EAGLE PITCHER	EXIST/MOD	36.4	64
CONTROL ASSY	GAC	NEW	18.7	23
SIGNAL COND	GAC	NEW	56	10
REMOTE UNIT	HARRIS	NEW	10	4
GND CH DIODE ASSY	GAC		.6	11
BUS PROT ASSY	GAC		.8	7
WIRE, CONN THERM	GAC		25.4	26
STRUCTURE				73
			202.9	245#
PROCURE, BUILD, TEST, LOAD			577.1	
	MODULE TOTAL		$780.0K	245#
SOLAR ARRAY			368	170
DRIVE			52	25
STRUCT, BUILD, TEST, PROCURE			335	
	SOLAR ARRAY TOTAL		755	195
	EPS TOTAL		1.53M	440#

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Table 3.1-4 ACS Module Summary

COMPONENT	MFG	STATUS	RECURRING COST (1ST UNIT)	WT
CSS (2)	BENDIX	EXIST	4.	.3
DSS	ADCOLE	EXIST/OAO	42	5.
RGA	BENDIX	EXIST/IUE	235	15
FHT	ITT	EXIST/ELMS	43	17
ELECT ASSY	ITHACO	NEW	193	13
REMOTE UNIT (2)	HARRIS	NEW	20	3
MAGNETOMETER	SCHOENSTEDT	EXIST		6.5
REAC WHEELS (3)	BENDIX	EXIST/OAO	90	30
MAG TORQ	ITHACO	MOD		30.6
SEN, THERM, WIRE, ETC			14.3	35
STRUCTURE				73
			641 K	229
PROCURE, BUILD, TEST, LOAD			648 K	
	TOTALS		1.26M	229

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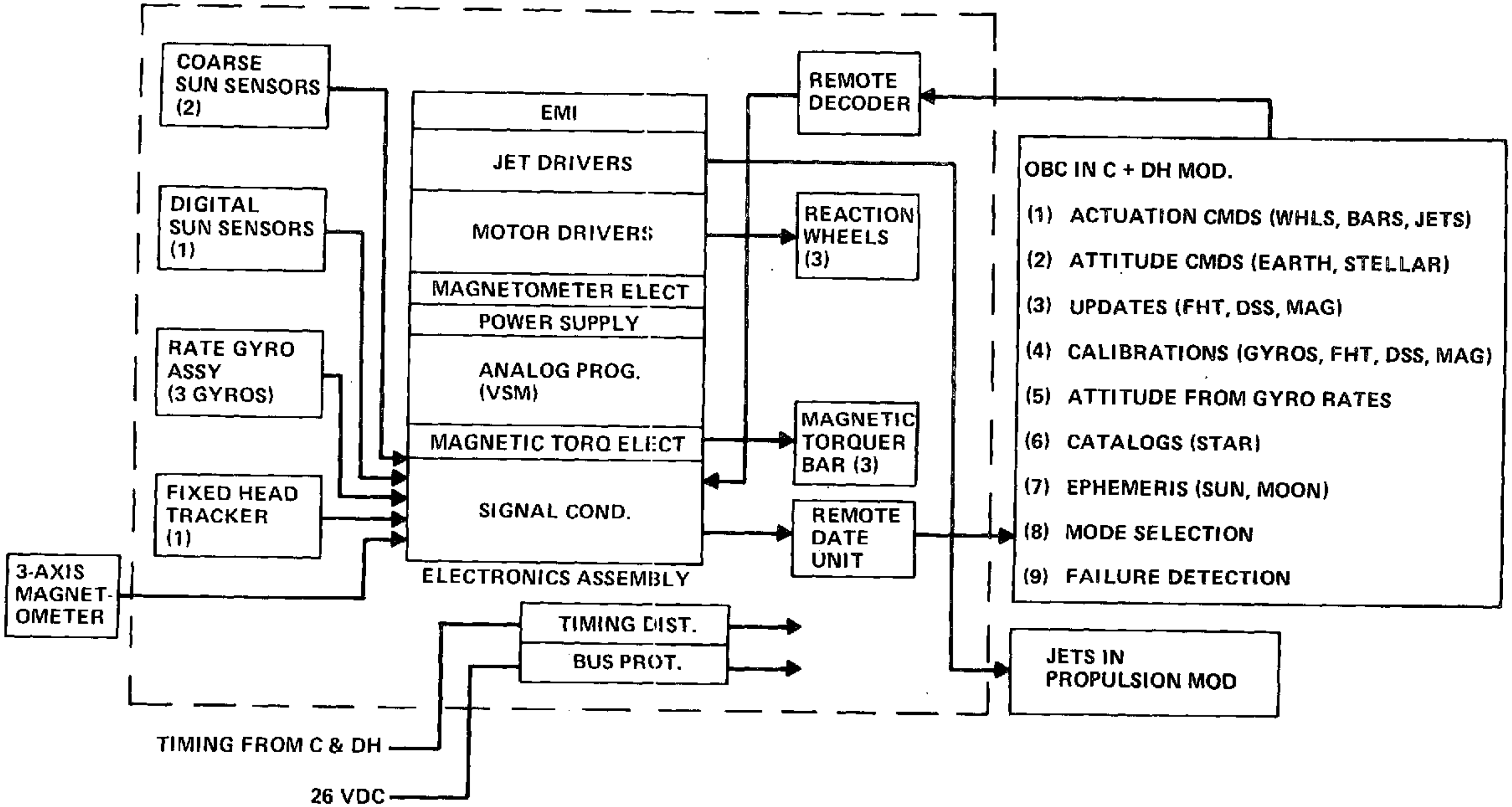


Fig. 3.1-7 Attitude Control Subsystem (No Redundancy)

3.2 PROGRAM REQUIREMENTS

EOS requirements originate from many sources. Those that come from the potential user community drive the instrument design and the data output requirements. Support of instruments in orbit and the transmission of data to the user generate many more requirements. The multitude of sources for requirements is obvious by scanning the reference documentation listed in the EOS RFP. Our design studies and system analysis expand this list.

The requirements must be organized and categorized so that the source of each requirement is available, and so that all software and hardware requirements are imposed during the design process. Further, by categorizing the requirements, conflicting or complementing requirements are highlighted for resolution. The initial categorization of requirements has been accomplished in an EOS Program Requirement Document, given in Appendix C, by collecting those imposed by NASA to initiate the EOS Design Study. Other requirements which are derived during the study as a result of analysis are also incorporated.

Imposed requirements guide the program plan and the design process. They are categorized into the following primary EOS elements: program, mission, system, and subsystem. The detailed breakdown includes such typical elements as mission and traffic models, safety, interfaces, test requirements, data management system, and logistic support requirements. Each requirement has its status defined (interim or verified) and the program option to which it applies. Further, the source of each requirement is coded to the source documentation or to an applicable trade study. See Figure 3.2-1 for a sample requirements document page.

Mission functional analysis has been done for three types of EOS Land Resources Management missions: Delta launched, Titan launched and Shuttle deployed. Functional analysis (Appendix B) permitted systematic identification of derived requirements. This process demands that decisions be made at the highest level at which functions are identified, therefore reducing the number of "garden paths" to be followed. When a lower level function uncovers a "show stopper," the higher level decision is immediately evident and can be changed if desired.

The functional analysis was used as an input to construct mission time-lines (Fig. 3.2-1), which revealed time related requirements (e.g. battery power needed until solar panel deployment).

Extensive bookkeeping is usually necessary to translate functions into requirements and then allocate these requirements to the subsystem level. However, EOS requirements already exist, some in extensive detail. Therefore, the method adopted was to identify derived requirements from the functional analysis and to compare the functions, and associated requirements, with the imposed requirements. As each functional requirement was identified in the existing Requirements Document, the function was checked off. When requirements are not in the Requirements Document, they are added, including suitable notation as to the source. Since each function in the functional analysis is identified by a number, and follows a standard numbering sequence, a convenient method of tracing functional requirements exists.

Compilation of the EOS Requirements Document will serve as the basis for the EOS System Specification which will be accomplished in the next study phase.

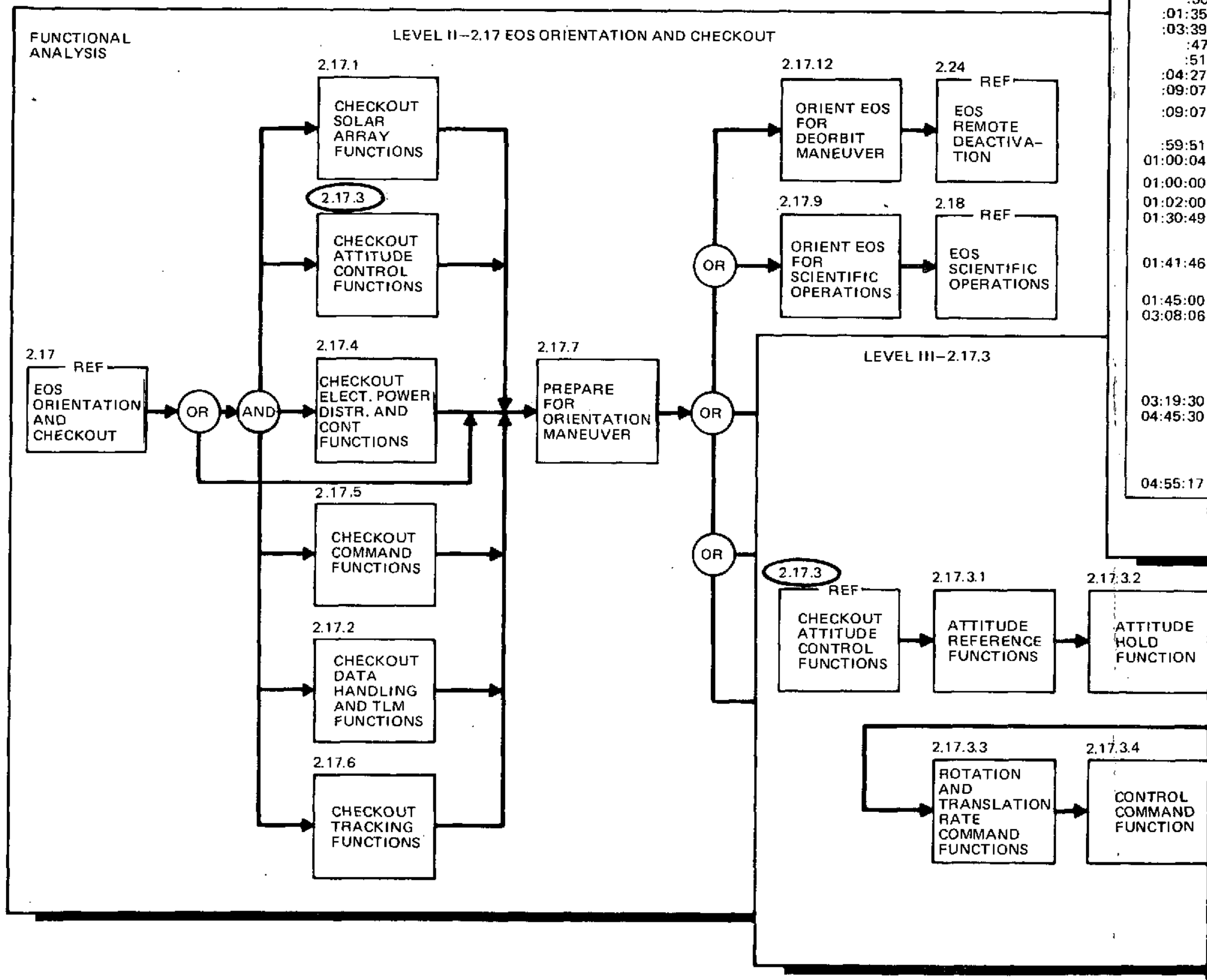
3.3 PROGRAM OPTIONS

3.3.1 CENTRAL DATA PROCESSING (CDP) DESIGN/COST OPTIONS

This study defines feasible functional configurations for various CDP options. The EOS program scenario postulates a number of EOS missions beginning with EOS-A and extending through A', B, B', C SEOS-A and SEOS B within five years (1979-1984), and encompassing both R&D and operational missions, as shown in Fig. 3.1-1. Instruments include operational MSS, R&D and operational TM, HRPI, SAR possibilities, and DCS. These instruments combined in various time-phased conditions, impose an initial daily data load on the DMS of approximately 3.5×10^{10} bits/day for 200 scenes of the operational MSS, and eventually could build to a much higher level (10^{12} bits/day), if International Data is processed in the CDP. In addition to data loading, the CDP is driven by:

- Levels of Processing (Level 1-Radiometric calibration and one-dimensional line correction; Level 2-Geometric corrections for Earth model, bestfit ophemeris, and two-dimensional resampling of all data to place them in the selected grid format; Level 3-Precision geometric correction, identical to Level 2, except Ground Control Points for precise resampling are used).
- The number of users
- The number of output products.

The DMS scenario, in concert with Figure 3.1-1, envisions an operational facility, not necessarily co-located with the R&D facility. The operational facility may be under control of DOI at Sioux Falls, while the R&D facility may be NASA operated at GSFC.



DPC 470-51 **LPM–DELTA DEDICATED S/C–MISSION MODEL 11B–OPTION 5** DATE 6/4/74

G.E.T. HR MIN	ΔT HR MIN SEC	EVENT OPERATION		REMARKS
00:00:00	00:09:07	LAUNCH TO ORBIT		
00:00:00		LIFTOFF		
:38		SOLID MOTOR BURN OUT		
:01:35		JETTISON SRM		
:03:39		MAIN ENGINE CUTOFF	VTS 6.0	(VANDENBERG)
:47		STAGE I & II SEPARATION		
:51		II STAGE IGNITION		
:04:27		FAIRING SEPARATION		
:09:07		II STAGE CUT-OFF		
:09:07	50:57	DELTA ORBITAL OPERATIONS		
	50:44	COAST		
:59:51		II STAGE IGNITION		
01:00:04		II STAGE CUT-OFF (366.1 NM ORBIT)		
01:00:00	:45:00	EOS SEPARATION		
01:02:00		DELTA II TO SEPARATION ATTITUDE		
01:30:49		COMMAND DELTA/EOS SEPARATION		VERIFY EOS HEALTH (FAIRBANKS)
		ENABLE ACS LOGIC	ULA 11.0	
		ENABLE ATTITUDE CONTROL		RATE STABILIZED
01:41:46		NULL SEPARATION RATES		
		DELTA II EVASIVE MANEUVER		TBD
01:45:00	72:00:00	SURVIVAL PHASE		ALLOW APPROX. 72 HR. FOR HIGH VOLTAGE EQUIP. OUTGASSING.
03:08:06		JETTISON DUST COVERS		
		COMMAND SOLAR ARRAY DEPLOYMENT ATTITUDE		
		ARM ARRAY DEPLOYMENT	ULA 11.4	
		DEPLOY SOLAR ARRAY		
		ENABLE ARRAY DRIVE		
03:19:30		ENABLE BATTERY CHARGING		
04:45:30		COMMAND COARSE SUN ORIENTATION		
		COMMAND INERTIAL HOLD		
		AUTO SWITCH TO FINE SUN ORIENTATION	ULA	
		ENGAGE REACTION WHEELS	9.8	
04:55:17				

	REQUIREMENT	SOURCE	OPTION 1	2	3	4	5
4.1.4	ATTITUDE AND CONTROL						
4.1.4.1	PROVIDE FOR SPACECRAFT CONTROL DURING INITIAL ACQUISITION, REACQUISITION, NORMAL OPERATIONS, ORBIT ADJUSTMENT MANEUVERS, AND COARSE ATTITUDE HOLD MODE.	F/I, 2.1/-	•	•	•	•	•
4.1.4.2	WHERE AN INTEGRAL TUG IS REQUIRED, THE ACS SHALL PROVIDE ATTITUDE AND RATE SIGNALS REQUIRED FOR ITS CONTROL	F/I, 2.1/-			•		
4.1.4.3	THE ACS SHALL BE DRIVEN DIRECTLY IN RESPONSE TO ERROR SIGNALS GENERATED BY SENSORS OR VIA THE ON-BOARD COMPUTER	F/I, 2.1/-	•	•	•	•	•
4.1.4.4	THE ACS SHALL BE CAPABLE OF OPERATING WITHIN SPECIFICATION CYCLIC DISTURBANCE TORQUE LIMITS OF: ORBIT ALT (N MI) — TORQUE (FT-LB): AVERAGE (SECULAR) / CYCLIC (PEAK)	F/I,2.2.5/-					
	300 - 900: 10^{-5} TO 0.1 / 2×10^{-4} TO 0.2		•	•	•	•	•
	GEOSYNCH: 10^{-6} TO 0.01 / 2×10^{-5} TO 0.02						•
4.1.4.5	THE ACS SHALL ACCOMMODATE THE FOLLOWING RANGE OF SPACECRAFT MASS PROPERTIES: WEIGHT 2500 - 25,000 LB; MOMENT OF INERTIA 500 - 100,000 SLUG - FT^2	F/I,2.1/-	o	o	o	o	o
4.1.4.6	ACS RESPONSE OUT-OFF FREQUENCY SHALL BE 0.1 Hz	F/I,2.1/-	o	o	o	o	o
4.1.4.7	IN ACQUISITION MODE: A. REDUCE S/C RATES TO LESS THAN ± 0.03 DEG/SEC B. DEFINE S/C INERTIAL ATTITUDE TO WITHIN ± 2 DEG	F/I,2.2.1/- (TS 3.2.7)	o	o	o	o	o

REVISION DATE

Fig. 3.2-1 Program Requirements Development

FOLDOUT FRAME

FOLDOUT FRAME

NASA would process data from the R&D instruments and develop the techniques and equipment required. As the instruments and techniques/equipment for processing became operational, DOI would incorporate this data processing capability in their expanded facility.

Table 3.3-1 depicts a set of CDP options consistent with the logical data and user build-up of the scenario. The initial options (A&B) conform to an operational MSS, using all digital processing, located at Sioux Falls. Option A was evaluated by NASA and documented in the report "Earth Resources Survey, Operational System Study", September 1973. Costing was based on 200 scenes per day with the OCC performing the functions of orbit determination and image annotation. The costs were listed as $9.54 million for configuring the facility and operating it for a five-year period, less civil service costs (Configuration 3D in the ERS study). Option B, with two operational MSS instruments sending data, would double the load on the processing center. Thus, since the second MSS (EOS-A') timeframe lags the first (EOS-A), it is reasonable to consider a doubling of the investment and operating cost of Option A with a reduction of 50% for the second set of spares and engineering service contracts. This would make a five-year Option B cost of $18.4 million.

The remaining options in Table 3.3-1 stem from the initial R&D nature of the instruments and their data processing requirements, as well as the levels of processing and the output products. Option C represents a minimum system (throughput, level of processing, output products), Option F a maximum system, while Options D&E are intermediate ones consistent with the need for transitioning from the R&D to the operation mode. Cost parametrics for these options can be obtained from the Data Operations tradeoff study, Section 6.5.

The parameters for Option C thru F were introduced into the cost/throughput model computer program of the Data Operations tradeoff study and exercised, in this case, for the program scenario. Concentration was on the EOS A & A' missions. Recurring costs were generated on a yearly basis.

Option C costs for two years of operation subsequent to EOS A' launch, making a total of 3 years of operation, plus the CDP configuration and based on a mini-computer modular system, were $26.7M. Similarly, Option D costs were $86M, Option E costs were $135M, and Option F costs were $208M. Thus, for the high-volume options, the use of standardized systems such as mini-computers, imposes a severe cost penalty. It is expected that a reduction in costs of approximately 4/1 to 10/1 can be obtained with special purpose processors, depending on throughput.

The cost interrelationship between CDP design and processing algorithm must be noted. Section 5 configured a CDP system based on a 185 km swath TM, 20 scenes a day. Processing was based on use of 50% bilinear and 50% cubic convolutional algorithms. Option C used the wide-swath TM, also 20 scenes a day, and 100% bilinear interpolation. The design configuration of the CDP in Section 5 will handle the throughput of Option C as indicated. Cost differences are mainly in the increased expendables (HDDT and CCT tapes) required to handle the wide swath TM as compared to the 185 km TM.

3.3.2 INTERNATIONAL DATA ACQUISITION

Data acquisition by foreign users has been established during the earlier ERS programs. To date, they have received data either in processed form (from NASA, DOI) or in raw from (NASA, direct acquisition-Canada, Brazil). Methods for placing EOS data in foreign user's hands have been defined during the current study with each having their own peculiar impact on the program. These options include:

Option 1: Direct transmission (D.T.) to foreign user ground stations.

Option 2: A wideband video tape recorder (WBVTR) system for collection of foreign data and processing and distribution from CONUS.

Option 3: A TDRSS Configuration for the relay of foreign data to CONUS for processing and distribution.

Option 4: A hybrid system consisting of a WBVTR, dumping to a primary ground station, and six low cost ground stations (LCGS). This configuration is primarily intended for use with an IDA mission involving relatively low data volume, such as wheat crop only.

The relative performance rating of each IDA option (less the hybrid) is shown in Table 3.3-2 based solely on the percentages of available data each alternative can provide for three data volumes of interest. The TDRSS configuration is clearly superior to the other configurations, followed by the 2-site (Alaska and NTTF) WBVTR configuration, the D.T. System and finally the single (Alaska) site WBVTR system.

The costs of each of the three primary IDA options and a hybrid system configuration are given in Table 3.3-3.

Table 3.3-1 CDP Options

Option A	Option B	Option C	Option D	Option E	Option F
MSS: (Operational) (EOS A) Data Vol: 200 Scenes/Day Level of Processing: Lev. 1: 200 Scenes Lev. 2: 0 Scenes Lev. 3: 200 Scenes Products: HDDT: 3 CCT: 20 Multi-Scene (6250 BPI) 200 Single Scene (1600) 1st Gen: B&W: 2000 Bands 2nd Gen: B&W Prints: As needed Color Film: As needed Color Prints: As needed Format: CCT: 2 Film: As needed	MSS: Operational (EOS A & A') Twice MSS (Operational EOS A)	TM: (330 KM) Data Vol: 20 Scenes Usable/Day Level of Process: Lev. 1: 20 Scenes Lev. 2: 10 Scenes Lev. 3: 10 Scenes Products: HDDT: 10 for DOI 10 for NASA CCT: 40 (6250 BPI) 40 (1600 BPI) 1st Gen: B&W: 280 Bands 2nd Gen: B&W Prints: 140 + 3 x 140 Color Film: 20 + 20 Color Prints: 20 + 60 Format: CCT: 2 Film: 1 Note: (1) 1 x 10" Bits/Day (2) 2 Users for HDDT (3) CCT, 10-users (4) HDDT has 2 scenes/Tape (5) CCT has 1/4 scene (6250 BPI) (6) 1/2 of scenes on 6250 CCT (7) All film prod. for NASA Use only, DOI makes own. NASA to supply B&W 1st gen.	TM: (330 KM) Data Vol: 60 Scenes Usable/Day Level of Process: Lev. 1: 60 Scenes Lev. 2: 0 Scenes Lev. 3: 60 Scenes Products: HDDT: 30 for DOI 30 for NASA CCT: 120 (6250) 120 (1600) 1st Gen: B&W: 840 Bands 2nd Gen: B&W Prints: 420 + 3 x 420 Color Film: 60 + 180 Color Prints: 60 + 180 Format: CCT: 3 Film: 3 NOte: (1) 3 x 10" (2) 2 Users for HDDT (3) 50 CCT Users (4) HDDT – 2 scenes/tape (5) CCT – 1/4 scene (6250 BPI) (6) 1/2 scenes on CCT (7) 1st Gen: 6 users (8) Films for NASA use DOI makes own. DOI to get 1st Gen B&W	2 TM + HRPI Data Vol: 100 Scenes Usable/Day (TM EQ) Level of Processing: Lev. 1: 100 Scenes Lev. 2: 25 Scenes Lev. 3: 75 Scenes Products: HDDT: 50 for DOI 50 for Nasa 50 for others CCT: 200 (6250) 200 (1600) 1st Gen: B&W: 1600 Bands 2nd Gen: B&W Prints: 700 + 3 x 700 Color Film: 100 + 3 x 100 Color Print: 100 + 4 x 200 Format: CCT: 5 Film: 3 Note: (1) 5 x 10" bits/day (2) 5 Users for HDDT (3) 75 CCT users (4) HDDT – 2 scenes/tape (5) CCT – 1/4 scene (6250 BPI) (6) 1/2 of scenes on CCT (7) 1st Gen: 25 Users (8) Films for NASA use. DOI makes own. DOI to get 1st Gen. B&W	TM + HRPI + SEOS Data Vol: 200 Scenes (TM EQ) Usable Daily Level of Processing: Lev. 1: 200 Scenes Lev. 2: 0 Lev. 3: 200 Scenes Products: HDDT: 100 for Nasa 100 for DOI CCT: 400 (6250) 400 (1600) 1st Gen: B&W: 1600 2 Bands 2nd Gen: B&W Prints: 1400 + 10 x 1400 Color Film: 200 + 10 x 200 Color Print: 200 + 10 x 200 Format: CCT: 5 Film: 3 Note: (1) 1 x 10^{12} Bits/Day (2) 5 Users for HDDT (3) 100 CCT users (4) HDDT – 2 scenes/tape (5) CCT – 1/4 scene of TM on a tape (6250) (6) 1/2 of scenes on CCT (7) 1st Gen: 50 users (8) Films for NASA use. DOI makes own from 1st Gen. B&W

Table 3.3-2 IDA System Performance Ratings

CONFIGURATION RATING	PERCENT ALL LAND	PERCENT TILLED LAND	PERCENT WHEAT CROP
TDRSS	1 (90%)	1 (98%)	1 (96%)
WBVTR 2 SITES	2 (61%)	2 (75%)	3 (87%)
D.T.	3 (53%)	3 (65%)	2 (91.5%)
WBVTR 1 SITE	4 (45.7%)	4 (56%)	4 (84%)

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Table 3.3-3 System Cost Breakdowns

OPTION	EARTH TERMINAL	SPACECRAFT COSTS +	$M/YEAR DATA PROCESSING & HANDLING COSTS	TOTAL COST (COST IMPACT TO EOS)**
1. D.T. WITH SIX REGIONAL STATIONS	$6M	---	$4.2M	$10.2M (0)
2. WBVTR (2 TR'S)	---	$2M	$4.2M	$ 6.2M ($2M)
3. TDRSS	* $25M (BW PRICING) $2.5M (BT PRICING)	$1M	$4.2M	$30.2M ($1M) $ 7.7M ($1M)
4. HYBRID *** 6 LCGS & WBVTR (1 TR)	$0.6M	$1M	$0.4M	$ 2.0M ($1M)

*TDRSS - PRORATED COSTS BASED ON BANDWIDTH (BW) PROPORTION USED BY EOS ($25M) OR BANDWIDTH TIME PRODUCT (BT)? $2.5M.

**EOS COST IMPACT INCLUDES ONLY SPACECRAFT EQUIPMENT COSTS.

***PRIMARILY INTENDED FOR LOW DATA VOLUME MISSIONS.

+NON-RECURRING COSTS PRORATED.

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As a result of the IDA program options study, the following conclusions emerged:

- The TDRSS configuration can be the most cost-effective solution for large data volume users, but with attendant technical risks. In addition, the cost sensitivity of the TDRSS approach is dependent on the way the TDRSS costs are apportioned between users.
- The hybrid configuration of Direct Transmission and WBVTR is a low risk and cost-effective option for low volume data missions such as single crop monitoring
- The remaining two options are technically low risk but of a higher program cost.

3.3.3 LUS PROGRAM OPTIONS

One version of the LUS is the LGCS. Its purpose is to acquire and record direct delivery (EOS-to-LCGS) compacted payload data. By this action, a data user with only local area (interests can receive his selected data in a more timely fashion than if he received the data via the CDPF, assuming mailing of the data.

However, during the study several alternative data delivery methods were considered. These delivery methods can be more cost effective than the direct delivery method, but particular LUS configurations must be defined prior to the cost effectiveness evaluation.

For example, dial-up 50 to 56 kb/sec wideband common carrier lines should be available throughout CONUS by the EOS time frame. Also, DOMSATs are becoming available and could be used to relay EOS high-rate payload data in near-real time from almost any point within CONUS to almost any other point. Therefore, relayed payload data via telecommunications (computer-to-computer) links could be less expensive than providing each LUS with the means to directly receive selected data, and within short time periods (i.e., several minutes to a few hours).

LUS processor and display subsystems with modular flexible hardware and software were conceptually developed to explore their costs and capabilities. Table 3.3-4 shows three subsystem models and their hardware cost as a function of quantity.

By replacing the RF/IF and data handling/recording LUS subsystems with a relatively inexpensive 50 kb/sec modem and computer interface (approximate cost $2K) the LCGS LUS terminal hardware cost would be reduced about $70K. This dollar saving plus operator and maintenance cost reductions, and enhanced simplicity of data reception could be attractive to several classes of local data users.

The processor and display telecommunications subsystem LUS concept can be used as a satellite processor with a colocated data processing facility or as a user terminal in a network of local users that would cover a region (for example the Western U.S., Eastern U.S., etc.) or the entire CONUS area.

With the preceding considerations in mind, it is envisioned that this innovative relayed data delivery concept could prove a most valuable option for enhancing the entire EOS program. Therefore, it is recommended that NASA consider the preceding data delivery concepts as well as the LCGS concept in future EOS studies and implementation planning.

3.3.4 SHUTTLE UTILIZATION

Although EOS will predate the Shuttle, it will be operational during the Shuttle era. The Shuttle has unique capabilities for deploying multiple payloads, retrieving orbiting payloads, and servicing payloads on-orbit. The potential exists to exploit these capabilities to realize an EOS system that is more flexible, more economically attractive, and/or more operationally effective than is possible using conventional, dedicated launch vehicles restricted to delivery only.

Table 3.3-4 LUS Processor & Display Hardware Model Capabilities With Costs as a Function of Quantity

LUS MODEL	HARDWARE	CAPABILITIES	COSTS ($K) VERSUS QUANTITY(2)		
			1	10	100
BASIC	1 - MINICOMPUTER (32K MEMORY) 1 - DISK AND DRIVE (29 MBy) 2 - MAGNETIC TAPES (1) AND DRIVES 1 - OPERATOR I/O CRT/KEYBOARD 1 - B & W IMAGE STORAGE DISPLAY 1 - DATA REPRODUCER INTERFACE	DISPLAY - B & W IMAGES (1024 x 1024 PIXELS) DATA PROCESSING - YES (SLOW) IMAGE ANALYSIS - YES (VERY SLOW) HARDCOPY - YES (CAMERA)	61	58	48
ENHANCED I	BASIC HARDWARE PLUS 1 - MINICOMPUTER (MULTIPROCESSOR) 1 - LINE PRINTER 1 - INTERACTIVE COLOR DISPLAY 1 - FLOATING POINT & HARDWARE MULTIPLY/DIVIDE FOR MINICOMPUTERS	DISPLAY - B&W OR COLOR IMAGES DATA PROCESSING - MODERATE SPEED IMAGE ANALYSIS - INTERACTIVE HARDCOPY - CAMERA AND LINE PRINTER FOR THEMATIC MAPS, ETC.	159	151	124
ENHANCED II	BASIC, ENHANCED I HARDWARE PLUS 1 - DISK AND DRIVE (29 MBY) 2 - MAGNETIC TAPES AND DRIVES 1 - B&W AND COLOR IMAGE RECORDER 1 - COLOR IMAGE DISPLAY	DISPLAY-2 SIMULTANEOUS B&W OR COLOR IMAGES DATA PROCESSING - REASONABLY FAST IMAGE ANALYSIS - INTERACTIVE, CHANGE ANALYSIS, MODERATE SPEED HARDCOPY - LINE PRINTER AND FIRST GENERATION PHOTO PRODUCTS (70 MM to 4"x5" SIZES)	240	228	187

NOTES: (1) 75 ips & 1600 bpi

(2) ALL COSTS ARE FOR HARDWARE ONLY, FOB MANUFACTURERS POINT OF SHIPMENT. ESTIMATED COSTS SUBJECT TO VARY WITH RESPECT TO MARKET CONDITIONS AND PARTICULAR MANUFACTURERS HARDWARE. OFF-THE-SHELF HARDWARE USED IN THE CONFIGURATIONS. HARDWARE ELEMENT INTERFACE COSTS INCLUDED COMPONENTS PLUG TOGETHER.

A variety of Shuttle utilization options are available for consideration. In order to determine the operational, economic, and cost impacts benefits of Shuttle utilization, each option must receive an objective assessment. The options scheduled for consideration in the utilization study are:

a. Deploy Only

The multiple payload deployment capability of Shuttle offers the potential for sharing transportation costs among several payloads, defraying individual user costs.

b. Deploy plus Retrieve

Payload retrieval offers the capability of recovering orbiting spacecraft for subsequent reuse on other missions or as a means of replacing exhausted or malfunctioning spacecraft to extend the on-station life of experiments. These applications can be attained by any one of the following alternatives, each offering specific advantages:

(1) Scheduled Retrieval

The spacecraft is retrieved on a regularly scheduled basis, regardless of its operating status. This approach is particularly attractive to ground operations, enabling realistic sizing and scheduling of support crews and facilities, and entails minimum impact on Shuttle flight scheduling and mission planning. Consideration of retrieval, refurbishment, and subsequent replacement with and without a spare spacecraft provides discrimination between the effects of fleet size and instrument on-orbit operating time.

(2) Non-scheduled Retrieval

The spacecraft is retrieved, refurbished, and replaced only when it ceases to operate. This approach takes maximum advantage of potential spacecraft operating life in excess of its design life (e.g. ERTS, OAO) to minimize required flights and down time. The concept of a spare spacecraft to accelerate replacement after retrieval also applies.

c. Deploy plus Resupply plus Retrieval

Instead of retrieval for refurbishment on the ground, the spacecraft is serviced in orbit, with replacement of selected modules. Upon completion of the assigned mission, or if a major malfunction occurs which cannot be repaired in flight, the spacecraft is retrieved, returned to earth for refurbishment, and re-deployed for a new mission or resumption of the interrupted mission. Options applicable to this mode are:

(1) Scheduled Resupply

This is the same philosophy which is explained under Scheduled Retrieval.

(2) Non-scheduled Resupply

This is the same philosophy which is explained under Non-scheduled Retrieval.

(3) Resupply Concept

Each of the resupply options listed above can be implemented using alternative resupply concepts. These concepts, which influence development and operations are:

(a) Fully manual
e.g. EVA, IVA

(b) Man-tended
Remote control of manipulator mechanisms by console operators.

(c) Fully-automated
Pre-programmed sequences for all operations.

The relative merits of the above options, considering the interactions among spacecraft design life vs resupply interval, resupply concept, design impact, and modularity level will be treated in Report No. 6, "Space Shuttle Interfaces/Utilization." An initial assessment of the design impact for Shuttle compatibility is reviewed in Section 6.3 of this document.

The investigation of Shuttle utilization is basically the integration of three individual trade studies to reflect their interactions. These studies are:

- Shuttle Compatibility
- Design Life/Resupply Interval
- Resupply Concept

Figure 3.3-1 illustrates the fundamental study tasks and their relation to the overall study logic. The objective of the study is to identify the relative merits of each alternative Shuttle mode in terms of operations, design, and cost impact, and the associated influence on Instrument "up" time. The major issues to be addressed, in order of significance, are:

- Is Shuttle utilization beneficial to the EOS program?
- What are the preferred modes of Shuttle utilization?
- What are the effects of each approach to Shuttle utilization on each participating element (e.g. Shuttle, EOS, user, operations)?

These questions are addressed in Report No. 6, "Space Shuttle Interfaces/Utilization".

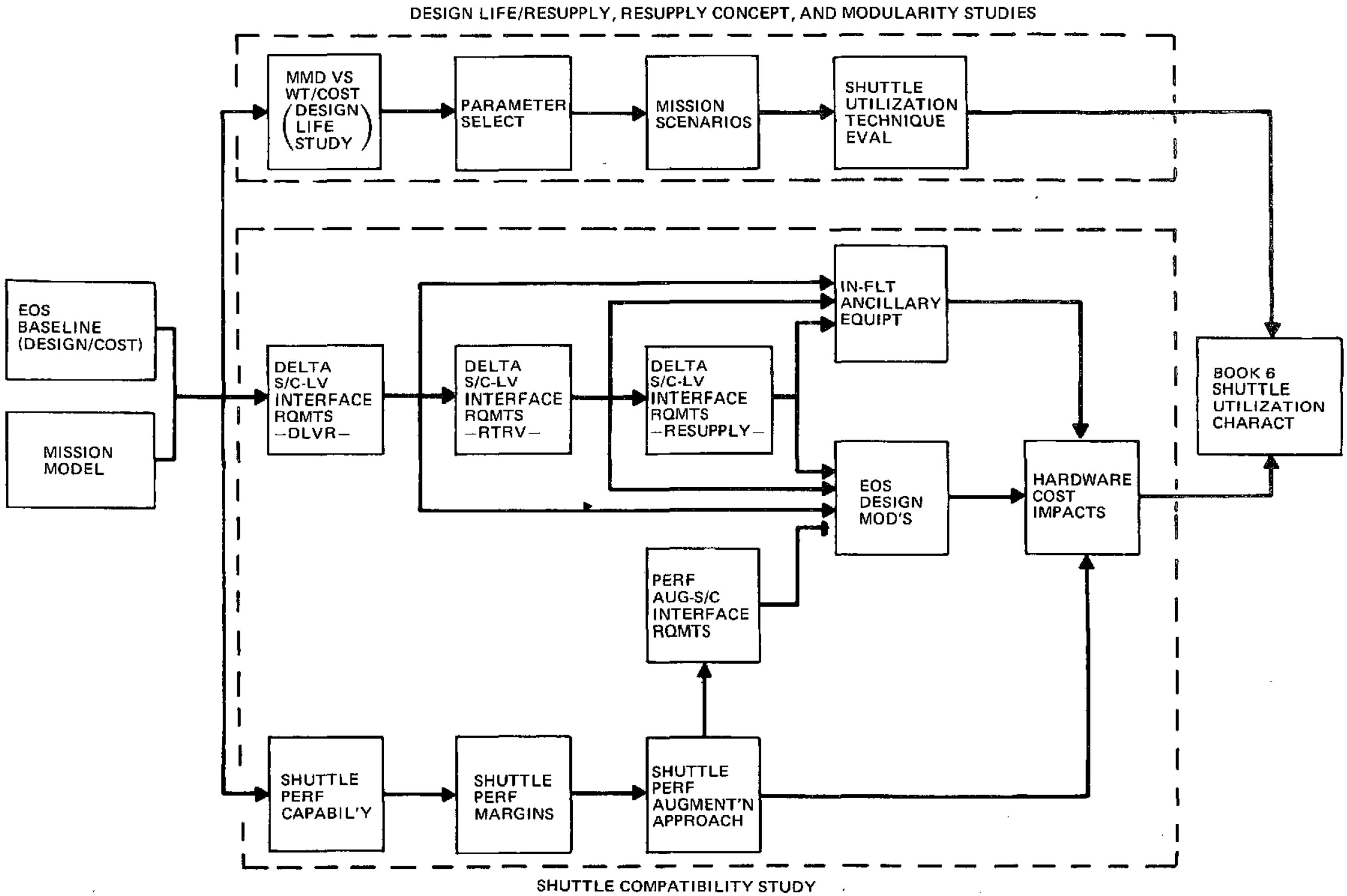

Fig. 3.3-1 Space Shuttle Interface/Utilization Study

3.3.5 OBSERVATORY WEIGHT/COST OPTIONS

Although the methodology described in Section 3.4 combine and compare the parameters of program cost, spacecraft/payload weight, and booster capabilities varies slightly depending on the mission, the resulting output of the system synthesis task is common to all missions. This output, shown in Figure 3.5-2, provides concise means of describing in a single chart what capability the space segment can provide in each facility and what the cost incurred to provide this capability is. A "facility" can be defined by the following characteristics:

- Launch Vehicle
- Instrument Payload
- Central Data Processing Capability
- Observatory Design Life
- Capability Options
  - Low Cost Ground Station Support
  - Support of International Data Acquisition (TDRSS or WBVTR)
  - Shuttle Display, Retrieval or Resupply

Since the booster choice in effect defines the capability the user sees, we have chosen to label the in-space segment of the facilities by the booster names. Thus we speak of the Delta 2910 "facility," the Titan IIIB "facility" etc. Each of these capture a given number of the intended missions as shown in Table 3.3-5, and each falls within some cost target. Our approach has been to allow for enough flexibility in the spacecraft design to decouple it as much as possible from these programmatic choices. In other words, our goal is to fly essentially the same spacecraft in establishing the space segment of each of the facilities mentioned.

Since a change of booster has major cost implications ($5M-$10M per spacecraft) the approach has been to provide as many capability options as possible within a given booster facility until it becomes more cost effective to change to a higher-payload booster. For example, the figure shows how we have, for the Delta 2910 facility, built up the "low-cost barebones" spacecraft, added the required instruments and then provided for the ad-

Table 3.3-5 EOS Facility Capabilities

FACILITY	MISSIONS CAPTURED	WEIGHT RELATED OPTIONS CAPTURED
DELTA 2910	• EOS-A AND A'	- ALL (WITH ROLL-OUT ARRAY) - REDUNDANCY + RETRIEVAL - MSS WIDE BAND TAPE RECORDERS (WITH RIGID ARRAY)
	• EOS-D (SEASAT-B)	- ALL (WITH ROLL-OUT ARRAY) - REDUNDANCY + RETRIEVAL (WITH RIGID ARRAY)
	• SMM	- ALL
DELTA 3910	• EOS-B AND B'	- ALL
	• SEASAT-A	- ALL
TITAN III-B	• EOS-C	- ALL
	• EOS-E (TIROS-O)	- ALL (WITH ROLL-OUT ARRAY) - REDUNDANCY + RETRIEVAL + INCREASED STRUCTURE (WITH RIGID ARRAY)
TITAN III-C7	• EOS-F (SEOS)	- ALL

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ditional capability options of redundancy, retrieval, resupply etc. Since this spacecraft is specifically designed for low cost, it is not the minimum weight design. The upper curve in the figure indicates the expected minimum weight configuration. The decreased weight and increased cost of this configuration results from the weight saving design changes given in Table 3.3-6. As can be seen from the table, the total cost of the changes can be considerable. In fact, if the Delta 3910 launch vehicle is available, with its increase of approximately 1000 pounds to orbit at a cost of $800K recurring and $500K non-recurring above the $4M cost of the D2910, it would not be cost effective to include any changes in the design which result in any more than a $1.3M increase in total program cost.

Table 3.3-6. Weight Savings Options, EOS-A

DESCRIPTION OF CHANGE	WEIGHT REDUCTION (LB)	COST IMPACT ($K)
CHANGE INSTRUMENT SUPPORT STRUCTURE MATERIAL FROM ALUMINUM TO HYBRID GR/EP COMPOSITE TUBULAR TRUSS	30	150
USE FLEXIBLE (ROLL-OUT) SOLAR ARRAY IN PLACE OF RIGID DEPLOYABLE SOLAR ARRAY	93	700
CHANGE BASIC SPACECRAFT CORE STRUCTURE MATERIAL FROM ALUMINUM TO HYBRID GR/EP COMPOSITE	82	400
MODULE STRUCTURE	62	300

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3.4 KEY TRADES

3.4.1 INSTRUMENT APPROACH

The key trades in this area revolve around the weight of the instruments as a function of altitude; the utility of the overall mission as a function of scanner swath width, and then the two remaining questions - impact of a particular scanner configuration on the ground data processing system and the detail questions concerning each individual scanner design.

All of the candidate scanners except the Westinghouse pushbroom approach are deemed feasible and within the current state of the art.

The impact of scanning technique on ground data processing is addressed in Section 3.4.3.

3.4.1.1 INSTRUMENT WEIGHT VS ALTITUDE (BASIC TRADE)

The most basic instrument trade arises if the radiometric performance is held fixed and the best available detectors are assumed for a given resolution. The collecting aperture of the telescope for the TM and HRPI is then proportional to the square of the altitude. Furthermore, for telescopes in this size range. the weight is also proportional to the aperture, i.e., the square of the altitude. In order to keep the booster size down, a strong preference for a lower altitude was conveyed to the orbit selection group. One of the orbits enumerated in our proposal then became preferred, for a number of reasons, 680 km (See Fig. 3.4-1).

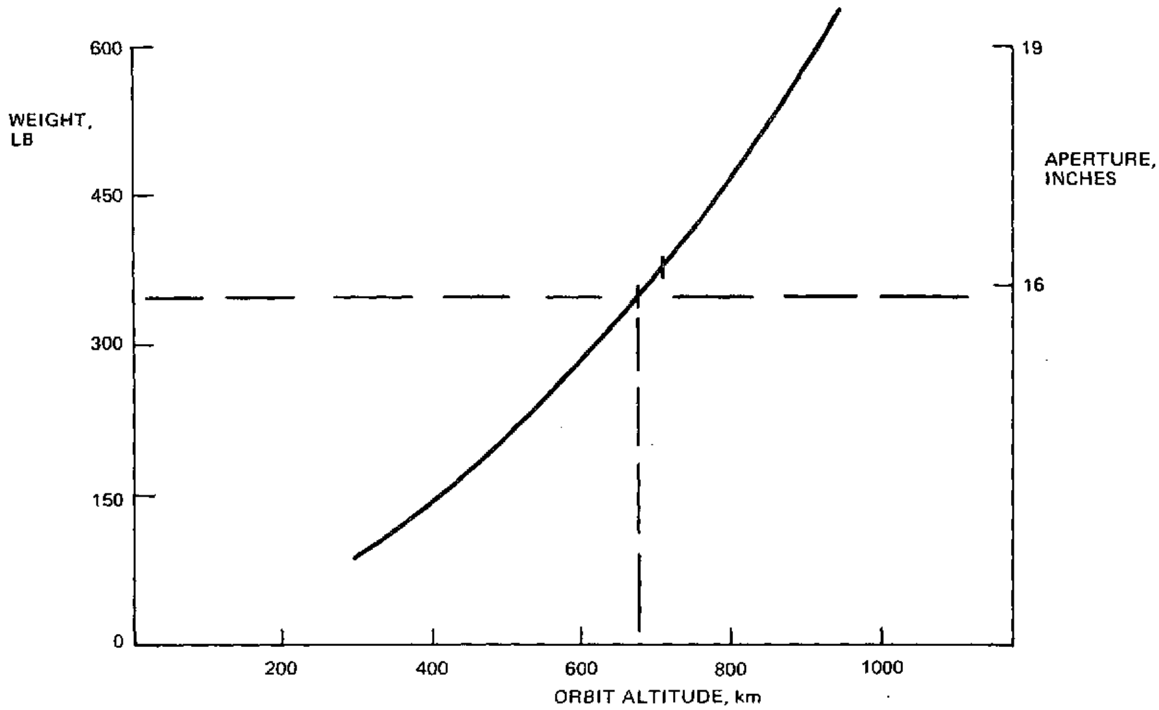
3.4.1.2 INSTRUMENT UTILITY vs SWATH

The baseline TM is quite under-utilized at 185km swath width ($\pm 8^{o}$ from nadir). This narrow swath is well below the point where geometric linearity or atmospheric path problems become important.

By increasing the swath width, considerably greater area is covered per day, the repeat cycle is shortened and more area is revisited on a 1-3 day interval. Thus, the utility of the mission is increased by about 3:1 by increasing the swath to 330 km (See Fig. 3.4-2).

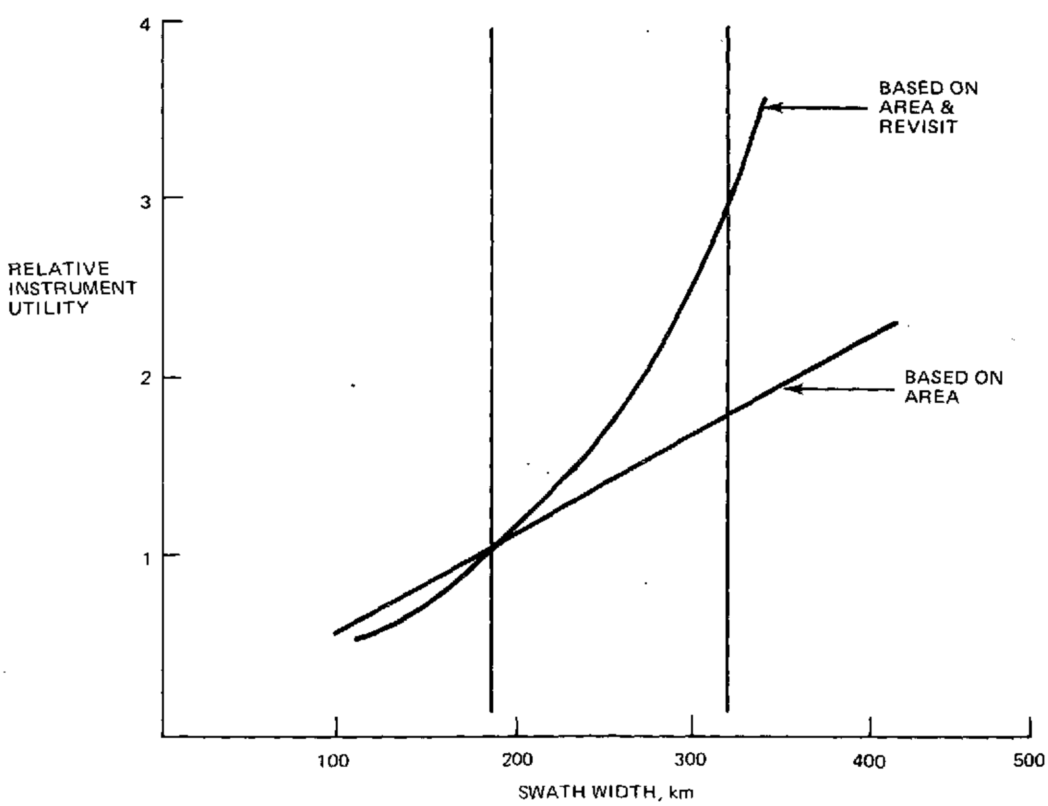
3.4.1.3 INSTRUMENT CAPABILITY

In view of the high desirability of a wider TM swath, this possibility was reviewed with the point design contractors.



3-126

Fig. 3.4-1 Instrument Weight vs Orbital Altitude



3-127

Fig. 3.4-2 Instrument Utility Vs Swath Width

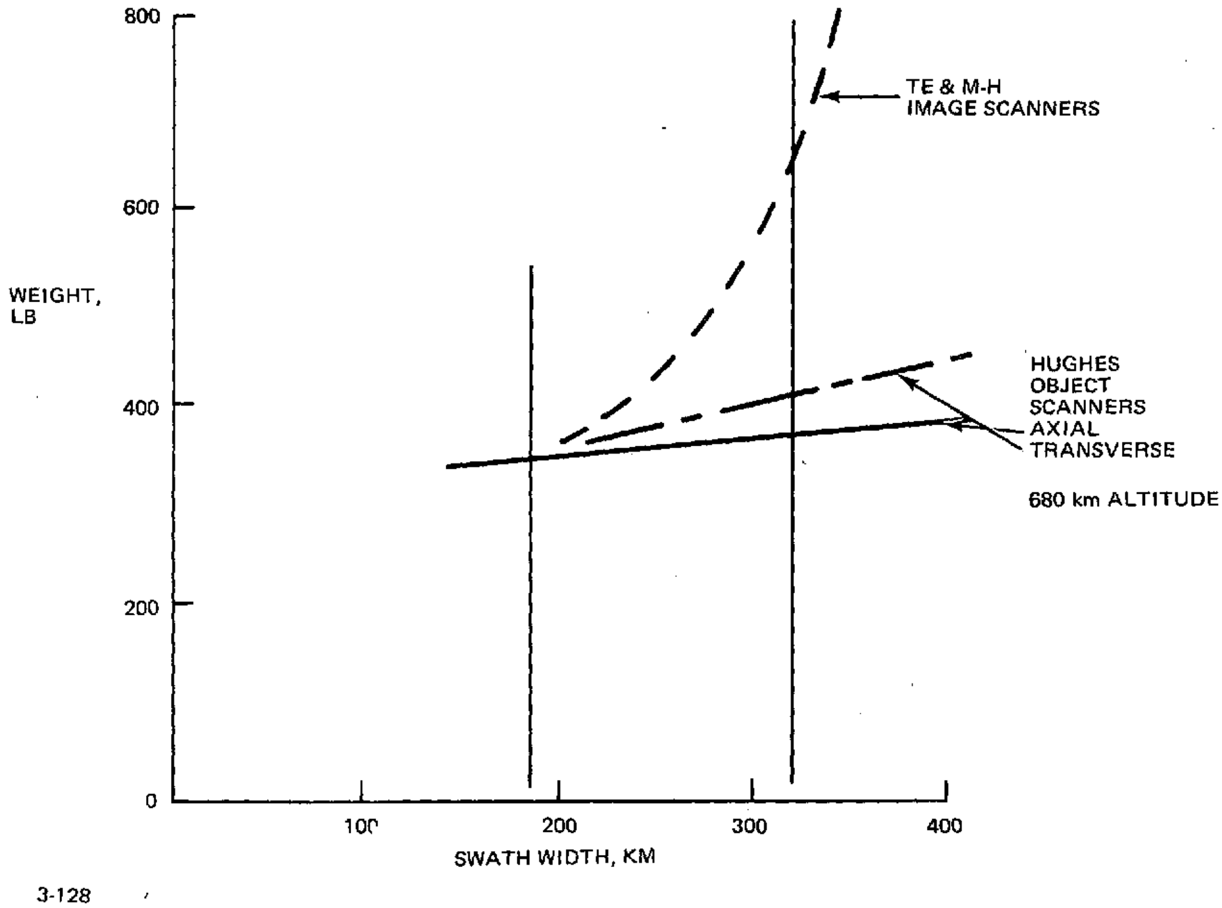


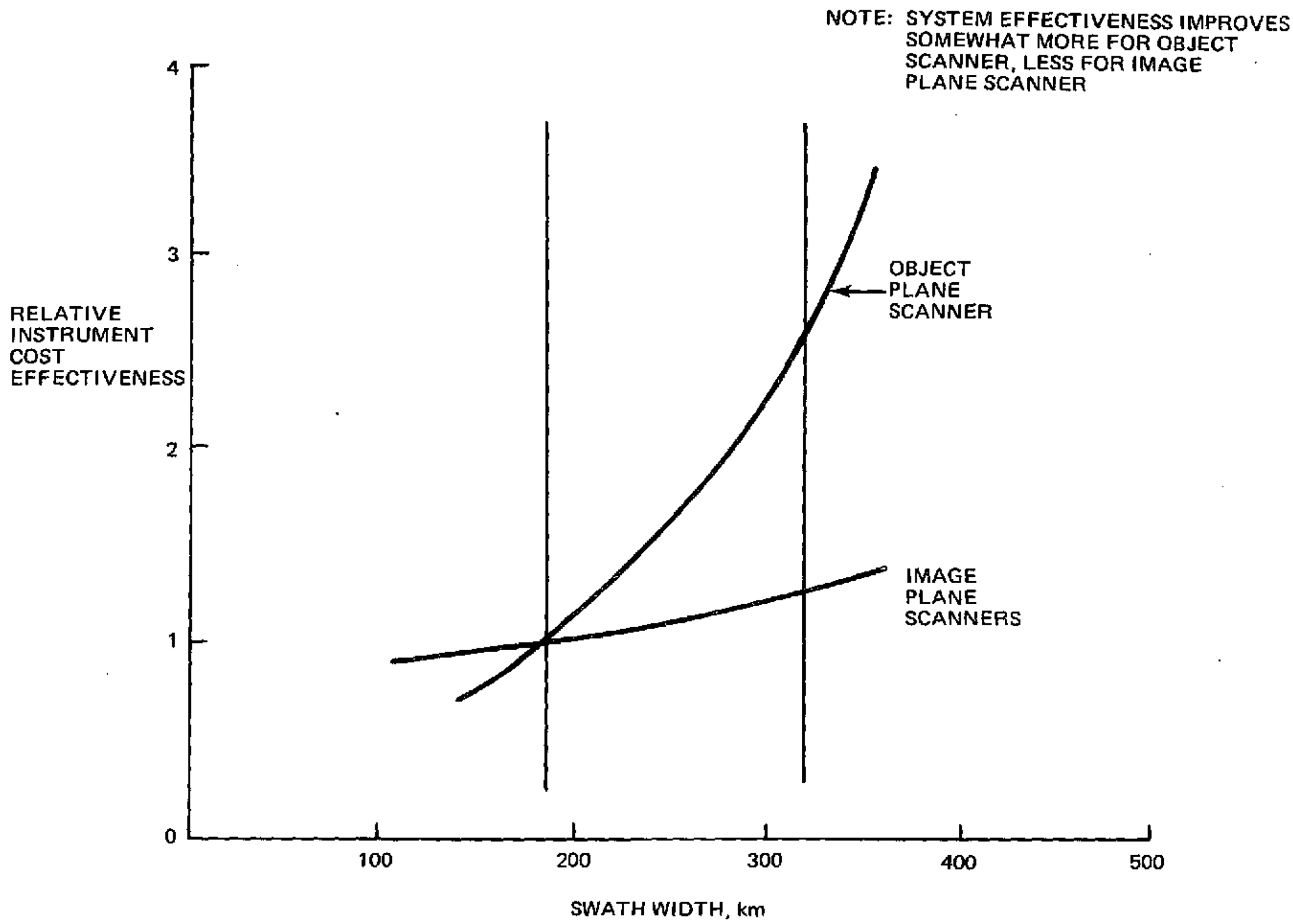
Fig. 3.4-3 Instrument Weight vs Swath Width

The object plane scanner, particularly in the version employing a telescope aligned axially to the launch and flight vector, was far more capable of meeting this higher utility level without incurring a large weight penalty. Fig. 3.4-3 shows swath width as a function of instrument weight.

3.4.1.4 INSTRUMENT COST EFFECTIVENESS

The increased utility of an advanced TM with a wide swath is obtained at virtually no weight increase or design cost in the case of the object plane scanner. Therefore, its cost effectiveness rises rapidly with swath.

The image plane scanners suffer a considerable entrance pupil growth and therefore weight growth with increasing swath. In addition, the design complexity grows. Therefore their cost effectiveness is much lower.



3-145

Fig. 3.4-4 Cost Effectiveness Vs Swath Width

As the weight of the image plane scanners increases, a change of booster may be required which would result in an overall loss in system cost effectiveness. Fig. 3.4-4 shows instrument cost effectiveness as a function of swath width.

3.4.1.5 UNCALIBRATED COST SAVINGS

There are a number of design decisions which can be made at this time which will lead to lower hardware, development and program management costs.

A significant saving in both hardware and design costs can be achieved by employing only solid state detectors. The performance achievable is within 20% of that of PMT's if nominal cooling is employed to lower preamplifier noise. The 20% can be recovered by increasing the aperture slightly. The weight penalty would be balanced by the detector weight saving.

The data system should employ a digital interface between the instruments and data link. This will result in a measurable cost saving in program management, design and checkout cost. It will also simplify evaluation of the equipment and its supplier's performance.

The choice of 1.0 samples per pixel recognizes the greater importance of radiometric accuracy than the ability to reproduce regular repetitive patterns in the land resources management mission.

The on-board data reduction equipment designed to meet the low-cost ground station LCGS requirement is equally capable of generating an output signal emulating the 5 (or 7) band MSS. Thus, on development flights, the advanced TM can act as a full backup to the MSS as far as DOI is concerned, while simultaneously generating a TM output. The same equipment can also generate a pseudo-HRPI output when not required to emulate the MSS. This should allow some HRPI mission simulation on early TM missions.

All of the above outputs will be available in an 18 mb/sec data stream which can be delivered over the DOI or LCGS data link.

The growth potential of the advanced TM leads directly to a HRPI as a minor hardware mod, change detector array, scan mirror rate, and mount the entire unit on an axle for offset (delete cooler).

The unit is also compatible with a future improvement in sensitivity by replacing the image plane detectors with a CCD or similar detector system.

See Table 3.4-1.

Table 3.4-1 Uncalibrated Cost Savings

• USE ALL SOLID STATE DETECTORS	
• COOL TO APPROACH PMT PERFORMANCE	
• USE A DIGITAL INTERFACE TO DATA LINK	
• USE 6 BIT ENCODING TO SAVE ON GROUND STATION	
• USE 1.0 SAMPLE/PIXEL	– TO SAVE ON GROUND STATION
	– TO RELIEVE DATA LINK
	– TO IMPROVE RADIOMETRIC QUALITY
• PROVIDE DOI COMPATIBLE OUTPUT (MSS)	
• PROVIDE PSEUDO-HRPI OUTPUT ON TM	
• ACQUIRE HRPI AS TM MODIFICATION	
• USE IMAGE PLANE SPECTRAL FILTER RING	

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3.4.2 ORBIT/LAUNCH VEHICLE

The recommended orbit for the EOS Land Resources Mission should be sun synchronous, with a minimum altitude having acceptable orbit decay, swath sideslip and ground station coverage. The maximum altitude should result in the selection of a low cost booster and be capable of direct shuttle service. The specific altitude selected should be optimized for instrument swath width and repeat cycle time. Our studies indicate that an altitude range of 365-385 nm. is best suited to these EOS requirements.

A promising sun synchronous orbit for EOS missions A, B and C is 680 km (366 nm) when using an instrument with a 185 km swath width. This orbit has a 17 day repeat cycle, and a 14 nautical mile swath overlap. The swath overlap to an adjacent swath occurs within three days. When using a HRPI instrument with 30° offset pointing for CONUS viewing, 90% of a reference swath may be viewed within three days.

For a 9 day repeat cycle time an acceptable orbit within the recommended altitude is 708 km (382 mn). This orbit should be operated with an instrument whose swath width is 330 Km (178 nm) and has a swath overlap of 14.8 nautical miles. The swath overlap with an adjacent swath will occur within two days.

The 680 km orbit was evaluated for orbit decay and was found operationally acceptable; that is, node side slip at the equator less 2.1 nm in 30 days and 8.5 nm for a 60 day period, for a 1979 nominal atmosphere. The orbit decay will become less severe as time progresses away from solar maximum 1978-79. The Jacchia Atmospheric Model was used for this analysis.

The lower of the two orbits, 680 km, was evaluated for ground station coverage. The most critical ground station for CONUS coverage is Sioux Falls. Since this Data Acquisition ground station is not yet operational, the site survey data, which predicts a clear field to within two degrees of the horizontal, was used. Our analysis indicates complete coverage of CONUS for the 680 km orbital altitude and higher altitudes. All of the state of Florida, the most critical location, is covered.

The projected EOS payloads were developed for each EOS mission, A through F. A detailed weights analysis of the spacecraft design was used in developing payload weights. For each mission, the lowest cost booster that was capable of lifting the payload to the

EOS orbit was selected. The Delta 2910 launch vehicle was selected for the A mission. Because of the heavy instrument for the B and C missions, launch vehicles with heavy lift capabilities were selected; the Delta 3910 for the B mission and a Titan III B/NUS with a circularization solid rocket motor, for the C mission. For the 680 km operational orbital altitude, recommended for mission A, B and C, direct shuttle service is possible. The selection of higher altitudes would require the addition of a multi SRM kick stage to transfer the spacecraft to a lower orbit for shuttle service.

All EOS missions are launched from the Western Test Range (WTR) except mission F, which will be an ETR launch. This mission will also require a 7-segmented SRM Titan launch vehicle T III C7. The T III C7 has the Transstage as the upper stage. A TE 364-4 SRM stage is used to circularize at the orbital altitude of 19,323 nautical miles, 0^o inclination. For shuttle servicing, missions E and F will require special provisions.

Because of the poor reliability (0.89) of the low cost Delta 2910 launch vehicle, identified for missions A and D, it is recommended that program planning take into account the possibility of a failed launch.

3.4.3 DATA OPERATIONS

The Central Data Processing Facility (CDP) performs the data operations for the EOS program. Cost drivers that impact the CDP include the daily data volume (throughput), the level of processing of this data (radiometric-Level 1; geometric correction and resampling-Level 2; ground control point location and grid resampling-Level 3), and the percent of data that is processed at the various levels, the number of users, and the amount of output products required by the users. In order to exercise the cost impact on the configuring and operation of a CDP of these and other parameters, a cost/throughput model was constructed with interrelates the pertinent drivers. The model was then reduced to a computer program. This program was exercised for a number of example cases and two CDP configurations (mini - computer systems and general purpose processor).

Table 3.4-2 shows total CDP costs as a function of input data load, output product level, and product mix for the two configurations. All data is processed to Level 1. The product mix is the percentage of data processed to Level 3 with the remainder of the data processed to Level 2 plus the Level 1 processing.

Tables 3.4-3 and 3.4-4 summarize the 5-year costs for product generation, and the costs of image processing equipment, for the example.

Table 3.4-2 Five-Year Total CDP Costs

($ Millions)

PRODUCT MIX	% LEVEL 3		SCENES/OUTPUT PRODUCT LEVEL 20 MIN	20 MAX	90 MIN	90 MAX	400 MIN	400 MAX
BILINEAR	0	CONFIGURATION 1	5.4	10.1	20.7	40.5	83.5	169.6
BILINEAR	100	CONFIGURATION 1	5.5	10.2	20.9	40.7	84.3	170.4
CUBIC CONVOLUTION	100	CONFIGURATION 1	7.9	12.7	33.7	53.4	142.1	228.1
BILINEAR	0	CONFIGURATION 2	9.5	14.2	41.4	61.1	177.3	263.3
	100	CONFIGURATION 2	9.6	14.3	41.9	61.6	179.6	265.6
CUBIC CONVOLUTION	100	CONFIGURATION 2	16.6	21.	77.0	96.8	339.1	425.1

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Table 3.4-3 Five-Year Total Expendables Plus Product Copier Costs

($ Millions)

TOTAL SCENES	OUTPUT PRODUCT LEVEL MINIMUM	MAXIMUM
20	1.48	4.92
90	5.51	21.19
400	23.30	92.97

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Table 3.4-4 Total Image Processing Equipment Costs

($ Millions)

TOTAL SCENES		% LEVEL 3 BILINEAR 0	INTERPOLATION 100	CUBIC CONVOLUTION 100
20	CONFIGURATION 1	0.88	0.9	2.09
90	CONFIGURATION 1	4.01	4.1	9.55
400	CONFIGURATION 1	17.84	18.2	42.45
20	CONFIGURATION 2	2.40	2.5	5.76
90	CONFIGURATION 2	10.95	11.2	26.29
400	CONFIGURATION 2	48.68	49.7	116.82

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Exercising the cost/throughput model and analyzing the results of the example leads to the following conclusions:

- There are a large number of potential cost drivers, any one of which can become a large cost contributor when its associated requirements parameters are increased.

- No significant cost breakpoints were found for flexible processors. The cost appears to behave roughly as a linear function of the throughput.

- The number of formats has a minimal impact on cost.
- The impact of the number of users depends on the average fraction of the data received by each user in each data product type.
- The data processing expendables can become a major cost driver.
- The detailed characteristics and mix of processing algorithms are a significant cost driver.
- The minicomputer was found to be uniformly lower in cost than the general purpose processor. However, for large data volume, neither machine represents an economical solution beyond the R&D stage. It is expected that special purpose processors will provide a significant reduction in the cost of processing at high data throughput volumes.

3.5 TRADE METHODOLOGY

Although at this time the design effort has not been completed, EOS candidate configurations and their associated mission models have been developed to cover the spectrum of required mission capabilities and selected program cost budgets. This set of configurations has been selected from a much larger set of feasible options using the results of design analyses and associated cost information to "screen out" those options which were judged to be deficient on the basis of either system cost or performance impact. From the outset, the objective of placing equal effort on both the initial EOS missions and the development of a low-cost spacecraft has influenced our approach toward the development and selection of options. The general procedures in the process are consistent throughout, but the approach has been tailored to the EOS programmatic mission model received from GSFC during course of the study. The approach taken for the initial missions depends heavily on groundrules input such as sensors to be carried, number of spacecraft and flights in the mission model, and the booster to be used. It is illustrated in Figure 3.5-1.

Since the EOS-B and B' missions carry a complete complements of instruments which have never flown, and include the new capability of offset pointing of an instrument, the process shown in the figure is modified for these missions. In this case we restrict the sensor complement and possible boosters by groundrule. However, the choice of a booster and the orbit had to be decided by reconciling the sometimes conflicting requirements of user revisit, atmospheric drag, and shuttle/booster payload to orbit capabilities.

The EOS missions downstream of B and B' have been grouped under the general category of "follow-on missions". These missions were often incompletely defined. Also, even in those cases where there was no lack of definition, the impacts of the missions on the EOS was unknown. Thus the option development and screening process shown in the figure must be again modified when applied to the "Follow-on EOS Missions". EOS configuration options which would cover the spectrum of requirements within the cost targets were developed. This common objective required that in each case a system synthesis task be performed. This task provides a final screen of options prior to design development by combining and comparing the parameters of program costs, spacecraft/payload weight, and booster capabilities for programmatic options providing varying capability. This task, as well as each process, is described in greater detail in the sections which follow.

3.5.1 EOS A AND A' OPTION DEVELOPMENT AND SCREENING

The EOS-A and -A' missions are the near-term, most-well-defined, and most-understood of the missions in the EOS program mission model provided by GSFC and shown in Figure 3.5-1. The characteristics of these missions are shown on Table 3.5-1. Since these are the best known, the establishment of the options for study relied the least on tradeoff information and was heavily dependent on GSFC groundrules, which were developed from

Table 3.5-1 EOS and Follow-On Mission Weight and L/V Performance

ITEM DESCRIPTION	EOS–A,A′	EOS–B,B′	EOS–C	EOS–D (SEASAT–B)	EOS–E (TIROS–0)	EOS–F (SEOS)	SEASAT–A	SMM
BASIC SPACECRAFT – LB*	1680	1722	1752	1680	1752	1776	1680	1680
S/C MISSION PECULIAR	31	44	705	198	1841	256	122	216
INST. MISSION PECULIAR	400	700	920	267	237	549	254	467
INSTRUMENTS	613	853	1753	706	770	2300	602	1431
SUBTOTAL	2724	3319	5130	2851	4600	4881	2658	3794
WEIGHT SAVING OPTIONS	112	–	–	166	126	–	–	–
TOTAL SPACECRAFT WEIGHT	2612	3319	5130	2685	4474	4881	2658	3794
LAUNCH VEHICLE CAPABILITY	2660	3730	5150	2825	4500	5600	3350	3900
PAYLOAD MARGIN – LB	48	411	20	140	26	719	692	106
LAUNCH VEHICLE	DELTA 2910	DELTA 3910	T III B SSB/NUS	DELTA 2910	T III B SSB/NUS	T IIIC-7 TE-364-4	DELTA 3910	DELTA 2910

*INCLUDES RETRIEVAL/RESUPPLY MECH, EXTRA BATTERY

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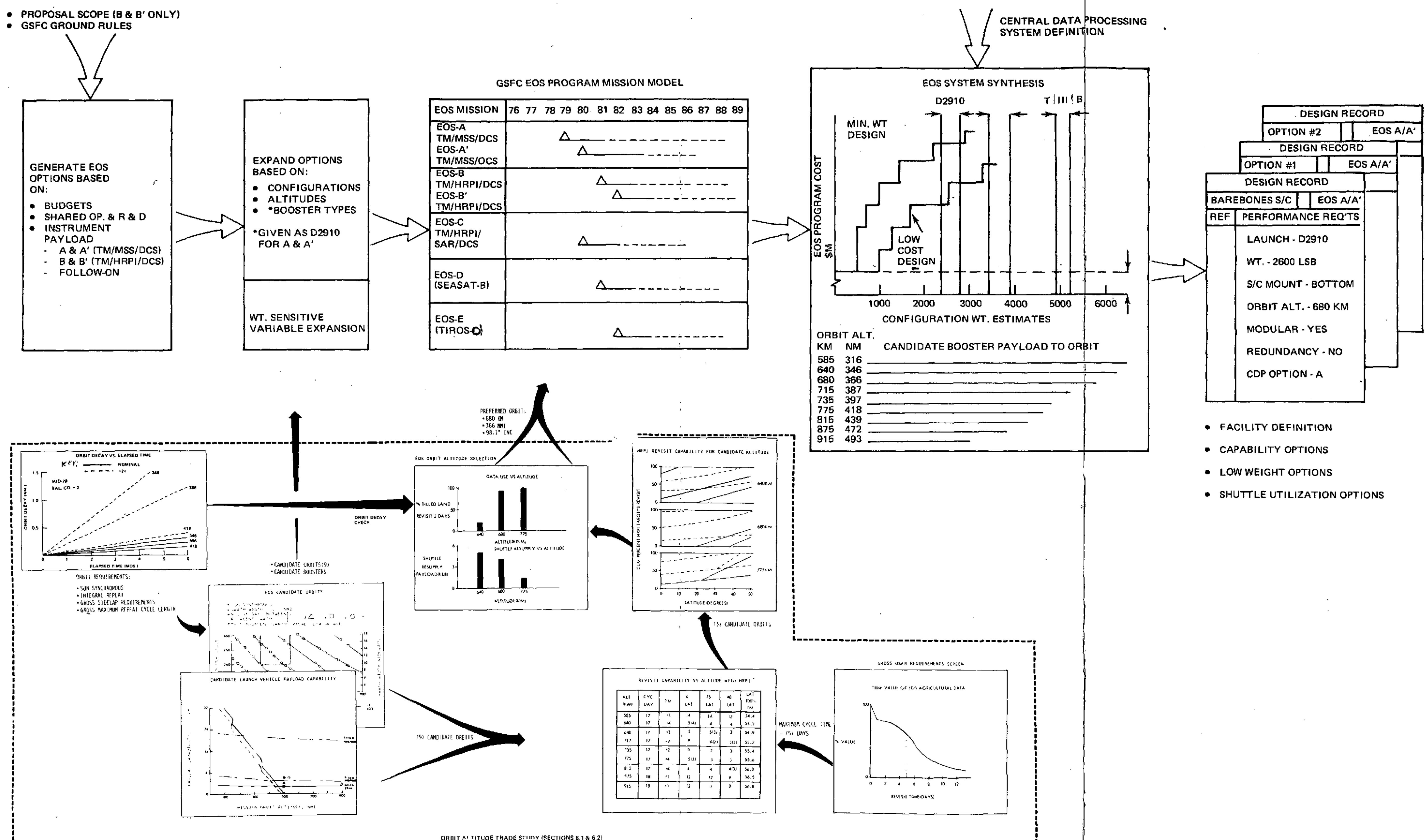

Fig. 3.5-1 Option Development and Screening Process

well known near-term requirements. For example, the need for near-term cost sharing between the NASA and user agencies necessitated the combining of an R&D sensor with an operational sensor for the first EOS mission. This requirement was enforced by the need of the user to provide a continuous operational capability for Earth observations. The near-term requirements also dictate the inclusion of the ERTS orbit of 915 km (493n. mi.) in the orbit considerations, even though the unassisted Shuttle payload to that orbit is negligible. These groundrules greatly restrict the options which could be considered, and thus greatly simplified the systems methodology applied. The groundrules which were used for these missions are given below:

- Maximum of two spacecraft launched at one-year intervals beginning in 1979.
- 5-Band MSS operational payload with a TM R&D payload.
- 915 km orbit in addition to the follow-on mission orbit.
- Use of the Delta 2910 launch vehicle.

The system synthesis task was performed to determine how much capability could be incorporated in the EOS design within the launch constraints of the Delta and the selected program cost targets. Results are shown in Figure 3.5-2, 3.5-3 and 3.5-4. In order to investigate this issue, the total program cost was plotted vs. weight for the "barebones" spacecraft design. This design included no redundancy, no special provisions for shuttle resupply and retrieval, and no operations cost. This option resulted in a cost of $14.6M and a weight of 1421 pounds. For the single spacecraft case, the costs of one year of operations (i.e., Network Ops, Control Center Ops, Data Processing, etc.) have been added to the fixed cost as well as the additional dollars for instrument accommodation effort. This results in a total cost of $45M and a total weight of 1421 pounds for a one-year spacecraft. To this weight and cost were added the OAS and the instrument support structure needed to support the 5-Band MSS, the TM and DCS, and the associated data handling equipment. Then the instruments themselves and the wideband communications were added, resulting in an observatory weight of 2400 pounds and a cost of $134M. To this weight were added optional capability features in the following order (where the order is an indication of their priority):

- Redundancy for a two-year MMD
- Resupply Capability
- Retrieval Capability
- Wide Band Video Tape Recorder.

With the addition of each capability increment, the payload weight approaches the limit of the Delta 2910 at the orbits of interest. First, the capability at 915 km is surpassed, and then, finally that of the lower orbits also. In order to maintain the weight below the payload-to-orbit-capability of the booster, it is necessary to include more expensive weight-saving features in design (e.g., the use of composite structure, flexible rolled-up arrays). These features raise the cost of the spacecraft design, but allow for a launch on the low-cost Delta 2910 booster. This approach toward weight savings is continued until either no more savings are possible, or until the cost increases exceed the cost increase of going to the Delta 3910 booster. At this point, the more expensive booster results in the more economical program. At present, the results of the study indicate that the maximum capability required for the A and A' mission can be incorporated into a configuration which meets the constraints of the Delta 2910. However, significant weight increases in the future might modify this result.

3.5.2 EOS-B AND -B' OPTION DEVELOPMENT AND SCREENING

The EOS-B and -B' missions require a more detailed systematic approach, since fewer requirements are fixed, and the offset capability of the HRPI influences the choice of orbit. The modified process shown in Figure 3.5-3 results in the selection of EOS B and B' candidate configurations and their associated mission models. The process expanded the number of options proposed for the instrument complement of TM and HRPI by considering lower-level design variables. This expansion was followed by the reduction of the number of options by successive application of high-level technical and cost screens. The screens applied were:

- Orbit Altitude/Booster Screen
- Gross Cost Screen
- Gross User Requirements Screen
- Gross Mission Model Screen

The application of these screens narrowed the number of options to several mission models. The configurations corresponding to these models are the primary configurations toward which our designers have directed their effort for the B and B' spacecraft.

3.5.3 EOS OPTION DEVELOPMENT AND SCREENING FOR FOLLOW-ON MISSIONS

The follow-on EOS missions are the least defined, and therefore require the greatest amount of systematic study to develop sets of reasonable non-conflicting design requirements.

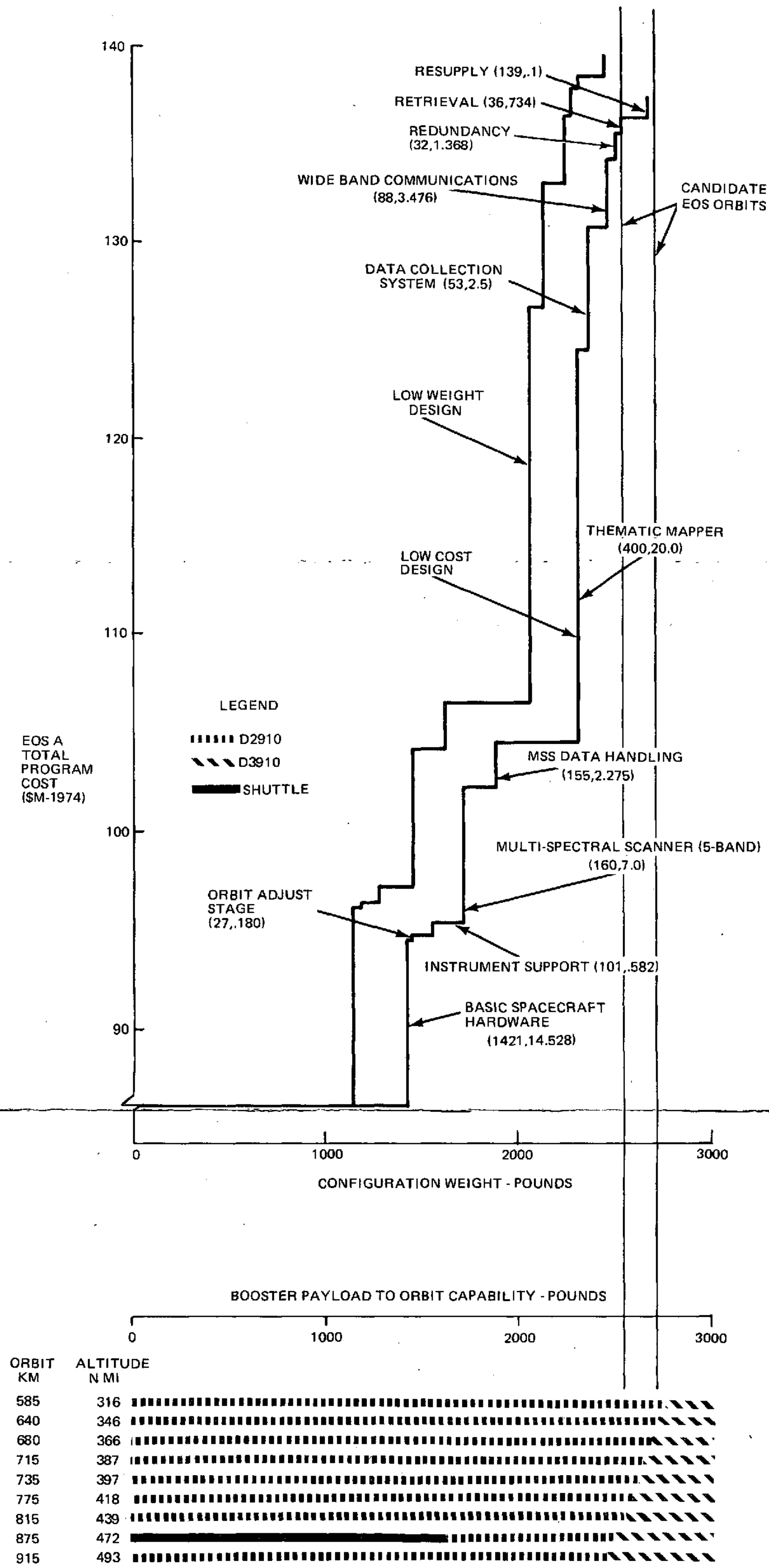

Fig. 3.5-2 EOS-A System Synthesis

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FOLDOUT FRAME 2

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FOLDOUT FRAME 1

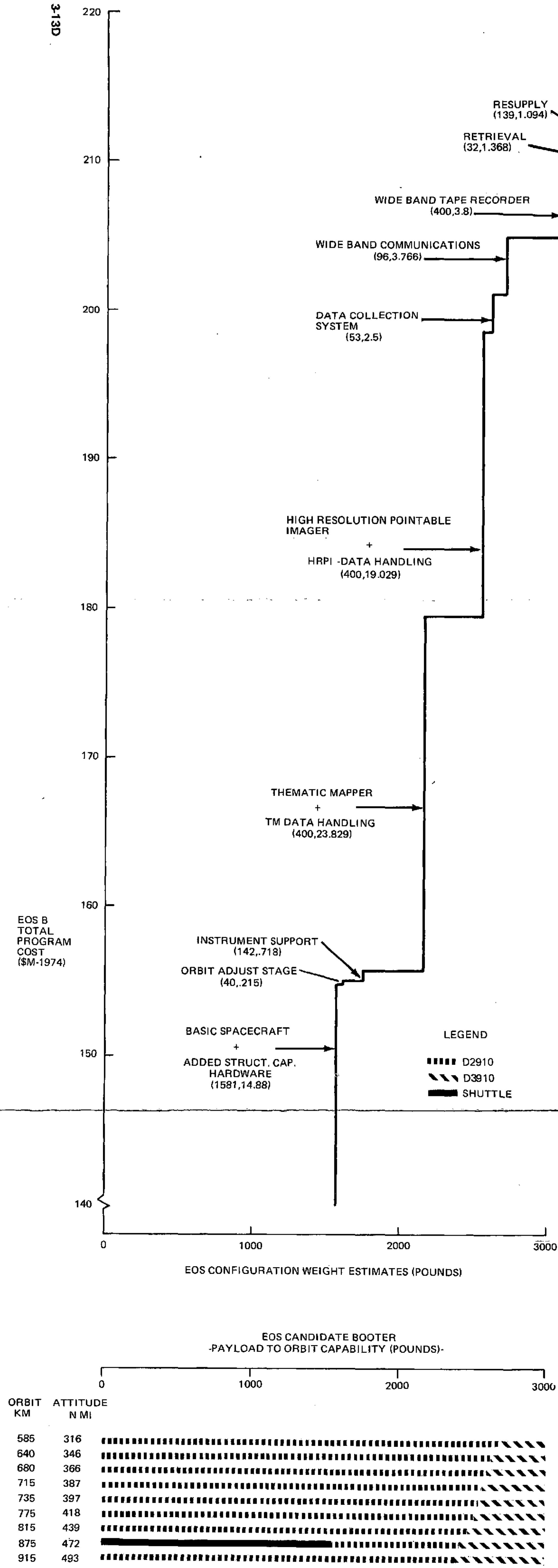

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Fig. 3.5-3 EOS-B System Synthesis

FOLDOUT FRAME 1

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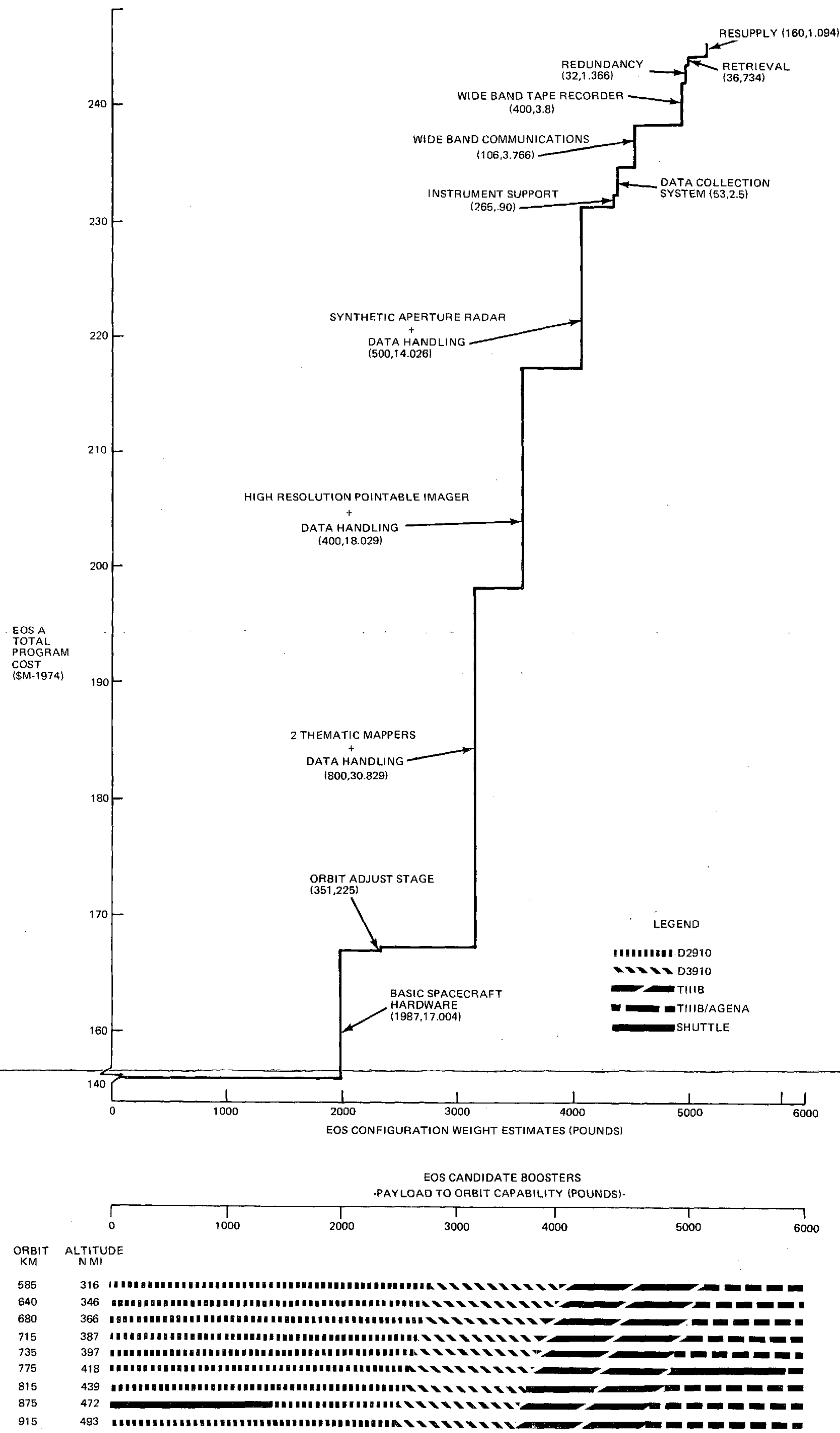

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Fig. 3.5-4 EOS-C System Synthesis

The development of the trade areas which have the most impact on each of these missions, and the level to which these design drivers should impact the baseline EOS-A design, is the purpose of the on-going Design Growth Economic Study reported is Section 6.17 of this volume. The results of this study will indicate how the systematic approach to design development described here should be tailored to each of the EOS follow-on missions.

3.5.4 TAILORING OF SYSTEM METHODOLOGY

- Early Missions
  - Well Defined Specific Objectives
  - Satisfy Operational as well as R&D needs
  - Design option development simplified
  - No extensive trades in well defined areas
  - Instrument payload and possibly booster defined by GSFC direction
- Later Missions
  - Objectives less firm
  - Conceptual Instrument Payload
  - Analysis and trades replace experience
  - Spectrum of boosters must be considered

3.5.5 INTEGRATION OF TRADE STUDY RESULTS

Figure 3.5-5, which appeared in our Proposal, depicts our original approach to the problem of integrating trade study results. Evaluation of the many different and loosely-constrained options was to be accomplished by means of a system Figure of Merit which would combine the results of trade studies for each candidate, and numerically evaluate each candidate.

As a result of the GSFC direction specifying the A and A' missions, the diagrammed approach was modified in the following ways:

- The MSS was included in the instrument complement for A and A'
- Other options were eliminated
- The booster was specified as the Delta 2910.

The system synthesis task has been performed, providing weight and cost data for the spacecraft and DMS options. The design/cost tradeoffs have been completed in at least preliminary fashion.

As shown in Figure 3.5-5, the original approach developed options in instruments, data transmission, data management, and data processing for different cost targets to establish the capability obtainable at different budget levels. The Figure of Merit combined the results of trade studies conducted for these options to establish the most effective configurations for each budget level.

Figure 3.5-6 shows the interfaces between the Program Effectiveness Model and the individual trade study areas. The trade studies provide data and inputs for each option to the Effectiveness Model. The model then combines data from all trade areas to evaluate options on a programmatic basis. One can then see the programmatic impact of individual trades.

Emphasis during the study to date has been on the EOS A, A', B, and B' missions. The instrument and data options were constrained by the detailed specifications of the missions and systems. Therefore, the System Effectiveness model has not yet been exercised for these well-defined missions. The degree of definition of the systems lessens the possibility of many interactions between trade study areas. Therefore, it is expected that the model will reveal very few results not discovered already by the individual trade studies.

Since configurations for A and A' are constrained, it is not expected that many changes will result from the use of the FOM. In this case, the FOM will be used to evaluate the performance of the resulting system.

During the next phase of the study, the System Effectiveness model will be used to evaluate the many alternative options to perform these missions independently and using spacecraft having common elements. The trade study data obtained so far will be supplemented with data for other options available on the less-well-defined missions. These results will be reviewed on a programmatic basis using the FOM and total program costs. This programmatic analysis of potential design paths will provide the data necessary for making design decisions.

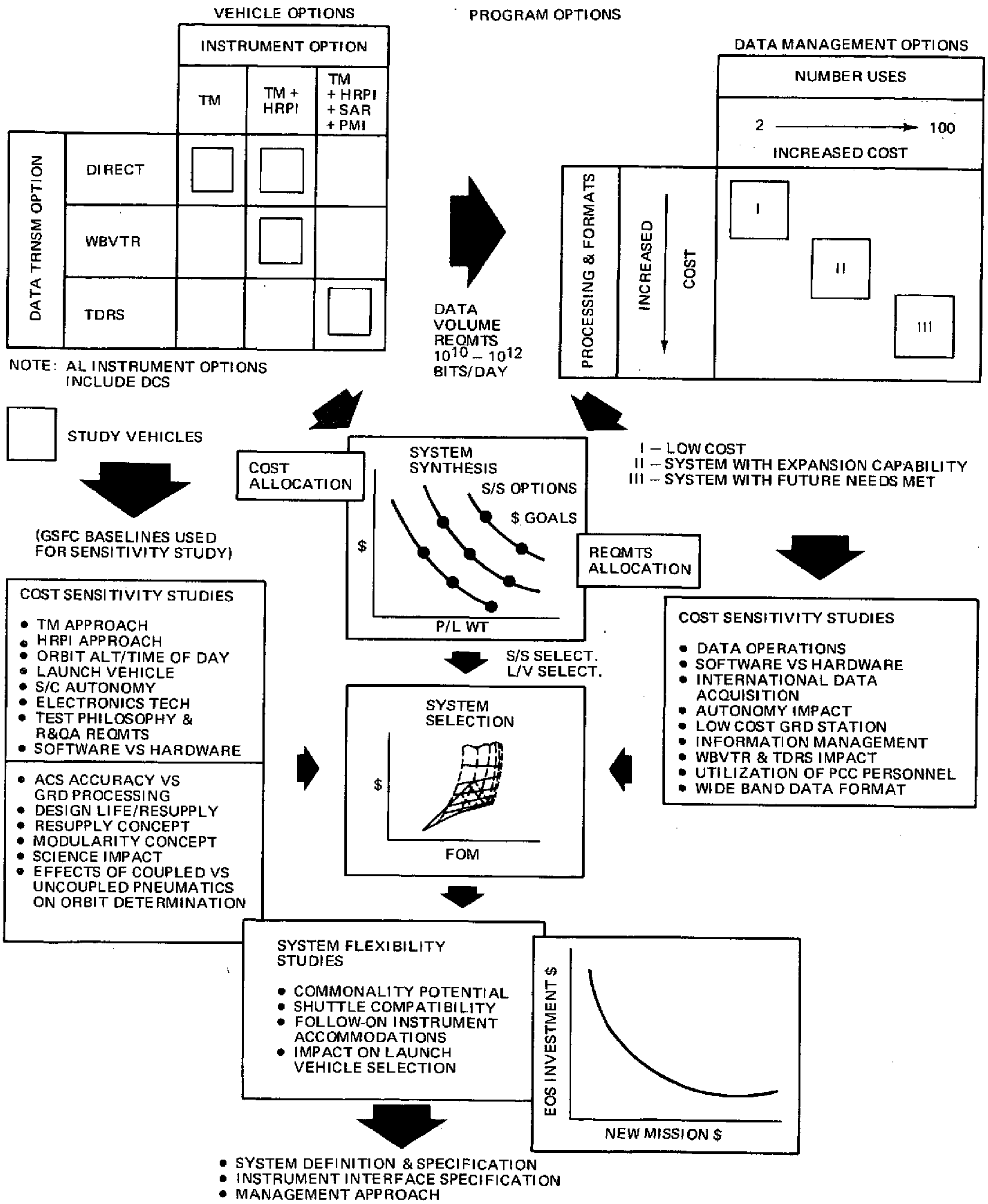

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Fig. 3.5-5 System Design Approach. Cost is a Significant Factor in Our Design Approach.

	REPORT NO. 1 ORBIT/LAUNCH VEHICLE												RPT #2		REPORT NO. 3 SYSTEM DESIGN/COST TRADEOFFS																									REPORT NO. 5 SYSTEM DESIGN & SPECIFICATIONS						RPT #6		
DATA AND OUTPUTS FROM THIS TRADEOFF OR DESIGN AREA . . . / , , , ARE INPUT TO THIS SECTION OF THE EFFECTIVENESS MODEL	ORBIT ALTITUDE	L/V PAYLOAD CAPABILITY	SHUTTLE PAYLOAD CAPABILITY	ORBIT TRANSFER/ADJUST REQUIREMENTS	GROUND TRACE PATTERN	FREQUENCY OF RE-COVERAGE	TRACKING STATION VIEW TIMES	SOLAR ILLUMINATION ANGLE	TIME OF LAUNCH	TIME SPENT IN EARTH SHADOW	ENERGY STORAGE REQUIREMENTS	DISTURBANCE TORQUES	INSTRUMENT CONSTRAINTS	INTERFACE SPECIFICATIONS	ORBITAL ALTITUDE/ORBIT TIME OF DAY	LAUNCH VEHICLE OPTIONS	SHUTTLE COMPATIBILITY	INSTRUMENTS APPROACHES	DATA OPERATIONS	SPACECRAFT AUTONOMY/SOFTWARE VS HARDWARE	ELECTRONIC TECHNOLOGY	INTERNATIONAL DATA ACQUISITION	FOLLOW-ON INSTRUMENT ACCOMMODATION	DESIGN LIFE/RESUPPLY INTERVAL	RESUPPLY CONCEPT	MODULARITY CONCEPT	SCIENCE IMPACT	ACS/GROUND PROCESSING	UTILIZATION OF CC PERSONNEL	EFFECTS OF COUPLED VS UNCOUPLED PNEUMATICS ON ORBIT DETERMINATION	WIDE BAND DATA FORMAT LINE	TEST PHILOSOPHY	R&QA REQUIREMENTS	COMMONALITY POTENTIAL	RELIABILITY	INSTRUMENT COMPLEMENT	GROUND LOGISTICS SYSTEM	ON-BOARD STORAGE CAPACITY	USER APPLICATIONS	INSTRUMENTS DESIGN, DCS	INSTRUMENT MISSION PECULIARS	SPACECRAFT DESIGN	SPACECRAFT MISSION PECULIARS	DATA MANAGEMENT SYSTEM DESIGN	VEHICLE INTERFACES	SPACE SHUTTLE INTERFACES	SPACE SHUTTLE UTILIZATION	FLEXIBILITY COST/BENEFIT ANALYSIS
EFFECTIVENESS OF SYSTEMPARAMETERS	•			•	•	•											•	•				•	•	•	•	•	•	•						•	•	•			•	•		•		•		•	•	•
EFFECTIVENESS OF LAUNCH VEHICLE		•	•	•											•	•	•								•					•						•						•	•		•	•	•	•
QUALITY OF INSTRUMENT OUTPUT	•			•	•	•		•	•	•		•	•	•	•		•	•	•	•	•	•	•		•	•	•	•		•		•				•			•	•	•							•
QUANTITY OF INSTRUMENT OUTPUT	•									•			•	•				•	•	•	•	•	•		•		•				•					•		•	•	•	•							•
AVAILABILITY OF INSTRUMENT OUTPUT	•		•				•			•	•		•	•	•	•	•	•	•	•	•	•	•	•	•	•						•	•	•	•	•		•		•	•	•	•		•	•	•	•
QUALITY OF S/C SUPPORT		•	•	•	•	•		•	•	•			•	•	•		•	•				•		•	•	•		•		•						•			•	•	•		•			•	•	•
AVAILABILITY OF S/C SUPPORT		•	•	•						•	•	•	•	•		•	•			•	•		•	•	•	•						•	•	•	•	•				•	•	•	•		•	•	•	•
QUALITY OF THROUGHPUT							•												•		•							•			•					•		•	•					•				•
QUANTITY OF THROUGHPUT							•												•		•	•							•		•			•		•		•	•					•				•
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AVAILABILITY OF DATA PRODUCTS																													•			•	•		•		•		•					•				•

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Fig. 3.5-6 Interfaces Between System Effectiveness Model and Tradeoffs. FOM is Focal Point for all Tradeoffs.

S/C CONFIGURATION
STUDIES

4 - SPACECRAFT CONFIGURATION STUDIES

4.1 SPACECRAFT AND INSTRUMENT CONFIGURATION

4.1.1 BASIC SPACECRAFT REQUIREMENTS

A basic spacecraft configuration compatible with Delta, Titan or Shuttle launch vehicles was designed to support a significant number of follow-on satellite missions. The general requirements for the structure subsystem were established to support this goal. These requirements are:

- One vehicle configuration shall support EOS missions A, A', B and C (Table 4.1-1) as a minimum, and be usable in a wide variety of other missions.
- The configuration shall support three discrete standard subsystem equipment modules which include EPS, ACS and C&DH, and a mission-peculiar propulsion module.
- The module and core structure configuration shall allow for shuttle resupply of the modules with little or no change.
- The vehicle shall be capable of mating with and be launched by a Delta or Titan launch vehicle and have optional shuttle launch and retrieval capability.
- The vehicle shall be capable of supporting and operating with a wide variety of instruments and instrument mission peculiar equipment.
- The basic spacecraft configuration shall meet dynamic and static load requirements as defined in Tables 4.1-2, 4.1-3, 4.1-4 and 4.1-5. The spacecraft coordinate system is defined in Figure 4.1-1.

Table 4.1-1. Instrument Section Requirements

EOS MISSION	SPACECRAFT PAYLOAD	INSTRUMENT MISSION PECULIARS	ANTENNAS	SOLAR ARRAY	LAUNCH VEHICLE
A	(1) MSS (1) TM DCS	(1) 11 X 25 X 32 INCH RECORDER MODULE (1) 14 X 36 X 36 INCH IMP MODULE	(1) X-BAND STEERABLE (1) X-BAND SHAPED BEAM	155 SQ. FT 516 WATTS	DELTA 2910
A'	(1) MSS (1) HRPI DCS	SAME AS A	SAME AS A	SAME AS A	DELTA 2910
B	(1) TM (1) HRPI DCS	(1) 22 X 30 X 36 INCH RECORDER MODULE (1) 14 X 36 X 36 INCH IMP MODULE	SAME AS A	SAME AS A	DELTA 3910
C	(2) TM (1) HRPI (1) SAR DCS	SAME AS B	SAME AS A	230 SQ. FT. 766 WATTS	TITAN III B

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Table 4.1-2 Delta 2910 and Delta 3910 Load Factors

CONDITION	LIMIT LOAD FACTORS		ULTIMATE LOAD FACTORS (1)	
	LONGITUDINAL X	LATERAL Y OR Z	LONGITUDINAL X	LATERAL Y OR Z
LIFT-OFF	+2.9 −1.0	2.0	+4.35 −1.5	3.0
MAIN ENGINE CUTOFF	+12.3	0.65	+18.45	1.0

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Table 4.1-3. Titan III B/NUS Load Factors

CONDITION	LIMIT LOAD FACTORS		ULTIMATE LOAD FACTORS (1)	
	LONGITUDINAL X	LATERAL Y OR Z	LONGITUDINAL X	LATERAL Y OR Z
LIFT-OFF	+2.3 −0.8	2.0	+3.45 −1.2	3.0
STAGE I SHUTDOWN (DEPLETION)	+8.2 −2.5	1.5	+12.3 −3.75	2.25
STAGE I SHUTDOWN (COMMAND)	+10.8 −2.0	1.5	+16.2 −3.0	2.25

NOTES: 1. LIMIT LOAD FACTOR TIMES 1.5.
2. LOAD FACTOR CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD.
3. INCLUDES BOTH STEADY STATE AND DYNAMIC CONDITIONS.

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Table 4.1-4. Shuttle - Payload Bay Load Factors

CONDITION	LIMIT LOAD FACTORS (4)			ULTIMATE LOAD FACTORS (1)(4)		
	DIRECTION (2)			DIRECTION (2)		
	X	Y	Z	X	Y	Z
LIFT-OFF (3)	+1.7 ± 0.6	±0.3	+0.8	+2.55 ± 0.9	±0.45	+1.2 +0.3
HIGH Q BOOST	+1.9	±0.2	−0.2 +0.5	+2.85	±0.3	- 0.3 +0.75
BOOSTER END BURN	+3.0 + 0.3	±0.2	+0.4	+4.5 ± 0.45	±0.3	+0.6
ORBITER END BURN	+3.0 ± 0.3	±0.2	+0.5	+4.5 ± 0.45	±0.3	+0.75
SPACE OPERATIONS	+0.2 −0.1	±0.1	±0.1	+0.3 −0.15	±0.15	±0.15
ENTRY	±0.25	±0.5	−3.0 +1.0	±0.38	±0.75	−4.5 +1.5
SUBSONIC MANEUVERING	±0.25	±0.5	−2.5 +1.0	±0.38	+0.75	−3.75 +1.5
LANDING AND BRAKING	±1.5	±1.5	−2.5	±2.25	±2.25	−3.75
CRASH (5)	–	–	–	−9.5 +1.5	±1.5	−4.5 +2.0

NOTES: 1. LIMIT LOAD FACTOR TIMES 1.5 EXCEPT FOR CRASH.
2. POSITIVE X, Y, Z DIRECTION EQUAL FORWARD, RIGHT AND DOWN.
3. THESE FACTORS INCLUDE DYNAMIC TRANSIENT LOAD FACTORS.
4. THESE FACTORS DO NOT INCLUDE DYNAMIC RESPONSE OF THE PAYLOAD.
5. CRASH LOAD FACTORS ARE ULTIMATE AND ONLY USED TO DESIGN LOCAL PAYLOAD SUPPORT.

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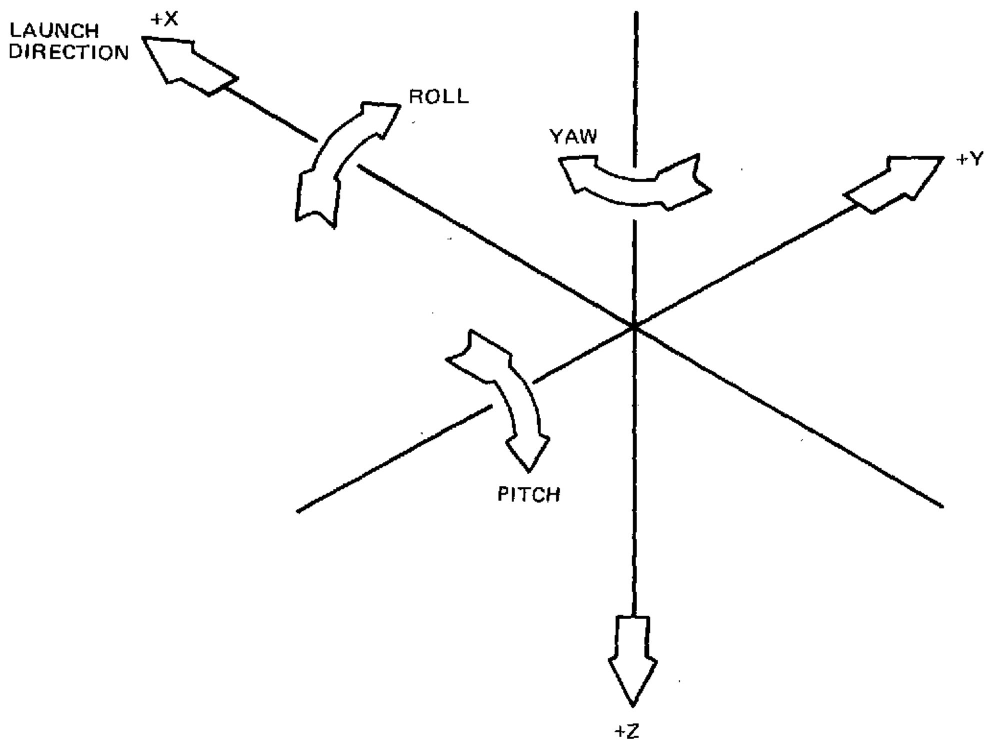
Table 4.1-5. Minimum Frequency Criteria

LAUNCH VEHICLE	MINIMUM FREQUENCY (Hz)		REFERENCE
	LONGITUDINAL	LATERAL	
DELTA	35	15	1
TITAN III B/NUS	20	10	2
SHUTTLE	N.D.	N.D.	3

N.D. = NOT DEFINED



3-1 Fig. 4.1-1 Spacecraft Coordinate System

4.1.2 SPACECRAFT MOUNTING ON LAUNCH VEHICLES

4.1.2.1 DELTA LAUNCH VEHICLE MOUNTING

The basic spacecraft is configured to be bottom-mounted when launched on a Delta launch vehicle, as shown in Figure 4.1-2. Provisions for transition ring mounting for launch or retrieval are inherent in the design and can be provided if required. We have selected the bottom mount for several reasons. The most significant one is ease of separation

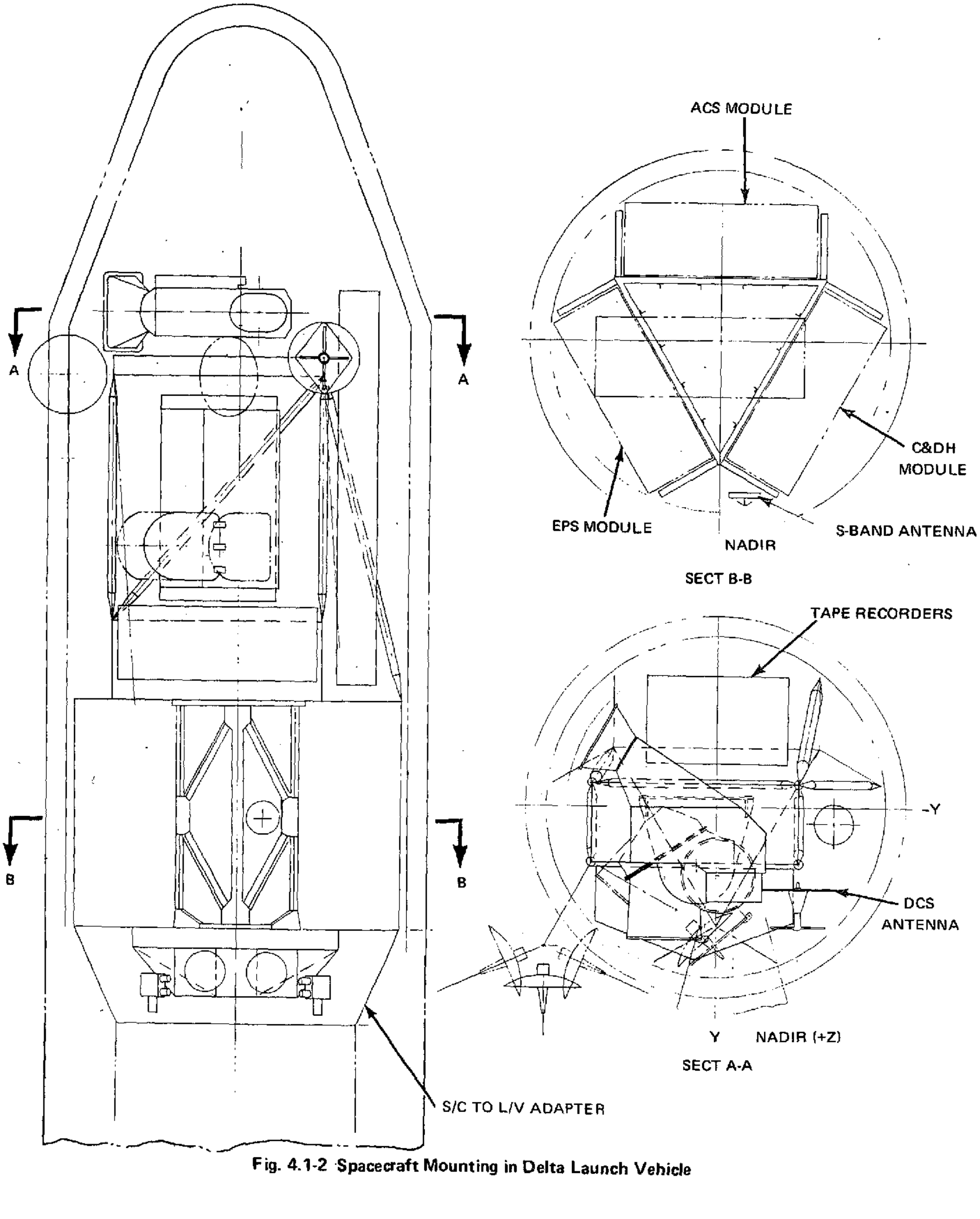


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Fig. 4.1-2 Spacecraft Mounting in Delta Launch Vehicle

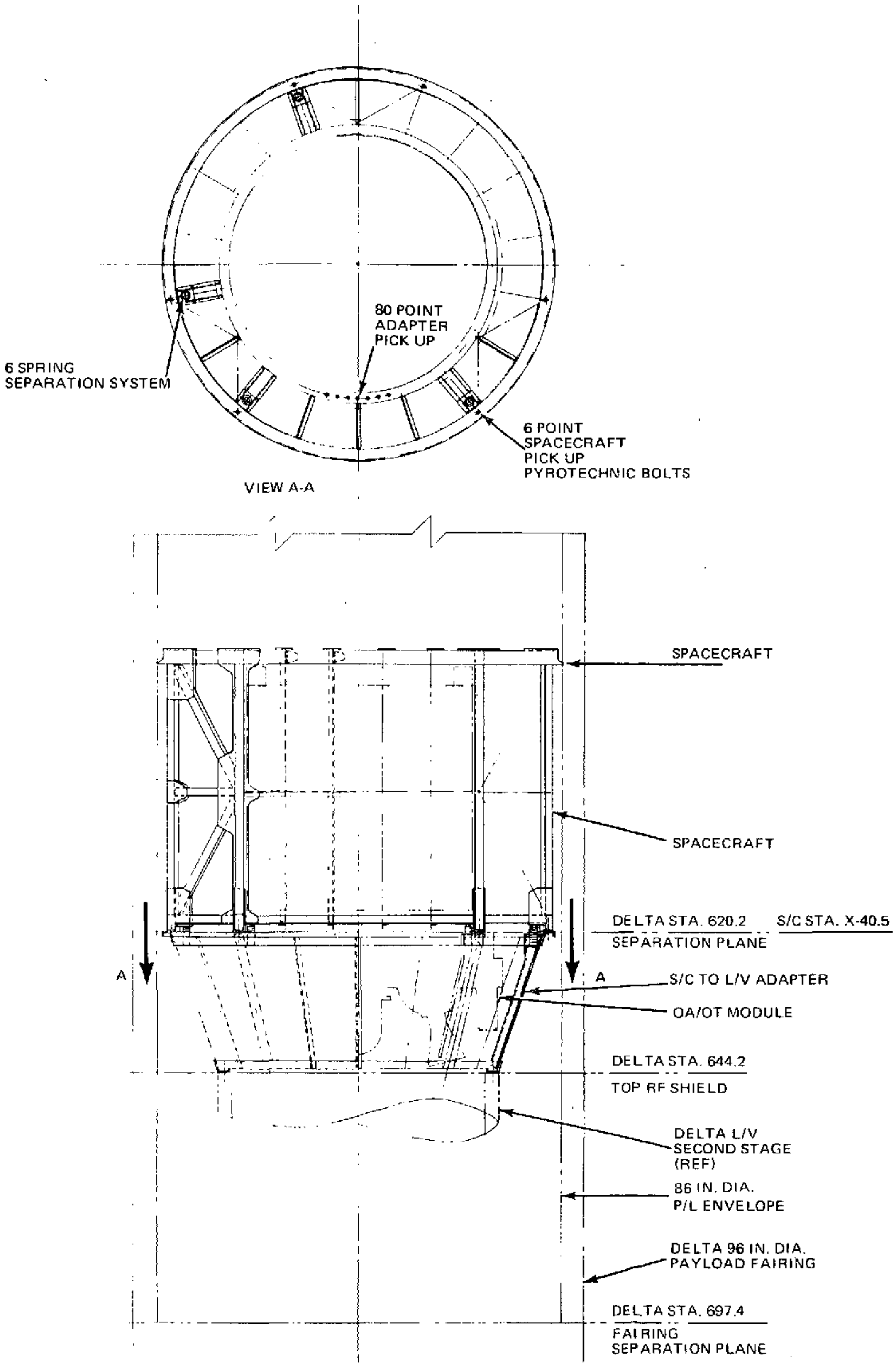
of the spacecraft from the launch vehicle adapter. The separation is performed in an unobstructed volume with an inexpensive, light-weight, well proven separation system. The interstage adapter is of conventional design (Figure 4.1-3), with possibility of using an existing design in some configurations. The adapter shown in Figure 4.1-3 would be a new design extending to the 86" periphery of the spacecraft as shown in Figure 4.1-2. This design allows clearance for the RCS/OAS module located at the spacecraft lower bulkhead to facilitate resupply. Estimated first unit recurring cost of this design would be \$54K. A standard Delta 5724 adapter of smaller diameter could be used costing approximately \$35K, but requires a smaller RCS/OAS system with consequent difficult mounting of the thrusters. Table 4.1-6 shows a cost and weight comparison of the two systems.

Table 4.1-6

Comparison of Adapter Costs

New Adapter	Recurring Cost	Non-Recurring	Weight
	\$54K (first unit) \$44K (4th unit)	\$150K + test	95 lb
Delta 5724	\$35K	Not avail.	113 lb

The transition ring amount requires the spacecraft to be withdrawn through a close clearance (approximately 1") adapter for a distance of over 5 feet. This will require a guided withdrawal to eliminate the possibility of a hangup between spacecraft and adapter. The guide mechanism could represent a significant weight penalty (estimated at 60 lb) and an expensive design and development program. An adapter of special configuration would have to be designed to reach from the launch vehicle to the transition ring. Our studies show an effective weight penalty of 205 lb if the external launch vehicle fairing is split to support the transition ring. A graphite/epoxy external adapter yields no reduction in effective launch weight compared with that of a bottom-mount design, and requires a significant increase in both cost and development difficulty. Table 4.1-7 shows this weight comparison. Cost estimates have not been developed because the weight penalty is prohibitive.



3-3

Fig. 4.1-3 Spacecraft/Launch Vehicle Adapter

Table 4.1-7. Weight Penalty for Delta L/V Split Shroud

FAIRING SECTIONS	CURRENT	SPLIT SHROUD – STA 581.7 AL LOWER	SPLIT SHROUD – STA 581.7 GFRP LOWER
UPPER FAIRING SECTION		(725)	(725)
FAIRING	825	460	460
BASE FRAME	70	70	70
LONGITUDINAL JOINT	60	40	40
SEPARATION	200	130	130
ELECTRICAL & MISC.	45	25	25
LOWER FAIRING SECTION		(515)	(310)
FAIRING	–	365	210
FRAMES	–	130	80
ELECTRICAL & MISC.	–	20	20
TOTAL PAYLOAD FAIRING	1200 LB	1240 LB	1035 LB
EFFECTIVE (25%) WEIGHT-STAGED	300	180	180
FAIRING CARRIED INTO ORBIT	–	515	310
TOTAL EFFECTIVE WEIGHT	300 LB	695 LB	490 LB
WEIGHT PENALTY FROM CURRENT	–	+395	+190
DELETE PAYLOAD ATTACH FITTING		-190	-190
NET PENALTY TO CURRENT EOS		+205 LB	– LB

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Our studies show no significant spacecraft structural penalty for a bottom mount. Since the instrument payload structural configurations differ significantly for the required combinations of payloads, the various payload support structures will attach to a variety of locations on the spacecraft. For the transition ring mount system, the load paths, and therefore the flexibility could be complex relative to a base mount system. Preliminary studies show that the basic spacecraft is approximately the same weight for either the transition ring system or the base mount system.

4.1.2.2 TITAN LAUNCH VEHICLE MOUNTING

When the basic spacecraft is launched by a Titan III vehicle, the clearance problem is reduced. The 86" O.D. of the spacecraft combined with the 110" I.D. of the shroud results in a 12" radial clearance, which significantly reduces the spacecraft extraction problem. For this installation, we recommend adding a 110"-diameter ring to the basic S/C at its transition ring station and mounting it on an extended booster adapter. The Martin Company has proposed such an extension of the booster skin to the ring level. The assembly weighs 350 lb and costs $300K recurring and $400K non-recurring. We do not recommend a bottom mount for this configuration because its narrow base would make the assembly top-heavy.

4.1.2.3 SPACECRAFT/SHUTTLE ATTACHMENT

The Space Shuttle launch and retrieval of the EOS requires a modified clamp type separation mechanism at the S/C upper bulkhead. This support configuration is compatible with the Flight Support System (FSS) suggested by the Shuttle contractor and the SPAR/DSMA designers of the Special Purpose Manipular System. The basic difference between the GAC concept and the GSFC baseline transition ring assembly is that the GAC concept supports six discrete mount fittings of the triangular S/C configuration and the GSFC has a continuous ring system. This is shown in Figure 4.1-4. Elimination of the continuous mounting ring results in a S/C weight saving of 75 lb.

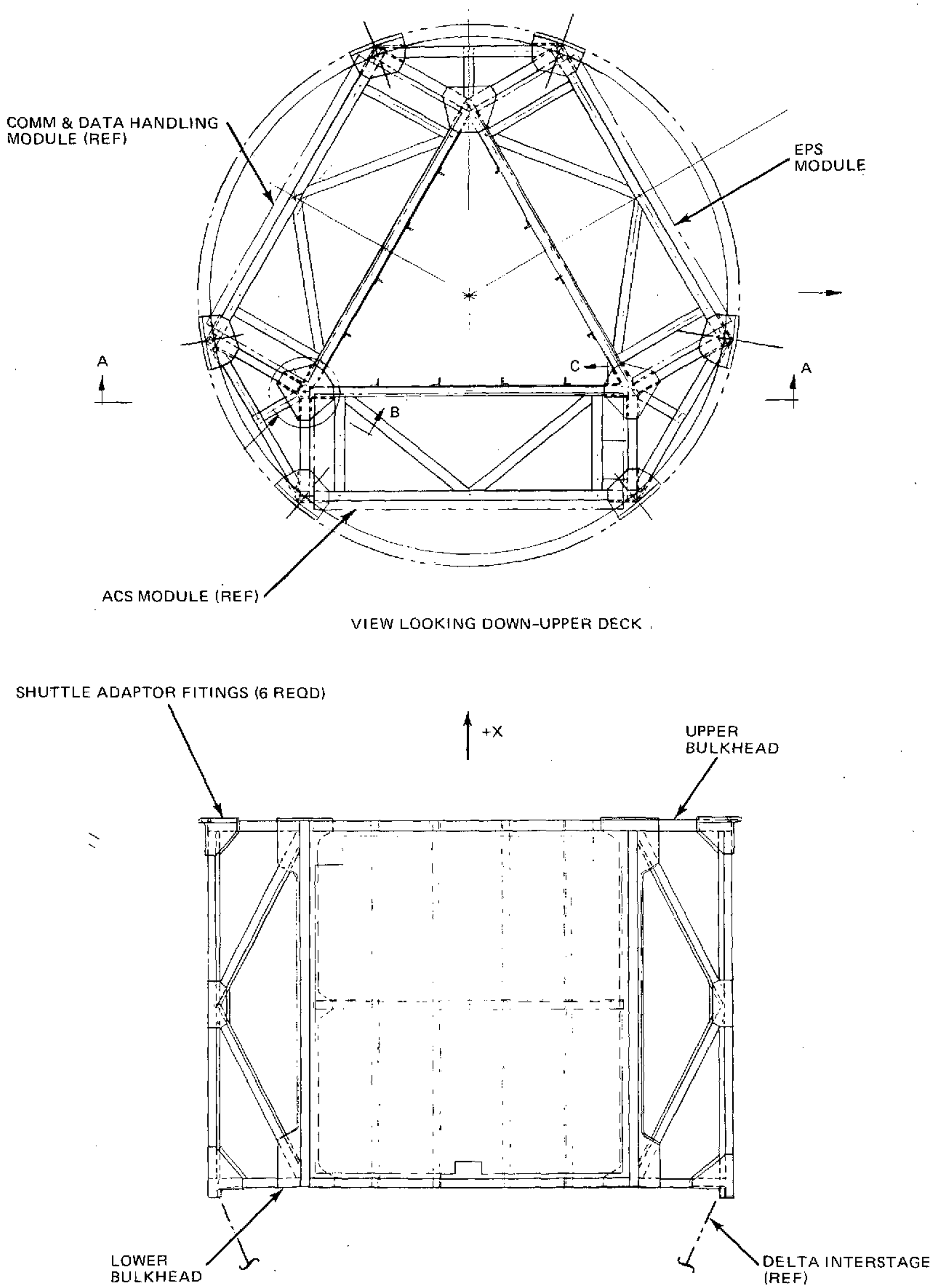
4.1.3 STRUCTURAL CONFIGURATION FOR TRIANGULAR VEHICLE

The Grumman alternative configuration is shown in Figure 4.1-4 and -5. The primary structure consists of three vertical shear webs forming a triangular-cross-section core vehicle. Extending from the webs are six vertical trusses which form the support for the three equipment modules. The equipment modules are supported at three points, as shown in Fig. 4.1-6. In this arrangement, primary structural loads are not induced in the S/S equipment modules, but are carried from the adapter hard points through the six rigid vertical trusses to the instrument support structure. This arrangement allows the subsystem modules to be easily removed for inflight or ground resupply, with no significant design or cost impact. Thus the vehicle can be initially designed and built for, or easily converted to, a Shuttle-resupply configuration, with insignificant cost or weight impact.

The basic core structure weights 186 lb and each module frame and honeycomb panel weighs 73 lb, for a total structural weight of 405 lb. The addition of resupply latch mechanisms adds 10 lb per module, and a segmented transition ring adds 36 lb, for a total of 471 lb. The structure has been conservatively designed. Weights and member sizes will be refined in the second phase of the study. A detailed structural weight breakdown is shown in Table 4.1-8.

The detail structural design has, for the most part, used standard structural sections, to effect a low-cost design. The recurring cost for the core structure first unit is estimated to be \$65K, and for each module structure, \$12.5K excluding honeycomb. This will be verified in Phase 2 of the Study. As quantities increase, the costs of succeeding units will decrease, as shown in Table 4.1-9.

Table 4.1-10 compares the above configuration with the GSFC baseline approach.



3-5

Fig. 4.1-4 Basic Spacecraft Structural Arrangement

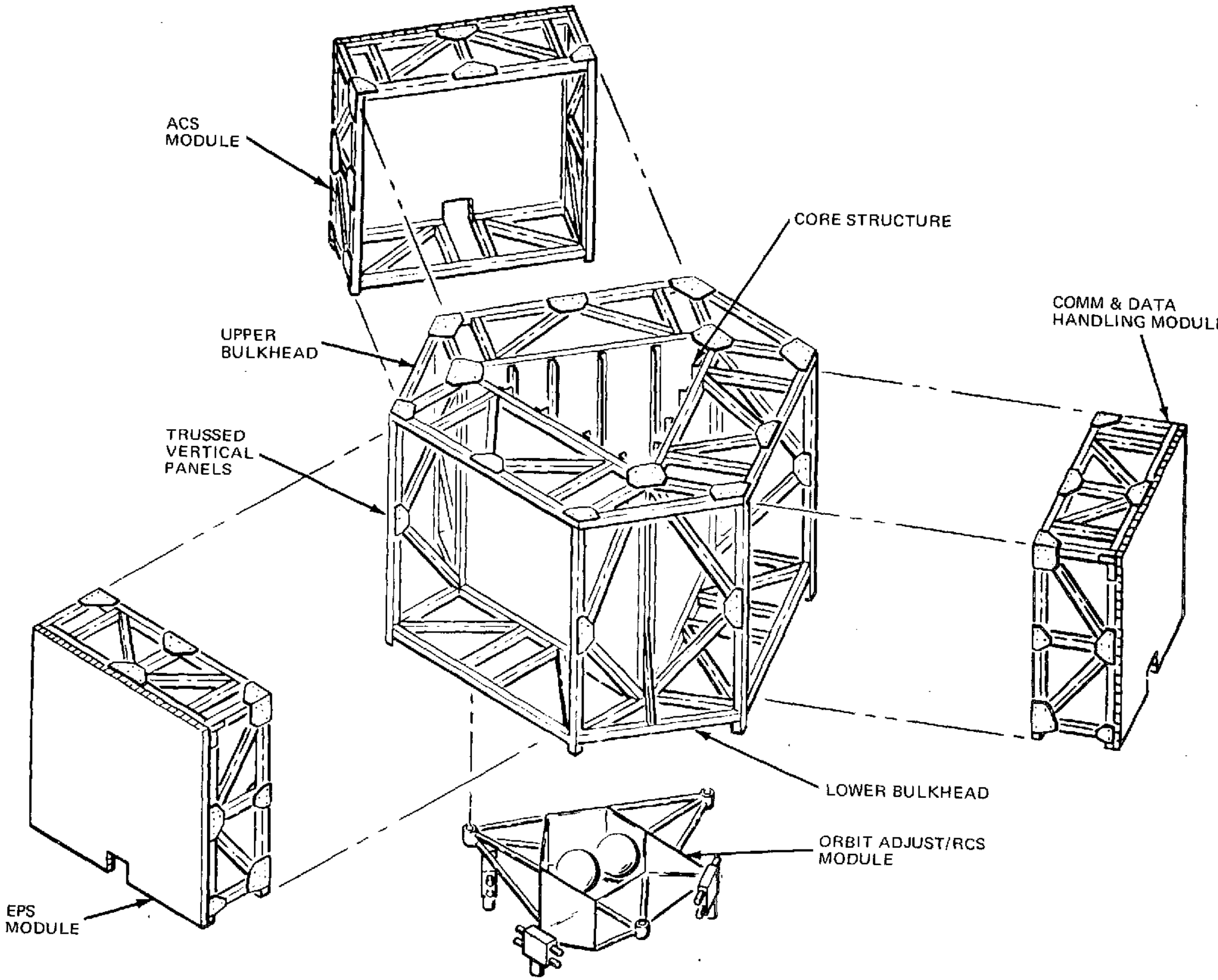


Fig. 4.1-5 Basic Spacecraft Structure

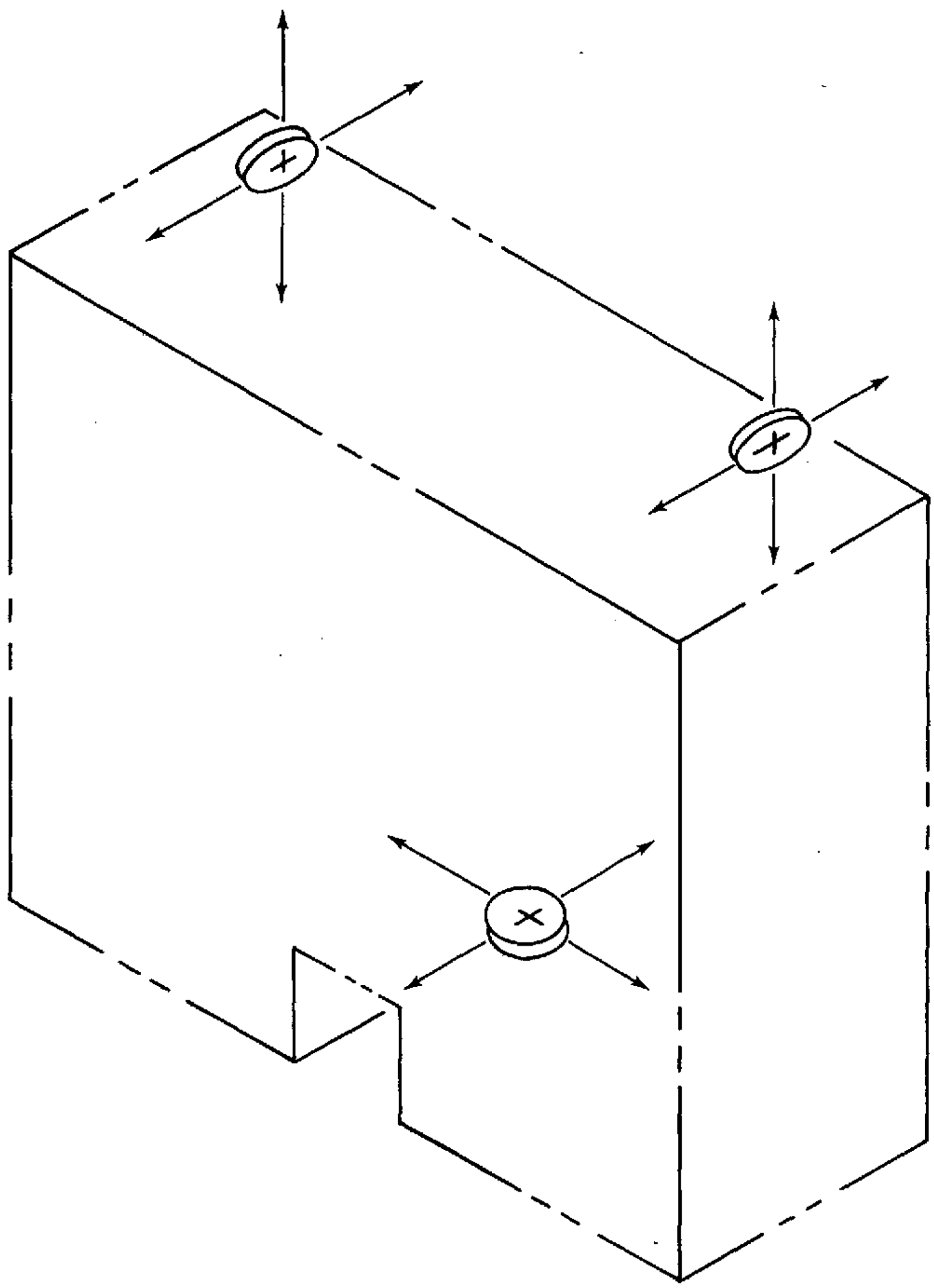


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Fig. 4.1-6 Equipment Module Support Points

Table 4.1-8. Basic Structure Weight Summary

	EOS-A	RESUPPLY	RETRIEVE
CORE STRUCTURE	(186 LB)		
CORE TRUSS	70.9		
FWD BULKHEAD STRUCT.	15.6		
AFT BULKHEAD	30.6		
OUTRIGGER TRUSS (6)	68.7		
ORBITER I/F (6)			36 LB
MODULE STRUCTURE (3)	(219)		
H/C BULKHEAD	79.2		
H/C SHELF	18.3		
SIDE TRUSS	121.5		
ORBIT ADJUST STAGE	(25)		
H/C BULKHEAD	3.7		
BEAM (6)	9.0		
SUPPORT TRUSS	10.1		
TANK SUPPORTS	2.3		
RESUPPLY MECHANISM		(52)	
CORE (TRACKS)		7.6	
MODULE (3)		30.0	
OAS		7.2	
SOLAR ARRAY		7.2	
TOTAL	430	52 LB	36 LB
TOTAL STRUCTURE	518 LB		

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Table 4.1-9. Unit Structural Cost vs Quantity ($ K)

	NUMBER OF UNITS			
	1	FOURTH	TENTH	30TH
CORE STRUCTURE	65	52	46.5	39
MODULE STRUCTURE	12.5	10.1	9	7.3

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Table 4.1-10 Comparison of Structural Concepts, GSFC Baseline Vs Grumman Alternative

	WT. OF CORE STRUCT.	MODULE WT. (3)	OAS SUPPORT WT.	RETRIEVE WT.	RESUPPLY WT.	RESUPPLY	BOOSTER MOUNT
GSFC BASELINE	130	324	20	107	0	NO	RING
GAC ALTER.	186	219	25	36	52	YES	BOTTOM

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Significant weight reduction and some recurring cost savings may be accomplished on the basic spacecraft structure by substituting advanced composite materials for aluminum. The particular composite which would be used in this case is hybrid Graphite/Eposy. This composite is a mix of UHM and LMS Graphite fibers in an epoxy matrix which offers the same stiffness as Boron/Epoxy but at a lower cost. In addition, the hybrid physical properties, such as thermal expansion, can be tailored by varying the UHM and LMS mixture. A weight saving of 169 lb can be realized. Table 4.1-11 lists the structural weight savings potential for the primary EOS structure. A cost comparison of the core structures for aluminum and composites is shown in Table 4.1-12. It is concluded that, although cost of initial tooling for composites is high, cost of succeeding units is competitive with aluminum and saves 80 lb.

Table 4.1-11. EOS Structural Weight Reduction Potential With Hybrid Composite

ITEM DESCRIPTION	PRIMARY STRUCTURE WEIGHT: ALUMINUM	COMPOSITE	WEIGHT DIFFERENCE
CORE STRUCTURE	(186 lb)	(104 lb)	(–82 lb)
CORE BEAM	70.9	39.7	–31.2
FWD BULKHEAD	15.6	8.7	– 6.9
AFT BULKHEAD	30.6	17.0	–13.6
OUTRIGGER TRUSS	68.7	38.1	–30.6
ORBITER INTERFACE	(36)	(31)	(– 5)
FWD FRAME	6.0	2.9	– 3.1
CORNER BRACE	4.5	2.2	– 2.3
I/F SEGMENT (6)	25.8	25.8*	
MODULE STRUCTURE	(219)	(146)	(–73)
H/C BULKHEAD	79.2	68.3	–10.9
H/C SHELF	18.3	18.2	– 0.1
SIDE TRUSS	121.5	59.4	–62.1
ORBIT ADJUST STAGE	(25)	(16)	(– 9)
H/C BULKHEAD	3.7	3.2	– 0.5
BEAM (6)	9.0	5.2	– 3.8
SUPPORT TRUSS	10.1	6.4	– 3.7
TANK SUPPORTS	2.3	1.4	– 0.9
RESUPPLY MECHANISM	52	52*	
TOTAL STRUCTURE	518	349	–169
% REDUCTION		32.6	

*Titanium for thermal expansion compatibility with GR/EP

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Table 4.1-12. Core Structure Cost/Weight Tradeoff

MATERIAL	WEIGHT	NON REC. TOOLING	MATERIAL COST	RECURRING COST (FIRST UNIT)
AL	186	$130K	$1K	$65K
HGFRA	104	500K	30K	43K

Weight Saving - 82 #
Cost - $377K

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4.1.4 RESUPPLY APPROACH

4.1.4.1 LATCHING MECHANISM

The GAC latch mechanism shown in Figure 4.1-7 consists of three hook-and-roller latches per module and utilizes a self-locking linkage. Initial alignment is accomplished by means of the latch roller guides provided on each of the three latches. The latch hooks are configured to supply the final pull-in force required for mating of the self-aligning electrical connector, and the latch-operated push-off rods supply the necessary demating force. Launch loads are carried via the three latch points only, and no loads are transmitted through the track and roller guide system. Module positioning and latch operation are accomplished by means of a single latch operator. The system is readily adaptable to a dual or triple latch operator arrangement. The latch operator consists of a holding knob rigidly fixed to the module and containing a centrally-located rotary driver which supplies rotary input to the worm gears operating the latches. A common latch operator is utilized for all the resupply latches.

This arrangement has many advantages:

- The single latch operator simplifies the Shuttle module exchange mechanism and increases its reliability
- Can be easily adapted to individual delatching.
- Can possibly be adapted to module exchange using Shuttle-Attached Manipulation only.
- Can delatch blind areas and around corners. (No line-of-sight needed.)
- Has integral push-off rod to eliminate cold welding and provide separation force for electrical connectors.
- Light weight: 3 lb. per latch, 10 lb. per module.
- High mechanical advantage, needing low actuator force.
- Simple, reliable, and economical.

4.1.4.2 MODULE RESUPPLY

The selected latching mechanism can be used for resupply of every required replaceable assembly. Typical are the concepts shown in the following figures:

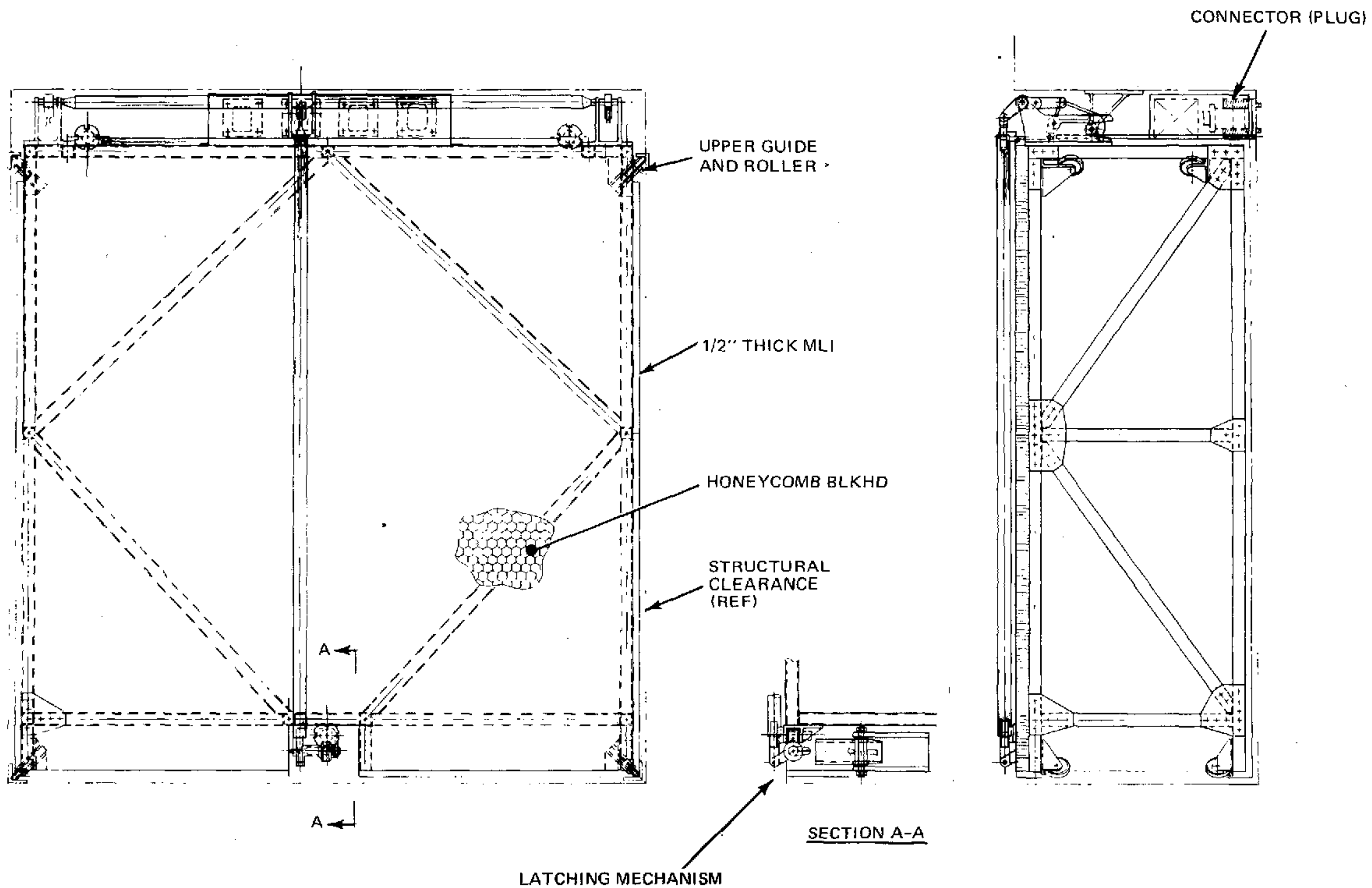

Fig. 4.1-7 Subsystem Module Structural Assembly Showing Latching Mechanism Detail

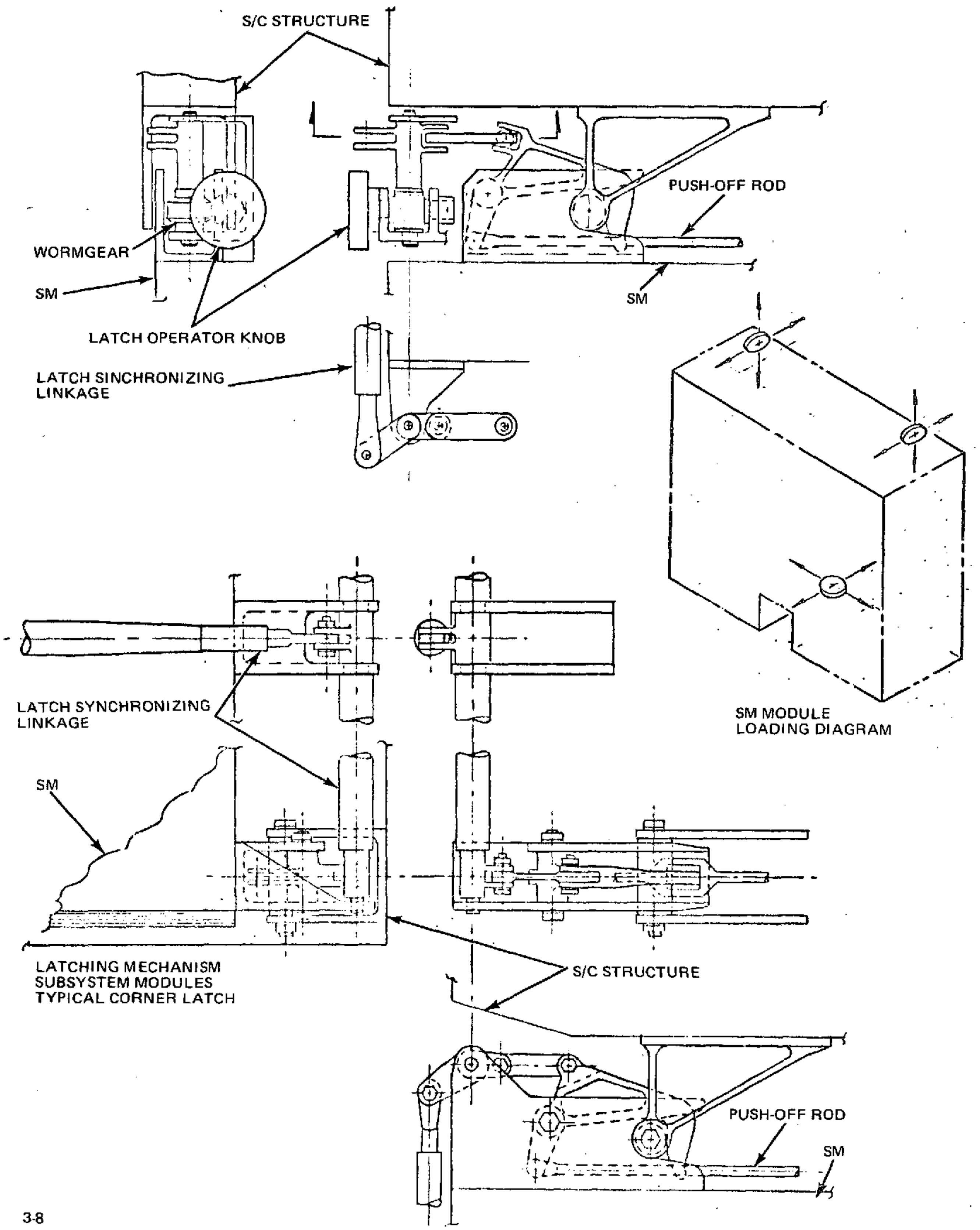

Figure 4.1-8 Subsystem Module, Typical Central Latch

Figures	Module
4.1-8	Subsystem Module
4.1-9	Thematic Mapper
4.1-10	Solar Array
4.1-11	RCS/OAS Module

In addition, other replaceable assemblies which can use one of the above concepts include the IMP, antenna, and tape recorder modules.

The resupply latch system for the Orbit Adjust Stage is similar to the subsystem, and differs only in that one of the three latches is replaced by a conical socket engagement. The Solar Array drive resupply latching is similarly accomplished with dual SM-type latches and a third point support provided by a conical socket engagement. Additional retract latches are provided on the Solar Array for the purpose of sustaining loads during launch, orbit adjust and shuttle re-entry.

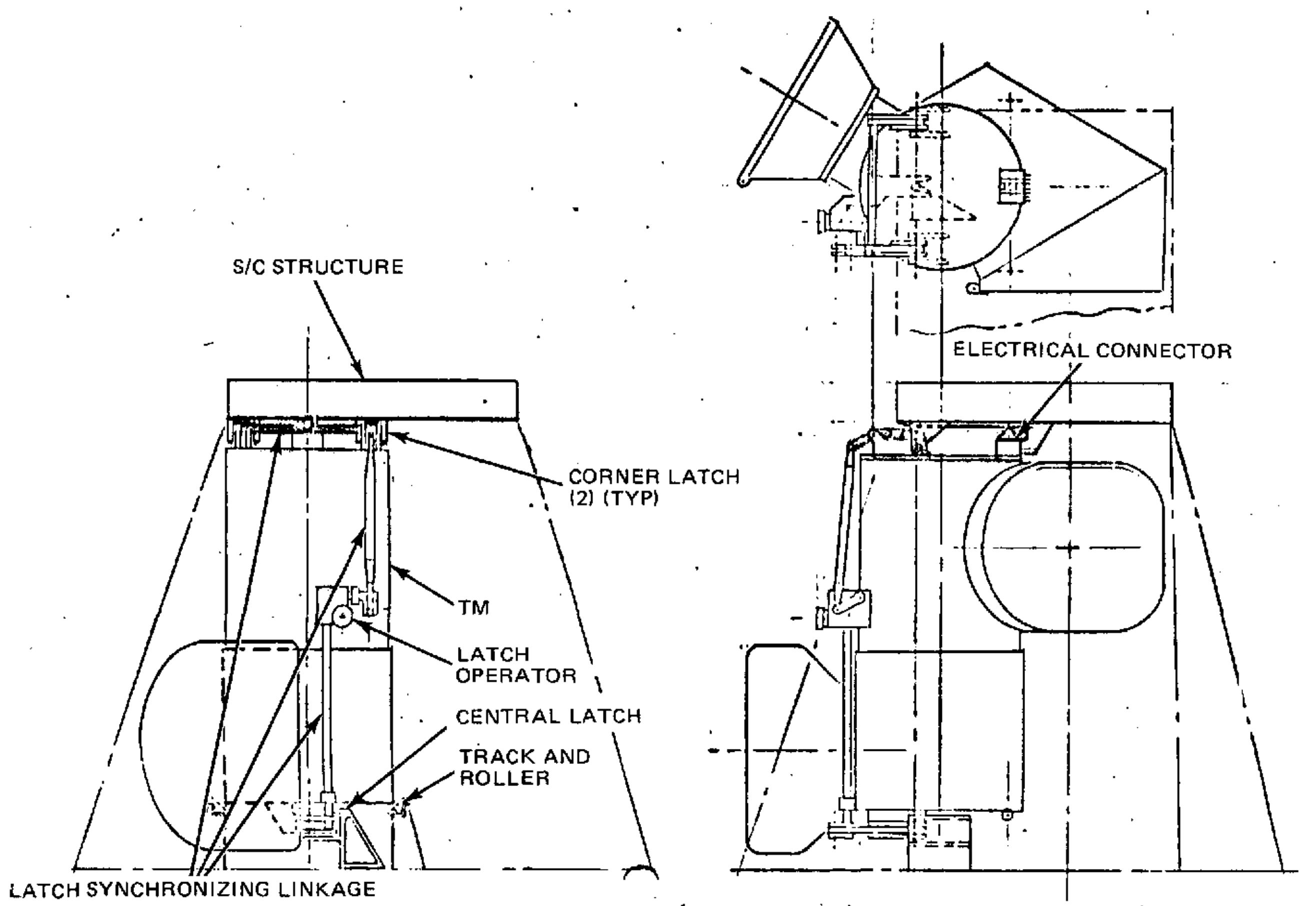

Fig. 4.1-9 Thematic Mapper Latching Concept

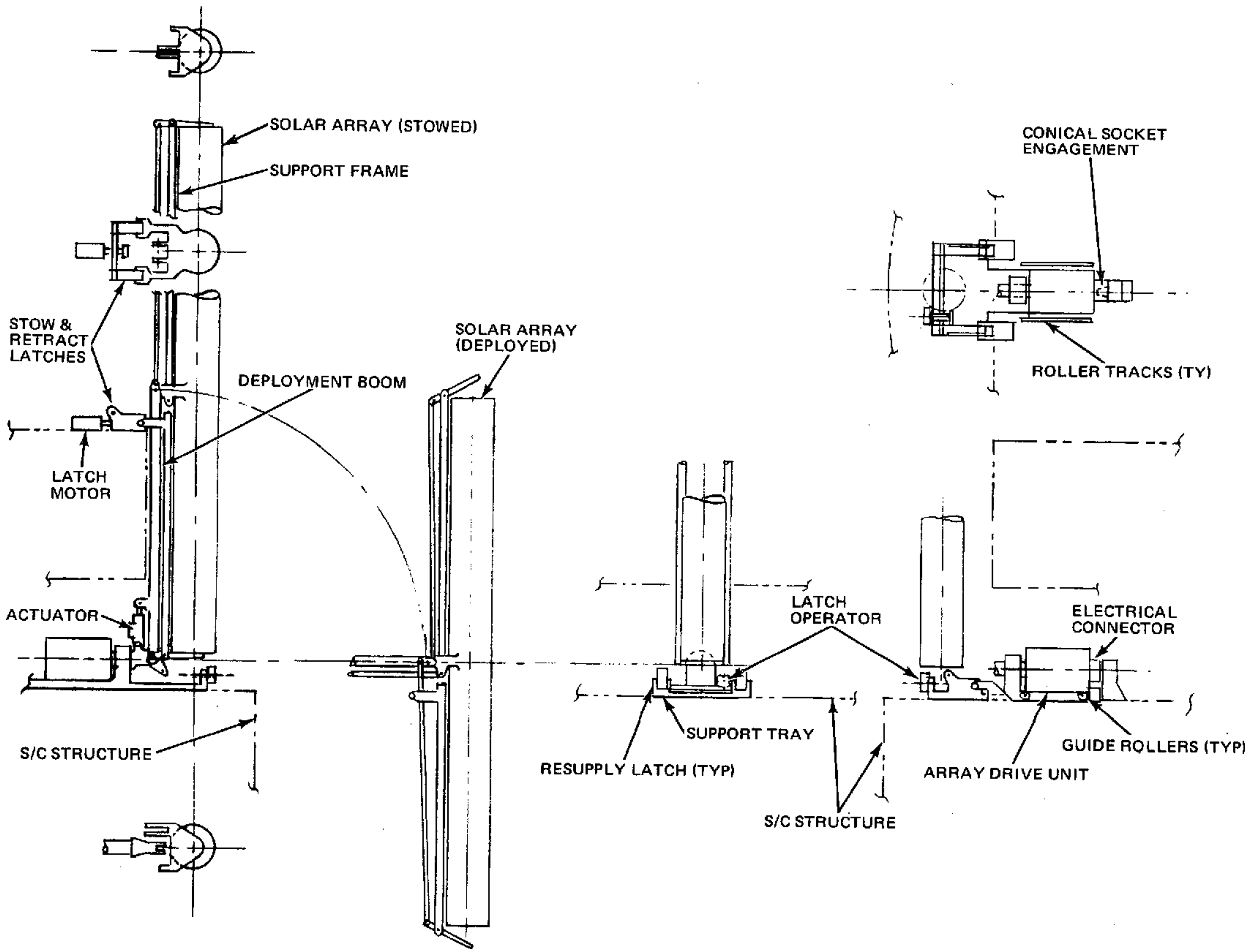

Fig. 4.1-10 Solar Array Mechanism for Stowage, Deployment, Retraction and Resupply

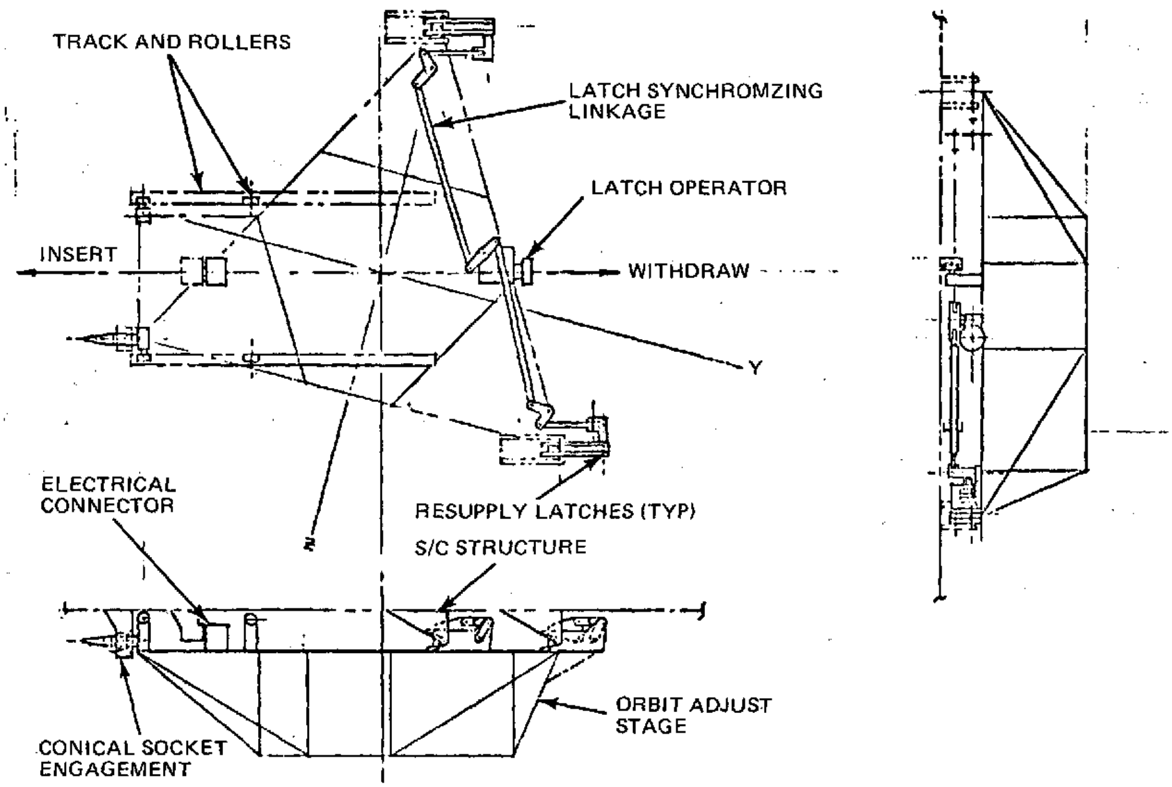

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Fig. 4.1-11 RCS/OAS Latching Concept

4.1.4.3 ELECTRICAL DISCONNECT OF MODULES

We propose to utilize the electrical connector shown in Fig. 4.1-12 for the electrical disconnect function in the subsystem or other modules. This disconnect has the capability of mating despite large misalignments. It is currently being used on the F-14 weapons rail and has application on the EOS resupply.

4.1.5 ORBIT ADJUST

The Orbit Adjust/RCS Module shown in Fig. 4.1-13 provides support for four thruster pods and two 10-inch diameter hydrazine propellant tanks. Each pod houses two 0.10 lb, two 1.0 lb, and one 5.0 lb thrusters. The module is mounted under the core structure to facilitate removal during resupply operation.

The stage consists of a central hexagonal module which contains the propellant tanks. The module is 12 inches deep and 24 inches across the flats. Six corner tee members are connected by stiffened sheet webs. A honeycomb shelf is attached to the bottom cap angles of the peripheral webs to support the propellant tanks. Five square tubes join the upper opposite corners of the hexagon with the aid of a splice plate at their intersection.

Four square tube struts extend off three alternate sides of the central hexagon and terminate at the stage attachment/latch fittings which are 120° apart on a 30 inch radius.

Two tapered, stiffened sheet beams extend off two of the remaining three sides of the central hexagon to provide support for two of the thruster pods. The other two pods are supported off the underside of two of the stage support strut assemblies by sheet metal brackets and angles.

The thruster pods consist of a sheet metal C-section which forms the back, top and bottom of the module, two removable end plates, to which are attached the low level thrusters, and an outer cover which serves as an access panel and module closure. The high-level thruster is attached to the bottom of the C-section.

The module structure is of relatively simple, straight line geometry, "standard" sections used wherever possible. Estimated weight of the structure is 25 lb.

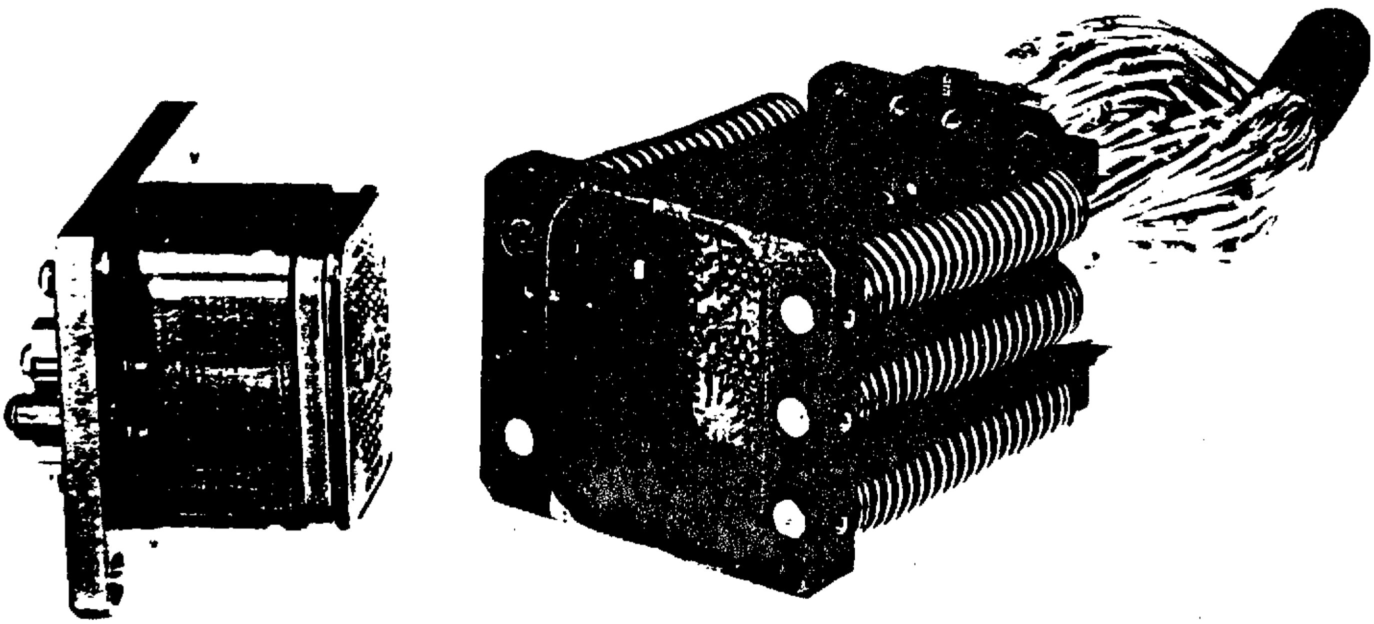

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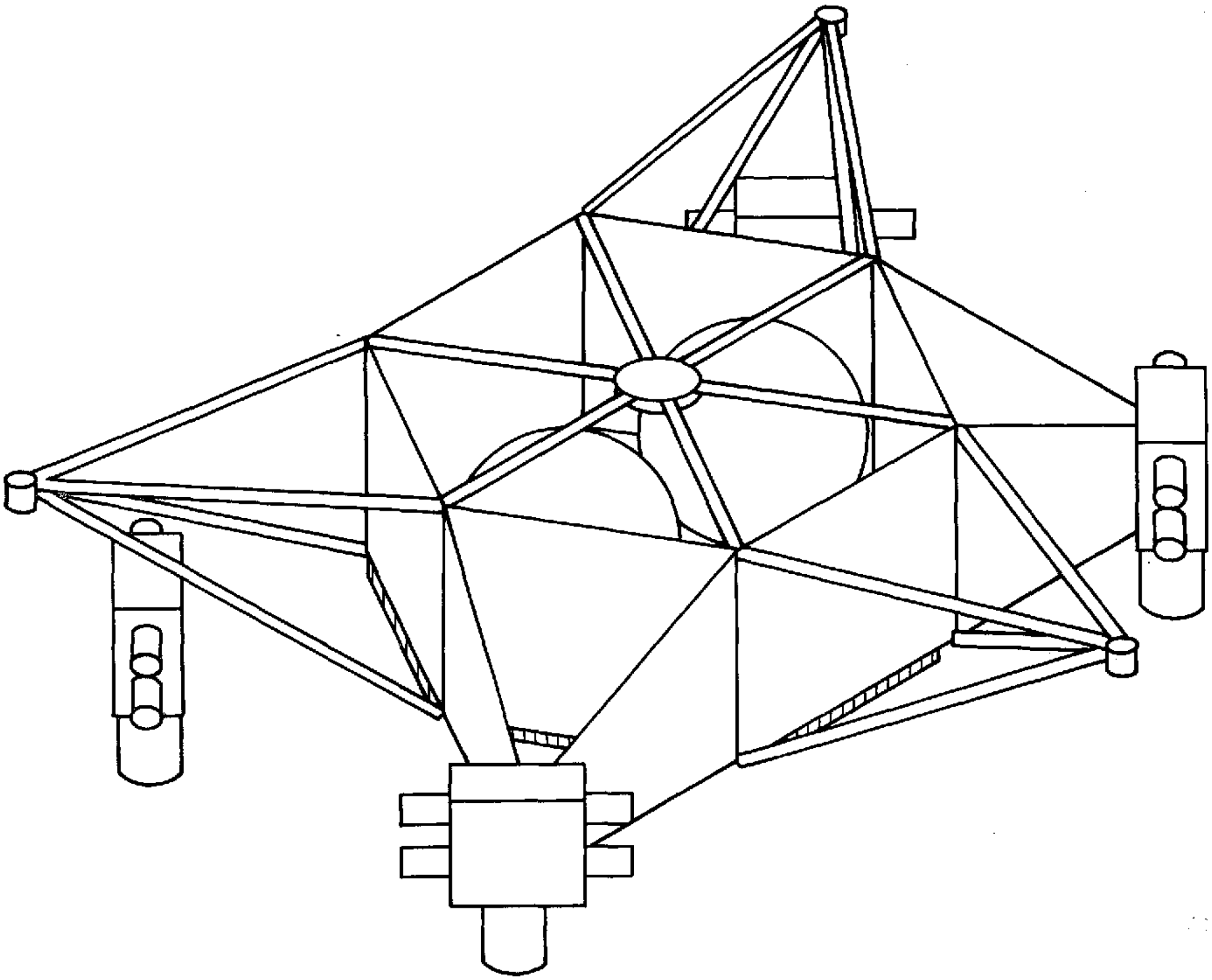
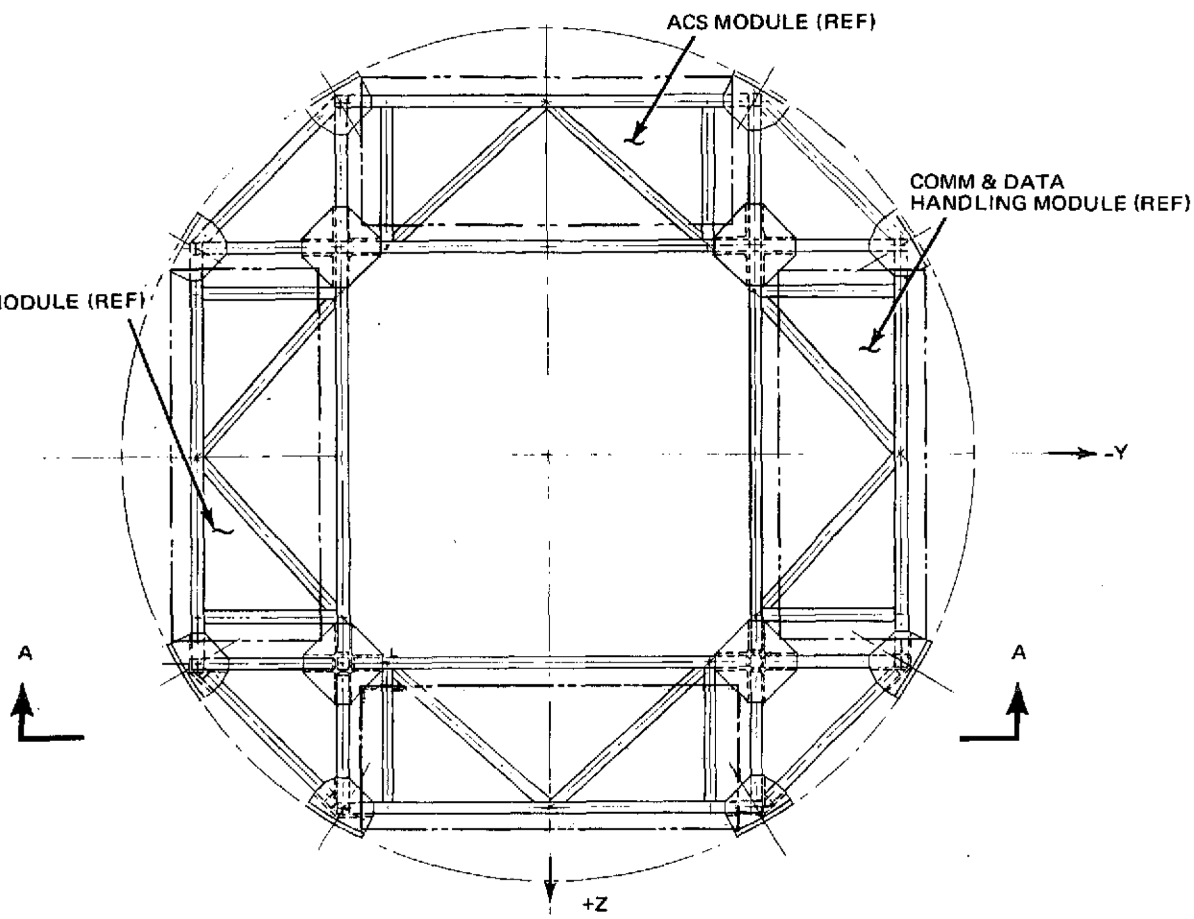
Fig. 4.1-12 Blind-Mate Umbilical

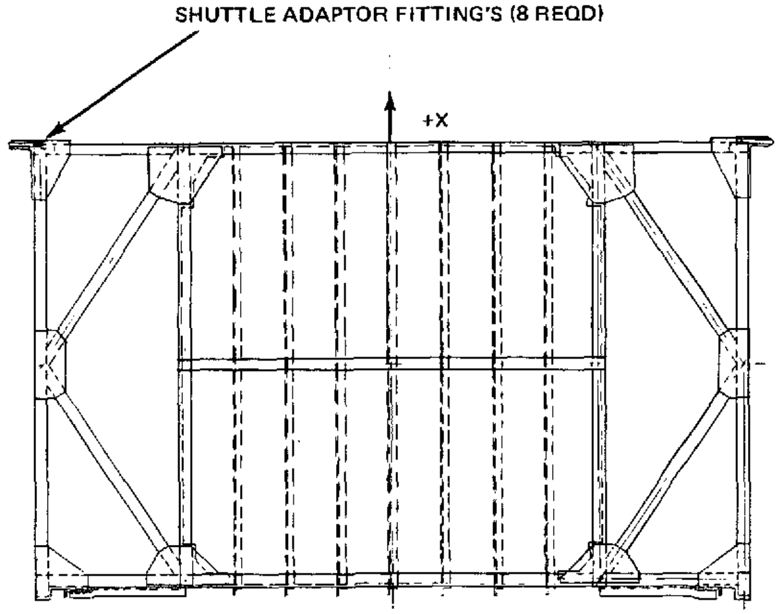


3-13 **Fig. 4.1-13 Orbit Adjust/RCS Module**

4.1.6 SOLAR ARRAY ACCOMMODATIONS

The Solar Array, shown in Fig. 4.1-10, is of the roll-up flexible type mounted on a deployable support frame. Rigid array configurations have been investigated and are feasible. The stowage and deployment mechanism is configured to provide retrieval and resupply capability. A single electric screwjack operates the deployment mechanism both during deployment and retraction. For stowage, dual hook and roller latches are provided to secure the array frame to the spacecraft structure during launch, orbit adjust and shuttle re-entry. In addition, both the frame and the deployment boom are snubbed against the structure in the stowed condition. The stowage latches are actuated by an electric motor driven worm





3-15

Figure 4.1-14 Titan Basic Spacecraft Structural Arrangement

gear set. The resupply system consists of dual hook and roller latches and a single conical socket engagement mounted on a support tray which houses the array drive motor and the deployment boom lower support. Guide rollers are provided on the tray to facilitate initial alignment during its insertion into the spacecraft. Insertion and removal is accomplished by grasping the single latch operator knob and latching or unlatching is affected by rotation of a drive socket within the knob. Thrust forces for mating and demating the electrical connector are supplied by a hook pull-in and a push-off rod respectively. Therefore, only a torquing force need be supplied by the SAMS and effector.

4.1.7 TITAN-DEDICATED STRUCTURE

A structure was designed to make full use of the Titan launch vehicle volume and configuration advantages. This structure is shown in Fig. 4.1-14. The capability of this configuration meets all the requirements of the triangular structure except for a launch on a Delta Vehicle. In addition, it can house a fourth subsystem module and support a total vehicle weight of 5100 lb in the Titan III environment. A weight breakdown of the basic structure compared to the GSFC baseline design is shown in Table 4.1-13. A total weight saving of 818 lb is indicated. This vehicle is bottom mounted for the reasons described in paragraph 4.1.2.1. Structural analysis has indicated no weight penalty for bottom mount. The spacecraft adapter and separation system are of conventional design and no spacecraft extraction problem is envisioned in this approach. The load paths are similar in concept to the triangular Delta spacecraft.

Table 4.1-13

Comparison of Titan Structure & Weights

Structure	GSFC Baseline	GAC
CORE	580	360
MODULE (4)	392	292
OA/OTS	170	75
RETRIEVE	225	80
RESUPPLY (5)	320	(6) 62
TOTAL	1687 LB.	869 LB.

4.1.8 INSTRUMENT ACCOMMODATION

The basic requirement for instrument accommodations is to allow for functional operation of the instruments and to allow for resupply of all items in the forward end of the spacecraft. These include steerable antennas, solar array, instrument mission peculiars, tape recorders and instruments. The instrument mission peculiars and tape recorders have been housed in modules, again to facilitate resupply.

4.1.8.1 EOS-A INSTRUMENTS

The combination of instruments, TM and MSS, results in the configuration as shown in Fig. 4.1-15. The drivers in this arrangement are the two viewing requirements of each:

- Radiator viewing to "Black Space"
- Sensor viewing to the nadir

The Thematic Mapper is supported between two box beam platforms. The lower beam reacts directly into two of the three shear webs of the subsystem structure below. The upper beam (+X) is attached to the lower by means of a bulkhead on the +Z side which is notched for TM sensor. In addition, six struts from five spacecraft "hardpoints" to the upper beam, add the required stiffness to keep the natural frequency above requirements. A three point determinant support has been assumed for this instrument and clearance to the support structure has been allowed for the resupply latches.

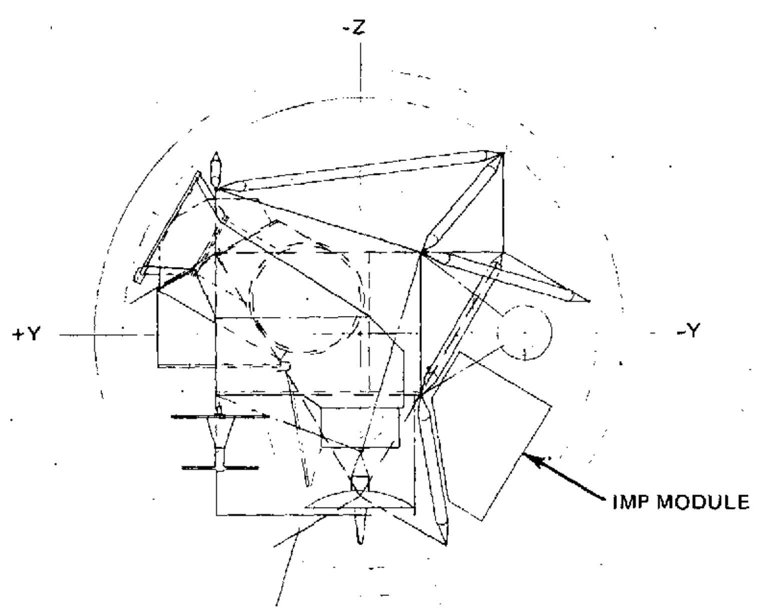
The MSS is supported on top of the upper beam with similar latches in a cantilevered fashion.

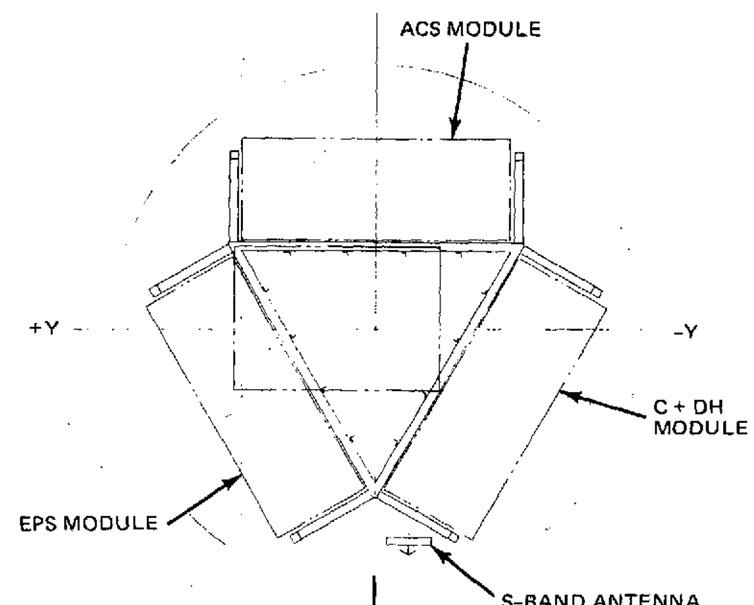
The solar array is supported on the subsystem structure utilizing an additional beam or a continuation of the TM support beam to balance the ± X loads. The Y and Z loads are balanced by two struts from the upper beam.

4.1.8.2 EOS-A' INSTRUMENTS

The viewing requirements of the HRPI and MSS result in the configuration shown in Fig. 4.1-16. Both instruments view the nadir while the MSS has the additional requirement of a radiator viewing "Black Space."

The basic support structure for both instruments is a five sided box where the lower and upper faces are box beams and the other three sides are strut-trusses. The HRPI is positioned between the lower and upper faces and the MSS is on top (+X) of the upper face.





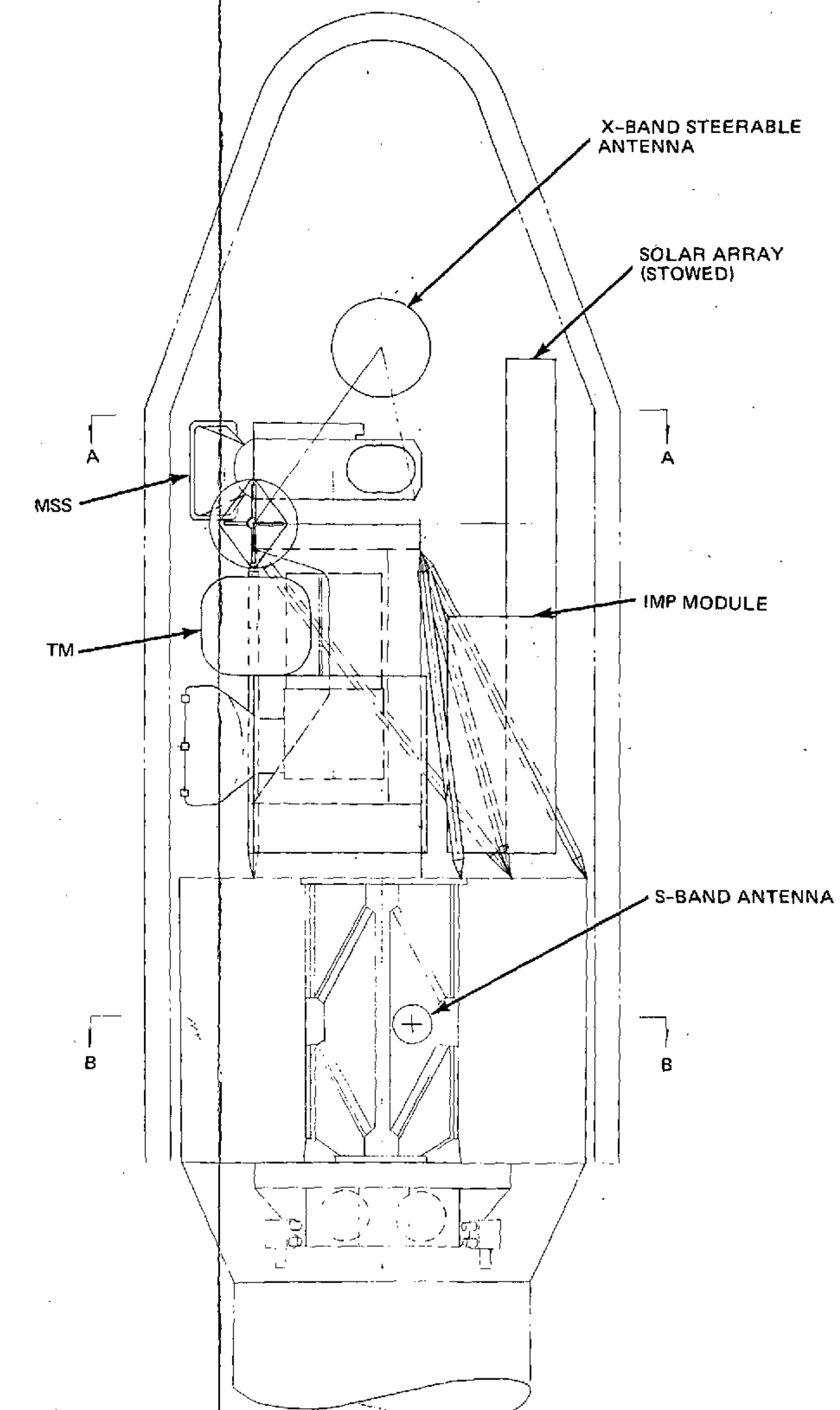
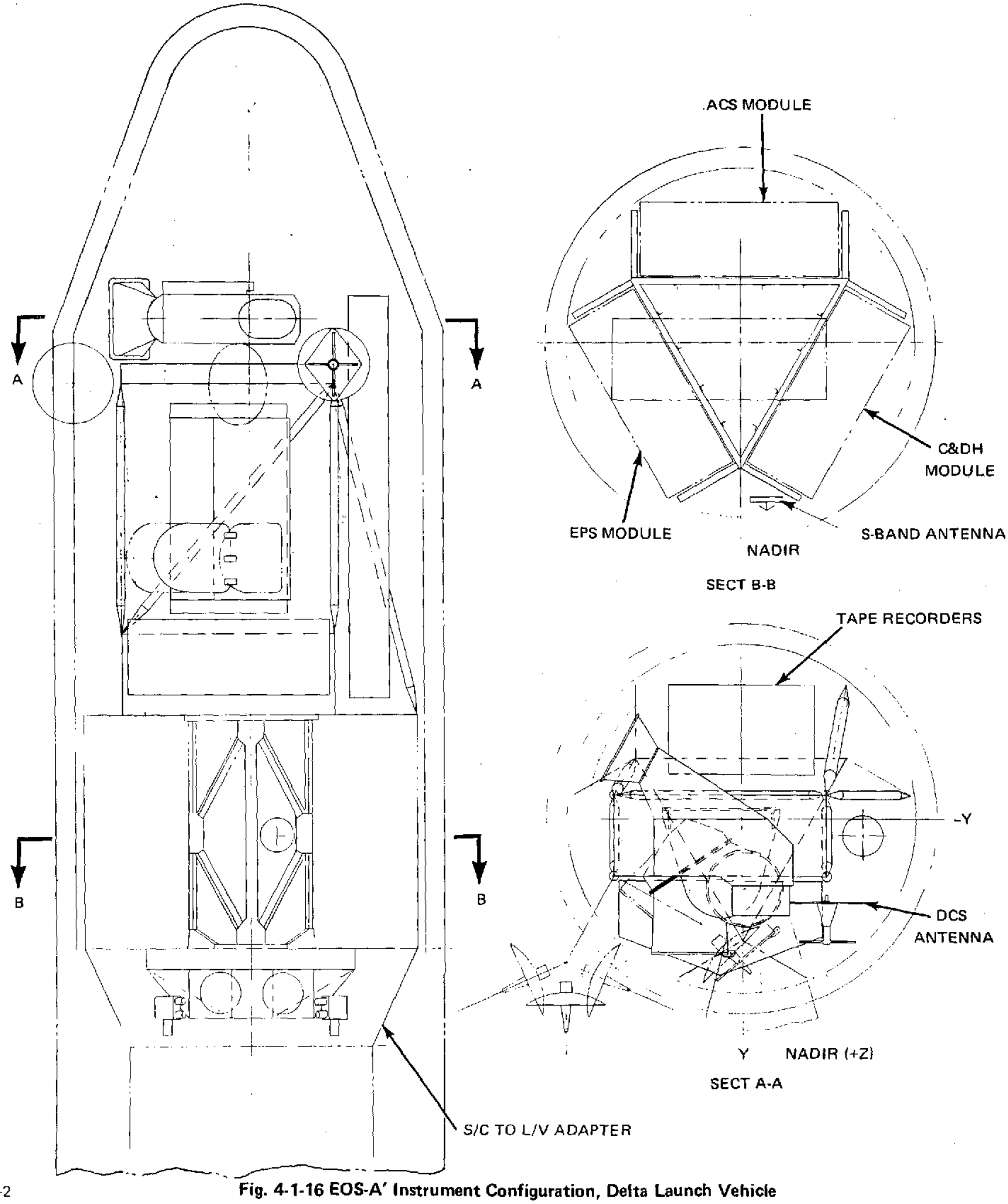


Fig. 4.1-15 EOS-A Instrument Configuration, Delta Launch Vehicle



3-2

Fig. 4-1-16 EOS-A' Instrument Configuration, Delta Launch Vehicle

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The lower box beam (-X) reacts directly into two of the three shear webs of the subsystem structure below. In addition, three struts from spacecraft hardpoints to the upper box beam, add stability in the Y, Z plane. The assumed three-point support and resupply capability of these instruments require a special latch/retention system, and clearances to the structure have been allowed for it. HRPI removal for resupply is in the -Z direction, the MSS in the +Z direction. The solar array is supported on the subsystem structure utilizing an additional beam to balance the ± X loads. The Y and Z loads are balanced by a fitting from array to upper beam.

The tape recorder is located on the -Z side and supported off the spacecraft structure via a beam or beams to pick up the resupply latches. Removal is in the -Z direction.

The IMP box is located on the +Z side and is supported similarly to the tape recorder. Removal for resupply is in the +Z direction.

The DCS and X-Band antennas are supported on the upper box beam via appropriate struts.

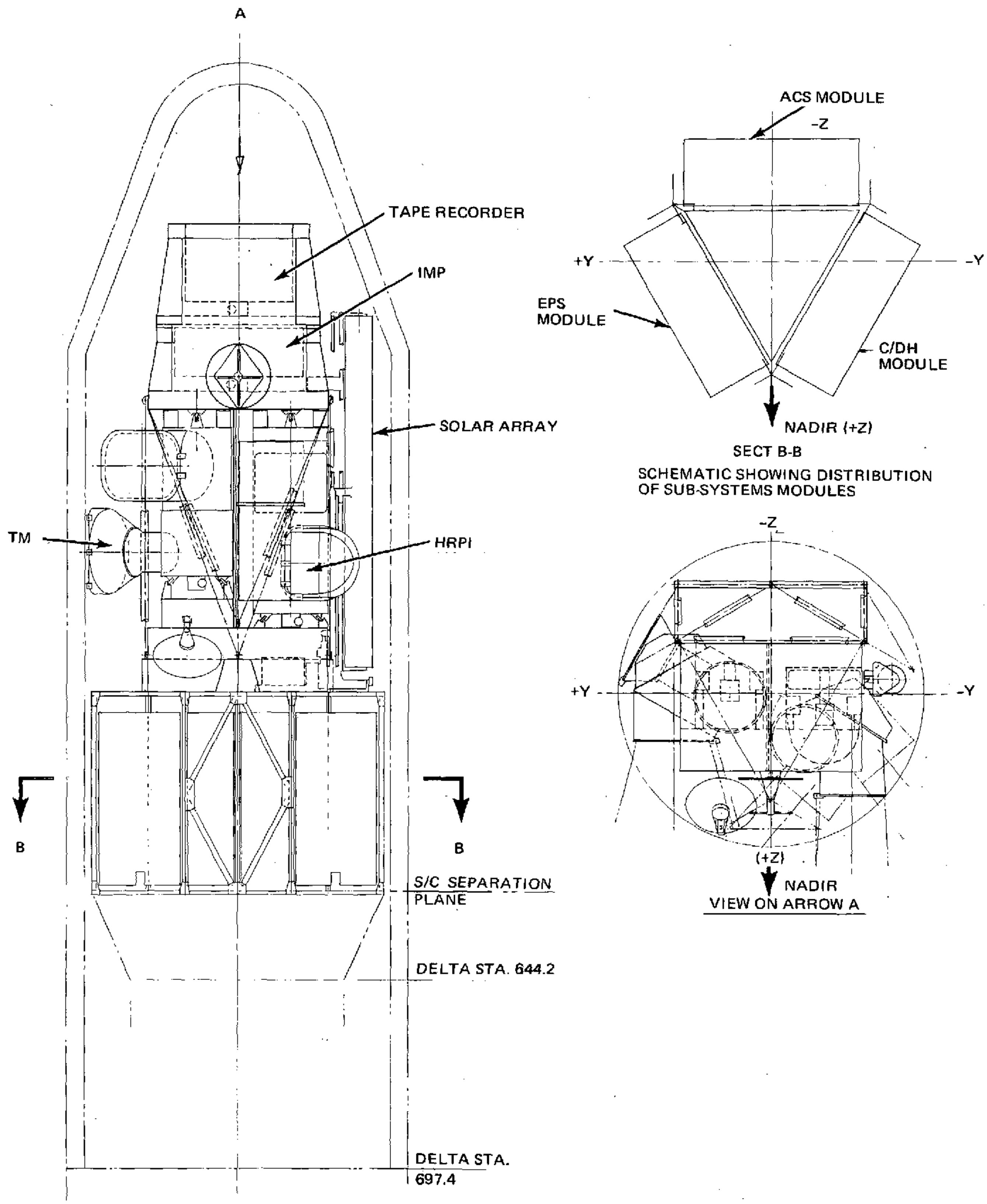
The tape recorder(s) is located on the +Z side of the "four sided box" adjacent to the notched bulkhead. Both the "box" and the lower subsystem structure are used to support this item. The IMP, located in the -Y, +Z quadrant, is supported similarly.

The DCS and X-Band antennas are supported on the upper box beam via appropriate struts.

For resupply, the TM is removed through the +Y side, MMS through -Z, tape recorder +Z and IMP +Z-Y (45°).

4.1.8.3 EOS-B INSTRUMENTS

The Thematic Mapper and the High Resolution Pointable Imager as configured by Hughes can be mounted side-by-side on the triangular spacecraft support structure as shown in Fig. 4.1-17. The TM is located on the +Y side to permit a 180-deg radiator field of view on the shaded side of the spacecraft. The TM scanner is therefore in the forward or +X end with respect to the velocity vector (X-axis) without interference with the adjacent TM scanner sunshield. The Data Collection System Antenna is located on the nadir side of the instrument support platform along with the steerable X-Band antenna for maximum earth exposure. The instrument mission peculiar electronics package is also mounted on



3-18 Fig. 4.1-17 EOS-B Instrument Configuration, Delta Launch Vehicle the upper platform central to all instruments and to the tape recorder package which is located above it.

Each of the instruments, the mission peculiar package and the tape recorder package are removable from the spacecraft using the Grumman resupply latching system. The TM and HRPI may be manipulated from the nadir (+Z) side and the mission peculiar electronics and tape recorder packages from the (-Z) side of the spacecraft.

The roll-up solar array is deployable in the -Y direction, the area of maximum solar energy potential. The actuating mechanism assembly is latch mounted to the forward bulkhead of the module structural support assembly. A latching mechanism attached to the instrument platform supports the other end of the undeployed array during launch. The Grumman latching system permits replacement of this unit in the -Y direction.

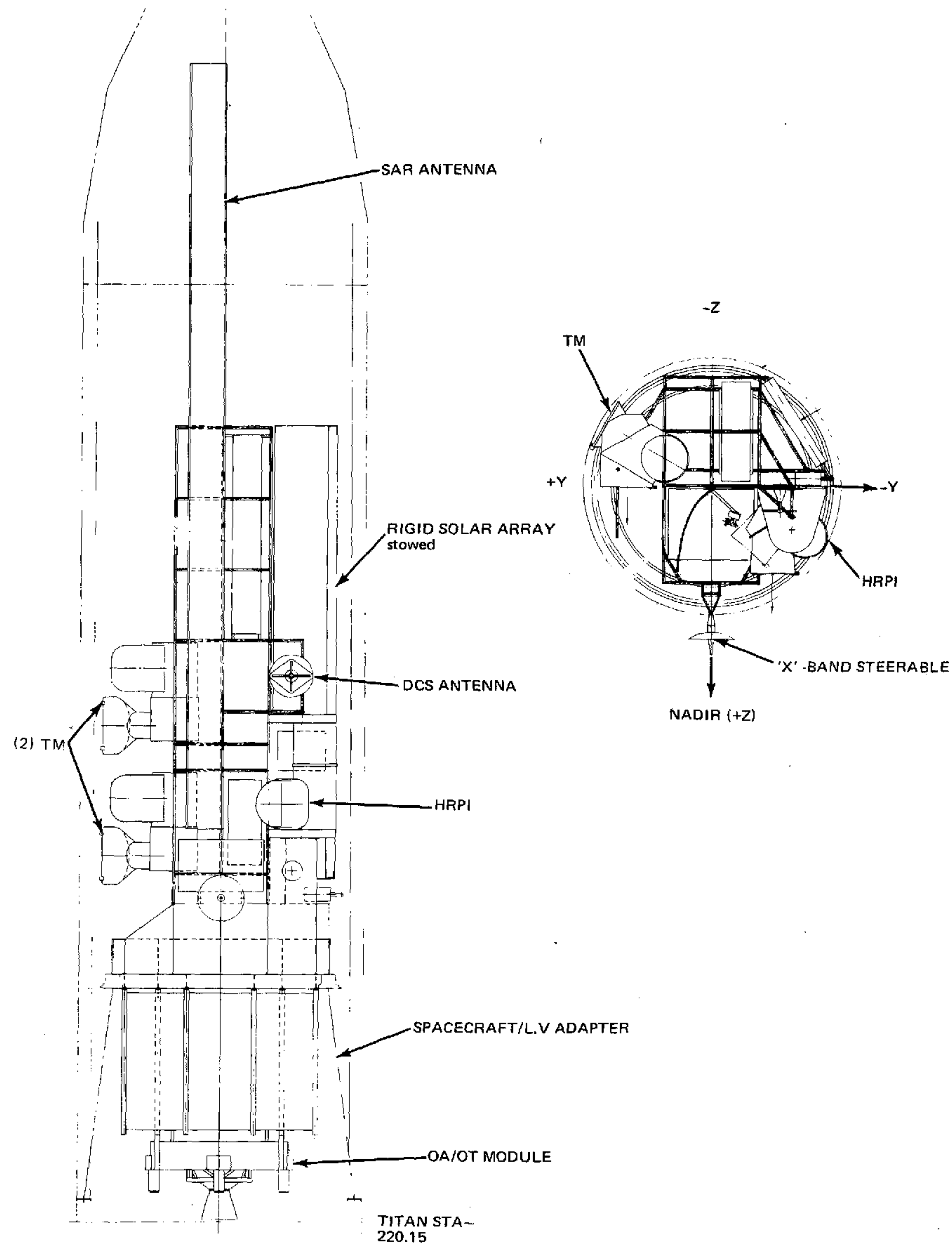
The X-Band antenna, a 20-inch diameter dish, is rotatable 62 deg in any nadir (+Z) direction and is rigidly mounted to the forward face of the instrument support platform.

The instrument support structure is essentially supported by a 26-inch wide by 51-inch long lower beam-platform, which is attached through the forward bulkhead to the upper caps of the triangular spacecraft structure assembly. The lower latches and track assemblies for the HRPI and TM are readily accommodated in the hollow interior of the beam-platform due to its 18-inch height. Stiffened sheet metal construction with extruded cap members and intercostals are envisioned for this structure.

The upper latches and guides for the major instruments and the array are supported on the underside of a sheet metal instrument support platform five inches deep. The X-Band and DCS antennae are attached to a longitudinal beam which is the forward edge. The side-face beams and intercostals of the platform provide support for the upper and lower latches of the mission peculiar electronics package. The former latches are supported on pylon fittings and the latter on the upper platform face.

4.1.8.4 EOS-C INSTRUMENTS

The instrument section for the EOS-C mission contains the following components: one HRPI, two TM's, one SAR antenna, one tape recorder module, one IMP module, one SAR electronics modules, a deployable solar array, a steerable X-Band antenna, a shaped beam X-Band antenna and a DCS antenna. (See Fig. 4.1-18)



3-19 **Fig. 4.1-18 EOS-C Instrument Configuration, Titan Launch Vehicle**

The structure required to support and house this complement of components consists of a base support/adapter ring, a rectangular arrangement of beams, 30 inches high, and a truss/beam tower, approximately 17 feet high to support the TM's, SAR, electronics modules, antennas, and solar array.

The base support/adapter ring serves as a base support for the complement of beams forming the primary support structure. It also serves as the interface/separation ring when launched in the Titan III B launch vehicle and the interface/support ring when launched in the STS orbiter.

The base support beams consist of three beams parallel to the Z axis, one on the Z axis and two 20 inches either side of the center beam. The ends of the beams terminate at the center of the adapter ring. Joining the ends of the ± 20 inch beams are two beams running parallel to the Y axis. The center beam terminates on the two transverse end beams. Another transverse beam spans across the adapter ring 6 1/2 inches off the center of the stage on the -Z side provides a center support for the three main beams and the forward (+2) support for the tower structure. All of these beams are 30 inches high.

Several auxiliary beams extend between the main beams and the base ring to provide support for the subsystem module stage below the instrument stage. Two additional beams extend upward from the main beams to support the HRPI.

The two TM's are mounted one above the other within the tower structure and may be removed laterally in the +Y direction.

The SAR antenna is supported by, and hinged off, the front end of center tower beam over half the antenna length. The remainder of the antenna is cantilevered from the top of the tower upward.

The tape recorder module is mounted on tracks atop the forward half of the base beam structure. The IMP and SAR electronics packages are mounted within the tower structure.

The solar array is stowed in the -Y, -Z quadrant and consists of four 42 ft x 16 ft panels. They are folded into a 42 in. x 16 ft package and supported off the base beams and the tower structure.

4.1.9 WEIGHT STATEMENT

The launch weights of the Barebones, EOS-A, B and C spacecraft are summarized in Table 4.1-14. This shows the functional weight breakdown for each spacecraft. The Barebones spacecraft is a basic modular spacecraft with the minimum practical redundancy, a power supply capable of providing 200 W average power to the instrument section interface, and having no provisions for retrieval or resupply.

Table 4.1-14. EOS Weight Summary

FUNCTIONAL ELEMENT	BAREBONES	EOS-A	EOS-B	EOS-C
BASIC STRUCTURE	430	518	548	538
ELECTRICAL POWER	172	204	204	268
ELEC. HARNESS & SC	35	90	90	90
SOLAR ARRAY & DRIVE	195	102*	195	279
ATTITUDE CONTROL	156	156	156	301
RCS (HYDRAZINE)	37	37	37	37
C & DH	107	111	111	111
THERMAL CONTROL	72	72	72	72
INTERSTAGE ADAPTER	95	95	100	135
• S/C SUBTOTAL	1299	1385	1513	1831
MISSION PECULIAR	( )	(403)	(710)	(1215)
ORBIT ADJUST/TRANSFER	–	27	40	351
INSTR SUPPORT	–	133	174	318
WB TAPE RCDR	–	155	400	400
WB COMM	–	88	96	146
INSTRUMENTS	( )	(613)	(853)	(1753)
MSS	–	160	–	–
TM	–	400	400	800
HRPI	–	–	400	400
SAR	–	–	–	500
DCS	–	53	53	53
CONTINGENCY	186	211	243	331
TOTAL S/C	1485 LB	2612 LB	3319 LB	5130 LB

3-74 * INCORPORATES LIGHTWEIGHT (ROLL-OUT) SOLAR ARRAY DESIGN.

Major differences between the Barebones and the EOS-A basic spacecraft are:

a) Basic Structure. Adds a segmented orbiter interface ring to provide for retrieval. Adds latches and mechanisms for on-orbit resupply of modules, OAS and solar array.

b) Electrical Power. Additional battery is added to increase service life by limiting depth-of-discharge.

c) Electrical Harness and Signal Cond. Adds weight penalty for replacing standard connectors (both halves) with electrical interface assemblies for on-orbit resupply of modules, OAS and solar array.

d) Solar Array and Drive. Replaces the rigid deployable solar array with a flexible, roll-out solar array.

e) Mission Peculiar. Adds propellant, tankage and thrusters for orbit correction and adjustment. Adds instrument support structure and insulation, and latches and mechanism for instrument, IMP/DCS, and tape recorder box resupply. Adds instrument data handling equipment.

f) Instruments. Adds instrument complement for EOS-A mission.

g) Contingency. Adds contingency impact of above changes.

The EOS-B and C launch weights are built up similarly with weight changes reflecting such things as local structure reinforcement for launch vehicle compatibility, additional batteries and larger solar arrays, enlarged ACS reaction wheels and torquer bars, increased orbit adjust propellant requirements and the installation of an SRM for circularization at the mission altitude, in the case of EOS-C, as well as changes to the instruments and instrument data handling equipment.

Table 4.1-15 depicts the weight build-up from the Barebones spacecraft for EOS-A, A', B, C and the follow-on missions. The resulting launch weights are compared to the payload capabilities of the selected launch vehicles, and weight saving options applied as required to obtain a positive payload margin. EOS-A, A', D and E make use of roll-out solar arrays to achieve the savings (including contingency reduction) shown. EOS-B requires a Delta 3910 launch vehicle in the configuration shown, but if necessary can be flown on a Delta 2910 launch vehicle by undertaking the weight reduction program described in Table 4.1-16 which includes the surrender of one or more program options. SEASAT-A could be launched on a Delta 2910 launch vehicle with fewer modifications required.

The weights described above were derived from a detailed weight analysis which was performed on the Barebones and EOS-A spacecraft. Preliminary stress and dynamic

Table 4.1-15 EOS and Follow-On Mission Weights and Launch Vehicle Performance

ITEM DESCRIPTION	EOS–A	EOS-A'	EOS-B	EOS-C	EOS-D (SEASAT-B)	EOS-E (TIROS-O)	EOS-F (SEOS)	SEASAT-A	SMM
BAREBONES SPACECRAFT WEIGHT-LB (1)	1485	1485	1485	1485	1485	1485	1485	1485	1485
ORBITER RETRIEVAL INTERFACE	36	36	36	36	36	36	36	36	36
ORBITER RESUPPLY MECHANISM	107	107	107	107	107	107	107	107	107
2 YEAR SERVICE LIFE (BATTERY)	32	32	32	32	32	32	32	32	32
INCREASED STRUCTURAL CAPABILITY	-	-	35	60	-	60	80	-	-
Δ CONTINGENCY	20	20	27	32	20	32	36	20	20
BASIC SPACECRAFT	1680	1680	1722	1752	1680	1752	1776	1680	1680
SPACECRAFT MISSION PECULIAR	(31)	(31)	(44)	(705)	(198)	(1841)	(256)	(122)	(216)
ELECTRICAL POWER (BATTERY)	-	-	-	64	64	32	-	32	-
SOLAR ARRAY	-	-	-	84	84	23	--	50	-
ATTITUDE CONTROL	-	-	-	145	-	145	145	-	145
COMM & DATA HANDLING	4	4	4	4	-	-	-	-	-
ORBIT ADJUST/TRANSFER	27	27	40	351	27	1589	27	27	47
Δ CONTINGENCY	-	-	-	57	23	52	84	13	24
INSTRUMENT MISSION PECULIAR	(400)	(400)	(700)	(920)	(267)	(237)	(549)	(254)	(467)
INSTRUMENT SUPPORT	133	133	174	318	132	133	411	121	190
WIDE BAND RECORDERS (2)	155	155	400	400	-	-	-	-	155
WIDE BAND COMMUNICATION	88	88	96	146	110	80	88	110	88
Δ CONTINGENCY	24	24	30	56	25	24	50	23	34
INSTRUMENTS	(613)	(613)	(853)	(1753)	(706)	(770)	(2300)	(602)	(1431)
MULTI-SPECTRAL SCANNER	160	160	-	-	-	-	-	-	-
THEMATIC MAPPER	400	400	400	800	-	-	-	-	-
HIGH RESOLUTION POINTABLE IMAGER	-	400(Alt)	400	400	-	-	-	-	-
SYNTHETIC APERTURE RADAR	-	-	-	500	-	-	-	-	-
DATA COLLECTION SYSTEM	53	53	53	53	-	-	-	-	-
SEASAT-B (OCEAN DYN. & SEA ICE)	-	-	-	-	706	-	-	-	-
TIROS-O (WEATHER & CLIMATE)	-	-	-	-	-	770	-	-	-
SEOS (GEOSYNCHRONOUS EOS) & OTHER EXPERIMENTS	-	-	-	-	-	-	2300	602	1431
SUBTOTAL - SPACECRAFT	2724	2724	3319	5130	2851	4600	4881	2658	3794
WEIGHT SAVING OPTIONS (3)	112	112	-	-	166	126	-	-	-
TOTAL SPACECRAFT WEIGHT–LB	2612	2612	3319	5130	2685	4474	4881	2658	3794
LAUNCH VEHICLE PAYLOAD CAPABILITY	2660	2660	3730	5150	2825	4500	5600	3350	3900
PAYLOAD MARGIN–LB	48	48	411	20	140	26	719	692	106
LAUNCH VEHICLE (5)	D 2910	D 2910	D 3910	TIIIB	D 2910	TIIIB	TIIIC-7	D 3910	D 2910

NOTES:
1. BAREBONES SPACECRAFT WEIGHT INCLUDES 186 LB CONTINGENCY.
2. SEASAT-B (EOS-D), TIROS-O (EOS-E) AND SEASAT-A MISSIONS UTILIZE TDRS.
3. WEIGHT SAVING OPTIONS EMPLOYED ARE:
   a. ROLL-OUT SOLAR ARRAY (EOS-A, A', D, E)
4. SHUTTLE PAYLOAD LIMIT REDUCED TO ACCOUNT FOR FLIGHT SUPPORT SYSTEM (1490)LB). IN ADDITION, SPACECRAFT WEIGHTS ABOVE INCLUDE 95 TO 135 LB FOR LAUNCH ADAPTER WEIGHT.
5. T III B PAYLOAD LIMITS ARE FOR TITAN IIIB (SSB)/NUS; TITAN III C-7 PAYLOAD LIMITS ARE BASED ON TE 364-4 3RD STAGE.

Table 4.1-16
Lightweight EOS-B Weight Derivation
(Delta 2910 Launch Vehicle)

CURRENT EOS-B SPACECRAFT WEIGHT	3319 LB
DELETE CONS TITAN COMPATIBILITY CHANGES:	(- 55)
STRUCTURE LOCAL REINFORCEMENT	- 35
ORBIT ADJUST PROPELLANT CHANGE	- 13
CONTINGENCY CHANGE	- 7
INCORPORATE LIGHTWEIGHT DESIGN FEATURES:	(-363)
ROLL-OUT SOLAR ARRAY	- 93
HYBRID GR/EP S/C STRUCTURE	-169
HYBRID GR/EP INSTRUMENT SUPT	- 40
CONTINGENCY CHANGE	- 61
LIGHTWEIGHT EOS-B (COMPLETE OPTIONS)	2901
ELIMINATE PROGRAM OPTIONS:	(-445)
RESUPPLY - BASIC SPACECRAFT	-107
- INSTRUMENT	- 32
RETRIEVAL	- 36
REDUNDANCY - S/C BATTERY	- 32
- WB TAPE RCDR	-200
CONTINGENCY CHANGE	- 38
LIGHTWEIGHT EOS-B S/C WEIGHT	2456
DELTA 2910 PAYLOAD CAPABILITY	2660
PAYLOAD MARGIN	+204 LB

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analyses were used to size the major structural members. Theoretical structure weights were computed, and non-optimum factors applied to determine total assembly weight. These factors have been developed over the course of several spacecraft (and numerous aircraft) programs, and have been proven quite accurate. Detailed calculations were performed on the latches and mechanisms, increasing our confidence in the lightweight latching concept. Subsystem equipment weights are based on actual weights of existing components, or quoted vendor weight. Thermal control weights are based on non-optimum factors developed from the actual weights of Lunar Module MLI blankets. Instrument and instrument data handling weights are either vendor estimates or government-specified.

The contingency weight is based on a detailed assessment of each assembly or component, generally using a factor of 20% for structure and new equipment, and 10% for modified existing equipment. No contingency exists for existing (off the shelf) equipment or specification weights. A detailed weight statement for EOS-A is found in Table 4.1-17. The mass properties for EOS-A have been tabulated for several significant configurations in Table 4.1-18.

Table 4.1-17 EOS-A Weight Statement (Sheet 1 of 3)

ITEM	WEIGHT - LB
1.0 STRUCTURE	518
CORE STRUCTURE	(186)
CORE BEAM	70.9
FWD BULKHEAD	15.6
AFT BULKHEAD	30.6
OUTRIGGER TRUSS (6)	68.7
ORBITER INTERFACE STRUCTURE	(36)
FWD FRAME	6.0
CORNER BRACE (3)	4.5
INTERFACE SEGMENT (6)	25.8
MODULE STRUCTURE (3)	(219)
H/C BULKHEAD	79.2
H/C SHELF	18.3
SIDE TRUSS	121.5
ORBIT ADJUST STAGE	(25)
H/C BULKHEAD	3.7
BEAM (6)	9.0
SUPPORT TRUSS	10.1
TANK SUPPORTS	2.3
RESUPPLY MECHANISM	(52)
CORE (TRACK INSTL (5))	7.6
MODULE (3)	30.0
ORBIT ADJUST STAGE	7.2
SOLAR ARRAY	7.2
2.0 ELECTRICAL POWER	204
POWER SUPPLY	(123)
BATTERY	96.0
BATTERY CHARGER	27.0
POWER DISTRIBUTION	(67)
CENTRAL POWER CONTROL UNIT	23.0
GND CHG. DIODE ASSY (2)	11.0
BUS PROTECTION ASSY	5.0
BUS ASSEMBLY (3)	2.0
CONNECTORS	3.0
WIRING & INSTL.	15.0
S/C INTERFACE ASSY	8.0
SIGNAL CONDITIONING	(14)
SIGNAL COND. ASSY.	10.0
REMOTE DECODER (2)	2.0
DUAL REMOTE MUX (2)	2.0
3.0 ELECTRICAL HARNESS & SC	90
BASIC HARNESS	(25)
SPG BUS	0.5
CONNECTORS	3.4
WIRING & INSTL.	21.3
RESUPPLY PENALTY (I/F DISCS.)	(55)
VEHICLE HARNESS	35.0
REPLACEABLE ASSY (5)	20.0
PYRO CONTROL	5
LAUNCH INSTRUMENTATION	5
4.0 SOLAR ARRAY & DRIVE	102
FLEXIBLE (ROLL-OUT) ARRAY	(77)
SOLAR CELL ASSY	44.0
STRUCTURE	8.0
MECHANISM	25.0
SOLAR ARRAY DRIVE	25

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Table 4.1-17 EOS- A Weight Statement (Sheet 2 of 3)

ITEM	WEIGHT - LB
5.0 ATTITUDE CONTROL	156
SENSORS	(44)
COARSE SUN SENSOR	0.3
DIGITAL SUN SENSOR	5.0
RATE GYRO ASSY	15.0
FIXED STAR TRACKER	17.0
MAGNETOMETER	6.5
EVALUATION & CONDITIONING	(16)
CONTROL LOGIC ASSY	13.0
REMOTE DECODER (2)	2.0
REMOTE MUX (2)	1.0
CONTROL	(61)
REACTION WHEEL (3)	30.0
TORQUER BAR (3)	30.6
ELECTRICAL INTEGRATION	(35)
BUS ASSY (3)	2.0
BUS PROTECTION ASSY	5.0
CONNECTORS	8.0
WIRING & INSTL.	20.5
6.0 REACTION CONTROL	37
HYDRAZINE SYSTEM	(14)
THRUSTER (16)	5.6
TANK	2.9
VALVES	1.4
LINES	3.6
SUPPORTS	1.0
SIGNAL CONDITIONING	(8)
SIGNAL COND. ASSY	6.0
REMOTE DECODER	1.0
REMOTE MUX	0.5
ELECTRICAL INTEGRATION	(12)
CONNECTORS	9.5
WIRING & INSTL.	3.0
PROPELLANT (N_2H_4)	3
7.0 COMMUNICATION & DATA HANDLING	111
COMMUNICATIONS	(15)
S-BAND ANTENNA (2)	2.4
INTEGRATED TRANSPONDER	7.8
COAXIAL CABLE INSTL.	5.0
DATA HANDLING	(59)
COMPUTER	20.0
MEMORY MODULE	4.0
CONTROLLER/FORMATTER	3.0
COMMAND DECODER	12.0
SENSORS	20.0
SIGNAL CONDITIONING	(12)
SIGNAL COND. ASSY.	10.0
REMOTE DECODER	1.0
REMOTE MUX	0.5
ELECTRICAL INTEGRATION	(25)
BUS ASSY. (3)	2.0
BUS PROTECTION ASSY.	5.0
CONNECTORS	11.0
WIRING & INSTL.	7.0
8.0 THERMAL CONTROL	72
THERMAL SKINS	(26)
CORE	10.1
MODULE (3)	12.4
RCS/ORBIT ADJUST STAGE	3.1
INSULATION	(43)
CORE	21.4
MODULE (3)	17.3
RCS/ORBIT ADJUST STAGE	4.6
INSTALLATION	1
HEATERS	(2)
CORE(5)	0.5
MODULE (15)	1.5

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Table 4.1-17 EOS-A Weight Statement (Sheet 3 of 3)

ITEM	WEIGHT - LB
9.0 INTERSTAGE ADAPTER	95
LAUNCH ADAPTER	(84)
SKIN	27.9
LONGERON (6)	4.6
FWD RING	10.9
AFT RING	8.1
STIFFENER (12)	11.2
SEPARATION SYS	21.5
SPACECRAFT INSTL.	(11)
SEPARATION I/F	10.7
10.0 MISSION PECULIAR	403
ORBIT ADJUST/TRANSFER	(27)
THRUSTER (4)	4.0
TANKAGE	2.9
PROPELLANT	20.0
INSTRUMENT SUPPORT	(133)
STRUCTURE - BOX BEAM	28.3
- TRUSS	22.8
- PLATFORM	29.3
- IMP BOX	10.0
RESUPPLY MECHANISM (4)	32.4
THERMAL INSULATION	10.9
WIDEBAND TAPE RECORDER (2)	155
WIDEBAND COMMUNICATIONS	(88)
X-BAND STEERABLE ANTENNA	12.0
ELECTRONICS	28.5
WB DATA HANDLING UNIT	14.0
SIGNAL CONDITIONING	33.5
11.0 INSTRUMENTS	613
MULTI-SPECTRAL SANNER	160
THEMATIC MAPPER	400
DATA COLLECTION SYSTEM	(53)
ANTENNA	9.0
ELECTRONICS	44.0
12.0 CONTINGENCY	211
STRUCTURE	(103)
BAREBONES STRUCTURE	86.0
RETRIEVAL I/F	7.0
RESUPPLY MECH	10.0
SUBSYSTEMS	(65)
ELECTRICAL POWER	15.0
ELECTRICAL INTEGRATION	3.0
SOLAR ARRAY & DRIVE	20.0
ATTITUDE CONTROL	14.0
REACTION CONTROL	3.0
C&DH	10.0
LAUNCH ADAPTER	19
MISSION PECULIAR	(24)
INSTR. SUPT.- STRUCTURE	18.0
- RESUPPLY MECH.	6.0
EOS - A SPACECRAFT LAUNCH WEIGHT	2612

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Table 4.1-18 EOS-A Mass Properties Summary

MISSION PHASE	WT	Center of (1) Gravity (in.)			Moment of Inertia (sl-ft^2)			Product of Inertia (sl-ft^2)		
	LB	X	Y	Z	Ixx	Iyy	Izz	Iyz	Ixz	Ixy
ARRAY STOWED										
• BASIC S/C	1483	73.6	-1.8	0.4	262	535	564	4.9	31.2	- 80
• S/C WITH MISSION PECULIAR ITEMS	1886	83.6	-1.5	2.9	328	798	822	7.5	76.7	- 78
• TOTAL S/C WITH INSTRUMENTS	2498	100.7	0.7	2.6	372	1369	1410	-1.1	63.1	- 3.5
• TOTAL S/C WITH INSTRUMENT AND LAUNCH ADAPTER (AT LAUNCH)	2612	97.6	0.7	2.5	407	1506	1546	-1.1	67.5	- 2.3
ARRAY DEPLOYED TOTAL S/C WITH INSTRUMENTS (ON ORBIT)	2498	100.7	-8.1	2.6	1484	1271	2432	11.3	67.3	23.6

NOTES: (1) + x IS IN THE DIRECTION OF FLIGHT (X100 LOCATED IN THE PLANE OF THE S/C FWD BULKHEAD)
+ z IS TOWARD EARTH
+ y COMPLETES RIGHT HAND RULE COORDINATE SYSTEM

3-78 (2) SOLAR ARRAY IS SHOWN FOR NOON DNTD (YZ PLANE)

4.2 SPACECRAFT THERMAL CONTROL

4.2.1 SUMMARY

A matrix of structural concepts has been considered for both the Delta and Titan vehicles. The number of instruments gives additional mission peculiar complexity. With the above considerations, the buildup and use of a detailed comprehensive thermal model, for this study phase, was not feasible. Each section of the structure (i.e. instrument structure, transition area, module support structure, orbit adjust stage) was individually analyzed for an available Delta configuration. Heater power as a function of structure temperature and insulation effective emittance was evaluated. It is clearly recognized that a specific configuration was evaluated, however, the approaches and results should be indicative for all configurations.

In support of the thermal analysis, an orbital heat flux study was conducted and maximum and minimum absorbed heat fluxes established. In addition, unit costs of thermal control hardware were obtained to support the design cost studies. These details and the structure thermal analysis are given in Appendix D, Subsection 1.2.

The results of the structure thermal analysis are summarized as follows:

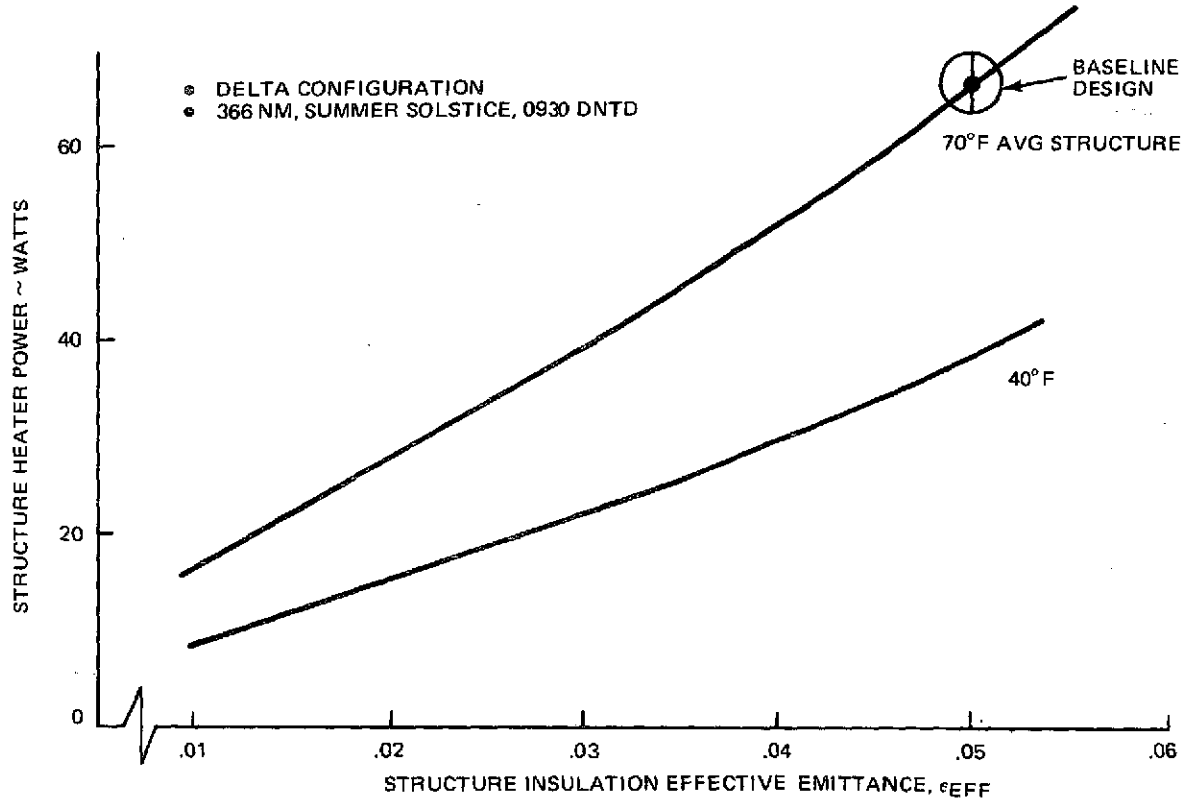
- Reductions in structure heater power from pre-study estimates have been achieved by structure thermal design approaches that minimize external surface area and maximize the use of multilayer thermal insulation. Deletion of thermal skins in the instrument areas and substitution of insulated trusses and decks result in significant reduction in weight, heater power and cost.

- For a baseline requirement of $70^{\circ}F$ structure and insulation effectiveness of .05, the total structure heater power is 66 watts. Using an insulation effectiveness of .02, which should be readily achievable, reduces the heater power to 28 watts. Reducing the structure temperature to $40^{\circ}F$ decreases the heater power requirements to the range of 15-38 watts (range of insulation effectiveness). Although 100 watts of structure heater power was assumed for preliminary solar array sizing, it is apparent that the total structure heater power penalty will be less than 40 watts.
- Preliminary feedback from the instrument contractors indicate concurrence with a thermally decoupled design interface and therefore acceptance of lower structure temperatures. Maintaining the transition ring at $70^{\circ}F$ should be only a transient condition, during contact periods. A module support structure of $40^{\circ}F$ is consistent with the minimum anticipated equipment operating temperatures. A $40^{\circ}F$ OAS structure is consistent with minimum propellant temperature requirements.

4.2.2 DESIGN COST EVALUATION

The structure insulation design/cost trade study is shown in the following figures. Figure 4.2-1 shows the total structure heater power required for a Delta configuration spacecraft as a function of structure insulation effective emittance. Structure temperatures of $70^{\circ}F$ and $40^{\circ}F$ are plotted as parameters. The baseline design ($70^{\circ}F$ structure temperature and .05 effective emittance) requires 66 watts of heater power.

Figure 4.2-2 shows the cost of structure insulation as a function of effective emittance. Improved insulation performance is achieved by different design and insulation techniques. Also shown in Fig. 4.2-2 are the added power subsystem costs for structure heater power. Two extremes for solar array costs are plotted for a range of structure temperature and type of solar array. The cost of insulation plus solar array is shown in this figure to have a minimum value at insulation effective emittance value less than the baseline design. Thus for the more costly roll-up solar array and $70^{\circ}F$ structure temperature, the optimum structure insulation has an effective emittance of .015 to .02. The less costly honeycomb array and $40^{\circ}F$ structure has the minimum total cost with effective emittance in the range of .02 to .03. Definite cost reductions from the baseline design are possible by the use of better performing insulation, the actual performance depending on the particular solar array selected and the structure temperature selected.



3-20

Fig. 4.2-1 Total Structure Heater Power

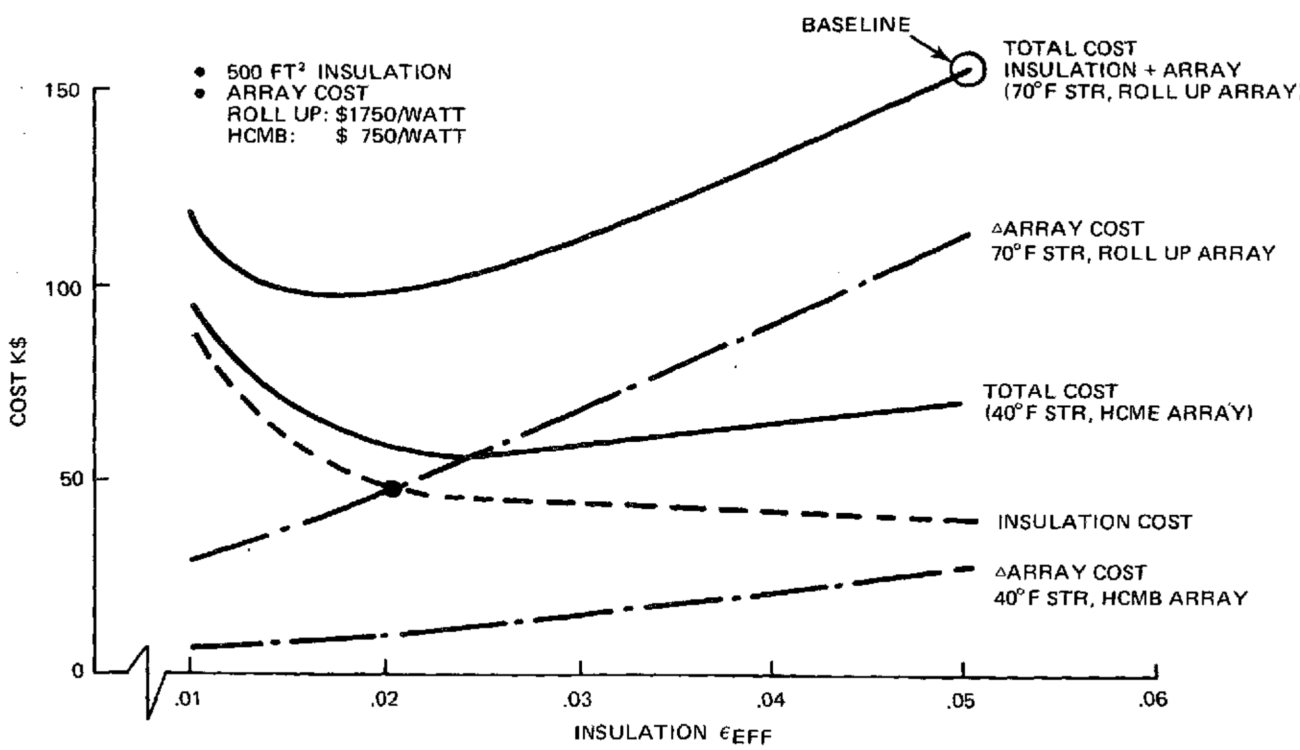


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Fig. 4.2-2 Structure Insulation Plus Delta Array Costs

Since a full system acceptance, test is not being considered, no recurring test costs are incurred. A qualification system test (non-recurring) is being considered in the program, but a wide temperature range will probably be used to evaluate the vehicle. Therefore the structure temperature/test cost impact of this non-recurring cost was not included in the tradeoff.

4.3 SUBSYSTEMS

4.3.1 ATTITUDE CONTROL

An attitude control subsystem (ACS) can be designed to meet the basic requirements of different missions, specifically:

- Earth Pointing
- Stellar
- Low or geosynchronous altitudes

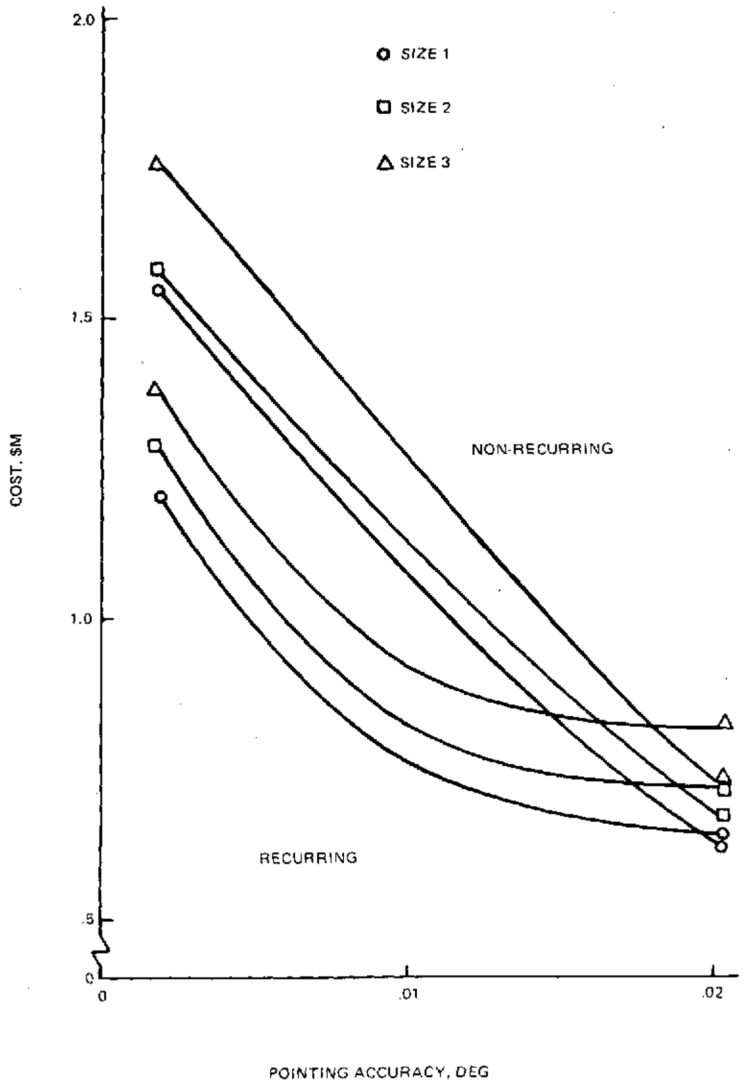
The basic requirements for the ACS are summarized in Table 4.3-1. The range of external disturbance torques results in the need for different sizes of reaction wheels and magnetic torquer bars. The range of missions, from Earth Pointing to Stellar and from low to geosynchronous altitude, results in the need for the update sensors to be capable of operating at low-altitude orbit rate, geosynchronous-altitude orbit rate, and at zero rate.

4.3.1.1 CANDIDATE CONFIGURATIONS

Three candidate ACS configurations were established as shown in Fig. 4.3-1. System (1) meets requirements lower than baseline (0.05 deg attitude accuracy and 5×10^{-6} deg/sec angular rate stability), (2) meets baseline requirements (0.01 deg attitude accuracy and 10^{-6} deg/sec angular rate stability), and (3) meets requirements higher than baseline (0.002 deg attitude accuracy and 0.2×10^{-6} deg/sec angular rate stability). These ACS configurations are summarized in terms of components, cost, weight, and performance in Fig. 4-3-1a, b and c. The components that change with configuration are the sensors (rate gyros, star-trackers, and earth sensor) and the associated software in the C&DH OBC (which is not included in the component listings). Each configuration has three different sizes of wheels and bars: size 1 for spacecraft up to approximately 8,500 lbs, size 2 for spacecraft between 8,500 lbs and 7,000 lbs, and size 3 for spacecraft between 17,000 and 25,000 lb. The size 1 magnetic torquer bars are used with the size 1 reaction wheels etc. Whenever possible, the components selected are space qualified, and when not space qualified are presently in

Table 4.3-1 ACS Basic Requirements

ITEM	REQUIREMENT
MISSIONS	EARTH, STELLAR, SOLAR
MISSION LIFETIME ALTITUDES SPACECRAFT WEIGHT SPACECRAFT INERTIAS SPACECRAFT EXTERNAL DISTURBANCE TORQUE	2 YEARS OPERATIONS PLUS 3 YRS SURVIVAL 300 TO 900 NMI & GEOSYNCHRONOUS 2500 TO 25000 LB 500 TO 100,000 SLUG-FT^2 300 TO 900 NMI: CYCLIC PEAK $\geq 2\times10^{-4}$ and ≤ 0.2 FT-LB AVERAGE $\geq 10^{-5}$ and ≤ 0.1 FT-LB GEOSYNCHRONOUS ALTITUDE: 10% OF THE VALUES GIVEN FOR 300 TO 900 NMI
ACS MODES	ACQUISITION, SLEW (SINGLE-AXIS), INERTIAL ATTITUDE HOLD, EARTH-ORIENTED MISSION, STELLAR MISSION, SOLAR MISSION, SURVIVAL, ORBIT TRIM, AND ORBIT ADJUST. ALSO MAY REQUIRE SRM BURN.
ACQUISITION MODE	SEPARATION RATES $\leq$ 1 DEG/SEC FINAL ATTITUDE $\leq$ 2 DEG FINAL ANGULAR RATES $\leq \pm$ 0.03 DEG/SEC
SLEW (SINGLE-AXIS) MODE	SLEW ANGLE $\leq$ 90 DEG. ACCUMULATED ERROR $\leq \pm$ 0.03 DEG. RATE OF SLEW $\geq$ 2 DEG/MIN
INERTIAL ATTITUDE HOLD MODE	DRIFT BEFORE IN-ORBIT CALIBRATION $< \pm$ 0.03 DEG/HR DRIFT AFTER IN-ORBIT CALIBRATION $< \pm$ 0.003 DEG/HR
EARTH-ORIENTED MISSION MODE	POINT YAW AXIS TO EARTH CENTROID ORBITS: (1) SUN-SYNCHRONOUS (9:30 AM - 12 NOON) CIRCULAR 300-900 NMI (2) GEOSYNCHRONOUS POINTING ACCURACY/AXIS $<$ 0.01 DEG POINTING STABILITY/AXIS: (1) AVERAGE RATE DEVIATION OVER 30 MIN $< \pm 10^{-6}$ DEG/SEC (2) ATTITUDE JITTER RELATIVE TO AVERAGE BASELINE: UP TO 30 SEC $< \pm$ 0.0003 DEG UP TO 20 MIN $< \pm$ 0.0006 DEG IN-ORBIT CALIBRATION ACCEPTABLE
STELLAR-MISSION MODE	TIME INTERVAL $\leq$ 1 HOUR POINTING ACCURACY/AXIS $<$ 0.01 DEG POINTING STABILITY/AXIS: (1) AVERAGE RATE DEVIATION OVER 30 MIN $< \pm 10^{-6}$ DEG/SEC (2) ATTITUDE JITTER RELATIVE TO AVERAGE BASELINE $< \pm$0.0006 DEG WITH PERFECT INSTRUMENT ERROR SIGNALS: (1) POINTING ACCURACY/AXIS $< \pm 3\times10^{-6}$ DEG (2) POINTING STABILITY/AXIS: ATTITUDE JITTER RELATIVE TO AVERAGE BASELINE $< \pm 10^{-7}$ DEG
SURVIVAL MODE	SOLAR ENERGY $\perp$ SUNLINE $< \pm$ 7 DEG ANGULAR RATE/AXIS $<$ 0.05 DEG/SEC TIME: CONTINUOUS. RELIABILITY: 95% SUPPORT SHUTTLE RESUPPLY AND RETRIEVAL
ACS INTERFACE	INSTRUMENT: HAVE CAPABILITY FOR USING INSTRUMENT POINTING ERROR SIGNALS PNEUMATICS: SEND ON-SIGNALS TO JETS C&DH OBC: SEND SIGNALS TO C&DH OBC & RECEIVE SIGNALS FROM OBC
ACS CUTOFF FREQUENCY	APPROXIMATELY 0.1 hz
REACTION WHEELS	NUMBER OF SELECTABLE UNITS $\leq$ 4 INTERCHANGEABLE ELECTRICALLY & PHYSICALLY
MAGNETIC TORQUERS	MAGNETIC FIELD AT EXTERNAL ENVELOPE OF ACS MODULE $\leq$ 0.1 GAUSS
ACS MODULE	DIMENSIONS 48" x 48" x 18" WEIGHT $\leq$ 600 LB POWER $<$ 150 WATTS
MASS EXPULSION	TWO TORQUE LEVELS FOR ON-ORBIT OPERATION: (1) HIGH TORQUE FOR INITIAL STABILIZATION, ORBIT ADJUST, & BACKUP FOR REACTIONWHEEL IN SURVIVAL MODE (2) LOW TORQUE FOR BACKUP MOMENTUM UNLOADING OF REACTIONWHEEL AND POSSIBLY A BACKUP FOR REACTIONWHEEL IN POINTING MODE IN CASE SRMS ARE USED, A THIRD LEVEL OF JET TORQUE IS GENERALLY REQUIRED.



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a. ACS Configuration

COMPONENT	NO. PER S/C	COST $K (Ea.)		WEIGHT EACH, LB.	PERFORMANCE
		NON-RECUR	RECUR		
COARSE SUNSENSOR (BENDIX)	2	5	2	0.156	FOV ± 90°, 2 AXES
DIGITAL SUNSENSOR (ADCOLE)	1	10	42	5	FOV ± 32°, ACCURACY 1' LSB 14 sec
RATE GYRO (BENDIX)	1	10	40	7.25	<1°/HR
EARTH SENSOR (QUANTIC)	1	100	125	45	0.02 DEG
• MAGNETOMETER (SCHOENSTEDT)	1			6.5	± 1.0 GAUSS RANGE
• ELECTRONIC ASSY (ITHACO)	1			13	
MULTIPLEX (HUGHES)	2	0	62	0.5	
DECODER (HUGHES)	2	0	10	1.0	
REACTIONWHEELS, SIZE 1 (BENDIX)	3	10	30	11.3	h = 2 FT-LB-SEC, T = 2 IN-OZ
REACTIONWHEELS, SIZE 2 (BENDIX)	3	10	40	20	h = 8.5 FT-LB-SEC, T - 7.5 IN-OZ
REACTIONWHEELS, SIZE 3 (BENDIX)	3	100	60	22	h = 25 FT-LB-SEC, T = 25 IN-OZ
• MAGTORQUER BARS, SIZE 1 (ITHACO)	3			10.2	m = 45,000 POLE-CM
MAGTORQUER BARS, SIZE 2 (ITHACO)	3			50	m = 450,000 POLE-CM
MAGTORQUER BARS, SIZE 3 (ITHACO)	3			112	m = 4,500,000 POLE-CM
• SET: 1 MAGNETOMETER, 1 ELEC ASSY, 3 MAGTORQUER BARS (SIZE 1)		370	103		
SET: 1 MAGNETOMETER, 1 ELEC ASSY, 3 MAGTORQUER BARS (SIZE 2)		422	229		
SET: 1 MAGNETOMETER, 1 ELEC ASSY, 3 MAGTORQUER BARS (SIZE 3)		474(1)	265(1)		

NOTES: (1) BARS CANNOT FIT IN ACS MODULE. ESTIMATES PROVIDED FOR COMPARISON PURPOSES ONLY.

c. ACS Configuration III

COMPONENTS	NO. PER S/C	COST (EA) $K		WEIGHT EA.	PERFORMANCE
		NON-RECUR	RECUR		
COARSE SUNSENSOR (BENDIX)	2	5	2	0.156	FOV ± 90°, 2 AXES
DIGITAL SUNSENSOR (ADCOLE)	1	10	42	5	LSB 14 SEC, ACCURACY 1 MIN FOV ± 32°
RATE GYRO ASSY (BENDIX)	1	650	235	15	RANDOM DRIFT 0.003°/HR (IOC)
GIMBALED STARTRACKER	1	500	500	50	GIMBAL TRACK ± 45° (2 AXES) FOV ± 30 MIN ACCURACY 5 SEC
MAGNETOMETER (SCHOENSTEDT)	1			6.5	RANGE ± 1.0 GAUSS
ELECTRONICS ASSY (ITHACO)	1			13	
MULTIPLEXER (HUGHES)	2		62	0.5	
DECODER (HUGHES)	2		10	1.0	
REACTION WHEEL (BENDIX, SIZE 1)	3	10	30	11.3	H = 2 FT-LB-SEC, T = 2 IN-OZ
REACTION WHEEL (BENDIX, SIZE 2)	3	10	40	20	H = 8.5 FT-LB-SEC, T = 7.5 IN-OZ
REACTION WHEEL (BENDIX, SIZE 3)	3	100	60	22	H = 25 FT-LB-SEC, T = 25 IN-OZ
TORQUER BARS (ITHACO, SIZE 1)	3			10.2	M = 45,000 POLE-CM
TORQUER BARS (ITHACO, SIZE 2)	3			50	M = 450,000 POLE-CM
TORQUER BARS (ITHACO, SIZE 3)	3			112	M = 4,500,000 POLE-CM
SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 1)		370	193		
SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 2)		422	229		
SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 3)		474	265		

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b. ACS Configuration II

COMPONENTS	NO. PER S/C	COST (EA) $K		WEIGHT EA.	PERFORMANCE
		NON-RECUR	RECUR		
COARSE SUNSENSOR (BENDIX)	2	5	2	.156	FOV ± 90 DEG, 2 AXES
DIGITAL SUNSENSOR (ADCOLE)	1	10	42	5	LBS/14 SEC, ACCURACY 1 MIN FOV ± 32°
RATE GYRO ASSY (BENDIX)	1	650	235	15.0	003° HR (IOC)
FIXED HEAD STAR TRACKER (ITT)	1	40	43	17.0	FOV 8 DEG CIRCULAR ACCURACY 20 SEC (COMPENSATED)
MAGNETOMETER (SCHOENSTEDT) •	1			6.5	RANGE:1 GAUSS
ELECTRONICS ASSY (ITHACO) •	1			13	
MULTIPLEXER (HUGHES)	2	–	62	0.5	
DECODER (HUGHES)	2	–	10	1.0	
REACTIONWHEEL (BENDIX, SIZE 1)	3	10	30	11.3	H = 2 FT-LB-SEC, T = 2 IN-OZ
REACTIONWHEEL (BENDIX, SIZE 2)	3	10	40	20	H = 8.5 FT-LB-SEC, T = 7.5 IN-OZ
REACTIONWHEEL (BENDIX, SIZE 3)	3	100	60	22	H = 25 FT-LB-SEC, T = 25 IN-OZ
TORQUER BAR (ITHACO, SIZE 1)	3			10.2	M = 45,000 POLE-CM
TORQUER BARS (I THACO, SIZE 2)	3			50	M = 450,000 POLE-CM
TORQUERS BAR (ITHACO, SIZE 3)	3			112	M = 4,500,000 POLE-CM
SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 1)		370	193		
SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 2)		422	229		
SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY 3 MAGTORQUERS (SIZE 3)		474	265		

d. Comparison of ACS Configuration

ACS CONFIGURATION	HARDWARE COST, PER SPACECRAFT, $M		WEIGHT PER SPACECRAFT, LB	SYSTEM PERFORMANCE	COMMENTS
	NON-RECUR	RECUR			
1 (low cost)	SIZE 1: 0.615 SIZE 2: 0.667 SIZE 3: 0.719	SIZE 1: 0.638 SIZE 2: 0.704 SIZE 3: 0.800	SIZE 1: 144 SIZE 2: 290 SIZE 3: 482	0.02 DEG 3×10^{-4} DEG/SEC	PERFORMANCE IS NOT AS GOOD AS FOR CONFIG 2 & 3. REQUIRES MORE GROUND PROCESSING.
2 (BASELINE)	SIZE 1: 1.085 SIZE 2: 1.137 SIZE 3: 1.279	SIZE 1: 0.751 SIZE 2: 0.817 SIZE 3: 0.913	SIZE 1: 124 SIZE 2: 270 SIZE 3: 462	0.01 DEG 10^{-6} DEG/SEC	PERFORMANCE SPECS ARE MET. CALCULATION OF STAR TRACKER IS DONE IN OBC.
3 (EXPANDED CAPABILITIES)	SIZE 1: 1.545 SIZE 2: 1.597 SIZE 3: 1.739	SIZE 1: 1.208 SIZE 2: 1.274 SIZE 3: 1.370	SIZE 1: 157 SIZE 2: 303 SIZE 3: 495	0.002 DEG 10^{-6} DEG/SEC	POINTING PERFORMANCE IS IMPROVED BEYOND THAT OF CONFIG. 2

Fig. 4.3-1 ACS Candidate Configurations

development. The capability to handle solar and stellar missions in addition to the earth pointing missions is present in ACS Configurations 2 and 3 but not in 1.

The capability of the ACS Configuration 1 exceeds that of the ERTS-A. This system is the least costly, complex, and versatile. ACS Configuration 2 is the baseline design, in which gyro control is normally maintained, with updates using a fixed-head star tracker. Extensive use is made of the C&DH OBC. This system is of medium cost, complexity, and versatility. The range of missions capable of being satisfied include earth-pointing, stellar, and solar. In ACS Configuration 3, a gimbaled star tracker having high resolution and accuracy is used to achieve the highest accuracy obtainable within the current state-of-the-art. Candidate gimbaled star trackers are the TRW proposed (unqualified) of PADS, the Bendix Skylab, and the Kollsman Instrument (OAO-Type). Both the Bendix and Kollsman gimbaled star trackers are space qualified but require modifications to incorporate high-resolution angle resolvers.

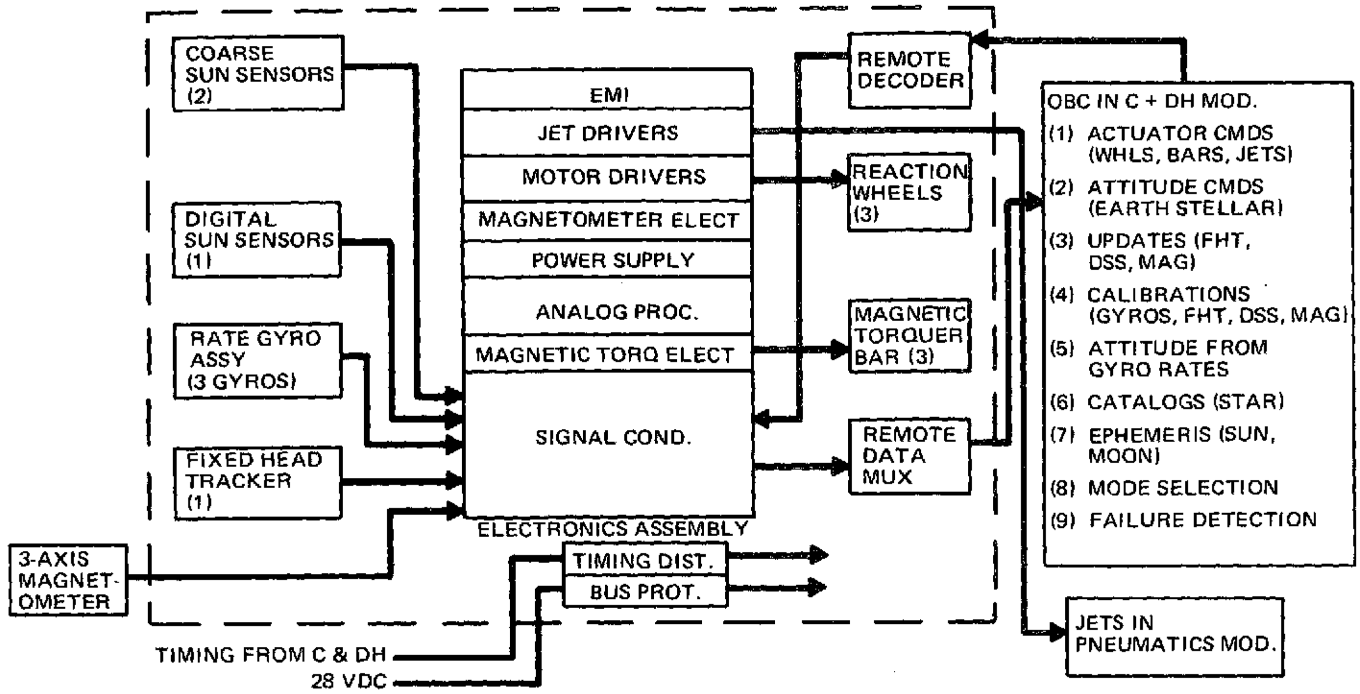
The three ACS configurations are compared on a cost, weight, and performance basis in Fig. 4.3-1d. The costs and weights vary with the use of different-size reaction wheels and magtorquer bars. The weights for all sizes remain below 600 lb. The recurring cost varies from 0.638 $M (ACS Config 1, size 1) to 1.370 $M (ACS Config 3, size 3).

4.3.1.2 SELECTED ACS CONFIGURATION

The selected configuration (2) (Fig. 4.3-2), is adequate, in most respects, for the earth-pointing (low and geosynchronous orbit altitudes), stellar/inertial, and solar missions. Sensors are available in flight-proven design with adequate accuracy and sensitivity. Reaction wheels providing up to 8.5 ft-lb-sec and 6 in-oz torque have also been flight qualified. These wheels can be easily qualified to 20 ft-lb-sec and 15 in-oz with minimal development cost/risk. Larger wheels capable of 50-100 ft-lb-sec and 25-50 in-oz are under development and would be available for those missions requiring them. The concept of providing the ACS control algorithms as a mission-peculiar software program to be processed in the OBC is viable.

Modal operations are functionally described, including a listing of the ACS sensors and actuators used in each mode, in Table 4.3-2.

Candidate components were assembled and compared on the basis of cost, performance, qualification status, availability, weight and power etc. Their features and cost are shown in Table 4.3-3.



BLOCK DIAGRAM OF BASELINE EOS ACS

COMPONENTS	NO. PER S/C	COST (EA) $K NON-RECUR	RECUR	WEIGHT, (EA)
COARSE SUNSENSOR (BENDIX)	2	5	2	0.156
DIGITAL SUNSENSOR (ADCOLE)	1	10	42	5
RATE GYRO ASSY (BENDIX)	1	650	235	15.0
FIXED HEAD STAR TRACKER (ITT)	1	40	43	17.0
MAGNETOMETER (SCHOENSTEDT)*	1			6.5
ELECTRONICS ASSY (ITHACO)*	1		193	13
REMOTE UNIT	2	–	10	1.0
REACTIONWHEEL (BENDIX, SIZE 1)	3	10	30	11.3
REACTIONWHEEL (BENDIX, SIZE 2)	3	10	40	20
REACTIONWHEEL (BENDIX, SIZE 3)	3	100	60	22
TORQUER BAR (ITHACO, SIZE 1)*	3			10.2
TORQUER BAR (ITHACO, SIZE 2)	3			50
TORQUER BAR (ITHACO, SIZE 3)	3			112
*SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 1)		370	193	
SET: 1 MAGNETOMETER, 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 2)		422	229	
SET: 1 MAGNETOMETER 1 ELECTRONICS ASSY, 3 MAGTORQUERS (SIZE 3)		474	265	

3-23

Fig. 4.3-2 Selected ACS Configuration

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Table 4.3-2. Model Operations

PHASE	MODE		ATTITUDE CONTROL REQUIREMENTS	CONTROL SENSORS	CONTROL ACTUATORS	DESCRIPTION OF MODE
EOS SEPARATION	NULL SEPARATION RATES		SEPARATION RATES < 1 d/s FINAL ANGULAR RATES ⩽ ± 0.03 d/s	R,P,Y RATE GYROS	R,P,Y 1-LB JETS	FOLLOWING SEPARATION FROM BOOSTER, SPACECRAFT RATES ARE NULLED
	INERTIAL HOLD		ANGULAR RATES ⩽ ± 0.03 d/s	R,P,Y RATE GRYOS	R,P,Y 1-LB JETS	
POWER UP	COARSE SOLAR POINT		ALIGN SPACECRAFT -Z AXIS TO SUN < 2 DEG. FINAL ANGULAR RATES < ± 0.03 d/s	R,P CSS MOUNTED ON -Z AXIS (MOUNTED ON ACS MODULE). R,P,Y RATE GYROS	R,P,Y 1-LB JETS	
	DEPLOY SOLAR ARRAY		MAINTAIN SPACECRAFT -Z AXIS TO SUN < 2 DEG. & ANGULAR RATES < ± 0.03 d/s	R,P CSS MOUNTED ON -Z AXIS (MOUNTED ON ACS MODULE). R,P,Y RATE GYROS	R,P,Y 1-LB JETS	
	ORIENT SOLAR ARRAY NORMAL TO SUN		ALIGN SOLAR ARRAY NORMAL TO SUN < 7 DEG. FINAL ANGULAR RATES < ± 0.03 d/s	CSS MOUNTED ON SOLAR ARRAY. R,P,Y GYROS	R,P,Y 1-LB JETS	SOLAR ARRAY DRIVE UNIT IS LOCKED
	MAINTAIN SOLAR ARRAY NORMAL TO SUN		ALIGN SOLAR ARRAY NORMAL TO SUN < 7 DEG. FINAL ANGULAR RATES < ± 0.03 d/s	CSS MOUNTED ON SOLAR ARRAY R,P,Y GYROS	R,P,Y REACTION WHEELS WITH MAGNETIC UN-LOADING	SOLAR ARRAY DRIVE UNIT IS LOCKED
CHECKOUT						APPROXIMATELY 1 MONTH. STAR TRACKERS ARE NOT TURNED ON FOR A MINIMUM OF 3 DAYS (UN-LESS STAR TRACKER DESIGN PERMITS IT) TO PREVENT ARC-ING DUE TO OUTGASSING
	INITIAL PRECISION UPDATE	COARSE SOLAR POINT	SAME AS ABOVE			
		FINE SOLAR POINT	ALIGN SPACECRAFT -Z AXIS TO SUN < 0.1 DEG. FINAL ANGULAR RATES < ± 0.03 d/s	R, P DSS MOUNTED ON -Z AXIS (MOUNTED ON ACS MODULE). R,P,Y RATE GYROS	R,P,Y FRACTION WHEELS. MAGNETIC UNLOADING	UPDATE ATTITUDE COMPUTATION IN OBC. UTILIZE DSS AND MAGNETO-METER SIGNALS FOR 3 AXIS UPDATE. (EPHEMERIS, & MODEL OF EARTH'S MAGNETIC FIELD IS USED).
		SLEW ABOUT Z AXIS (SUNLINE)	ACHIEVE SLEW RATE ⩽ 4 MIN SLEW RATE ⩾ 2 d/m REDUCE TO ZERO RATE ⩽ 4 MIN	R,P,Y RATE GYROS. R,P DSS	R,P,Y 1-LB JETS	THE SLEW ANGLE IS CALCULATED TO ARRIVE AT AN ATTITUDE IN WHICH THE FHT SHOULD SEE A GUIDE STAR
		INERTIAL HOLD	ANGULAR RATES ⩽ ± 0.03 d/s	R,P,Y RATE GYROS	R,P,Y REACTION WHEELS. MAG-NETIC UNLOAD-ING	FHT SEARCHES FOR ITS STAR. IF FOUND, FHT GOES INTO TRACK MODE. UPDATE USING DSS & FHT. IF STAR NOT FOUND, ASSESS & USE ALTERNATE PROCEDURE
	CALI-BRATE GYROS	INERTIAL HOLD	ANGULAR RATE ⩽ ± 0.03 d/s	R,P,Y RATE GYROS	R,P,Y REACTION WHEELS. MAG-NETIC UNLOAD-ING	USING UPDATES FROM FHT AND DSS, CALIBRATE GYROS. INSERT RATE BIAS ERRORS IN OBC
	EARTH-POINT	SLEW	ACHIEVE SLEW RATE ⩽ 4 MIN. SLEW RATE ⩾ 2 d/m REDUCT TO ZERO RATE ⩽ 4 MIN	R,P,Y RATE GYROS	R,P,Y REACTION WHEELS	USING SEQUENTIAL SLEWS, SLEW TO ATTITUDE WHICH AT A LATER POINT IN THE ORBIT RESULTS IN AN EARTH POINTING ATTI-TUDE. UPDATE WITH FHT DURING SLEWS.
		INERTIAL HOLD	ANGULAR RATES ⩽ ± 0.003 d/s	R,P,Y RATE GYROS	R,P,Y REACTION WHEELS. MAG-NETIC UNLOAD-ING	HOLD ATTITUDE.
		DEVELOP ORBIT RATE ABOUT PITCH AXIS	PITCH RATE = ORBIT RATE (≈ 0.06 °/s) ± 10^{-6} d/s	R,P,Y RATE GYROS	R,P,Y JETS	ALLOWING TIME TO BUILDUP PITCH RATE TO ORBIT RATE, COMMENCE AT PROPER MOMENT TO BUILDUP PITCH RATE TO OR-BIT RATE, SO THAT WHEN PITCH RATE = ORBIT RATE, SPACECRAFT ATTITUDE IS SUCH THAT INSTRU-MENTS POINT AT EARTH AND SPACECRAFT X AXIS POINTS IN THE DIRECTION OF FLIGHT. UPDATE USING FHT AT EACH OPPORTUNITY FROM THIS POINT ON.
		HOLD ORBIT RATE. CORRECT FOR ATTITUDE ERRORS.	PITCH RATE = ORBIT RATE ROLL AND YAW RATES OSCILLATE SINUSOIDALLY WITH MAGNITUDE 1 °/DAY, THE SINUSOID VARYING AT ORBIT RATE. POINT AT EARTH < 0.01 d RATE STABILITY OVER 30 MIN <±10^{-6} d/s JITTER UP TO 30 SEC < ± 1 sec UP TO 20 MIN < ± 2 SEC	R,P,Y RATE GYROS	R,P,Y REACTION WHEELS. MAG-NETIC UNLOAD-ING	HOLD PITCH RATE = ORBIT RATE. CONTINUE TO UPDATE WITH FHT. MAKE CORRECTIONS TO RATE COMMANDS BASED ON UPDATES. CALIBRATE GYROS USING FHT UPDATES. SLIGHT OSCILLATORY RATE COMMANDS ARE PUT INTO ROLL & YAW AXES FOR ORBIT REGRESSION. AN EPHEMERIS CALCULATION IS USED TO COM-PUTE ATTITUDE COMMANDS. THE DSS IS NOT USED FOR UP-DATING WHEN THE FHT PRO-VIDES SUFFICIENT UPDATES.
EARTH-POINT MISSION	EARTH-POINT					AS DESCRIBED ABOVE UNDER CHECKOUT
STELLER MISSION	STELLAR-POINT		POINTING ACCURACY < 0.01 d POINTING STABILITY OVER 30 MIN < ±10^{-6} d/s JITTER < ± 2 SEC WITH INSTRUMENT SIGNALS: POINTING ACCURACY < ± 0.01 SEC JITTER < ± 0.003 SEC	R,P,Y RATE GYROS	R,P,Y REACTION WHEELS. MAG-NETIC UNLOAD-ING	A SEQUENCE OF SLEWS ARE MADE TO A DESIRED ATTITUDE. IN THE CASE THAT THE INSTRU-MENT ERROR SIGNALS ARE USED, SETTLING IS REQUIRED. UPDATES ARE PERFORMED USING FHT AND INSTRUMENT TELESCOPE ERROR SIGNALS WHEN AVAILABLE. SLEWS BE-TWEEN TARGETS ARE RELA-TIVELY SLOW FOR TARGETS SCATTERED IN THE SKY, USING THE 2 d/m REQUIREMENT.

FOLDOUT FRAME 2

Table 4.3-3 Candidate Components and Associated Characteristics

Candidate Components				Significant Technical Features				Procurement Cost $K		
Function	Seller	Model	Status	Function/Capability	Wt Ea	Avg Pwr	"Qual" Status	Recur	Non-Recur	Comments
COARSE SUN SENSOR	BENDIX	1771858 (SKYLAB)	EXIST	4 π STERADIAN COVERAGE (2 SENSORS)	0.156	–	SIMILARITY	4	7	
	ADCOLE	C-1694 (OAO)	EXIST	4 π STERADIAN COVERAGE (9 SENSORS)	0.19	–	SIMILARITY	40	10	
	ADCOLE	C-1702 ATS	EXIST	4 π STERADIAN COVERAGE (2 BLOCKS)		–	SIMILARITY	40	10	
DIGITAL SUN SENSOR	ADCOLE	C-1594	EXIST	± 32° FOV, 1 MIN ACCURACY 14 BITS, LSB 14 SEC	5		SIMILARITY	40	10	
GYRO ASSY (3 GYROS; 1/AXIS)	HONEYWELL	GG334	EXIST	DIRECT ROLL, PITCH AND YAW SENSING • JEWEL DITHERED SUSPENSION • GAS BEARING • SHORT TERM DRAFT .002°/HR	15		Δ QUAL			
	NORTHROP	K76 3C	EXIST	• TAUT WIRE SUSPENSION • GAS BEARING • SHORT TERM DRAFT .002°/HR	–	–	Δ QUAL			
	BENDIX	64 PM RIG	EXIST	• MAGNETIC SUSPENSION • GAS BEARING	15	25	Δ QUAL	205	634	
FIXED HEAD TRACKER	BENDIX	OAO	MOD.	20 ARC SEC ACCY, + 5 MAG	16.7	8.7	Δ QUAL	100	394	
	BALL BROS.	SAS C	MOD.	20 ARC SEC ACCY, 3, 4, 5 MAG COMMANDABLE	13.8	6.0	Δ QUAL	77	65	
	ITT	ELMS	EXIST	20 ARC SEC ACCY, 3, 4, 5 MAG COMMANDABLE	17	9	SIMILARITY	43	83	
ELECTRONICS ASSY	ITHACA		NEW	ASSY CONSISTS OF 1) SIGNAL CONDITIONING 2) ANALOG PROCESSOR 3) WHEEL DRIVERS 4) MAG. TORQUER DRIVERS 5) MAGNETOMETER ELECT.	13	15	REQUIRED	193.2	370.4	*THREE SIZES ARE PROPOSED 1) 45 AMP - MET² 2) 450 AMP - MET² 3) 4500 AMP - MET²
	BENDIX							229.1	422.4	
REACTION WHEELS (3)	BENDIX	1880272 OAO	EXISTS	TORQ = 2 IN OZ H = 2.06 FT-LB-SEC NLS = 1200 RPM	10	23	SIMILARITY	30	40	
	BENDIX	188026 ATS	EXISTS	TORQ = 20 IN-OZ H = 8.47 FT-LB-SEC NLS = 1500 RPM	19.5	10	SIMILARITY	40	40	
	BENDIX		MOD	TORQ = 25 IN-OZ H = 25 FT-LB-SEC NLS = 3000 RPM	22	18	Δ QUAL	60	100	
GIMBALED STAR TRACKER	KIC	OAO	EXIST	OFV 1' x 1' GIMBAL ± 43° 5 SEC ACCY	50	15.4	SIMILARITY			
	BENDIX	OAO	EXIST (DID NOT FLY)	FOV 1' x 1' GIMBAL ± 43°			SIMILARITY			
	TRW	PADS	MOD	FOV 5' x 5' GIMBAL INNER ± 15° OUTER ± 45°	50	28	Δ QUAL			
HORIZON SENSOR	BARNES QUANTIC	13-166	MOD EXIST	2-AXIS EARTH SENSING ± 1° ± 0.02°	21.2 45	20 20	SIMILARITY SIMILARITY	195.5 125	16.6 100	

4.3.1.3 SELECTED ACS COMPONENTS

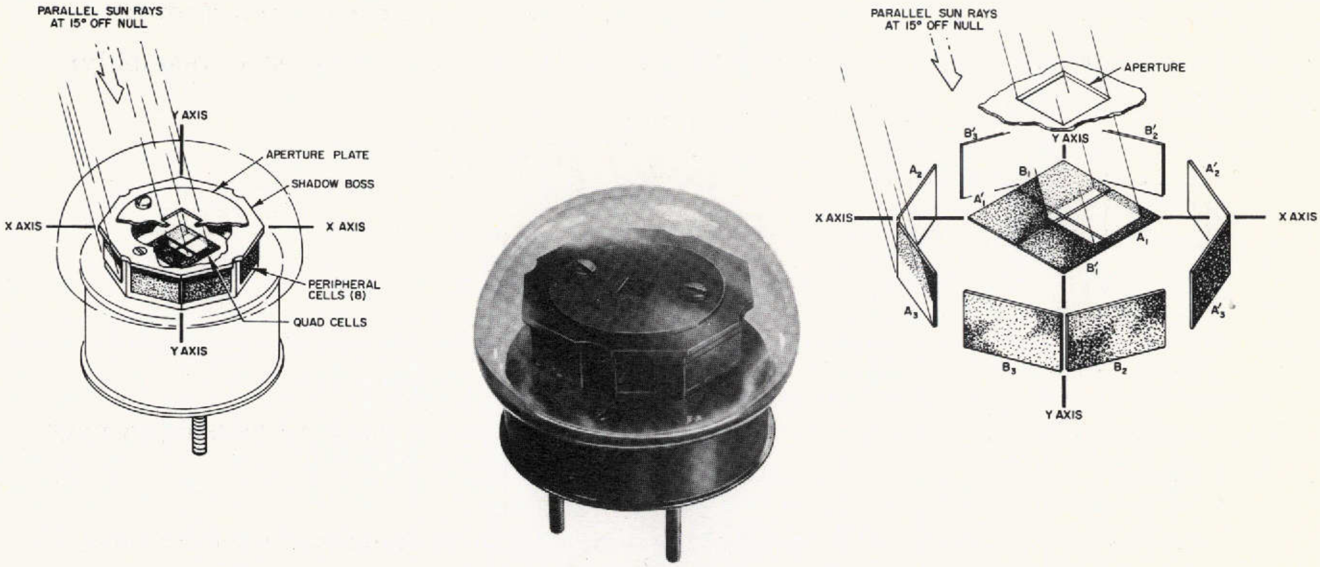
COARSE SUN SENSORS - Sun Sensor information is required along two axes, namely the axes which lie in the plane of the solar array. During acquisition, the sun sensors provide 4π steradian coverage so that the sun can be acquired from any orientation of the vehicle. The coarse sun sensor system proposed is the Bendix WASS Model 1771858. The wide angle sun sensor field of view is considerably in excess of a hemisphere so that two such units would suffice for acquisition over the full sphere without precise alignment. The unit consists of a basic four-quadrant set of photocells arranged in dual opposition, and a set of peripheral cells for angles far off the axis. See Fig. 4.3-3a.

DIGITAL SUN SENSOR - The fine sun sensor proposed for EOS is a high-resolution digital solar (Ref. Fig. 4.3-3b) with an accuracy exceeding 1 minute of arc and a field of view of 32 x 32 degrees. The resolution of this device is 1/256 degrees, or 14 arc seconds. The Digital Sun Sensor is used during initial acquisition, and for gyro update. A gray-coded pattern on the bottom of a quartz block screens light passing through a slit on the top of the block to either illuminate or not illuminate each of the photocells. The angle of incidence determines which photocells are illuminated. The photocell outputs are amplified and presence of a "1" or "0" is stored in a register to provide the output for use in the attitude computer as well as in telemetry. The unit, manufactured by Adcole, is fully space qualified.

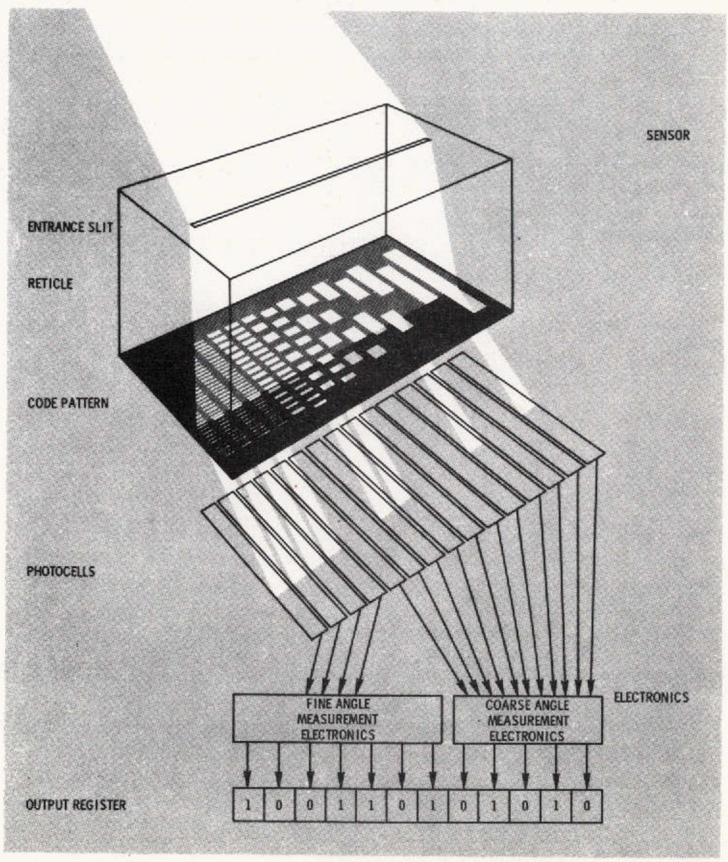
GYRO ASSEMBLY - The Gyro Assembly (Ref. Fig. 4.3-4) functional output is three digital 16-bit voltages proportional to the input inertial rates. The assembly consists of an orthogonal triad of rate-integrating gyros configured for closed-loop (rate) operation. The gyro triad is hard-mounted to the vehicle so that the gyro input axes are colinear with the roll, pitch and yaw axes of the spacecraft.

The Gyro Assembly is configured around the Bendix 64 PM RIG-60 single degree-of-freedom gyroscopic sensing unit. The gyroscopic unit incorporates a hydrodynamic spin motor (wheel) within a cylindrical float which is suspended from a self-contained, hydrostatic liquid bearing and electromagnetic suspension. The unit will require a delta qualification program to satisfy EOS needs.

FIXED-HEAD TRACKER - The Fixed-Head Tracker (FHT, Ref. Fig. 4.3-4) provides attitude sensing information for attitude determination and gyro assembly update. The selected FHT is built by ITT and is based on their existing Fine Guidance Error System which



A) COARSE SUN SENSOR



B) FINE SUN SENSOR

Fig. 4.3-3 Selected Sun Sensors

has successfully flown on over 17 Aerobee rocket flights and is presently being modified and qualified for use on the ELMS satellite program. The SRA (Star Reference Assembly) consists of the following subassemblies:

- Star Aspect Sensor (SAS)
- Power Supply (PS)
- Bright Object Sensor (BOS)
- Earth Albedo/Sun Shade (EA/SS)

The FHT provides two-axis star position signals for identifiable stars passing through the field of view.

The SRA has four commandable threshold levels. The SRA operates normally when the target stars are within the minimum and maximum range described as follows:

- Minimum star; +5.30 visual magnitude, FOV or bluer spectral class
- Maximum star; +2.0 visual magnitude, FOV or redder spectral class

ELECTRONICS ASSEMBLY - The Electronic Assembly (EA) proposed by Ithaco, Inc. (Ref. Fig. 4.3-5a) contains the major portion of the control system electronics. It has within one envelope the following operations:

- Signal conditioning
- Analog processor
- Wheel drivers
- Magnetic torquer drivers
- Jet drivers
- Magnetometer electronics

The EA will be made in a modular arrangement of electronic cards in metal frames with interconnections made by a wiring harness. The unit requires a qualification program to satisfy EOS needs.

REACTION WHEELS - The Bendix N&C division has been selected for the proposed reaction wheels. The size 1 wheel will be similar to the one flown on OSO, which is space qualified. (See Fig. 4.3-5b.)

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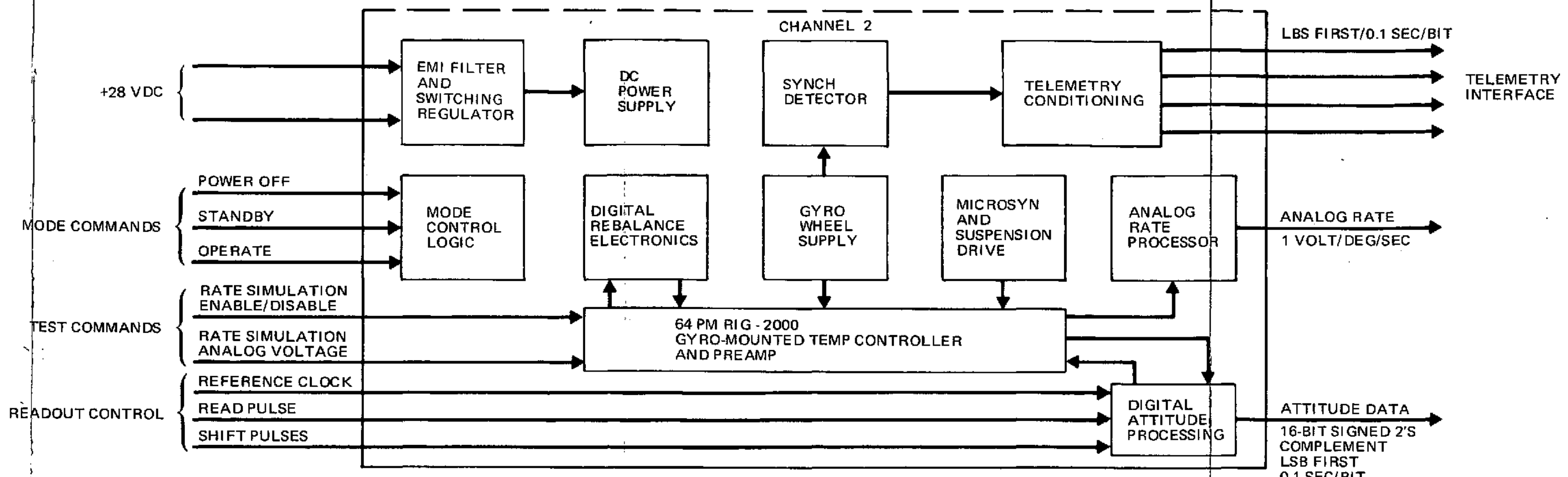
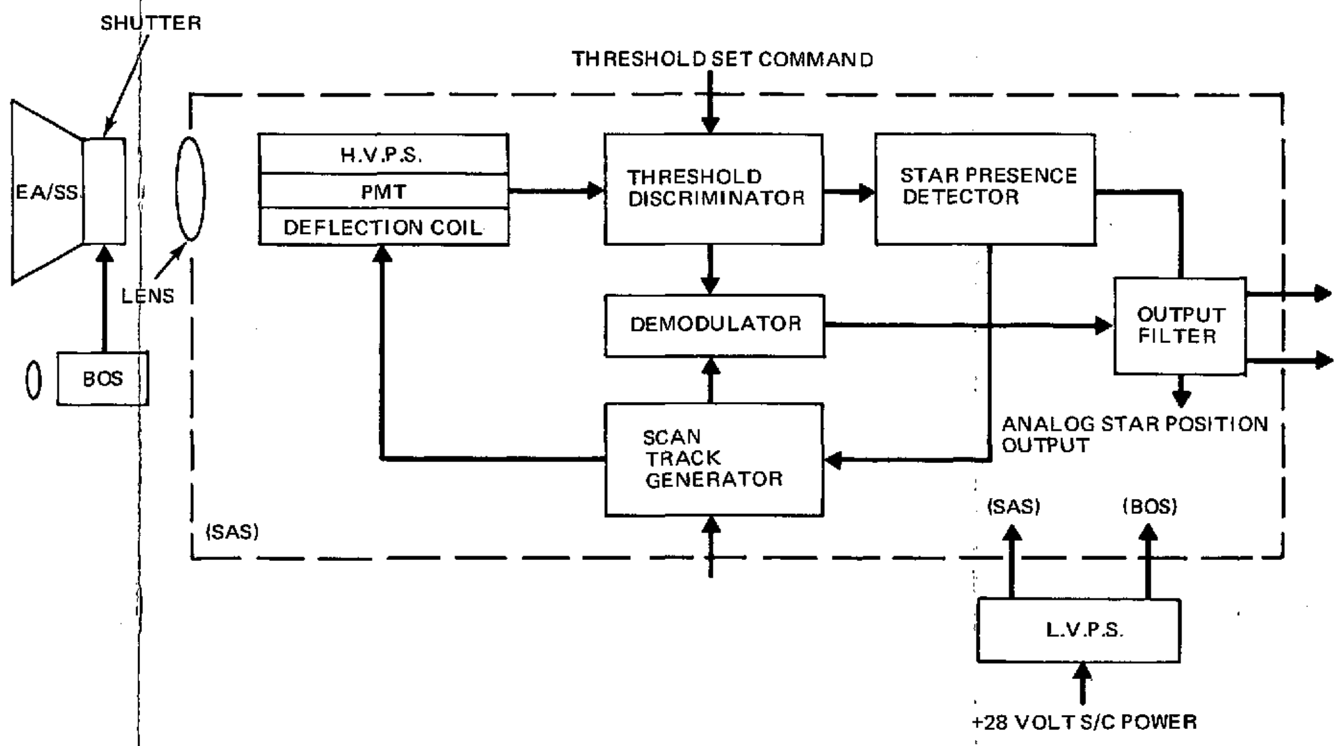
GYRO PERFORMANCE CHARACTERISTICS

PARAMETER	EGA CAPABILITY
BANDWIDTH	> 10 Hz
GAIN MARGIN	18 db
PHASE MARGIN	87 DEG
ATTITUDE MEAS	–
QUANTIZATION	0.1 ARC SEC/COUNT
INPUT RANGE	± 10 DEG/SEC
ATTITUDE SCALE FACTOR	–
SCALE FACTOR NON LINEARITY	< 100 PPM ± 0.05-2.25°/SEC INPUT
SCALE FACTOR STABILITY	< 10 PPM
RANDOMNESS	< 100 PPM
2 DAY STABILITY WITH SHUTDOWN	
LONG TERM STABILITY	< 1300 PPM
DRIFT CHARACTERISTICS	–
G-INSENSITIVE DRIFT	
MAX VALUE	< 1.0 DEG/HR
SHORT TERM DRIFT	< .001 DEG/HR
DRIFT RATE RAMP	< .0001 DEG/HR^2
PEAK ATTITUDE NOISE	< 3 ARC-SEC
SHORT TERM NOISE	< 0.5 ARC-SEC
2 DAY STABILITY WITH SHUTDOWN	< 0.05 ARC-SEC
DRIFT RATE (DYNAMIC ENVIRONMENT)	< .003 DEG/HR (MAX)
G-SENSITIVE DRIFT RATE MAX VALUE	< 1.3 DEG/HR-G
2 DAY STABILITY WITH SHUTDOWN	< 0.4 DEG/HR-G
TEMPERATURE SENSITIVITY	–
G-INSENSITIVE DRIFT	.00555 DEG/HR/°F
SCALE FACTOR	8.3 PPM/°F
ALIGNMENT	1.5 ARC SEC/DEG F
VOLTAGE SENSITIVITY	–
G-INSENSITIVE DRIFT	< .0005 DEG/HR-VOLT
SCALE FACTOR	< 5 PPM/VOLT
MAGNETIC FIELD SENSITIVITY	
G-INSENSITIVE DRIFT	.022 DEG/HR
SCALE FACTOR	19 PPM
WHEEL SPIN POWER	< 5 PERCENT



REFERENCE GYRO ASSEMBLY (RGA) SINGLE AXIS
FUNCTIONAL BLOCK DIAGRAM



BLOCK DIAGRAM OF ITT SRA (FHT)

STAR SENSOR ACCURACY

ERROR SOURCE		VALUE, ARC-SEC (1σ) WITHOUT	WITH CORRECTIONS
TEMPERATURE		90	12.0
MAG FIELD		30	6.0
FIELD REPEATABILITY		10	10.0
HYSTERESIS		2	2.0
LAG ERROR – TRK LOOP	BANDWIDTH 4 Hz	2	2
– ERROR AMP	BANDWIDTH 4 Hz	4	4
NEA		22	6.0+
TOTAL (1σ) (PRELAUNCH)			18.4 ARC-SEC‡
NULL SHIFT (DUE TO LAUNCH)			10.0
TOTAL (1σ) (ON-ORBIT)			18.4 ARC-SEC
ON-ORBIT ALIGNMENT CALIBRATION			4*
TELEMETRY RESOLUTION			4
TOTAL (1σ)			

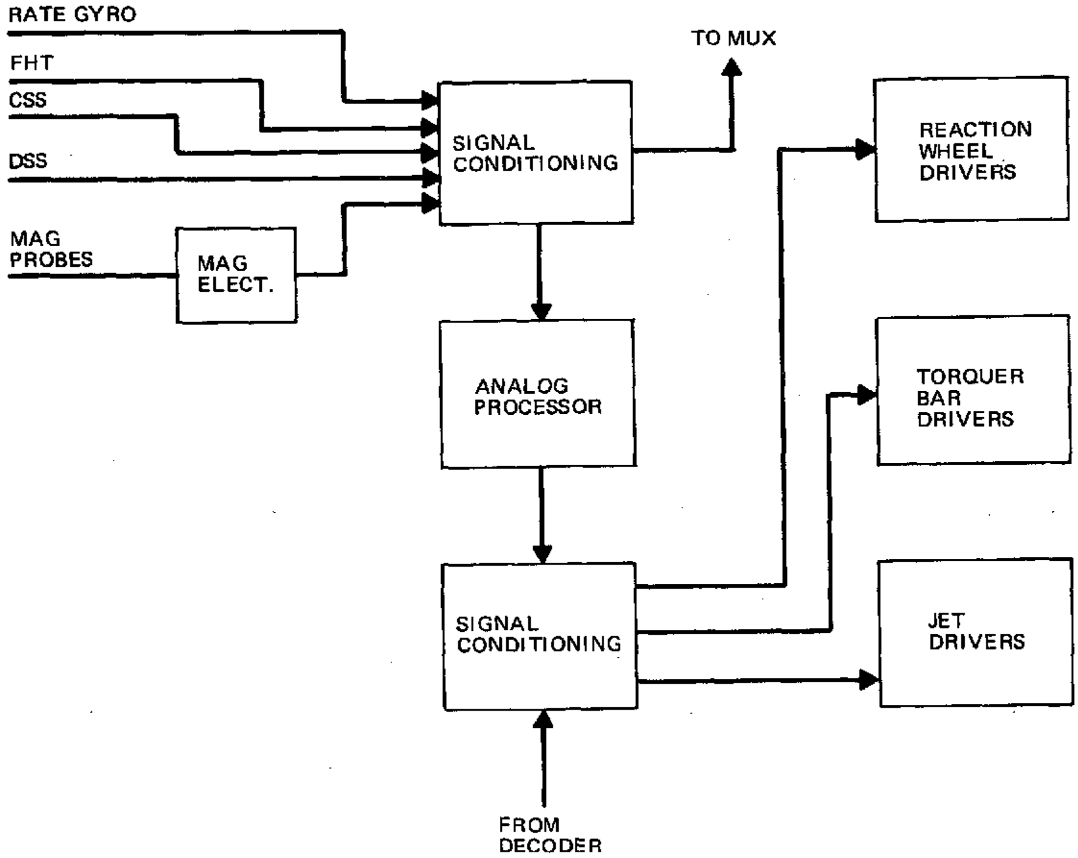
*IN FLIGHT ALIGNMENT ACCURACY FOR ROLL
+AFTER FILTERING
‡EXCLUSIVE OF NULL SHIFT

Fig. 4.3-4 Gyro Assembly and Fixed-Head Tracker

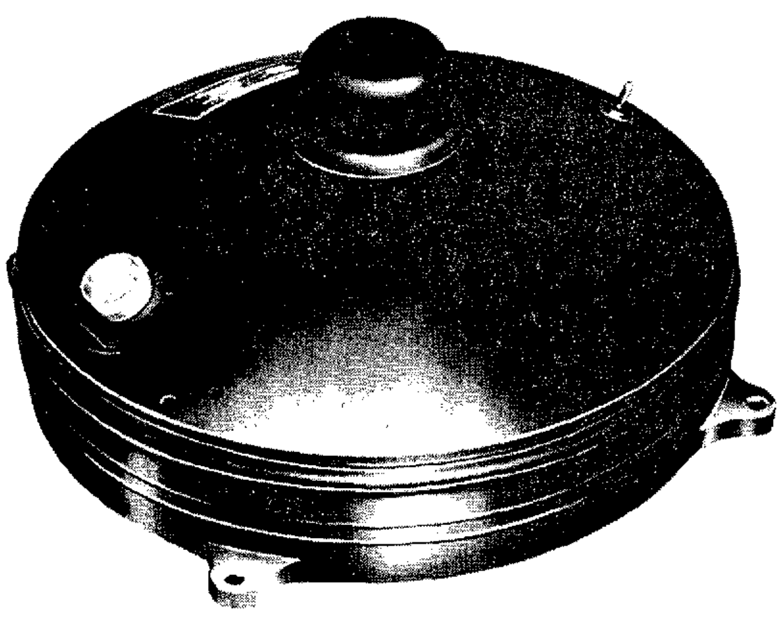
FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3



a. **Functional Block Diagram of Electronics Assembly**



b. **Reaction Wheel**

Fig. 4.3-5 Electronics Assembly and Reaction Wheel

MAGNETOMETER - The magnetometer (Ref. Fig. 4.3-6a) operates on the flux gate principle. The probe is excited with a 200 Hz signal. The probe output contains even harmonics of 2 kHz whose amplitude is proportional to the vector component of the magnetic field aligned along the probe axis. The probe contains a very thin sliver of magnetic material which is the core of a transformer. One winding of the transformer is driven by the 2 kHz sine wave. The presence of the earth's steady dc field saturates the thin sliver of magnetic material, and the resulting ac magnetic field, which is sensed by a second winding on the transformer, is distorted and contains a large second harmonic component. The phase of this component in relation to the 2 kHz drive signal is a function of the direction of the earth's steady field. A block diagram of the magnetometer is shown in Fig. 4.3-6a. This unit is space qualified.

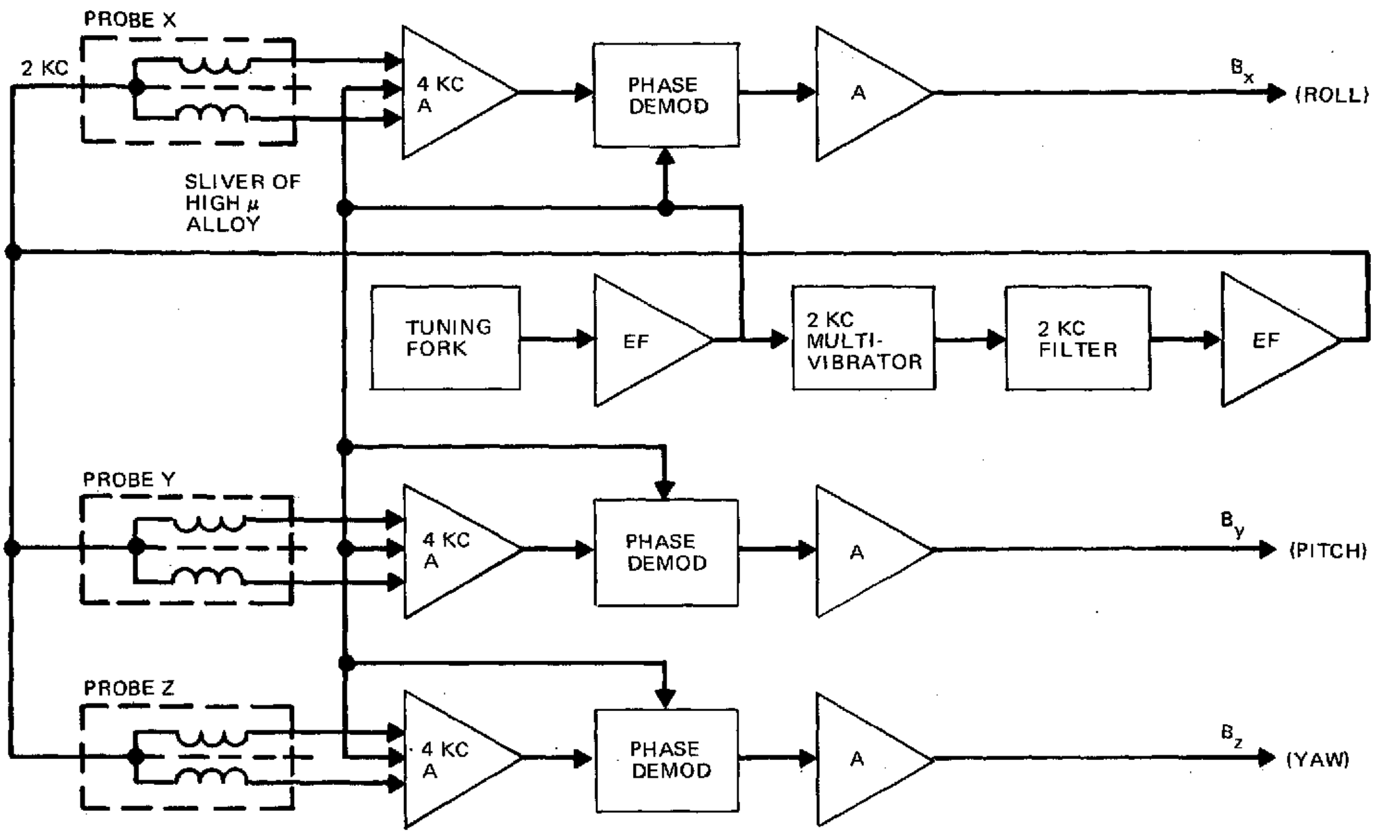
MAGNETIC TORQUER BARS - Three torquer bars, one parallel to each axis, generate the magnetic fields that interact with the earth's magnetic field to produce a torque on the observatory system (Ref. Fig. 4.3-6b).

The torquer bars are electromagnets which develop a magnetic moment capable of developing torque when operated in the earth's magnetic field (which is approximately 0.3 gauss for an orbit of 500 nm). There are three of these bars, one along each of the spacecraft's control axes.

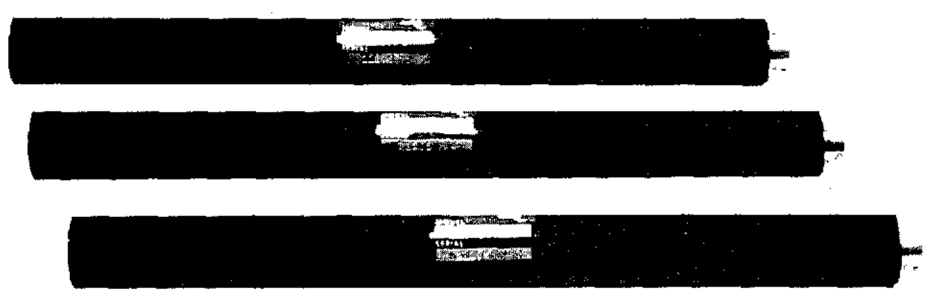
4.3.2 COMMUNICATIONS AND DATA HANDLING

The communications and data handling (C&DH) subsystem must satisfy the EOS requirements, Appendix C, Vol. 3, and be compatible with the operational requirements defined in the GSFC STDN Users Guide No. 101.1 and the GSFC Aerospace Data System Standards X-560-63-2.

The C&DH subsystem will provide the means of commanding the spacecraft and payload instruments via the uplink, provide onboard data required for ground monitoring of the spacecraft and payload status via downlink telemetry, and transpond ranging signals for ground tracking of the spacecraft. This subsystem shall be located in the Communications and Data Handling Module except for the antennas. The antenna locations will depend on radiation pattern coverage requirements. Other items such as signal conditioning and remote units which are elements of the C&DH subsystem are charged to the module in which they are located and serviced. The subsystem shall be functionally separate and operate independent of the wide-band communications subsystem.



a. Magnetic Unloading System Magnetometer



b. Magnetic Torquer Bars

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Fig. 4.3-6 Magnetometer and Magnetic Torquer Bars

4.3.2.1 COMMUNICATIONS GROUP

The Communications Group of the C&DH module provides telemetry, tracking and command link compatibility with STDN, Shuttle Orbiter, TDRS (option) and DOI (option). Table 4.3-4 tabulates the significant communication link requirements for these four interfaces. Only the interface with STDN at S-Band is fully defined at this time. Table 4.3-4 will be updated to include all of the detail interface requirements for the other links as soon as data are available.

COMMUNICATIONS GROUP CONFIGURATION ALTERNATIVES - Seven alternative communication configurations were derived. They vary in capability and complexity from the single thread configuration of Fig. 4.3-7a to the sophisticated multimode configurations of Fig. 4.3.7g. The basic parameters of these alternates are compared in Table 4.3-5. Some of the comparison data is currently not available for configurations 3, 4 and 5 since the evaluation of these configurations is still in process.

The primary difference between configurations 1 and 2 is that configuration 1 provides spherical antenna coverage on the uplink and hemispherical antenna coverage on the downlink, whereas configuration 2 provides spherical antenna coverage on both the uplink and downlink. The "A" versions of configurations 1 and 2 have dual redundant transponders.

In configuration 3, an improvement in uplink command reliability is achieved by combining the outputs of receiver/demodulator and selecting the best signal, or by cross-strapping the inputs of two demodulators. Results of preliminary analysis indicates that cross-strapping the demodulator inputs is the preferred approach.

Configuration 4 is configuration 2 plus a TDRS S-Band terminal. The terminal includes a S-Band transceiver package and a steerable antenna. Since wideband communications (refer to appendices to this book) will have an interface to the TDRS at Ku-band, a dual frequency S/Ku-Band steerable antenna is being considered to satisfy both the narrow band and wideband communications requirements. Portions of the TDRS S-Band equipment (i.e., receiver front end, transmitter) may be co-located with the steerable antenna to reduce RF transmission line losses while other portions (i.e., demodulator, baseband assembly) may be located in the C&DH module. The steerable S-Band antenna (7 to 11 ft. diameter requirement) will probably be located on a boom to minimize vehicle blockage problems.

Table 4.3-4 EOS Communications Link Requirements

Downlink Telemetry Parameter	Value	Communication Link STDN	TDRS*	SHUTTLE	DOI*
• FREQUENCY	TBD MHz ± .001% (2200 to 2300 MHz)	X	X	X	TBD
• NARROW BAND DATA RATE	SELECTABLE, 32 KBPS, 16 KBPS, 8 KBPS, 4 KBPS, 2 KBPS, 1 KBPS	X	X	X	X
• NARROW BAND MODULATION	SPLIT PHASE PCM/PM ON 1.024 MHz SUBCARRIER	X	X	X	X
• MEDIUM BAND DATA RATE	128 KBPS, 640 KBPS OPTIONAL	X	N/A	N/A	X
• MEDIUM BAND MODULATION	SPLIT PHASE PCM/PM ON CARRIER	X	N/A	N/A	X
• CCIR POWER FLUX DENSITY LIMITS IN ANY 4 KHz BAND	-144 $dBW/m^2/4KHz$	X	X	X	X
OVERHEAD S-BAND TRANSMISSION					
• LINK MARGIN	6dB MINIMUM	X	X	X	X
• GROUND ANTENNA SIZE	30 FT DISH	X	N/A	N/A	TBD
• GND SYSTEM NOISE TEMPERATURE	125° K	X	N/A	N/A	X
• BIT ERROR RATE	$\leq 10^{-5}$	X	X	X	X
• REQUIRED E/N_0	12dB	X	X	X	X
• MAXIMUM SLANT RANGE	3040 KM	X	TBD	TBD	TBD
• ATMOSPHERE LOSS	0.6dB	X	N/A	N/A	X
• POLARIZATION	RHCP	X	X	X	X
UPLINK COMMAND					
• FREQUENCY	TBD MHz (2025 TO 2120 MHz)	X	X	X	TBD
• COMMAND BITE RATE:	2000 bps	X	X	X	X
• COMMAND MODULATION:	PCM/PSK - Σ/FM/PM (USES 70 KHz SUBCARRIER)	X	X	X	X
• TRANS PWR:	10 KW	X	TBD	TBD	X
• TRANS. ANTENNA SIZE:	30 FT. DISH	X	TBD	TBD	TBD
• LINK MARGIN:	6dB MINIMUM	X	X	X	X
• BIT ERROR RATE:	$\leq 10^{-5}$	X	X	X	X
• REQUIRED E/N_0:	12dB	X	X	X	X
• MAXIMUM SLANT RANGE:	3040 KM	X	TBD	TBD	TBD
• ATMOSPHERE LOSS:	0.6 DB	X	N/A	N/A	X
• POLARIZATION:	RHCP	X	X	X	X
RANGING CHANNEL					
• FREQUENCY:	DOWNLINK $= \frac{240}{221} \times$ UPLINK	X	TBD	X	TBD
• RANGING MODULATION:	PM ON CARRIER	X	X	X	X
• RANGING TECHNIQUE:	HARMONIC TONES (500 KHz MAX) PN	X N/A	N/A X	X N/A	TBD N/A
• TURNAROUND RATIO:	221/240	X	TBD	X	TBD
• GROUP RELAY UNCERTAINTY:	$\leq$ 5 NANOSECONDS	X	TBD	TBD	TBD
• OTHER PARAMETERS:	REFER TO UPLINK & DOWNLINK REQUIREMENTS				

*OPTIONAL INTERFACES

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Table 4.3-5 Communications Configuration Comparison

PARAMETER	CONFIGURATION 1	1A	2*	2A	3	4	5
o S-BAND ANTENNA COVERAGE							
SPHERICAL UPLINK/DOWNLINK-1 ANT	X	X					
SPHERICAL UPLINK/DOWNLINK-2 ANT			X	X	X	X	X
o DUAL REDUNDANT TRANSPONDERS		X		X	X		
SIGNAL COMBINING OR CROSS-STRAPPING					X		
o TDRS INTERFACE						X	
o X-BAND DOWNLINK							X
o WEIGHT (LBS)	9.5	14.2	10.2	14.9	TBD	10.2+ TDRS	TBD
o POWER (WATTS)	12.0	14.5	12.0	14.5	TBD	TBD	TBD
o COST ($K)	283	390	274	381	TBD	TBD	TBD
o RISK	MIN	MIN	MIN	MIN	MOD	MAX	MIN
o SPACECRAFT INTEGRATION COMPLEXITY	MIN	MIN	MIN	MIN	MIN	MOST	MOD

* SELECTED CONFIGURATION

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The last configuration, 5, provides a downlink in the 8.0 to 8.4 GHz frequency band allocated for operational earth resource satellite programs. The uplink command would still be at S-Band. A downlink capability would be retained at S-Band in order to provide NASA with maximum command and control capability of the EOS spacecraft from all STDN ground stations.

4.3.2.2 SELECTED COMMUNICATIONS GROUP CONFIGURATION

Analysis of alternate configurations, summarized in Table 4.3-5, resulted in the selection of configuration 2, Fig. 4.3-7c. This configuration utilizes a single S-Band transponder with an integrated hybrid, coaxial switch and two diplexers, along with two broadband S-Band shaped-beam antennas. The hardware was selected from available candidate components shown in Table 4.3-6. The equipment list and cost allocation for the selected configuration is shown in Table 4.3-7. Selection of this configuration satisfies the functional and performance/design requirements shown in Table 4.3-4 for a STDN S-Band interface and is considered a low-risk design because it uses space proven off-the-shelf components.

4.3.2.3 DATA HANDLING GROUP

The Data Handling Group (DHG) must acquire, process, record, format and route data/commands from and to the appropriate EOS subsystem (Communications, ACS, Elec.

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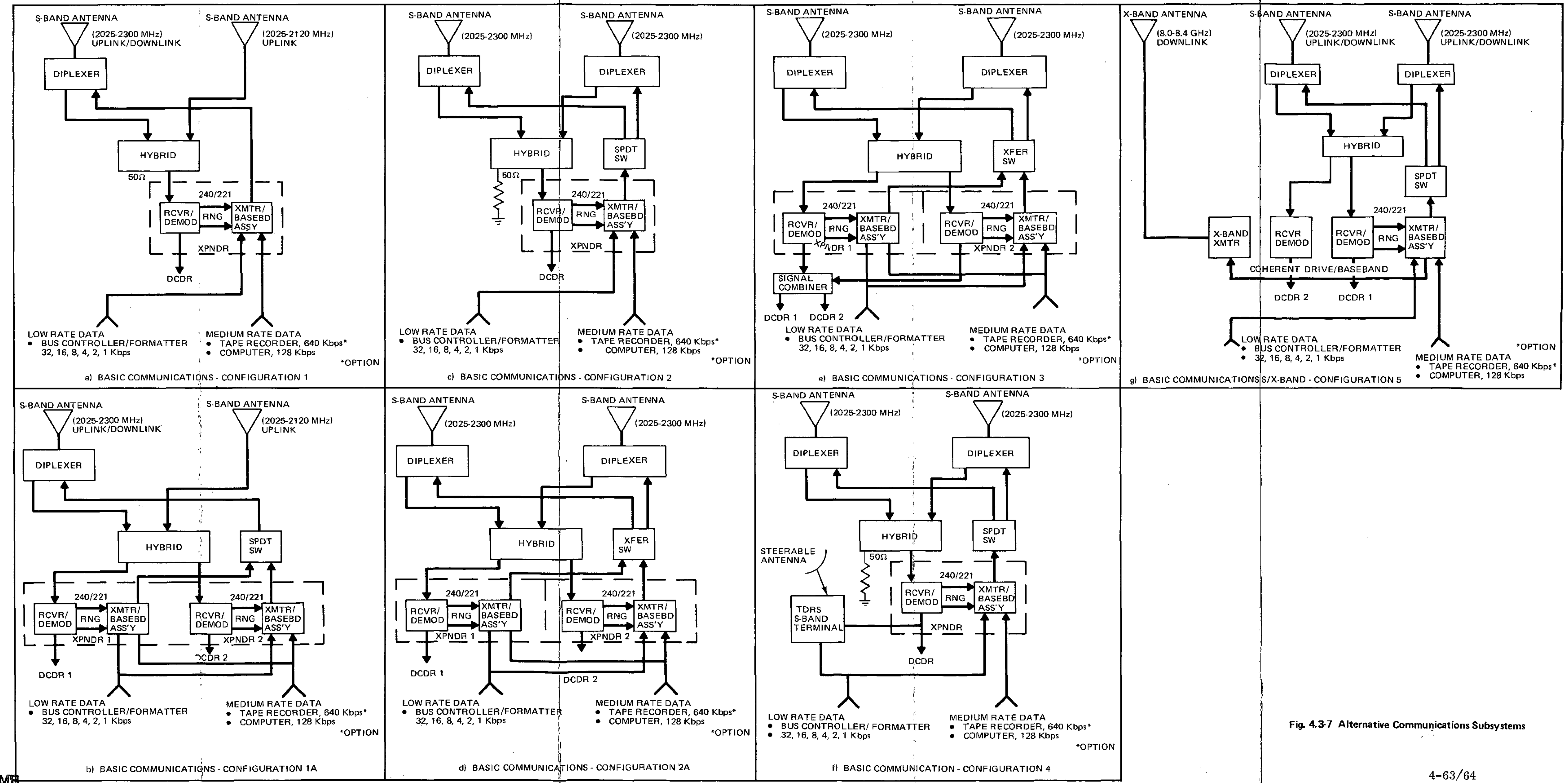

Fig. 4.3-7 Alternative Communications Subsystems

FOLDOUT FRAME

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FOLDOUT FRAME

FOLDOUT FRAME

Table 4.3-6. Communications Group Candidate Components

Integrated Electronics	Mfr.	Previous Program	Wt. (lbs)	Av Pwr (watts)	Vol (in³)	Cost ($K) N.R.	R	Total	Comments
S-BAND TRANSPONDER	MOTOROLA	JBS	12.5	5/14.5*	563	141	214	355	DUAL UNIT
	MOTOROLA	JBS(MOD)	7.8	2.5/12*	253	141	107	248	SINGLE UNIT
	MOTOROLA	ERTS	25.0	7/23*	624	115	175	290	DUAL UNIT
	CUBIC								
	CINCIN. ELECT.		REQUEST FOR QUOTES IN PROCESS						
	PHILCO FORD								
SEPARATE ELECTRONICS									
S-BAND TRANSMITTER/	TELEDYNE	ELMS	3.5	33.6	72	19	20	39	
BASEBAND	CONIC	LCRU	2.5	34.0	51	35	33	68	ELMS QUOTE
S-BAND RCVR/DEMOD	CINCIN. ELECT.	ELMS	3.6	5.0	93	40	34	74	
S-BAND HYBRID	WAVECOM	ELMS	1.0	–	8	6	2	8	
	SANDER ASSOC.	F-14	0.5	–	8	72	2	80	GPS QUOTE
S-BAND DIPLEXER	WAVECOM	LCRU	1.0	–	27	10	3	13	
S-BAND SWITCH	TRANSCO	ERTS	0.1	–	0.8	6	2	8	
ANTENNAS									
BROADBAND	GE	ELMS	1.2	–	50	33	8	41	ELMS
	RCA	VIKING	0.8	–	12	113	36	149	QUOTES
	GRUMMAN	F-14	1.5	–	40	49	9	58	
NARROWBAND	BALL BROS	ELMS	0.5	–	1.3	15	2	17	

* Rx/Rx + Tx

Table 4.3-7 Selected Communications Group Configuration

COMPONENT	QTY/VEH	WT. EA. (LBS)	VOL. EA. (IN^3)	PWR EA. (WATTS)	STATUS	SOURCE	COSTS NR	(PER VEH) R
S-BAND TRANSPONDER ASSEMBLY				12W(R_X&T_X) 2.5 W (R_X)	M	MOTOROLA/ JBS	$141K	$107K
• TRANSPONDER	1	7.8	253					
• DIPLEXERS	2	6.8	253					
• HYBRID								
• COAXIAL SWITCH								
TLM/CMD ANTENNAS	2	1.2	50	N/A	E	GE/ELMS	$10K	$16K
COAXIAL CABLE ASSEMBLIES	2	TBD	TBD	N/A	E	TWC/ELMS	–	$0.2K

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Pwr, Orbit Adjust and Transfer, etc.) and the support vehicle (e.g., Shuttle Orbiter, etc.). In addition, the group must perform the required attitude control computations issuing the necessary commands, receive commands from the ground and distribute or execute these in real time, or store them for delayed execution on a time or event basis.

Detailed DHG requirements and their origin are outlined in Table 4.3-8. The DHG shall be capable of transmitting to the Shuttle Orbiter or ground variable data rates of 32/16/8/4/2/1 KBPS and receiving 2 KBPS in commands. Commands uplinked from ground are 40 bits in length, 24 bits of which are defined by NASA as the computer data word, thereby requiring two locations in storage in any 16 to 18 bit word length computer.

GAC software sizing estimates for command storage, spacecraft control, systems monitoring, etc. define 23.3K eighteen bit words including margin as required for storage in the computers main memory. Resolution to 30 meters is required for MSS image processing. Twenty-four bit word length provides resolution to seven meters while still accommodating earth orbit dimensions with margin. Throughput requirements range from 6-13 KOPS (Kilo Operations per Second). The IRU Service routine is the main driver utilizing 3 KOPS.

The basic spacecraft's approximately 300 measurements and 200 commands are handled by five remote units (64 inputs and 64 outputs each) while two more remotes are dedicated to the instruments. The TM, HRPI, and MSS all require approximately 118 measurements, plus 48 discrete commands and 4 instruction words each. The MOMS is estimated to require 16 measurements and 16 discrete commands.

Recording requirements are driven by telemetry line data rates; maximum time EOS is out of ground contact is 5 to 7 hours (GAC estimates based on GAC mission trajectory analysis of EOS Sun Synchronous mission) and 11 minutes max that EOS is in ground contact following such a period.

Table 4.3-8. Data Handling Subsystem Requirements

ITEM	REQUIREMENT	ORIGIN
COMPUTER		
MAIN MEMORY (K WORDS)	23.3	GAC SOFTWARE SIZING ESTIMATE
WORD SIZE (BITS)	18-24	NASA, GAC SIZING ESTIMATE
THRUPUT (KOPS)	6-13	GAC SFW SIZING ESTIMATE
LANGUAGE	aASSEMBLY	GAC MEMORY EFFICIENCY
CLOCK STABILITY	±1 PART IN 10^6 PER DAY	NASA
TLM RATES (KBPS)	32/16/8/4/2/1	NASA GSFC & JSC ORBITER
CMD RATES (KBPS)	2, 2.4	NASA GSFC & JSC ORBITER
REMOTE UNITS (#/SPACECRAFT)	7	GAC MEASUREMENTS & COMMANDS SIZING
MEASUREMENTS	300	GAC MEASUREMENTS & COMMANDS SIZING
COMMANDS	200	GAC MEASUREMENTS & COMMANDS SIZING
CAUTION & WARNING FUNCTIONS	9-12	GAC SIZING (LAUNCH VEHICLE DEPENDENT)
TAPE RECORDER (OPTIONAL)		
CAPACITY (MBITS)	10^6	GAC
RECORD RATES (KBPS)	32/16/8/4/2/1	NASA
REPRODUCE RATE (KBPS)	640	NASA
RECORD TIME (MINUTES)	560	GAC

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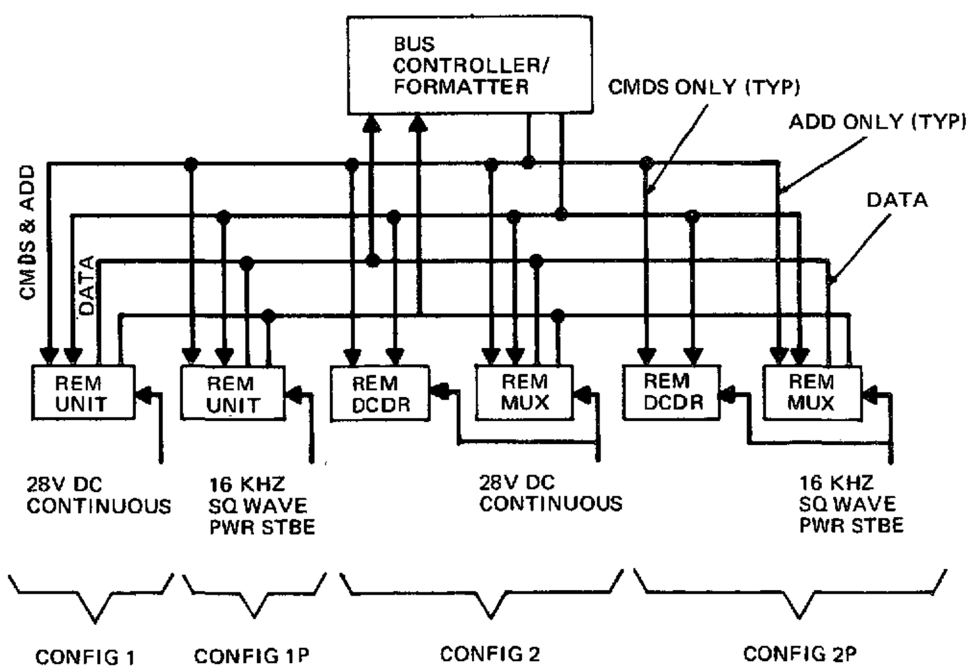
While EOS is attached to the Orbiter, the Orbiter crew must be alerted and have the capability of monitoring any EOS parameters which will indicate a potentially hazardous condition. Nine to twelve EOS caution and warning functions have been identified by GAC for EOS. Three Caution and Warning functions vary for alternate orbit transfer subsystems which vary as a function of the launch vehicle configuration.

DATA HANDLING ALTERNATE CONFIGURATIONS - The objective of this study is to obtain accurate costing for a data bus system suitable for EOS application. Data Bus system configuration alternatives are many. These include full duplex versus half duplex, separate command and address line versus common lines, data rates, formats, combined versus separate remotes, etc. In order to limit the scope of the study to output which would yield accurate costing, a baseline data bus system for EOS was selected. Alternative configurations within the baseline were defined and vendors requested to quote on the baseline and these alternatives. Vendors were also invited to quote on their own alternative configuration providing it met the overall operational parameters of the baseline system.

The NASA Standard Full Duplex System with commands and addresses sharing a common bus was selected and merged with the NASA EOS baseline equipment characteristics. The decision to incorporate the NASA Standard data bus features into the baseline system was made for the following reasons and assumptions. The non-recurring development costs for such a system are assumed not chargeable to the EOS program, thereby, significantly reducing a major cost element of the EOS Data Handling Group. Also, a review of its features showed a strong resemblance to the present system being developed for the Shuttle Orbiter. Orbiter commonality would also reduce program costs. In addition, the NASA Standard operating at 1 MBPS rate and using self-synching Manchester II Bi-Phase L code easily fulfills EOS requirements.

Since the full duplex system uses a common bus for both commands and addresses, it was determined that a single central unit (Controller/Formatter) which would control the bus issuing both addresses and commands would reduce system complexity. The alternative is to operate the bus under control of two separate and distinct units.

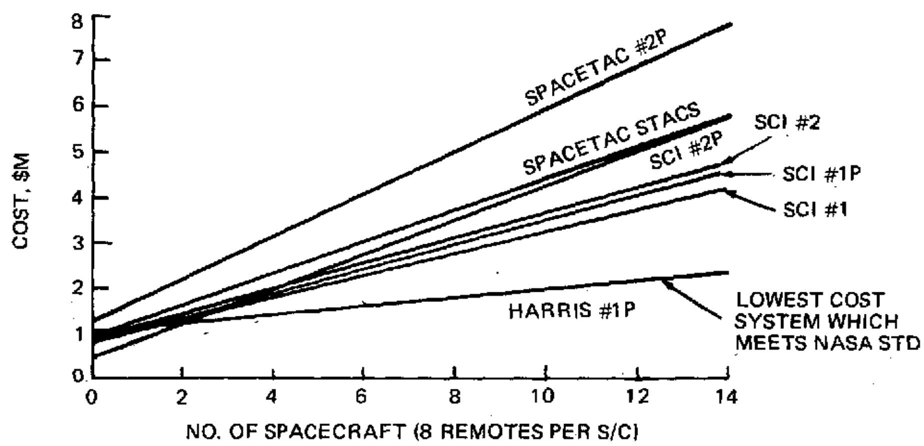
The selected baseline is shown in Figure 4.3-8. Configuration alternatives to the system are Configuration 1 which uses a remote unit that incorporates both a remote decoder and remote multiplexer (Mux). Configuration 1P is the same remote unit but power strobed with a 16 kHz square wave. Configuration 2 uses separate remote decoders and remote mux's while 2P is the 16 KHz square wave power strobed version of #2 (the NASA EOS baseline for remotes).



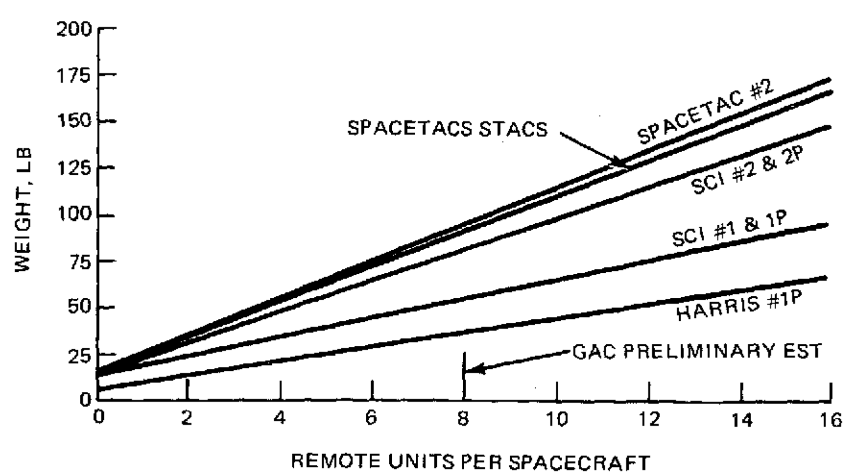
3-29 **Fig. 4.3-8 EOS Data Bus System Configurations**

Preliminary conclusions are drawn from three companies, Harris (Radiation), SCI and Spacetac, who have responded to date. These results are plotted in Figures 4.3-9 and 4.3-10. The Harris 1P configuration is the lowest-cost system for a program where two or more spacecraft are procured. (Figure 4.3-9). It is also the lowest weight system (Figure 4.3-10) and therefore is the most attractive candidate for EOS.

Assuming that this system becomes the NASA standard and its one million dollar non-recurring cost is not chargeable to the EOS program, then this system would be the prime choice due to its very low recurring unit cost.



3-30 **Fig. 4.3-9 Total Program Cost (Non Recurring + Recurring) of Candidate Data Bus Systems**



3-31 **Fig. 4.3-10 Weight of Candidate Data Bus Systems**

The SCI 2P configuration is the lowest-cost system for a one spacecraft program due to its lower non-recurring cost. Results of this study are not conclusive since not all of the solicited vendors (including the orbiter prime data bus contractor) have yet responded.

DHG AOP MEMORY ALTERNATIVES - The Advanced On-Board Processor (AOP) is available with three memory types: core, plated wire and CMOS (Complimentary Metal Oxide Semi-conductor). Core and plated wire are both considered to be acceptable memory types for EOS application while CMOS is conditionally acceptable.

CMOS RAMs (Random Access Memories) are volatile, requiring quiescent power at all times to retain their stored data. A power interrupt or loss (e.g. due to high peak load transients including fault loads, clearing shorts, power transfer from orbiter to EOS, battery explosion, etc.) could cause loss of all data in the RAM. If a spacecraft design without an analog backup is selected, then it may be desirable to design the computer with its own back-up battery which would provide power during any spacecraft power interrupts or shutdown. Another alternative is to store the entire contents of the RAM on an on-board tape recorder and, following a shutdown, a routine stored in a CMOS ROM (Read Only Memory) module could direct the recorder to reload the main memory.

The primary driver for memory selection is total program cost. Assuming the value of one watt of spacecraft power is 1K dollars, a plot of total program cost (including power costs) appears in Figure 4.3-11. As shown, selection of core memory for a single spacecraft requiring 24K memory saves 91K dollars over plated wire and 57K dollars over CMOS.

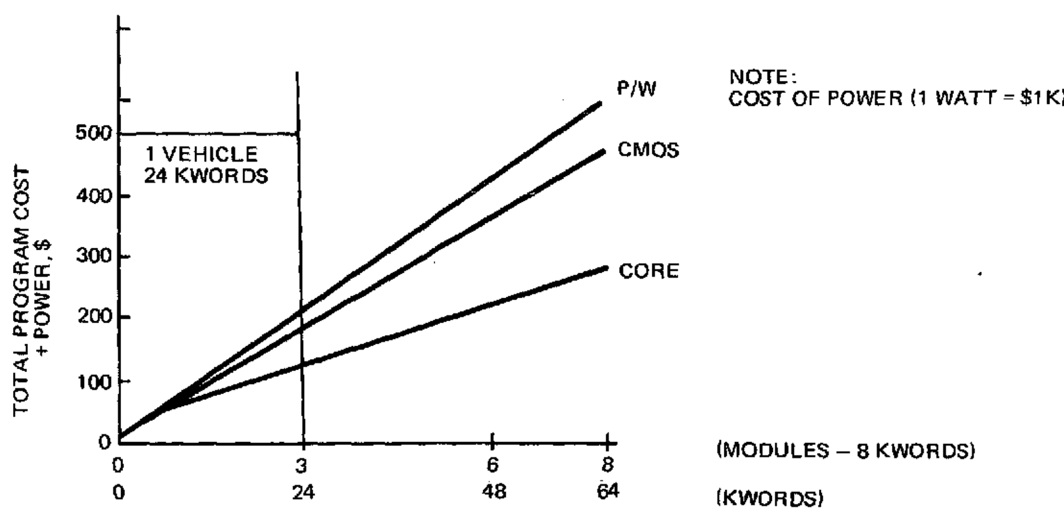


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Fig. 4.3-11 Computer Memory Type Total Program Cost

4.3.2.4 SELECTED DATA HANDLING CONFIGURATION

The baseline single thread DHG as depicted in Figure 4.3-12 is comprised of a 24k word Advanced Onboard Processor (AOP) with core memory, command decoder, bus controller/formatter unit, seven remote units (one located in the C&DH module, the remaining six distributed throughout the spacecraft), a 4.096 mHz central clock and signal conditioning units which condition Hi- and Lo-level signals to 0-5vdc, and also contain D/A (Digital to Analog) conversion and latching relays for implementation of commands.

The AOP computer using the Harris CMMA chips will be flown aboard ERTS-B. A space-qualified AOP minimizes non-recurring costs. Assuming AOP procurement efforts progress as planned, the AOP should be well proven prior to the first EOS flight, thereby minimizing program risk.

Using a standard Aerospace instruction mix of 80% shorts (adds) and 20% longs (multiplies) the AOP's throughput is computed to be 85KOPS, seven to eight times the current maximum requirement for EOS.

The AOP's capability to perform data compression is utilized on housekeeping data, thereby obviating the need for the optional tape recorder. This represents a savings of approximately 80K dollars, 8 watts and 14 pound per spacecraft. The AOP computes and/or stores each measurement's high, low, mean, mean variance and current value. Utilizing this technique, 150 measurements require 750 storage locations in main memory. It is executed via the 30 words software routine flow charted, see Figure 6.7-2.

The selected Harris full duplex data bus system, #1P configuration, has combined remote units which are power strobed with either 16 kHz square wave or 28 vdc. Remote units have dual receivers and transmitters which operate off the dual-redundant command/address busses and data reply busses respectively. Each unit has 64 input channels that can be used for analog, bilevel or serial digital signals as defined in the NASA EOS C&DH specification. Each unit also has 64 output channels for pulse commands plus 4 serial magnitude command outputs. Output levels are also as defined in the NASA C&DH specification. Remote units weigh 4 pounds each and draw 4 watts of power when "ON."

The controller/formatter also has dual receivers and transmitters which interface to the dual redundant busses. This unit can accept and interleave 50 commands/second from the command decoder with 62.5 commands/second from the AOP, and transmit these to the remote units. Telemetry output rates are command selectable at 32/16/8/4/2/1 KBPS and format consists of minor frames of 128 eight bit words.

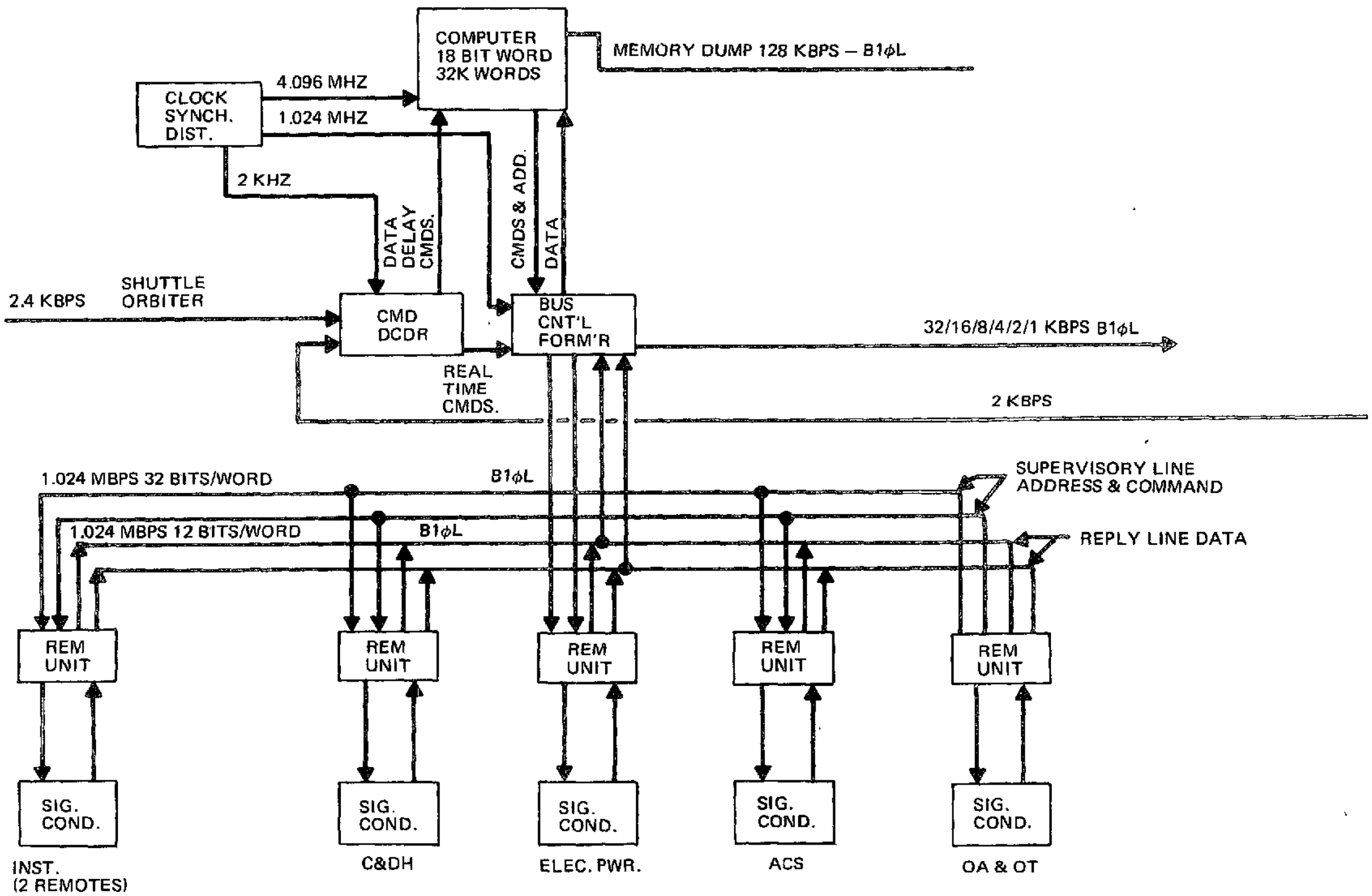

Fig. 4.3-12 Data Handling System

4.3.3 ELECTRICAL POWER

An electrical power subsystem (EPS) can be configured to meet the basic and expanded requirements of the EOS and follow-on missions.

The EPS will consist of a standardized power subsystem module and a mission-peculiar solar array. The power module must be capable of controlling, storing, distributing and monitoring the power derived from the solar array. Energy storage and control functions will be of a modular design to permit optimization of subsystem performance, weight, reliability and cost to specific mission requirements. Command and telemetry requirements will be compatible with remote command decoding and telemetry multiplexing equipment contained in the power module.

Table 4.3-9 defines known spacecraft/mission electrical power requirements. The basic spacecraft (exclusive of mission-peculiar payloads) is estimated to require approximately 300 watts of orbital average power. EOS instruments and other associated payload equipment can range from an average of 150 watts to over 350 watts. Including missions other than EOS could result in an average payload power of up to 500 watts. Therefore, the electrical power subsystem design load capability, based upon this tentative load analysis, should be in the range of 400 to 1000 watts orbital average.

Table 4.3-9. Summary Load Analysis

LOAD	NORMAL OPS ORBITAL AVG POWER (W)	SURVIVAL ORBITAL AVG POWER (W)
SPACECRAFT		
ACS MODULE	92	20
C&DH MODULE	80	63
POWER MODULE	44	44
THERMAL CONTROL (ALLOCATION)q	100	100
PNEUMATICS/OAM	NEGL.	NEGL.
TOTAL	316 W	227 W
INSTRUMENTS/PAYLOAD		
EOS	180-350[1]	–
OTHER (SMM, SEASAT, ETC)	110-500	

[1] THE SAR INSTRUMENT ACCOUNTS FOR 200 WATTS OF ORBITAL AVERAGE POWER. ACTUAL POWER WILL BE APPROXIMATELY 1.2 KW FOR APPROX. 10-20 MINUTES PER ORBIT.

TOTAL EOS POWER REQUIRED;

ORB AVG 500 – 875 W

MAX. PEAK (DELTA) 1.2 KW

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The peak load requirement was defined by NASA in the EPS subsystem specification to be 5.6 kw for 10 minutes, either day or night. This requirement, which is assumed to be contributable to the peak power required by a Synthetic Aperature Radar, was reviewed and now appears to be somewhat high. Present estimates of the SAR delta-peak power required are in the order of 1.3 kw over the normal spacecraft load. Therefore, maximum peak loads for the EOS are not expected to exceed approximately 2 kw.

These power requirements, as well as other requirements which have a major impact on the electrical power subsystem design, configuration, performance weight and cost, arc summarized in Table 4.3-10, Major Driving Requirements. The general subsystem design requirements common to all subsystems have not been included in the table.

4.3.3.1 EPS CANDIDATE CONFIGURATIONS

General forms of EPS configurations were evaluated with respect to the basic and expanded requirements of the EOS and follow-on missions. One of the key evaluation criteria was flexibility to optimize the configuration to mission-peculiar requirements and options without cost penalties and still maintain a high degree of standardization.

Table 4.3-11 summarizes some of the key advantages and disadvantages of various alternative configurations. Any of these configurations could be designed to satisfy the EOS as well as follow-on mission requirements, with the optimum configuration being dependant upon specific mission and spacecraft requirements.

Table 4.3-10. Major EPS Driving Requirements

REQMT/RANGE		INFLUENCE
MISSION		
ORBIT:	a) 200 TO 900 N MI, 0-90° INCLINATION INCLUDING SUN-SYNCH @ 9:30 AM TO 12 NOON b) GEOSYNCHRONOUS	AFFECTS DARK & LIGHT DURATIONS AND RADIATION ENVIRONMENT WHICH IN TURN AFFECT EPS SIZING, WEIGHT AND COST.
LIFE:	a) OPERATIONAL 2 TO 5 YEARS b) DESIGN 5 YEARS	AFFECTS BATTERY LIFE (DEPTH-OF-DISCHARGE) AND SOLAR ARRAY DEGRADATION – DIRECT INFLUENCE ON SIZE, WEIGHT AND COST OF EPS
POINTING:	a) EARTH-POINTING b) INERTIAL POINTING	AFFECTS SOLAR ARRAY CONFIGURATION, ORIENTATION AND SIZE
SPACECRAFT		
POWER:	a) ORBITAL AVERAGE 400 TO 1000 WATTS b) PEAK (DELTA) 1.3 KW, FOR 10 MINUTES	DIRECT INFLUENCE ON SIZE, WEIGHT COST OF OVERALL EPS
PRIME BUS VOLTAGE:	28 ± 7 BDC	AFFECTS EPS CONFIGURATION BY ELIMINATING NEED FOR PRIME POWER CONDITIONING

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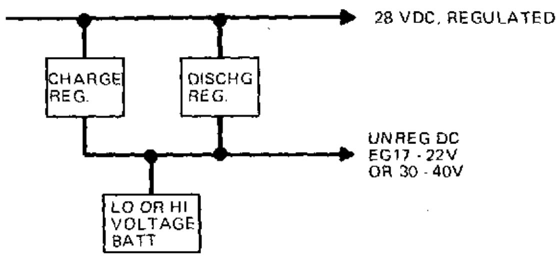
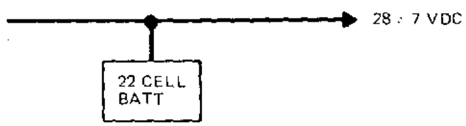
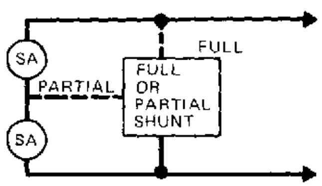
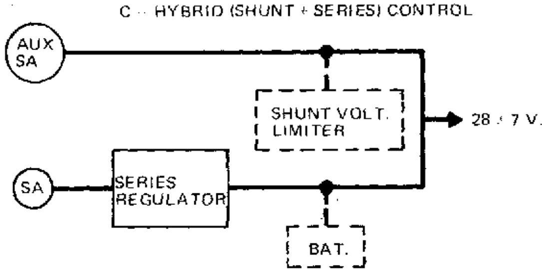
Table 4.3-11 Comparison of Alternative EPS Configuration

EPS SYSTEM CONFIG. ALTERNATIVES	ADVANTAGES	DISADVANTAGES
I. BATTERY CHARGE/DISCHARGE CONTROL A – CHG/DISCH REGULATION 	• PRIME BUS VOLTAGE REGULATION GENERALLY BETTER THAN DIRECT TRANSFER CONTROL • THE SERIES CHARGE REGULATOR NEED PASS ONLY THE BATTERY CHARGE POWER.	• DISCHARGE REGULATOR INCREASES BATTERY DISCHARGE ENERGY REQUIRED • PEAK LOADS SHOULD OPERATE AT DIFFERENT THAN "STANDARD" BUS VOLTAGE • SOURCE IMPEDANCE AND NOISE GENERALLY HIGHER THAN WITH DIRECT TRANSFER.
B – DIRECT BATTERY ENERGY TRANSFER 	• MINIMUM BATTERY ENERGY REQUIRED • MINIMUM SOURCE IMPEDANCE • PEAK LOADS OPERATE FROM STANDARD BUS VOLTAGE	• VOLTAGE REGULATION NOT AS GOOD AS WITH CHG/DISCHG REGULATOR.
II. SOLAR ARRAY CONTROL A – TOTAL SHUNT CONTROL 	• MAXIMUM POWER TRANSFER OF SOLAR ARRAY POWER OUTPUT TO LOAD AND BATTERY.	• REQUIRES CLOSE MATCH OF SOLAR ARRAY CHARACTERISTICS AND SYSTEM VOLTAGE FOR EFFICIENT SOLAR ARRAY POWER UTILIZATION. – SOLAR ARRAY FLEXIBILITY LIMITED IN ORDER TO MAINTAIN HIGH OVERALL SYSTEM EFFICIENCY • EXCESS SOLAR ARRAY POWER DISSIPATED IN S/C
B TOTAL SERIES CONTROL 	• SOLAR ARRAY CHARACTERISTICS/SYSTEM MISMATCH LESS CRITICAL • CAN BE MADE TO TRACK THE MAX. POWER CAPABILITY OF SOLAR ARRAY. • EXCESS SOLAR ARRAY POWER DISSIPATED ON-ARRAY.	• ALL S/C POWER (LOAD + BATTERY CHARGE) PASSES THROUGH SERIES REGULATOR. • SYSTEM LOAD CAPABILITY MORE DIRECTLY AFFECTED BY POWER HANDLING CAPABILITY OF SERIES REGULATOR.
C – HYBRID (SHUNT + SERIES) CONTROL 	• PROVIDES MAXIMUM FLEXIBILITY IN: - SYSTEM LOAD CAPABILITY SOLAR/ARRAY DESIGN/CONFIGURATION - OVERALL SYSTEM OPTIMIZATION • BY CHANGING RATIO OF AUX. SA TO MAIN SA, SYSTEM CAN BECOME A TOTAL SERIES SYSTEM (NO AUX. SA) TO TOTAL SHUNT SYSTEM WITH DEDICATED LOAD & BATTERY RECHARGE SOLAR ARRAY SEGMENTS.	

4.3.3.2 SELECTED EPS CONFIGURATION

The preferred EPS configuration for EOS is the hybrid system where both series and shunt (direct-energy-transfer) solar array control and direct battery energy transfer is utilized. This system configuration is a modularized EPS that is custom-designed to mission and spacecraft requirements using standard components. It is basically the same as that used for OAO, and is shown at a functional block level in Fig. 4.3-13. Options which are available include:

- Supplying none, part or all of the spacecraft load with a dedicated mission-peculiar auxiliary solar array that is operated in the direct-energy transfer mode. Control of this portion of the solar array can be with inherent battery voltage limiting (with appropriate voltage clamp circuits) with on-off control of segments of the auxiliary array, or on-array voltage limiting with zener diodes.

- A series regulator that can efficiently support the entire spacecraft and battery charge power requirements or down to just battery recharge.

- Capability to max-power-track the solar array or operate in direct-energy-transfer mode for initial battery charging.

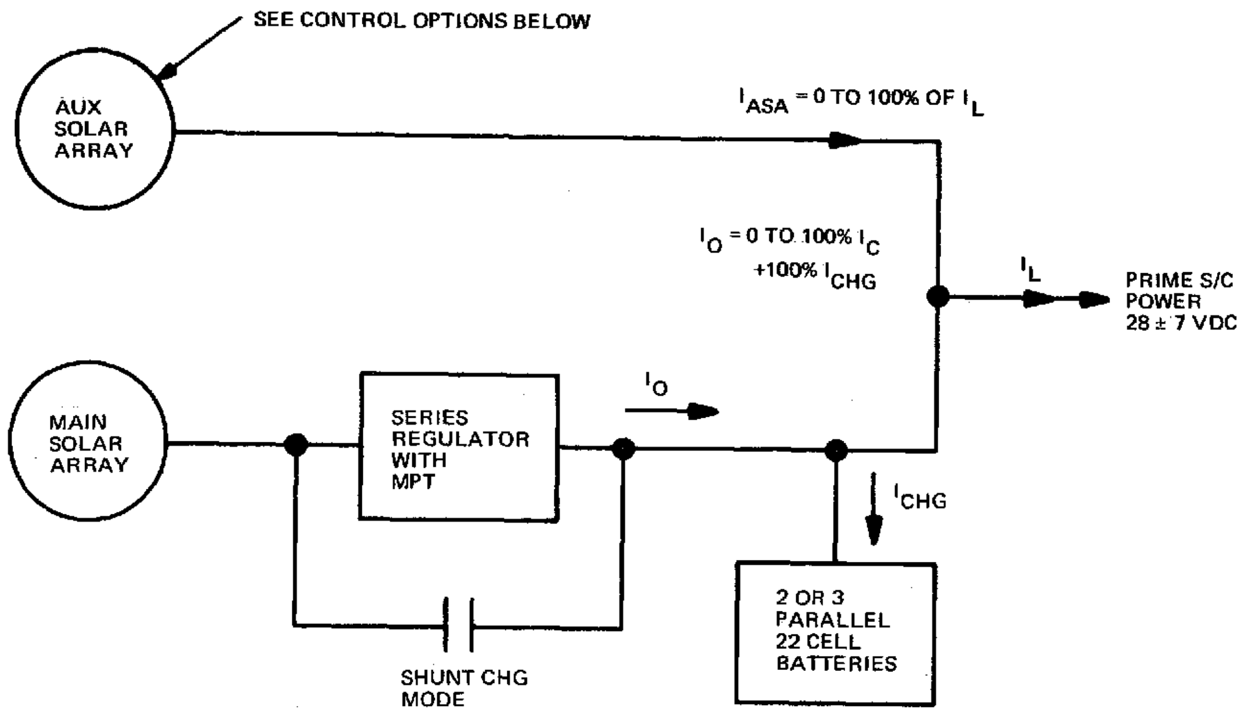
- Flexibility to choose array control that minimizes solar array cost. Existing and/or fixed solar arrays which have mismatch between array characteristics and system can be used efficiently with series regulation.

- Option of using the 20 A.H. or 36 A.H. batteries thereby satisfying 40 to approx. 120 A.H. capacity option requirement with only 2 or 3 batteries.

POWER MODULE ALTERNATIVES - The basic functions included in the power module are:

- Solar array control
- Energy storage control
- Energy storage
- Interface control
- Command, telemetry and monitoring

The major EPS functional requirements of energy storage and control and solar array control can all be implemented with existing or slightly modified equipment with little or no risk in developing new equipment. A case in point is the demonstration model EPS fabricated by NASA. This baseline system satisfies the basic EOS requirements for a modular, multi-mission EPS. Therefore, the major thrust of the Grumman EPS design-cost trades was to identify cost effective improvements to the basic NASA configuration.



AUX SOLAR ARRAY CONTROL OPTIONS:

1 NONE – ARRAY IS SIZED SUCH THAT IS CAN NEVER SUPPLY THE S/C LOAD – BATTERY VOLTAGE ESTABLISHES OPERATING POINT

2 NONE – WITH ON-OFF CONTROL – SAME AS 1 EXCEPT SECTIONS OF AUX. ARRAY ARE SHORTED OUT WITH CONTRACTORS AND ENABLED WHEN S/C LOAD CONDITIONS ALLOW.

3 ON-ARRAY VOLTAGE LIMITING – ARRAY CAN BE SIZED TO SUPPLY THE ENTIRE S/C LOAD IF ON-ARRAY VOLTAGE LIMITING WITH ZENER DIODES IS ADDED.

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Fig. 4.3-13 Selected EPS Configuration

Two subsystem functions were considered as likely candidates for improving the cost/performance characteristics of the demonstration power module.

- Battery alternatives (See Table 4.3-12)
- Solar array/battery control (See Table 4.3-13)

SOLAR ARRAY ALTERNATIVES - One of the basic requirements imposed on the solar array is that it must be compatible with the power module configuration and capabilities. The power module solar array/battery charge equipment was selected, among other reasons, to offer flexibility and latitude in defining a solar array that is optimized to particular mission requirements. The alternatives are shown in Table 4.3-14.

Table 4.3-12 EOS 22-Cell, Ni-Cd Battery Alternatives

BATTERY	MFG.	RATED CAPACITY AH	WEIGHT PER CELL (LBS)	BATTERY WEIGHT (LBS)	WEIGHT PER A-H (LB/AH)	DIMENSIONS (INCHES) L W H	RECURRING COST	$/AH
OAO, 3 20 AH	GULTON	3 20 AH	2.1	164	2.73	2" (18-5/8 X 11 X 10)	85K	1.42K
MODIFIED OAO-SINGLE 20AH	GULTON	20 AH	2.1	54.7	2.73	12.6 X 11 X 10	30K	1.50K
MODIFIED ELMS	GULTON	20 AH	2.1	53.1	2.66	12.6 X 7.4 X 7.4	20.6K	1.03K
MODIFIED SAR 8022-5	EAGLE PICHER	20 AH	2.3	65.8	3.29	20.5 X 8.5 X 6.5	13K	0.65K
MODIFIED NATO III	E-P	20 AH	1.28	32.4	1.62	12.4 X 7.5 X 5.3	18.2K	0.91K
MODIFIED SAR 8055-19	E-P	36 AH	2.80	89.9	2.50	19.8 X 8.5 X 7.0	17K	0.47K
SAR 8054	E-P	45 AH	3.60	103	2.29	22.3 X 9.0 X 8.2	18K	0.4K

3-90

Table 4.3-13 Alternative Solar Array/Battery Control Comparison

NAME	HISTORY/USAGE	KEY FEATURES	SIZE/VOLUME	WEIGHT	NON-RECURRING COST
POWER CONTROL POWER REGULATOR UNITS	FLOWN ON OAO	• SEPARATE REGULATOR PACKAGE (PRU) AND CONTROL PACKAGE (PCU) • PCU ALSO INCLUDES MISC. BATTERY CONTROL & SWITCHING FUNCTIONS • PRU & PCU USES STANDBY UNIT REDUNDANCY • OUTPUT POWER CAPABILITY 2350 WATTS	EACH UNIT 21-5/8 X 10-5/8 X 4 TOTAL 1840 IN^3	PCU 34.5 LBS PRU 40.5 LBS TOTAL 75 LBS	203K
MODIFIED ELMS BATTERY CHARGER	1 ORIGINALLY USED ON SKYLAB – OWS 2 MODIFIED FOR UPCOMING FLIGHT ON USAF-ELMS 3 EOS CHARGER TO BE MODIFIED VERSION OF ELMS	• SELF CONTAINED UNIT • MODULAR CONSTRUCTION • REDUNDANT MAX. POWER TRACKERS (CLOSED LOOP, ACCURATE TO > 97%) • INCLUDES VOLTAGES FROM 32 TO 125V (LOWER VOLTAGE LIMIT ONE OF MODS TO ELMS UNIT) • INCORPORATES UP TO 5 POWER REGULATOR MODULES WHICH, ARE PARALLEL OPERATED, # OF MODULES CAN BE OPTIMIZED TO MAINTAIN HIGH EFFICIENCY • CURRENT LIMITED @ 15A/ POWER MODULE • POWER OUTPUT CAPABILITY, MAX 2350 WATTS	10 X 10 X 7 (700 IN^3)	24 LBS	ELMS-55.4K EOS – 65K* *INCLUDES PERCENTAGE OF DISTRIBUTION UNIT COST REQUIRED BY CHARGER

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Table 4.3-14 Typical Solar Array Candidates Suitable for EOS

SOLAR ARRAY TYPE/DESCRIPTION	SIZE	WEIGHT	POWER	RECURRING COST	REMARKS
REQ'RD FLAT PANEL – XEROX P-95 0.300 ALUM HONEYCOMB WITH ALUM FACE SHEETS	17 X 66 IN 7.8 FT^2	7.8 LBS (1.0 LB/FT^2)	92.5 WATTS @ 28°C (11.9 W/FT^2)	$23K $2.95 K/FT^2	
REQ'D FLAT PANEL – XEROX TIMATION IIIA (ALUM HONEYCOMB)	54" X 21.75" (8.2 FT^2)	17.2 LBS (2.1 LBS/FT^2)	96 W @ 28°C (12 W/FT^2)	37K $4.5K/FT^2	
SOLAR CELL MODULES – SPECTROLAB ATM (ALUM HONEYCOMB WITH ALUM FACE SHEETS)	20" X 24.63" (3.4 FT^2)	EST. 5 LBS ~1.5 LB/FT^2	35 W @ 28°C (10 W/FT^2)	APPROX $10K $3K/FT^2	NATURAL FREQ 0.25 Hz
FLEXIBLE ROLL-UP SOLAR ARRAY-HUGHES FRUSA LAMINATED KAPTON-H FILM AND FIBERGLASS SUBSTRATES (FLOWN ON USAF MISSION, 1971)	DRUM 5-1/2 FT X 8" DIA. ARRAY - TWO BLANKETS EACH 5-1/2 FT X 16 FT (166 FT^2)	70 LBS DRUM ASSY 36 LBS BLANKET 34 LBS 0.205 LB/FT^2 + DRUM	1600W (~ 10 W/FT^2)	$1.2M	
FLEXIBLE ROLL-UP SOLAR ARRAY – GE (RA250)	DRUM 8.2' X 8" DIA. ARRAY 8.2' X 34 FT	82 LBS DRUM 36 LBS BLANKET 46 LBS (.19 LBS/FT^2 + DRUM)	(2500 WATTS) 10 W/FT^2)	NOT AVAILABLE	NATURAL FREQ. LESS THAN 0.1 Hz

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General alternatives that must be considered in defining a spacecraft solar array include:

- Rigid versus flexible
- Fixed versus oriented
- Continuous versus limited rotation

The optimum rotation selection is a continuous drive system compatible with the sensitivity of the Attitude Control System. The major determinant for this choice is the necessity to minimize resultant disturbance torques created by periodic solar array stops, starts, and reversals.

4.3.3.3 SELECTED EPS COMPONENTS

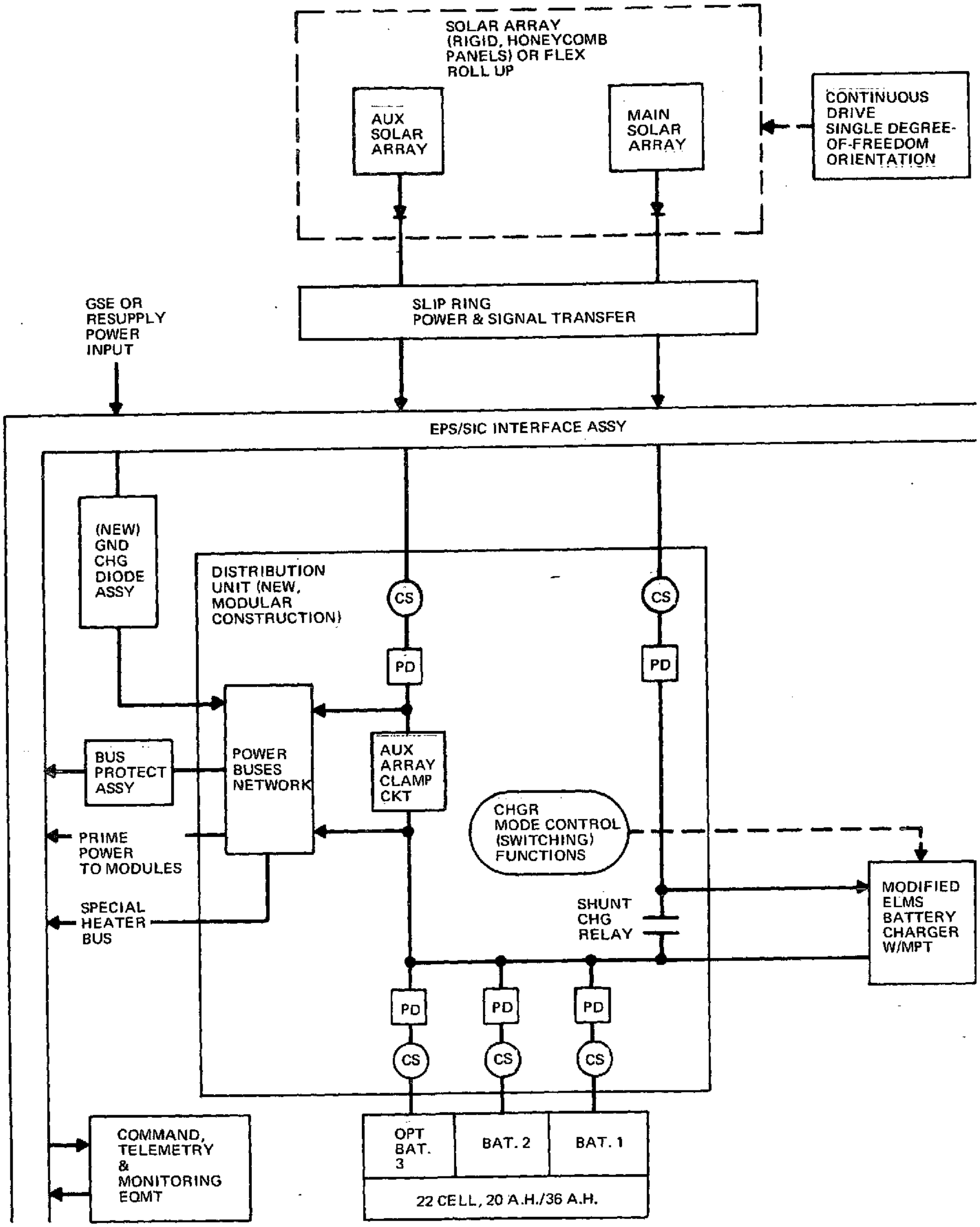
A detailed, functional/component block diagram of the selected EPS is shown in Fig. 4.3-14. A summary of selected components are identified in Table 4.3-15.

4.3.4 PROPULSION

The requirements having significant influence on the design of the propulsion subsystems are:

- Orbit Adjust
  - Launch vehicle (L/V) injection errors
  - Orbital decay
- Reaction Control
  - Initial stabilization & restabilization
  - Wheel unloading
- Orbit Transfer
  - Circularization
  - Deorbit

Table 4.3-16 shows the anticipated impulse requirements and fluid quantities. Note that modularity will influence the design of each of the subsystems.



3-35

Fig. 4.3-14 EPS Functional Component Block Diagram

Table 4.3-15 Selected EPS Components

POWER MODULE		RECURRING COST, $K	WEIGHT, LB	
BATTERY 2-22 CELL, 20 AH		37	64	MODIFIED NATO III
BATTERY CHARGER		55	24	MODIFIED ELMS
CENTRAL POWER CONT. UNIT		50	23	NEW – GRUMMAN
SIGNAL COND. ASSY		25	10	
GRD CHG DIODE ASSY (1)		5	7	NEW – GRUMMAN
BUS PROTECT. ASSY		10	5	NEW – GRUMMAN
S/C INTERFACE ASSY		10	12	NEW – GRUMMAN
BUS ASSY (3)	}		2	NEW – GRUMMAN
CONNECTORS		4	4	NEW – GRUMMAN
WIRING & MISC			18	NEW – GRUMMAN
REMOTE DECODER				
DUAL REMOTE MUX		10	2	NEW
POWER MODULE	TOTAL	206	171*	
SOLAR ARRAY DRIVE		52	25	BENDIX, MODIFIED ELMS
SOLAR ARRAY 125 FT^2 (16 PANELS)		390	132	XEROX, P95 RIGID PANELS
	TOTAL	648	332	

*EXCLUDES WEIGHT OF:
- POWER MODULE STRUCTURAL FRAME
- BATTERY HEAT SINKS, AND LOUVERS

3-93

The propellant required to correct the L/V injection errors represents 97% of the total translational propellant on board the spacecraft. The propellant required to perform the orbit-keep function varies with altitude, increasing at the lower altitudes. For the orbit selected for our spacecraft, the orbit-keep propellant represents 3% of the translational propellant.

Vehicle stabilization and restabilization have a small impact on the total RCS propellant loading. However, the need for vehicle stabilization initially, and during injection error firings, establishes the 1.0 lb thrust level. The burn times using smaller thrusters (e.g., 0.1 lb thrusters) would be excessive.

Wheel unloading requires approximately 73% of the RCS (rotational) propellant. The quantity of wheel unloading propellant is based on performing 20% of the total unloading using reaction jets. The requirement for very low impulse bits for unloading established the need for low thrust level thrusters on the order of 0.05 to 0.1 lb of thrust. Analysis

Table 4.3-16 Anticipated Impulse Requirements and Fluid Quantities

ORBIT ADJUST/RCS REQUIREMENTS

MISSION PHASE	IMPULSE (LB-SEC)					
	TRANSLATION			ROTATION		
	X	Y	Z	R	P	Y
INITIAL STABILIZATION				7	14	14
CORRECT INJECTION ERROR	4070					
CONTROL DURING CORRECTION FOR INJECTION ERROR				0.5	21	21
STABILIZE AFTER SOLAR ARRAY DEPLOYMENT				1	2	2
ORBIT KEEP	140					
GRAVITY-GRADIENT, JETS				105	236	0
TOTALS	4210			113.5	273	37
CONTINGENCY 10%	421			11	27	4
OVERALL TOTALS	4631			125	300	41
TOTAL 5097						

FLUID QUANTITY

QUANTITY (LB)	TRANSLATION	ROTATION
N_2H_4	20	2.9
N_2	76.7	11.8

ORBIT TRANSFER REQUIREMENTS

DELTA VEHICLE:	WEIGHT-2450#, ORBIT – 493 NM (RESUPPLY)
IMPULSE	85000#SEC
N_2H_4	370#
SRM PROP.	308#
TITAN VEHICLE:	WEIGHT-5100#, ORBIT – 493 NM (RESUPPLY)
IMPULSE	347000#SEC
N_2H_4	1110#
SRM PROP.	1224#

3-94

showed that the minimum impulse bit (MIB) capability of existing 0.1 lb thrusters (0.002 lb-sec) is acceptable for wheel unloading. The 0.1 lb thruster was, therefore, selected.

The Shuttle payload capability as defined by NASA-JSC establishes the requirement for an orbit transfer subsystem (OTS) or kick stage when the operational orbit exceeds approximately 400 nm. Our studies selected an operational altitude of 366 nm, eliminating the need for the OTS. However, propellant loading to transfer to and from a 493 nm orbit was established. SRM's, a N_2H_4 fueled system and a bipropellant system were considered.

The requirement for modularity and, potentially, resupply, results in the propulsion subsystems being installed in a separate structure on the aft end of the spacecraft. The modular approach provides several advantages.

- Mounting of OAS thrusters provides desired thrusting along vehicle flight path
- RCS thrusters easily oriented to provide pitch, yaw and roll control
- Eliminates the need for fluid interfaces between main spacecraft structure and thruster pads
- Minimizes possibility of impingement or interaction of thruster exhaust plumes with solar array or instruments

The main disadvantage of the modular design is that pitch and yaw firings result in small translational movements of the vehicle in addition to the desired rotation. This may not produce significant orbit decay in view of the very low RCS usage expected.

4.3.4.1 OAS CONFIGURATION TRADEOFF

Two alternatives, one using hydrazine (N_2H_4) and the other gaseous nitrogen (GN_2), were considered to fulfill the OAS function. The results of the trade study are shown in Table 4.3-16. While the GN_2 system provides a less complex and slightly lower cost OAS, it is a much heavier system. Since weight is a major consideration in the Delta 2910 spacecraft configuration, the lighter weight N_2H_4 system was selected. It should be noted that the weights shown include only component and propellant weights. The structural weight penalty associated with the GN_2 system is not included.

The selected orbit adjust subsystem is a hydrazine fueled system utilizing four 5 lb thrusters and operating in a blow-down mode. The equipment is mounted in a module, mounted on the aft end of the spacecraft as shown in Fig. 4.3-15.

Adding redundancy to the subsystem to provide fail-safe operation requires the addition of two latching solenoid valves and a second solenoid/seat assembly to each of the thruster valves, as shown schematically in Fig. 4.3-16.

CONCLUSION - N_2H_4 orbit adjust saves 140.1 lb at a cost penalty of $27K

COST OF REDUNDANCY

Weight penalty = 3.22 lb. Cost penalty = $11K

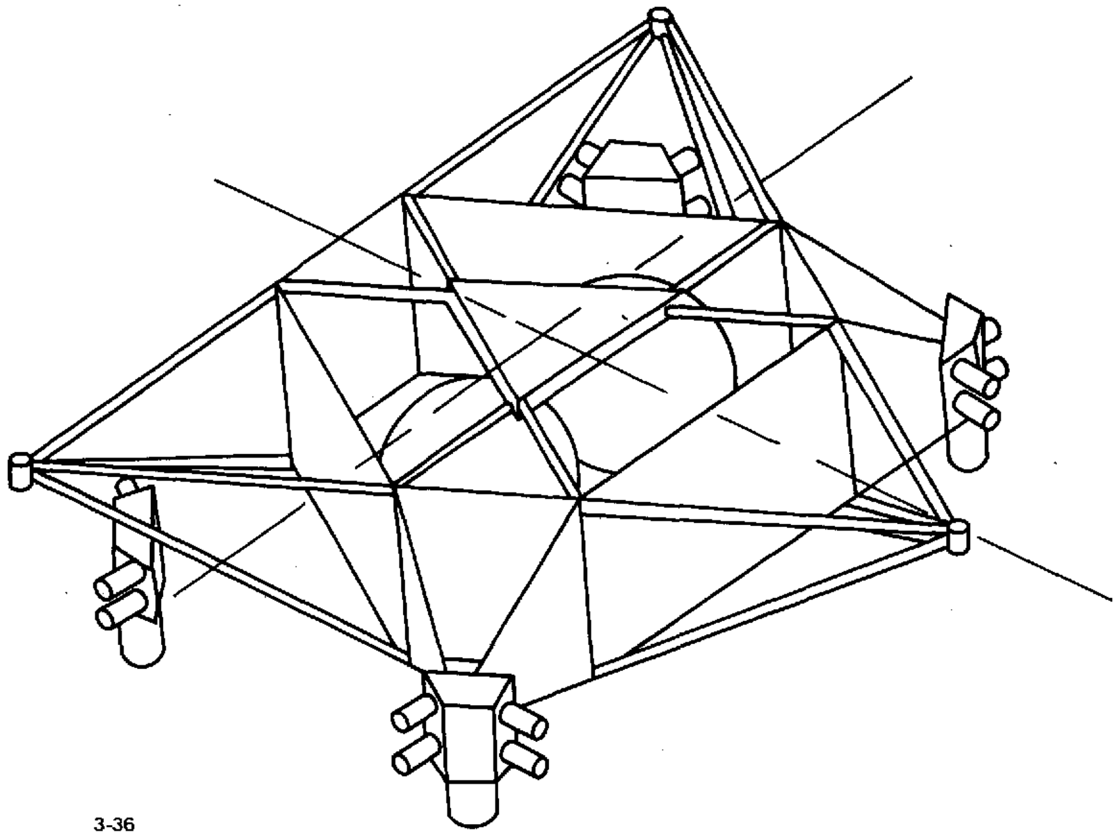


Fig. 4.3-15 OAS Module Mounting

4.3.4.2 RCS CONFIGURATION TRADEOFF

Two alternatives were considered to fulfill the reaction control (pneumatics) subsystem function. The first of these assumed the use of GN_2 as the propellant. The GN_2 system is a simple design carrying 11.8 lb of GN_2, with the capability to provide initial stabilization and restabilization of the vehicle, as well as its allotted wheel unloading requirement. The logical alternative to using GN_2 was the use of hydrazine as the propellant. Since the vehicle is already carrying a hydrazine-fueled OAS, it follows that combining the reaction control subsystem with the OAS should be considered.

In order to make a fair comparison, the combined GN_2 reaction control and N_2H_4 subsystem weights and costs were compared to the all-N_2H_4 subsystem. The results of the trade study are shown in Fig. 4.3-17.

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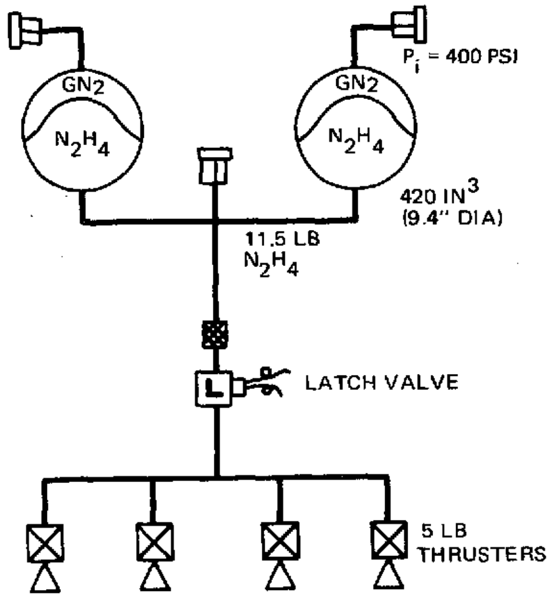
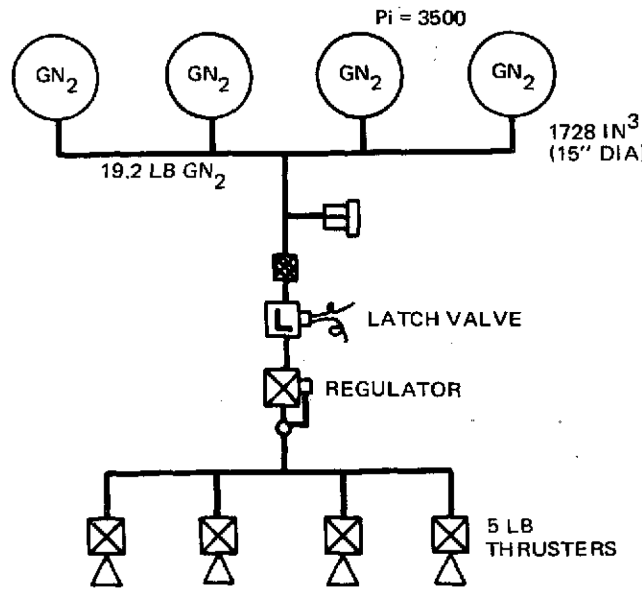
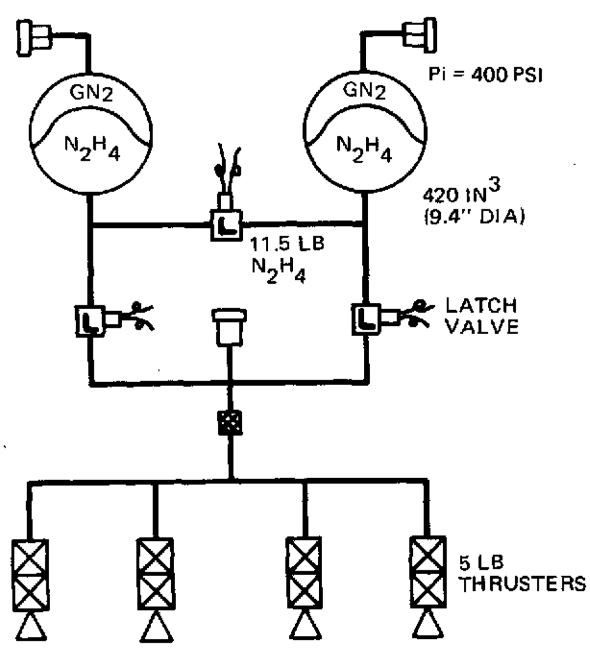


HYDRAZINE
ORBIT ADJUST SUBSYSTEM
(SINGLE THREAD)



NITROGEN
ORBIT ADJUST SUBSYSTEM
(SINGLE THREAD)

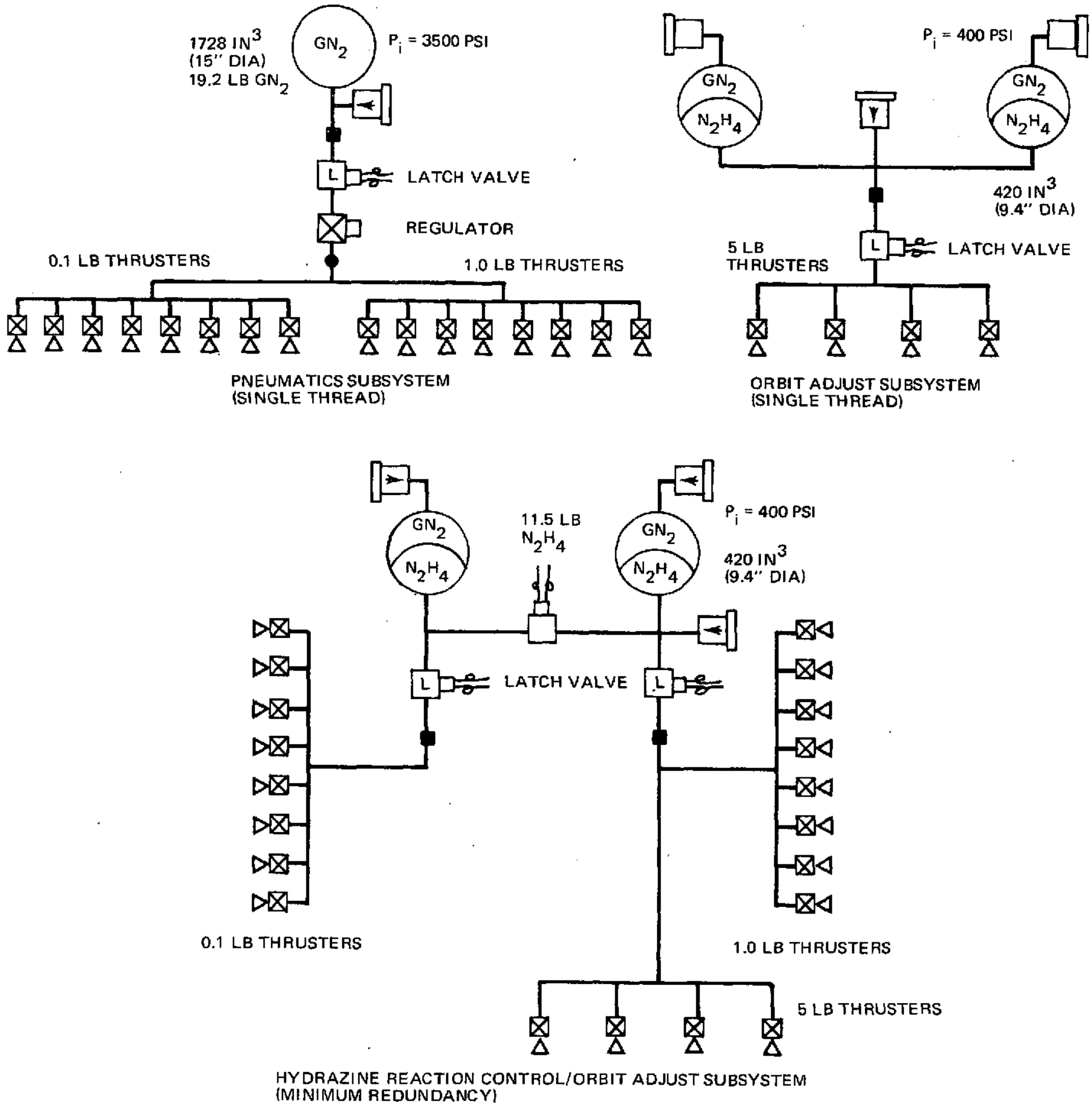
N_2H_4 VS GN_2 ORBIT ADJUST SUBSYSTEM			
	N_2H_4 SYSTEM	GN_2 SYSTEM	DELTA (N_2H_4 VS. GN_2)
WEIGHT, LB	30.2	170.3	-140.1
COST, $K	148.0	121.0	+27



a) REDUNDANT SYSTEMS

3-37

Fig. 4.3-16 Orbit-Adjust Subsystems



GN_2 Vs. N_2H_4 REACTION CONTROL SUBSYSTEM			
	COMBINED GN_2/N_2H_4	ALL N_2H_4	DELTA (COMBINED VS. ALL N_2H_4)
WEIGHT, LB.	78.2	40.2	-38
COST, $K	310.4	336.4	+26

3-39

Fig. 4.3-17 RCS Tradeoff

On an individual basis it appears that the GN_2 reaction control system is lower in complexity as well as in cost. However, when the total propulsion module is considered, the N_2H_4 reaction control/orbit adjust subsystem is the least complex system. The GN_2 regulator and the high-pressure (3500 psi) GN_2 tank are eliminated.

The selected reaction control subsystem is a hydrazine-fueled system which is combined with the orbit adjust subsystem. Common tankage is manifolded to 0.1 and 1.0 lb thrusters as well as the 5 lb. OAS thrusters. The subsystem operates in a blowdown mode. The equipment is installed in a module mounted on the aft end of the S/C as shown in Fig. 4.3-18.

CONCLUSION

- Integrate RCS with OAS
- All N_2H_4 RCS/OAS
  - Saves 38 lb
  - Costs $26K
  - Simpler installation
  - Higher safety and reliability

4.3.4.3 RCS/OAS REDUNDANCY - The subsystem selected has a minimum of redundancy. Further redundancy to provide fail-safe operation can be added to the subsystem for minimal weight and cost. The addition of a second cross over manifold and a second solenoid/seat assembly to each of the thruster valves, as shown schematically in Fig. 4.3-19, provides fail-safe operation.

COST OF REDUNDANCY

- Weight penalty = 4.4 lb
- Cost penalty = $20K

4.3.4.4 OTS CONFIGURATION TRADEOFF

The primary means of providing orbit transfer capability if required* is the use of solid rocket motors (SRM) shown in Fig. 4.3-20.

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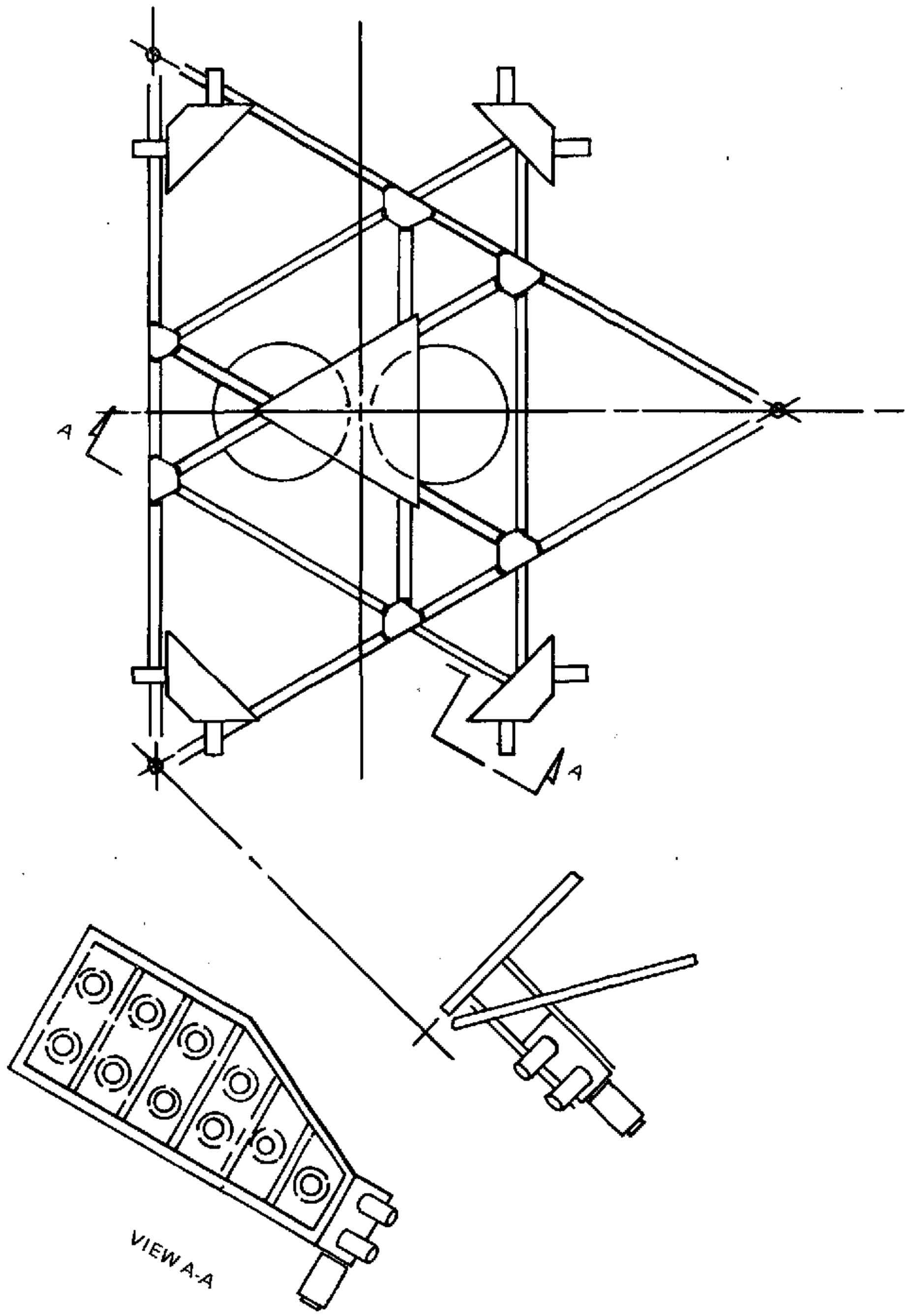
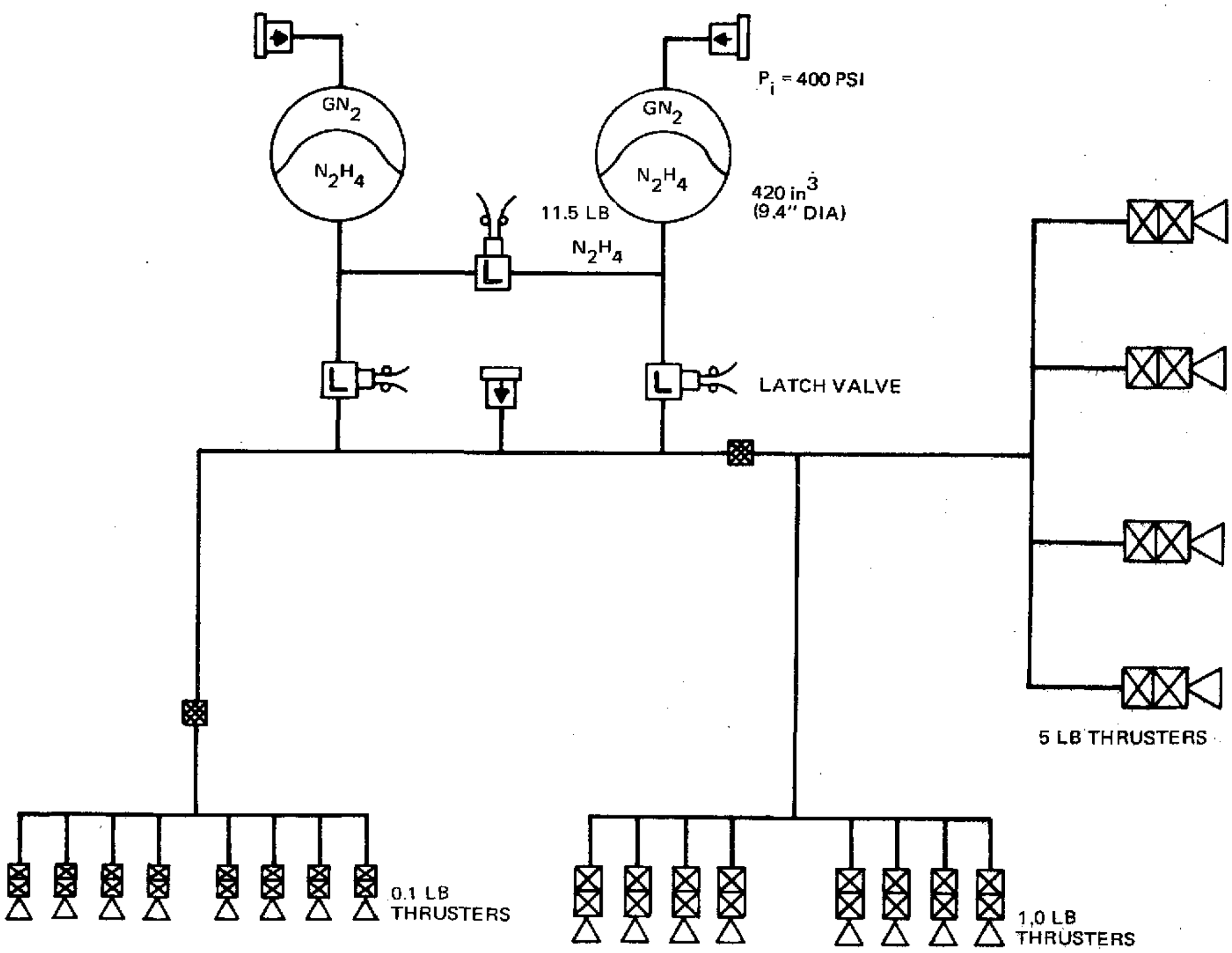
*The GAC baseline does not require orbit transfer for a 366 nm orbit.

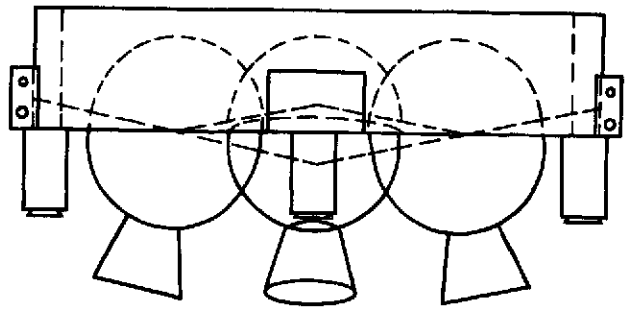


3-121 **Fig. 4.3-18 RCS/OAS Module**



3-40

Fig. 4.3-19 Fail-Safe Redundant RCS/OAS



3-41 Fig. 4.3-20 OTS/SRM Module

The alternatives studied were an orbit adjust subsystem using from 75 lb SRM thrusters and a bipropellant system based on the Shuttle orbit maneuvering subsystem (OMS). For each case, it was assumed that the Shuttle would operate in a 300 nm orbit with the EOS being transferred to and from a 493 nm orbit.

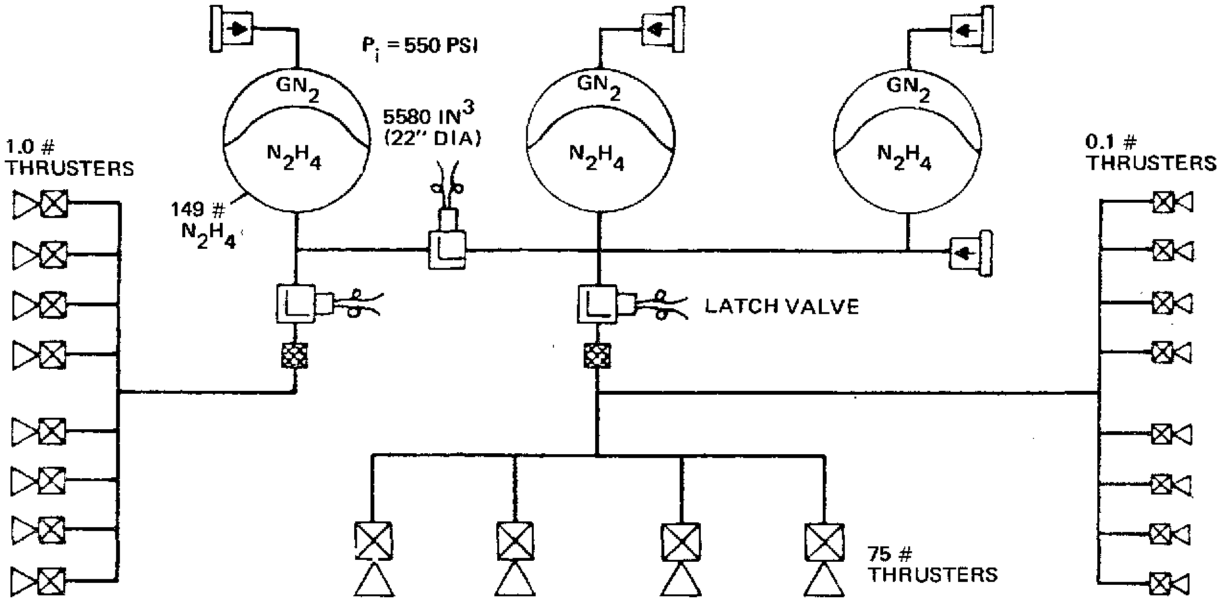
DELTA 2910 LAUNCH VEHICLE - The use of the OAS for orbit transfer requires the replacement of 5 lb thrusters with 75 lb thrusters. In addition, because of the much higher propellant load required, the two 9.4 inch tanks are replaced by three 22 inch tanks. With the exception of the three tanks, the subsystem is schematically identical to the OAS/RCS shown in Fig. 4.3-21. In order to obtain an equal comparison, the combined SRM/OAS weight and cost was compared to the all-N_2H_4 system. The results of the trade are shown in the table in Fig. 4-3-21.

It should be noted that the costs are based on a four-vehicle/four-flight program. As the number of flights increases, the cost differential becomes extremely large. At 12 flights, the cost differential exceeds $1M (see the Program Cost savings curve on Fig. 4.3-21.)

CONCLUSIONS (DELTA L/V OTS.)

- N_2H_4 OTS - Increases weight 59 lb
  Saves $306K (4 flights)
- Cost savings increases with number of flights

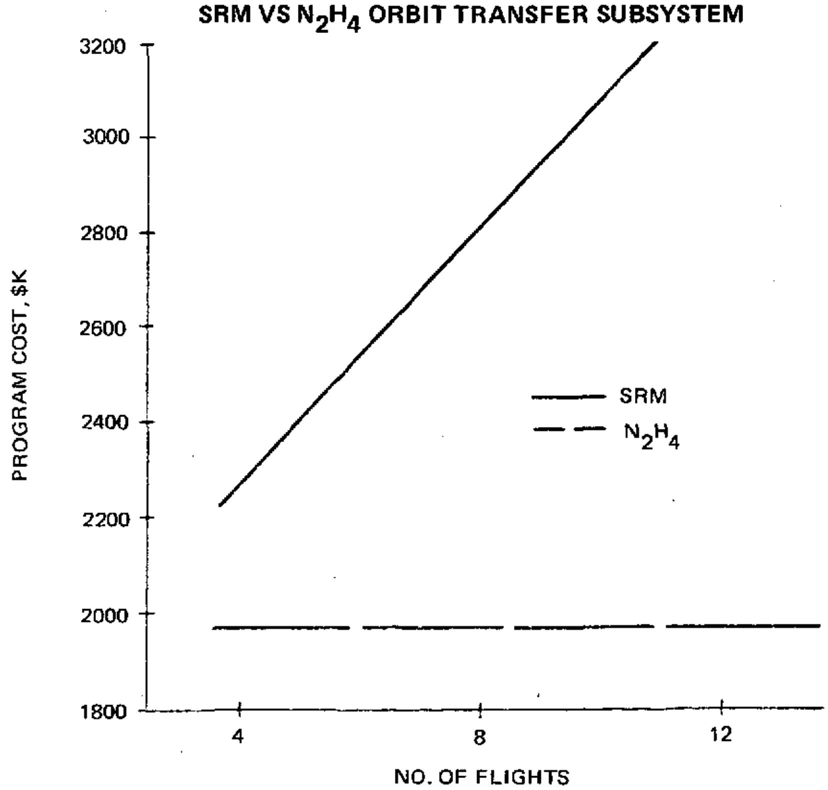
TITAN LAUNCH VEHICLE - The use of a bipropellant OTS appears to be viable only for the larger EOS spacecraft being studied, vehicles which require orbit transfer stages such as the Boeing Burner II type design. This study assumed the use of the SRM-2 motors called



N_2H_4 ORBIT TRANSFER SUBSYSTEM

	COMBINED SRM/OAS	ALL N_2H_4	DELTA (COMBINED VS ALL N_2H_4)
Weight, lb.	405.6	464.6	+59
Cost, $K	2,283	1,977	-306

SRM VS N_2H_4 ORBIT TRANSFER SUBSYSTEM



PROGRAM COST SAVINGS

Fig. 4.3-21 Delta 2910 L/V OTS Tradeoff

for in the Boeing design. A bipropellant system using N_2O_4 and MMH and sized to the same total impulse as the 4 SVM-2's was assumed. The system is shown schematically in the figure on the facing page. A four vehicle/four flight program was also assumed. The results are shown in the table in Fig. 4.3-22.

At first glance, the bipropellant system appears to be a poor choice. However, this system uses Shuttle hardware which is designed to operate for 100 missions. It is, therefore, capable of operating over the full lifetime of the EOS. The Total Program Cost curve on the facing page shows that a cross-over point occurs in total program costs at the 10-11 flight point in the program.

CONCLUSIONS (TITAN L/V OTS)

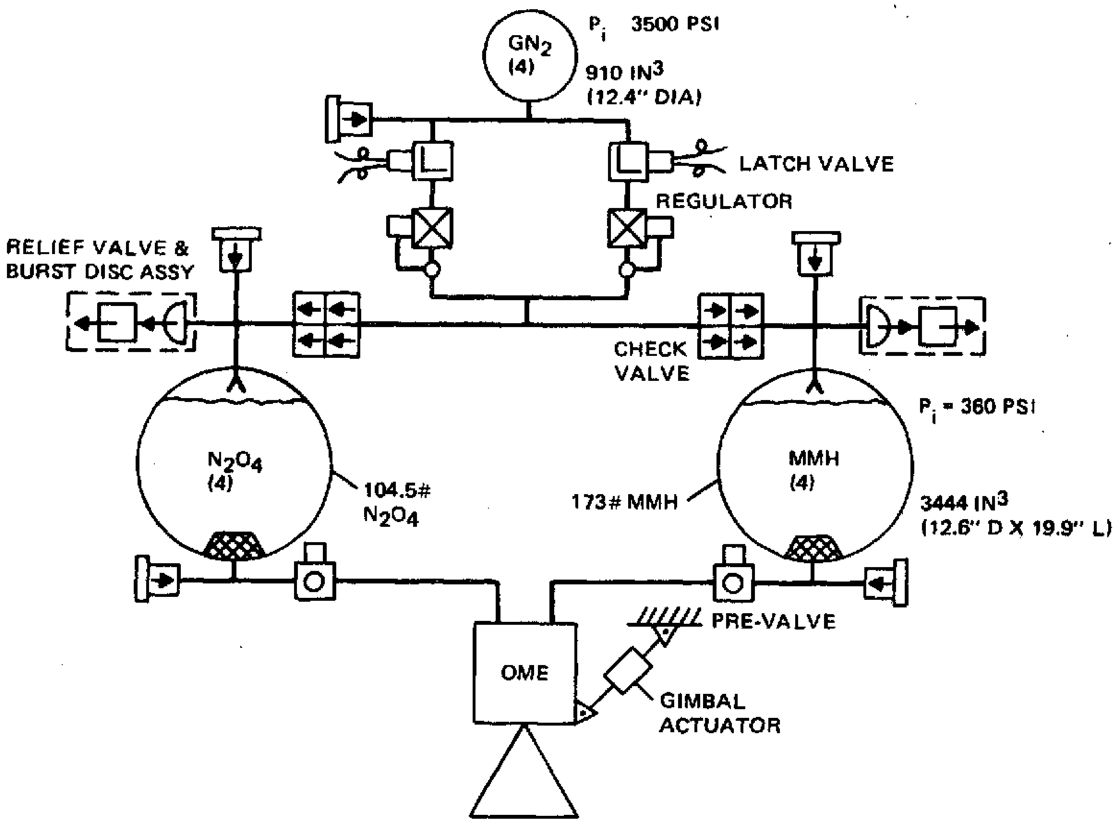
- SRM's are lower cost for 10 flights or less
- Bipropellant OTS - Increases weight 77.1 lb
  Costs $1,458K (4 flights)

4.3.5 THERMAL CONTROL

The thermal evaluation of the subsystems was based on a modular configuration. Two module configurations were considered for the Delta triangular arrangement and a square configuration was considered for the Titan arrangement.

Evaluations were conducted for the Land Resources Mission. Worst case min/max environment heat fluxes were used for each module (see appendix). An altitude range of 300 nm to 500 nm and DNTD range of 9:30 am to 12:00 Noon was used as the basis for determining the worst case heat fluxes. Where applicable, heat input from the solar array was also included.

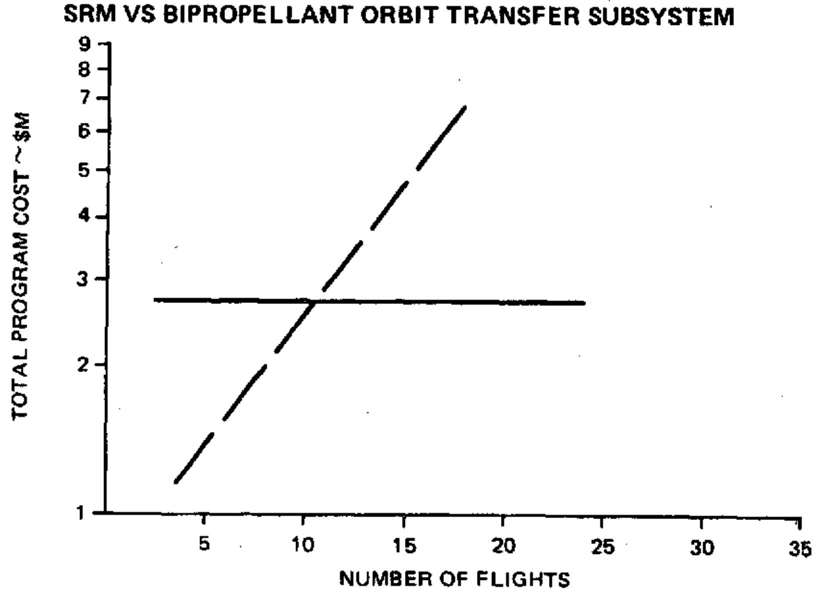
Up to this point in the study, the subsystems were treated on a cost/capability trade off basis. The large matrix of black box candidates within a subsystem has not allowed finalization of load analysis or module equipment arrangement. Therefore the analysis of the modules could only be considered on a lumped parameter, parametric basis. The ability to reject heat was studied as a function of alternate thermal options for each location. This technique established module location and feasibility of passive control, supplemented with heater power during low power dissipating modes.



ORBIT TRANSFER SUBSYSTEM (LIQUID ENGINE ~ OME)

	SRM'S	BIPROPELLANT	DELTA (SRM VS BIPROP)
Weight, Lb	1400.0	1477.1	+77.1
Cost, $K	1,247	2,705	+1,458

SRM VS BIPROPELLANT ORBIT TRANSFER SUBSYSTEM



SRM VS BIPROPELLANT OTS TOTAL PROGRAM COSTS

3-43

Fig. 4.3-22 Titan L/V OTS Tradeoff

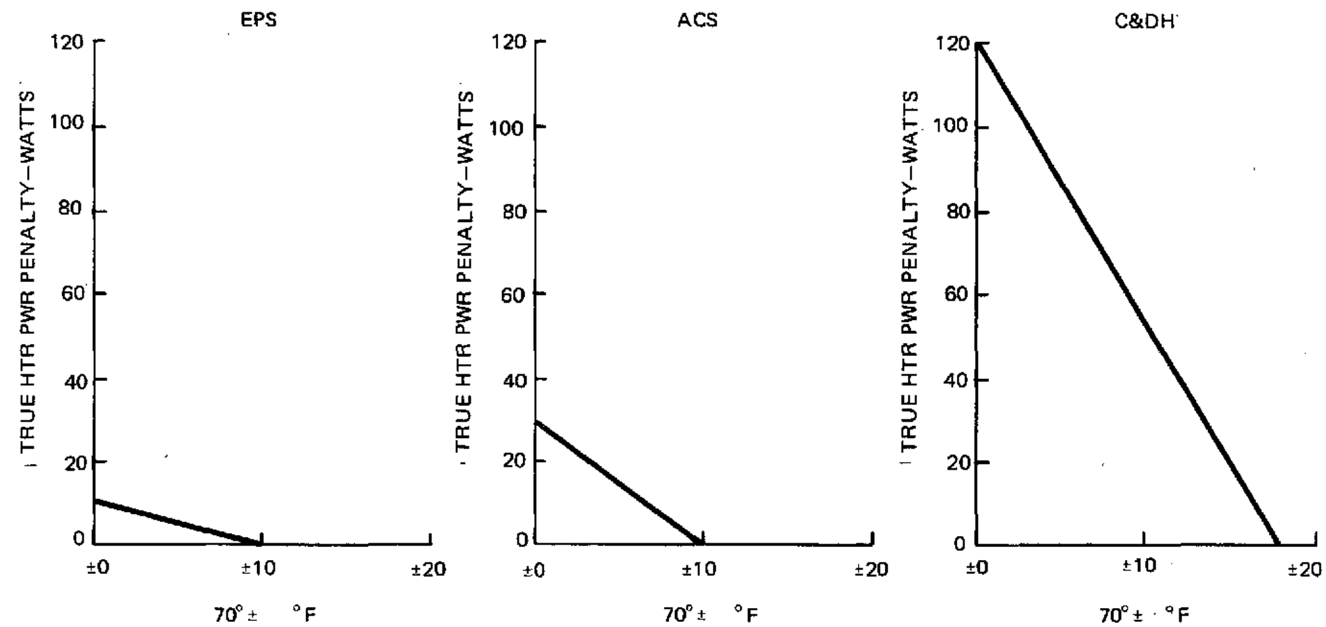
The cost per watt can vary between \$0.75K and \$1.75K per Watt (see appendix), depending on the array selected. The savings in module acceptance test costs resulting from a narrow operating temperature range (± 10^{o}F vs ± 50^{o}F) can be as much as \$16K. The fundamental passive design cost trade off is therefore the impact of equipment operating temperature range on power subsystem and test costs. The cost of active control to reduce heater power (if a penalty) must then be considered. These trade offs are used to achieve the design-to-cost targets.

Figures 4.3-23 and -24 show the design-cost trade offs conducted for the selected Delta module locations (apex toward nadir, Delta 1 Configuration). The evaluations were conducted for the LRM. A hot case heat rejection capability of 150 watts was assumed for each module and the true cold case heater power penalties were determined for various module operating temperature ranges about a mean of 70^{o}F. True heater power penalties in this case would be the power in excess of 150 watts for each module.

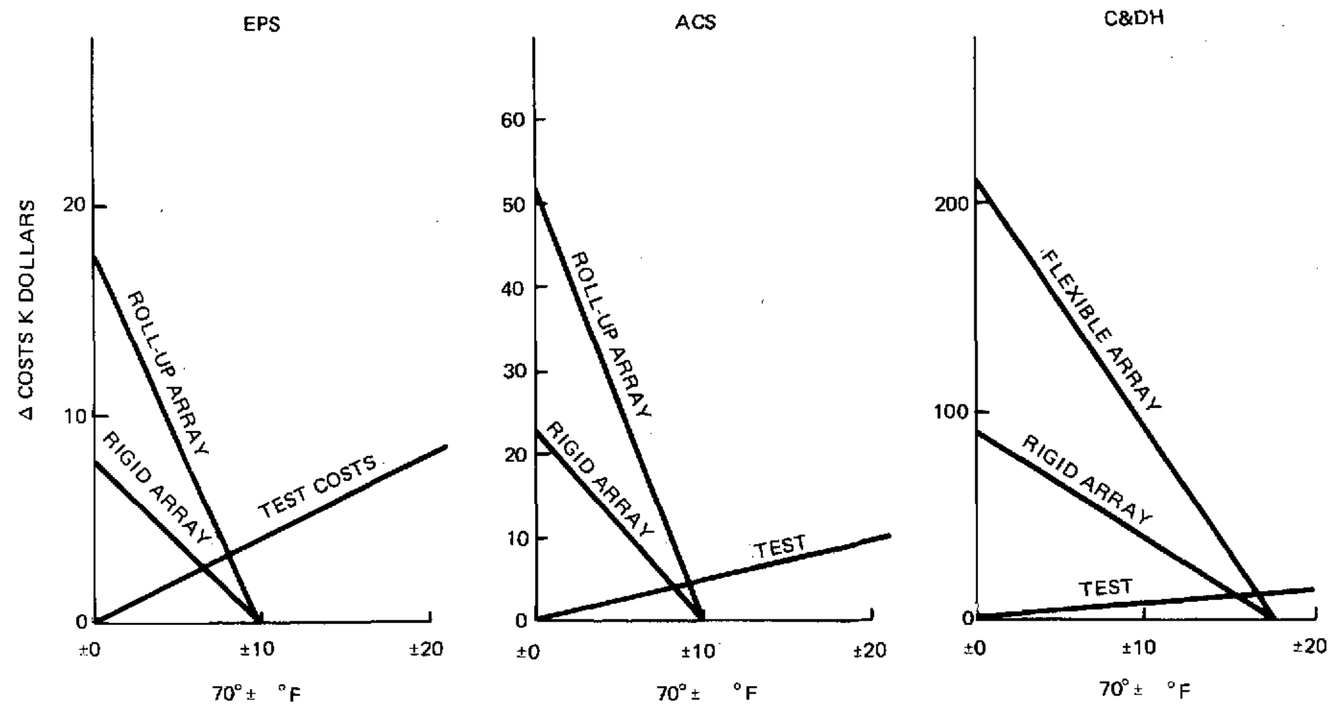
The increase in power subsystem costs at .75K/watt for a rigid array and 1.75K/watt for a flexible array were then determined. The increase in module acceptance test costs (20K at ± 50^{o}F) as a function of operating temperature range is also shown. The curves show that a minimum cost is achieved for each module when heater power penalty costs are eliminated (i.e. ± 10^{o}F for EPS and ACS and ± 20^{o}F for C&DH). The results of a similar evaluation, shown in the appendix, for the Titan configuration of modules (with the C&DH module facing the earth), are quite close to the above and yield the same conclusions.

It must be emphasized that these conclusions are subject to modifications resulting from detailed module design evaluations. Local power loading within a module may require further trading off of active control costs to achieve these narrow operating temperature ranges. Designing for failure modes (such as a solar array hang-up) would modify these results due to designing with different min/max heat fluxes. Future mission considerations would have a similar impact.

Although common modules for each mission are the goal, thermal tailoring of the modules for each mission is the most cost-effective approach. The ability for all modules to be tailored for each mission would be a design requirement. It is envisioned that a thermal design handbook will be developed to define the thermal changes required for each mission. These modifications will be limited to the module external heat sink and skin.



3-44 **Fig. 4.3-23 Passive Design For 150 Watts Hot Case Heat Rejection**



3-45 **Fig. 4.3-24 Change in Subsystem and Acceptance Test Costs with Temperature**

4.4 INSTRUMENT MISSION-PECULIARS

4.4.1 WIDEBAND DATA HANDLING AND COMPACTION

The function of the spacecraft wideband data handling and compaction subsystem is to convert, format, multiplex and select multichannel analog video data from multispectral scanning instruments and produce serial digital NRZ data streams at suitable rates for transmission to primary and low cost ground stations via radio links.

Figure 4.4-1 depicts the overall data handling subsystem block diagram. The functions shown within the dotted lines are the data handling subsystem functions required to handle the baseline instrument payloads and include appropriate commandable switching functions to apply output data to WBVTR/TDRS, QPSK modulator or BPSK modulators. The data handling subsystem will have appropriate interfaces with the instruments, S/C prime power, S/C on-board computer, the WBVTR/TDRS option function and the direct primary ground station or LCGS radio link modulation functions. In addition, an auxiliary low data rate interface is envisioned to handle appropriate low rate S/C telemetry and/or PMMR instrument data to the extent that it can be inserted during available TM overhead format time. A speed buffer function is included to provide for a partial scene data compaction option for either TM or HRPI. In general, the output rates from either TM or HRPI data handling units will be constrained to be equal at a value R megabits per second. Similarly, the compacted rates from either instrument will be constrained to some convenient rate R/X megabits per second. Due to the high rate (R) and the probable physical separation of units from the QPSK modulator, in-phase (I) and quadrature (Q), retiming functions are included to properly condition the NRZ signals for QPSK modulation. The synthetic aperture radar (SAR) signals will also be constrained in rate to equal value R, to provide compatibility and commonality of modulation equipment. Subsequent paragraphs will treat the exact alternative approaches to dividing and modularizing the circuitry.

4.4.1.1 ALTERNATIVE CONFIGURATIONS

There are several factors contributing to the determination of the data handling and data compaction subsystem configuration.

- Instrument manufacturer's desire for a digital interface.
- Magnitude of the digital rate at the instrument interface (if digital).
- Size of the electronics package that can be placed inside or in external contact with the instrument.

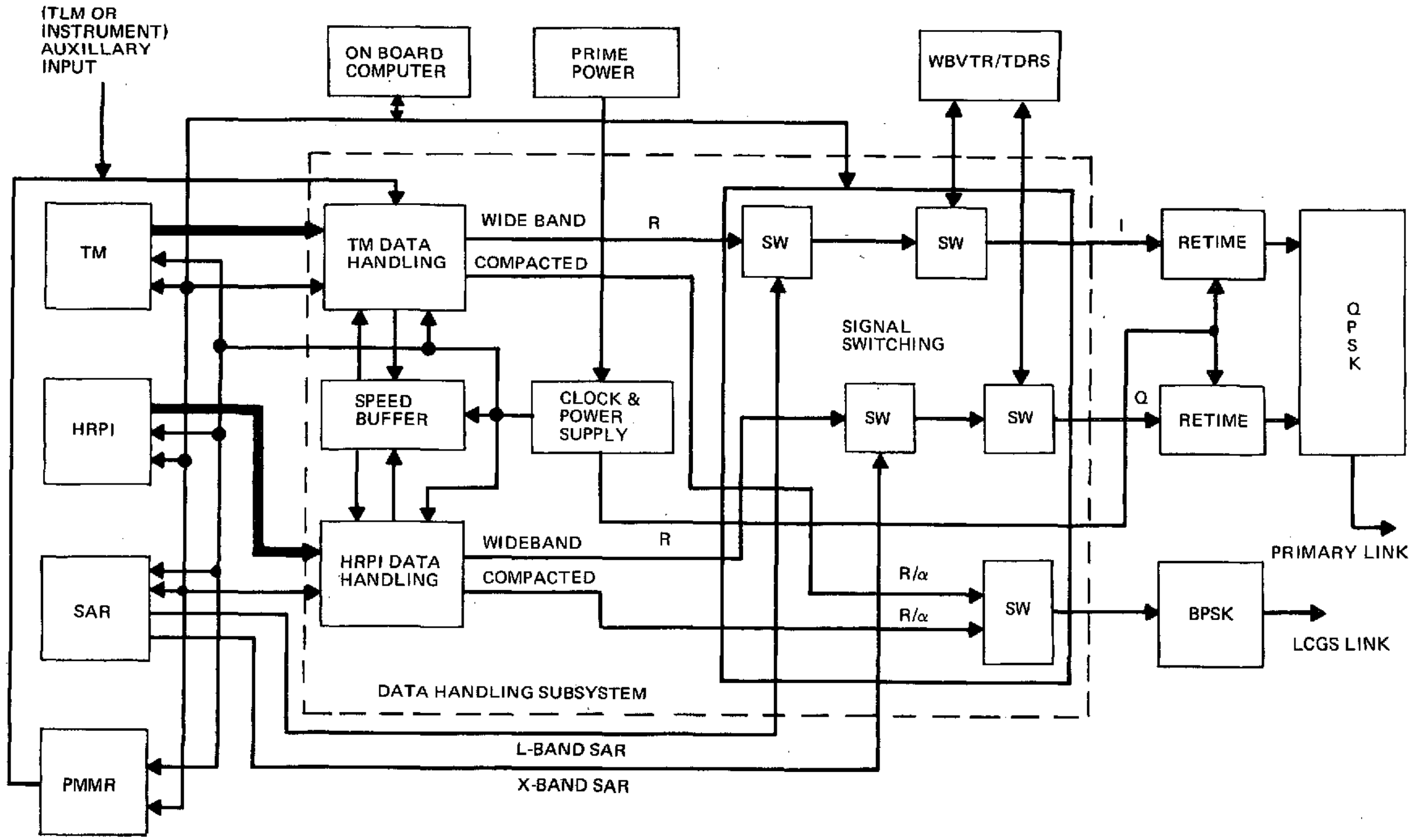


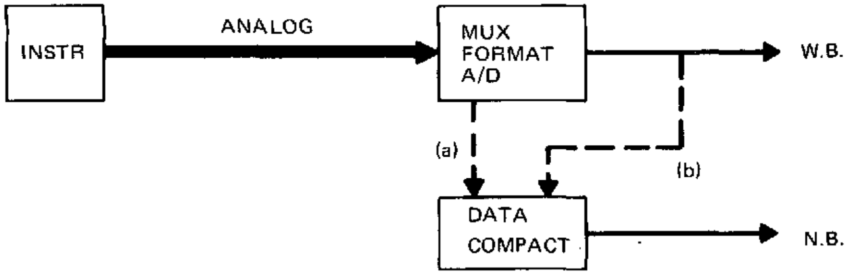
Fig. 4.4-1 Data Handling Subsystem Functional Block Diagram

- Consideration for multiple instrument data handling function requiring modular flexibility.
- Consideration for multiple instrument data compaction where only one compacted instrument output can be selected and sent at one time.
- Presence or absence of high capacity speed buffering in the data compaction functions.

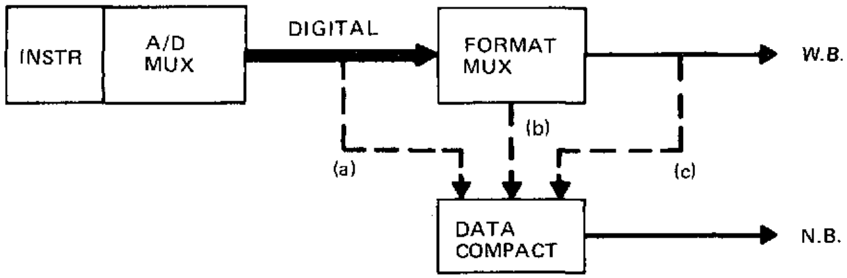
Figure 4.4-2 portrays the single-instrument data handling and compaction alternatives that could be considered for EOS. Figure 4.4-2a shows an analog interface (multiple bands and multiple detectors per band) feeding a data multiplexer, formatter and A/D converter. Analog cabling run lengths of several feet are assumed for this case and the number of analog lines could run as high as 100 per instrument. The Radiation Inc. MOMS point design assumed such a configuration with analog inputs. In discussions with Hughes Aircraft instrument people, it became apparent that they would prefer to have a digital interface with the instruments due to the difficulty in guaranteeing performance at the end of long analog lines. Their concept of this would be to provide a single digital serial bit stream per color band for either the TM or HRPI at a moderate data rate (approximately 12 Mbps for TM and approximately 26 Mbps for HRPI). The electronics for multiplexing and A/D conversion would either be supplied by them or someone else. The key point being that they (Hughes) would be responsible for the instrument output performance to the digital per band interface level. This low-rate digital instrument interface is depicted in Figure 4.4-2b. It is assumed that the electronics associated with the instrument is in very close proximity to the instrument and is tested and delivered with the instrument.

A logical extension of Figure 4.4-2b is to include the entire data handling function in the electronics package to be delivered with the instrument, as shown in Figure 4.4-2c. This approach, however, requires a high digital data rate interface to be reckoned with by the instrument manufacturer (approximately 100 Mbps) and is less attractive from this viewpoint. However, only a single data line (two with clock) would be required.

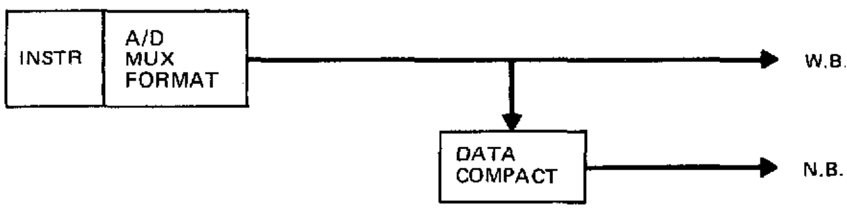
The configurations shown in Fig. 4.2-2a, -2b and -2c assume that the data compaction function is separate from the handling function. There are options here as to where to sample the wideband sensor data for compaction to the lower narrow band rate (approximately 20 Mbps) which depends on the data compaction options implemented and whether a partial scene option is included, thus requiring a data buffer. Simple options requiring no data buffer could be treated differently than those that included a partial scene option.



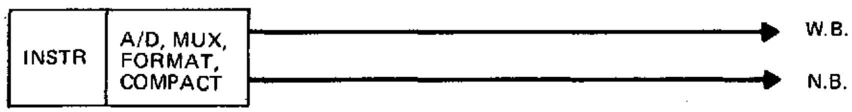
(a) ANALOG INSTRUMENT INTERFACE



(b) LOW RATE DIGITAL INSTRUMENT INTERFACE



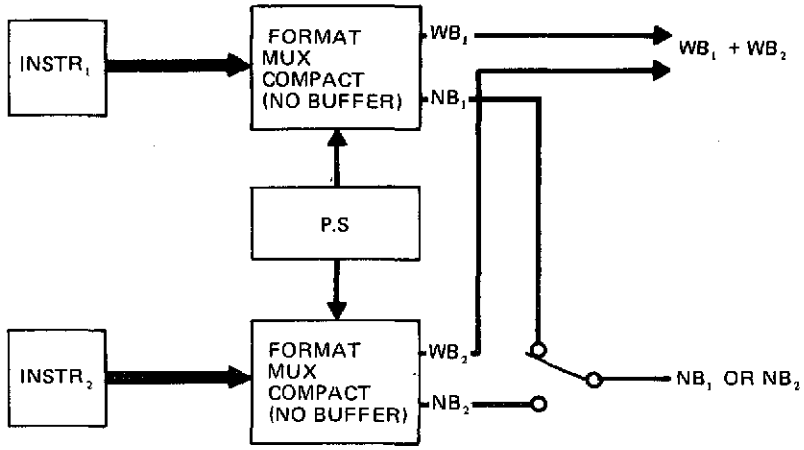
(c) WIDEBAND HANDLING COLOCATED WITH INSTRUMENT



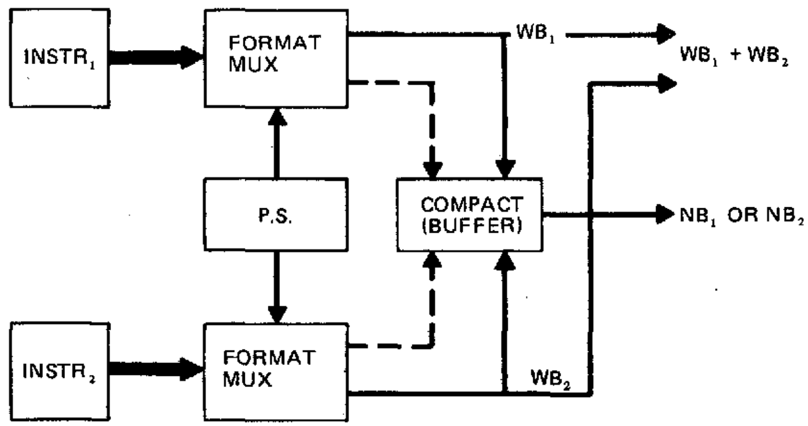
(d) WIDEBAND HANDLING & NB COMPACTION COLOCATED WITH INSTRUMENT

WB: WIDEBAND NB: NARROW BAND

Fig. 4.4-2 Single-Instrument Data Handling and Compaction Alternatives



(a) WITHOUT COMPACTION BUFFERING



(b) WITH COMPACTION BUFFERING

Fig. 4.4-3 Multiple-Instrument Data Handling and Compaction Alternatives

A final concept that could also be considered follows directly from this discussion and Figure 4.4-2c. That is, combine the entire data handling and compaction functions in one unit in close proximity to the instrument (Figure 4.4-2d).

Figure 4.4-3 shows the considerations of multiple instrument data handling and data compaction. At the outset of the EOS study, it was supposed that multiple instruments meant more than two. It soon became apparent that probably only two instruments (TM and HRPI) existed that require high rate data handling and data compaction. The desire for modularity and flexibility have indicated separate formatting and multiplexing modules for the TM and HRPI. The data compaction function is another matter. If no partial scene compaction option is offered, no buffer is required and the functions for band selection or detector averaging can best be combined with the respective instrument data formatting and multiplexing functions. If partial scene options are offered for each instrument, the large buffer required makes a separate shared compaction function module an important candidate consideration. Another possibility is to just provide a common shared buffer module with the remaining compaction circuitry located with each instrument formatting-multiplexing module. It is not yet clear whether the HRPI compaction function should include a partial scene option. If it does not, any buffer would be strictly associated with the TM instrument, with appropriate modification to the argument for a separate compaction buffer circuit module.

Block diagram examples in more detail for the analog and low-rate digital instrument interface cases are shown in Figure 4.4-4 and 4.4-5 respectively. Figure 4.4-4 is essentially the MOMS approach for the analog digital interface and a separate compaction (with speed buffer) module approach. Note also that the compaction function with speed buffering is shared with the TM or HRPI in Figure 4.4-4. In the original MOMS point design, only TM was assumed to be compacted. Figure 4.4-5 depicts the block diagram for a TM wideband data handling and compaction unit for the low rate digital interface case. The compaction circuit shown is integral with the formatting and multiplexing functions corresponding to the Figure 4.4-3a concept. At this juncture it is not clear whether partial scene compaction is to be employed. Hence, the storage block in the lower right hand corner of Figure 4.4-5 may or may not be a significant hardware item. If it is and both TM and HRPI compaction is involved, then a separate compaction circuitry/buffer module may be entertained in accordance with the Figure 4.4-3b concept.

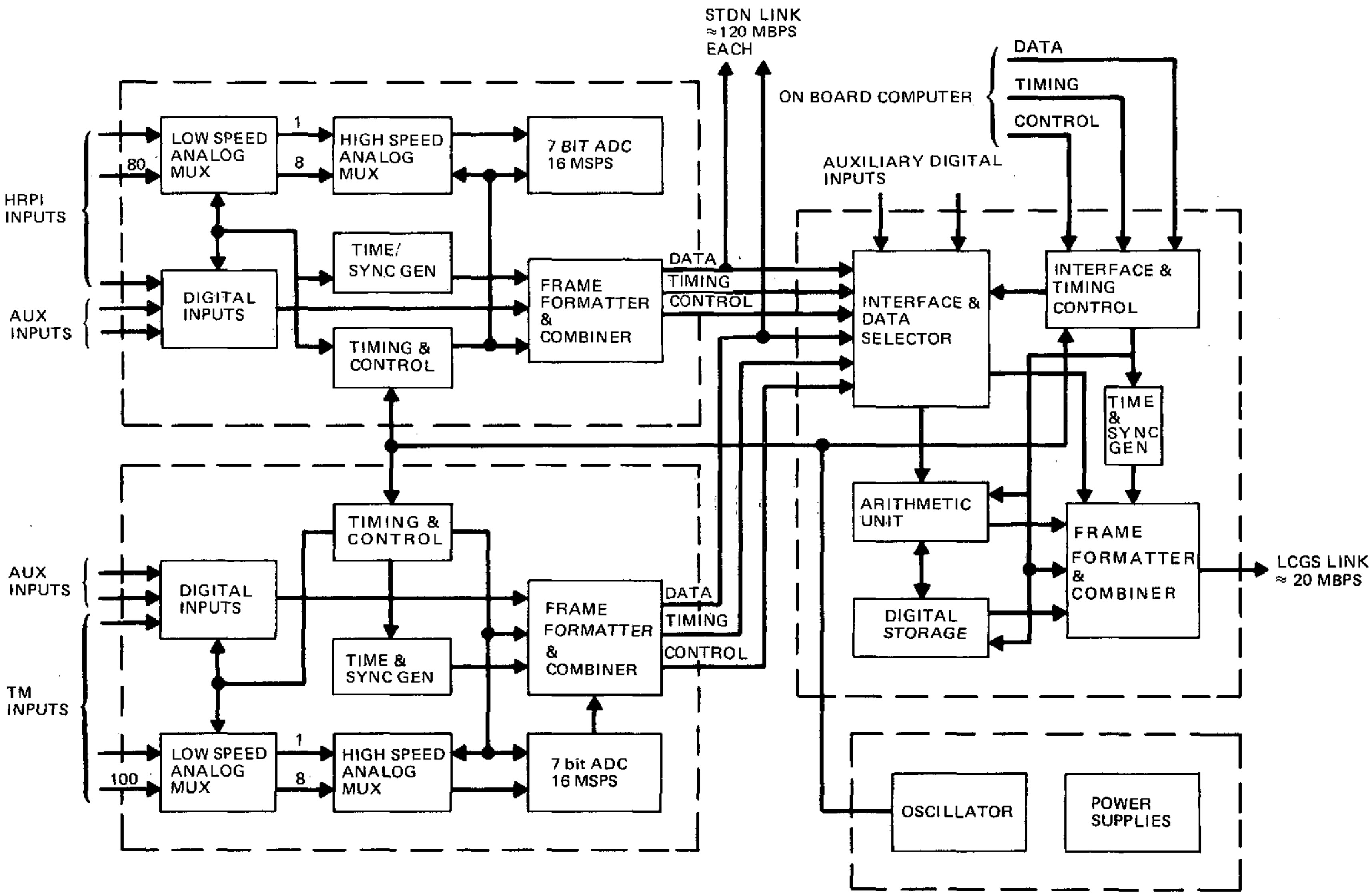


Fig. 4.4-4 Wideband Data Handling and Compaction Subsystem With Analog Instrument Interface

Figure 4.4-5 represents the latest thinking on the data handling configuration. The HRPI handling and compaction would be similar to the TM handling and compaction shown in Figure 4.4-5 with appropriate modification to the number of input color bands and the detailed manner of inserting overhead information (formatting).

4.4.1.2 ALTERNATIVE APPROACHES

INSTRUMENT INTERFACE - Three alternatives exist for the instrument interface. These are: (1) analog, (2) non-formatted digital plus control lines and (3) transparent digital containing all required overhead. The analog interface has tentatively been eliminated, primarily from the instrument manufacturer's point of view. The remaining approaches are both acceptable to comply with a digital interface. The non-formatted digital plus control line approach for this interface has been selected, thus performing all of the formatting and combining of overhead information in the data handling unit for the respective instrument.

OVERHEAD - Included in each line with the instrument sensor data is a small amount of non-sensor data. This non-sensor data and that part of the sensor data not occurring during the active scan interval are referred to as overhead. The latter can be used for optically calibrating the instrument. An electrical calibration of the post sensor electronics can also be obtained during this time.

In both the TM and scanning HRPI, the active scan interval (the interval between the start of line (SOL) and end of line (EOL)), is eighty percent of the scan time. This allows twenty percent for calibration data and the non-sensor overhead. Five to ten percent should be sufficient for the calibration.

The active scan is initiated at the SOL signal or some short known time after it. A code is inserted in the output bit stream in place of the sensor data that identified the SOL. Additional overhead data may also be inserted after the SOL code, e.g., relative time. At the EOL another code is inserted to identify the end of the active scan. Additional overhead may also be inserted at this time.

Ground processing of the received wideband data starts with the detection of the SOL code. To make this code stand out, an idle code can precede it. This code would provide a good background against which the SOL code can easily be found. The SOL code should provide the maximum distance from the idle code to allow error correction and prevent false acquisition on noise. A simple combination would be a seven bit code and its complement. However, such a short code could cause an LCGS radio link flux density problem. A

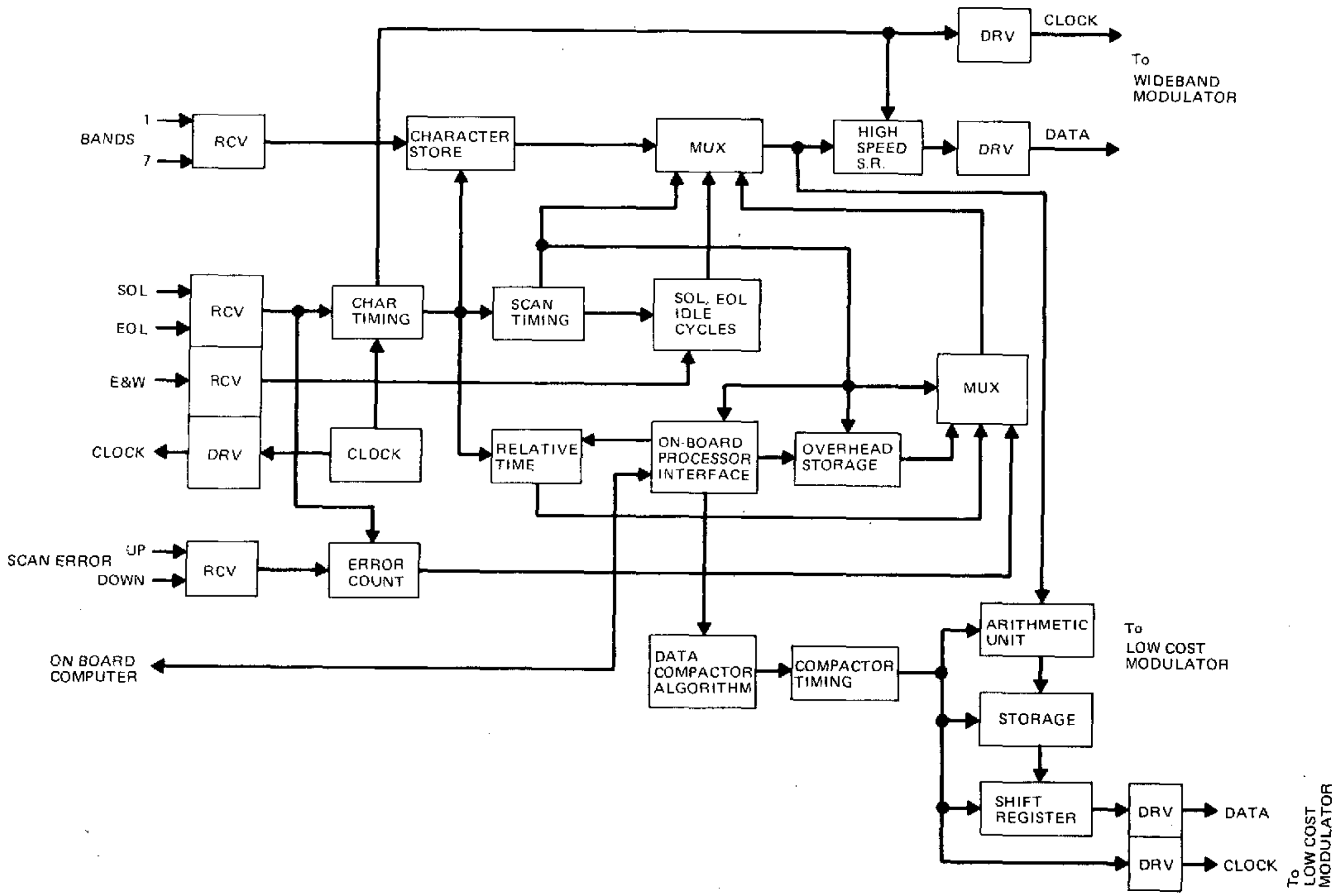


Fig. 4.4-5 Wideband Data Handling and Compaction Subsystem with Low-Rate Digital Instrument Interface, TM Case

PN sequence could also be used which would not create the flux density problem, but would be more costly to implement. Start and end of lines should be accurate to at least 0.1 pixel elements. This allows these codes to occur on character boundries (7 bits) within the output data stream. It appears that a seven bit code is sufficient for SOL, but EOL should be 14 or 21 bits since it does not follow an idle code.

In order to process the data on the ground, parameters such as pitch, roll and yaw and their time derivatives are required. This information is available in the on-board processor (AOP) and would be supplied to the data handling equipment. Its inclusion is probably best accomplished after the EOL code and before the calibration data.

Another requirement for ground processing is the elimination of scan non-linearities. It is assumed that a scan error signal can be derived from the instrument. Such a signal could be a delta modulation signal that can be accumulated in the data handling equipment or sent as it is received. Overhead slots exist in the format of the TM during the active scan time due to the reduced resolution of band 7. The accumulated error can easily be sent during these times. In the HRPI no overhead exists during the active scan (unless the total wideband rate exceeds four times the single band rate) and either a data bit has to be preempted now and then or the scan error must be transmitted after EOL or calibration. The latter approach requires a large buffer to store the error signal. In the former case, preempting a single bit every second pixel would suffice, and require a buffer of less than four thousand bits. Another implementation of the former case could be to preempt the least significant bit of a particular detector substituting the scan error signal. The detector A/D accuracy is reduced but this may not be a problem especially if seven bit quantization is used.

WIDEBAND RATE - The data rate from the instrument is determined by a number of parameters. For a square pixel, the pixel dimension or resolution has the largest effect. Spacecraft altitude has the least effect, a variation of about 10 percent for an altitude of 500 to 1000 kilometers. The equation below expresses the bit rate as a function of these parameters.

$$R = \frac{Vg \bullet Dg \bullet S \bullet N \bullet K}{(RE)^2 \bullet E}$$

where Vg = ground speed of spacecraft = $\dfrac{7.91}{(1 + \frac{h}{6378})}$

Dg = ground distance covered in a scan

S = number of bits per sample

RE = dimension of resolution element

E = instrument efficiency

N = number of bands per instrument

K = number of samples taken for each resolution element in scanning direction

h = S/C orbit altitude

In the current design, the parameter K seems to be the one whose value is least certain. With an assumed value of K = 1 and

h = 680 Km

Dg = 185 Km

S = 7 bits/sample

N = 7

E = 0.8

RE = 30m

the rate, R, for the TM is 85.75 mbps. With the TM and HRPI at the same wideband rate and

K = 1 sample/resolution element

S = 7 bits/sample

N = 4

E = .8

RE = 10m

the ground distance covered by a HRPI scan is about 36 Km. This is about 25 percent less than the point design of the HRPI, a result due mainly to the switch from a high efficiency (100 percent) push broom HRPI to the lower efficiency scanning HRPI.

COMPACTED DATA RATE - The compacted data rate should be in the order of 20 megabits per second. At the wideband rate of 86 mbps a division by four produces a rate of 21.5 mbps. This is easily produced by counting down the wideband clock and provides for some very good compaction schemes. The actual rate used for compacted data should be examined from both the LCGS radio link and S/C data handling viewpoints. Increased cost for a two dB increase in power, e.g., to handle 30 mbps rather than 20 mbps, may be more than offset by decreased complexity in the data handling equipment.

COMPACTION - Data compaction schemes with a digital instrument interface require only digital processing techniques. Three types of compaction are considered for both the HRPI and the TM; band selection, resolution reduction, and partial coverage. The specifics of these are obviously influenced by the rate used and the amount of buffering available.

Resolution reduction from 10 meters to 20 meters in the HRPI and from 30 meters to 60 meters in the TM decreases the data rate to one quarter of the wideband value. This is the minimum reduction and one quarter rate (20 to 30 mbps) can probably be supported by a low cost ground station. With this compaction scheme, very small buffer storage is required, resulting in about one half the number of bits produced each sample time. These factors favor a compacted rate of one quarter the wideband rate.

Band selection for the HRPI is rather limited since there are but four bands. The only feasible selection is a single band, and this would require a rate one quarter of the wideband. This is another factor supporting the one quarter rate recommendation. For the TM a combination band selection and resolution reduction would provide a single band at 30 meters resolution and three bands at 60 meters resolution at the same rate. As an alternative, two bands at 30 meters resolution could be provided at two-sevenths of the wideband rate. This rate, however, is not compatible with the HRPI unless the data is buffered and fill or overhead inserted into the bit stream.

Band selection and resolution reduction described above at one quarter the wideband rate are attractive since they do not require a large buffer memory to smooth the difference between the input and output rates. The compaction hardware could be included in the same physical package with the wideband, reducing interfaces and power dissipation.

The third alternative, partial coverage, requires a buffer memory. Its size is dependent upon the number of bands involved, the compacted rate and the amount of coverage. With a compacted rate of 21.44 mbps, Figure 4.4-6 indicates the minimum buffer size as a function of swath width covered for TM and HRPI instruments. Maximum coverage is limited by either swath width in the case of two bands of the TM, or amount of data for N = 3 through 7 of the TM and N = 2 through 4 for the HRPI. It is interesting to note that

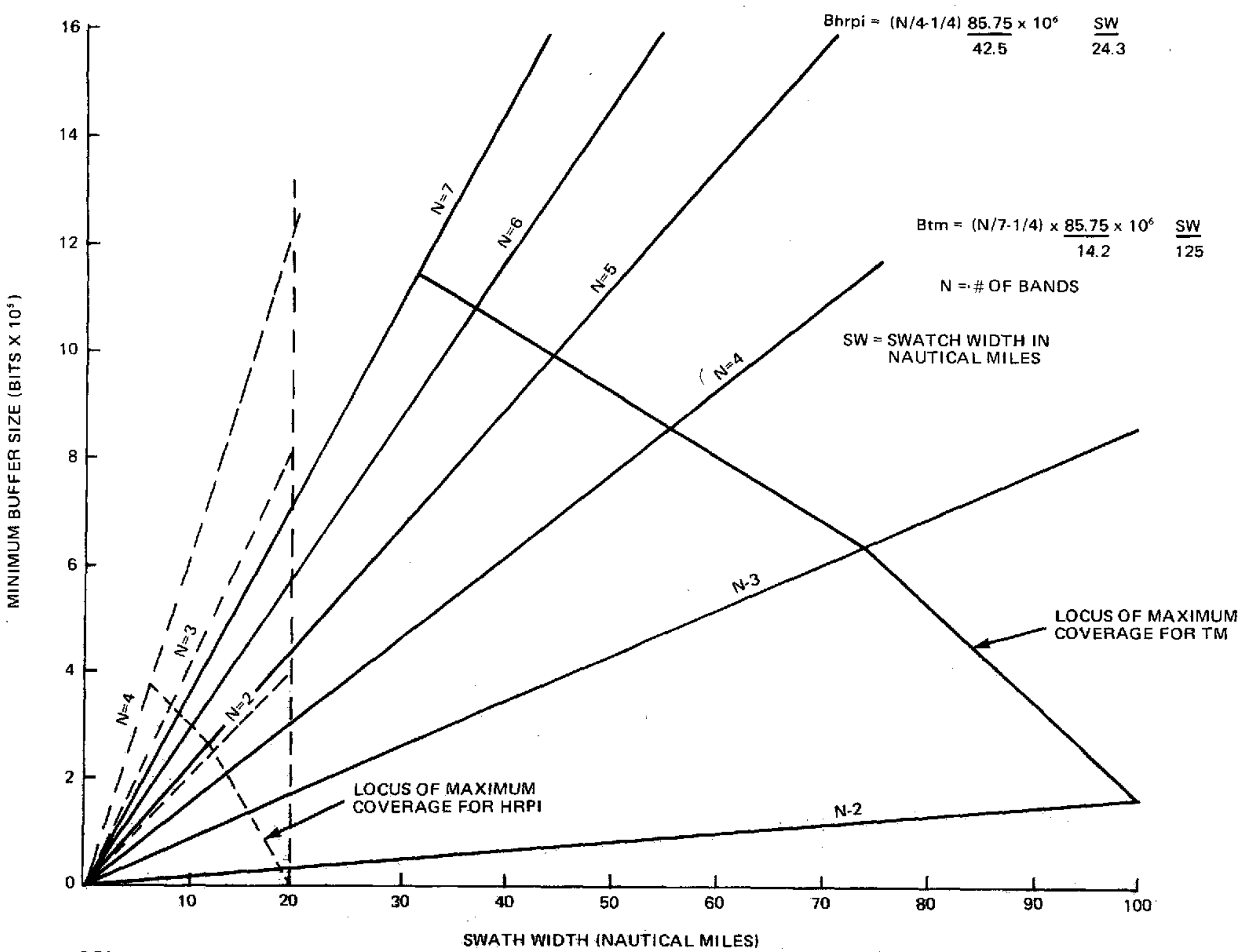


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Fig. 4.4-6 Buffer Size Vs Swatch Width

a HRPI reduced resolution supplies a type of TM partial coverage. A swath width of 19 nautical miles or less involving the first four bands of the TM (same as the HRPI bands) is more economically produced by the HRPI than the TM, since no buffer is required for reduced resolution. In order to cover the same swath width for four bands with the TM, a 300 kilobit memory is required and the resolution is degraded from 20 meters to 30 meters.

4.4.1.3 COST, SIZE, WEIGHT AND POWER CONSIDERATIONS

Preliminary cost, size, weight and power information for the Data Handling and Compaction unit shows:

• Cost	
Recurring	\$610K - \$2.3M
Non-Recurring	\$250K - \$500K (4 Units)
• Size	860 - 950 in^3
• Weight	25 - 50 lbs.
• Power Consumption	90 - 120 watts

The range of values is provided to indicate the uncertainty in the $0.5 - 1.0 \times 10^6$ bit speed buffer which would be required if partial scenes are desired from the data compaction unit. This speed buffer is considered a development risk at this time.

4.4.2 WIDEBAND COMMUNICATIONS

4.4.2.1 INTRODUCTION

Wideband communications is here defined as the complement of spacecraft communication subsystems required to communicate sensor data, both uncompacted and compacted, from the EOS spacecraft to earth. Two basic direct communication link requirements have been identified, distinguished by the magnitude of the date rate to be transmitted and the earth terminal resources to receive each transmission. The primary link has been sized at 240 Mbps and is required to be received by STDN earth terminal sites having an antenna diameter of at least nine meters and a system noise temperature of 200°K. The low cost ground station (LCGS) link has been sized to handle a reduced data rate of 20 Mbps and is to be received by small earth terminals having a G/T of 11dB/°K with antenna diameters of approximately six feet and system noise temperatures in the order of 900°K. The key issue in the design of the small earth terminals is low cost.

In addition to the direct communication link requirements the EOS spacecraft may also be required to relay sensor data to earth through the proposed NASA Tracking Data Relay Satellite System (TDRSS). Use of the TDRSS capability will entail an IF interface at the TDRSS location. The entire TDRSS ground station complex will be leased by the Government.

The TDRSS link and the two direct links represent the baseline EOS spacecraft wideband communication subsystem requirements. Alternative spacecraft (S/C) subsystem designs were considered including the use of a wideband video tape recorder (WBVTR) and MSS recorder to replace the TDRSS link requirement and alternative approaches for establishing the primary ground station (PGS) and LCGS links. The alternatives considered for establishing the direct links involved the selection of spacecraft antenna diameters and amplifier power requirements and the choice of either X or Ku-band for the downlink frequency.

4.4.2.2 ALTERNATIVE SUBSYSTEM CONFIGURATIONS

The wideband communication subsystem configurations considered include the TDRSS link to transmit a total of 240 Mbps of data at Ku-band to the TDRSS for relay to ground stations, a tape recorder option in lieu of the TDRSS relay and direct link configurations for PGS and LCGS at X-band and Ku-band. (The 240 Mbps capability allows for the accommodation of missions with two instruments, each operating at 120 Mbps. For Mission A with TM and MSS, the bit streams at 100 and 20 Mbps, respectively, will be bit-stuffed or multiple-sampled into the two-quadrature 120 Mbps channels.) Each of these configurations will be described below.

TDRSS LINK - The TDRSS subsystem provides the means of transmitting a total of up to 240 Mbps data at Ku-band to the TDRSS for relay to ground stations. The TDRSS subsystem operates in conjunction with an antenna tracking subsystem. The EIRP from the spacecraft is specified to be at least 61.3 dBW for the 12.5 foot diameter steerable antenna and 7 dBW for the omni antenna. The subsystem components are:

- QPSK* modulator for two 120 Mbps data inputs
- Up converter/driver
- RF amplifier
- Omni antenna for the tracking system between the EOS and the TDRSS
- Directional antenna

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* Modulation choice is treated in Appendix D.

TAPE RECORDER OPTION - An option in lieu of the TDRSS relay of data acquired while the EOS spacecraft is not in view of primary or local user ground stations is to tape record these data and read-out later when the EOS is in view. This option employs three recorders: one wideband video tape recorder (WBVTR) for the Thematic Mapper (TM) instrument output and two ERTS-type recorders for either the MSS or the compacted Thematic Mapper (CTM) data. The WBVTR has a read in and read out rate of approximately 120 Mbps for periods of up to 15 minutes and a total data volume capacity of 10^{11} bits. The two ERTS-type recorders are capable of data rates of 16 or 20 Mbps and have a capacity of 15 minutes of data at these rates. These tape recorders interface with the data sensors and the direct link wideband spacecraft communication subsystems.

X-BAND DIRECT LINKS - The basic requirements that must be satisfied by the direct link wideband S/C communication subsystem are a 100 Mbps TM and 16 Mbps MSS link to PGS sites and a 20 Mbps link for CTM data to low cost ground stations. However, the PGS link subsystem has been sized to provide for 240 Mbps channels in order to accommodate future higher rate EOS missions and a redundant high rate channel for high reliability and to mitigate the data handover problem between sites. The frequency plan is discussed in Appendix D paragraph D. 2.

There are two alternative approaches for establishing the primary and LCGS links. Approach 1 employs two narrowbeam steerable antennas for both the primary and LCGS links; Approach 2 uses a steerable antenna for the primary link and a fixed antenna for the LCGS link. Further design choices involve the selection of RF power amplifier levels and efficiencies and the inherent backup capability of a particular configuration in the event of failure. The subsystem components consist of:

- QPSK* modulator for PGS link and DPSK* modulator for LCGS link
- Up converters/drivers
- RF filters
- RF power amplifiers (PGS and LCGS)
- Directional Antenna(s); Approach 1
  Fixed Antenna; Approach 2
- DC to DC converters
- RF switches
- Combiners or multi-coupler

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*Modulation choice is treated in Appendix D.

KU-BAND DIRECT LINKS (OPTION) - An option with the X-band wideband communication subsystem for direct transmission to both PGS and LCGS is the Ku-band Direct Link Transmission Subsystem. Alternative configurations for this option are constrained by the limited availability of space-capable power amplifier devices in the band of interest (14 GHz). The subsystem component types are basically the same as those required for the X-band subsystem configuration.

4.4.2.3 ALTERNATIVE SUBSYSTEM TRADES AND ISSUES

Several factors influence the design of the baseline S/C wideband communication links: the choice of frequency band, the power flux density limit at the earth's surface for a particular band, the available power sources, propagation losses, antenna diameters and receiver noise temperatures and the back-up capability in the event of failure that a particular configuration permits. These and other technical factors are discussed in this section for each S/C communication subsystem and its alternative configurations (options); cost, weight and size considerations are treated in the following section.

ALTERNATIVE SUBSYSTEM PERFORMANCE CONSIDERATIONS - Three frequency bands are potentially applicable for space transmission at the high bandwidths involved here: X-band (specifically, 8.025 - 8.4 GHz), Ku-band (specifically, 14.5 - 15.35 GHz), and K-band (21.5 - 22 GHz). Of these, only X and K are approved for operational use, whereas Ku is for R & D only. However, TDRSS is planned for Ku band operation. (This is space-space communication, not space to ground.) Power calculations for X and Ku-band operation are shown in Table 4.4-1. The reason why K band has not been pursued further or shown in the table is clear: the additional propagation losses and higher noise temperatures of receivers in this band make adequate link margins impossible. In the table, signal margins have been calculated for the TDRSS link, direct links to PGS sites and LCGS sites at both X and Ku-band and alternative X-band LCGS configurations using both steerable and fixed antenna designs. The various configuration power margins are calculated from the following relationship:

$$\text{MARGIN (dB)} = (10 \log_{10} \text{EIRP} + 10 \log_{10} G/T + 10 \log_{10} K) - 10 \log_{10} \text{Propagation losses} + 10 \log_{10} \text{Pointing losses} + 10 \log_{10} E_b/N_o + 10 \log_{10} R)$$

where:

EIRP = Effective radiated power relative to an isotropic radiator

G/T = Ratio of net ground station antenna gain excluding pointing losses to the system noise temperature

K = Boltzmann's constant, 228.6 $dBW/K^o/Hz$

Propagation losses = Free space + O_2/H_2O + Rain + Cloud losses

Pointing losses = Ground antenna pointing losses

R = Data rate required (240 or 20 Mbps)

E_b/N_o = Required energy per bit to noise density ratio for the demodulator type and error rate specified.

In all but the Ku-band direct link design with a fixed spacecraft antenna, the resulting margins are at least 3 dB for the specified EIRP and G/T parameters indicated in Table 4.4-1. In this latter case a link margin of only 2.6 dB is realized under the worst case loss conditions assumed in the calculations. It should be noted however that the EOS system design specifications on EIRP and G/T are minimum acceptable values and hence careful design and component selection procedures could result in higher values of these parameters and a concomitant higher link margin. The magnitudes of the various loss terms used in the power budget calculations to arrive at EIRP and G/T are only representative of the losses that might be encountered with a given design and could be reduced with a concerted design effort.

Detailed design tradeoffs and considerations for each of these links will be discussed in subsequent paragraphs. However, it should be noted that, in the case of the Ku-band options for the PGS and LCGS links, the limited availability of power amplifier devices at this frequency (15 GHz) constrains the possible alternative configurations and the resulting link margins for reasonable ground station G/T values. The corresponding constraints at X-band are not so severe and hence admit to design trades between power amplifier and S/C antenna sizes for a given EIRP requirement and a corresponding set of alternatives for the ground station complex of equipments.

DIRECT LINK TRADES AND ISSUES - The two alternative approaches for establishing the primary and LCGS links at X-band are depicted in Figure 4.4-7. Alternative 1 employs two narrow beam steerable antennas with a one-foot diameter and 28 dBi gain, each fed by a 4 watt power amplifier. Due to ITU power flux density limits the LCGS link must be

Table 4.4-1 Signal Margins with EOS Wideband Links

PARAMETER / OPTION		X-BAND (8.25 GHz)			KU-BAND (15 GHz)		
		PGS 240 MBPS	20 MBPS (LCGS)		TDRSS LINK 240 MBPS	DIRECT LINK 240 MBPS	DIRECT LINK 20 MBPS
		STEERABLE S/C ANT.		FIXED S/C ANT.	STEERABLE S/C ANT.		FIXED S/C ANT.
S/C TRANS. PWR.	dB	6.0 (4W)	1.0(2) (4W)	17.0 (50W)	12.0 (16W)	12.0 (16W)	14.0(3)
CIRCUIT LOSS	dB	1.5	1.5	1.5	1.2	3.0	1.0
S/C ANT. GAIN	dB	28.0 (1')	28.0 (1')	7.0 (±30°)	51.0 (12.5')	30.0 (1')	7.0 (±30°)
ANT. POINT LOSS	dB	2.5	2.5	0.5 (AXIAL RATIO)	0.5	3.0	—
S/C EIRP(1)	dBW	30.0	25.0(2)	22.0	61.3	36.0	20.0
FSL	dB	180.0 (2° EL)	173.3 (30° EL)	171.0 (50° EL)		186.0 (2° EL)	176.0 (50° EL)
O_2/H_2O	dB	1.0	0.5	0.2		1.0	1.0
RAIN	dB	3.1	1.0	0.5		7.0	3.0
CLOUD	dB	3.0	0.5	0.3		5.0	3.0
PROPAGATION LOSS	dB	187.1	175.0	172.0		199.0	183.0
GROUND ANT. GAIN	dB	55.4 (30')	41.5 (6')	41.5 (6')		60.5 (30')	52.0 (12')
POINT LOSS	dB	0.5	1.5	1.5	REF. (4)	0.5	0.5
SURF. TOLER. LOSS	dB	0.3	0.5	0.5		0.5	0.5
CIRCUIT LOSS	dB	0.5	0.5	0.5		0.5	0.5
DUAL FEED LOSS	dB	0.5	—	—		0.7	—
NET ANT. GAIN	dB	53.6	39.0	39.0	$(R_{dB}$-25) = EIRP, dBW REQ'D	58.3	50.5
K	dBW/K°/Hz	-228.6	-228.6	-228.6		-228.6	-228.6
T	dB° K	23.0 (200° K)	29.5 (900° K)	29.5 (900° K)		24.0 (250° K)(5)	28.5 (710° K)(6)
C/KT	dB/Hz	102.1	88.1	88.1	INCLUDES	99.9	87.6
R	dB/Hz	83.8	73.0	73.0	(83.8	83.8	73.0
E_b/N_o @ 10^{-6} PGS @ 10^{-5} LCGS	dB	13.0	12.0	12.0	12.5	13.0	12.0
MARGIN	dB	5.3	3.1	3.1	3.0)	3.1	2.6

NOTES: (1) EIRP'S ARE MINIMUM REQUIRED; ANY COMBINATION OF POWER, GAIN AND LOSSES THAT SATISFIES EIRP IS PERMITTED. COMBINATIONS SHOWN ARE REPRESENTATIVE ONLY.
(2) BACK-OFF DUE PFDL = 25 dBW @ 20 MBPS
(3) TWO 16W TUBES IN PARALLEL
(4) TDRSS USERS' GUIDE RETURN LINK CALCULATION. (NO CODING FEASIBLE AT THIS DATA RATE.)
(5) COOLED PARAMP
(6) TDA

power controlled to 1 dBW as indicated in the power budget calculations of Table 4.4-1. Alternative 2 employs a narrow beam steerable antenna one foot in diameter (28 dBi), and a 4W power amplifier for the PGS link, and a fixed S/C antenna with a $\pm 30^{\circ}$ beam width and 7 dBi gain and a 50 watt power amplifier for the LCGS link.* The link power budget calculations indicate that both alternatives yield the same margins with the specified EIRPs and G/Ts shown in Table 4.4-1. The primary performance consideration between the two alternatives involves the different coverage areas for the LUS. Coverage/look angle charts are provided in Appendix D. Whereas the fixed antenna covers $\pm 30^{\circ}$ as seen from the spacecraft, the additional EIRP of the steerable antenna allows coverage of about $\pm 50^{\circ}$ for the same link margin. This can be of considerable utility to the Local User community. Cost, weight, size and power consumption considerations for these Alternatives will be addressed in the next section.

The Ku-band option for the direct links to PGS and LCGS sites demands higher S/C EIRP and ground station G/T ratios due to the larger propagation losses at this frequency. The performance penalties for the Ku-band option relative to the X-band configuration, Alternative 2, are the reduced margins for both the PGS and LCGS links as shown in Table 4.4-1. The PGS link at Ku-band has an acceptable 3.1 dB margin relative to the 5.3 dB margin for the X-band approach; the LCGS link has only a 2.6 dB margin in comparison to the 3.1 dB margin for the corresponding fixed S/C antenna X-band configuration. An advantage of the Ku-band option is that the required modifications to the PGS sites for operation at Ku-band are already being planned and hence no further modifications to accommodate X-band would have to be made if the EOS operated at this downlink frequency. On the other hand the cost differential for LCGS sites may well dictate the most cost-effective approach depending on the number of local user stations in the system. Cost, weight, size and power considerations will be addressed in a subsequent section.

TDRSS/WBVTR OPTION - The baseline EOS wideband communication subsystem includes a relay capability wideband sensor data to earth via the TDRSS. This capability is primarily intended for use with International Data Acquisition (IDA) missions so that the EOS can

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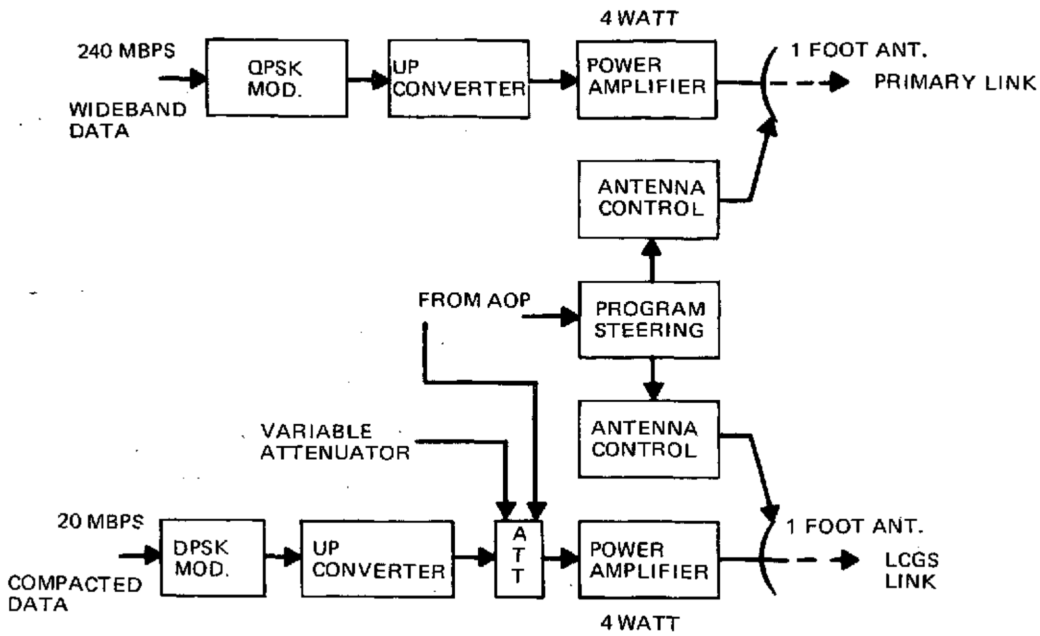
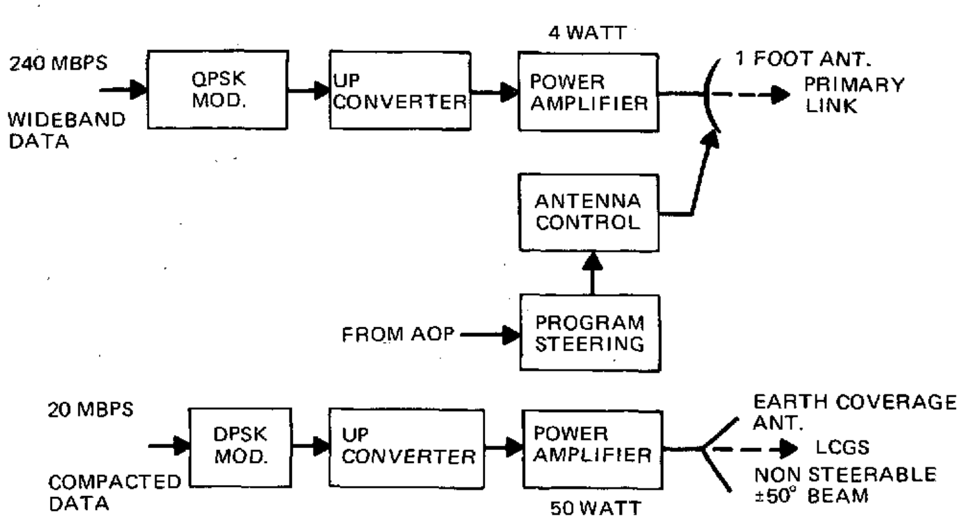
* Note: The baseline spacecraft approach differs from this Alternative 2 in that two identical narrowbeam antenna subsystems, in addition to the fixed antenna subsystem, are required. Here only one narrowbeam antenna subsystem is considered, because to evaluate Alternative 1 versus the baseline would present a very distorted picture. The two alternatives as given show the tradeoffs in serving the LUS via a fixed versus a steerable system on the spacecraft.



(A) ALTERNATIVE NO. 1



(B) ALTERNATIVE NO. 2

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Fig. 4.4-7 X-Band Direct Link Communication Alternatives

relay these foreign data to CONUS for subsequent data reduction and distribution. An alternative to this configuration is to include on-board tape recorders of the wideband video and ERTS type for storage of the foreign data until the EOS is in view of a PGS location. The TDRSS configuration has been shown, by the IDA analysis study in Section 6.9 to be able to provide almost continuous coverage (approximately 96%) of all of the land areas of interest in the foreign data collection mission.

The tape recorder option was shown to provide coverage of approximately 75% of the land areas of interest in the same study. The reduced coverage relative to the TDRSS configuration was traceable to the requirement to dump data on the next orbit following collection or pay the price of storage for 2 to 3 orbit periods until the EOS was in view of one of two assumed data dump stations (Alaska or Goddard). Hence, on the basis of the percentage of data each option provides, the TDRSS alternative clearly outperforms the tape recorder option.

The power budget calculations for the TDRSS link and the X-band direct link, which is the assumed communication link for the tape recorder option given in Table 4.4-1, demonstrate that both alternatives provide adequate signal margin and hence acceptable communication link performance.

Final selection of a preferred approach will depend upon the cost, weight, size and power consumption considerations for these alternatives which will be addressed in the next section and the technical risks associated with the TDRSS data acquisition and tracking problem.

4.4.2.4 COST, WEIGHT, SIZE AND POWER CONSIDERATIONS

The preceding section has discussed the technical performance trades and issues for alternative wideband communication subsystems. This section will address the cost, weight, size and power consumption impacts of each of these alternatives and hence complete the primary comparative analysis considerations for the various communication subsystems and options. Tables 4.4-2 through -6 present the cost, weight, size and power consumption data of the TDRSS X-band direct link, Alternatives 1 and 2, Ku-band direct link and tape recorder subsystems respectively. The cost data for each subsystem or alternative is broken down into non-recurring and recurring costs per unit. Non-recurring costs generally include design and development, qualification modeling and fabrication, test equipment and tooling and qualification test costs. Recurring costs include production

Table 4.4-2 TDRSS Subsystem

SUBSYSTEM / COMPONENT	COSTS ($K) NON-RECUR	COSTS ($K) RECUR/UNIT	WEIGHT (LB)	SIZE (CU IN.)	POWER (WATTS)
12.5′ S/C ANT. SUBSYSTEM	1500	515	80	–	–
16W KU-BAND TWTA + SUPPLY	–	100	10	11.8X6.6X4.4	140
L.O.					
QPSK MOD/EXC.	500	250	3	100	3
UP-CONVERTER					
D.C. CONVERTER					
TOTALS	$2000K	$865K	93 LB	442 CU IN.	143W

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Table 4.4-3 X-Band Direct Link Subsystem (Alternative 1)

SUBSYSTEM / COMPONENT (QUANTITY)	COSTS ($K) NON-RECUR	COSTS ($K) RECUR/UNIT	WEIGHT (LB)	SIZE (CU IN.)	POWER (WATTS)
(2) 1′ S/C ANT.	401	(2) 172	(2) 20	–	–
(2) 4W–TWTA + SUPPLY	400	(2) 60	(2) 8.2	12X6.2X4.2	14
(2) L.O.					
(2) UP-CONV.					
(2) RF FILTERS					
(1) QPSK MOD/EXC.	491	223	3	100	3.3
(1) DPSK MOD/EXC.	343	128	1.5	50	2
(2) D.C. CONV.					
TOTALS	$1635K	$815K	60.9 LB	774 CU IN.	33.3W

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Table 4.4-4 X-Band Direct Link Subsystem (Alternative 2)

COMPONENT (QUANTITY) \ SUBSYSTEM	COSTS ($K) NON-RECUR	COSTS ($K) RECUR/UNIT	WEIGHT (LB)	SIZE (CU IN.)	POWER (WATTS)
(1) 1' S/C ANT.	401	172	20	–	–
(1) E.C. S/C ANT.	18.5	1.2	4	–	–
(1) 4W TWTA/SUPPLY	400	60	8.2	12X6.2X4.2	14
(1) 50W TWTA/SUPPLY	400	100	9.5	11.8X6.4X4.4	172
(2) L.O.	↑	↑	↑	↑	↑
(2) UP-CONV.	↑	↑	↑	↑	↑
(1) QPSK MOD/EXC.	491	223	3	100	3.3
(1) DPSK MOD/EXC.	343	128	1.5	50	2
(2) D.C. CONV.	↓	↓	↓	↓	↓
TOTALS	$2053.5K	$684.2K	46.2 LB	804 CU IN.	191.3W

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Table 4.4-5 Ku-Band Direct Link Subsystem (Option)

COMPONENT (QUANTITY) \ SUBSYSTEM	COSTS ($K) NON-RECUR	COSTS ($K) RECUR/UNIT	WEIGHT (LB)	SIZE (CU IN.)	POWER (WATTS)
(1) 1' S/C ANT.	401	172	20	–	–
(1) E.C. S/C ANT.	18.5	1.2	4	–	–
(3) 16W TWTA	–	(3) 100	10	11.8X6.4X4.4	140
(2) L.O.	↑	↑	↑	↑	↑
(2) UP. CONV.	↑	↑	↑	↑	↑
(1) QPSK MOD/EXC.	500	250	3	100	3
(1) DPSK MOD/EXC.	350	130	1.5	50	2
(2) D.C. CONV.	↓	↓	↓	↓	↓
TOTALS	$1269.5K	$853.2K	58.5 LB	1176 LB	425W

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Table 4.4-6 Tape Recorder Option

TYPE	COSTS ($K) NON-RECUR	RECUR/UNIT	WEIGHT (LB)	SIZE (CU FT)	POWER (WATTS)
(1) WBVTR: BER < 10^{-6} 100 MBPS 9 X 10^{10} BITS, CAPACITY	NOT AVAILABLE	500	200	5.3	205 PEAK RECORD 270 PEAK REPRODUCE 60 ORBIT AVG.
(2) ERTS: BER < 10^{-6} R = 16/20 MBPS 15 MIN. CAPACITY	(ALREADY DEVELOPED)	(2) 300	72	1.9	50 PEAK RECORD 90 PEAK REPRODUCE 30 ORBIT AVG.
TOTAL (IMPACT ON EOS) a. WBVTR b. ERTS-TYPE		$1100K	344 LB (2) ERTS & (1) WBVTR	9.1 CU FT	450W PEAK REPRODUCE

units costs, fabrication, assembly and installation and acceptance tests for production rate/quantity. In the case of the EOS subsystems, the primary costs are associated with space qualifying the hardware and not the innovative nature of the required equipment and the lack of quantity discounts for the spacecraft equipment. Furthermore the cost data represent rough order of magnitude costs from responsive manufacturers of S/C hardware and accompanying ground system equipment. The data tabulated for each subsystem/option represents a compilation of information obtained from a number of manufacturers coupled with best engineering judgment where the data conflicted or was lacking. The following manufacturers responded to requests for the resulting tabulated information in the areas indicated.

Manufacturer	S/C Wideband Comm. Equipment
Motorola Government Electronics Div.	X-band direct link Alternative 1 & 2, except antennas & TWTA's
Harris Corporation Electronic Systems Div.	S/C Antennas: 1' steerable & E.C. fixed, X-band and 12.5' steerable, Ku-band
Cubic Corp.	TDRSS link, except antenna and data items
RCA	WBVTR Option
Raytheon	4-watt X-band solid state RF amplifiers
Hughes Electron Dynamic Div.	RF Amplifiers (TWTA): 4W & 50W, X-band and 16W, Ku-band

A summary of wideband communication subsystem/option data is presented in Table 4.4-7. The summary data represents the total subsystem costs, weights, sizes and powers. All totals reflect the product of the individual component values and the quantities of those respective items required to implement a particular subsystem or option, non-recurring costs have not been multiplied by the quantities of components since these represent one time development costs.

By summing the appropriate combinations of subsystem data items found in Table 4.4-7, various wideband communication subsystem data elements can be determined. This has been done for the TDRSS and tape recorder options in conjunction with the alternative direct link configurations (X and Ku-band) and the results tabulated in Table 4.4-8. Using recurring costs, weight and power consumption impacts as the primary basis of subsystem discrimination, the following conclusions can be drawn from these data:

(1) Any option involving the tape recorders in lieu of the TDRSS subsystem results in severe penalties in spacecraft weight and power requirements.

(2) The TDRSS and Ku-band direct link configuration requires substantially more power than either of the X-band direct link configurations.

(3) The first configuration, consisting of a TDRSS link in conjunction with the Alternative 1 (two steerable S/C antennas) X-band direct link subsystem, requires less power than the Alternative 2 configuration for the X-band direct link, with no substantial penalty in recurring costs or weight. The power requirement differences are traceable to the 50 watt TWTA demands in the Alternative 2 configuration.

(4) Although non-recurring costs and size were not used as the primary basis for subsystem option discrimination, the non-recurring cost data are not significantly different for the competing subsystems and the size factors reinforce the conclusions already drawn concerning combinations of tape recorder and Ku-band direct link configurations.

4.4.2.5 SUMMARY OF PREFERRED APPROACH AND ADDITIONAL CONSIDERATIONS

The "decision tree" which can be followed to select the preferred approach is shown in Figure 4.4-8. The factors influencing each node, as numbered in the figure are discussed in the paragraphs that follow:

Table 4.4-7 Summary of Subsystem/Option Data

SUBSYSTEM/OPTION	COSTS ($K) NON-RECUR	COSTS ($K) RECUR	WEIGHT (LB)	SIZE (CU IN.)(1)	POWER (WATTS)
TDRSS	2000	865	93	442	143
X-BAND DIRECT LINK					
ALT. 1: 2 STEERABLE S/C ANTENNAS	1635	811	60.9	774	33.3
ALT. 2: 1 STEERABLE + 1 EC/S/C ANT.	2053.5	684.2	46.2	804	191.3
KU-BAND DIRECT LINK (OPTION)	1269.5(2)	853.2	58.5	1176	425
TAPE RECORDERS:(5) (WBVTR + 2 ERTS-TYPE)	NOT AVAILABLE(3)	1100	34.4	15700 (9.1 CU FT)	450(4) (PEAK REPRODUCE)

NOTES: (1) SIZES SHOWN DO NOT INCLUDE ANTENNA SIZE.
(2) LOW DUE TO THE FACT THAT THE 16W-TWTA REQUIRES NO NEW DEVELOPMENT.
(3) NON-RECURRING COSTS FOR WBVTR WERE NOT SUPPLIED BY MANUFACTURER AND ERTS TYPE RECORDERS ALREADY EXIST.
(4) PEAK RECORD = 305W, ORBIT AVERAGE = 120W.
(5) THESE DATA DO NOT INCLUDE THE REQUIRED DIRECT LINK COMMUNICATION SUBSYSTEM ELEMENTS.

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Table 4.4-8 Summary of Total Subsystem Options

WIDEBAND COMMUNICATION SUBSYSTEM OPTIONS	RECURRING COSTS ($K)	WEIGHT (LB)	POWER (WATTS)
1. TDRSS + X-BAND DIRECT LINK, ALT. 1	1676	153.9	176.3(143)(1)
2. TDRSS + X-BAND DIRECT LINK, ALT. 2	1549.2	139.2	334.3 (191.3)
3. TDRSS + KU-BAND DIRECT LINK	1718.2	151.5	568 (425)
4. TAPE REC. + X-BAND DIRECT LINK, ALT. 1	1911	404.9	483.3 (450)
5. TAPE REC. + X-BAND LINK, ALT. 2	1784.2	390.2	641.3 (450)
6. TAPE REC. + KU-BAND DIRECT LINK	1953.2	402.5	875 (450)

NOTE: (1) FIRST NUMBER REPRESENTS THE SUM OF THE POWERS FOR THE INDIVIDUAL SUBSYSTEMS; NUMBER IN PARENTHESIS REPRESENTS THE POWER REQUIRED IF BOTH TDRSS OR TAPE RECORDED SUBSYSTEMS ARE NOT ASSUMED TO OPERATE CONCURRENT WITH THE DIRECT LINK COMMUNICATION SUBSYSTEM.

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NODE 1: WBVTR OR TDRSS - The data of the previous paragraph shows that TDRSS is preferable to WBVTR in terms of spacecraft size, weight, and power. TDRSS has some risk element with acquisition of two 12:5 ft. antennas at Ku band on two satellites, along with attendant reliability problems. On the other hand, WBVTRs have their own reliability problems and certainly some risk is attached to the 100 Mbps, 10^{-11}-bit recorder itself. In favor of the WBVTR approach is the fact that the data are "delivered" to the primary ground stations directly. With TDRSS, the data reception occurs at the TDRSS ground station, from which it must be relayed thousands of miles to the processing center. The costs for doing this are estimated at $2M/year for domestic satellite, microwave relay, or leased high speed aircraft. On the other hand, if the data delivery requirement can be relaxed to "overnight," commercial flights can be used to transfer taped data for negligible expense.

A firm decision cannot be made at this time. If the TDRSS program is in being and progressing at the time of decision, and assuming this TDRSS program will successfully solve the antenna pointing problem, it would appear that the TDRSS option is preferable.

NODES 2A/2B: X OR KU-BAND FOR DIRECT LINK* - The X-band solution is better than at Ku-band, in terms of spacecraft power, while being essentially equivalent in cost and weight. It will be shown that the LCGS for Ku-band costs some $10K more than at X-band, due mainly to the greater gain required. Ku-band antennas will be harder to point than X-band. X-band availability will be higher than at Ku. On the side of Ku-band is the $300K or more savings brought about by the existence of Ku-band feeds on the future STDN stations. On balance, it is concluded that X-band operation is preferred.

NODE 3A: ALTERNATIVE 1 OR 2 - The question here is whether the LUSs should be served via a fixed or steerable antenna on EOS. The following are pertinent factors:

- Steerable antenna must be pointed at the LUS in question.
- With a steerable antenna, only one LUS per pass is served.
- The fixed antenna alternative consumes substantially more spacecraft prime power.

The first and third items involve tradeoffs with other spacecraft subsystems, and must be considered in this context. Since both alternatives do use a steerable antenna for the primary direct link, the feasibility of doing so is not at issue. However, with the PGSs, the steering can be "fine tuned" in a closed loop mode, i.e., the PGSs also have command capability, whereas pointing towards LUSs must be open loop.

A firm preference is not possible at this time. It is recommended that present planning be a baseline with two steerable, one fixed antenna, X-band design. If further developments in system definition confirm that LCGSs can be readily served by a steerable antenna, the fixed antenna and its transmitter can simply be deleted from the design. If not, then the two steerable systems provide handover and backup advantages.

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* As noted, K-band is not viable due to high losses and noise temperatures which lead to unacceptable performance.

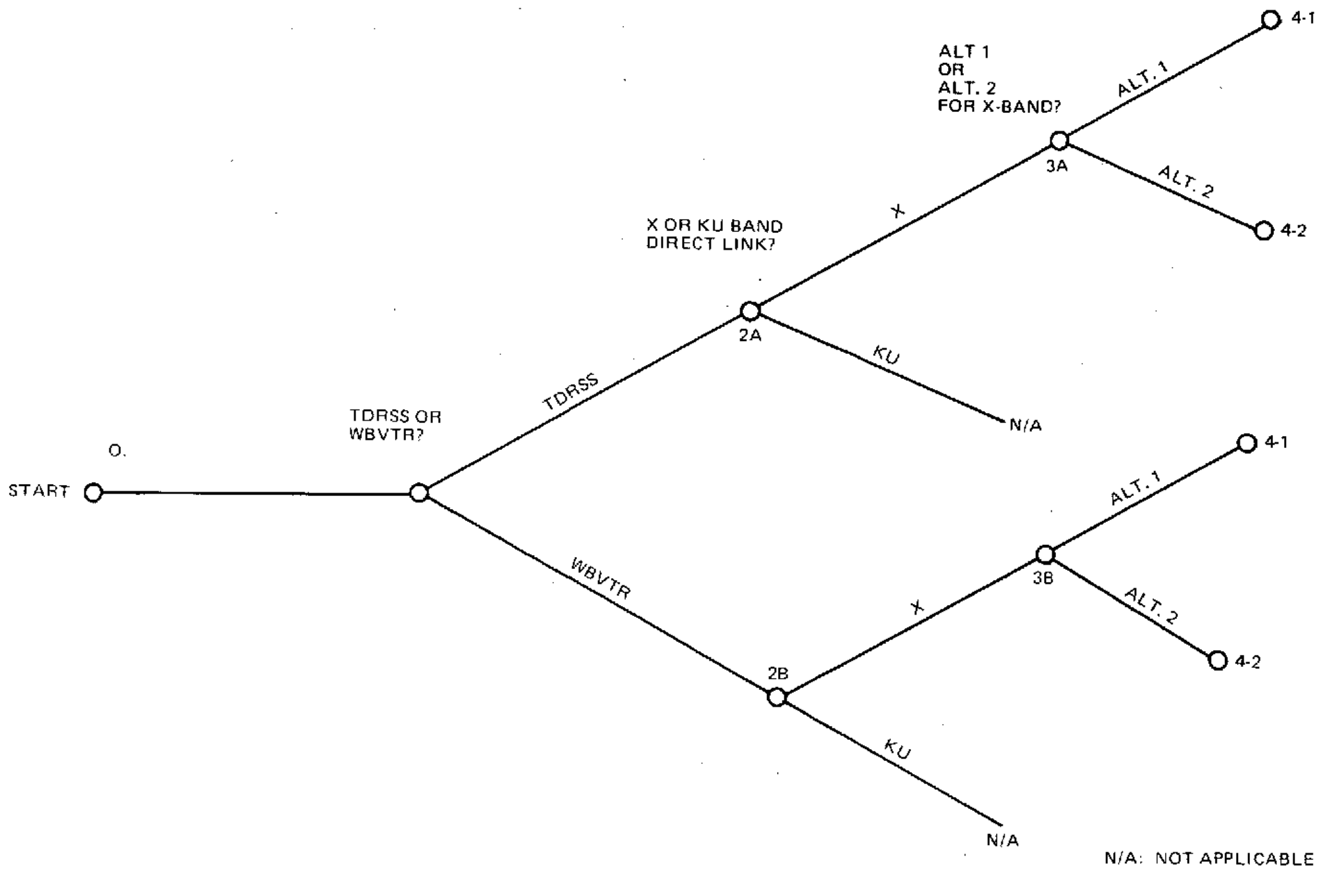

Fig. 4.4-8 Decision Tree

4.5 FOLLOW-ON INSTRUMENT/MISSION ACCOMMODATIONS

4.5.1 SUMMARY OF EOS SPACECRAFT WEIGHT/COST IMPACTS

The EOS A and A' mission spacecraft have a broad enough capability to accommodate the instruments and their mission-peculiars for the SEOS, SEASAT, and SMM missions with no major changes in the basic spacecraft design. Because of the diversity of orbit-on-orbit attitude, pointing requirements and instrument complements included in the above missions, some changes must be expected in even a flexible spacecraft design. However, the basic subsystem configuration remains intact for all these missions.

A tabular listing of these changes and their hardware costs is given in Table 4.5-1. Since these follow-on missions were not of primary concern during the first phase of this study, the impact areas indicated were determined using the reference data available to define the mission and instrument payloads. In cases where a clear definition was not available, reasonable assumptions were made by the instrument design group, based on the overall mission objectives, to establish a complete set of instrument requirements for each mission. To this extent, the impact areas identified in the table should be considered preliminary. In the next phase of the study the investigation of these follow-on missions will be continued. The impact areas indicated will be updated and the results included in the final report.

4.5.2 INSTRUMENT/MISSION DEFINITION

A listing of the instruments and their associated requirements is given in Table 4.5-2. The instruments for each mission have been included in the configurations shown in Section 4.5.3. A review of the impact of the driver follow-on requirements on each subsystem in the EOS A and A' spacecraft is also given in Table 4.5-1.

4.5.2.1 SYNCHRONOUS EARTH OBSERVATORY SATELLITE (SEOS)

The SEOS is a geosynchronous satellite designed to supplement earth observations made from lower-orbiting non-synchronous satellites, or from synchronous satellites with lower resolution. The area of observation for the spacecraft is considered to be the continental and coastal regions of the U.S. The SEOS will serve specific applications in the fields of:

- Earth resources
- Mesoscale weather phenomena
- Timely warnings and alerts of severe phenomena

SPECIFIC APPLICATION AREAS - Some of the applications which the SEOS may serve are:

- Earth Resources
  - Detection and monitoring of water-suspended solid pollutant
  - Estuarine dynamics and pollutant dispersal
  - Monitoring extent, distribution and change of snow cover
  - Detecting and monitoring of fish location and movement

Table 4.5-1 Follow-On Mission Weight and Cost Impact on Basic EOS Spacecraft

MISSION	STRUCTURE IMPACE AREA	WEIGHT	COST	ATTITUDE CONTROL SYSTEM IMPACT AREA	WEIGHT	COST	COMM. AND DATA HANDLING IMPACT AREA	WEIGHT	COST	ELECTRICAL POWER IMPACT AREA	WEIGHT	COST	PROPULSION IMPACT AREA	WEIGHT	COST	THERMAL IMPACT AREA	WEIGHT	COST
SEOS	• TITAN ADAPTER			• GIMBALLED STAR TRACKER	35#	$1.4M	• HIGH GAIN* ANTENNA	3.3#	$95K	• REDUCED POWER REQUIREMENTS COMPARED TO BASIC MODULE	ARRAY SAVINGS IN STRUCTURE	NONE	• ADDITIONAL FUEL REQUIRED TO PERFORM WHEEL UNLOADING	11.5#	NEG.	• EPS MODULE 4 VCHP	25#	$24K
	• TRANSITION RING	75#	$30K													• ACS MODULE 4 VCHP	25#	$24K
	• GIMBALLED STAR TRACKER MOUNT ON MODULE	2#	$10K	OPERATIONAL UPDATE OF GYROS TO MEET MOLDING REQ'TS	—	—	• COAXIAL SWITCH									• C & DH MODULE 4 VCHP	25#	$24K
	• REDUCE ARRAY SIZE TO 60 FT²	-60#	NO CHANGE COST SAVING IN HARDWARE EFFECT BY DESIGN EFFECT													• STRUCTURE HEATER POWER	3-8#	$23-53K
	• ADDITIONAL N_2H_4 TANK	+2.92#	$13K				• STAR TRACKER SOFTWARE & HARDWARE POSSIBLE	6#	$25K									
SEASAT	• SYNTHETIC APERTURE RADAR A PROBLEM DUE TO VOLUME LIMITS OF DELTA SHROUD, BUT NOT DUE TO EOS SPACECRAFT			CONTROL PROBLEMS POSSIBLE DUE TO ANTENNA MOTION DYNAMICS (WILL BE INVESTIGATED FURTHER)			NONE			• NEW SOLAR ARRAY DRIVE	NEG.	$300K	NONE			• EPS MODULE 4 VCHP & OSR	30#	$50K
	• INCREASE IN ARRAY SIZE	75#	$300K													• C & DH MODULE 4 VCHP & OSR	30#	$50K
							*MAY BE ELIMINATED BY DUAL FEED WITH WIDE BAND DATA OVER INSTRUMENT X-BAND ANTENNA ADDITIONAL											
SMM	• TRANSITION RING	75#	$30K	• GIMBALLED STAR TRACKER	35#	$1.4M	• ADDITIONAL STAR TRACKER SOFTWARE & POSSIBLE HARDWARE	6#	$25K	• ELIMINATE ARRAY DRIVE DUE TO SOLAR ATTITUDE	SAVINGS IN STRUCTURE	$100K	NONE			• EPS MODULE 4 VCHP & OSR	30#	$50K
	• GIMBALLED STAR TRACKER MOUNT	2#	$10K							• REDUCE POWER REQUIREMENTS						• C & DH MODULE 4 VCHP & & OSR	30#	$50K
	• REDUCE ARRAY SIZE	-55#																

FOLDOUT FRAME 1

FOLDOUT FRAME 2

- Detection and assessment of disease and insect damage to forest species
- Flood prediction, survey and damage assessment
- Determination of optimum crop planting dates
- Exploration of geothermal sources

• Weather Phenomena

- Detection, monitoring and prediction of thunderstorms and related tornadoes, hail, and excessive rainfall
- Detection, monitoring and prediction of tropical cyclones
- Predictions and monitoring of frost and freeze conditions

• Warnings and Alerts

- Floods
- Storms
- Frosts and freezes
- Fog

INSTRUMENT PAYLOAD - The prime payload for the SEOS will be a multispectral of Ritchey-Cretian Cassegrainian Telescope approximately 1.5 meter aperture. This telescope which is called the Large Earth Survey Telescope (LEST), will be used in conjunction with one or more of the following:

• Advanced Atmospheric Sounder and Imaging Radiometer (ASSIR)

• Microwave sounder

• Data Collection System (DCS)

• Framing camera

A brief description of the SEOS payload for those instruments which are somewhat defined follows:

• LEST

- 1.5 meter aperture
- Imaging in the visible and IR, and IR sounding (between 0.2um and 15 um)
- Multiband
- Theoretical ground resolution is 100 m in the visible and 800 m in the IR

- Microwave Sounder
  - 2.0 meter aperture or larger
  - 50 to 180 GHz
  - Four frequency bands
  - Theoretical ground resolution would be 200 km at 50 GHz and 50 km at 180 GHz
- Framing Camera
  - TV - type device
  - Resolution and coverage
    - o 216 meters per TV line for 1000x1000 km
    - o 45 meters per TV line for 200x200 km
  - Highlight range programmable for greater than 1000 times
  - A sequential color frame every 15 seconds
  - Limited radiometric capability

A summary of the remaining SEOS mission characteristics is given in Table 4.5-2.

4.5.2.2 SEA SATELLITE (SEASAT)

SEASAT is a low-altitude non-sunsynchronous earth-orbiting spacecraft that will fulfill the need for information on several oceanographic phenomena including sea state, currents, circulation, pileup, storm surges, tsumanis, air/sea interaction, surface winds, temperature and ice formations. The spacecraft will carry a complement of active and passive remote sensing instruments operating mostly at microwave wavelengths capable of all weather observations. The active facility performs the primary ocean dynamic measurements and the passive provides path length corrections for atmospheric water content. The payload also includes a visible/IR imager for high-resolution mapping of sea surface temperature and cloud cover and a laser reflectometer for tracking. The active microwave sensors have a capability for altimetry and wave directional spectrometry and a synthetic aperture capability for side-looking coherent imaging. The passive microwave sensors include radiometers operating at six bands and providing a capability of measuring atmospheric properties, sea ice, sea surface roughness and atmospheric attenuation to correct active scatterometer data.

4.5.2.3 SOLAR MAXIMUM MISSION (SMM)

The SMM is a low earth-orbit solar pointing satellite designed for solar observations during the period of maximum solar activity (expected about 1978). Its general mission objective is to make solar observations in all areas of the spectrum from IR to Gamma Rays and obtain data to supplement data acquired during the SKYLAB/ATM mission. The SMM will serve specific applications in the fields of:

- Solar flares
- Flare-associated X-and Gamma-radiation as well as high energy particles
- Solar interior to corona energy transfer
- Solar and Stellar evolution

INSTRUMENT PAYLOAD - The instrument payload of SMM is made up X-ray and UV spectrometers, spectroheliographs (images) spectrographs, and a coronagraph as described in Table 4.5-1. A summary of the remaining SMM mission characteristics is given in Table 4.5-2.

4.5.3 SUBSYSTEM DEFINITIONS

4.5.3.1 STRUCTURES AND CONFIGURATIONS

SYNCHRONOUS EARTH OBSERVATORY SATELLITE (SEOS) - The approach is to meet all of the SEOS requirements defined by GSFC instrument definition table (Reference II. D. 3_ and in the MSFC, "Payload Discriptions - Volume I - Automated Payloads", (October 1973), and to use the EOS-A modular design which involves minimum risk and cost growth.

The following constraints and guidelines were used to implement this approach:

- The spacecraft is sized for a Titan III C-7 launch from ETR into a circular orbit altitude of 19323 n. mi. at an inclination of 0 degrees. The nominal orbit positioning will be 96° west longitude (geostationary equatorial over CONUS)
- Existing spacecraft system technology is used.
- The spacecraft is designed so that the EOS subsystems can be fabricated, tested and integrated independently and are as identical to EOS-A and-A' as possible.
- No on-orbit servicing or retrieval is planned.

INSTRUMENT PAYLOAD REQUIREMENT - The following complement of instruments and associated equipment are carried to fulfill the SEOS program objectives:

- Large Earth Survey Telescope (LEST)

Table 4.5-2 EOS Follow-On Mission Instrument Definition

MISSIONS/ INSTRUMENTS	WEIGHT LBS	POWER W	BAND	RESOL. M	SWATH, KM OR FOV	DATA RATE	ATTIT. CNTRL POINT/ JITTER	SPECIAL COOLER	APERTURE DIAM M	POINTING	VOLUME CU. Ft.
I. SEOS (TITAN/ SHUTT. L'V.)											
1. LEST (PRIME)	2,300	100 AV	VISIBLE IR	100 800	3,000	50 MB/S	.0016/	NONE PASSIVE	1.5		
2. MW SOUNDER			50 GHz 180 GHz	200,000 50,000		1	.85 SEC FOR 2 SEC &	NONE	2		
3. FRAMING TV CAMERA			.4-9NM	216 45	$1{,}000^2$ 200^2	6 MB/S	2 SEC for 20 MIN	NONE			
TOTAL (INCL. S/C)	3,500 TO 6,600	600 TO 750				60					360
II. SEASAT-A											
1. ALTIMETER	100	125	13.9 GHz	3 SEC		.5 KB/S	$.5^{\circ}$	NONE	1	NADIR	.71
2. MW SCATTER-	200	86	13.9 GHz			2 KB/S	.5	NONE	5.3 (STICKS)	FWD	58.3
3. MW RADIO-meter	110	65	6.6 GHz 10 GHz 18 GHz 22 GHz 37 GHz		866 866 866 866 866	4 KB/S	.5		1.25	38° OFF NADIR NADIR NADIR NADIR	
4. VISIBLE/ IR RADIO-METER	20	8	VISIBLE IR	2.5-7.5KM 5-7.5KM	2,000 2,000	50 KB/S	.5	NONE PASSIVE		NADIR	
5. SAR	88	200	1.7 GHz	25	100	.5 TO 16 MB/S		NONE	17 M^2	SIDE LOOKING	
66. LASER REFLECTIOM.	20			100	200			NONE			
TOTAL (PAYLOAD ONLY)	538	484				56 KB/S W/O SAR				SUN POINTING AREA-FT^2	
III. SMM* (DELTA L'V.)											
1. UV MAGNETO-GRAPH	100	20	.11-22NM		2°	.5 KB/S	$.0014^{\circ}$	--	7 x 10 (INCHES)	.50	3.00
2. EUV SPECT-	100	20	0.02-.07		2o	1.0	.0014	--	10 x 10	.70	4.20
3. HI-RESOL. X-RAY SPECTROM.	100	15	1.25 A		5°	.35	.0014	--	7 x 10	.50	3.25
4. HARD X-RAY IMAGING	100	10			5	.200	.0028	--	6 x 5	.20	1.30
5. X-RAY POLARIMTR	16	10			5	.400	.0168	--	8 x 8	.45	1.35
6. GAMMA RAY DETECTOR	200	12			20	.5	1°	--	18 x 18	2.25	6.75
7. X-RAY SPECTRA-METER	70	12			20	.5	1°	--	12 x 12	1.00	3.00
8. X-RAY DETECTOR	20	5			10	.2	1°	--	12 x 12	1.00	1.00
9. CORONO GRA	100	10			20	.5	.0336	--	5 x 12	.42	2.52
10. UV SPEC-ROMETER	110	20			2	.5	.0014	--	8 x 12	.67	4.00
11. NEUTRON DETECTOR	205	15			20	.2	10	--	10 x 20	1.40	4.20
12. H-POTOMTR	20	10			2	.125	.0014	--	4 x 4	.11	.33
13. FLARE FINDER	30	10			2	5	.0028	--	4 x 4	.11	.67
TOTAL	1,171	169				10				9.30	35.57

*All experiments (1 to 13) must be accomdated simultaneously on SMM.
Ref. GSFC Report X-703-74-42 Jan 74.

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- Advanced Atmospheric Sounder and Imaging Radiometer (AASIR)
- Multibanned Microwave Sounder
- Data Collection System (DCS)
- Framing Camera (assumed part of LEST)

INSTRUMENT ACCOMMODATIONS - The pertinent spacecraft interface characteristics of these instruments are listed in Table 4.5-2. A configuration of a modular EOS-A configuration accommodating these instruments is shown in Figure 4.5-1. Our study indicates that the primary instrument, the LEST, fits remarkably well into the EOS A spacecraft. The back focal surface area can be fit quite nicely within the triangular area between the modular structure. The primary mirror support for equal mirror loading during launch can be interfaced easily with the slightly larger spacecraft structure by a transition ring which picks up the primary load points on the spacecraft. The Delta sized spacecraft would require a tapered adapter to join the larger Titan diameter. It is expected that this adapter will be non-standard and thus it is a structural impact.

All the other instruments represent only the normal packaging problems which would exist for any SEOS design. One possible problem may be the mounting of the gimballed star tracker. Mounting it on the ACS module might impose a thermal constraint on the module. At first cut it seems that this problem is not too severe. However, it will be investigated further during the next phase of the study and alternative mounting will be provided if required.

SEA SATELLITE (SEASAT) - The approach is to meet all of the SEASAT-A requirements defined in the GSFC Phase A Study Report and to use a modular EOS design that involves minimum risk and cost growth.

The following constraints and guidelines are used to implement this approach.

1. The spacecraft is sized to be Delta launched into a 700 to 800 km altitude, 82° inclination non-sun-synchronous orbit, carring the required complement of instruments defined below.
2. Existing spacecraft system technology is used.
3. The spacecraft is designed so that modularized EOS subsystems can be fabricated, tested and integrated independently.
4. The number of deployable structures is minimized.

INSTRUMENT PAYLOAD REQUIREMENT - The following complement of instruments and associated equipment is carried in order to fulfill SEASAT-A program objectives.

1. Altimeter
2. Scattermeter - Spectrometer
3. Microwave Radiometer
4. Very High Resolution Radiometer (VHRR)
5. Synthetic Aperture Radar (SAR)
6. Laser Reflectometer
7. Satellite to Satellite Tracking Antenna
8. Data Collection System

INSTRUMENT ACCOMMODATION - The pertinent spacecraft interface characteristics of these instruments and their logistic requirements are listed in Table 4.5-2. A configuration of a modular Deltaborne EOS accommodating these instruments is shown in Fig. 4.5-2. The study indicates that the integration of the SEASAT payload with the EOS platform does not present a severe packaging problem except for the volume required by the SAR antennas and the clearance required by the VHRR cooler. It may be necessary to make the SAR antennas deployable. The VHRR cooler problem is complicated by the non-sun-synchronous nature of the SEASAT orbit. Possible solutions to this problem include cryocooling, rotation of spacecraft, a two-cooler system, or a reduced duty cycle. Additional studies are needed. Other study areas include the rotational and pointing requirements of the microwave antennas which give rise to important dynamic problems in the control of the spacecraft.

SOLAR MAXIMUM MISSION - The approach has been to meet all of the SMM requirements defined by the GSFC report X-703-74-42, Solar Maximum Mission (SMM) Conceptual Study Report, dated January 1974, and to use an EOS-A design which involves minimum risk and cost growth. (See Fig. 4.5-3)

The following constraints and guidelines are used to implement this approach:

- The spacecraft will be launched on a Delta Vehicle. Subsequent retrieval and redeployment is planned for the Shuttle. The nominal orbit is 275-300 n.mi., circular, at an inclination of 28-33 degrees.
- Existing spacecraft system technology is used.
- The spacecraft is designed so that modularized EOS subsystems can be fabricated, tested, and integrated independently and are as identical to EOS A and A' as possible.

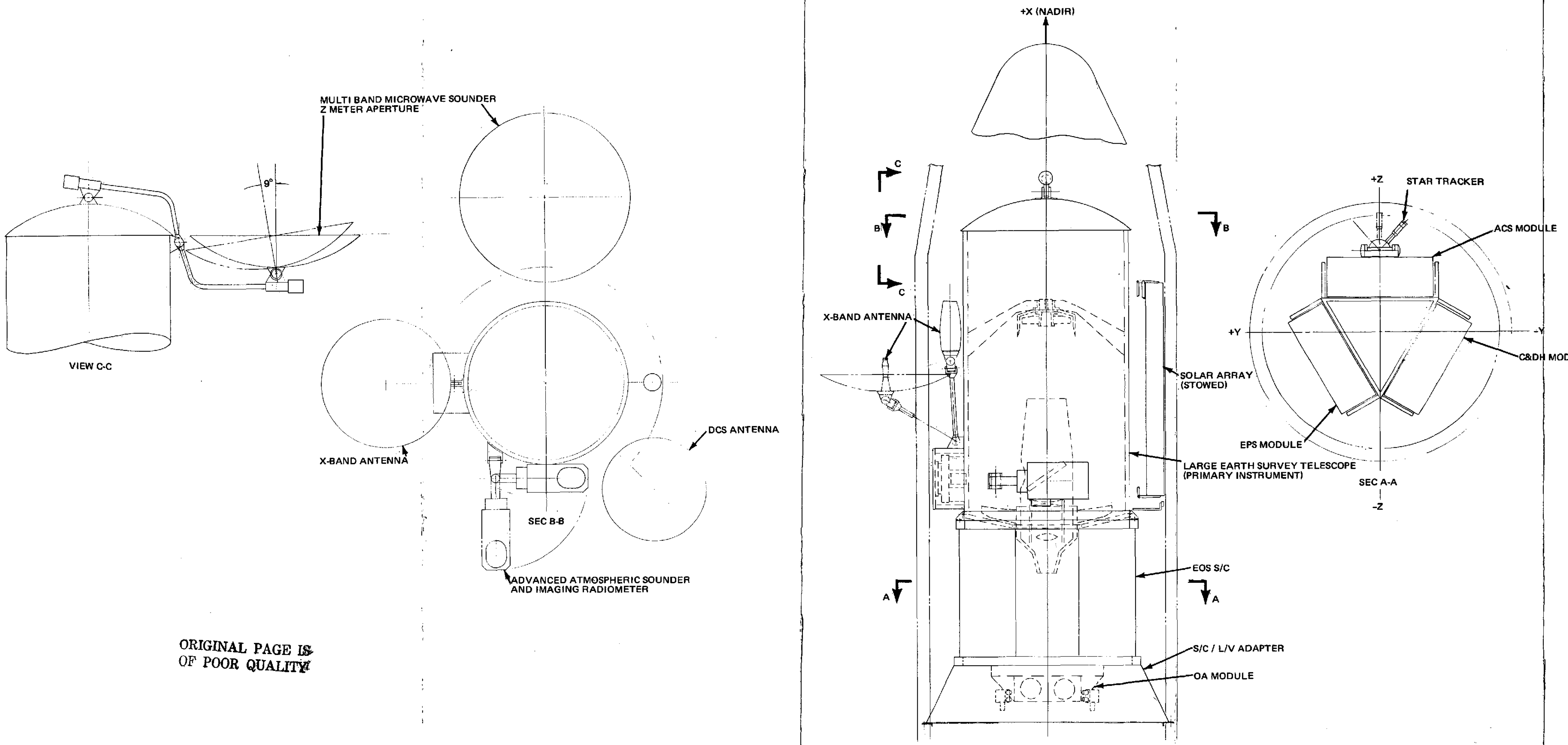

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Fig. 4.5-1 EOS/SEOS Configuration

3

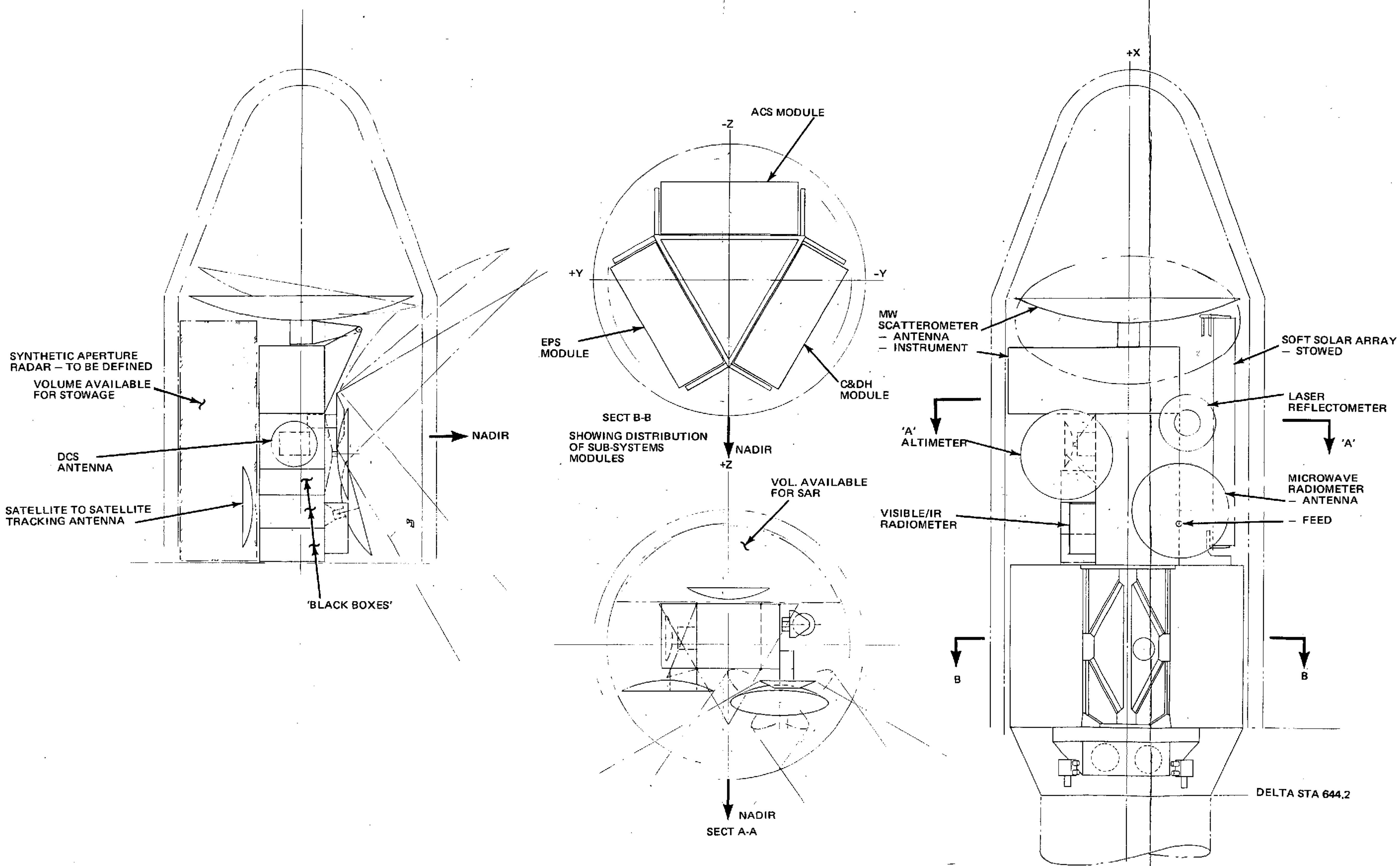


Fig. 4.5-2 EOS-SEASAT Configuration

FOLDOUT FRAME 1

FOLDOUT FRAME 2

INSTRUMENT PAYLOAD REQUIREMENT - The following complement of instruments and associated equipment is carried in order to fulfill the SMM program objectives:

- UV Magnetograph
- Hi-Resolution X-Ray Spectrometer
- Hard X-Ray Imager
- X-Ray Polarimeter
- Gamma Ray Detector
- X-Ray Spectrometer
- X-Ray Detector
- Coronagraph
- UV Spectrometer
- Neutron Detector
- H-Photometer
- Flare Finder

Instrument Accommodation - The pertinent spacecraft interface characteristics of these instruments is given in Table 4.5-2. A configuration of a modular Delta-borne EOS accommodating these instruments is shown in Fig. 4.5-3. The study indicates that the SMM payload can be packaged well using the EOS-A basic spacecraft. Two alternative payload configurations are possible. The first, which is shown in the drawing, has the instruments mounted transverse to the spacecraft axis. This configuration is basically an EOS with the instruments always pointing at the sun rather than at the earth. The advantages of this configuration include the ability to easily mount the required gimballed star tracker close to the instruments thus reducing alignment and distortion problems, the fact that the instrument integration orientation relative to the spacecraft axis is the same as EOS-A, and the ability for additional payload growth within the delta shroud.

The other alternative locates the instruments along the spacecraft axis in a cylindrical or box structure. This configuration has thermal advantages and allows for growth in individual instrument length but is constrained in the number of additional instruments which can be added.

In either case the instruments can be easily mounted to the spacecraft hardpoints via a transition ring. The constant solar pointing altitude required for the mission allows for a reduction in array size and elimination of the array drive.

4.5.3.2 ATTITUDE CONTROL SYSTEM

In order to meet the most severe pointing accuracy requirements for the follow-on missions, a gimballed star tracker is required. These requirements result from the SEOS (.0016) and the SSM (.0014). As can be seen from Fig 4.3-1, the addition of this star tracker will reduce the pointing error of the baseline system of approximately .01 deg to that of the expanded capabilities system, approximately .002 deg. However, this addition increases the ACS module cost by:

$ 950K Recurring
470K Non-Recurring

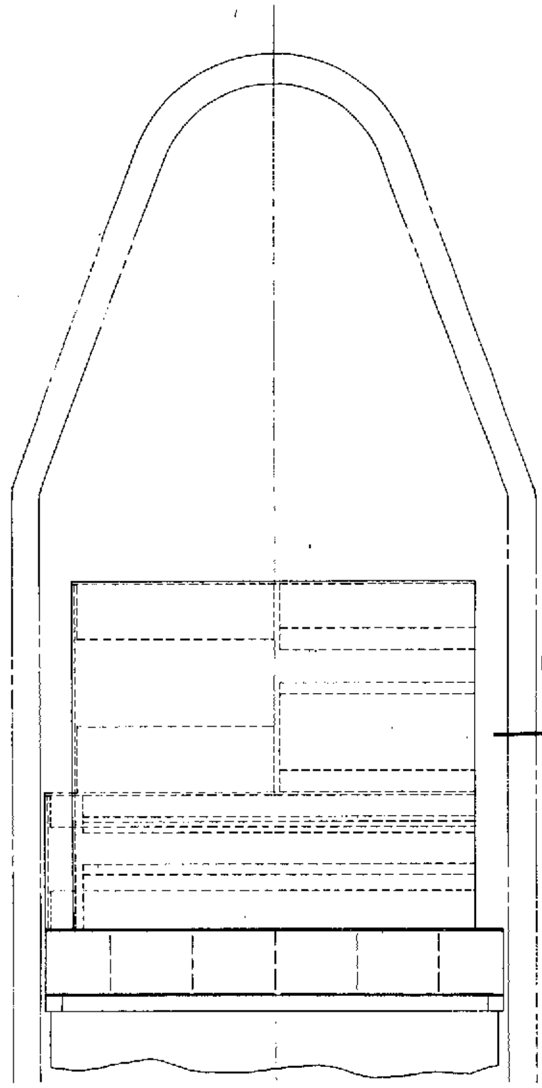
$ 1,420K Total

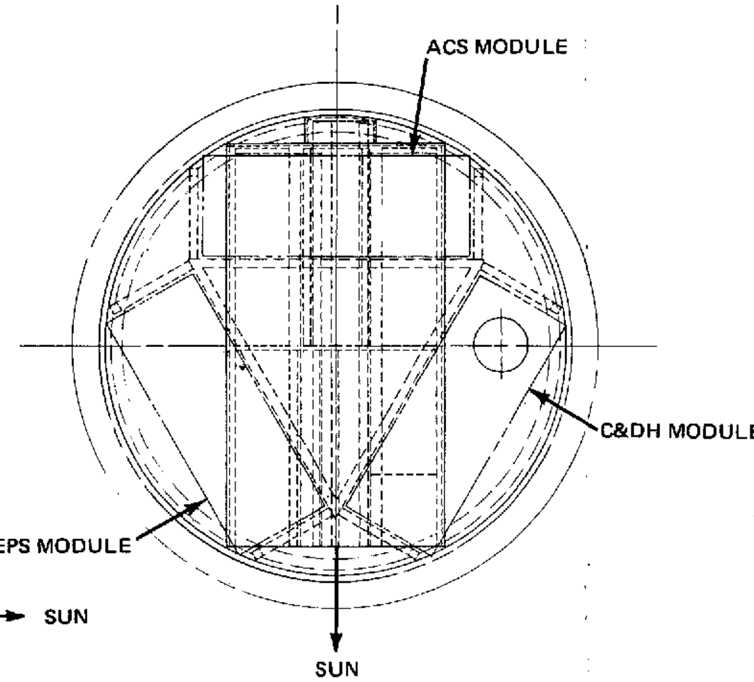
or about $1.5M for the hardware cost of a single unit.

The critical holding requirement for the follow-on missions results from SEOS. One of the SEOS application objectives is the observation of mesoscale weather phenomena such as early warning of developing tornadoes and other storms. These phenomena are on the scale of 1 km, which translates to a pointing requirement of 0.0016° (6 or 28 μrad). In order to track the progress of these storms, the SEOS may be required to hold this pointing to 2 arc-sec for 20 minutes of picture-taking as well as 0.85 arc-sec for a 2 second exposure. With the state-of-the art gyro which is proposed, we can expect a random drift of .003 arc-sec/sec over a 30 minute time interval, a jitter of 1 arc-sec over 30 seconds or less, and a jitter of 2 arc-sec over 20 minutes. Thus the requirement can be met during the 2-second exposure time and the 20 minute picture-taking session.

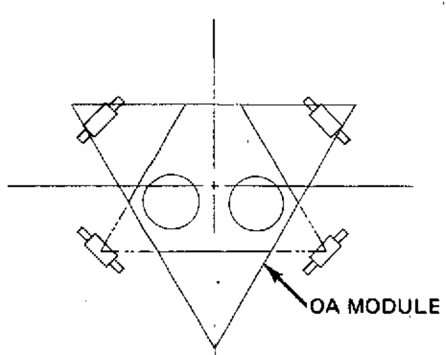
4.5.3.3 COMMUNICATIONS AND DATA HANDLING

The communications necessary to handle 50 Mbps of SEOS at bit error rate of 10^{-5} will require an increase in transmitted EIRP of approximately 10 dB over the baseline X-band communications for the 678 km EOS mission (to approximately 41 dBW). This assumes a ground antenna of 30 ft with an effective noise temperature of 166°K and results in about 6dB margin. This increased EIRP can easily be provided by increasing the X-band S/C antenna size to about 5 ft with a 4 watt transmitter. This size should fit easily within the launch vehicle shroud. A 5 ft antenna would have a beamwidth of approximately 1.6 deg and would therefore not provide full earth coverage from synchronous altitude. An ability to point the S/C antenna through approximately ±9 deg





SMM
INSTRUMENT PAYLOAD
1 UV MAGNETOGRAPH
2 EUV SPECTROMETER
3 HI-RESOL. X-RAY SPECTROM
4 HARD X-RAY IMAGING
5 X-RAY POLARIMTR
6 GAMA RAY DETECTOR
7 X-RAY SPECTRAMETER
8 X-RAY DETECTOR
9 CORONOGRAPH
10 UV SPECTROMETER
11 NEUTRON DETECTOR
12 H-POTOMTR
13 FLARE FINDER
E INSTRUMENT ELECTRONICS



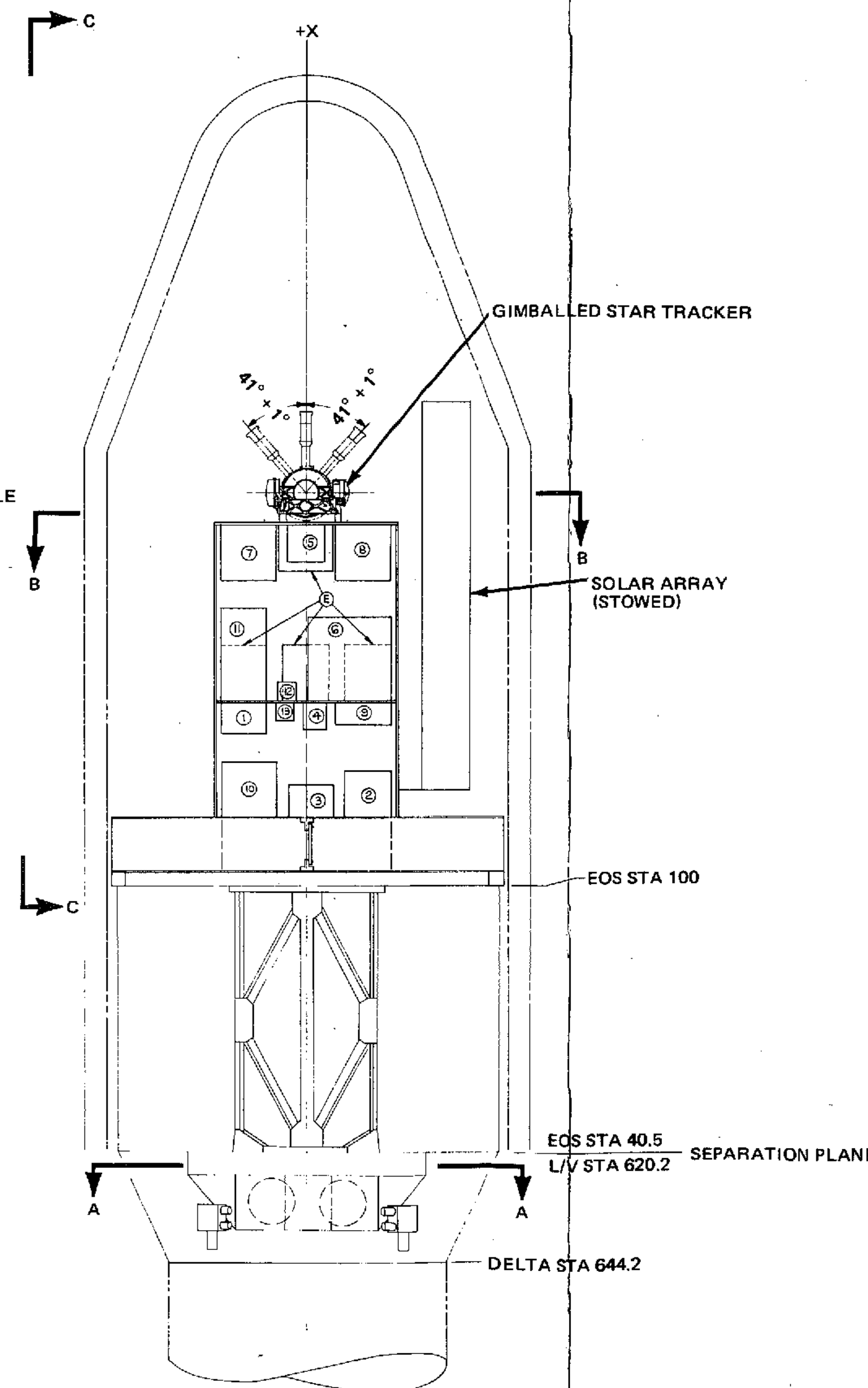


Fig. 4.5-3 EOS Solar Maximum Mission Configuration

FOLDOUT FRAME 1

FOLDOUT FRAME 2

is required. The antenna size could be reduced with commensurate increases in the transmitter power, however this would not eliminate the requirement for a steering capabiltty.

COMMUNICATIONS - The selected communications configuration, consisting of two shaped beam antennas and an S-band transponder assembly with an integrated hybrid coaxial switch and two diplexers, will satisfy the basic communication requirements for telemetry, tracking and command compatibility with STDN for the SEASAT and SMM EOS missions. The SEOS mission, however, will require a high gain, earth coverage, antenna. A candiaate antenna type is a short backfire antenna mounted on a four-foot boom in order to minimize blockage from the LEST payload. This antenna provides 15 dB gain and has a 34 degree half-power beamwidth. It is 11 inches in diameter by 2.7 inches in height and weighs 3.3 pounds. The existing S-band transponder can satisfy the downlink transmitter RF power requirements since medium rate data from the computer or tape recorder is not required and the low rate data can be transmitted at the high output power (2 watts) transpoder mode of operation. If medium rate data is determined in the future to be a requirement, a 6 to 10 dB increase in transmitter power will be required. A coaxial switch will also have to be utilized to switch from the low gain broad beam antenna (ascent/orbit insertion/transfer orbit) to the high gain antenna (synchronous orbit).

The changes to the communications configuration for the SEOS mission will result in the following hardware costs:

	Non-recurring	Recurring	Total Cost
• High Gain Antenna	\$70K	25K	\$ 95K
• Coaxial Switch	\$ 6K	2K	\$ 8K
			Total 103K

DATA HANDLING - The EOS Data Handling Group (DHG) is capable of executing payload comands in both real time or delayed time. In addition, it is capable of acquiring and routing for telemetry (or optional onboard recording) housekeeping and low bit rate scientific data (less than 32 kbps). The full duplex multiplex data bus system is capable of interfacing with up to 32 remote units (multiplexers and decoders) which can have 64 input and 64 output channels each. The system provides the flexibility required for handling varied payloads without impact to the basic EOS data handling design.

At present the basic spacecraft utilizes five of these remote units while two more are dedicated for utilization by the EOS, A, A', B, C, D and E mission payloads. These two remote units are also capable of satisfying the payload housekeeping data handling requirements of the advanced missions such as SEOS, SEASAT and SMM. If more remotes are required they can be added without impact to the EOS basic Data Handling Group.

The addition of the ACS gimballed startracker to the ACS subsystem for the SEOS and SMM may increase the required AOP software. The AOP's main memory is expandable in 8K word modules to 65K words. Since present EOS missions require only 23.3K words, any software increase greater than 1 K words may be accommodated by adding additional 8K word memory modules.

4.5.3.4 ELECTRICAL POWER

The basic impact on EPS for all missions is in the mission peculiar solar array and quality of batteries. For the missions described herein the following figures apply.

- SEOS - The combination of a load of 425 watts and synchronous altitude reduces the solar array area to 60 ft instead of 125 ft for the baseline. These changes will result in a weight reduction of 60 lbs.

No net increase in price, reduction of hardware costs are offset by a design effort.

- SEASAT - This type of mission will require an increase in solar array area because of a power increase. Also, the mission-peculiar solar vectors will require a two-degree of freedom rotation for the solar drive. The solar drive mechanism can be considered a new item with an estimated cost of non recurring and recurring of $300K. The total increase in cost above the baseline is $600K witha weight increase of 75 lb.

- SMM - This mission is pointing toward the sun continually, thus eliminating the need for a rotable solar array. Also, total power is reduced by approximately 100 watts. The net effect results in a cost reduction of $100K and a weight reduction of 55 lb.

4.5.3.5 THERMAL

A preliminary evaluation was made of the impact of future missions on the thermal design of EOS. The future missions considered were SEASAT (non-sun-synchronous), SEOS (geo-synchronous), and SSM (solar pointing). An estimate was made of the external heat flux for each of the above mission orbits. The heat rejection capability of the modules for the Delta 1 configuration was analyzed and the impact on the structure temperature control was considered. Module heat rejection capability was studied parametrically as a function of design approach and temperature level. For cost/weight impacts, a module hot case heat rejection capability of 150 watts and an operating temperature range of 70° F $+20^{\circ}$F was assumed. In general, the orbits of the future missions considered, result in significant impact on thermal control when compared to the benign environment of the LRM, sun-synchronous orbit.

SEOS MISSION - The SEOS mission is geo-synchronous. This orbit reduces module heat rejection capability and causes large changes in orbital environment over a 24 hour period. Detector cooling, if required, should not be a significant problem. The cost and weight impact of thermal control is as follows:

Item	Control	Cost Increase	Weight Increase
EPS Module	4 VCHP	$24K	25lb
ACS Module	4 VCHP	24K	25lb
C&DH Module	4 VCHP	24K	25lb
Structure	Heater Power	23-53K*	3-8 lbs
		$95K-125K	78-83 lbs

* Depending on Array

SOLAR MAXIMUM MISSION - This solar pointing mission reduces heat rejection capability (solar load), but the orbital swings in environment are not as severe as the other orbits considered. The cost weight impact is estimated as follows:

Item	Control		Cost Increase	Weight Increases
EPS Module	4VCHP		$24K	30lb
ACS Module	Passive		-	-
C&DH Module	4VCHP		24K	30lb
Structure	Passive		-	-
		Total	$48K	60lbs

SEASAT MISSION - The SEASAT mission is non-sun-synchronous. In this orbit, the sun moves relative to the spacecraft, exposing all spacecraft surfaces to the sun (over a period of time). This causes reduction in heat rejection capability and large swings in environment from hot to cold. As a result, active control and in some cases expensive stable coatings (such as optical solar reflector) will be required to achieve thermal control. The cost and weight impact for thermal control estimated as follows:

Item	Control	Cost Increase	Weight Increase
EPS Module	4VCHP & + OSR	$50K	30lb
ACS Module	Passive	-	-
C&DH Module	4VCHP & +OSR	50K	30lb
Structure	Passive	-	-
		$100K	60lbs

It should also be noted that instrument detector cooling will be a significant problem. The design of a passive cooler alone appears prohibitive. A passive cooler with supplemental cooling (i.e.: cryogen) or preferred spacecraft orientation are possible solutions.

4.5.3.6 REACTION CONTROL/ORBIT ADJUST SUBSYSTEM

The RCS/OAS is capable of performing the SEASAT-A and SMM missions with no changes to the subsystem. This is not the case for the SEOS mission. At synchronous altitude the magnetic unloading system (MUS) is unable to completely unload the reaction wheels because of the weakness of the earth's magnetic field. The RCS must therefore perform the wheel unloading function. The tankage in the subyystem as presently designed is filled to capacity. A tank must be added to provide the additional capacity to handle the increased wheel unloading operation. The addition of one tank provides the capability to perform 100% of the wheel unloading. The tank capacity is 11.5lb of N_2H_4, its weight is 2.92 lb. The cost penalty is $13K.

4.6 GROUND SUPPORT EQUIPMENT

The requirements for GSE depend upon the EOS design and the test program necessary to place the EOS into orbit. For the system definition study, it was found that alternate spacecraft configurations had little impact in the type and quantity of GSE required. The major drivers were the tests to be performed, the time element for test results, the manner in which the test would be conducted and the supporting equipment required for handling the spacecraft.

Except for instrument operation while installed in the spacecraft, all instrument GSE was assumed to be provided by the respective instrument vendor.

4.6.1 GSE REQUIREMENTS

4.6.1.1 Spacecraft Level

Real time analysis of S/C data and operation, less sensor outputs, is required at all test sites. The real time analysis permits instant evaluation of S/C anomalies which may be corrected as test progresses with minimum down time. This results in lower test costs and a shorter test period. In order to perform real time analysis of the S/C during test, uplink and downlink communications is required. This results in a test station with an RF front end capability, which has as a result, the added benefit of use during S/C pre-launch and launch test at VAFB, the primary launch site.
All S/C equipment is required to operate during test. Where equipments do not receive interface signals as a result of non-orbital operation, such as the solar array sensor, stimulation is required. The stimulation need not be to orbital levels in all cases, but enough to provide an output sufficient to verify equipment operation. Provision may also be made in the software program of the test station to account for the reduced output for non-orbital level stimulation.

Real time analysis of vehicle performance during thermal vacuum tests of the S/C is required. This analysis will not require any addition to the test station which will be used for nominal S/C testing. Interface cabling with a thermal vacuum chamber will be required and consitutes additional equipment.

During vibration and thermal vacuum tests, additional instrumentation will be added to the S/C (i.e., thermal and vibration sensors). These signals are not part of the normal downlink housekeeping data and will require monitoring during the test.

Spacecraft power will be on during testing. In order to conserve flight batteries (lower cots), ground power, simulating the flight battery characteristics, must be provided for the S/C. The capability must also be provided to simulate the S/C load profile.

Prior to installation of the flight batteries they must be conditioned by discharge and charge cycles.

Assembly of the S/C will be in the vertical position, but its movement inter-and intra-site will be in the horizontal position. Transportation dollies, handling equipment and work stands must accommodate this requirement. In addition, environmental protection must be provided during S/C movement.

All S/C pyro signals, in response to pyro initiation commands, must be simulated during test.

Spacecraft tanks require actual fluid or pneumatics or, when this is not practical, a simulated but inert substitute to the same cleanliness level required for flight. Handling of such fluids should not cause its contamination or contamination to S/C tanks and lines.

4.6.1.2 SPACECRAFT MODULE LEVEL

With a modular approach to S/C assembly, bench checkout of the modules is now required in order to provide for the integration of the equipment within a module. This provides for an overall cost reduction by removing a large portion of S/C checkout from the period of time when S/C integration delay is costly. Each module now is complete and operable in itself, given S/C interface simulation, including appropriate power profiles.

The module checkout benches should be provided with sufficient flexibility so that they can be used as module maintenance benches during the Shuttle operational period.

4.6.2 EQUIPMENT IDENTIFICATION

A review and analysis of the test trade study report summary, combined with the top level GSE requirements has resulted in the identification of the GSE for EOS to a sufficient depth to permit its costing. These equipments have been divided into three categories, electrical, mechanical and fluid. A review of the test schedule revealed no interference between S/C production and test. As a result, only one of a GSE end item is required. The items with an asterisk have been identified as candidates for added quantities in the event a follow-on mission scenario calls for more than one launch per year.

4.6.3 GSE vs. SPACECRAFT ACTIVITIES

A review was made to determine what GSE would be required during each spacecraft activity from assembly to launch. The results are presented in Fig 4.6-1. The chart also reveals the multiple use being made of the GSE which is moved with the vehicle.

DEVELOPMENT & QUAL

	MODULE FAB & ASSEMBLY	MODULE MASS REP. & INSTALLATION	MODULE STRUCT. DEV & ACOUSTIC QUAL	MODULE THERMAL VACUUM DEV. TEST	S/C STRUCTURE FAB & ASSEMBLY	MODULE & INSTR MASS REP. INSTALL.	STRUCTURE LOADS ACCEL TEST	ACOUSTIC TEST (LRM CONFIG)	MECHANICAL VIBRATION & SHOCK	MODEL SURVEY	RECONFIG INSTR MASS REPS	MODEL SURVEY	MECHANICAL VIBRATION & SHOCK	ACOUSTIC TEST	SEPARATION & DEPLOYMENT QUAL TEST	MODULE WIRING / COMPONENT INTEGR	INST INTEGR & OBS FUNCT TEST	ACOUSTIC VIBRATION / PYRO SHOCK	THERMAL VACUUM, INCLUDING EMC	POST T/V SEP TESTS
ELECTRICAL																				
1. TEST & INTEGRATION STATION																•	•	•	•	•
2. BREAKOUT BOX SET																•	•	•	•	•
3. BATTERY CONDITIONER																				
4. TEST BATTERY SET																	•	•	•	•
5. S/C POWER SET & CABLES																	•	•	•	•
6. RANGING TEST ASSEMBLY																•				
7. PYRO TEST SET																		•		
8. INTERFACE CABLE SET																	•			
9. SOLAR SIMULATOR																	•			
10. INSTRUMENT INTERFACE SIMULATOR																	•			
11. UMBILICAL SIMULATOR																	•			
12. DITMICO PROGRAM & CABLE SET																•				
13. POWER MODULE C/O BENCH																•				
14. C & DH MODULE C/O BENCH																•				
15. ACS MODULE C/O BENCH																•				
16. PROPULSION C/O BENCH (RCS)																•				
17. DCS SIMULATOR																	•	•	•	•
18. S/C MONITOR & CONTROL																	•	•	•	•
19. IMP MODULE C/O BENCH																•	•			
MECHANICAL																				
1. INTERFACE ADAPTER SET							•	•	•	•	•	•	•	•	•			•	•	•
2. HOIST BAR & SLING SET	•	•			•	•	•	•	•	•	•	•	•	•	•		•			
3. SUPPORT DOLLY VERTICAL S/C					•	•				•		•			•		•			•
4. SUPPORT DOLLY HORIZONTAL							•	•	•				•	•				•	•	
5. ACCESS WORK STAND - S/C					•	•											•			•
6. SKIN STORAGE RACK																				
7. WT & CG FIXTURE S/C						•														
8. MASS SIMULATOR SET		•				•														
9. SUPPORT DOLLY S/C MODULES			•	•												•				
10. SHIPPING CONTAINERS S/C MODULES																				
11. S/C COVER SET																				
12. HUMIDITY CONTROL KIT																				
13. SHIPPING CONTAINER - SOLAR ARRAY																				
14. SOLAR ARRAY INST & DEPLOY. FIX															•					
15. TRANSPORTERS																				
16. INDICATING ACCELEROMETER KIT																				
17. PYRO INST. TOOL KIT																	•			
18. STAGE MOTOR INST FIXTURE																				
19. EOS/SHUTTLE SIMULATOR ADAPTER															•					
20. TIE DOWN KIT SET																				
21. BATTERY SHIPPING CONTAINER																				
22. BATTERY INSTALLATION TOOL																				
FLUID																				
1. GN_2 CONDITIONING UNIT																	•	•	•	•
2. GN_2 REGULATION UNIT																	•	•	•	•
3. VOLUMETRIC LEAK DETECTOR																	•	•		•
4. RCS VACUUM TEST CART																	•	•		
5. FLUID DIST SYS, GRUMMAN																	•	•		
6. PRESSURE MAINTENANCE UNIT																				
7. GN_2 STORAGE SYS (TRANSPORTER)																				
8. GN_2 MANIFOLD & SUPPLY PLATFORM																				
9. FLUID DIST SYS, LAUNCH SITE																				
10. PROPELLANT TRANS ASSEMBLY																•			•	
11. MASS SPECTROMETER LEAK DETECTOR																	•			

FLIGHT S/C EGR

	MODULE FAB ASSY & WIRING	COMPONENT INSTALLATION & INTEGR	S/S FUNCTIONAL C/O	MODULE ENVIRONMENTAL ACCEPT. TEST	POST ENVIRONMENTAL TEST MODULE C/O	S/C STRUCTURAL FAB & ASSY	S/C WIRING & C/O	S/C MODULE INTEGR & FUNCT C/O	INSTR INTEG & C/O, IMP & OAS INTEGR	OBS SYS TEST & EMC	SEPARATION & DEPLOYMENT MECH C/O	OBS SPECIAL TEST	WORKMANSHIP ACOUSTIC, PACK & SHIP	PRELAUNCH OPERATIONS	LAUNCH
ELECTRICAL															
1. TEST & INTEGRATION STATION		•	•					•	•	•	•	•	•	•	•
2. BREAKOUT BOX SET		•	•	•				•	•	•		•		•	
3. BATTERY CONDITIONER										•				•	
4. TEST BATTERY SET								•	•	•	•			•	
5. S/C POWER SET & CABLES								•	•	•	•	•	•	•	
6. RANGING TEST ASSEMBLY		•													
7. PYRO TEST SET										•					
8. INTERFACE CABLE SET								•	•						
9. SOLAR SIMULATOR									•			•			
10. INSTRUMENT INTERFACE SIMULATOR								•	•						
11. UMBILICAL SIMULATOR										•				•	
12. DITMICO PROGRAM & CABLE SET							•								
13. POWER MODULE C/O BENCH		•	•	•	•										
14. C & DH MODULE C/O BENCH		•	•	•	•										
15. ACS MODULE C/O BENCH		•	•	•	•										
16. PROPULSION C/O BENCH (RCS)		•	•	•	•										
17. DCS SIMULATOR								•	•	•				•	
18. S/C MONITOR & CONTROL								•	•	•	•	•	•	•	
19. IMP MODULE C/O BENCH		•	•	•	•				•						
MECHANICAL															
1. INTERFACE ADAPTER SET										•	•		•	•	•
2. HOIST BAR & SLING SET				•				•		•	•		•	•	•
3. SUPPORT DOLLY VERTICAL S/C						•	•	•	•	•	•	•		•	
4. SUPPORT DOLLY HORIZONTAL										•			•		
5. ACCESS WORK STAND - S/C						•	•	•	•	•		•			
6. SKIN STORAGE RACK						•	•	•	•						
7. WT & CG FIXTURE S/C										•					
8. MASS SIMULATOR SET															
9. SUPPORT DOLLY S/C MODULES	•	•	•	•	•			•							
10. SHIPPING CONTAINERS S/C MODULES													•		
11. S/C COVER SET													•		
12. HUMIDITY CONTROL KIT													•		
13. SHIPPING CONTAINER - SOLAR ARRAY													•		
14. SOLAR ARRAY INST & DEPLOY. FIX											•				
15. TRANSPORTERS													•		
16. INDICATING ACCELEROMETER KIT													•		
17. PYRO INST. TOOL KIT									•						
18. STAGE MOTOR INST FIXTURE								•							
19. EOS/SHUTTLE SIMULATOR ADAPTER											•				
20. TIE DOWN KIT SET													•		
21. BATTERY SHIPPING CONTAINER		•													
22. BATTERY INSTALLATION TOOL		•													
FLUID															
1. GN_2 CONDITIONING UNIT								•	•	•		•		•	•
2. GN_2 REGULATION UNIT								•	•	•		•		•	•
3. VOLUMETRIC LEAK DETECTOR								•	•	•					
4. RCS VACUUM TEST CART								•	•	•					
5. FLUID DIST SYS, GRUMMAN								•	•	•					
6. PRESSURE MAINTENANCE UNIT													•		
7. GN_2 STORAGE SYS (TRANSPORTER)													•		
8. GN_2 MANIFOLD & SUPPLY PLATFORM													•		
9. FLUID DIST SYS, LAUNCH SITE														•	•
10. PROPELLANT TRANS ASSEMBLY															•
11. MASS SPECTROMETER LEAK DETECTOR								•	•	•					

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Fig. 4.6-1 Ground Support Equipment Requirements Vs Spacecraft Activity Schedule

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DMS CONFIGURATION
STUDIES

5 - DMS CONFIGURATION STUDIES

5.1 GENERAL

The EOS DMS is composed of several system elements that are connected together so that the DMS supports the EOS Program by providing:

- Payload Data Acquisition and Recording
- Data Processing and Product Generation
- Spacecraft and Data Processing Management and Control
- Data User Services

Two types of data acquisition and processing configurations exist. A primary or high data rate configuration is madeup of Primary Ground Stations (ULA, GDS, ETC) and the Central Data Processing Facility (CDPF). Several secondary or Local User Systems (LUSs) are composed of low cost receiving, recording, and processing and display subsystems that make up Low Cost Ground Stations (LCGSs)

The CDPF is composed of several systems that process payload data, produce data products, enable management and control, and provide information and data retrieval services for the data users. Two sub-areas of the CDPF are the Information Services System (ISS) and the Central Processing System (CPS). System management and control are exercised through the Information Management System (IMS), part of the ISS. Other services are packing and shipping of data products and a data products scheduling and ordering capability.

The CPS provides radiometric and geometric image data corrections and processed data archive capabilities. It is a modular system that can be expanded as the need arises to process increasing payload data loads.

5.2 DMS OVERVIEW

Figure 5.2-1 indicates the major DMS elements. Design tradeoff areas that have been considered include:

- CPSs that process 20 and more scenes per day
- Minicomputer versus large scale CPS computers
- R&D, prototype, and production CPS equipment
- An automated, semiautomated, and manual IMS
- Special versus conventional NASCOM communications
- New versus modified PGSs
- Modular LUS designs
- Centralized LUS support elements

The current conclusions are that a 20 scene CPS should be adequate to handle NASA's EOS payload data processing needs based on only supplying processed data products for R&D users. Using minicomputers in the CPS will cost less than using large scale computers. The CDPF should be an R & D system, capable of expansion, rather than a prototype or production facility. The IMS should be semiautomated and convertible to an automated system for a production CDPF.

Further conclusions are that the planned NASCOM communications are adequate to handle the EOS command, housekeeping, and tracking data needs for the prototype system. Modified STDN PGSs that acquire and record the EOS payload data are less expensive than developing new PGSs. A modular LUS that can serve several user applications areas is relatively inexpensive with respect to regional stations, and it can be a LCGS or be a subset of the LCGS equipment that is only used to process and analyze the image data. Assuming that the LUS population is between 10 and 100 terminals, centralized application program development and equipment diagnostic capabilities can reduce the LUS maintenance costs and enhance LUS utilization. Computer program development equipment is not required in the LUS terminals. Therefore these terminals can be operated by applications personnel rather than computer operators and programmers. An online automated data archive is considered a necessity for a production CPS, and a prototype archive that is modularly expandable is considered as a development option.

5.3 STDN MODIFICATIONS/PGS DATA ACQUISITION AND RECORDING

Figure 5.3-1 shows the network modifications that are to be implemented at each PGS. Appendix D.2 provides the study details.

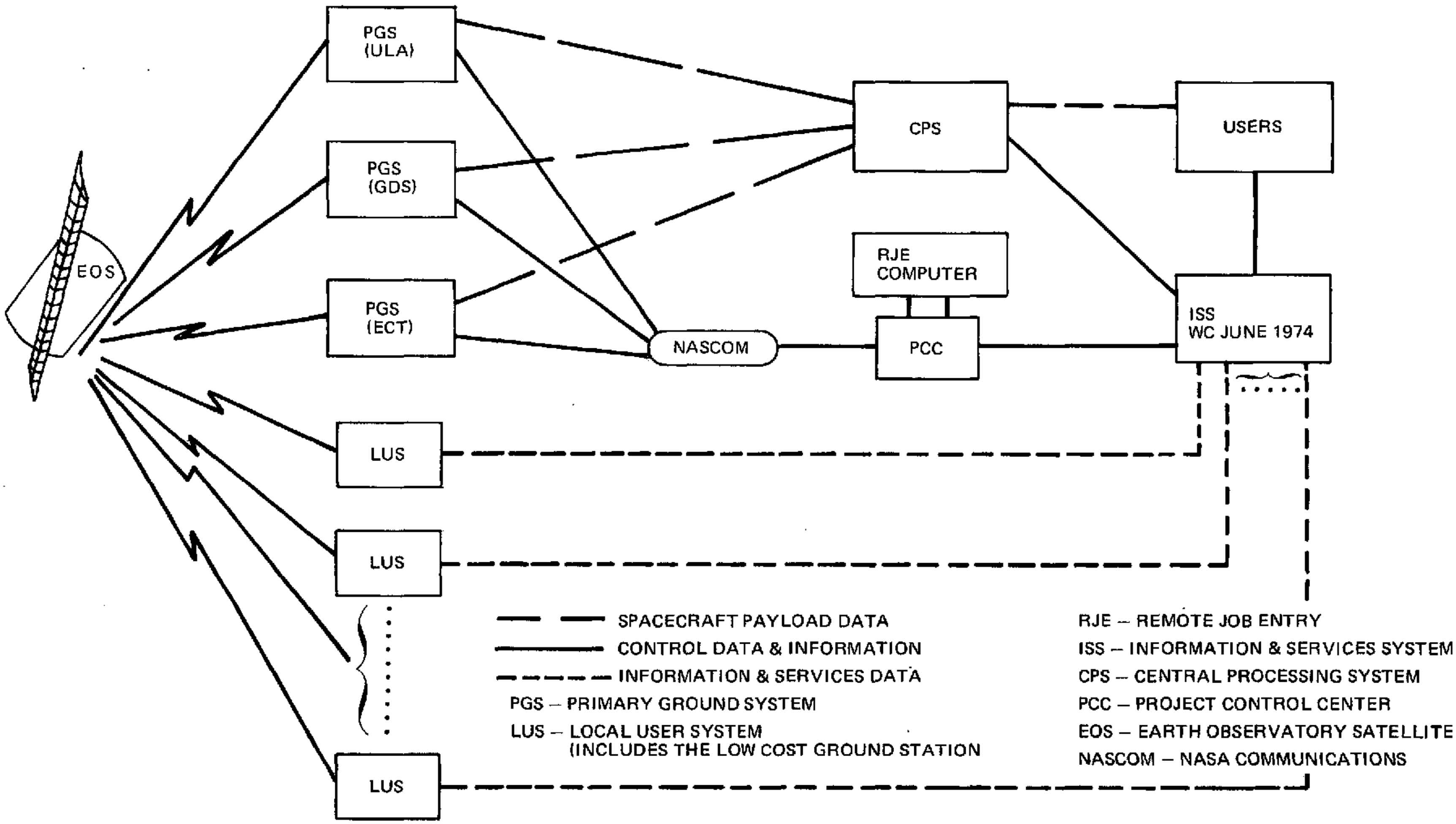

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Fig. 5.2-1 EOS Data Management System (Data Communications, Control, Processing and Information Services)

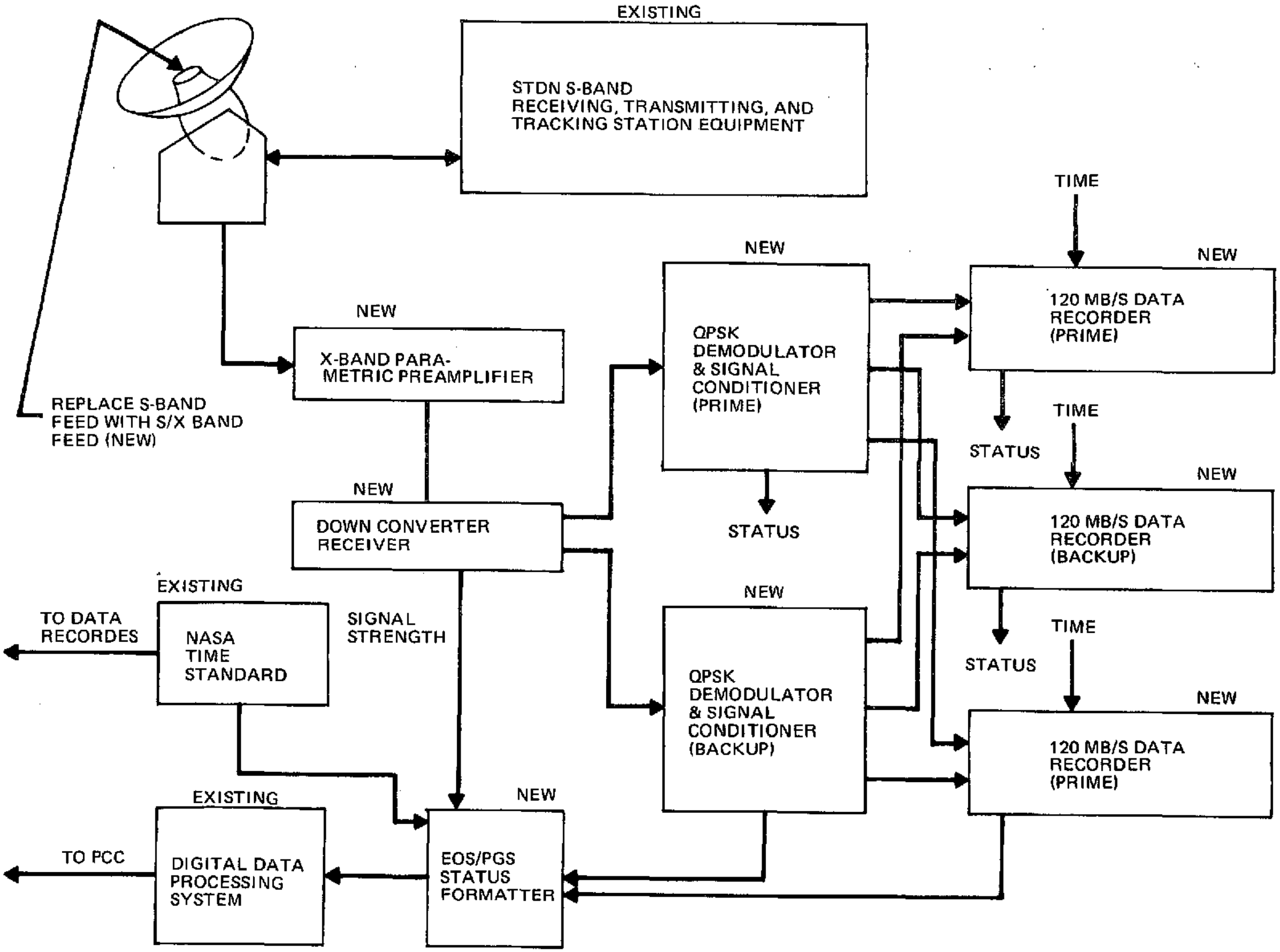

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Fig. 5.3-1 Primary Ground Station Modifications to STDN Stations

Purpose of Modifications

- Provide X-Band Receiving Capability
- Provide QPSK Demodulation Capability
- Provide up to 240 Mb/s Data Recording Capability
- Provide Receiving/Recording Equipment Status Indications

Tradeoff Areas

1. A new dual S/X-Band feed installed in the existing 30/40-foot STDN reflectors vs. a new X-Band antenna system. The dual S/X Band feed was selected because it saves the cost of a new antenna subsystem and results in negligible degradation to the existing S-Band system.

2. A new uncooled parametric preamplifier vs. a new cooled preamplifier. The uncooled unit was selected because it yields adequate performance at a minimum cost and maintenance.

3. A new receiver vs. modification to the existing site S-Band receivers. The new unit was selected because of design simplicity and installation, and increased reliability.

4. Suppressed carrier QPSK modulation with digital encoding for ambiguity resolution versus residual-carrier modulation. The digital resolution approach was selected to simplify the recording systems and to recover the loss of approximately 0.5 dB which is incurred with the residual-carrier approach.

In addition a Status Formatter was selected because it enables the PCC to receive near-real-time indications of equipment and communications link operational quality.

Additional tradeoff areas that were considered are:

1. The possible use of an S/X/Ku-Band feed in the existing STDN reflectors to prepare for future S/C modification.

2. The use of an independent (from the existing STDN site) X-Band ground terminal to provide flexibility of site location.

3. The use of an X/Ku-Band feed in the existing STDN reflector or a new X-Band reflector to prepare for future S/C modifications.

4. The feasibility of making greater use of existing STDN equipment in the new X-Band receiver to reduce cost.

The status of these areas is discussed in the Appendix.

5.4 CENTRAL DATA PROCESSING FACILITY-OVERVIEW

Figure 5.4-1 shows processed data flow. Figure 5.4-2 shows the processing time required for one TM scene. Appendix D.2 details the technical studies performed in the data processing area. Significant conclusions from tradeoffs are:

- A practical breakpoint from general purpose to special design processors occurs when more than 20 scenes of payload data per day must be processed.
- Throughput strongly depends on the geometric interpolation algorithm used.
- Data Input/Output (I/O) transfer through the system must be optimized (balanced between processors and working storage) to maximize throughput. I/O transfer can dominate throughput considerations at rates greater than 20 scenes per day.

Processing operations required to meet various throughput - algorithm (indicated in Figure 5.4-2) combinations are:

- 20 Scenes/Day - 1 Mips (Million Instructions Per Second) processors and parallel data transfer required to provide nearest neighbor interpolation

  10 Mips processors, parallel transfer, provides all interpolation algorithms
- 90 Scenes/Day - 100 Mips processor, nearest neighbor interpolation only, (becomes I/O limited with other algorithms)
- 400 Scenes/Day - All systems I/O limited, cannot meet with practical general purpose processors

The CDPF activities include:

- Pre-processing and Level I processing for radiometric/linear corrections
- Prototype Archive
- Level II geometric correction processing with precision ephemeris and S/C attitude data
- Level III geometric correction processing with Ground Control Points
- Digital and Photo output products
- Information Management System
- LUS Program Development and Equipment Diagnostic Laboratories

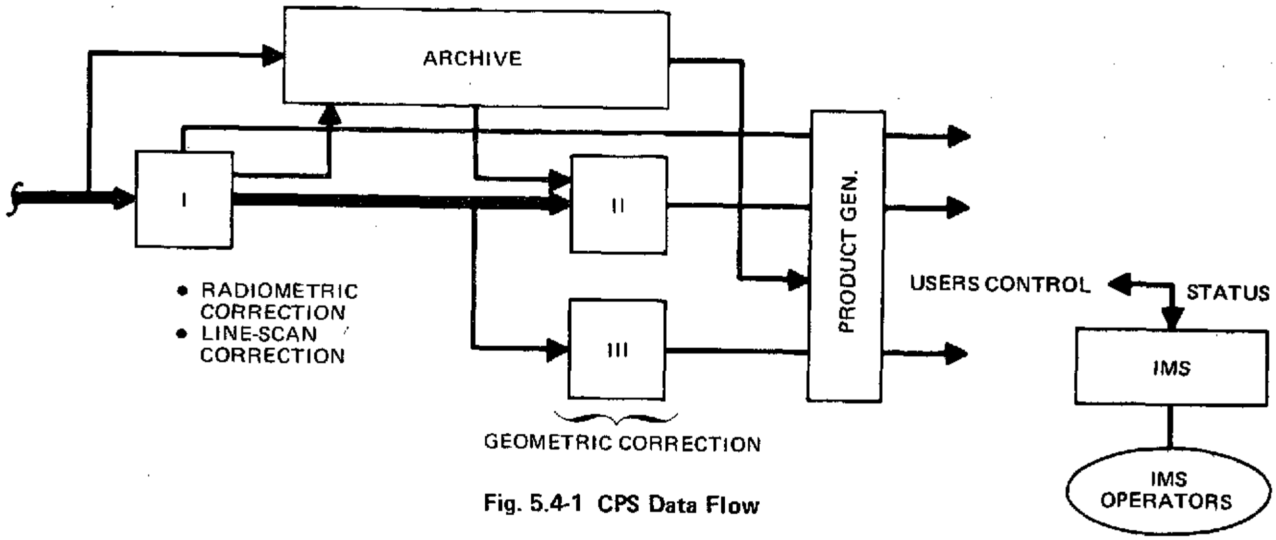

Fig. 5.4-1 CPS Data Flow

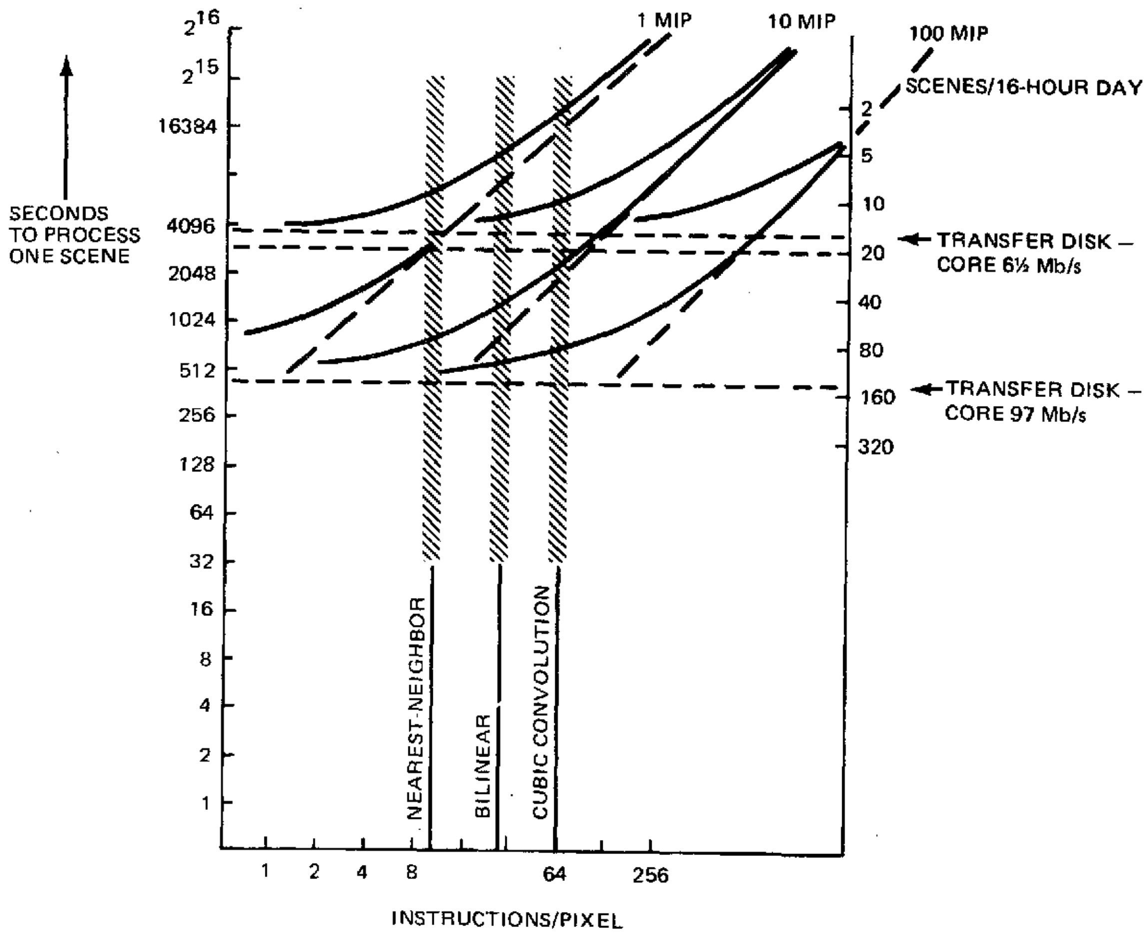

Fig. 5.4-2 Time to Process One TM Scene Vs Number of Instructions/Pixel

5.4.1 DATA PREPROCESSING AND LEVEL I PROCESSING CONCLUSIONS

- Modular approach required for expansion because reprocessing and Level I processing are performed on all input scene data (see Figure 5.4-3)
- Preprocessing must provide pixel decommutation, cloud cover display, and input of precision ephemeris and S/C attitude data from IMS for prototype CPS
- Radiometric Correction (Level I)
  - Table Look-up performed in special design hardware for production CPS
  - Line-Scan Correction (one-dimensional), if required, performed in special design hardware for production CPS
  - Scan error data supplied with the image pixels
- Multiport Disk Units used for pixel data transfer between prototype CPS processors to enable fast I/O data exchange.

5.4.2 ARCHIVE TRADEOFFS, CONCLUSIONS, AND FACTS

- Several digital archiving systems available
- Ampex TBM system chosen for detailed study because
  - Minimum cost for initial basic system development
  - Ease of expansion to meet increased on line storage demand
- Minimum basic system is shown in Figure 5.4-4
- Record and reproduce simulataneous data streams, each at 5.6 Mbps Projected update to 50 Mbps
- Uses standard magnetic video tape "TBMTAPE"
- Tape capacity is 45×10^9 bits (about 15 scenes) Projected update to 10^{12} bits/tape
- Two independent transport units (i.e., two tape reels)
- Switching from one tape to another is done automatically when one is full
- Average access time is 15 sec., worst case is 45 seconds
- Independent read and write channels in Data Channel (DC), thus giving input/output rate of 11.2 Mbps
- Uncorrectable error rate 1.5×10^{-11}
- Storage Control Processor (SCP) maintains master file directory (MFD) of all data files (scenes or images)

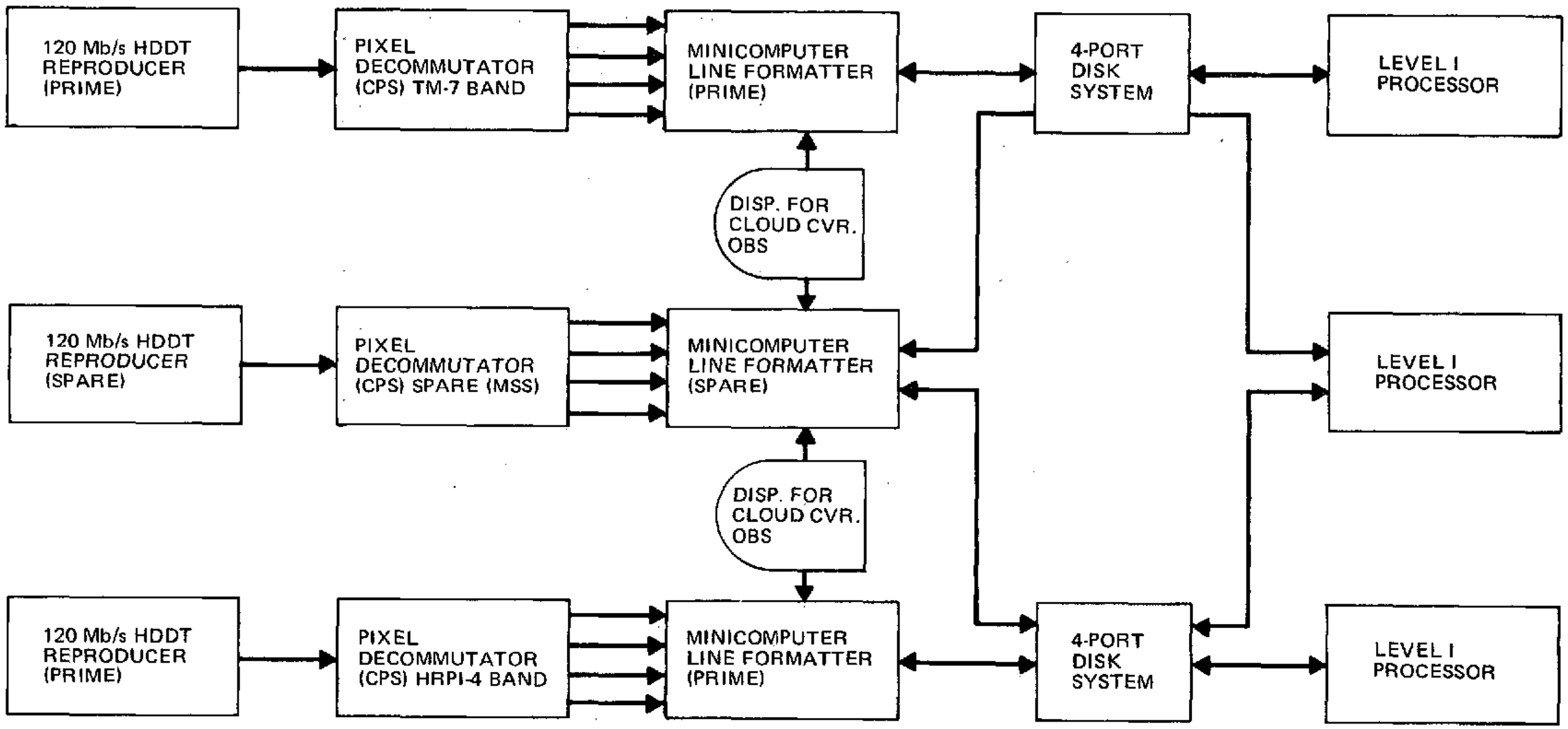

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Fig. 5.4-3 Preprocessing and Level I Processor Configuration

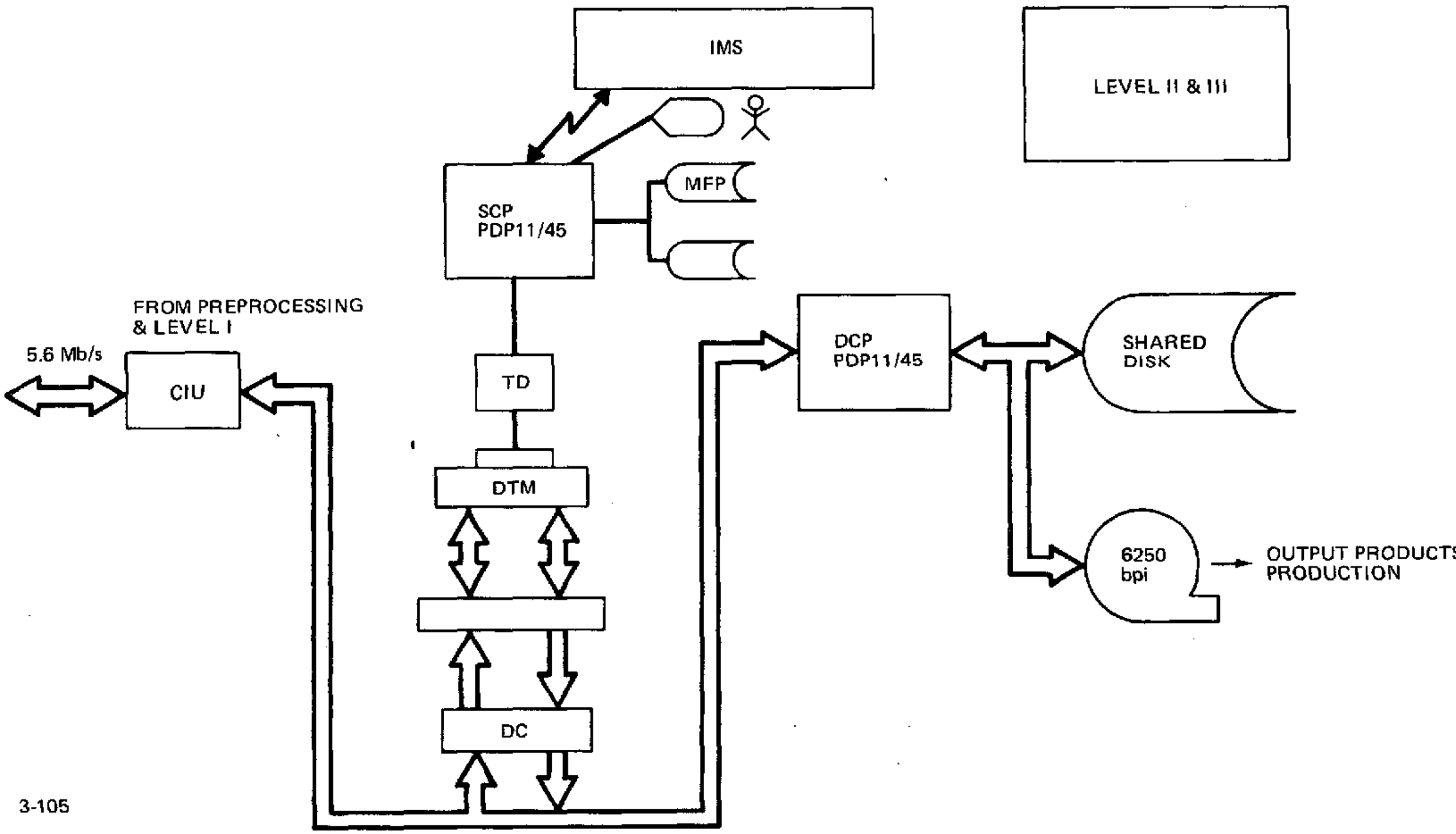

Fig. 5.4-4 Minimum System for Prototype and Easy Expansion

- SCP also manages internal work queues generated by requests from IMS CPU and other CPUs.
- Data Channel Processor (DCP) can directly transfer scenes or images to IMS CPU
- DCP transfers indirectly the data files to other CPUs through shared disk
- DCP provides for generating off-line computer compatible tapes (CCT)
- Prototype allows development to production system that uses developed software and expanded hardware.

5.4.3 LEVEL II PROCESSING

Purpose:

Geometric corrections are made using the best available estimates of S/C attitude and ephemeris, and models of Earth's rotation and curvature. Two-dimensional resampling/interpolation of the original data is then used to produce the final UTM (Universal Transverse Mercator) map projections.

Description:

Processing is carried out in two steps. (Figures 5.4-5 and 5.4-6)

1. A resampling grid is determined by locating the intersections of latitude and longitude lines on the earth's surface in the scanner coordinate system, which is identified by line number and pixel number within the line. A number of coordinate transformations are required to do this. The grid provides a coarse subdivision of the original data (say into 100-1000 blocks) so that linear coordinate computation can be performed over each block.
2. Once the parameters of the resampling grid are determined, we compute the coordinates of the desired output data samples (x, y), as one moves through the original data samples. Then we can use either (a) nearest neighbor interpolation, (b) bilinear interpolation or (c) cubic convolution to obtain desired output samples from the true data samples.

Conclusions:

1. The resampling grid computation requires about 3×10^4 machine instructions (MIs) per grid point. This number increases for more accurate grids, which will require many more iterations of the coordinate transforms. (1 MI = 1 Integer Add Time)

2. Numbers of machine instructions per pixel of approximately 5, 28 and 100 are required for the nearest neighbor, bilinear and cubic interpolations, respectively. These numbers are machine dependent; computers with special capabilities reduce these numbers.

3. The sensor scan technique of conical rather than linear increases processing time significantly for nearest neighbor interpolation, because of the additional complexity of the coordinate computation, and moderately (20%) for cubic convolution.

4. Impact of conical scan impact on LUS costs requires that the number of local user terminals be defined.

5.4.4 LEVEL III PROCESSING

Purpose:

This processing uses the same two steps described on the Level II processing with a significant difference in that the resampling grid is determined more precisely. To obtain this precision resampling grid a number of GCPs are, as far as possible, uniformly distributed over the scene (an area of 185 x 185 km^2). GCP location can be automatic using four possible techniques: (1) straightforward correlation (2) 2-D fast fourier transform (FFT) (3) sequential similarity detection algorithm (SSDA), and (4) special procedures that use edges/contours in the images. After the GCPs are located, a linearized least squares differential correction procedure can be used at GCP locations with S/C attitude and ephemeris as barometers, to obtain corrections and more precisely determine the resampling grid. (Figure 5.4-7)

Conclusions:

1. If the SSDA algorithm proves to be feasible in locating GCPs, the operations required to locate a moderate number of GCPs per scene should have a negligible impact on throughput rate. Simulation studies will be required to verify this.

2. Resampling grid accuracy after GCP correction is an order of magnitude better than that in Level II processing. Actual simulation is required to confirm this.

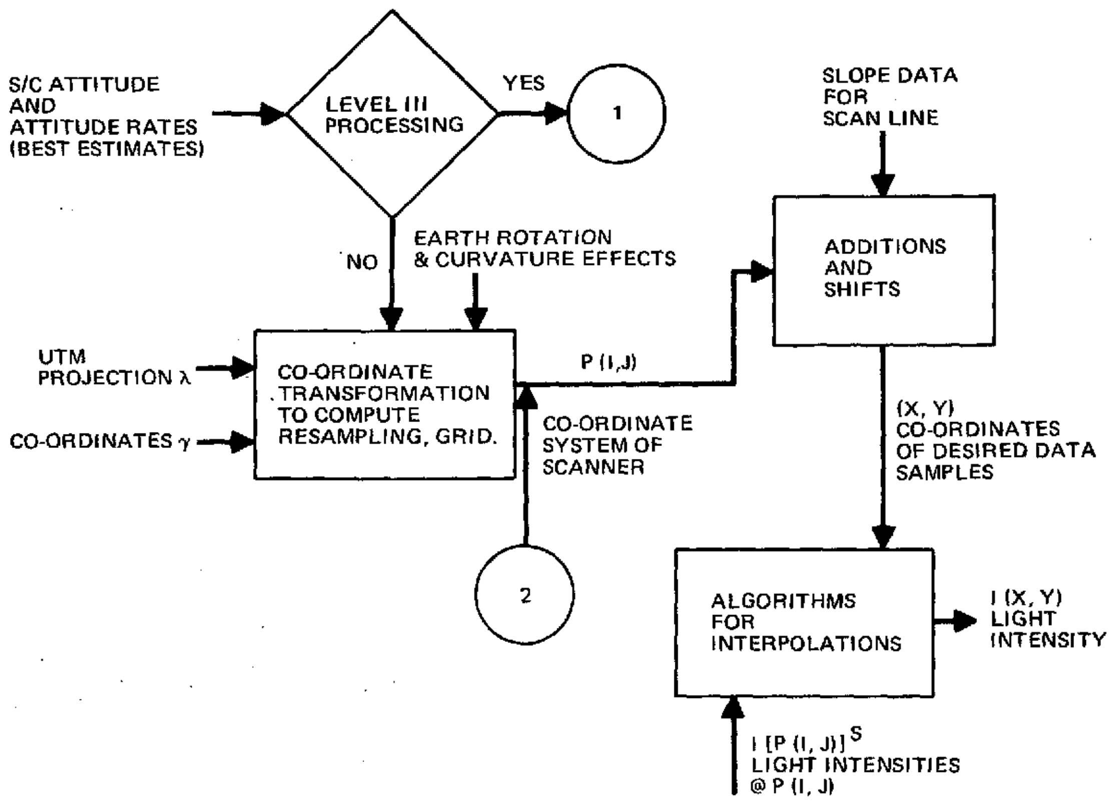

Fig. 5.4-5 Overall Picture of Level II Processing

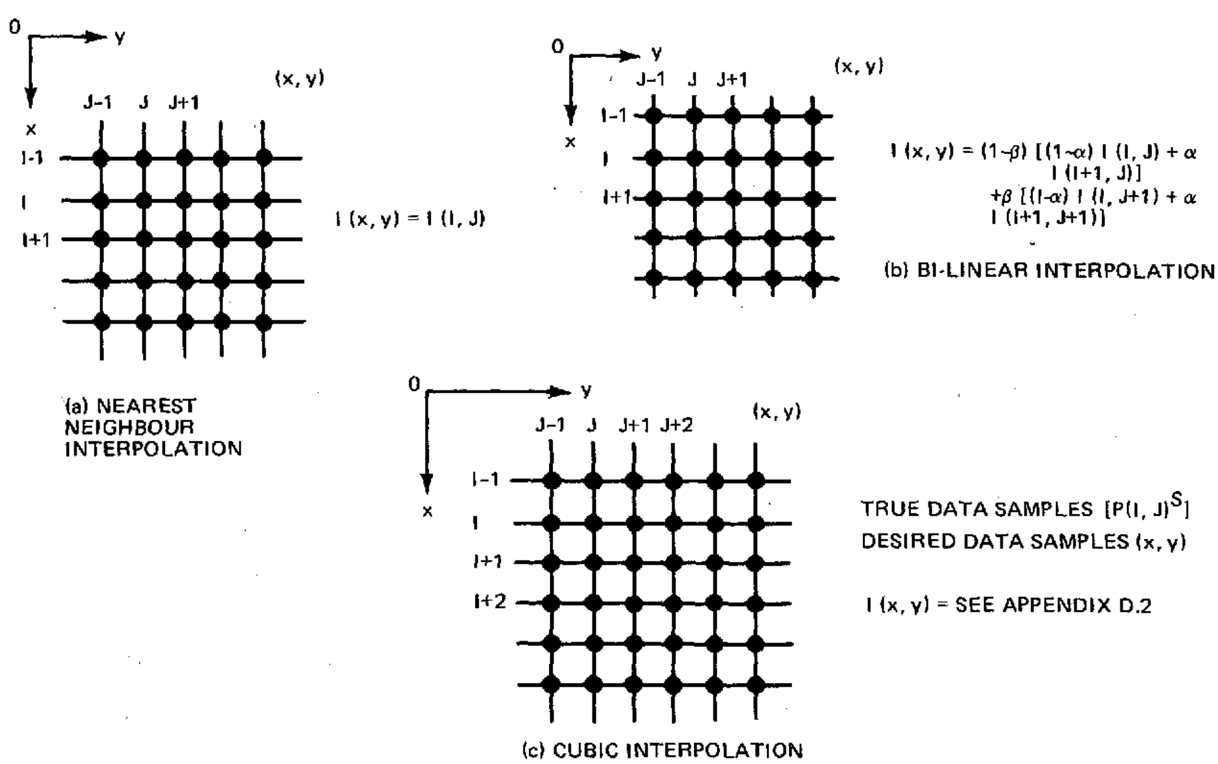

Fig. 5.4-6 Geometries for Various Interpolations

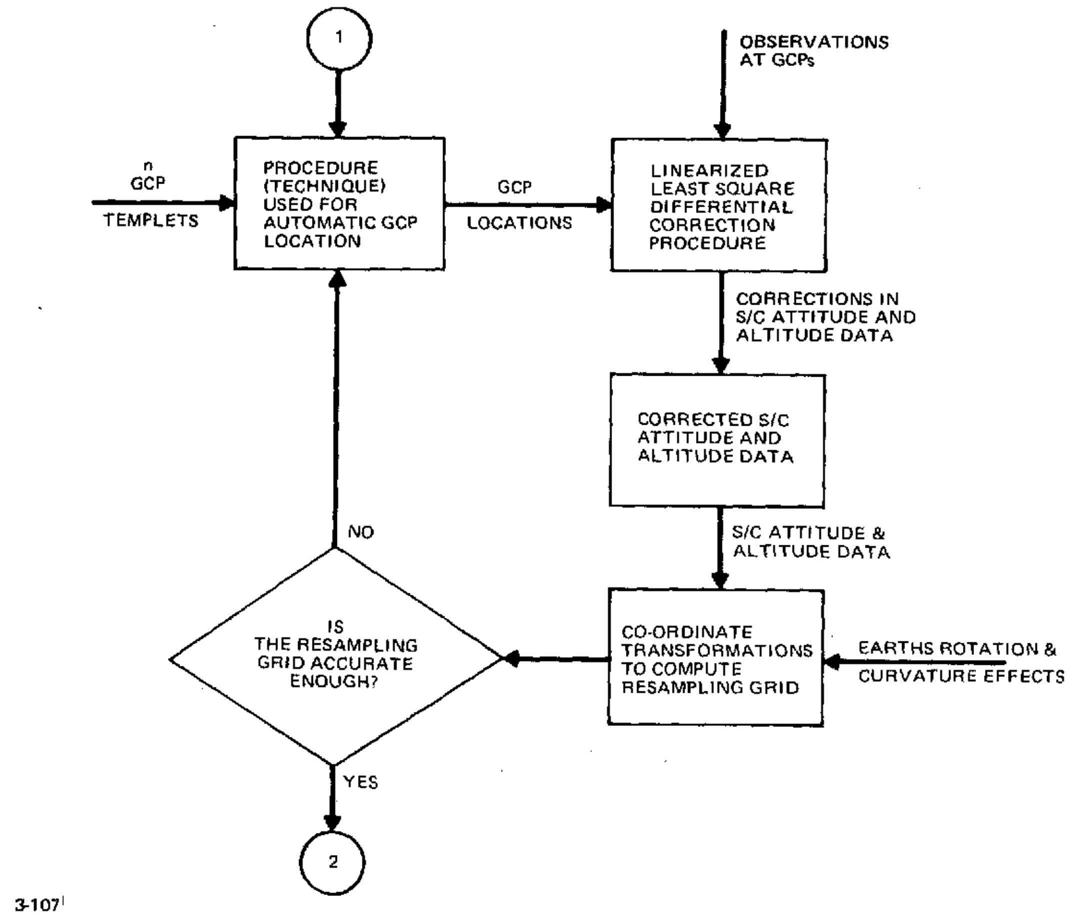

Fig. 5.4-7 Overall Picture of Level III Processing (Refer to Figure 5.7 of Level II Processing for Interpolations and Links)

5.4.5 INFORMATION MANAGEMENT SYSTEM

The ISS provides user interface and overall DMS central system control as summarized in Table 5.4-1.

Three functional options are presented in Table 5.4-2. Further details are provided in Appendix D, Subsection 2.3.8.

5.4.6 LUS CENTRALIZED SUPPORT CAPABILITIES

The systems concept is to provide modular hardware and software capabilities for the LUSs that would be complemented by centralized support capabilities. The centralized support elements are assumed to be located within the GSFC complex, and are collocated with (and within) the Information Services System (ISS), the Project Control Center (PCC), and the Central Processing System (CPS).

Table 5.4-1 Summary of IMS Functions

FUNCTION	ACTIVITIES
• IMAGE CATALOG AND DATA INVENTORY	• IMAGE CATALOG/DIRECTORY • IMAGE DESCRIPTOR INDEX • IMAGE ORIGINAL AND DATA PRODUCT INVENTORY/LOCATOR
• ORDERING FOR OBSERVATIONS AND DATA PRODUCTS	• STANDING ORDERS • DATA REQUESTS • OBSERVATION REQUESTS • ORDER STATUS INQUIRIES • OVERALL SYSTEM CONTROL
• SCHEDULING AND CONTROL	• SCHEDULES • WORK ORDERS • OPERATOR INTERFACE • PRODUCT QUALITY CONTROL
• ACCOUNTING, REPORTING, AND HISTORICAL DATA	• SYSTEM UTILIZATION REPORTS • USER ACCOUNTING • USER/PRODUCT CROSS TABULATION
• PRODUCT ROUTINE AND DELIVERY	• MAILING LABELS • DIRECT TRANSMISSION

Table 5.4-2 Summary of IMS Options

	1	2	3
STANDING ORDER PRODUCT LIMITATION	LIMITED	EXTENDED	UNLIMITED
SENSOR OBSERVATION REQUEST TIME FRAME	3 MONTHS	1 YEAR	5 YEARS
USER ACCESS TO SYSTEM (ON-LINE)	LOCAL OPERATOR TERMINAL	REMOTE USER TERMINAL	REMOTE USER TERMINAL
TRANSACTIONS ALLOWED ON LINE:			
CATALOG QUERY	SIMPLE	EXTENDED	EXTENDED
PRODUCT REQUEST	YES	YES	YES
ORDER STATUS REQUEST	SIMPLE	SIMPLE	EXTENDED
IMAGE DESCRIPTOR ENTRY	NO	NO	YES
ORDER PRIORITY	FIFO	FIFO AND SPECIAL	PRIORITY LEVELS
ACCOUNTING DATA REQUEST	NO	NO	YES
LIMITED DIGITAL PRODUCT DELIVERY	NO	NO	YES
PRODUCT/USER CROSS-TABULATION	NO	YES	YES
ACCOUNTING/REPORTING CYCLE	MONTHLY	MONTHLY	ON-LINE DAILY SUMMARY
CDP SYSTEM CONTROL	PRINT DAILY ORDER LIST	ORDER LISTING ON	DETAILED SCHEDULE
CATALOG LEVEL OF DETAIL	SIMPLE	SIMPLE	CURRENT DATA LOCATION
SENSOR REQUEST LEAD TIME REQUIRED	HIGH	MEDIUM	LOW

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Two centralized support elements are the Applications Program Development Laboratory (APDL) and the LUS Diagnostic and Equipment Laboratory (LDEL). The APDL provides a computerized capability for the development of LUS Applications Programs and the conversion of previously developed programs for use with the LUSs. Additionally, scientific consultation services would be available from the APDL personnel. Remote LUS processing and analysis equipment testing is provided by the LDEL via low-speed digital data dial-up telephone lines. The LDEL operators would be experts with the operational LUS hardware and software and would be able to exercise the local computerized equipment via low-speed digital communications from their central location.

Note that a basic assumption for the centralized/local system concept is that the LUS operators are primarily applications oriented (i.e., the operators are not necessarily computer programmers or computer operator experts). Therefore, the applications and diagnostic support which is necessary to maintain operational LUSs is provided by the shared centralized system elements. This concept would be cost-effective if there were at least 10 LUSs. If only a few LUSs were deployed, the APDL and LDEL would not be economical. The exact break-point for the cost-effectiveness has not yet been determined.

Adding or eliminating the APDL and LDEL elements does not affect the acquisition, display, and processing capabilities of the LUS. However, one centralized element that is necessary for LUS operation is the IMS. The LUS operators communicate with the IMS via dial-up voice or digital low-speed telephone lines to receive precision EOS orbit and attitude data as well as make known their requests for CPS processed computer compatible tapes (CCTs) and picture products. Additionally, the operators would receive EOS orbit predictions and coverage time information from the IMS to point and acquire the direct EOS-to-LUS data transmissions. Figure 5.4-8 shows how the centralized and local system elements are interfaced.

5.4.7 BASIC CDPF CONCEPTIONAL HARDWARE AND SOFTWARE

General:

Hardware selected for the CDPF basic configuration represents current performance/price relationships for a multi-configuration minicomputer approach for the radiometric and geometric image data processing. The immediate future appears to offer significant improvements for this direction considering a prototype CPS. Characteristics for each processor (Level I, II, III) reflect the characteristics of the specific processing tasks. A single supercomputer configuration approach was not included for the CPS design because of the cost and complexity to expand the CPS workload beyond that which can be performed within one configuration.

The initial cost for all the processors in this system design is a small portion of the total hardware cost. The largest single cost item is for high and medium speed processor memories to handle and rearrange the large sequences of data and to buffer I/O operations that would otherwise waste processor time. Disk hardware is the next largest cost item followed by the tape unit hardware. Further examination of this particular processing approach may result in additional cost reductions.

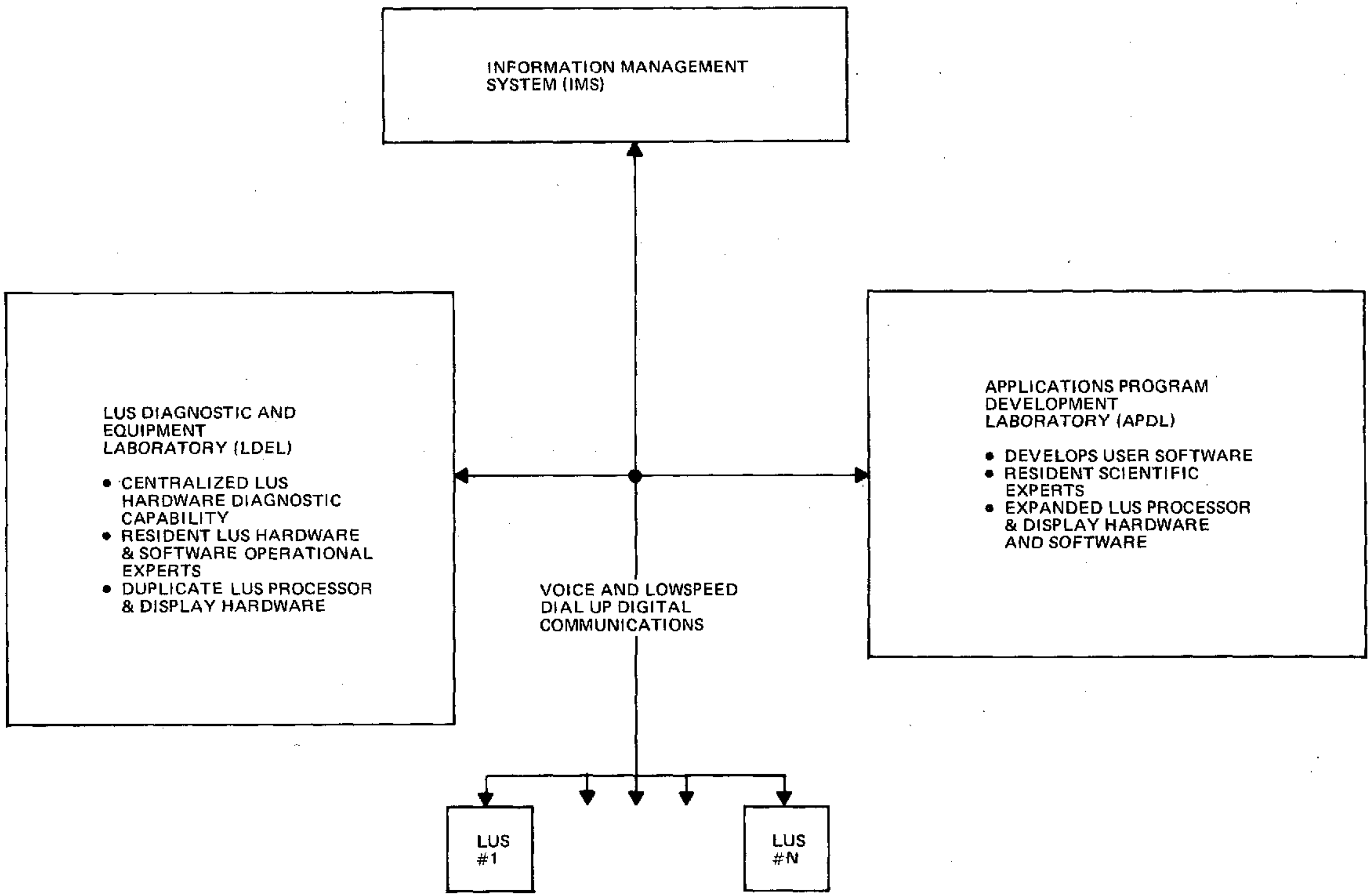


Fig. 5.4-8 LUS Centralized Support Elements

Software development and cost is not a major driver for the CPS. This is because a basic proceessing set of software is produced once and then used in additional processors for expansion.

The major software cost element is that for the IMS. Here we are dealing with pure development areas, such as data user request scheduling, information (rather than data) retrieval, and the status and control of a complex system. The system management software problem must be faced if a truly large scale production processing system is to ever be developed.

Hardware Configuration:

The basic CDPF hardware configuration and facility area layout is shown in Figure 5.4-9. An online data archive is not included for the basic configuration because it is not needed in an R & D system.

Processed data flow through the CPS begins at the Data Input Station, which is the preprocessing area. Operators view data images to reject those containing complete cloud cover. Formatters package the data and all ancillary information needed for Level I, II, and III processing.

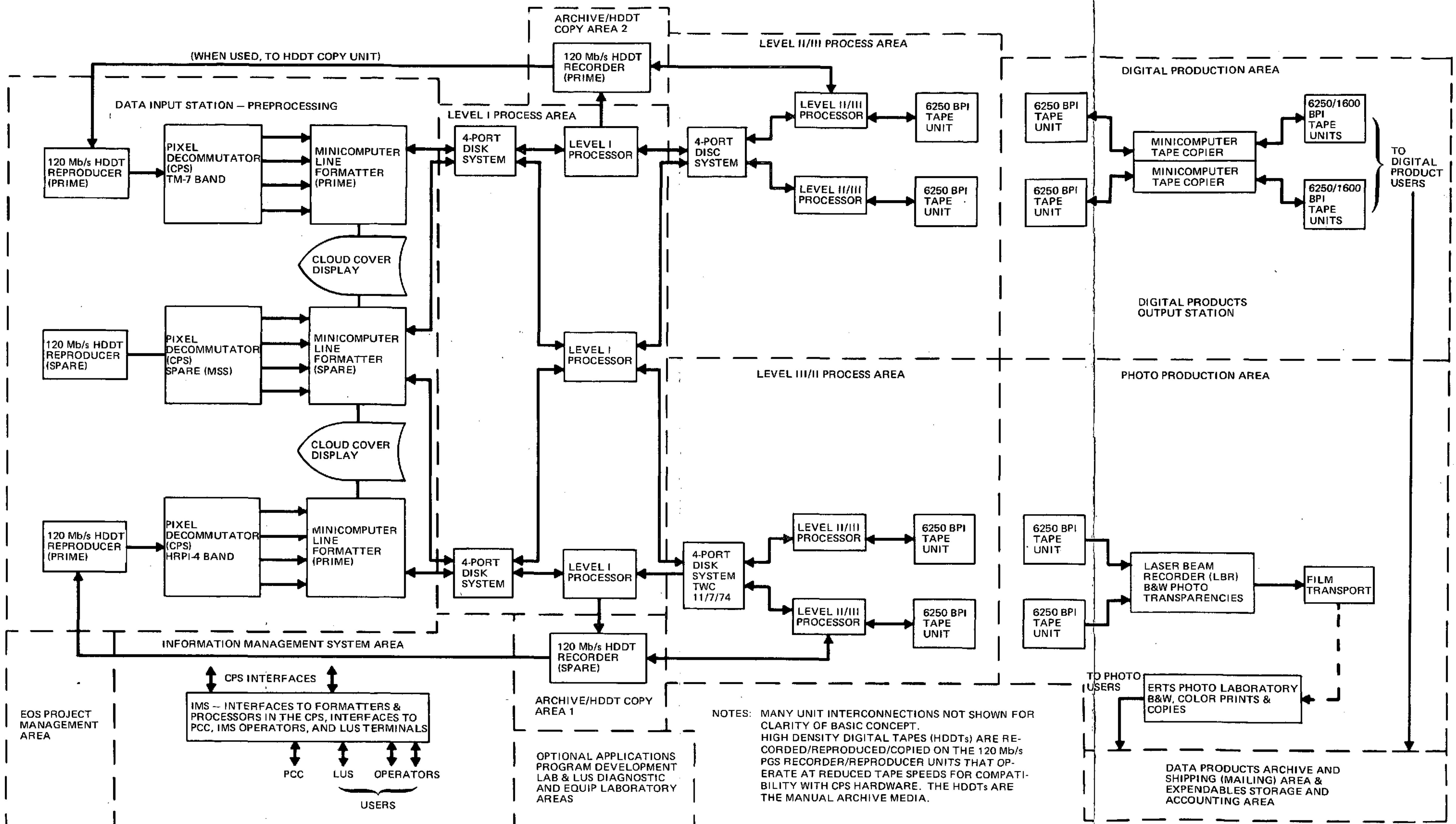
Multiport disk systems are used for interlevel working storage data transfer. All Level I processed data are recorded on HDDTs.

Level II/III processes are performed in identical hardware configurations for simplicity. Processed image data are output to Computer Compatible Tapes (CCTs). Processed data may also be recorded on HDDTs.

Digital and Photo Production areas use the CCTs for data input in the prototype CPS. For a production system the transfer from Level II/III processors to the production areas would be performed through an online archive (considered only as an option for now).

Two types of product CCTs are provided in several formats (pixel interleaved, band interleaved, etc.). The 6250 bytes per inch (bpi) tape density and 1600 bpi density tape systems are provided.

Original (first generation) B&W transparencies are provided to photo product users via the Laser Beam Recorder (LBR) digital-to-analog converter. Second generation B&W and color prints are developed in the ERTS photo laboratory assumed as GFE for this study.



NOTES: MANY UNIT INTERCONNECTIONS NOT SHOWN FOR CLARITY OF BASIC CONCEPT.
HIGH DENSITY DIGITAL TAPES (HDDTs) ARE RECORDED/REPRODUCED/COPIED ON THE 120 Mb/s PGS RECORDER/REPRODUCER UNITS THAT OPERATE AT REDUCED TAPE SPEEDS FOR COMPATIBILITY WITH CPS HARDWARE. THE HDDTs ARE THE MANUAL ARCHIVE MEDIA.

Fig. 5.4-9 Basic CDPF Conceptual Hardware Configuration (20 Scene/Day Processing – 185 km TM Swath Width Model) and Facility Areas

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FOLDOUT FRAME

FOLDOUT FRAME 2

Software:

Table 5.4-3 indicates the basic software routines required to operate the CDPF. Of these, the IMS routines require the most programmer time for development because system scheduling and management via the semiautomated IMS has never been done before.

Table 5.4-3 Basic CDPF Software Configuration

SUBSYSTEM	SOFTWARE ELEMENTS
PREPROCESSING	MONITOR CONTROL & STATUS SENSOR CALIBRATION I/O DRIVERS – PCC REQUEST CLOUD COVER DISPLAY LINE FORMAT AND ID INPUT TAPE ID CHECK EXECUTIVE OS
LEVEL I	MONITOR CONTROL & STATUS I/O DRIVERS (DISK, HDDT) PRE-SCAN FORMATTING LINEAR INTERPRETATION EXECUTIVE OS
LEVEL II/III	SYSTEM SUPERVISOR MONITOR CONTROL & STATUS I/O DRIVERS EPHEMERIS PROCESSOR ATTITUDE PROCESSOR GRID POINT UTM/LP CONVERSION RESAMPLE & PARAMETER DEV. PIXEL SORTING EXECUTIVE OS GCP GRID DEVELOPMENT LBR FORMATTER/SCENE OPTIONS
PRODUCTS PRODUCTION	EXECUTIVE OS I/O DRIVERS BPI SELECTION & AUTO FORMAT SELECTION OPTIONS LBR FORMATTER/IMAGE OPTIONS
IMS	SYSTEM SUPERVISOR USER/OPERATOR INTERFACE ORDER CONTROL, STATUS, HANDLING SCHEDULING & CONTROL ACCOUNTING/RPG EXECUTIVE OS

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5.5 LOCAL USER SYSTEM (LUS) OVERVIEW

A systems viewpoint is taken with respect to a wide family of LUSs which includes the Low Cost Ground Station concept. Centralized (see Paragraph 5.4.6) as well as local operations are necessary to assure systems' viability, and these operations have been considered.

Applications data users are assumed to operate the LUS terminals, and the users do not need to be computer programmers or operators. Therefore centralized applications program development is performed at the APDL eliminating the need for expensive program development equipment (card readers, card punches, extensive development system software, etc.) and the need for computer programmers at each LUS site.

Centralized checkout and diagnostic capability in the LDEL will eliminate the need for maintenance personnel at each LUS site for computerized equipment testing and diagnostic analyses. Detected problems will require local maintenance personnel to be sent to a LUS site, however, to effect repairs and to perform routine preventative maintenance.

Figure 5.5-1 shows the three basic subsystem elements that compose a terminal. The processing and display subsystem has a modular, expandable capability depending on the application user's needs. The other subsystems would be standard for the LCGS models.

The basic cost conclusions are that minimum (Basic) capability LCGSs can be provided for an equipment (hardware) cost in quantities of 10 or more of less than $150K, and that the enhanced processor and display subsystems, increasing the hardware cost to about $300K in quantity, should provide as much local processing and analysis capabilities as most local area analysis specialists would need.

5.5.1 LUS RF/IF SUBSYSTEM

Purpose

- To acquire and program track the EOS S/C
- To down convert X-Band Carrier Signals to 70 MHz for input to the Data Handling/ Recording Subsystem

Tradeoffs Considered

- Programmed S/C Tracking versus Manual and Autotrack
- Fixed S/C antenna versus steerable antenna effects on LUS cost
- Parametric versus FET preamplifiers

Conclusions

Manual tracking provides the least cost for the antenna system but is deemed impractical for most users. Autotracking capability requires excessive cost receiver and has been eliminated from further consideration because of cost. Programmed track capability has been selected as the candidate tracking method because the LUS processor can be used (no need for a new computer) with a special interface to the antenna drive servos resulting in a moderate system cost.

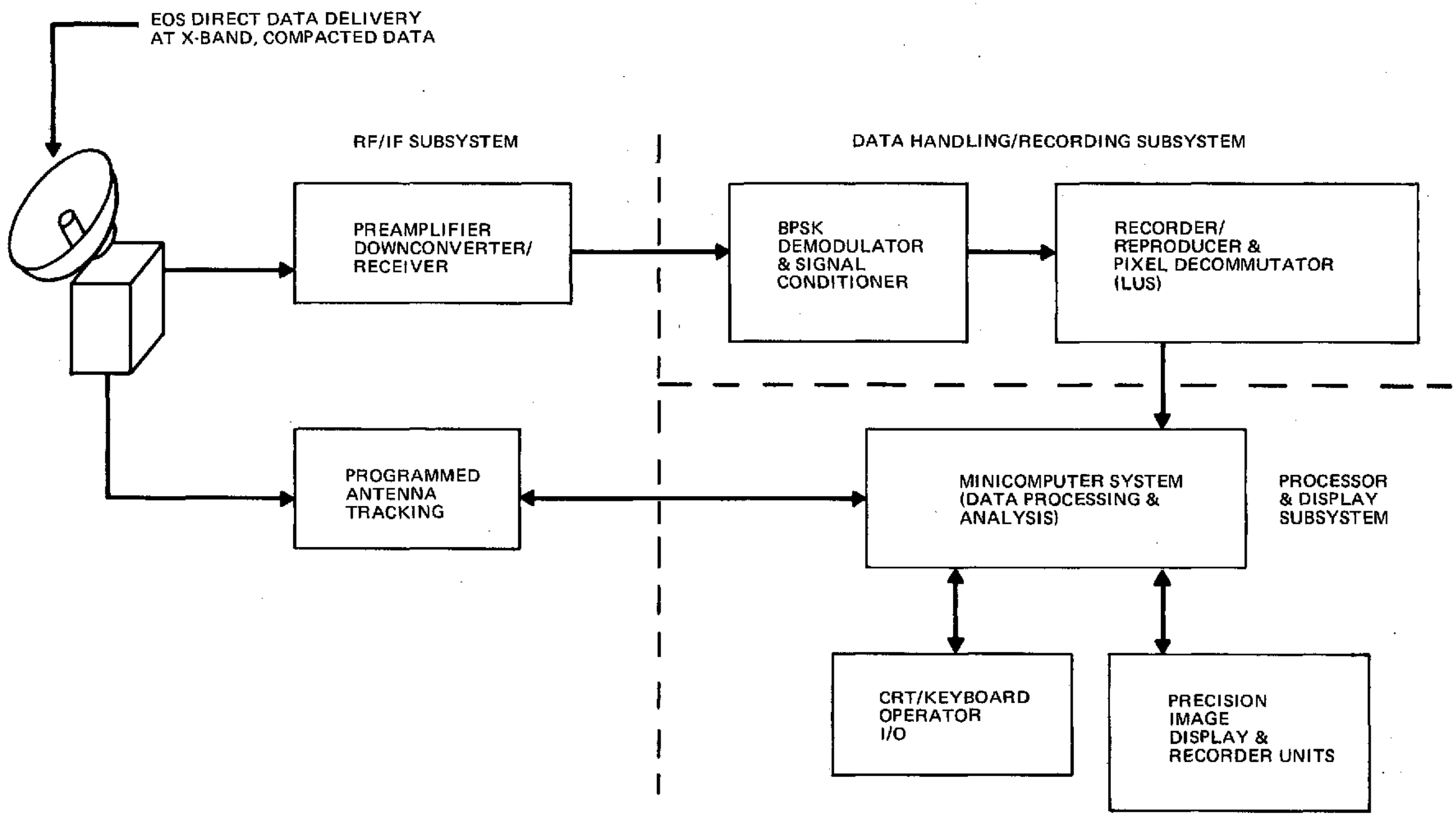


Fig. 5.5-1 Basic LUS Subsystems Make Up the Terminals that Have Software and Hardware Modular, Expandable Processor and Display Subsystems

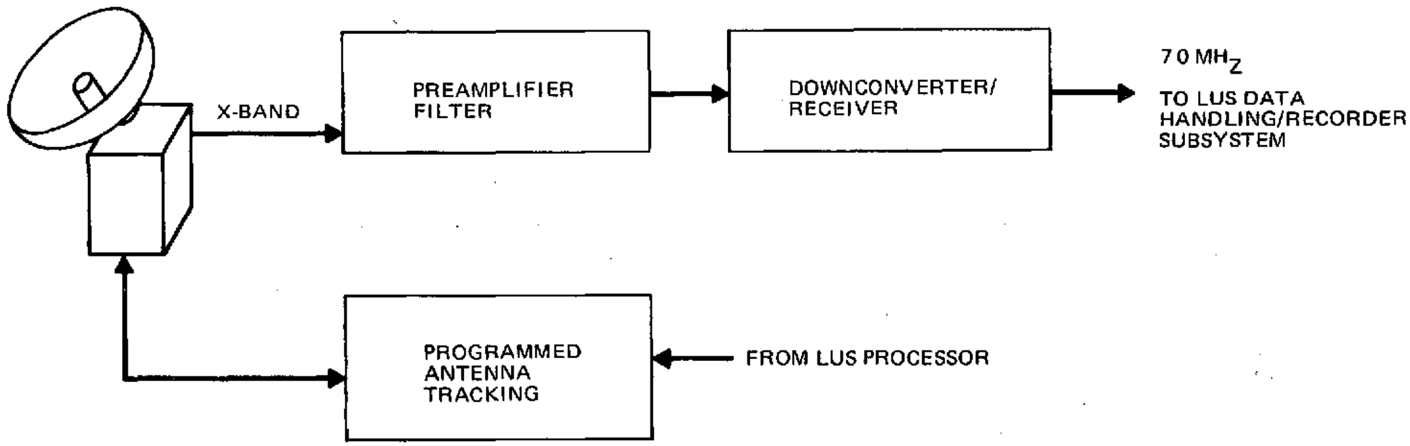
EOS S/C antenna (fixed or steerable) has a significant LCGS cost impact. A fixed antenna requires the RF/IF subsystem to have an 11' dish and an expensive parametric amplifier compared to a 6' dish and FET preamplifier that can be used with a S/C steerable antenna. See Figure 5.5-2.

5.5.2 LUS DATA HANDLING AND RECORDING SUBSYSTEM

Tradeoffs Considered

- Two different cost version for BPSK Demodulation and Signal Conditioner units available
  - First involves off-the-shelf subsystem without new technology. This is low nonrecurring but high recurring cost unit.

    Second is high nonrecurring and low recurring cost unit. It is commercial quality and is selected for LUS implementation
- Two concepts for acquiring and playback of data
  - Under first concept: (Figure 5.5-3)
    - o Data are demultiplexed in band format before recording on 16 track, 20 Mb/s recorder.
    - o Data reproduced at slower speed and entered through 16 DMA channels to the main memory of the mini-computer.
    - o For each buffer in input channel synchronizer there is double software buffer in core.
    - o While second core buffer is being filled, the first buffer contents are transferred to a CCT.
  - Under second concept: (Figure 5.5-4)
    - o Demodulated data are directly recorded on the 20 Mb/s unit.
    - o Serial data are converted into parallel before recording.
    - o Playback data at lower speed are converted from parallel to serial data stream.
    - o Serial data stream is decommutated before entering into minicomputer core buffers as in first concept.
- First concept selected because of lower acquisition recorder subsystem cost.



BASELINE CONCEPT CONFIGURATION
ANTENNA 6' DIAMETER-PROGRAMMED TRACKING
PREAMPLIFIER FIELD EFFECT TRANSISTOR
5 dB N.F.
S/C ANTENNA STEERING

ALTERNATIVE CONFIGURATION
11' DIAMETER-PROGRAMMED
TRACKING
PARAMETRIC 2.8 dB NOISE
FIGURE
S/C ANTENNA FIXED

Fig. 5.5-2 LUS RF/IF Subsystem and Alternative Configuration

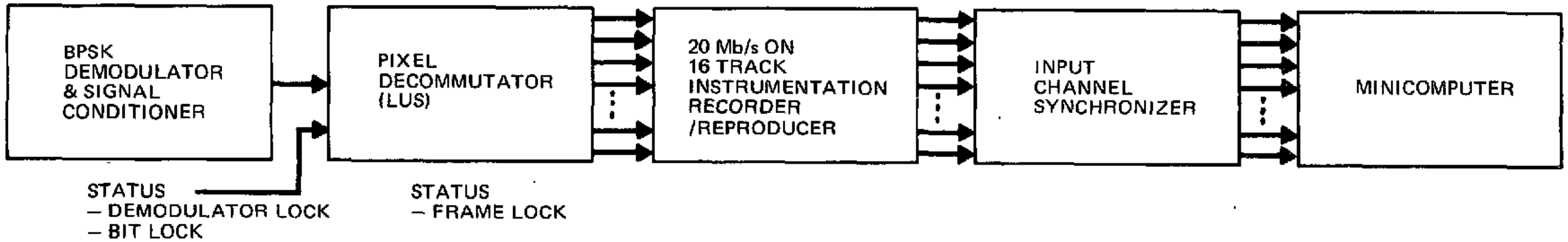


Fig. 5.5-3 LUS 20 Mb/s Data Acquisition & Recorder Playback (Concept 1)

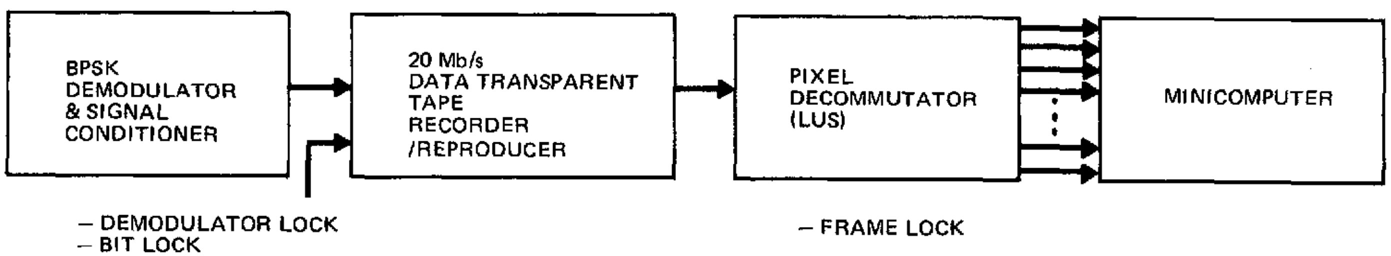


Fig. 5.5-4 LUS 20 Mb/s Data Acquisition and Recorder Playback (Concept 2)

5.5.3 LUS PROCESSOR AND DISPLAY SUBSYSTEM

Tradeoffs Considered

The following tradeoffs were considered:

- Three cost targets: $125K, $220K, and $300K for recurring (quantity 10 or more) hardware costs for LCGS LUSs that includes about $70K for the two previous subsystems.
- A single family of equipment.
- RF/IF and data handling/recording subsystems common for all LUS models.
- Processor and display subsystem with modular software, expandable to meet a variety of user applications needs.

Conclusions

For quantity 10, recurring LCGS hardware units are $130K for Basic Model, $223K for Enhanced I Model, and $300K for Enhanced II Model. Table 5.5-1 summarizes the processor and display hardware and model capabilities.

5.6 DMS ESTIMATED COST SUMMARY

Estimated DMS development and recurring costs are summarized in Table 5.6-1 for the basic DCPF Configuration illustrated in Figure 5.4-9 and the STDN and LUS concepts previously discussed. These costs do not include contractor General and Administrative (G & A) expenses and profit.

The online archives (TBM development) is not considered in the summarized costs because it would add about $800K to the estimate hardware and software development price plus increasing the expendables cost somewhat. It is not needed in the 20 scene per day CPS currently recommended for NASA implementation.

Tables 5.6-2, 5.6-3, and 5.6-4 show the estimated STDN modification, CDPF, and LUS cost estimates, respectively, in more detail than possible in Table 5.6-1. Note that these are current estimates and are subject to change as the study continues because of system optimization and further program design to cost functions.

Table 5.5-1 LUS Processor and Display Hardware Model Capabilities

LUS MODEL	HARDWARE	CAPABILITIES
BASIC	1 – MINICOMPUTER (32K MEMORY) 1 – DISK AND DRIVE (29 M BY) 2 – MAGNETIC TAPES* AND DRIVES 1 – OPERATOR I/O CRT/KEYBOARD 1 – B & W IMAGE STORAGE DISPLAY 1 – DATA REPRODUCER INTERFACE	DISPLAY – B & W IMAGES (1024 X 1024 PIXELS) DATA PROCESSING – YES (SLOW) IMAGE ANALYSIS – YES (VERY SLOW) HARDCOPY – YES (CAMERA)
ENHANCED I	BASIC HARDWARE PLUS 1 – MINICOMPUTER (MULTIPROCESSOR) 1 – LINE PRINTER 1 – INTERACTIVE COLOR DISPLAY 1 – FLOATING POINT & HARDWARE MULTIPLY/DIVIDE FOR MINICOMPUTERS	DISPLAY – B&W OR COLOR IMAGES DATA PROCESSING – MODERATE SPEED IMAGE ANALYSIS – INTERACTIVE HARDCOPY – CAMERA AND LINE PRINTER FOR THEMATIC MAPS, ETC.
ENHANCED II	BASIC, ENHANCED I HARDWARE PLUS 1 – DISK AND DRIVE (29 MBY) 2 – MAGNETIC TAPES AND DRIVES 1 – B&W AND COLOR IMAGE RECORDER 1 – COLOR IMAGE DISPLAY	DISPLAY-2 SIMULTANEOUS B&W OR COLOR IMAGES DATA PROCESSING – REASONABLY FAST IMAGE ANALYSIS – INTERACTIVE, CHANGE ANALYSIS, MODERATE SPEED HARDCOPY – LINE PRINTER AND FIRST GENERATION PHOTO PRODUCTS (70mm TO 4" X 5" SIZES)

*75 IPS & 1600 BPI

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Table 5.6-1 Estimated DMS Cost Summary

DMS ELEMENT	NONRECURRING	RECURRING/YEAR		
		SPARES	EXPENDABLES	O&M
	($K)	($K)	($K)	($K)
STDN MODIFICATIONS	2733.0	98.0[5]	0.0[1]	195.0[2]
CDPF	10159.2	342.8[6]	903.7	1576.7[3]
SUBTOTAL	12892.2	440.8	903.7[10]	1771.7
ADD FOR APDL & LDEL	1645.0	49.0[6]	15.0[7]	330.0[4]
SUBTOTAL	14537.2	489.8	918.7	2101.7
ADD FOR FIRST ENHANCED II LCGS	1266.6	15.0	10.0	100.0[8]
TOTAL ESTIMATED COST, INCLUDING ENHANCED II LCGS OPTIONS	15803.8	504.8[9]	928.7[9]	2201.7[9]

1. PAYLOAD DATA RECORDING TAPES & COST INCLUDED IN CDPF EXPENDABLES.
2. ASSUMES PROJECT PAYS 25% OF O&M COST, STDN PAYS 75% OF COST.
3. PROJECT MANAGEMENT ($256.7K/YR) PLUS 3 SHIFTS FOR O&M ($1,320. K/YR) FOR 16 HR PER DAY, 7 DAY/WEEK CDPF OPERATIONS
4. INCLUDES 2-40 HR/WEEK LDEL OPERATOR (6 PEOPLE) SHIFTS AND 40 HR/WEEK FOR 5 APDL OPERATORS AT $30.0K AVERAGE COST/YEAR/EMPLOYEE.
5. FIVE PERCENT OF MODIFICATION HARDWARE COST.
6. SEVEN PERCENT OF EQUIPMENT COST.
7. MISC.-PUNCHED CARDS, PRINTER PAPER, FILM, ETC.
8. ASSUMES 4-40-HR/WEEK OPERATORS.
9. ESTIMATED OPERATIONAL COST PER YEAR, $3635.2K INCLUDING OPTIONAL LUS ELEMENTS.
10. DAILY PRODUCTS ARE 21 1"–HDDT'S, 100 CCTs, 360 B&W POS/NEG TRANSPARENCIES, 150 B&W PRINTS, 20 COLOR POS/NEG TRANSPARENCIES, AND 20 COLOR PRINTS AT AN EXPENDABLE COST OF $2475.80 PER DAY.

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Table 5.6-2 STDN Modification Cost Estimates

SUBSYSTEM	HARDWARE NON-RECURRING PROTOTYPE DEVEL. ($K)	HARDWARE COST FOR 3 PCSs ($K)	REMARKS
RF/IF	267.5	546.0	INCLUDES ANTENNA FEEDS, PARAMETRIC AMPS, DOWN-CONVERTERS, BASIC SPARES, DOCUMENTATION, INSTALLATION AND CHECKOUT.[1]
DATA HANDLING/ RECORDING	415.0	1504.5	INCLUDES QPSK DEMODS & SIG. CONDS., 120 Mb/s HDDT RECORDERS, STATUS FORMATTERS, DESIGN, INSTALLATION, TEST, ETC.[1,2]
SUBTOTAL	682.5	2050.5	
SPARES/YR	–	98.0	AT 5% OF HARDWARE COST
O&M/YR	–	195.0	AT 25% OF TOTAL COST ASSUMING STDN USES PEOPLE 75% OF TIME

1. DOES NOT INCLUDE HARDWARE SHIPPING COSTS NOR INSTALLATION PERSONNEL PER DIEM AND TRAVEL CHARGES.
2. QUANTITY DISCOUNTS FOR HDDT UNITS, IF ANY, NOT APPLIED. HDDT DEVELOPMENT COST, INCLUDED HERE AND NOT ELSEWHERE WHERE HDDT UNITS ARE USED.

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Table 5.6-3 Basic CDPF Hardware & Software Development Cost Estimates

SUBSYSTEM	HARDWARE ($K)	SOFTWARE ($K)	REMARKS
PREPROCESSING	608.1	76.8	INCLUDES 3 HDDT UNITS, 3 PIXEL DOCOMMUTATORS, AND 3 FORMATTERS
LEVEL I PROCESSING	884.4	153.6	INCLUDES 2 4-PORT DISK SYSTEMS AND 3 LEVEL I PROCESSORS
HDDT ARCHIVE (MANUAL)	270.0	0.0	INCLUDES 2 HDDT UNITS
LEVEL II/III PROCESSING	2347.2	260.0	INCLUDES 2 4-PORT DISK SYSTEMS AND 4 LEVEL II/III PROCESSORS
DIGITAL PRODUCTS	209.6	40.0	INCLUDES INPUT & OUTPUT TAPE UNITS AND 2 MINICOMPUTER COPIERS
PHOTO PRODUCTS (ERTS PHOTO LAB, GFE)	348.5	60.0	INCLUDES INPUT TAPE UNITS AND ONE RCA LBR (LR72)
INFOR. MGMT SYSTEM	208.3	740.0	INCLUDES IMAGE CATALOGE, INVENTORY FILES, ETC., CPS, AND OPERATOR/USER INTERFACE PORTS (2 MINICOMPUTER SYS.)
SUBTOTAL	4876.1	1330.4	HARDWARE & SOFTWARE $6,206.5K
SYSTEMS ENGINEERING & INTEGRATION	$1,292.6K		10% OF HARDWARE COSTS PLUS ESTIMATED HARDWARE DEVELOPMENT COSTS
SYSTEMS DOCUMENTATION	$ 620.7K		10% OF HARDWARE & SOFTWARE COSTS
INSTALLATION & TEST	$ 931.1K		15% OF HARDWARE & SOFTWARE COSTS
MANAGEMENT	$ 620.7K		10% OF HARDWARE & SOFTWARE COSTS
FACILITIES	GFE		ASSUMED INSTALLATION AT GSFC
SPARES	$ 487.6K		10% OF HARDWARE COSTS
TOTAL COST ESTIMATE WITHOUT APDL & LDEL $10,159.2K			
ESTIMATED APDL & LDEL HARDWARE COST $700.0K, SOFTWARE COST INC DOCUMENTATION, ETC. $945.0 TOTAL COST ESTIMATE WITH APDL & LDEL $11,804.2K			
AMPEX TBM ONLINE ARCHIVE, $800.0K FOR HARDWARE & SOFTWARE			

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Table 5.6-4 LUS/LCGS Hardware and Software Cost Estimates (Enhanced II Model)

SUBSYSTEM	HARDWARE		REMARKS
	NON RECURRING PLUS FIRST UNIT ($K)	RECURRING FOR 10TH UNIT ($K)	
RF/IF	293.1	23.0	INCLUDES 6' DISK, PROGRAMMED TRACKING INTERFACE, FET PREAMP & DOWNCONVERTER/ RECEIVER
DATA HANDLING/RECORDING	211.0	49.0	DEMOD/SIG. COND., LUS PIXEL DECOM., RECORDING UNIT & COMP. INTERFACE
PROCESSOR/DISPLAY (ENHANCED II MODEL)	315.5	228.0	SEE TABLE 5.5.3-1 FOR HARDWARE/CAPABILITIES
SUBTOTAL	819.6	300.0	
SYSTEMS ENGINEERING, DOCUMENTATION, TEST FOR ESTABLISHMENT OF FIRST SYSTEM, ETC.	250.0	–	–
BASIC OPERATIONAL & PROCESSING SOFTWARE	200.0	NO APPLICATIONS SOFTWARE	IMAGE ANALYSIS SOFTWARE NOT INCLUDED
MISC. INSTALLATIONS HARDWARE & SUPPLIES	–	60.0	NOMINAL SHIPPING & INSTALLATIONS COSTS ASSUMED
ESTIMATED TOTAL COSTS	1266.6[1]	360.0[2]	ENHANCED II PROCESSOR & DISPLAY SUBSYSTEM

1. SUBCONTRACT $490K FOR STRIPPED DOWN, $379K FOR BASIC, & $181K FOR ENHANCED I MODELS.
2. SUBCONTRACT $243K FOR STRIPPED DOWN, $210K FOR BASIC, & $ 97K FOR ENHANCED I MODELS.

SUPPORTING SYSTEM
TRADE STUDIES

6 - SUPPORTING SYSTEM TRADE STUDIES

6.1 ORBIT ALTITUDE SELECTION

Purpose:

This study provides the rationale for selecting an EOS Mission altitude.

Summary:

TM overlap, TM revisit, HRPI revisit and Shuttle spacecraft payload capability are considered in order to narrow the field of viable orbit altitudes. When considered in this manner and checked against the effect of aerodynamic drag, the recommended orbit altitude for a mission employing a 100 nm TM swath width is 366 nm. The recommended altitude range of 365 to 385 nm will satisfy EOS coverage requirements with TM swath widths up to 235 nm (435 Km).

Conclusions and Recommendations:

The following material was developed in Report No. 1 -

With 100 nm selected for the TM swath width, and orbit altitudes constrained to the range 300 to 500 nm, Figure 6.1-1a shows all the available orbits when the repeat cycle time is 16, 17, or 18 days. The recommended orbit altitude, 366 nm, is chosen by the process outlined in Figure 6.1-1c. First, TM adjacent swath overlap of 10 to 20 nm and revisit intervals within 4 or less days reduces the available orbit altitudes to 12 in number. Then, to obtain an assured HRPI revisit interval of 5 days or less, all but 4 are eliminated (see Figure 6.1-1b). To retain a reasonable shuttle payload capability only orbit altitudes under 400 nm are retained. This leaves only 346 and 366 nm as viable EOS orbit altitude candidates. Altitude 366 nm is favored since (1) orbit decay from aerodynamic drag and (2) tracking coverage intervals are improved with greater altitude. Other repeat cycle times allow EOS altitudes in the range 365 to 385 nm with the TM swath width increased to up to as much as 235 nm.

Figure 6.1-1d shows the orbit decay resulting from aero drag during the first 6 months for both a nominal and a nominal +2 atmosphere (Jacchia Model). The sideslip in the longitude of the orbit node for the initially 366 nm altitude orbit appears in Figure 6.1-1e. If corresponding swaths are permitted to accumulate a nodal sideslip up to $\pm$ 20 nm, this may take 1.25 to 3 months to achieve, depending on the severity of the atmospheric drag. Figure 6.1-1f shows the Δ V need for each orbit adjust, 0.3 fps for the nominal atmosphere and 0.8 fps for the nominal + 2 σ . At 3 month intervals over 2 years, 7 orbit adjusts are needed for a total of 2.1 fps. At 1.25 month intervals, 19 adjusts require a total of 15.2 fps. The frequency of orbit correction is more likely to be governed by the nominal, and therefore more expected atmosphere. For purposes of mission reliability, however, the Δ V budget should reflect the needs of the more severe atmosphere.

Satisfactory behavior under aero drag in addition to survivability under the prior eliminating factors drives the recommended EOS orbit altitude to 366 nm for the 100 nm TM swath width.

6.2 LAUNCH VEHICLE SELECTIONS

Purpose:

This study determined the payload insertion capability of launch vehicle which show promise as feasible EOS boosters.

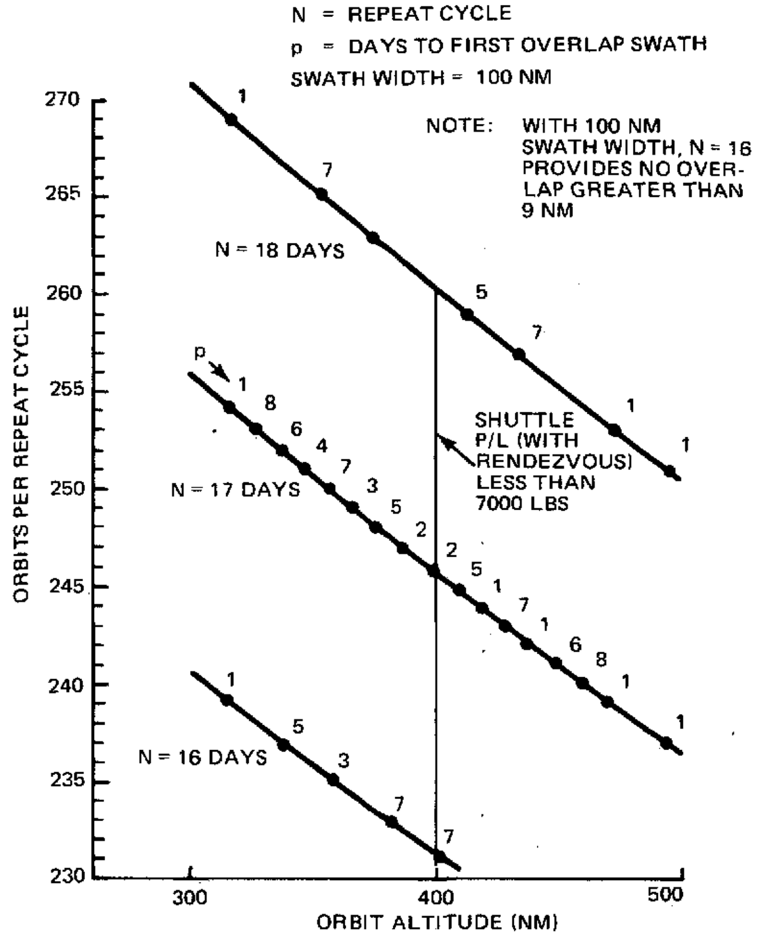
Summary:

The EOS-A, B and C missions can utilize the Delta 2910, a Constrained Titan, or a Titan III B (SSB) launch vehicle, depending on the program option selection. When available, the Shuttle will be capable of inserting any of the proposed EOS configurations into the candidate 366 nm orbit.

Conclusions and Recommendations:

The various EOS configurations, when taken with all and with none of their program options fall within the payload weight range 1951 to 6406 lb. Included are the weights of either a launch adapter or a flight support system and, where required, the weight of an apogee kick motor. Performance of the recommended booster is shown in Figure 6.2-1.

A. Candidate Orbit Altitudes



B. Days to Assure a HRPI Revisit+

ORBIT ALT (NM)	REPEAT CYCLE DAYS	0°	25°	48°
439	17	4	4	4 (3)
418	↓	5 (3)*	3	3
397		9 (7)	9 (7)	3
387		9	9 (7)	5 (3)
366		5	5 (3)	3
346		5 (4)	4	4
317	17	14	14 (13)	12 (11)
493	18	13 (12)	12 (11)	9 (8)
372	18	13 (12)	12 (11)	9 (8)

*NUMBER IN PARENTHESIS SHOWS DAYS TO ASSURE HRPI REVISIT TO AT LEAST 90% OF HRPI ACCESSIBLE TARGETS IF THAT OCCURS EARLIER THAN THE ASSURED ASSURED 100% DATE

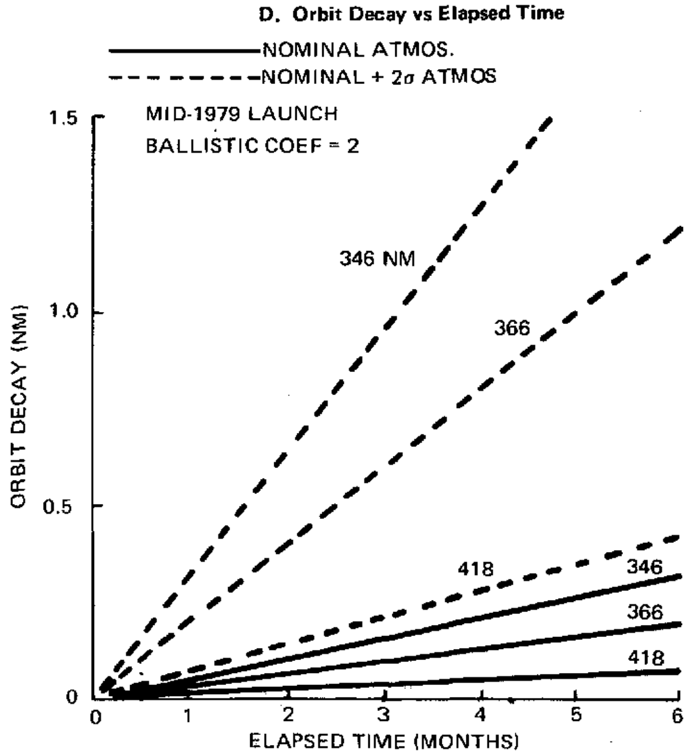
+MAXIMUM HRPI OFFSET ANGLE = 30°

C. Orbit Altitude Selection Process

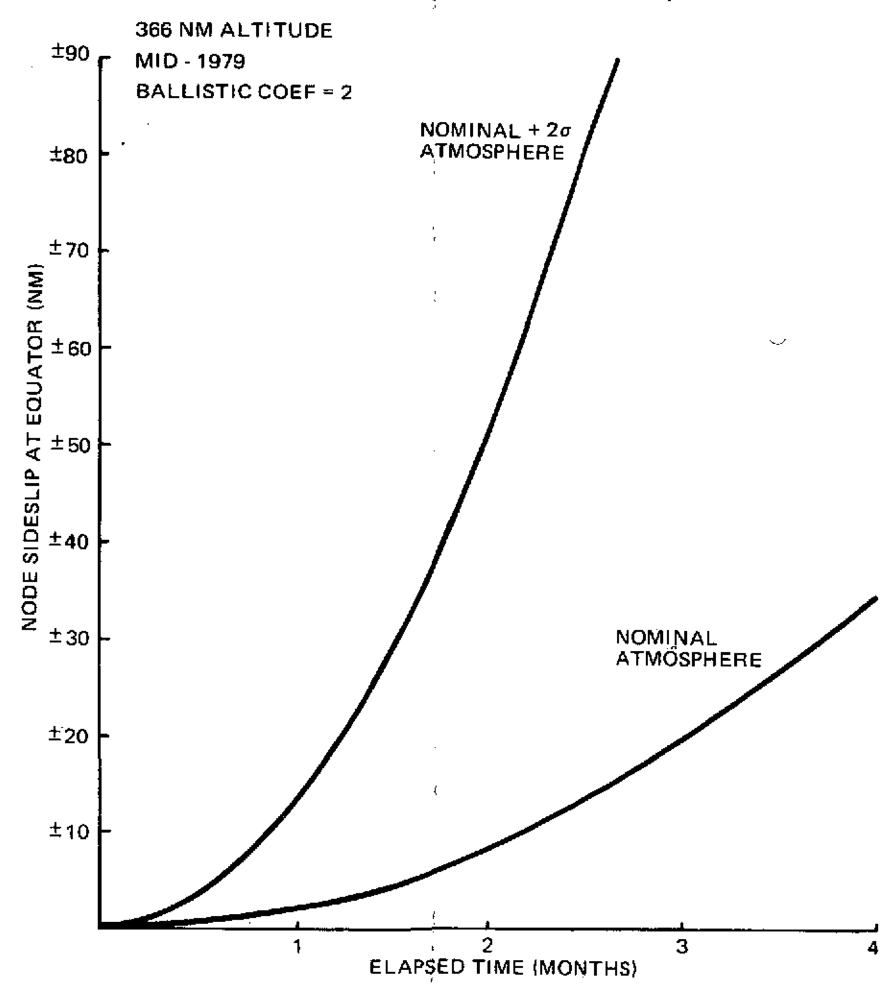
NO. OF CANDIDATE ORBIT ALTITUDES REMAINING

- EOS CONSTRAINTS (SEE FIG A)
  SUN SYNCHRONOUS
  300 NM < ALTITUDE < 500 NM
  10 NM < ADJACENT SWATH OVERLAP < 20 NM
  100 NM SWATH WIDTH
  N = 17 DAYS REPEAT CYCLE — 17
  N = 18 DAYS REPEAT CYCLE — 7
- TM ADJACENT SWATH REVISIT TIME ≤ 4 DAYS
  ELIMINATES p > 4
  N = 17 — 9
  N = 18 — 3
- HRPI (30° OFFSET) ASSURED REVISIT (SEE FIG. B)
  N = 17 — 4
  N = 18 — 0
- SHUTTLE P/L vs ALTITUDE CAPABILITY
  ELIMINATES ALTITUDES > 400 NM
  N = 17 — 2
- OF THE TWO REMAINING ALTITUDES, 346 AND 366 NM, ORBIT DECAY AND TRACKING COVERAGE FAVOR THE HIGHER ALTITUDE — 1
- SUGGESTED EOS ORBIT ALTITUDE = 366 NM

D. Orbit Decay vs Elapsed Time



E. Node Sideslip due to Orbit Decay



F. Velocity Increment Required for Orbit Adjust Due to Orbit Decay

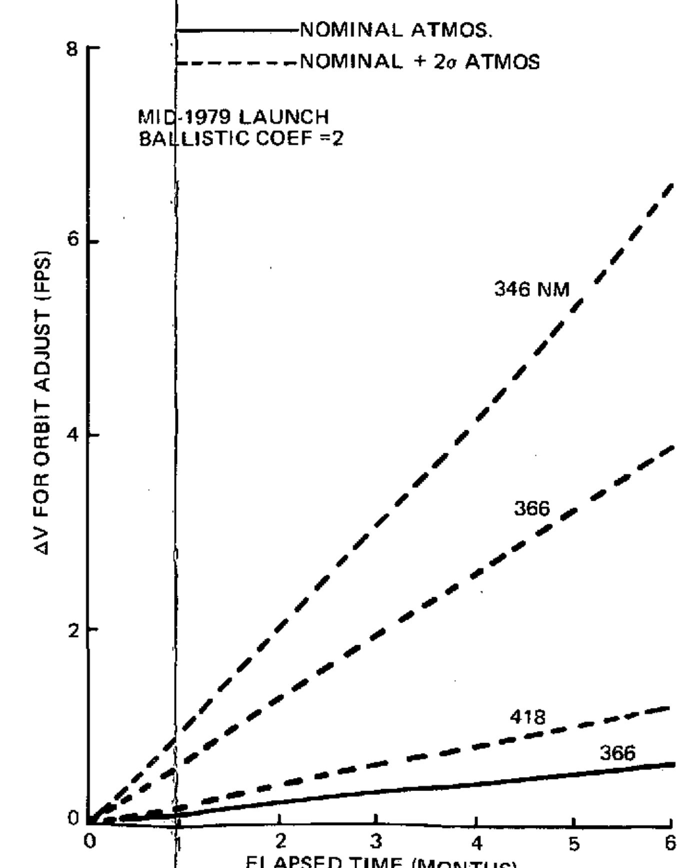


Fig. 6.1-1 Orbit Altitude Selection

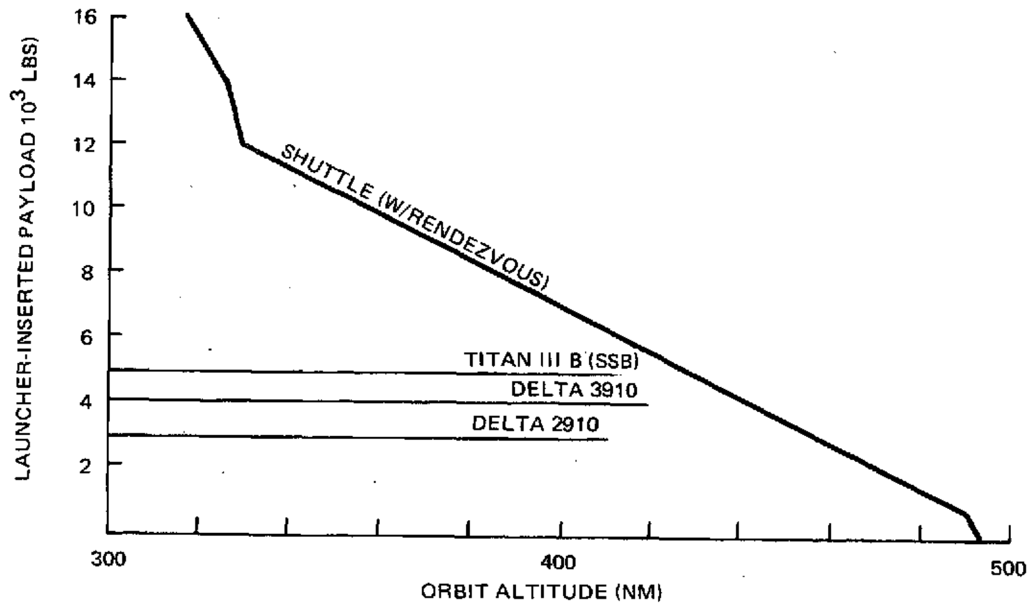


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Fig. 6.2-1 Performance of Recommended Booster

The non-Shuttle EOS-A mission, depending on the choice of program options and the extent of contingency weight actually required to complete the design will range from 1951 to 2612 lb. A Delta 2910 can launch and circularize at 366 nm, 98° inclination, a payload weight up to 2660 lb and therefore, this launch vehicle is the recommended booster for the EOS-A mission. The EOS-B weight ranges from 2373 to 3319 lb. The lower weights can be handled by the Delta 2910; the higher weights by a DELTA 3910 whose maximummum payload capability at 366 nm is 3730 lb. The non-Shuttle EOS-C weight range is 4016 to 5130 lb and its suggested launch vehicle is the Titan III B (SSB) with a minimum throw weight of 5150 lb into this orbit. When flown on the Shuttle in a deploy/retrieve mission the weight range spread is 3521 to 6406 lb. This is easily accommodated by the Shuttle, as may be seen by the bar chart on Figure 6.2-2. A resupply mission with the Shuttle, with payload range 5813 to 8684 lb, is also well within the Shuttle's 9600 lb lift-circularize-and-rendezvous capability at the 366 nm altitude.

6.3 SHUTTLE COMPATIBILITY

Purpose:

The purpose of this study was to identify the design requirements and associated cost impacts of using the Shuttle for EOS delivery, and the additional impact of achieving full compatibility for resupply and retrieval.

Conclusions:

EOS - Shuttle compatibility can be attained for all missions at reasonable impact to spacecraft design and cost. Excluding Missions E and F, all missions lie within the inherent performance capabilities of the Shuttle. Mission F (SEOS-A) requires a Tug for Shuttle compatibility, while Mission E (TIROS-O) can be accomplished using either an integral EOS Orbit Transfer System (OTS) or a Tug. Impacts for Shuttle delivery and retrieval are minimal except for the Mission E peculiar OTS. Resupply entails a significant impact, approximately 200 lb and $2 million non-recurring/$430K recurring for Missions A-B, reflecting module/assembly replacement mechanisms, a Shuttle demonstration model, and associated Engineering. The Shuttle Demo Model spacecraft, the System Qualification Spacecraft updated to flight status, is deemed necessary only for in-flight verification of resupply mechanisms and techniques. Weight penalties associated with Shuttle compatibility for any mode do not preclude initial delivery by assigned launch vehicles for any mission configuration.

Discussion:

Study scope was limited to defining the impact of configuring the EOS for physical compatibility with the Shuttle. Table 6.3-1 lists the visible weight and cost impacts for each spacecraft functional area, mission application and projected Shuttle utilization mode. These impacts assume a single spacecraft program and reflect a three subsystem module spacecraft design, the baseline Flight Support System (FSS), and the Module Exchange Mechanism (MEM) resupply concept. Of the eight missions included in the mission model, only Missions E and F require performance beyond the inherent capabilities of the Shuttle. For Mission F (SEOS), the required additional performance increment is too large to be accommodated by an orbit transfer system integral to the EOS. Accordingly, it was assumed that a Tug would be available and, for simplicity, it was further assumed that additional interface provisions were not required. On the other hand, Mission E could be accommodated by a Tug, if available, or by a moderately sized EOS orbit transfer capability (i.e. an integral SRM kick stage). The kick stage has been included in the impact assessment. The baseline FSS employs the same mechanisms for deployment and retrieval, accounting for the consistency in weight impact between the two modes. Resupply impacts reflect a significant increase, resulting from the addition of structural attach/release mechanisms, wiring disconnects, and a Shuttle Demonstration Model. Impact

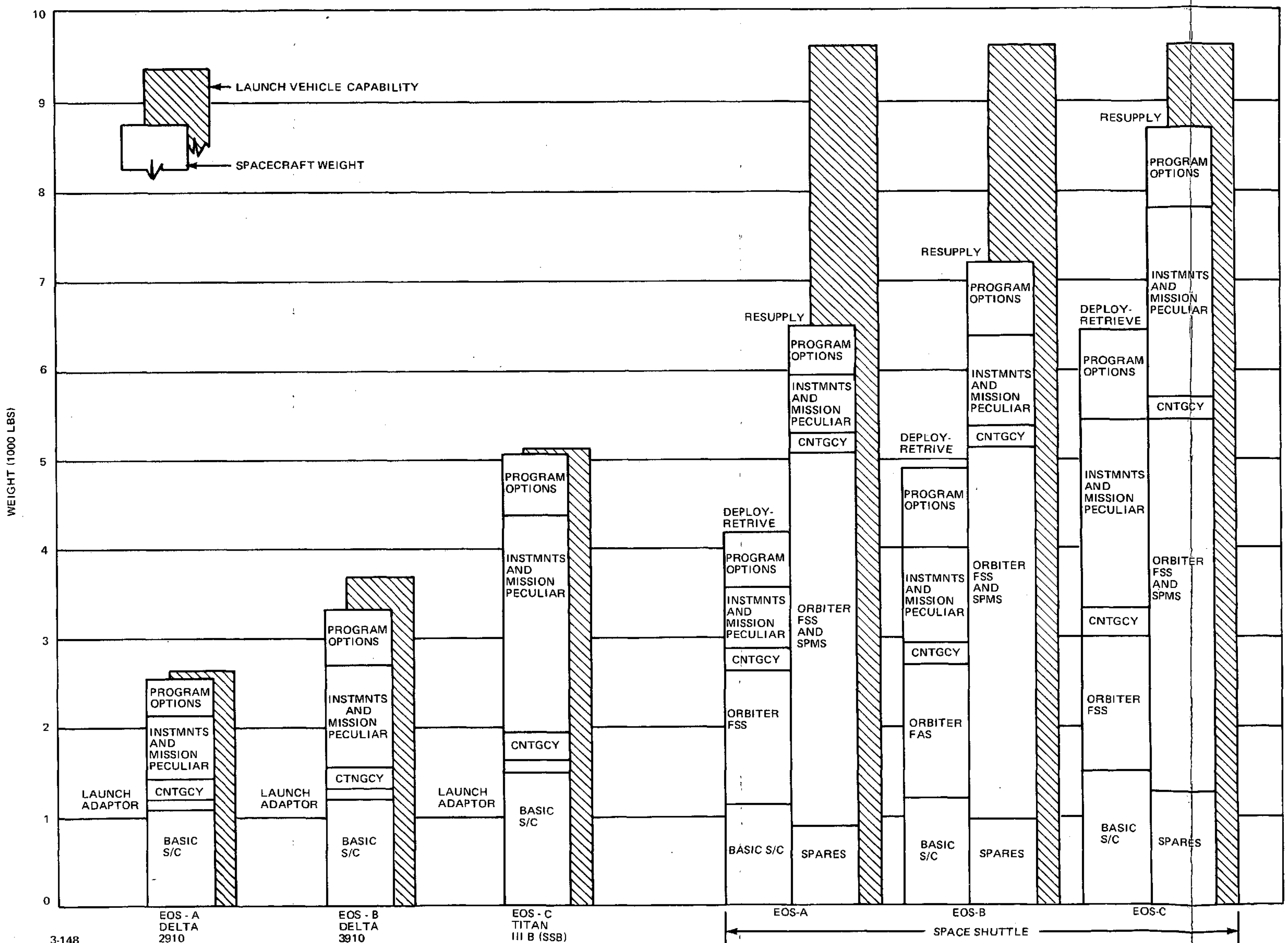

Fig. 6.2-2 Launch Vehicle Capability

FOLDOUT FRAME 1

FOLDOUT FRAME 2

C-5

3

6-9

Table 6.3-1 Shuttle Compatibility Impact Assessment

CONSIDERATION	MISSION A	MISSION A	MISSION B	MISSION B	MISSION C	MISSION D	MISSION E	MISSION F	DELIVER WT (LB)	DELIVER COST ($K) NON-REC	DELIVER COST ($K) RECUR	RETRIEVE WT (LB)	RETRIEVE COST ($K) NON-REC	RETRIEVE COST ($K) RECUR	RESUPPLY WT (LB)	RESUPPLY COST ($K) NON-REC	RESUPPLY COST ($K) RECUR
COMM & DATA HANDLING																	
• C & W INTERFACE	X	X	X	X	X	X	X	X	2	INSIG	INSIG	2	INSIG	INSIG	2	INSIG	INSIG
ELECTRICAL POWER																	
• SOLAR ARRAY STOW	X	X	X	X	X	X	X	X				1	74	2	1	74	2
• MODULE INTERFACE CONNECTORS/ RECEPTACLES	X	X	X	X	X	X	X	X							53	157	90
ALTITUDE CONTROL																	
• ANALOG PROCESSOR	X	X	X	X	X	X	X	X				1	55	40	1	55	40
STRUCT/MECH/THERM																	
• CRADLE ATTACH FITTINGS	X	X	X	X	X	X	X	X	36	126	61	36	126	61	36	126	61
• DOCK/DEPLOY TABLE PROBES	X	X	X	X	X	X	X	X	2	8	3	2	8	3	2	8	3
• LATCHES/PINS																	
– BASIC S/C	X	X	X	X	X	X	X	X							38	138	65
– INSTRUMENTS	X	X	X	X											26	94	44
					X										43	155	73
						X									46	166	78
							X								36	130	61
								X							18	65	31
• ROLLERS/ TRACKS																	
– BASIC S/C	X	X	X	X	X	X	X	X							14	51	24
– INSTRUMENTS	X	X	X	X											77	25	12
					X										10	36	17
						X									7	25	12
							X								6	22	10
								X							3	11	5
• PROPULSION																	
– PRESSURE RELIEF	X	X	X	X	X	X	X	X	2	33	5.4	2	33	5.4	2	33	5.4
– REDUNDANT S/O VLV	X	X	X	X	X	X	X	X				1	0	3.5	1	0	3.5
– KICK STAGE							X		548	300	120	1589	300	240	1589	300	240
SHUTTLE DEMO MODEL	X	X	X	X	X	X	X	X							N/A	265	N/A
SYSTEM ENG'G & INTEG	X	X	X	X	X	X	X	X	–	120	0	–	224	0	–	428	0
REL & QUAL	X	X	X	X	X	X	X	X	–	0	0	–	160	80	–	320	80
TOTAL	X	X	X	X					42	287	69	45	680	195	183	2039	430
					X				42			45	680	195	203	2111	464
						X			42			45	680	195	203	2111	464
							X		590	587	189	1634	980	435	1781	2390	445
								X	42	287	69	45	680	195	171	1996	410

variations among missions reflect different Instrument complements for which resupply mechanisms must be provided.

Ground operations were not considered in this assessment since they are dependent upon flight frequency and traffic density. The full spectrum of effects will be covered in Report No. 6, "Space Shuttle Interfaces/Utilization".

Development of the design and cost impacts of Shuttle compatibility itemized in Table 6.3-1 began with identification of functional requirements beyond those for EOS spacecraft configured for conventional launch vehicles as shown in Fig. 6.3-1. Differences were considered for each functional area of the spacecraft, as well as for the Instruments and Operations Potential variations between mission concepts were considered, but became evident only when performance augmentation was required, i.e., for Mission E. Shuttle utilization modes were considered in order of increasing operational complexity, hence a requirement cited for Deliver also applies to Retrieve and Resupply unless there are unique circumstances.

As shown in Fig. 6.3-1, the functional requirements were translated into specific hardware changes. For example, Propulsion requirement 3 cites replacement of the propulsion (i.e. combined RCS, OAS, OTS) module. The corresponding hardware requirements entail attach release mechanisms, signal connectors, and power connectors. These design implications were the basis for estimating the weight and cost impacts cited in Table 6.3-1. For this phase of study, Instruments and Operations were considered only to the extent that basic S/C design was affected. In addition, provisions for Instrument appendage retraction and/or replacement were assumed to be inherent in Instrument design.

Study analyses are detailed in Appendix E, Section 2.3, to this document.

6.4 INSTRUMENT APPROACH

Purpose:

a) To evaluate the competitive point designs provided for the proposed instruments; thematic mapper, high resolution pointing imager, synthetic aperture radar, and passive multichannel microwave radiometer.

b) To evaluate overall system designs applicable to the EOS-A instrument package.

c) To evaluate the utility, reliability, and costs related to each sensor point design proposed for EOS-A sensors.

Table 3.2.3-4 (Cont)
SHUTTLE COMPATIBILITY - REQUIREMENTS IMPACT

Trade Study Report 3.2.3
WBG Number 1.2.1.4.4

CONSIDERATION	MISSION APPLICATION A	A'	B	B'	C	D	E	F	REQUIREMENT DELIVER	RETRIEVE	RESUPPLY	REMARKS
1.5 PROPULSION	X	X	X	X	X	X	X	X	1. PROVIDE FOR PROPELLANT TANK PRESSURE RELIEF			
							X	X	2. PROVIDE FOR EOS TRANSFER FROM PARKING ORBIT TO MISSION ORBIT	2. SAME PLUS PROVIDE FOR TRANSFER FROM MISSION ORBIT TO ORBITER PARKING ORBIT		INTEGRAL EOS CAPABILITY REQUIRED ONLY IF TUG IS UNAVAILABLE
	X	X	X	X	X	X	X	X			3. PROVIDE FOR ON-ORBIT REPLACEMENT OF PROPULSION MODULE	RCS PNEUMATICS, GAS, AND/OR OTS, AS NEEDED FOR MISSION, CONTAINED IN COMMON MODULE
2.0 INSTRUMENTS	X	X	X	X	X	X	X	X		1. PROVIDE FOR RETRACTION OF ALL DEPLOYABLE ELEMENTS		
											2. PROVIDE FOR ON-ORBIT REPLACEMENT OF INSTRUMENT MODULES/ ASSEMBLIES	
3.0 OPERATIONS 3.1 FLIGHT OPERATIONS	X	X	X	X	X	X	X	X			1. PROVIDE FOR IN-FLIGHT DEMONSTRATION OF EOS SERVICING/RESUPPLY	ASSUMES THAT DEPLOYMENT AND RETRIEVAL TECHNIQUE DEMONSTRATED WITH PRIOR S/C

Table 3.2.3-4 (Cont)
SHUTTLE COMPATIBILITY - DESIGN IMPLICATIONS

Trade Study Report 3.2.3
WNG Number 1.2.1.4.4

CONSIDERATION	MISSION APPLICATION A	A'	B	B'	C	D	E	F	DESIGN DELIVER		RESUPPLY	REMARKS
1.5 PROPULSION	X	X	X	X	X	X	X	X	1. PRESSURE RELIEF A. ADD GN_2 PRESSURE RELIEF ASSEMBLY TO PROPELLANT TANK PRESSURANT STORAGE B. ADD GN_2 VENT TO ORBITER P/L BAY.			
							X		2. ORBIT TRANSFER A. ADD SRM'S TO RAISE SPARES FROM 300 N MI TO 915 N MI B. ADD SRM'S TO CIRCULARIZE AT 915 N MI	2. ORBIT TRANSFER A. SAME PLUS ADD SRM TO LOWER APOGEE FROM 915 N MI TO 300 N MI AND CIRCULARIZE		MISSION F UTILIZES A TUG. INTERFACES ARE ASSUMED CONSISTANT WITH INITIAL LAUNCH VEHICLE REQUIRING NO ADDITIONAL PROVISIONS
	X	X	X	X	X	X	X	X			3. REPLACE MODULE A. ADD STRUCTURAL ATTACH/ RELEASE MECHANISMS B. PROVIDE UNOBSTRUCTED LATERAL MOTION C. ADD SIGNAL/POWER DISCONNECTS	
2. INSTRUMENTS	X	X	X	X	X	X	X	X		1. APPENDAGE RETRACTION NOT CONSIDERED AT THIS TIME		FOR THIS PHASE OF STUDY, ONLY THE LATCH MECHANISMS HAVE BEEN CONSIDERED
											2. REPLACE INSTRUMENTS A. PROVIDE STRUCTURAL ATTACH/RELEASE MECHANISMS	

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Figure 6.3-1 Impact Evolution-Functional Requirements Imply Design Changes

d) To provide an evaluation of the available data collection system and recommendations to increase its utility when used on EOS if applicable.

e) To provide designs compatible with later EOS missions with regard to the follow-on instruments and to identify the operational and cost impacts of providing this capability.

Summary:

During the course of the study, a broad range of considerations were addressed in selecting the most useful, reliable and high growth potential instrument designs and data handling concepts.

Because of direction received during the study and the continual development of the various TM and HRPI point designs during the study, effort was concentrated on the further evolution of the TM and HRPI relative to the ERTS MULTI-SPECTRAL scanner and their various configurations and utilizations in the EOS-A mission.

The result of these studies are as follows:

1. No single point design is considered optimum in the form proposed by the suppliers.

2. The object plane scanner as a class offers significant growth potential relative to the EOS baseline without significant weight growth.

3. Spectral band selection by filtration techniques offers significantly more between potential than does the spectrometer (dispersion) approach.

4. The reduction in preamplifier noise by cooling down to 200°K promises performance improvements for silicon detectors even in Band 1, which makes them highly competitive with photo-multiplier tubes.

5. The lower cost, higher reliability, simpler design, lighter weight and higher growth potential of an all solid state detector array make this the preferred approach even if a slightly larger telescope aperture is felt necessary to meet minimum S/N ratio requirements.

6. In the land resources mission, the need for maximum radiometric data accuracy requires that the data transmission system sample the data stream once per pixel.

7. There are significant economies in obtaining the TM and HRPI from the same supplier due to a possible commonality factor as high as 80%.

8. A new TM has been defined which can provide a 330 KM swath at 27 meters resolution, provide an output at 80 meters completely compatible with and providing a backup to the operational MSS, and providing a pseudo-HRPI output covering a selectable 35 kilometer swath at 30 meters. Both the MSS backup and pseudo-HRPI signal would be compatible with the present DOI and planned Low Cost Ground Stations. (LCGS).

9. Only 6-bit encoding of the data is required. Provision for modification of the dynamic range of the data encoders can provide higher quality data at less cost.

10. As the land resources mission (LRM) matures, the desirability of obtaining stereo coverage will increase and a ± 50 N. M. drift in the orbit repeat cycle prior to orbit adjust will become preferred orbit.

All of the studies associated with the instruments assumed a nominal satellite altitude of 680 kilometers.

6.5 DATA OPERATIONS

6.5.1 DATA OPERATIONS COST/THROUGHPUT MODEL

Data operations refers to the activity and costs related to the establishment and operation of the CDP. These are impacted by a number of parameters (drivers) among which are the daily data volume, the level of processing of this data, the number of users and the amount of output products, and the data output format. In order to analyze the impact of these and other parameters on the cost of configuring and operating a CDP, a cost/throughput model of the CDP was constructed and reduced to a computer program. This model was exercised for two CDP configurations by varying the parametrics. Conclusions were drawn (documented below) based on the results of this model run.

Figure 6.5-1 is a flow chart of the cost/throughput model. Table 6.5-1 lists the cost estimating relationships used in the model.

6.5.2 LOAD FACTORS AND LEVELS OF PROCESSING: EXAMPLE

Load factors for the product processing load on the CDP are based on the number of users, a "replication factor" and the number of scenes of data processed daily.

For purposes of the cost-throughput model, the data product loadings were calculated as the product of a base data volume and the replication factor. The replication factor can be regarded as the product of the number of users and the average fraction of the base data volume ordered by each user. For example, 100 users each getting an average of 1 percent of the data would result in a replication factor of 1. This also assumes that the extent of overlap in the selection of images is not a factor. (If copying configurations were designed to make simultaneous multiple copies from the same original, then the degree of multiplicity in user ordering would become a relevant parameter in the calculation).

The base data volume for HDDT was taken as the sum of the number of images produced by Level 1, 2, and 3 processing. Note that this can never exceed twice the number of images originally acquired because the Level 2 and Level 3 processes are assumed to be mutually exclusive alternatives.

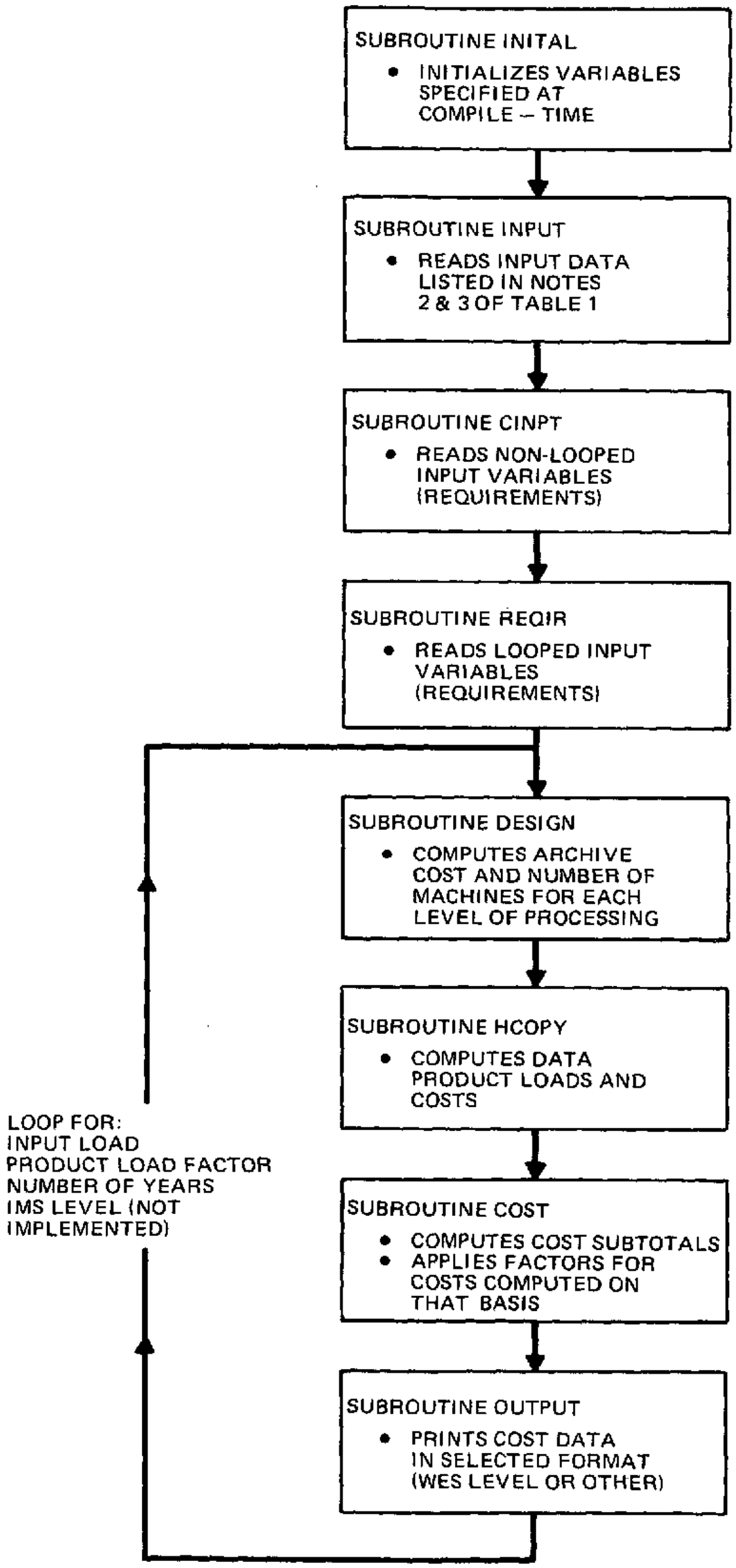

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Fig. 6.5-1 Cost/Throughput Model

Table 6.5-1 Cost Throughput Model Estimating Relationships

ITEM	BASIS	ESTIMATING FACTORS: MINI SYSTEM	ESTIMATING FACTORS: MAXI SYSTEM
PROCESSING COSTS	TOTAL MIPS	3	10
	BYTES PER PROCESSOR ACCESS	2	4
	EQUIP INSTRUCTIONS PER 1/10 ACCESS	1/4	0
	PROCESSOR COST	$150	$3M
	MEMORY TRANSFER RATE	1M BYTES/SEC	10M BYTES/ SEC
	BYTES PER MEMORY ACCESS	1	1
	STORAGE COST	$500	$3M
	FLOOR AREA	100 SQ. FT.	500 SQ. FT.
	POWER CONSUMPTION	2K WATTS	30K WATTS
	PEOPLE PER SHIFT	1	2
PHOTO ORIGINALS	MATERIALS COST	SAME AS B&W TRANSPARENCIES	
	LASER BEAM RECORDER	$350K	
	TIME PER IMAGE	20 SEC	
	FLOOR AREA	30 SQ. FT.	
	POWER CONSUMPTION	6 K WATTS	
	PEOPLE PER SHIFT	1	
PHOTO COPIES	COST PER ITEM		
B&W TRANSPARENCIES		$3.00 EACH	
B&W PRINTS		$1.25 EACH	
COLOR TRANSPARENCIES		$4.00 EACH	
COLOR PRINTS		$3.00 EACH	
HDDT COPIES	IMAGES PER REEL	200	
	COST PER REEL	$150	
	COPY TIME PER REEL	600 SEC	
	COPIER COST	$500K	
	FLOOR AREA	30 SQ. FT.	
	POWER CONSUMPTION	6K WATTS	
	PEOPLE PER SHIFT	1	
CCT COPIES	IMAGES PER REEL	2	
	COST PER REEL	$9	
	COPY TIME PER REEL	100 SEC	
	COPIER COST	$175K	
	FLOOR AREA	30 SQ. FT.	
	POWER CONSUMPTION	6K WATTS	
	PEOPLE PER SHIFT	1	
FACILITIES	FLOOR AREA	200 SQ FT PER PERSON	
	EQUIPMENT & PERSONNEL	$55. PER SQ FT	
SPARE PARTS	HARDWARE COST	15% ANNUAL SPARES	
TOOLS	HARDWARE COST	5% OF TOTAL	
PERSONNEL	NASA EXPERIENCE AVERAGE BURDENED COST	$30K/PERSON/YR. 2 SHIFTS	
DOCUMENTATION	HARDWARE & SOFTWARE COST	20% OF HDWRE 35% OF SOFWRE	
POWER & UTILITIES	EQUIPMENT PLUS 7.5 WATTS/SQ FT. OF FLOOR AREA (LIGHT, HEAT, AIR-CONDITION)	$200 PER KW YR	
SYSTEM ENGINEERING & INTEGRATION	HARDWARE COST	35% OF HDWRE	
PROJECT MANAGEMENT	TOTAL INITIAL & ANNUAL COSTS	10% OF TOTAL	
SOFTWARE			
INFORMATION MANAGEMENT SYSTEM	TOTAL SYSTEM COST	10% OF TOTAL	

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The base data volume for all other products was taken as the number of images originally acquired.

The following cases were run:

Product	Replication Factor Case 1 = Minima	Replication Factor Case 2 = All Maxima
HDDT	2	10
CCT	0.01	1
B&W Originals	1	1
B&W Transparencies	0.1	1
W&W Prints	0.1	1
Color Transparencies	1/60	1/6

The number of scenes of data processed daily can be considered a combination of TM & HRPI scenes. These can be reduced to a common factor called "equivalent TM images" which is defined as one reflectance band of the 185KM swath, 30 meter resolution TM (viz, 6168x6168 = 3.8x10 pixels). All HRPI images are thus referenced to a multiple of 3.8x10 pixels for purposes of data loading.

Table 6.5-2a defines the three cases used in an example calculation. These represent a range in data load of from (approximately) 10^{10} to 10^{12} bits per day. Case 1 and Case 2

Table 6.5-2a Data Product Loads

	CASE A MIN	CASE A MAX	CASE A MIN	CASE A MAX	CASE C MIN	CASE C MAX
TM SCENES	20		45		200	
HRPI SCENES	0		45		200	
COMBINED PERCENTAGE LEVEL II & III	100		100		100	
TOTAL EQUIVALENT TM IMAGES	121		553		2458	
HDDT IMAGES (EQUIV TM)	242	1212	1106	5531	4916	24582
REELS	1.21	6	5.53	27.7	24.6	123
CCT IMAGES (EQUIV TM)	1.21	121	5.53	553	24.6	2458
REELS	0.61	61	2.77	277	12.3	1229
B&W ORIGINALS (TRANSPARENCIES)	121		553		2458	
B&W TRANSPARENCY COPIES	12	121	55	553	246	2458
B&W PRINTS	12	121	55	553	246	2458
COLOR TRANSPARENCIES	2	20	9	92	41	411
COLOR PRINTS	2	20	9	92	41	411

replication factors were used to determine minimum and maximum output products. Tables 6.5-2b and 6.5-2c list the processing levels used in computing the throughput in the sample case. (Appendix D Section D.2 defines the Levels of Processing and their algorithms in detail.)

Table 6.5-2b Processing Levels

ITEM	PREPROCESS	PROCESSING LEVEL 1	2	3	ARCHIVE
AVERAGE OPERATIONS PER PIXEL	10	10	30	100	1
PERCENTAGE OF DATA PROCESSED	100	100	TABLE 6.5.2c		100
PERCENTAGE OF DATA ARCHIVED	100	100	100	100	-
ARCHIVE PURGE PERIOD (MONTHS)	∞	1	3	4	-

Table 6.5-2c Percent Data Processed

CASE	LEVEL	
(a)	2	3
(A)	100	0
(B)	50	50
(C)	0	100

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6.5.3 DATA OPERATIONS SUMMARY AND CONCLUSIONS

Table 6.5-3a lists the five year total costs (non-recurring and recurring) for the example case as calculated by the cost/throughput model for a mini-computer system (configuration 1) and a general purpose processor system (configuration 2). Tables 6.5-3b lists the costs for output products based on the replication factor. Table 6.5-3c tabulates the processing equipment (digital and photographic) costs.

Table 6.5-3a Five-Year Total CDP Costs

($ MILLIONS)

PRODUCT MIX % LEVEL 3		SCENES/OUTPUT PRODUCT LEVEL 20 MIN	20 MAX	90 MIN	90 MAX	400 MIN	400 MAX
BILINEAR 0	CONFIG. 1	5.4	10.1	20.7	40.5	83.5	169.6
100		5.5	10.2	20.9	40.7	84.3	170.4
CUBIC CONVOLUTION 100		7.9	12.7	33.7	53.4	142.1	228.1
BILINEAR 0	CONFIG. 2	9.5	14.2	41.4	61.1	177.3	263.3
100		9.6	14.3	41.9	61.6	179.6	256.6
CUBIC CONVOLUTION 100		16.6	21.	77.0	96.8	339.1	425.1

Table 6.5-3b Five-Year Total Expendables Plus Product Copier

($ MILLIONS)

TOTAL SCENES	OUTPUT PRODUCT LEVEL	
	MINIMUM	MAXIMUM
20	1.48	4.92
90	5.51	21.19
400	23.30	92.97

Table 6.5-3c Total Image Processing Equipment Costs

($ MILLIONS)

TOTAL SCENES		% LEVEL 3		
		BILINEAR 0	INTERPOLATION 100	CUBIC CONVO-LUTION 100
20	CONFIG. 1	0.88	0.9	2.09
90	CONFIG. 1	4.01	4.1	9.55
400	CONFIG. 1	17.84	18.2	442.45
20	CONFIG. 2	2.40	2.5	5.76
90	CONFIG. 2	10.95	11.2	26.29
400	CONFIG. 2	48.68	49.7	116.82

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The following conclusions have been drawn from the operation of the cost/throughput model:

- Although there are nonlinear effects in the costing that would contribute to the establishment of breakpoints, their effects tend to be heavily diluted. Thus, the costs are generally linear functions of the various requirements parameters. Undoubtedly, many of the linear relationships would become non-linear when higher order effects are included, but these considerations represent a degree of refinement that is probably not justified by the quality of the estimating factors used.
- The number of user formats has a minimal impact on cost.
- The number of users does not by itself define the product processing load. The replication factor must be considered.
- A large scale general purpose computer configuration and a mini-computer configuration were considered in the analysis. The mini-computer was found to be uniformly lower in cost. However, for large data volume neither the mini nor the general purpose machine represents an economical solution beyond the R&D stage. It is expected that special purpose processors will afford up to 10 to 1 reduction in the costs of processing at these high data flows.

- The replication factor and the data load have a large impact on data processing expendables. These expendable costs can easily become a major cost driver on the CDP.

- The detailed processing mix and processing algorithms were found to be a very significant cost driver. For example, the results show that the image processing equipment costs more than double when cubic convolution interpolation is used as compared to bilinear interpolation.

6.6 ACS/CPF TRADEOFF

Purpose:

To determine the Attitude Control System performance requirements which result in the lowest ACS/Central Processing Facility cost for a program of selected missions, while at the same time providing flexibility for meeting varying mission requirements.

Conclusion:

The ACS considered to be best on the basis of lowest ACS/CPF cost and mission flexibility is the baseline system, which has the following performance requirements: pointing accuracy $\leq \pm 0.01$ degree and angular rate stability $\leq \pm 10^{-6}$ degree/second over 30 minutes.

Discussion:

As shown in Fig. 6.6-1, the ACS/CPF cost is minimum for ACS Configurations 1 (low cost) and 2 (baseline) for each of the three programs. These configurations have the following performance requirements:

Item	ACS Configuration 1	ACS Configuration 2
ACS Pointing Error, deg,	≤ 0.05	≤ 0.01
ACS Angular Rate Error, deg/sec, (1)	$\leq 5 \times 10^{-6}$	$\leq 10^{-6}$

(1) average over 30 minutes

Since ACS Configuration 2 has a performance which is 5 times better than that of ACS Configuration 1, ACS Configuration 2 is best on the basis of ACS/CPF cost and mission flexibility.

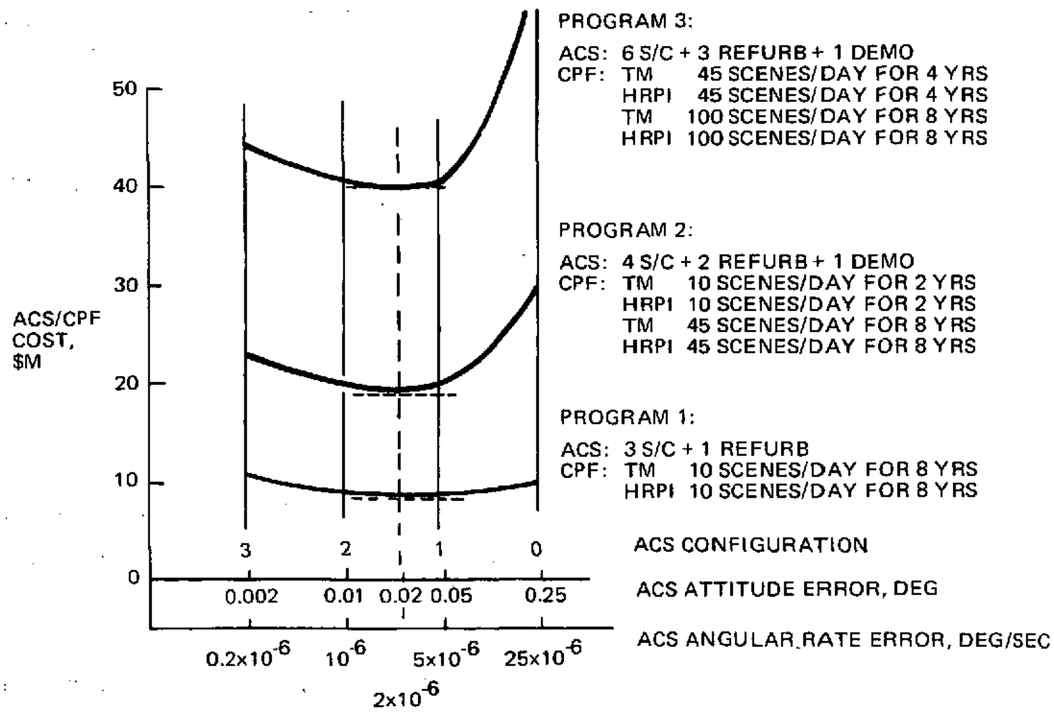

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Fig. 6.6-1 ACS/CPF Cost Vs. ACS Performance

The results on a per-spacecraft basis are similar. As shown in Fig. 6.6-2, with decreasing ACS performance, the ΔACS cost goes down and the ΔCPF cost goes up. ACS Configurations 3, 2, 1 and 0 have errors that are 0.2, 1, 5 and 25 times those of baseline at 0.01° and 10^{-6} $^{\circ}$/s. The net ΔACS/CPF cost decreases in going from ACS Configuration 3 to 2; remains approximately the same in going to ACS Configuration 1 and increases sharply in going to ACS Configuration 0. Thus the net ΔACS/CPF cost is lowest for ACS Configurations 1 and 2.

When the effects of increasing the number of scenes/day are examined, the results are again similar. As shown in Fig. 6.6-3, the recurring ACS hardware/manpower costs for one spacecraft are plotted at 0 scenes/day. The ACS/CPF cost increases from these points as the number of scenes/day increases from zero. When the number of scenes/day is below 20, ACS Configuration 0 is cost competititve with ACS Configurations 1, 2 and 3. When the number of scenes/day is higher than 20, ACS Configuration is not cost-competitive and ACS Configurations 1 and 2 are lowest in cost, with ACS Configuration 3 somewhat higher in cost.

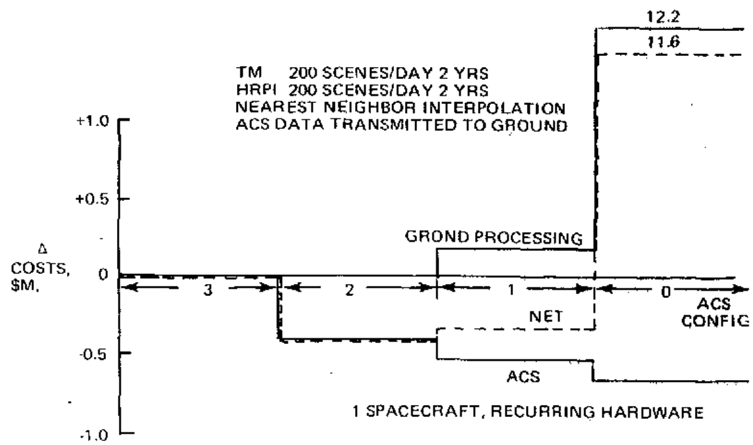

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Fig. 6.6-2 Δ Costs Vs ACS Configuration

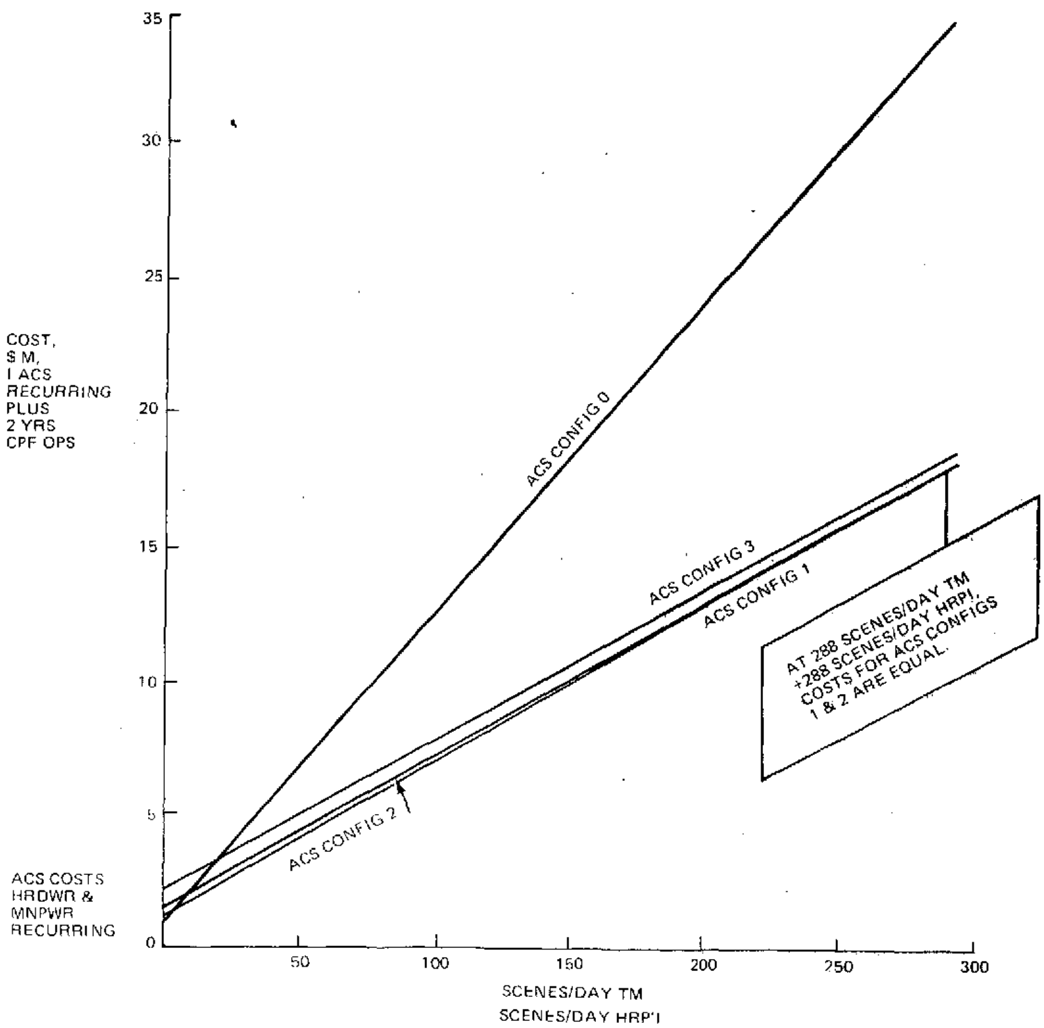

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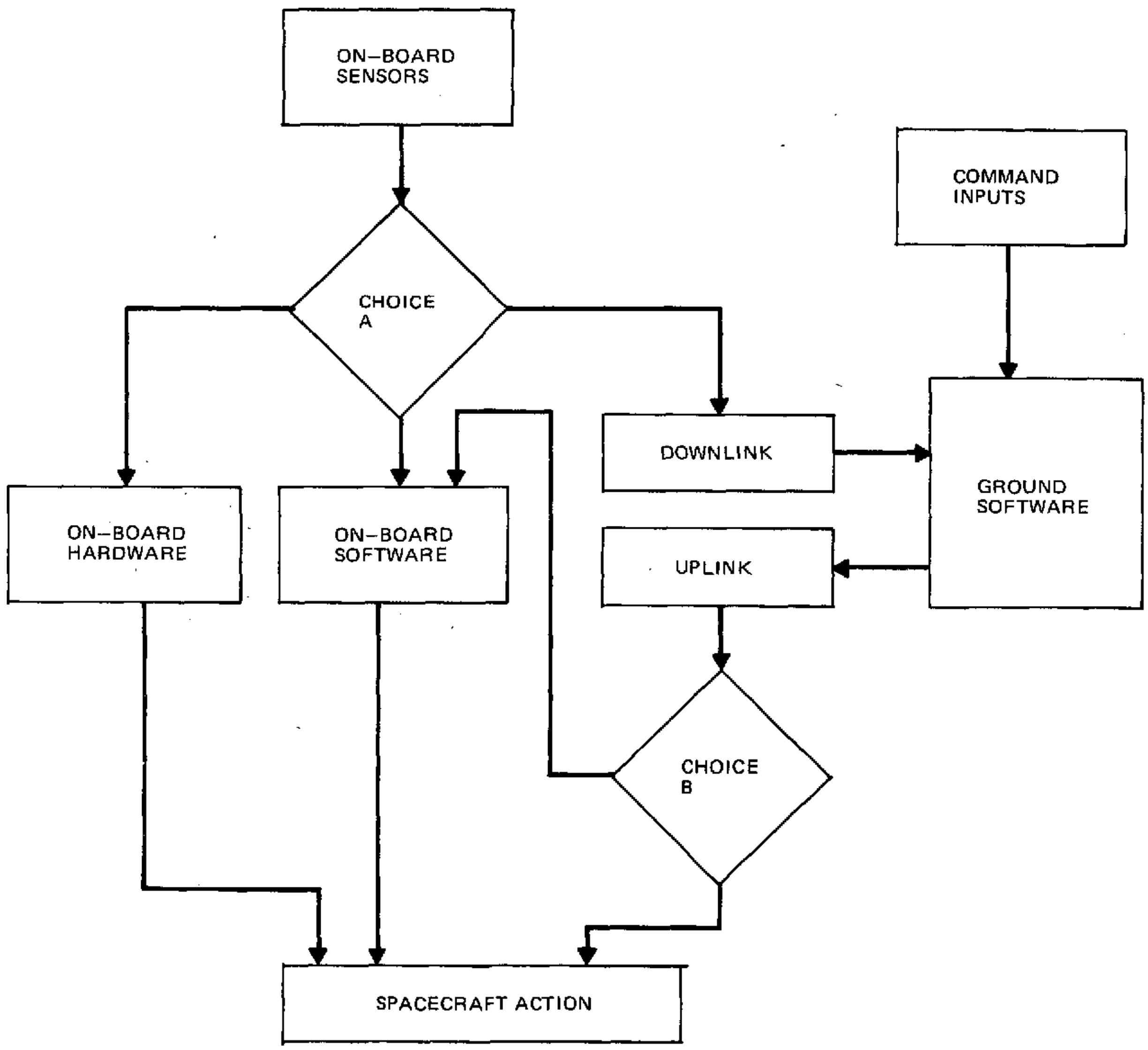
Fig. 6.6-3 Cost, A CS/CPF Vs Scenes/Day

6.7 SPACECRAFT AUTONOMY/HARDWARE VS. SOFTWARE

Purpose

The trade study of Spacecraft Autonomy and of Hardware versus Software is a function-by-function resolution of the choices illustrated in Figure 6.7-1. While each function of the spacecraft is a candidate for examination, care should be taken in the implementation of each autonomous function so as to assure that ground-control is not inhibited and remains available as a backup. This study considers representative functions, and on the basis of the choices, develops an on-board software budget which allocates computer memory space and computer running time of the functions. The result details the size and complexity of the recommended on-board software package.



3-158 **Fig. 6.7-1 Autonomy/Hardware/Software Trades**

Conclusions:

Of all the major functions considered, only one, the determination of spacecraft orbit parameters, is found to be inappropriate for on-board performance. The major element in this choice is the cost of facilities and manpower in the performance of ground computation, which, in turn, is at least partially controlled by the volume of uplink and downlink information to be handled.

Summary

The choice of implementation for the functions studies has fallen into four classes, which are listed here with the functions assigned to them:

Perform on-board with software
- RGA calibration (and other instrument calibration)
- Star Tracker data reduction
- Sub-satellite position computation
- Orbit counting
- Antenna Steering
- Solar array drive
- Attitude control

Perform on-board with hardware
- Thermal heater control

Perform on ground with on-board software implementation
- Orbit maneuvers

Perform on ground with data uplink to satellite
- Orbit determination

Some functions are sufficiently critical that a redundant choice is recommended as backup for the on-board software:

On-board attitude control by backup analog autopilot.

CHOSEN CONFIGURATION

The configuration which has been selected as an outcome of the trade studies is illustrated in Figure 6.7-2. The major features of this configuration are:

- Uplink commands are identified by latitude/longitude rather than time for most commands. This relieves a substantial effort otherwise required in ground simulation of orbit timing.
- Downlink wideband data are tagged with latitude/longitude/altitude and time. This relieves the need for on-the-ground correlation between ground orbit prediction and image processing computations.

- Downlink housekeeping data are compressed before transmission. This relieves the need for ground storage of large volumes of data and permits quicker quick-look satellite evaluation.

Commands from the ground station will normally require two forms: experiment identification data, and orbit control data. The experiments will be primarily identified by location coordinates and experimenter ID serialization. With this as an input, the time when the satellite is actually viewing the desired location can be easily determined as a real-time function in the on-board computer. The selection of ground stations for the receipt of downlink data can similarly be controlled on-board. Orbit control data will consist of a list of ground-determined orbit parameters which permit the on-board computer to compute its current location as a function of time. An orbit change will require the uplink of time-specified thrusting commands and the subsequent uplink of new orbit parameters.

Tagging of downlinked experiment data with position and time data will permit much more convenient scene identification for low-cost users, who would otherwise require detailed satellite ephemerides and precise timing for each orbit pass.

The primary on-board benefit of data compression before downlink is that no on-board tape recorder is required. This economy, added to the saving from reduced ground data reduction, results from the use of a digital filter which is also required for on-board stabilization routines.

RELIABILITY EFFECTS

The transfer of software functions from ground computers to the spacecraft computer does not affect the reliability of the spacecraft per se, since there is no modification to the spacecraft hardware. At some software size, however, it becomes necessary to expand the size of the on-board computer memory, which does modify the hardware. The memory to be added (assuming AOP implementation) is available in modules of 8K words. The 11. x 10^{-6} failures per hour estimated for an additional memory module has little impact on system reliability, however, because of the inherent redundancy of the AOP memory configuration.

EFFECT OF AUTONOMY ON MOCC

At the time of S/C launch there will be few (if any) critical S/C functions entrusted to the OBP (On Board Processor). But as the mission progresses and confidence in the

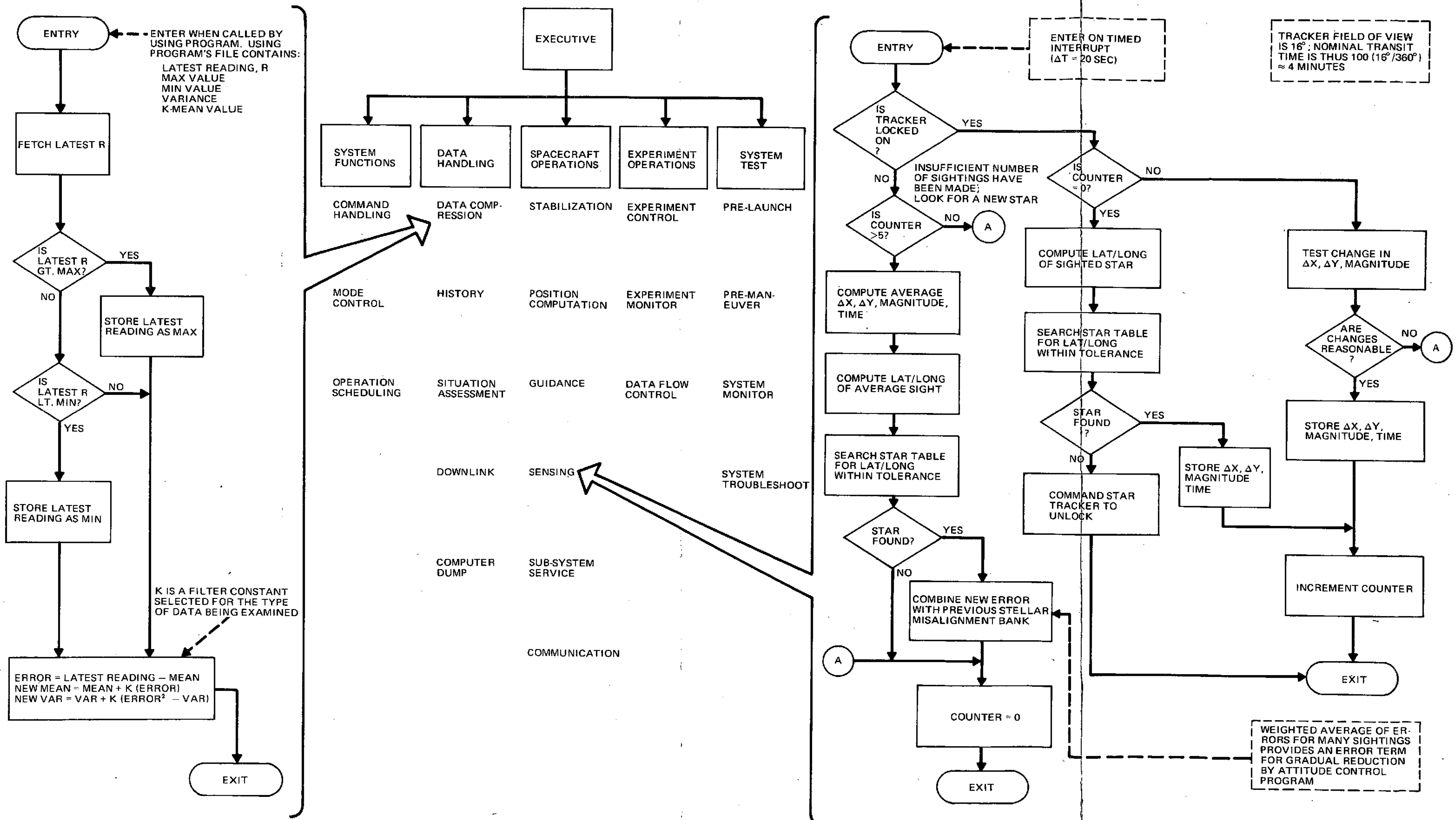


Fig. 6.7-2 On-Board Software Organization

OBP is developed it will be possible to off-load ground operations onto the OBP. Typical areas would be as follows:

(a) RGA calibration
(b) Star Tracker service
(c) Position Determination
(d) Antenna Steering
(e) Instrument Planning
(f) Command Memory Management
(g) Housekeeping Data Compression and Storage

As the OBP takes on more of the ground functions it should be possible to make some reduction. Referring to the MOCC organization diagram for on-going operations (see Appendix E.2.13), it should be possible (given a successful OBP capability) to eliminate four subsystem operators, four contact controllers, one SCPS liaison engineer, and one scheduler. This is a reduction of ten people, which is 18% of the original MOCC complement of 56 people.

COMPUTER SIZING

The memory requirements of the chosen configurations are presented in tabular form in Figure 6.7-3. The functions are presented in three groups in terms of their expected applicability to individual missions.

	FUNCTION	MEMORY (18-BIT WORDS)
BASIC	EXECUTIVE	2100
	SELF-TEST	200
	PROGRAM CHANGE	200
	COMMAND HANDLING	4000
	MODE CONTROL	800
	OPS SCHEDULING	1200
	DATA COMPRESSION	400
	HISTORY	1000
	SITUATION ASSESSMENT	300
	COMP DUMP	100
	STABILIZATION	800
	POSITION COMP	1600
	SUB-SYS SERVICE	1800
ADAPTABLE BASIC	DOWNLINK	800
	GUIDANCE	300
	SENSING	800
	PRE-LAUNCH TEST	4000*
	SYST MONITOR	800
	SYST TROUBLESHOOT	1200
MISSION PECULIAR	EXPERIMENT	400
	EXP CONTROL	2600
	EXP MAINT	700
	EXP DATA	600
	TOTAL	23300

*USES COMMAND HANDLING MEMORY AREA
NOTE: MEMORY SIZE INCLUDES 30% SPARE

3-160 **Figure 6.7-3 Spacecraft Computer Functions**

The basis software is that which will form a fixed library of programs applicable for all of the EOS missions. Such changes as are required will be those implemented by the link editor of the support software system, so that no continuing software effort will be required once these programs achieve their final form.

The Mission Peculiar software is that which is required for the support of the sensing experiments, and is thus applicable to only those missions which use the specific experiments. The entries in the budget presented represent the software required to support the MSS and TM.

The intermediate group of functions is made up of basic software which requires modifications to accommodate specific mission configurations. It will be taken from the basic library, however, and adapted to the specific mission by change of tabulated values and constants.

To some extent, the size of the software package will depend upon the configuration of the computer hardware. The estimates of Figure 6.7-3 are based on the use of the AOP (Advanced On-board Processor), which is a follow-on development from the OBP of the OAO program, and has an 18-bit word. Use of a computer with shorter word length will require somewhat more memory because some angle computations will need to use double-precision computation to meet the system requirements. On the other hand, a computer with word size of 22 or more bits will permit some economy by reducing the need for double-precision computations in the position computation routines. Computers with larger word size may also permit smaller software size because the address field of each instruction may have greater range than the 12-bit address of the AOP.

6.8 ELECTRONIC TECHNOLOGY

Purpose:

To evaluate existing versus new technology with trade-offs for selecting an approach for each element of hardware. Trades are to be performed recognizing that obsolete components do not necessarily require new design concepts for the function.

Summary:

This study is deferred, since it is anticipated that discussions for each element of hardware which differ from the current configurations preferences can be incorporated into the design very readily before the design specifications volume is issued. Before the final design specifications are finalized, each hardware element will be reviewed as to the availability and cost impact of new technology equipment. Each equipment possibility will be reviewed as to its possible design and cost impact and the risk of specifying it in the design will be assessed.

6.9 INTERNATIONAL DATA ACQUISITION

This study established the relative merits of several international data acquisition (IDA) alternatives for EOS and rated these alternatives on a cost-effectiveness basis. The following summarizes the options, approaches, and cost parameters involved in the trade study. A complete discussion and analysis is presented in Appendix E.2.11. The primary alternatives under consideration were:

Option 1: Direct transmission (D.T.) to foreign user ground stations.

Option 2: A wideband video tape recorder (WBVTR) system for collection of foreign data and processing and distribution from CONUS.

Option 3: A TDRSS configuration for the relay of foreign data to CONUS for processing and distribution.

The relative performance rating of each IDA configuration is shown in Table 1 in Appendix E, based solely on the percentages of available data each alternative can provide for three data volumes of interest. The TDRSS configuration is clearly superior to the other configurations, followed by the 2-site (Alaska and NTTF) WBVTR configuration, the D.T. system and finally the single (Alaska) site WBVTR system.

The costs of each of the three primary IDA options and a hybrid system configuration are given in Table 2 in Appendix E.

The direct transmission case includes the costs of the six regional stations and the data processing and handling costs (per year) required to produce and deliver high quality TM or HRPI pictures every day to each of 20 users per regional station location.

The WBVTR option costs include the costs of 2 recorders on the spacecraft and an equivalent data processing and handling cost to distribute the finished picture products to foreign users. Two tape recorders have been assumed to accommodate cases in which two orbit periods pass before data can be dumped at either Goddard or Alaska.

The TDRSS costs are a function of the cost allocation algorithm assumed for the TDRSS. If the costs are based on percent bandwidth occupancy for the IDA mission, then one-half ($25M) of the projected yearly TDRSS costs can be assumed. That is, the IDA missions (200 Mbps) use all of the single access capability of a single TDRSS satellite and there are two such satellites in the system. Proportioning costs on the basis of time-bandwidth utilization on the other hand, would reduce this figure by a factor of 1/10th (100 to 150 (worst case) minutes per day projected load), or $2.5M per year.

The hybrid system option includes six low cost ground stations (LCGS) and a WBVTR configuration primarily intended for use with a low data volume, wheat crop only, type IDA mission.

These cost data, together with the performance data of Table 1, in Appendix E, provided the basis for rating the cost-effectiveness of the IDA configurations.

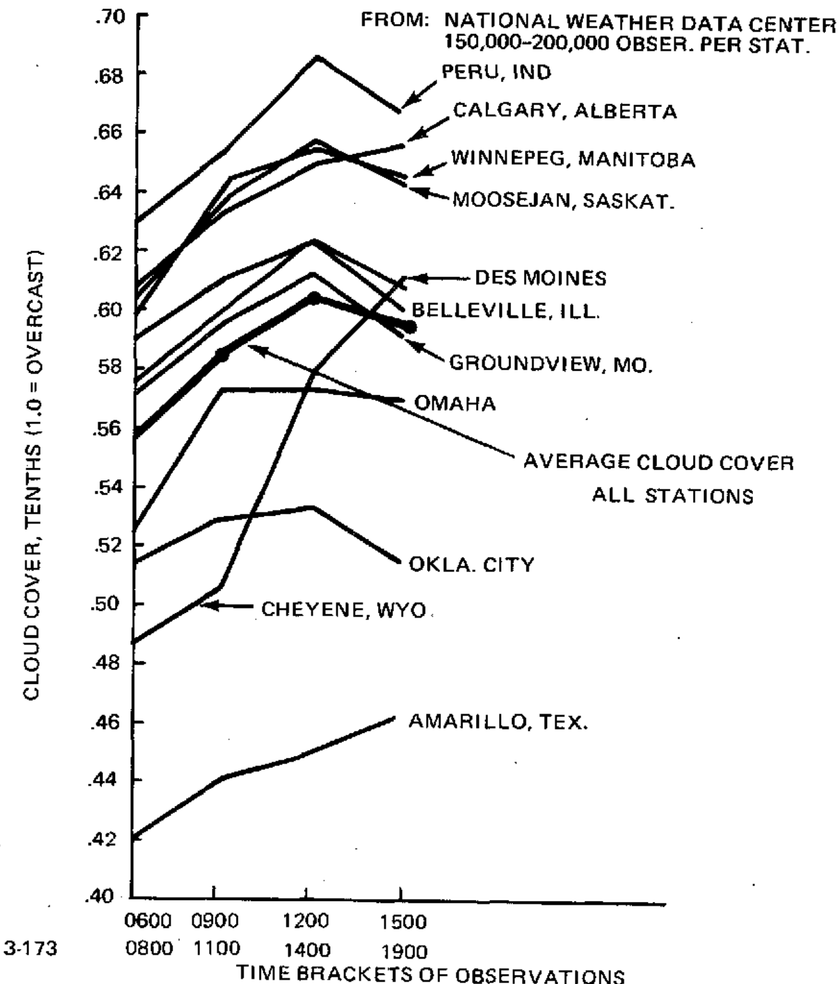
6.10 USER/SCIENCE AND ORBIT TIME OF DAY STUDIES

6.10.1 PURPOSE

To organize the user requirements for the spacecraft and instruments to provide guidelines for design evaluation.

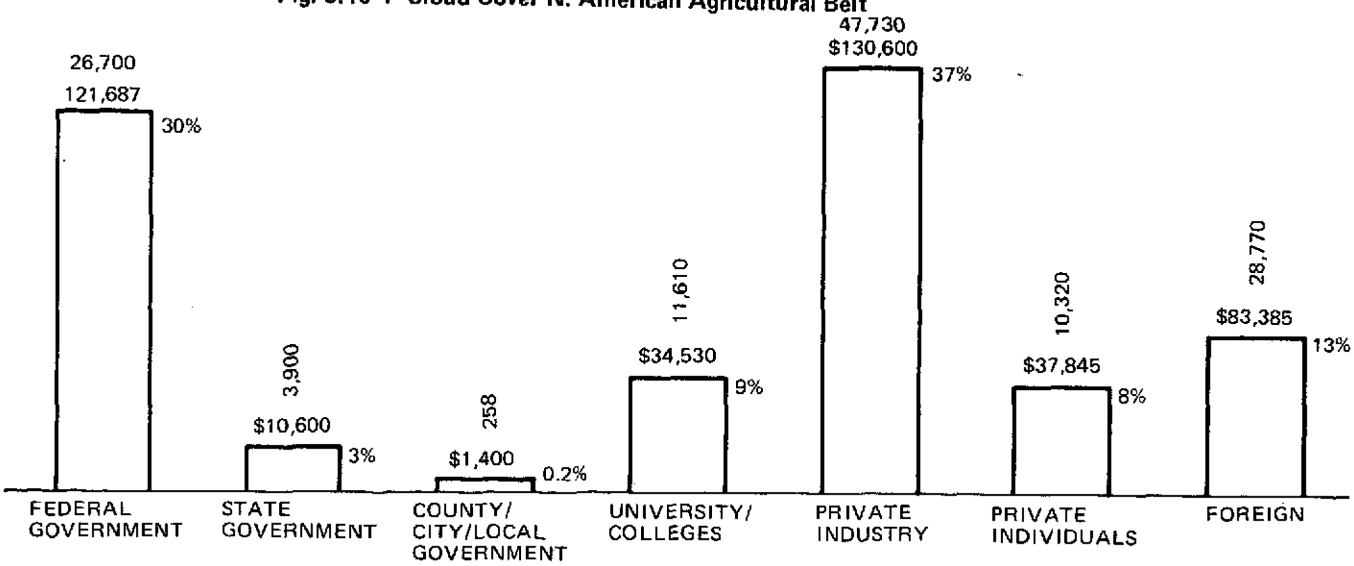
6.10.2 CONCLUSION

- EOS spacecraft design should be flexible with respect to orbit time of day.
  - Cloud cover for the 900-1100 time period averages 5% less than the 1200-1400 period for CONUS midwest agricultural region, Reference Figure 6.10-1.
  - Atmospheric modeling has been used to predict maximum and minimum signal levels in each spectral band. NASA specifications for minimum radiance levels appear to be higher than the calculated values; i.e., for some cases viewing will be instrument limited. See discussion in Appendix E, Section 2.12.
  - Sun angle versus orbit time of day does not change rapidly for low sun angles at high latitudes; however, at lower latitudes nearer noon orbits give significantly higher average sun angles, Reference Fig. 6.10-2.
  - Near noon orbits yield best photometric information (maximum brightness). However, water areas will be affected adversely by sun glint within approximately a 10^{o} cone, while recognition of some types of vegetation is facilitated at or near solar opposition.
  - Maximum daytime temperature difference for soils occurs at about 1330, Reference Fig. 6.10-6.
  - Shadowing at low sun angle is beneficial for such applications as topography and landform.
- EOS system data provided at a frequency of at least 2 weeks will satisfy 72% of the users. It is very desirable to provide data every week or 10 days in which case over 90% of the user applications will be satisfied, Reference Figure 6.10-3a.



3-173

Fig. 6.10-1 Cloud Cover N. American Agricultural Belt



TOTAL FRAMES – 129,288

3-176

TOTAL VALUE $410,117.00

Fig. 6.10-2 Customer Profile (July '72 – Dec '73)

- EOS data of 30 meters resolution will satisfy 77% of the user applications. Capability of providing 10 meter resolution is desirable to meet the requirements of the remaining 23% applications, Reference Fig. 6-10-3b.

- The 4 MSS spectral bands will satisfy 72% of the user applications. The additional 3 bands provided by the TM are desirable in order to satisfy the remaining user applications, Reference Figure 6.10-3c.

- Spectral bands specified for the TM are all useful. Relative priority of the 7 bands are MSS Bands 1, 2, 3 and 4 first priority, and the thermal IR Band 7, (10.4 to 12.6) second priority. Signal to noise problems in band 6 (2.08 to 2.35) may make this band of marginal value.

- Radiometric corrections increase in complexity with wider scan angles. The variations in sun angle, atmospheric profiles, ground reflectivity, etc., over the field of view will be investigated and discussed in the final report.

- All spectral bands of one sensor must be registered within one pixel.

- It is desirable that each quadrant of a scene have a data point specified with its geographic coordinates.

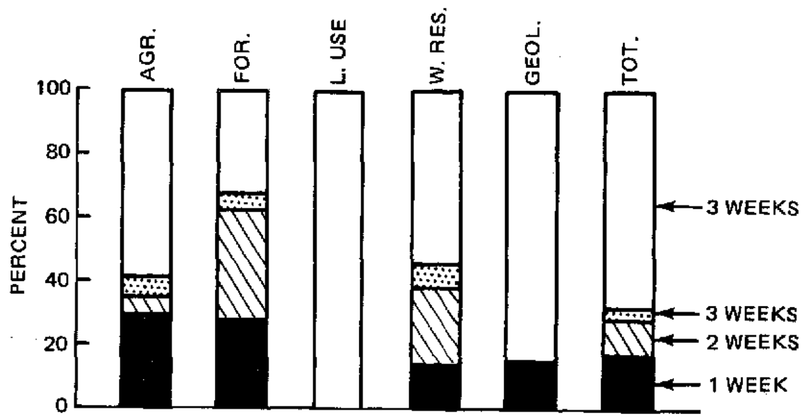
- The major products will probably be 70 mm B&W negatives and CCT's, once technology is disseminated.

- Industrial users now account for 37% of Sioux Falls output. This percentage will probably exceed 60% when EOS is launched, due to an anticipated large increase in technology transfer resulting in exponential increase in demand for data. Reference Figure 6.10-2.

- Monitoring of world food production regions is a very visible application of EOS and warrants emphasis, Reference Figure 6.10-4.

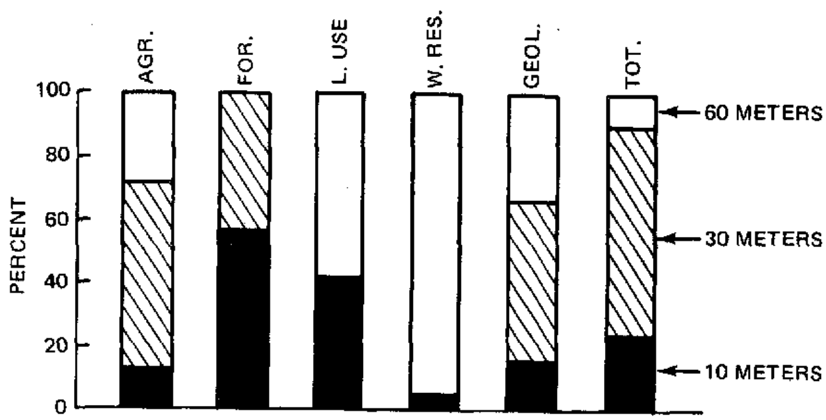
6.10.3 DISCUSSION

Recognized and accepted user applications were employed as the basis for the trade study against which we established system requirements and operational parameters useful in measuring the effectiveness of the EOS system. The frequency, spatial and spectral requirements versus approximately 235 user applications were developed by Dr. M. F. Baumgardner. Cloud statistics were obtained for the CONUS and Canadian major agricultural region and other major agricultural regions of the world. Local orbital time of day was studied within the context of solar illumination, shadowing, target brightness, atmospheric scattering and absorption. Also, the most effective time of day for acquiring thermal IR data was investigated. Atmospheric models now under refinement in Grumman Research are being employed to derive theoretical radiance levels for EOS. Contrast



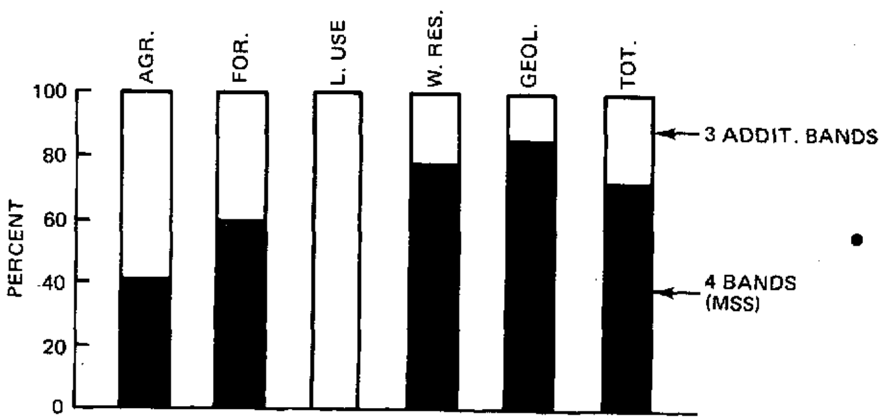
• LARGEST PERCENTAGE GREATER THAN 3 WEEKS

Fig. 6.10-3a User Revisit Time Requirements



• DEVELOP MULTI-RESOLUTION SYSTEMS TO SATISFY USER REQUIREMENTS

Fig. 6.10-3b User Resolution Requirements



• LARGE PERCENTAGE REQUIRES ADDITIONAL BANDS

3-175

Fig. 6.10-3c User Spectral Requirements

degradation for the various spectral bands are being investigated over the sensor scan field of view (ground swath) and HRPI offset angle.

6.10.3.1 DATA FREQUENCY, SPATIAL RESOLUTION AND SPECTRAL BANDS

In appendix E, the matrices for the five disciplines, Agriculture, Land Use, Water Resources, Forestry, and Geology included approximately 235 applications for which data requirements were listed. The data requirements considered to by EOS system drivers are frequency of data, spatial resolution and spectral bands. Figure 6.10-3 a,b, and c are bar charts showing the percent of the applications of each discipline which require data at the specified frequency (3a), spatial resolution (3b) and spectral bands (3c). The last bar in each figure, titled "Total", is an average of the five disciplines. The table below summarizes the results of this investigation for consideration in the EOS system trades.

Data Requirements Summary
(averaged without weighting)

Frequency:		
	Greater than 3 weeks satisfies	69% of applications
	3 weeks satisfies	72% of applications
	2 weeks satisfies	84% of applications
	1 week satisfies	100% of applications
Resolution:		
	60 meters satisfies	11% of applications
	30 meters satisfies	77% of applications
	10 meters satisfies	100% of applications
Spectral Bands:		
	4 MSS Bands satisfy 72% of applications	
	4 MSS Bands pluss 3 additional TM bands satisfy 100% of application.	

6.10.3.2 USER DATA COVERAGE REQUIREMENTS

The 14 + orbital passes of the EOS (680 KM orbit) are shown on Figure 6.10-4. A sample of the procedure for determining the frames of data acquired, load on the tape recorder and time over ground stations for data dump is shown in tabular form, Table 6.10-1. The details are given in Appendix E.

The outer bound of data acquisition may be set at a maximum load of 413 frames per day based upon covering all of the land mass of the world once every 17 day cycle. Due to overlap greater than 100% above 60^{o} latitude and also the present lack of demand for

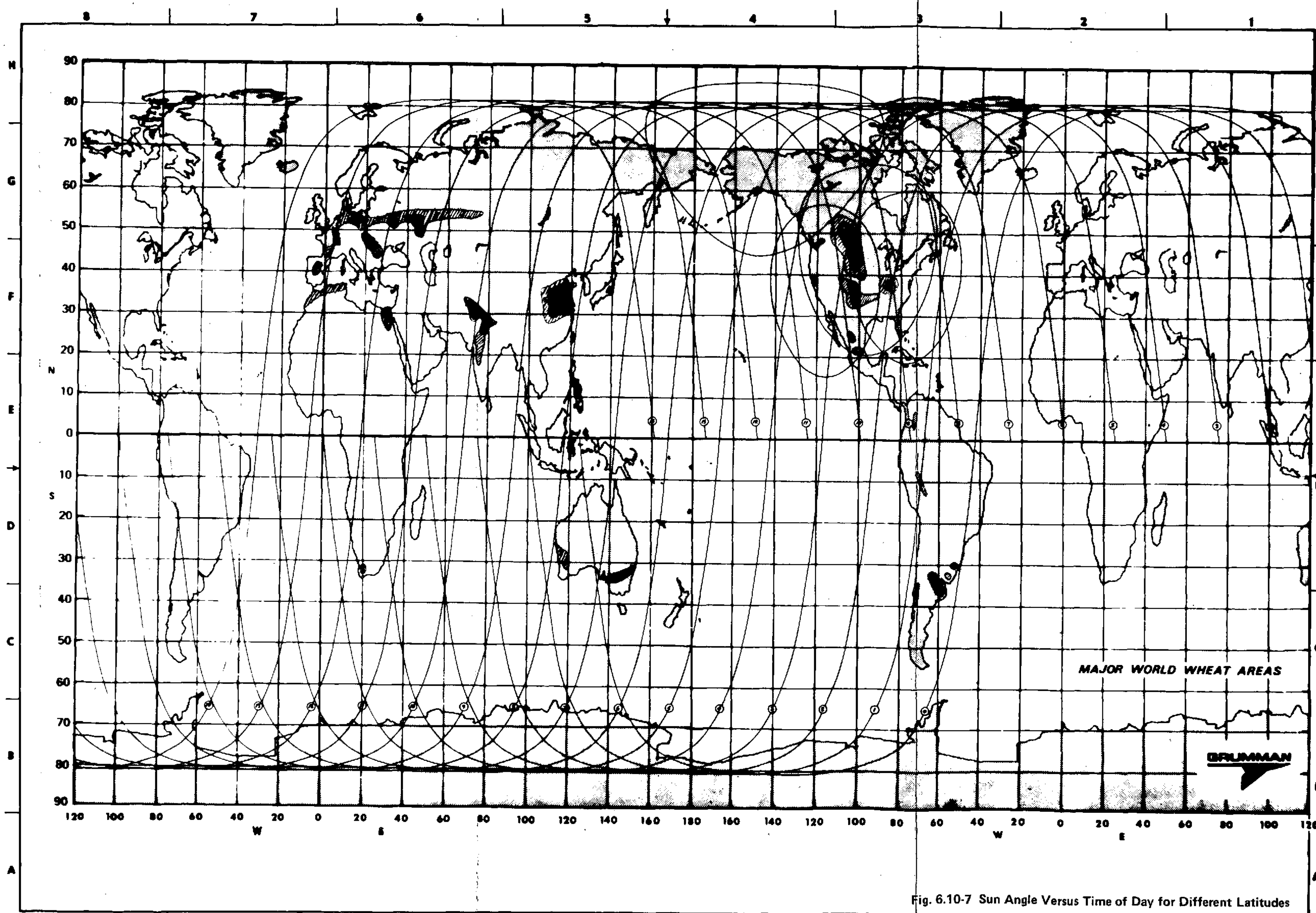


Fig. 6.10-7 Sun Angle Versus Time of Day for Different Latitudes

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 6.10-1 Potential Agricultural Applications of EOS Information Systems

APPLICATIONS	A: HIGH SUN ANGLE	A: LOW SUN ANGLE	B: FREQUENCY WEEKS	C: RESOLUTION METERS	D: RAD. CORRECTIONS	D: GEOM. CORRECTIONS	E: SPECTRAL BANDS, THEMATIC MAPPER 0.5-0.6 u	E: TM 0.6-0.7 u	E: TM 0.7-0.8 u	E: TM 0.8-1.1 u	E: TM 1.55-1.75 u	E: TM 2.08-2.35 u	E: TM 10.04-12.64 u	F: HRPI 0.5-0.6 u	F: HRPI 0.6-0.7 u	F: HRPI 0.7-0.8 u	F: HRPI 0.8-1.1 u	G: SAR "X" BAND	G: SAR "Y" BAND	H: REGISTR. T.M. BANDS	H: SCENE GEOGR. LOCAT.	I: PRODUCTS 70 uu B&W	I: 9x9 B&W	I: 9x9 COLOR	I: CCT	I: MICRO FILM COMP. PRINTOUT	J: DISTRIBUTION & USER AGENCIES INTERIOR	J: AGRICULTURE	J: NOAA	J: USACE	J: ONR	J: EPA	REMARKS
A. AGRICULTURAL LAND RESOURCES																																	
1. AGRICULTURAL LAND USE																																	
a. PROPERTY BOUNDARIES	P	A	52	30	N	Y	X	X	X	X	X	X		X	X	X	X					+			+			X					
b. MAJOR LAND UNITS	P	A	104	60	N	Y	X	1	X	1	X	X		X	X	X	X																
c. USES OF AGRICULTURAL STRUCTURES	P	A	52	10	N	Y	X	X	X	X			X	X	X	X	X					+	–	–	+		X	X	X				RADIOMETRIC CORRECTION IS NEEDED IF COMPARING DIFFERENT DATES.
d. CROPPING SYSTEMS	P	A	13	60	N	Y	X	1	X	1	1	1	1	X	1	X	1					+	–	–	+			X					
2. MAPPING OF GENERAL COVER TYPE & CONDITION																																	
a. VEGETATION, BARE SOIL, WATER	P	A	3-6	60	N	Y	X	1	X	1	1	1	1	X	1	X	1					+	–	–	+		X	X					
b. CULTIVATED VS. NON-CULTIVATED	P	A	26	60	N	N	X	1	X	1	X	X	1	X	1	X	1					+	–	–	+			X					RADIOMETRIC CORRECTION IS NEEDED WHEN DIFFERENT DATES ARE COMPARED
c. IRRIGATED VS. NON-IRRIGATED	P	A	13	60	N	N	X	1	X	1	1	1	1	X	1	X	1					+	–	–	+			X					
d. FALL PLOWED VS SPRING PLOWED	P	A	10-15	60	N	N	X	1	X	1	X	X	1	X	1	X	1					+			+			X					
e. CONVENTIONAL TILLAGE VS MINIMUM TILLAGE	P	A	52	60	N	N	X	X	X	X	X	X	X	X	X	X	X					+			+			X					
f. HARVESTED VS NON-HARVESTED	P	A	1-2	30	N	N	X	X	X	X	X	X	X	X	X	X	X					+			+			X					APPLICABLE ONLY DURING HARVEST SEASON.
g. DOUBLE-CROPPED VS SINGLE CROPPED	P	A	3-6	30	N	N	X	X	X	X	X	X	X	X	X	X	X					+			+			X					
h. DROUGHTY VS NON-DROUGHTY	P	A	8-12	30	N	N	X	X	X	1	1	1	1	X	X	X	X					+			+			X					
i. WET SURFACE VS DRY SURFACE	P	A	2-3	30	N	N	X	X	X	1	1	1	1	X	X	X	X					+			+			X					
3. LAND USE CAPABILITY CLASSIFICATION & MAPPING																																	
a. EROSION ASSESSMENT	P	A	2-3	30	N	Y	X	1	X	X	X	X	X	X	1	X	X					+	–	–	+		X	X		X		X	
b. SURFACE DRAINAGE	P	A	52	60	N	Y	X	X	X	X	X	X	X	X	X	X	X					+	–	–	+		X	X		X			
c. LAND FORMS	A	P	104	30	N	Y	X	1	X	1	X	X	X	X	1	X	1					+	–	–	+		X	X		X			
d. VEGETATIVE COVER	P	A	4-6	60	N	Y	X	1	1	1	1	1	1	X	1	1	1					+	–	–	+		X	X		X			
B. CROP RESOURCES																																	
1. CROP IDENTIFICATION AND MAPPING																																	
a. FIELDS GREATER THAN 4 HECTARES	P	A	12-15	60	N	Y	1	1	1	1	1	1	1	1	1	1	1					+	–	–	+			X					NO DOUBT LARGE DATA USERS WILL USE DIGITAL ANALYSIS. THEIR PRODUCT DEMAND WILL BE FOR CCT AND 70 MM REGS.

COL A: P = PREFERABLE, A = ACCEPTABLE

COL D: N = NO, Y = YES

COL. E, F, G: 1 = DEFINITELY USEFUL

X = MAY OR MAY NOT BE USEFUL

COL I: + = MAJOR DEMAND

– = LESSER DEMAND

monitoring all of the land mass we can conclude that the average daily data load will be less than 413 frames.

The major food producing regions of the world can be monitored by acquiring approximately 300 frames of data per day. The load on the EOS Data Management System is discussed in greater detail in Appendix E, International Data Acquisition and Section 3.3.2.

6.10.3.3 ORBIT TIME OF DAY SELECTION

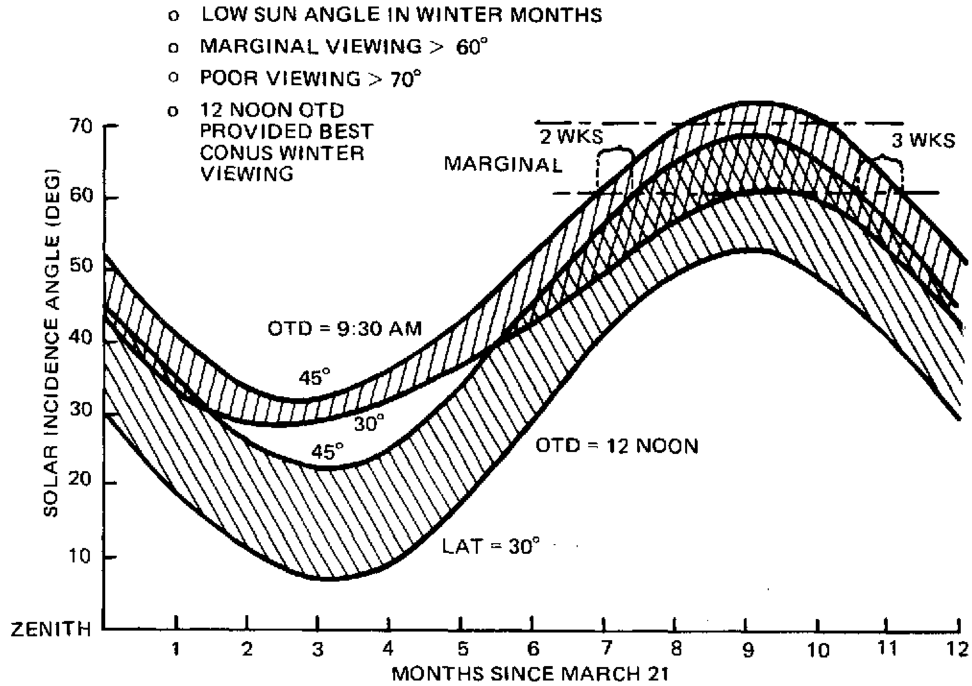
The driving considerations for selection of the best time of day for EOS passage are cloud cover, solar angle above the horizon, and the thermal IR period of maximum temperature difference.

Average cloud cover over the major N. American midwest agricultural area increases from 0.55 during the early morning time period 0600-0800, to a peak of 0.60 at 1200 to 1400 then drops to 0.59 at 1500-1700. See plot of cloud cover statistics Figure 6.10.5. Cloud cover changes less than 5% for the range of times under consideration, and is not an overriding factor.

Solar angle affects target shadowing and target brightness. Assuming a 60^{o} sun angle from zenith (30^{o} above the horizon) as an arbitrary limit, the noon orbit will provide 5 weeks more observation time per year at 45^{o} latitude, Figure 6.10-5. The same basic data was plotted as sun angle versus time of day for different latitudes, Figure 6.10-7, in order to present the impact of time of day on the shape of the sun angle curve at the higher latitudes. Time of day has no overriding impact above 50^{o} latitude.

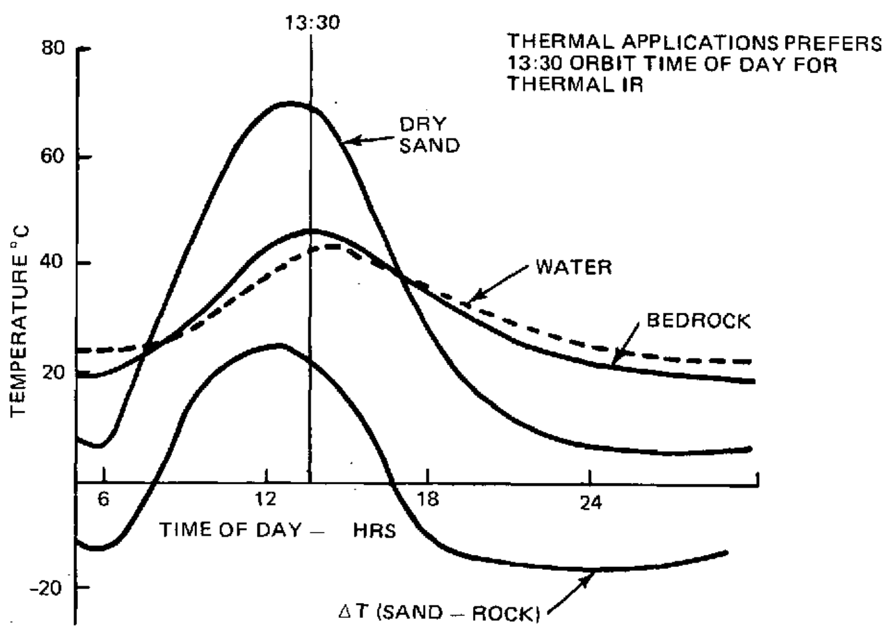
For target recognition, the spectral reflectance characteristics are of primary importance. A secondary factor is the photometric property of the target. Higher sun angles facilitate differentiation between specular and diffuse targets, e.g. soils.

For many applications the thermal IR period of maximum temperature difference occurs about 1330 and after midnight to dawn, see Figure 6.10-6. This is due to changes in thermal emission during a diurnal or seasonal cycle. In many cases the thermal emissions of two or more materials undergo a reversal relative to each other during a heating and cooling cycle. These effects can be correlated with geophysical and geothermal properties of soil, moisture, plant stress, marine processes, etc. For maximum utilization of the thermal IR band the orbit time of day should be about 1330.



3-179

Fig. 6.10-5 Orbit Time of Day Vs Seasons



3-180

Fig. 6.10-6 Orbit Time of Day — Thermal Applications

3

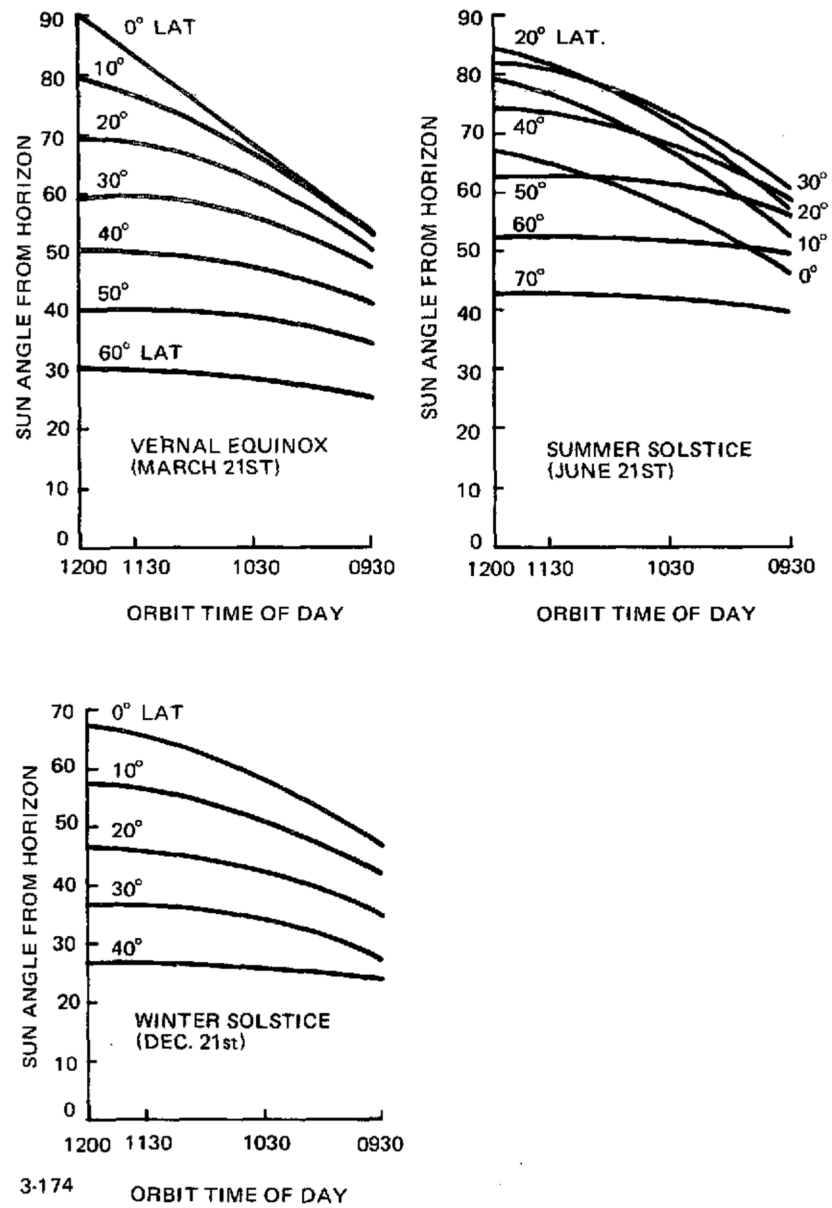


Fig. 6.10-7 Sun Angle Versus Time of Day for Different Latitudes

6.10.3.4 DATA REQUIREMENTS VERSUS USER APPLICATIONS

Data requirements versus user applications were defined by Dr. Marion F. Baumgardner of the Laboratory for Applications of Remote Sensing (LARS), Purdue University in consultation with Grumman. Matrices were developed for five (5) disciplines, Agriculture, Land Use, Water Resources (Fisheries), Forestry and Geology. A total of 235 applications were identified in terms familiar to the users and scientists involved in analyzing and applying spacecraft data. Table 6.10-1 is representative of these matrices. Refer to Appendix E, Section 2.10 for a complete listing of the requirements data. Some of the requirements must be met for specific applications or the data will be useless, while other applications do not have a sharp cut-off point. The following brief comments pertain to the requirements columns of the matrices, Reference Table 6.10-1A.

Col. A-Sun Angle - In general a high sun angle is preferred with the exception of some applications where topographic landform and height information is desired, and shadowing is beneficial.

Col. B-Frequency of Coverage - Where frequency of data is short, less than 10 days, it is assumed that the time from EOS passage to delivery of data to the user is also short, 3 to 5 days.

Col. C-Spatial Resolution - The resolution number entered does not imply that a larger resolution is worthless, but rather that the smaller number 30m or 10m for example will be useful for the particular application.

Col. D-Radiometric and Geometric Corrections - Many applications do not require radiometrically or geometrically corrected data for analysis and interpretation of a single scene for a single date. However, applications utilizing comparisons or overlay of EOS scenes acquired on different passes require radiometric and geometric corrections. Geometric corrections are essential for HRPI Nadir pointing where 10 meter resolution is required.

Col. E,F-Spectral Bands - As yet the research community has not developed an adequate definition of the spectral bands which are most useful for many applications. For soils studies bands 0.6-0.7 and 0.8-1.1 have been found particularly useful. For crop species identification, a thermal band, one or two reflective IR bands, and the upper visible region have been found useful. For vegetation under stress and many geological applications near IR, middle IR and thermal IR have been found to be important.

Experience with aircraft data indicate that thermal scanning may be important for studying internal drainage properties of soils and for studying moisture stress in plants. The middle IR Bands may also be useful in characterizing plant moisture stress. For snow areal extent measurements and determination of moisture equivalent the middle IR bands of the TM are essential.

Col. G-Synthetic Aperture-Radar - No very convincing evidence has yet been presented for the use of SAR other than providing all weather capability. The capability of SAR to identify and characterize earth surface features does not approach the capability of the multi-spectral scanner to perform such tasks.

The SAR does lend itself to the mapping of gross features and geometric patterns such as lakes, rivers, and land forms in regions of perpetual cloud cover.

Col. H-Registration - Since most applications will require the analysis of more than a single band of spectral data, it is essential that all Thematic Mapper and HRPI Bands be registered.

It would be most helpful if each quadrant of a frame of EOS TM data has a data point which is registered precisely with a specific geographical coordinate or address.

6.11 UTILIZATION OF CONTROL CENTER PERSONNEL

Purpose:

To Define Mission Operations and Mission Operations Control Center (MOCC) Concepts and Personnel Utilization for EOS.

Ground Rules:

- MOCC will handle S/C Housekeeping Data Only
- Hardware Interface with CPF-IMS via Common Read-Write Device
- Eliminate Mission Peculiar Ground Equipment to the greatest extent possible
- Minimize Magnetic Tape Requirements and Tape Carry Operations
- MOCC Flexibility is required to support
  - Varied S/C Designs (ERTS & EOS A-F in Particular)
  - Phase over to operational system at DoI
  - Multiple S/C Support
- Contact Message edit capability is required

Conclusions:

- MOCC manpower requirements are 1100 M-M pre-launch, 52 people to support 1 S/C post-launch, and 17 additional people to support a second S/C in the same control center.
- MOCC personnel will work a four shift operation, 24 hours a day. Primary activities will be mission planning, real time operation, and mission analysis. MOCC personnel will train in the T & I area as test conductors, and when necessary, MOCC personnel will be off loaded into the T & I area.
- The baseline MOCC design is structured around a shared memory/grouped-minicomputer configuration. All console designs will be identical, with a minicomputer and interactive CRT in each console.

- There will be an R & D MOCC at NASA/GSFC, and operations will be phased into an operational MOCC at DoI. DoI operation personnel should train in the R&D MOCC at NASA/GSFC.

MOCC Functional Configuration

The functional diagram for the MOCC is presented in Figure 6.11-1. The large rectangular section in the middle areas of the diagram represents MOCC computing capability. This software may be centralized in a MIDI computer or decentralized in a grouped mini configuration (the second choice is our baseline approach).

The functional flow is broken down into two areas-mission planning and real time operations.

Mission Planning:

The focal point in mission planning phase is the MOCC-IMS buffer, which is the storage medium for communication between the MOCC and the CPF. Mission planning is accomplished by coordinating the requirements from the IMS with NASA/GSFC MISCON, SCPS and the orbit determination group. The final result of the planning activity is a contact message residing on the Mission Operations disk where review and edit functions take place.

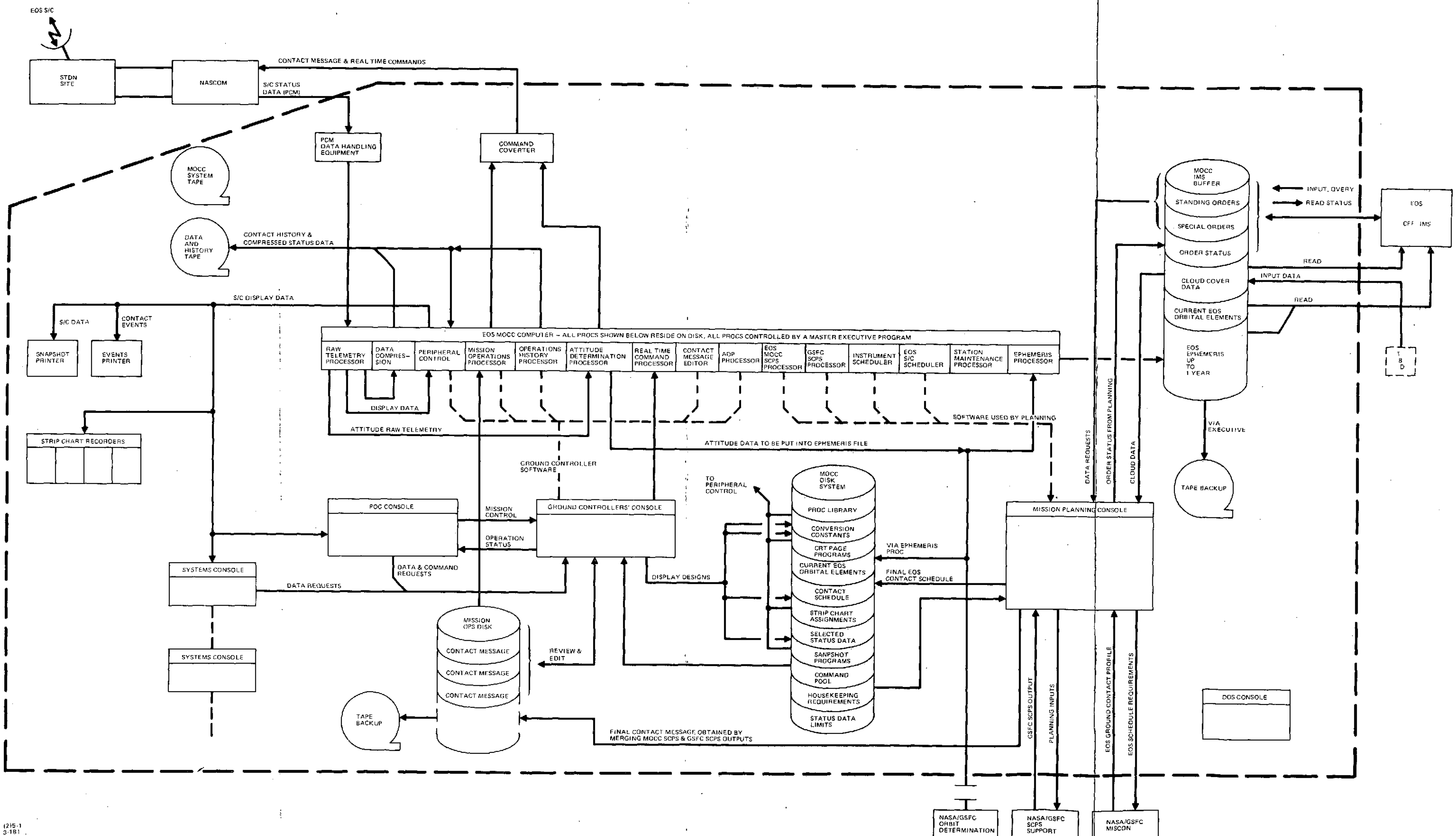
Real-Time Operations:

As real-time operations commence, housekeeping data enters the MOCC via STDN and NASCOM. The data stream is manipulated to drive the various displays and peripherals. During these real-time operations there are two individuals who are the central figures.

- the POC, who is responsible for the health and efficient operation of the spacecraft. The ultimate responsibility for all real-time decisions rests with this individual.
- the ground controller, who is responsible for enacting all pre-planned and real-time operational decisions.

Control Center - Front End Diagram:

The front end portion of the control center (Figure 6.11-2) will be the same for either of the configurations of computing complexes that follow. The only unique part will be the front end interface unit that will interface with the computer performing the front end processing function.



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3-181

Fig. 6.11-1 EOS Mission Control Center Functional Diagram

FOLDOUT FRAME 1

FOLDOUT FRAME 2

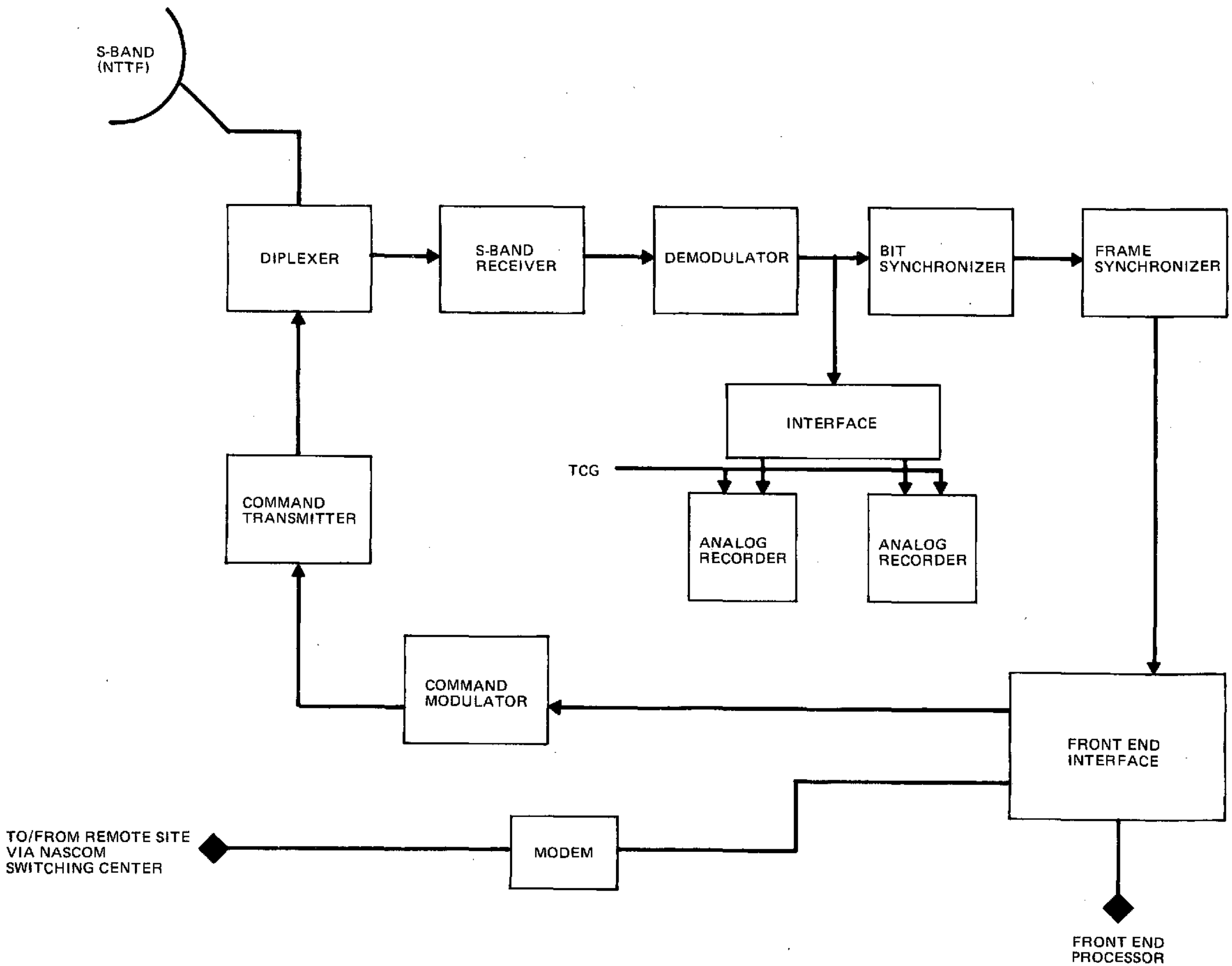


Fig. 6.11-2 EOS – MOCC System Diagram Front End

Data may enter the control center two ways. If the contact is made over Goddard the data will be acquired by NTTF routed to the control center where it is demodulated and synchronized. It then enters the front end interface unit to be further processed by the front end processor. If the contact is made by a remote site, synchronized data will enter the control center via the NASCOM switching center through a modem to the front end interface unit.

Commands to the spacecraft will be handled by the front end processor sent to the front end interface unit and routed either to the NTTF via the command transmitter or the remote site via the NASCOM switching center. Raw PCM data from the NTTF will be recorded on analog tape recorders at the control center.

Control Center - Computing Complex Diagram:

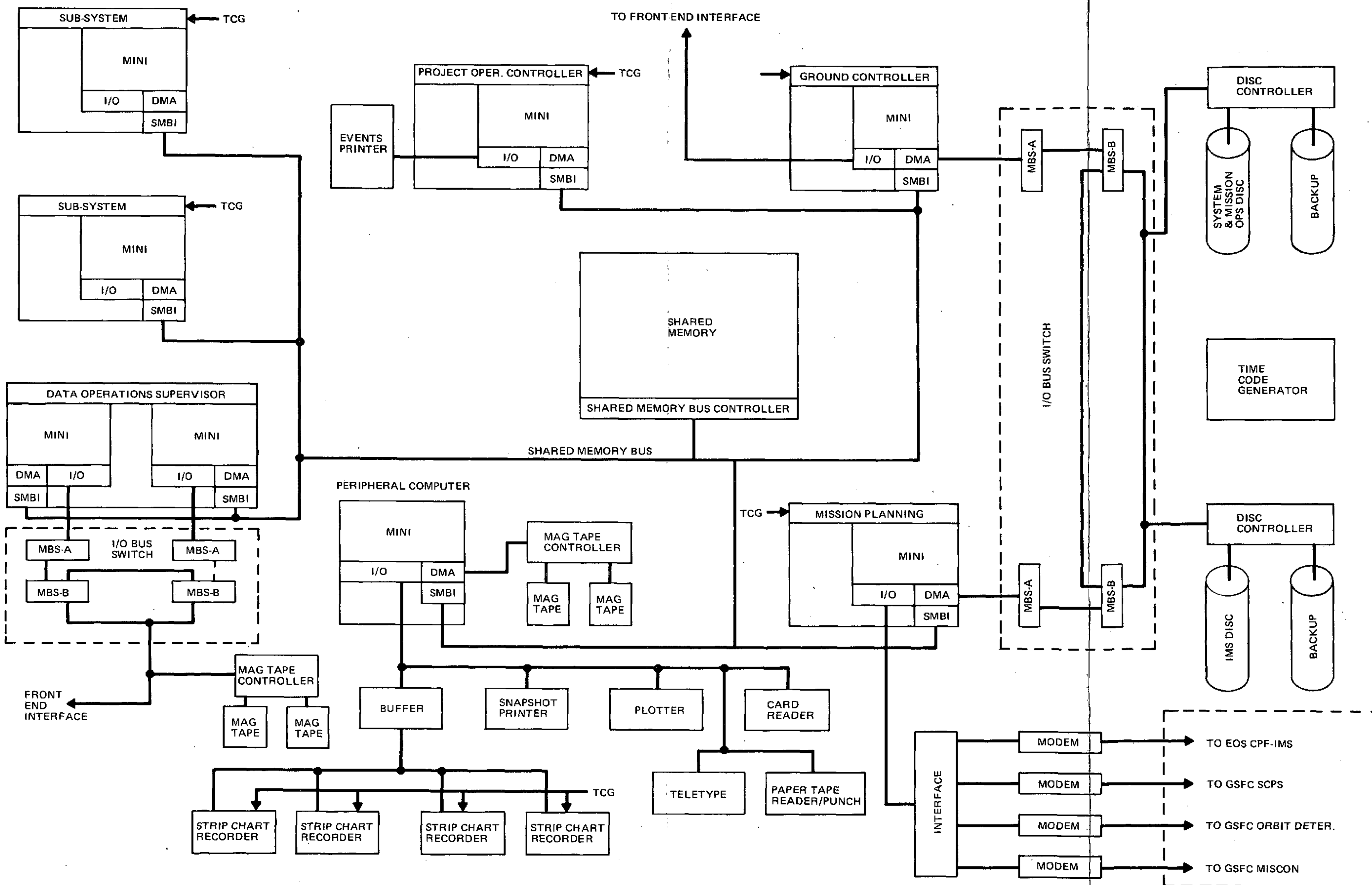
Configuration No. 1 of 2 - Real-time Grouped Mini (Figure 6.11-3)

The grouped mini configuration is an innovative approach to a control center. It consists of multiple mini computers operating in a multi-processing environment. The computers are physically located in each functional console and perform that part of the overall processing requirements for the control center dictated by that consoles function. Each computer (Console) communicates with the other computers via a shared memory. Access to the shared memory is on a priority basis. Protection is afforded in accessing (write mode) the shared memory by allowing each computer to write in only certain blocks of memory dependent upon the function(s) being performed by that computer. This protection is controlled by hardware. Two front end processors (for backup capability) in the DOS console process the incoming PCM data and store it in shared memory for use by the other consoles.

A separate peripheral computer performs all processing required by the peripheral pool units.

Commands are processed by the ground controllers console and sent to the front end interface unit for transmission to the spacecraft. This configuration provides backup capability for any console in event of failure. The consoles are functionally interchangeable via software.

Each console is identical, excepting the DOS console, and its major elements include an interactive CRT and command control and display panel. The panel is computer controlled; therefore, the switches and indicators are not dedicated specific functions, making the console extremely flexible with regard to function.



3-183

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Fig. 6.11-3 EOS-MDCC System Diagram Grouped Mini

Configuration No. 2 of 2 - Real-time Central MIDI (Figure 6.11-4)

The central midi configuration is the conventional approach to a control center. Two midi computers operate in a multi-processing environment. The capability exists for either computer to sustain the activities of the control center, in some satisfactory but reduced mode, in the event of failure of the other computer.

Normally every piece of equipment driven by the computing system will be dedicated to a specific computer. However, the capability exists for any piece of equipment to be switched to the other computer should its primary driver fail.

The consoles in this configuration will be identical, excepting the DOS console; however, they will not contain an internal computer. This will necessitate additional logic to be incorporated in the console in order to interface the console to the computer.

Summary:

The overall concept of a grouped mini configuration provides for an extremely flexible system that is most tolerant to changes and growth. Additionally the grouped mini concept lends itself more easily to the implementation of on-line diagnostics. Since each console contains its own computer, it does not need that interface with a central computer in order to perform diagnostic routines.

MOCC Manpower Requirements

MOCC manpower has been estimated for two project phases - pre-launch and post-launch. The pre-launch effort is estimated to require 1150 man-months, broken down as follows:

MOCC Design and Development	325 Man-Months
Software Design and Development	350 Man-Months
Mission Planning	175 Man-Months
Mission Preparation	300 Man-Months

The software estimate is for the grouped mini approach, and the software effort carries a learning curve because it is an innovative approach. If a dual midi computer configuration is used the software design and development can be reduced to 300 man-months.

The post launch manpower requirement is nominally 52 people to support a single spacecraft on a four shift, 24 hours per day basis. A second spacecraft can be supported in the same MOCC via the addition of 17 people.

6.12 COUPLED VS. UNCOUPLED PNEUMATICS

Purpose:

To evaluate the impact of incorporating coupled or uncoupled pneumatics on: the propulsion system cost, weight, and reliability; the orbit effects when jets are fired for rotation; the computer processing on the ground to compensate for orbit determination degradation; and the program cost.

Conclusions:

Since the impact on program cost is expected to be small, the study and conclusions are deferred.

Discussion:

The near-polar orbits for the low-altitude sun-synchronous missions are ideal for magnetic unloading of reaction wheels. By proper sizing of the magnetic torquer bars, it should be possible to avoid completely the necessity for jet unloading of the reaction wheels. Using reaction wheels unloaded by magnetic torquer bars at all times, the use of jets for rotation would not be required during the 2 years of operations. In this case, the orbit is not disturbed, even if uncoupled jets are used for rotation, since these jets would not be used during the operations period.

6.13 WIDE BAND DATA FORMAT

6.13.1 INTRODUCTION

The purpose of this trade is to outline the various factors that enter into the choice of formats for the wideband data. By considering all of these factors together, it may be possible to find optimum format(s) in the sense that data acquisition, processing, and user product generation are accomplished as efficiently as possible with a minimum amount of time spent in reformatting and handling the wideband data.

The various parts of the data formatting problem can be identified as shown in Figure 6.13-1.

When the processing flow is viewed from an overall standpoint, it is clear that the format of the data at the various stages can have an impact on the efficiency with which data is processed and products are generated. The overall goal of the wideband data format study is to identify the constraints involved and to select the best possible format for the data. These constraints include:

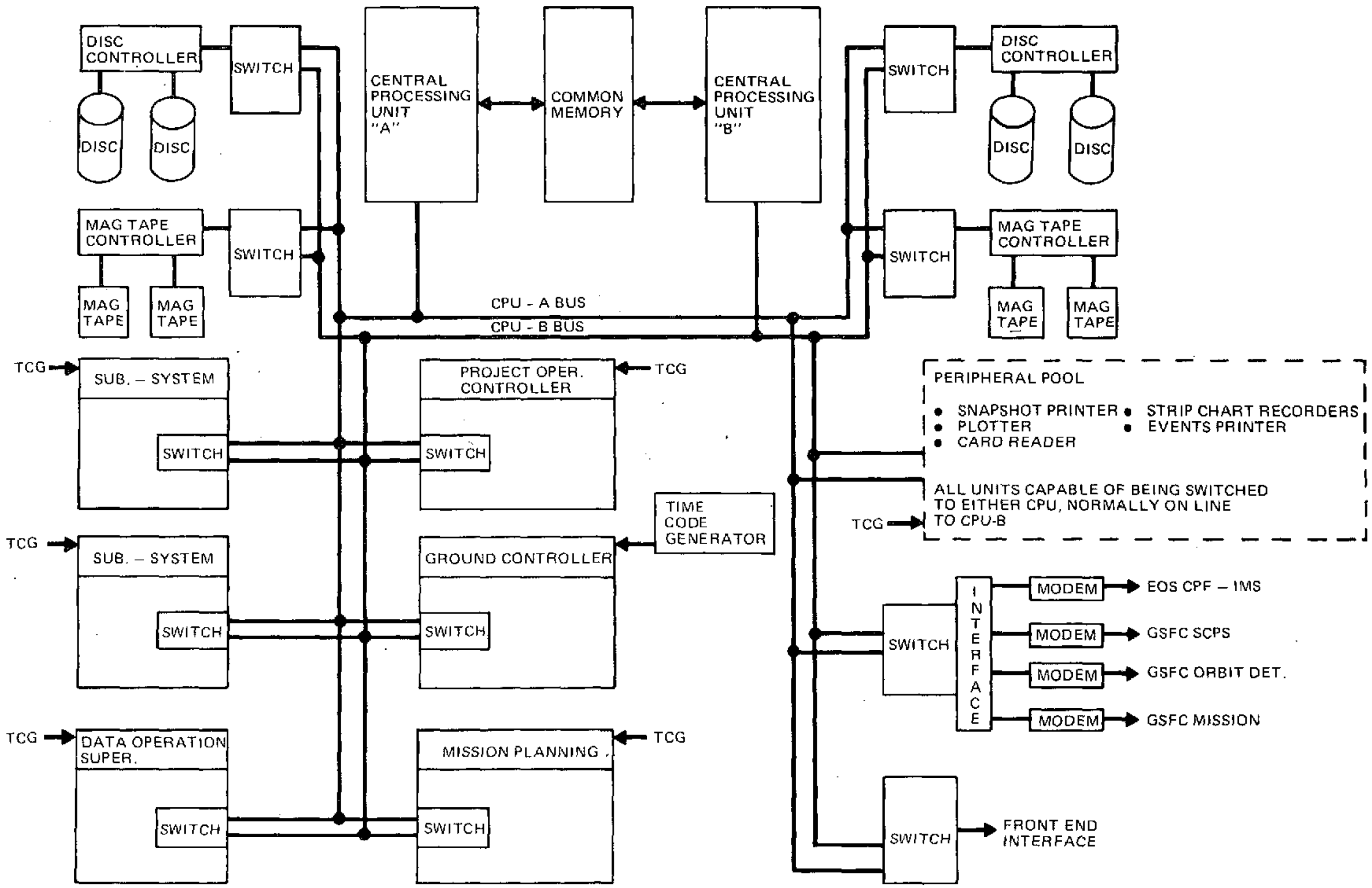


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Fig. 6.11-4 EOS – MCC System Diagram Central Midi

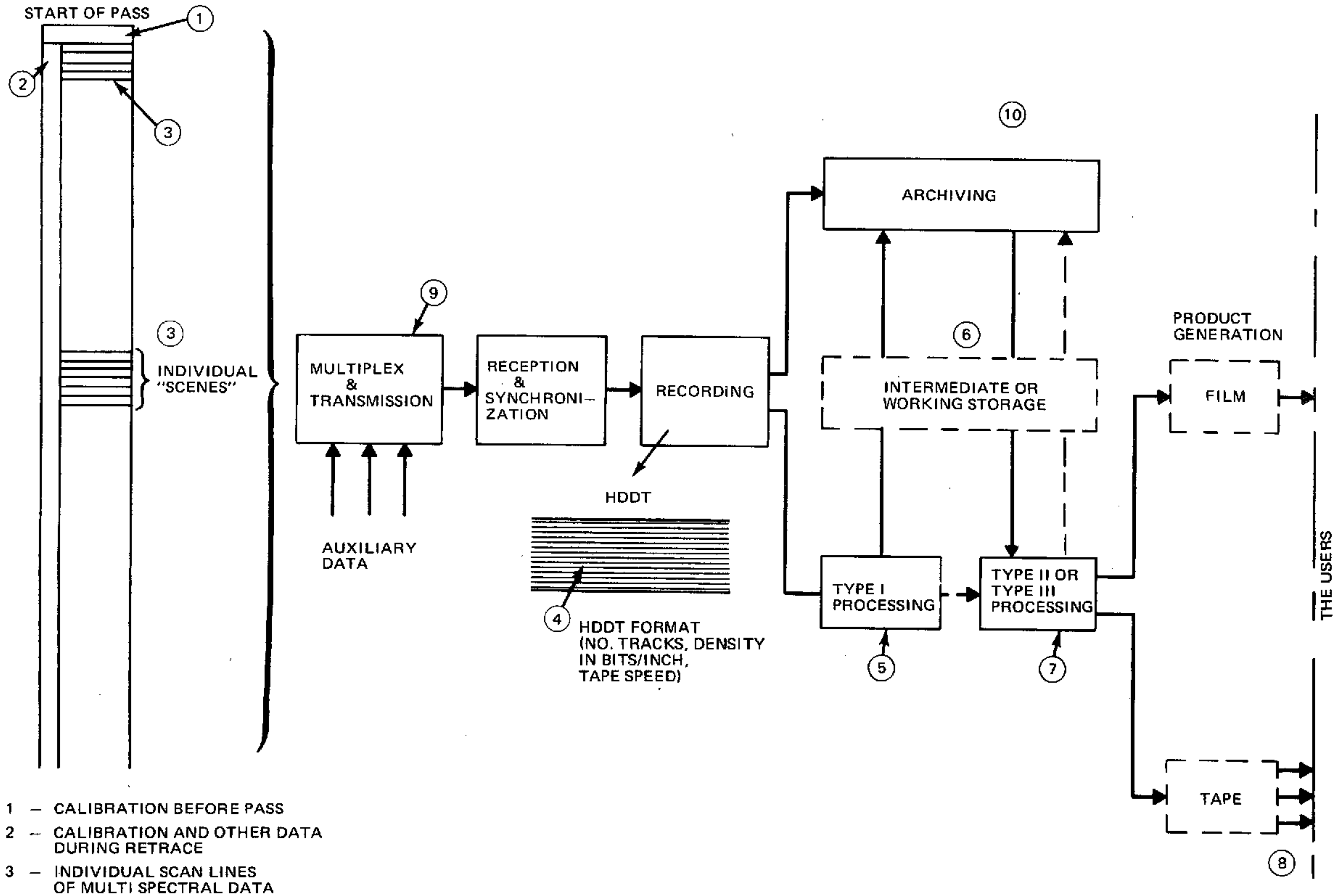

Fig. 6.13-1 The Various Points in the Processing Sequence Where Data Format Can Have an Impact

1. Constraints in the Instruments

   a. Instrument configurations, sampling rates, the sequence for sampling the detectors.

   b. The availability and form of the calibration data.

   c. The requirement that a reduced data set be made available for the low-cost ground station (LCGS) including sufficient calibration data.

2. Constraints at Ground Station

   a. Requirements for acquiring, synchronizing to, and recording the data including any standards that have been imposed on high-density digital tape (HDDT) recording.

   b. The format(s) that are best suited for processing through Stages I, II and III.

   c. Format(s) best suited for archiving the data.

   d. Formats required for product generation.

3. Constraints Imposed by the Users of the Data

   a. Multiple formats required by the processing that the user will perform on the data.

   b. Constraints imposed by the size of the user's processing facility.

6.13.2 CONSTRAINTS IMPOSED BY THE INSTRUMENTS

If we take the Thematic Mapper (TM) as an example, the instrument itself imposes a "natural" format on the wideband data. A typical layout of the detectors is shown in Figure 6.13-2a. From this physical arrangement, several possible formats follow which are shown in Figures 6.13-2b and 6.13-2c.

We begin by assuming a basic sampling rate of each detector within each channel of 150,000 samples/second. We will assume a sampling rate of 1.0 times the IGFOV and that samples are quantized to 6 bits. Within 6.666... μ seconds, therefore, each of the visible detectors must be sampled once; this corresponds to the time required for the scanner to advance 30 meters over the ground. With this arrangement, the thermal channels must be sampled every 26.666 ... μ seconds.

Two possible data formats are shown in Figure 6.13-2; in (a) we assume that space must be left within the data for scan-calibration data, in (b) no such data is included. With each format, a minor frame contains 4 pixels from each visible channel. A line contains

1542 of these minor frames. These lines can be further subdivided into six major frames each of which contains 257 minor frames. Following each line is the equivalent of 272 minor frames during the retrace interval (84.99% efficiency).

We will refer to the formats in Figure 6.13-2 as the pixel/detector interleaved format or, for short, as the "natural" format implying the sampling of the various detectors in each band in a natural order.

6.13.3 CONSTRAINTS IMPOSED BY THE PROCESSING

An assumed format for the pixel/detector interleaved data on 24-track HDDT is shown in Figure 6.13-3. This format requires a minimum amount of buffering (storage) to convert from the serial data stream from the demodulator to the tape format which stores complete minor frames down the length of the tape. At a recording density of 20,000 bpi, a minor frame occupies 20.6/20,000 or 6/1000 inches and an entire line occupies 9-1/4 inches of tape. tape. An entire scene (185 KM TM swath) can be stored on 315 feet of tape.

To perform Type I processing, it is best to retain either the "natural" format, which has all pixels from one sweep of the scanner together in an interleaved format, or possibly the pixel interleaved format. One sweep constitutes a complete scan (west to east) of the detectors and the one-dimensional line scan corrections would be performed in an identical manner on all lines at once. If line scan corrections are unnecessary, then data format has little effect on the Type I processing.

For Type II geometric corrections (two-dimensional) certain processing steps are common to all spectral bands. Therefore, it is best to have the multispectral pixels in close proximity. Either the natural or pixel interleaved formats should be suitable with only a slight penalty incurred by the line sequential format.

To locate ground control points (GCP's) (Type III processing), the band sequential format is best. However, only a slight penalty is paid in accessing the data for one band if the other formats are used.

Alternative formats for HDDT recording are shown in Figures 6.13-4 and 6.13-5. In Figure 6.13-4, the pixel interleaved format is shown where the detector interleaved feature in Figure 6.13-3 has been removed. Note that an entire line (all bands) appears on tape before the second line (detector #2 in all bands) appears. An entire scan line must be stored to perform this reformatting. In Figure 6.13-5, we show the line sequential format where an entire line of one band appears on tape before the same line of the next band ap-

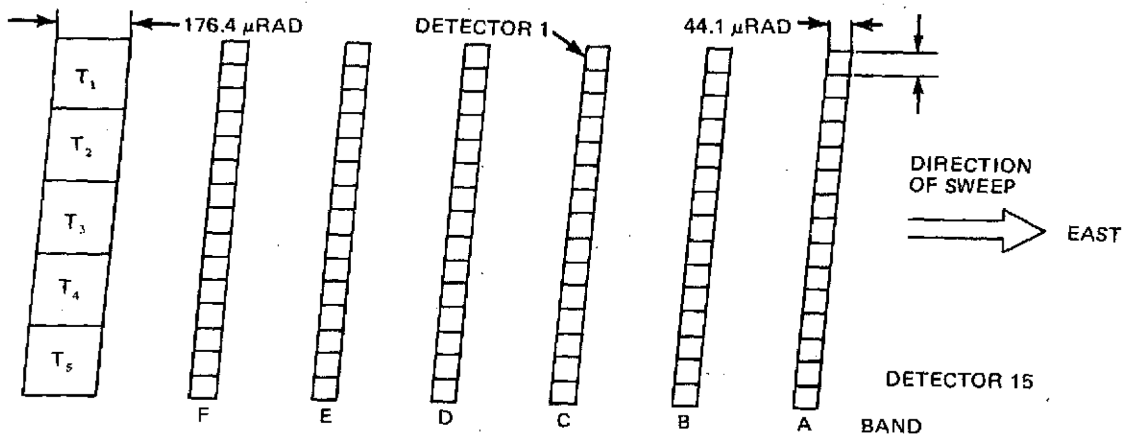

Fig. 6.13-2a Typical Layout of Detectors

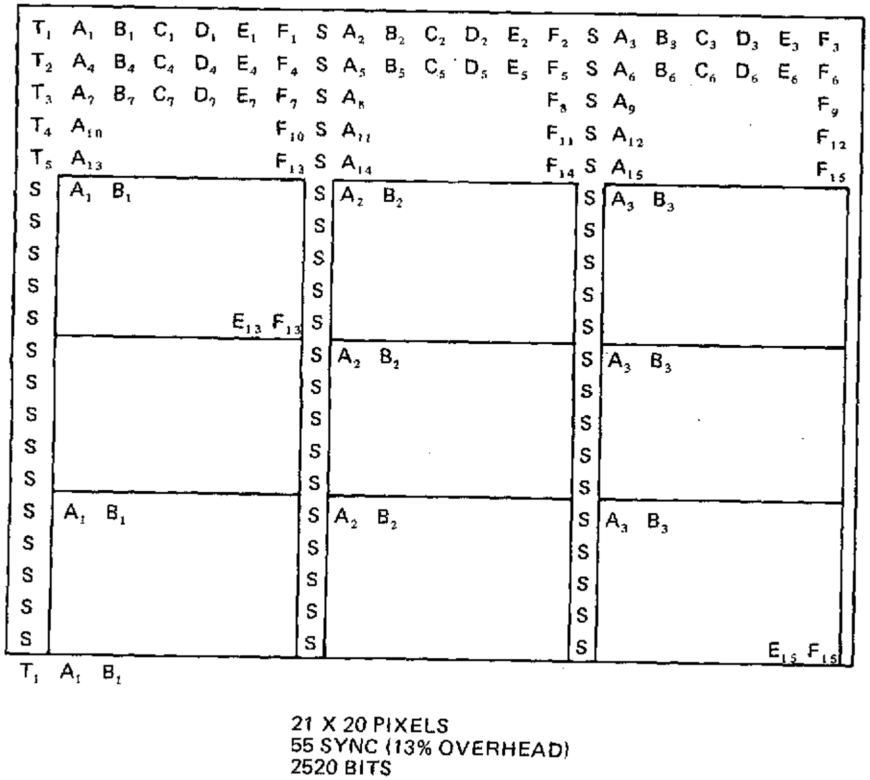

Fig. 6.13-2b Possible Format of Minor Frame

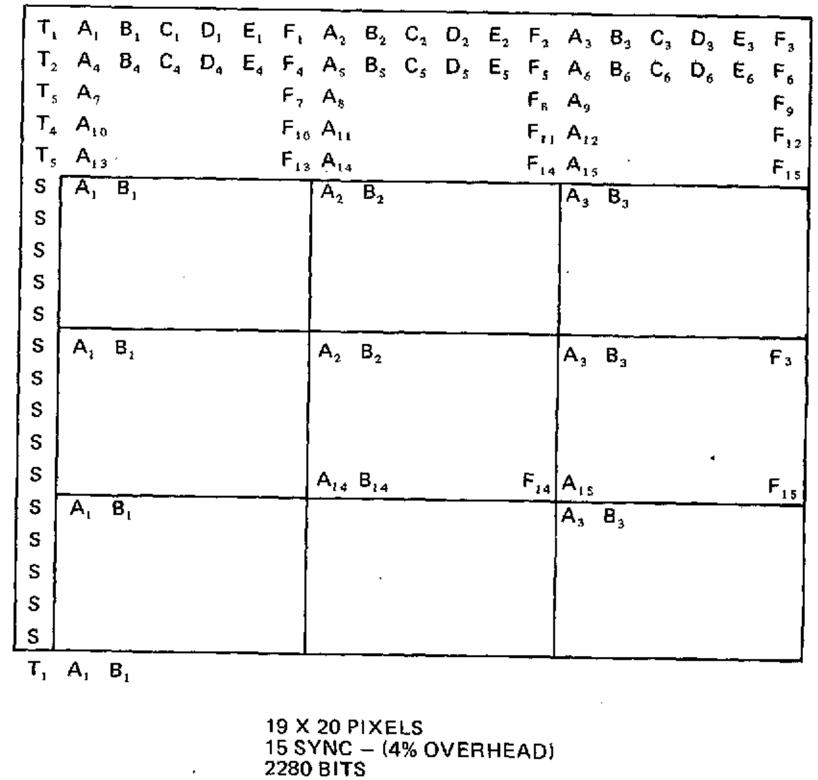

Fig. 6.13-2c Alternative Format for Minor Frame

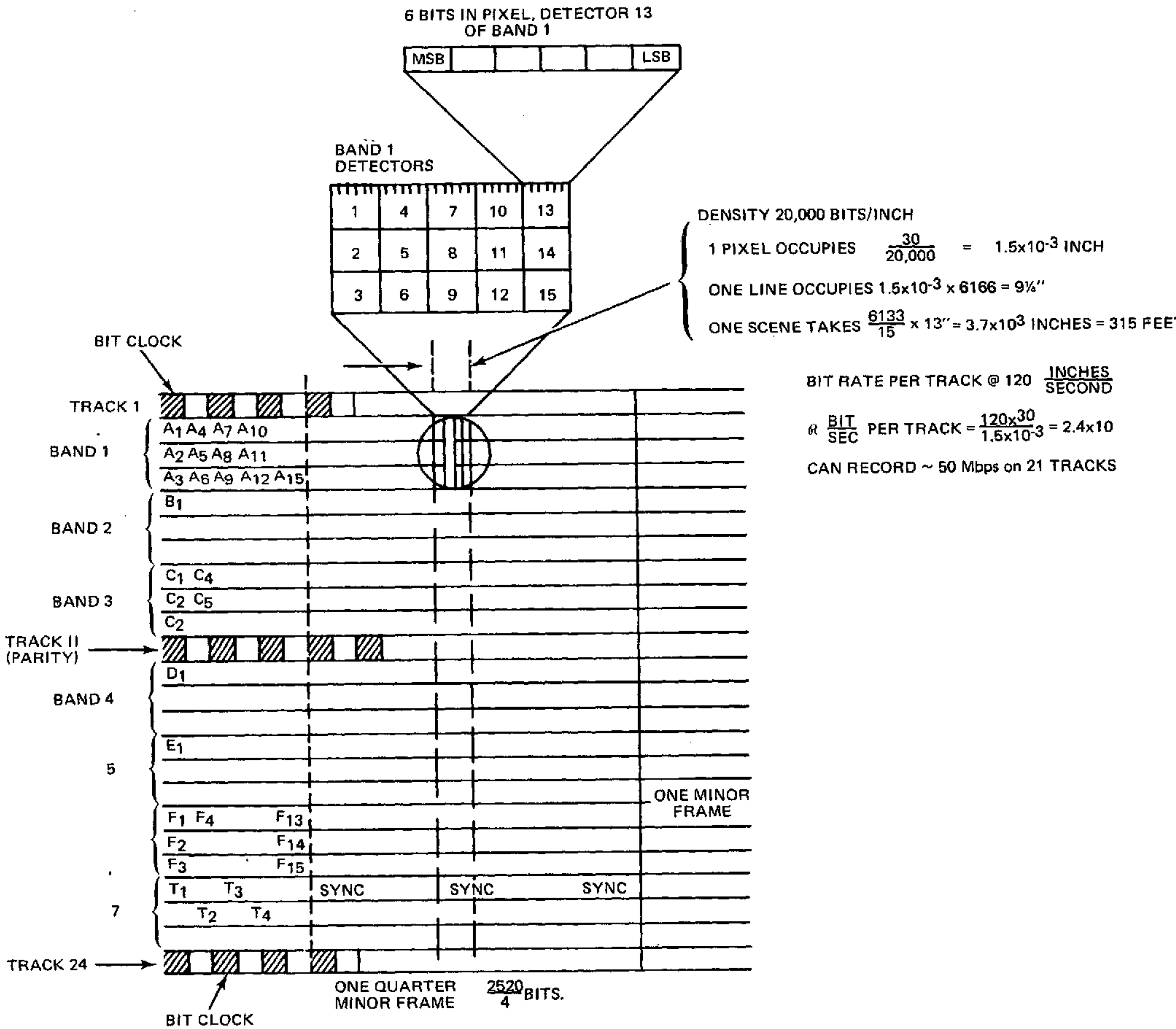

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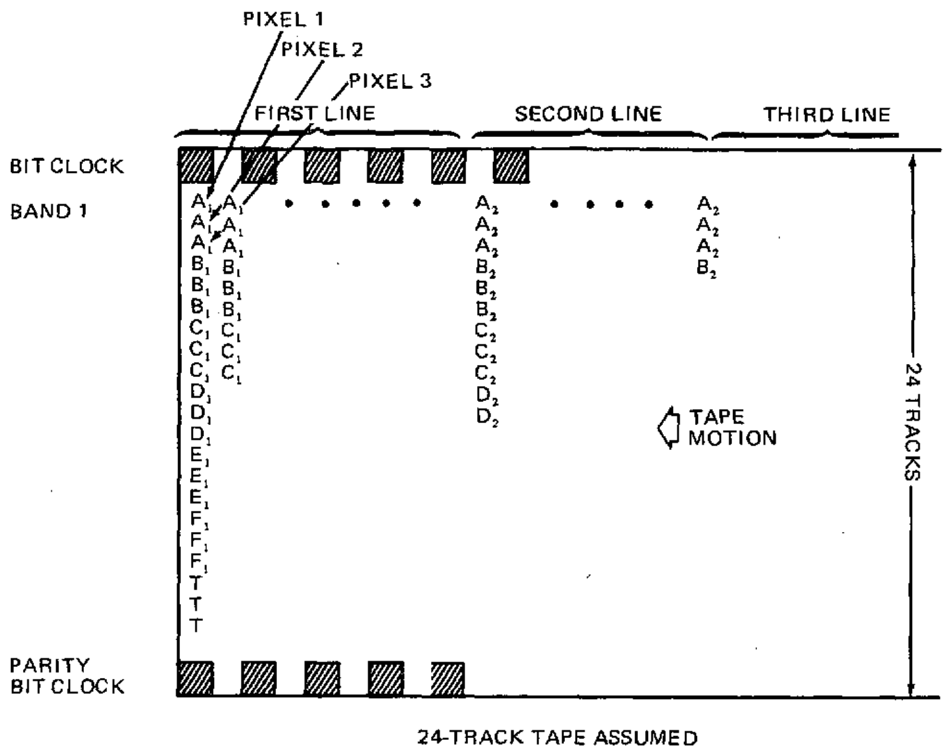
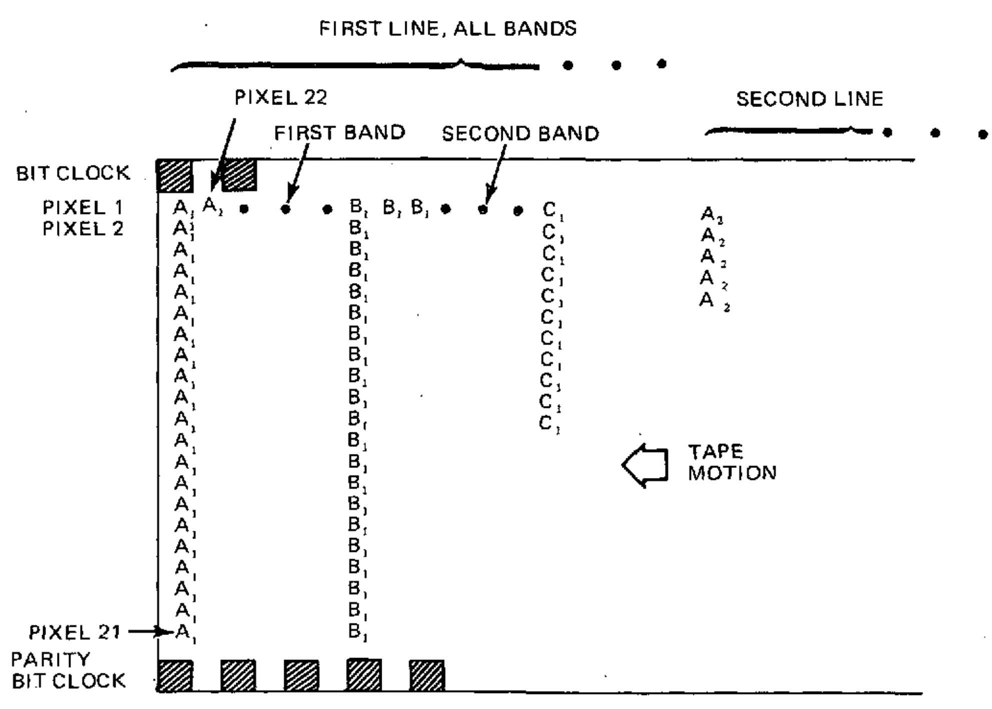
Fig. 6.13-3 Assumed Format of HDDT — Natural "Format" (Detector and Pixel Interlaced)



3-189 **Fig. 6.13-4 Assumed Format of HDDT Pixel Interleaved Format**



3-190 **Fig. 6.13-5 Assumed Format of HDDT Line Sequential Format** pears. Note that an entire scan line must also be stored in going from the natural to line sequential format but that much less storage is required in going from pixel interleaved to line sequential format but that much less storage is required in going from pixel interleaved to line sequential form.

We can summarize the suitability of the various formats for the levels of processing as shown in Table 6.13-1. Also we can summarize the ease (storage required) in going from one format to another, assuming tape $\Rightarrow$ processor $\Rightarrow$ tape transfer, as shown in Table 6.13-2.

6.13.4 CONSTRAINTS IMPOSED BY OUTPUT PRODUCTS

DMS formats are restricted to the tape products used to transfer the processed (corrected) imagery data from the output of the DMS to the user for further analysis and do not necessarily apply to the format to be used within the DMS for digital correction or archival storage.

Three formats identified are:

Pixel-Interleaved: Pixel 1 of band 1, Pixel 1 of band 2, ... Pixel 1 of band N, Pixel 2 of band 1, 111, repeat for each line. (Probably best for analysis where all spectral bands are required.)

Line-Sequential: Pixel 1 of band 1, Pixel 2 of band 1 . . . Pixel M of band 1, Pixel 1 of band 2, 111, Pixel M of band 2, ..., Pixel M of band N, repeat for each line.

Band-Sequential: Pixel 1 of line 1, Pixel 2 of line 1, 111, Pixel M of line 1, Pixel 1 of line 2, ..., Pixel M of line 2, 111, Pixel M of line L, repeat for each band. (Probably best when only one spectral band is needed.)

The application of these formats is:

Format	HDDT	CCT
Pixel-Interleaved	X	X
Line-Sequential	X	X
Band-Sequential		X

Further detail of these formats is shown in Figure 6.13-6a, b, c, where the following nomenclature is used:

B_N - band number: except for the IR thermal band these are similar and may be arbitrarily assigned - B_1 will be associated with IR band. N = 7 for TM and 4 for HRPI.

Table 6.13-1 Summary of Format/Processing Options

PROCESSING \ FORMAT	PIXEL/DETECTOR INTERLEAVED P/D I	PIXEL INTERLEAVED PI	LINE SEQUENTIAL LS	BAND SEQUENTIAL BS
TYPE I	BEST	← SECOND BEST →		VERY INEFFICIENT
TYPE II	← PROBABLY EQUALLY GOOD →		SLIGHTLY LESS EFFICIENT	RELATIVELY INEFFICIENT
TYPE III	← THIRD BEST →		SECOND BEST	BEST

Table 6.13-2 Storage Required to Format

FROM \ TO	P/D 1	PI	LS	BS
P/D I	–	ONE SWEEP 3.8×10^6 BITS	ONE SWEEP 3.8×10^6 BITS	ONE SCENE 2×10^9 BITS
PI	ONE SWEEP	–	ONE LINE ALL BANDS 2.5×10^5 BITS	ONE SCENE 2×10^9 BITS
LS	ONE SWEEP	ONE LINE ALL BANDS		ONE SCENE 2×10^9 BITS
BS	← ONE SCENE →			–

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L_L - scan line number: for the TM, L will range from 1 through approximately 6167 and for the HRPI from 1 through approximately 12,333.

P_M - pixel number for each scan line: for the TM, M will range from 1 through 6167 and for the HRPI from 1 through up to 3200 from left to right.

6.13.5 DATA FORMAT SUMMARY

We can now postulate several alternatives for reformatting the data. The notation introduced earlier

P/D I = Pixel/detector interleaved or natural format

PI = Pixel interleaved

LS = Line sequential

BS = Band sequential

are retained. The options are distinguished by the format that exists at each stage of processing and the point at which conversion is made. These options are shown in Table 6.13-3a.

Table 6.13-3a Candidate Data Reformats

OPTION	ACQUISITION INITIAL RECORDING	STAGE I	ARCHIVE	STAGE II	STAGE III	USER PRODUCT (TAPE) GENERATION
A	P/D I	P/D I	P/D I	P/D I	P/D I	PI, LS, BS
B	P/D I	PI	PI	PI	PI	PI, LS, BS
C	P/D I	LS	LS	LS	LS	PI, LS, BS
D	PI	PI	PI	PI	PI	PI, LS, BS

Option A retains the data in natural format through all processing with reformatting performed only when tape products are made. With Option C, the data is converted to line sequential (LS) format after acquisition but before Stage I processing is performed, and remains in this format until product generation.

We can now evaluate these alternatives by assigning a score to each option in the following categories:

- Intermediate reformatting effort (storage required and reduction in throughput)
- Type I processing efficiency
- Type II processing efficiency
- Type III processing efficiency
- Final reformatting efficiency (assumes 50 percent products are LS, 30 percent PI, 20 percent BS).

Note that Option C, which reformats the data to line-sequential format early in the processing, appears best primarily because most of the output tape products (50 percent) are assumed to be required in this format. The second-best option is to leave the data in the "natural" format (P/D I) throughout the processing and reformat only at the completion of all processing.

It is clear that the evaluation in Table 6.13-3b is rather arbitrary and has resulted from only a preliminary treatment of the overall problem. The evaluation criteria may not carry equal weights in terms of overall cost/throughput impact upon the overall system. Also, the choice between options C and A, (possibly B should also be retained) is dependent upon the assumption that the LS format is preferred by most users.

Table 6.13-3b Evaluation of Format Options

OPTION \ EVALUATION CRIT.	INTERMEDIATE REFORMAT EFFORT	TYPE I	TYPE II	TYPE III	FINAL REFORMAT EFFICIENCY	TOTAL
A	10	10	10	8	6	44
B	8	9	10	8	8	43
C	8	9	9	9	10	45
D	5	9	10	8	8	40

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6.14 MODULARITY LEVEL

Purpose:

To assess the baseline and alternative modularity levels and determine the most economic for EOS-A.

Conclusions:

- Integrated subsystems have potential weight savings but precludes on-orbit servicing.
- Subsystem modules have potential program cost savings (Spares, refurbishment), but the weight penalty precludes launch on the Delta 2910.
- Baseline modularity level (subsystem modules) provides for both on-orbit service and Delta 2910 launch.

Discussion:

This study was based on the NASA/TITAN EOS Configuration. In order not to perturb the basic spacecraft design, the subsystem module configuration considered smaller modules that would fit within the 48" x 48" x 18" envelope of the baseline subsystem modules. Each subsystem was partitioned in several submodules on the basis of equipment size, functional relationship, thermal load and redundancy. In almost all cases redundancy was placed in a separate, but identical module to the prime equipment, resulting in multi-application of modules. Figure 6.14-1 shows the preliminary distribution of equipment within the sub-modules. Of the 21 modules, there are only 11 different types, indicating a high degree of multi-application. The figure also shows how the 21 modules might be designed to fit within the baseline subsystem module envelopes. The weight penalty for the subsystem module precluded the launch on the Delta 2910, therefore, further design and system studies were terminated.

6.15 FOLLOW-ON MISSION ECONOMIC STUDY

Purpose:

The purpose of this study is to determine the economic benefits in utilizing multi-mission spacecraft to capture varying numbers of earth observation missions, and to evaluate the cost impact of extending the GAC baseline design to capture the EOS missions B through E plus SEASAT A, SEOS, and SMM.

Discussion:

The study is basically a cost comparison of multiple-mission spacecraft (subsystem-modular designs) against the corresponding single-mission spacecraft (subsystem modular design) for the same mission set. In the multiple-mission spacecraft case the subsystem modules are designed to meet the most stringent performance requirements in the mission set. Thus there are instances when the subsystems will operate below their design performance level. In the single-mission spacecraft case no such instances occur because the subsystem modules are matched to the particular mission requirements.

Extensions of the GAC baseline design were also evaluated against the corresponding single-mission spacecraft. The GAC baseline extension approach was not to build in subsystem performance to meet the most stringent mission in a set, but to capture additional missions by adding mission-peculiar subsystem performance capability as required.

The analysis was based upon the following groundrules:

- Design life of 2 years for all spacecraft except D (5 years).
- Shuttle on-orbit service available WTR in 1983

The cost model includes spacecraft design/development/test/engineering, other non-recurring costs, launch vehicle and support, spacecraft production and annual operation. DMS costs were assumed to be insensitive to the modular subsystem design of the spacecraft. Operational programs (EOS-B, D, and E) were costed to their stipulated runout years (1992, 1986 and 1992 respectively) and the operations costs included the Shuttle resupply costs.

Preliminary results are presented in Table 6.15-1. For spacecraft of varying mission capabilities, Table 6.15-1 indicates the percent saving for all missions. The percentages are given in reference to single-mission spacecraft, and are based on the total program costs.

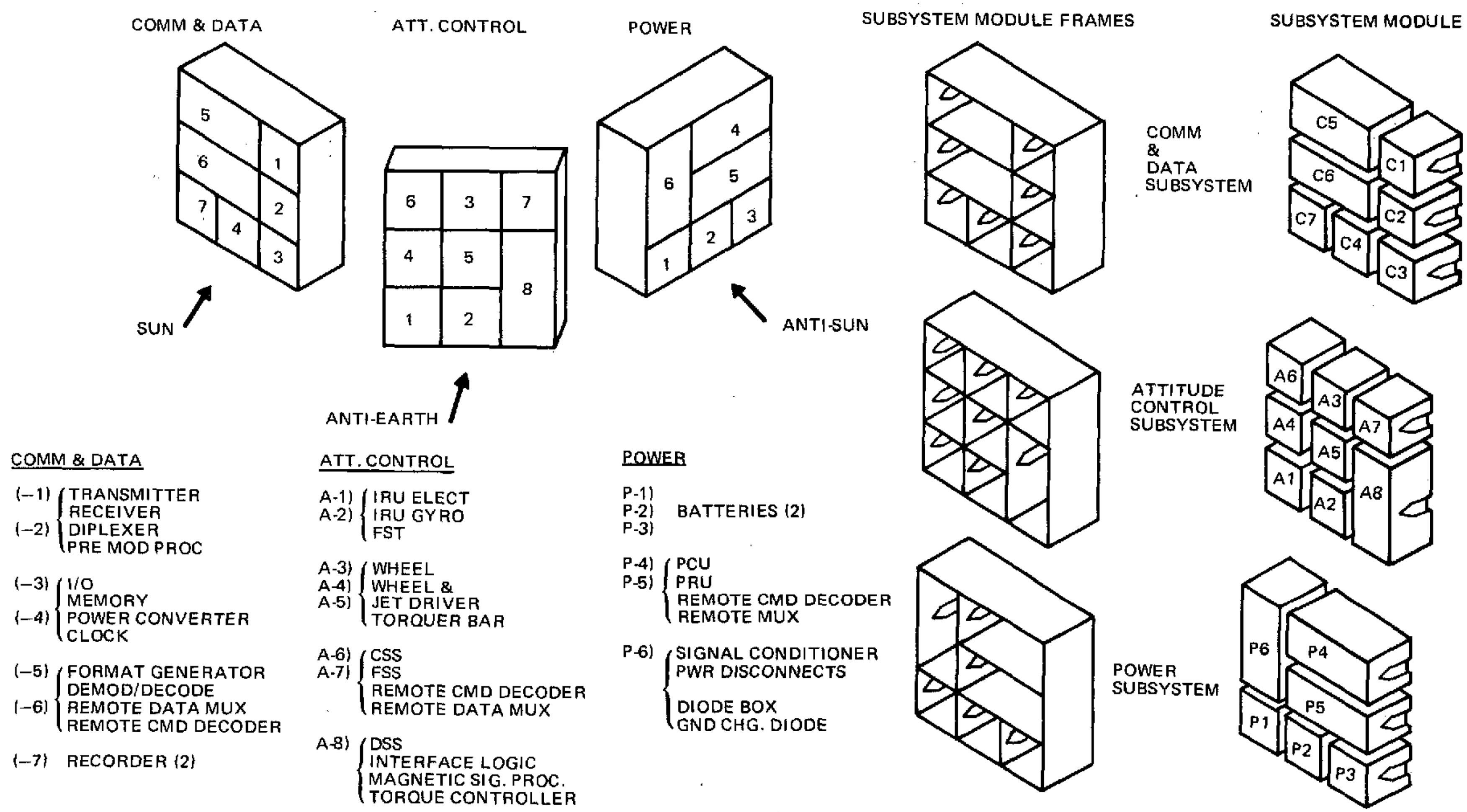

Fig. 6.14-1 EOS – Subsystem Module Configuration

Table 6.15-1 Projected Cost Savings

S/C MISSION CAPABILITY LEVEL	MISSION	% SAVINGS IN DDT&E FOR ALL % MISSIONS	% SAVINGS IN DDT&E AND PRODUCTION FOR ALL MISSIONS
A ONLY	A		
B ONLY	B		
C ONLY	C		
D ONLY	D	0	0
E ONLY	E		
SEASAT A ONLY	SEASAT A		
SEOS ONLY	SEOS		
SMM ONLY	SMM		
A TO C	A		
	B		
	C		
D ONLY	D		
E ONLY	E		
SEASAT A ONLY	SEASAT A	4%	1%
SEOS ONLY	SEOS		
SMM ONLY	SMM		
A TO E	A		
	B		
	C		
	D		
	E	19%	10%
SEASAT A ONLY	SEASAT A		
SEOS ONLY	SEOS		
SMM ONLY	SMM		
GAC B/L	A		
GAC B/L EXTENDED	B		
	C		
	D	31%	22%
	E		
	SEASAT A		
	SEOS		
	SMM		

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It is seen that a 4 percent and 19 percent saving of DDT & E is indicated when the EOS-A spacecraft performance capability is increased to capture missions A to C, and A to E, respectively.

The extended GAC baseline approach yields a 31 percent saving in DDT&E cost in capturing missions B through E plus SEOS, SEASAT-A and SMM. The modifications to the Grumman baseline design to capture various missions are indicated in Table 6.15-2.

Table 6.15-2 Modifications to the Basic EOS Spacecraft to Capture Additional Missions

MISSION CAPTURED	SUBSYSTEM CHANGES REQUIRED					IMPACT ON BASIC S/C COST	
	EPS	ACS	COMM & DH	OA	STRUCTURE	DDT&E	PROD.
A	NONE	NONE	NONE	NONE	NONE	0	0
B	NONE	NONE	NONE	ADD 1 TANK.	INCREASE CAPABILITY.	4%	2%
C	ADD 2 BATTERIES (EACH 20 AMPHRS.) & SOLAR ARRAY AREA	HEAVIER WHEELS AND TORQUERS	NONE	ADD 2 TANKS. ADD SRM.	INCREASE CAPABILITY.	25%	45%
D	ADD 2 BATTERIES & SOLAR ARRAY AREA	NONE	NONE	NONE	NONE	12%	7%
E	ADD SOLAR ARRAY AREA	HEAVIER WHEELS TORQUERS	NONE	ADD 2 TANKS. ADD SRM.	INCREASE CAPABILITY	31%	41%
SEOS	NONE	HEAVIER WHEELS & TORQUERS	NONE	ADD 1 TANK	INCREASE CAPABILITY	58%	35%
SEASAT A	ADD 1 BATTERY & SOLAR ARRAY AREA	NONE	NONE	NONE	NONE	14%	4%
SMM	NONE	HEAVIER WHEELS & TORQUERS	NONE	ADD 1 TANK	NONE	6%	32%

Table 6.15-2 also presents the accompanying DDT&E and production cost impacts. The GAC baseline extension achieves a greater cost saving than the multi-mission spacecraft considered.

Conclusions:

- Conducting all EOS missions with single-mission spacecraft is the most expensive approach.
- Program cost savings increase with increased mission capture capability of multiple-mission spacecraft.
- Greatest program cost savings compared to single-mission spacecraft approach were achieved through addition of performance capability to the Grumman basic spacecraft to capture EOS missions B through E plus SEASAT A, SMM, SEOS.

6.16 SINGLE-SATELLITE VS MULTIPLE SATELLITES

Purpose:

To investigate the total program function and performance advantages as well as the cost impacts of single-satellite vs. multiple-satellite EOS operational missions.

Conclusions: TBD

Summary:

During the course of the study, it became apparent that multiple satellite missions offered many operational advantages over a single-satellite mission. This occurred because of the need for EOS missions to satisfy the dual requirements of an operational and R&D system simultaneously, as well as other considerations. Obviously, the multiple spacecraft system is more costly, but the high non-recurring cost of the payload, as well as DMS considerations, would seem to indicate that the cost differential may not be as significant as was first envisioned.

An important consideration in this issue is what the total weight of the EOS B and B' spacecraft will be. At present it appears that the weight will exceed the payload-to-orbit capability of the Delta 2910. In this case it appears that all of the weight savings discussed in Section 4.1.9 would be cost-effective to incorporate in the design if the Titan III B must be used. The expected cost differential between a weight-constrained Titan and a normal Titan III B design is reflected primarily in the booster cost difference of about $3.7M. The significance of this cost difference may be clearly seen when it is compared to the recurring cost of an entire "barebones" spacecraft, which is of the order of $6M.

An an example, consider the possibility of having no booster available with a payload-to-orbit capability and a price between the Delta 2910 and the Titan III B, interesting options for EOS B and B' result. One case is the trade between four (4) Delta 2910 space segments and two (2) Titan III B space segments. If we assume that in both cases we will fly the same instrument complement (2TM, 2HRPI, and 2DCS), the cost of the instruments are the same. However, in the first case we fly the following:

Delta 2910 Space Segment

Spacecraft Number	Instrument Complement
1	TM, DCS
2	TM, DCS
3	HRPI
4	HRPI

and in the second case the following Titan III B Space Segment

Spacecraft Number	Instrument Complement
1	TM, HRPI, DCS
2	TM, HRPI, DCS

The costs of the first case (without instruments) are:

4 x 5.9M (spacecraft) + 4 x \$4M (Boosters = \$23.6M + \$16M
+ \$39.6M

and for the second are:

2 x \$5.9M (spacecraft) + 2 x \$9.0M (Boosters) = \$11.8M = \$18.0M
= \$29.8M

Thus for an extra \$9.2M a four spacecraft segment can be obtained with the following advantages to the program:

- Capability to manipulate yearly funding by changing the dates of four smaller launches in response to funding changes
- Capability to split the R&D and operational missions, allowing schedule changes in one independent of the other, thus reducing significantly the risk of an R&D instrument holding up an operational launch, and/or eliminating the risk of being required to go with an R&D instrument which is not ready to fly, due to operational commitments.

- Greater flexibility in phasing the space segment for the benefit of the users, and to lower peaks in the high data rate load on the Central Data Processing Facility.
- With a typical phasing of each satellite of about 60° as shown in Figure 6.16-1 there is a possibility of multiple service missions with one flight. For a sample case allowing for a reasonable coast time of 18 hours the Δ V required would be 200 ft/sec.

Another interesting result can be seen if we consider the case of three Delta launches vs. two Titan III B launches. In this case we eliminate the last R&D launch. The launch and spacecraft savings alone amount to $9.9M; thus a 3 Delta 2910 space segment is actually equivalent in cost to a 2 Titan III B space segment. This is not to say that the fourth flight would be eliminated in practice, but if enough R&D information is obtained with the first HRPI flight, then the second flight could be an operational HRPI or an R&D flight for some other sensor.

Of course the analysis presented has been performed only on a gross basis. We have not considered the cost of operations and have not investigated in any depth the additional payload capacity as well as other advantages provided by the Titan III B and the Delta 2910. We have also not considered the effect of the difference in launch vehicle reliability. These additional areas will be included and a more detailed investigation of this trade will be provided in the final study report.

6.17 MANAGEMENT APPROACH

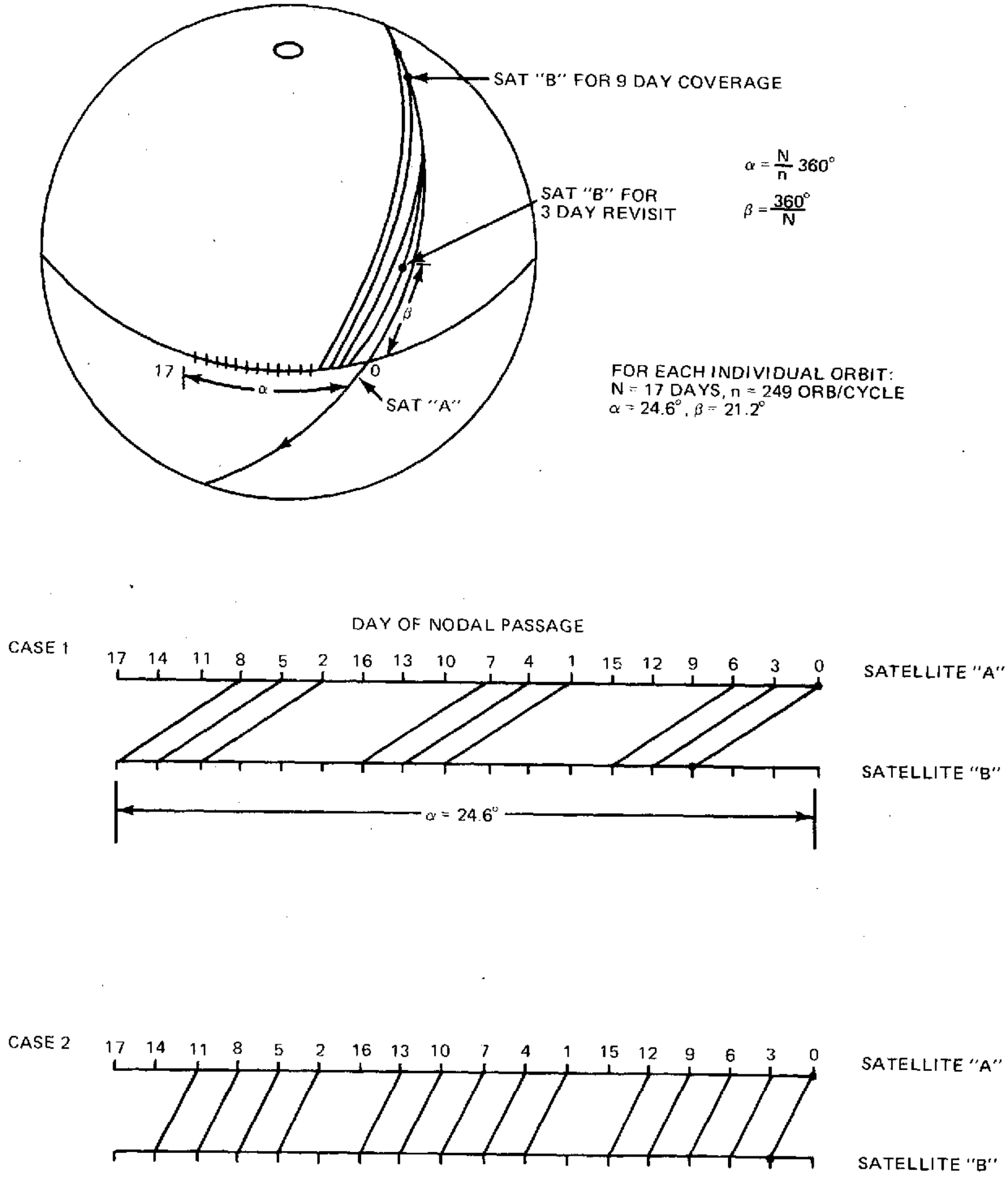
Purpose:

This trade is to determine a practical low cost way of managing and controlling the EOS program.

Conclusion:

It is recommended that target costs be established for the Basic Spacecraft and for the EOS-A, A' program. A Design-to-Cost (DTC) program should then be implemented to achieve this cost. The proposed DTC program requires:

- A System Integration team concept with direct participation and functional tasks performed by NASA personnel and associate contractor personnel.
- Associated simplification of controls and documentation.
- Direct purchase by NASA of the high technology instruments to reduce the added costs incurred when a prime contractor assumes responsibility for development risk.



3-197 **Fig. 6.16-1 Ground Tracks for Satellites Simultaneously Placed in Same Orbit Plane**

The complete discussion of Management Approach is given in report No. 4 and summarized on the following pages.

Summary:

The Earth Observatory Satellite (EOS) Program and the follow-on earth observation mission programs will be conducted in a controlled target cost environment. In this environment the program approach must insure program requirements are met within allocated budget.

Experience has shown that program requirements within specified ranges can be obtained within specific budget costs. Although programs have achieved these results most commonly through a Design-to-Cost (DTC) approach to unit production costs, they have achieved similar results through a DTC approach for the total program. Since the EOS Program has relatively low production volume and development cost is a major fraction of program cost, the recommended program approach is Design-to-Cost on a total Program Acquisition Cost basis.

In this approach, the system definition studies will have established program requirements and design-to-cost goals. The program requirements will be categorized as mandatory or desirable. The design-to-cost goals will be target budgets for major program WBS elements such as Spacecraft, Instruments, Ground Station, Data Processing, etc. Where the program implementation produces an out-of-tolerance condition, the problem will be resolved by reallocation between WBS elements and/or modification of desirable requirements. The net effect will be to maintain a total program cost within prescribed limits by designing to established cost goals and trading performance against cost for selected program requirements.

To manage the program implemented in accordance with the approach described above, we recommend a centralized program manager which we have designated as the System Integrator. This contractor, responsible to the NASA/Goddard EOS project manager, is the basic system contractor for the EOS - Basic Spacecraft, Control Center and Mission Controls, Mission Peculiar Spacecraft, Central Data Processing Facility and Low Cost Ground Station. In addition to the above responsibilities, the System Integrator is responsible for assessing the performance of the Instrument and System GFE contractors. The scope of this assessment includes cost, schedule and technical performance. Where cost/schedule or technical problems develop which cannot be handled

within the latitude of the specific contract, the System Integrator will perform an in-depth analysis of the problem, conducting cost/performance/requirements trades as required. Resultant recommended program modifications to maintain total program costs within established goals are forwarded by the System Integrator to the NASA EOS Program Manager for review, approval and implementation. The System Integrator Concept is also effective in the event program requirements and projected costs reach an incompatible impasse. In this case, the System Integrator would flag the problem with potential alternate solutions for early corrective action by the NASA EOS Project Manager. This concept of program management is shown in Figure 6.17-1.

We envision the System Integrator in his total program role functioning through a working team concept. This working team, under the leadership of the System Integrator, will include personnel from NASA/Goddard, user groups, GFE contractors, and the Instrument contractor as well as the System Integrator. Through this team, it is possible to address all functions of the EOS program and either resolve program problems or conduct the in-depth analysis/trades to formulate problem solving recommendations for the NASA/Goddard project manager, Fig. 6.17-2 and Table 6.17-1

The working team concept will reduce documentation requirements since the various program groups will be intimately involved in program assessment and modification as active team members. Other advantages of this concept are shortened response times and ability to vary team mix as program focus varies through the program phases. As a matter of fact, the System Integrator responsibility may very well be assigned to other contractors for follow-on earth observation missions. NASA/Goddard, System Integrator, and team member responsibilities will be detailed through contractual interface documents and memoranda of agreement.

In addition to the normal expertise contributed by Government personnel, other tasks directly applicable to the EOS program will be performed by Government team members. Verification requirements definition/planning review, residual flight and ground support equipment survey for EOS use and cost effective utilization of Government facilities are examples of the tasks which could be performed. Proposed utilization of Government facilities is the type of recommendation that would be made to the NASA/ Goddard project manager. The System Integrator and his team also provides a central source of current program information which will assist in future mission planning by the NASA/Goddard project manager and other Governmental agencies.

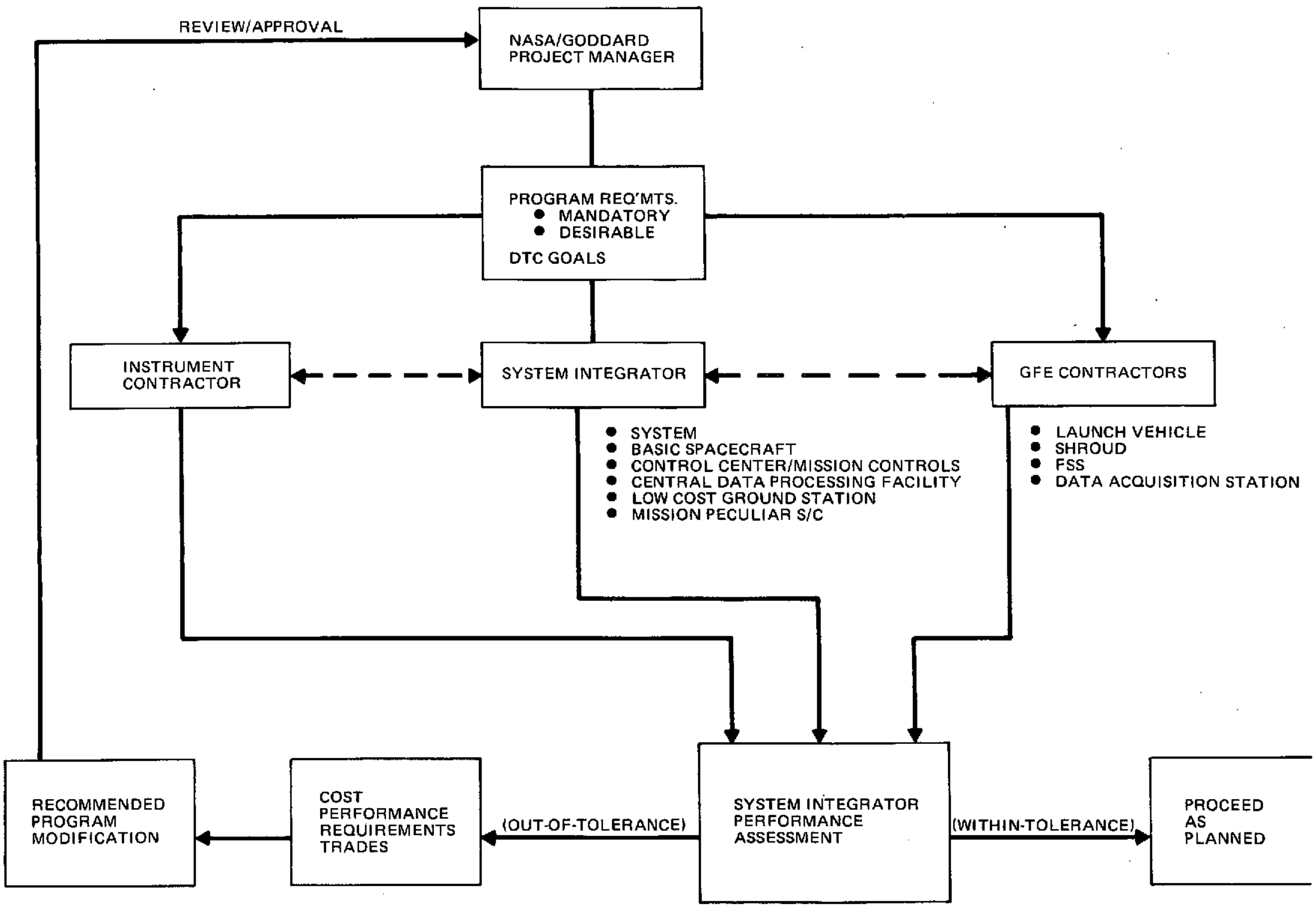
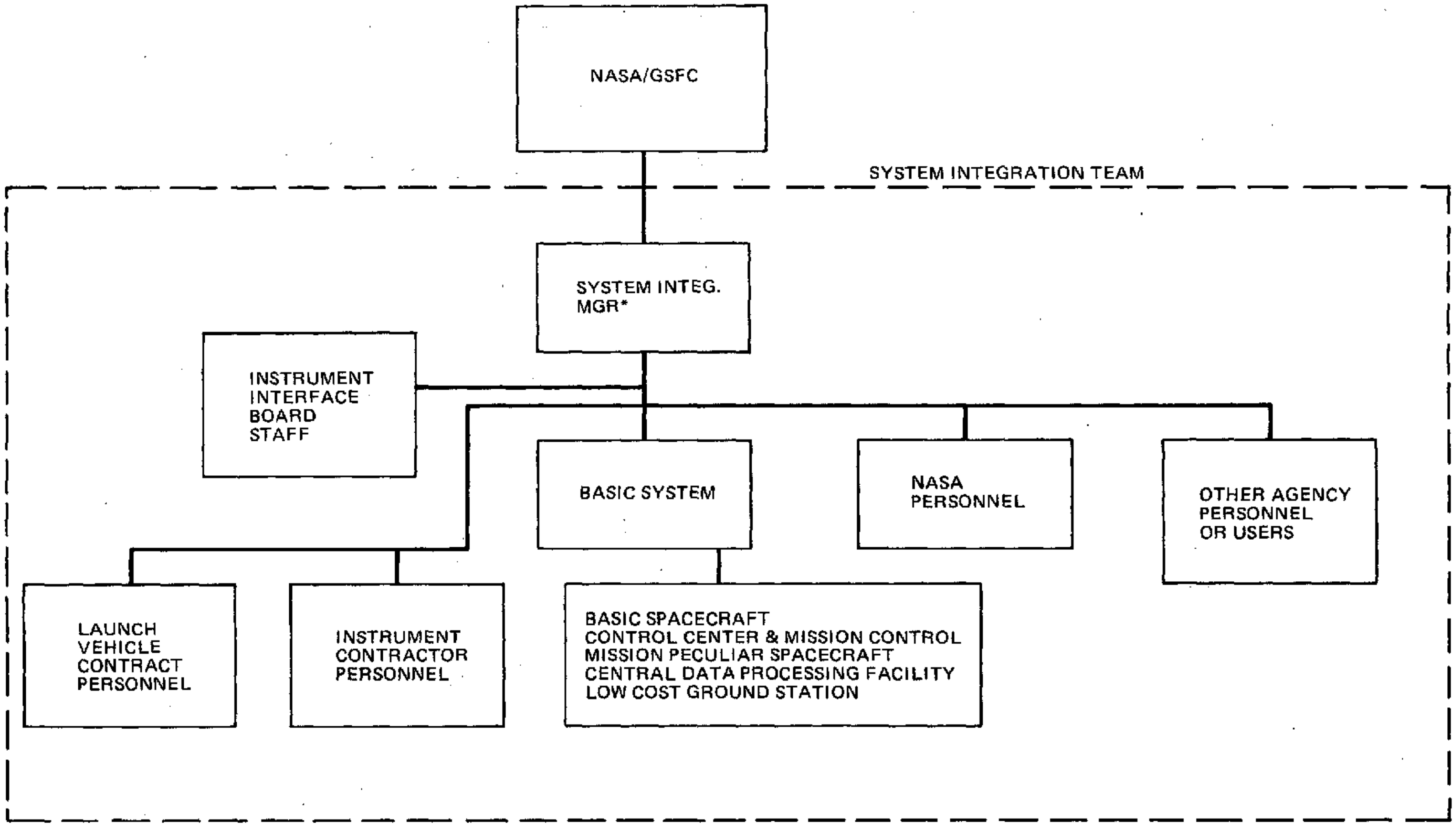


Fig. 6.17-1 EOS Program Management



* BASIC SYSTEM CONTRACTOR FOR EOS A, A'.
FOR FOLLOW-ON MISSIONS MAY BE OTHER
CONTRACTOR OR AGENCY.

Fig. 6.17-2 EOS Working Team

Table 6.17-1 EOS System Integrator Team Members, A Typical Distribution

	EOS A AND A' LRM	OPERATIONAL LRM	MARINE RESOURCES	WEATHER OBSERVATION
SYS. INTEG-CONTR. (3)	20	20	30	30
GOVERNMENT				
NASA/GSFC	15	5	2	10
LOW COST SYS.(1)	1	1	1	1
JPL	–	–	8	–
DEPT. INTERIOR	2	5	2	–
D. AGRICULTURE	2	2	–	–
NOAA, D. COMM.	–	–	5	4
NASA/ULO (1)	1	1	1	1
SCIENCE CONSULTANTS (2)	2	2	2	2
INSTR. CONTR.	4	5	4	4
BASIC SPACECRAFT	INCLUDED IN SYSTEM INTEGRATION	2	2	2
LAUNCH VEHICLE (1)	1	1	1	1

(1) PART TIME
(2) EQUIV. MEN MIX CHANGES BASED ON MISSION
(3) SYSTEM INTEGRATOR SELECTED FOR EACH MISSION

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Management of a DTC/Program Acquisition Cost program requires a total management system for effective implementation. The management system must include a Design-to-Cost system whereby major WBS cost budgets are subdivided down to the lowest level where work is performed - the Work Package Level. The Design-to-Cost system must provide budget visibility for design, manufacturing, test and procurement personnel as well as program management. It must provide a cost visibility so that design iterations and innovative trade studies are performed to establish configuration and detail designs consistent with allocated budgets. Designers must have total responsibility for both cost and performance of their work package. To efficiently carry out these responsibilities they are armed with tools such as Designer's Cost Manuals, Equipment Data Bank, etc. These tools provide the designer with the capability to estimate the cost of a particular design prior to release of a design for manufacturing, procurement, and test activities. This system also provides the capability of flagging for higher level action those areas where budgets/requirements are incompatible.

The Action Center is recommended as the most cost effective way to display EOS program plans, status and trends. This is a working session area displaying cost, schedule, and technical performance data for the total EOS program. Here lower level WBS data is available to Goddard to support the top level reports.

CONTRACTING TECHNIQUES BETWEEN THE GOVERNMENT AND THE PRIME CONTRACTOR

The contractual techniques recommended for the Design-to-Cost EOS A and A' phase of the program considers the cost risk of the major elements of the EOS program and shares this cost risk between the Government and the contractor to reduce the program cost to the lowest level. The EOS A and A' phase is also planned to permit the introduction of multiple procurements of the Basic Spacecraft and the Low Cost Ground Station and provides alternate methods for future procurement. Fig. 6.17-3.

The Instruments for the initial flights are procured by the Government and provided to the System Integrator as GFE. The System Integrator will manage the Instrument contractors through the System Integrator Team and will resolve interfaces within the team or by the Interface Board. For any problems requiring NASA/Goddard Project Management approvals, recommendations will be provided by the System Integrator and the Instrument contractor. The System Integrator will supply the necessary assistance to the NASA/Goddard Project Manager for the procurement and interface in order to fully integrate the Instruments into the Design-to-Cost goals.

The Launch Vehicle, Shroud, FSS, MEMS and modifications to the Data Acquisition Station are to be procured under the normal Government procurement practices. As members of the System Integrator team, representatives of these procurements will participate in the EOS program as associate contractors and the funding for these efforts will be part of the System Integrator Design-to-Cost goals.

The System Integrator is the prime contractor for the EOS A and A' mission, including the Basic Spacecraft, Control Center and Mission Control, Mission Peculiar Spacecraft, Central Data Processing Facility and Low Cost Ground Station.

It is recommended that this selection be made at the earliest time to begin the development of the Basic Spacecraft and to establish the System Integration of the Instruments. To expedite this selection, it is recommended that a preliminary RFP be issued to the contractors for comments. This review will provide a better understanding by Goddard and the contractor when the official RFP is issued.

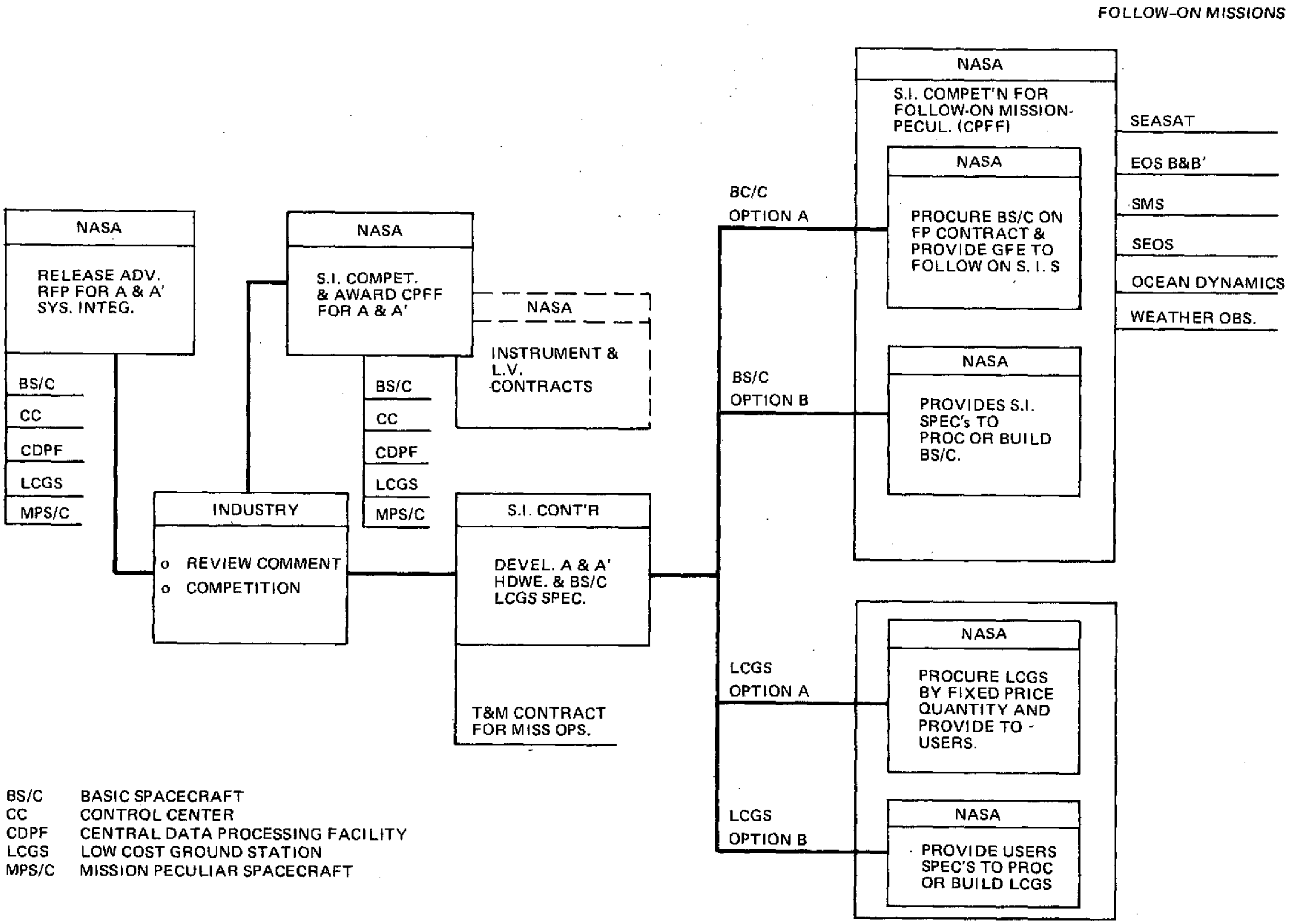
The competition for the EOS A and A' execution phase will be a management and technical competition with the Design-to-Cost goals fixed from information NASA has received from the System Definition Study and the funds allocated for the program defined. Costs will be allocated in this proposal to assist in understanding of the management approach. Total funding and fiscal funding requirements may be established.

It is recommended that a cost plus fixed fee contract by used for this procurement. In accordance to the objectives of a Design-to-Cost program and as required by the System Integrator's responsibility to manage within the Design-to-Cost goals, cost tradeoffs will be a continuous requirement. An incentive could be provided for this management if it is simple and does not hamper the trades which may be required during the program.

Several candidates for fixed price contracting are identified with alternate procurement techniques. The Basic Spacecraft, Modules and the Low Cost Ground Station may be procured by fixed price contracts following their development. For follow-on mission, the Basic Spacecraft or selected Modules may be procured by the Government and supplied to a System Integrator GFE or a procurement package including drawings and specifications which may be supplied for use by the System Integrator. The Low Cost Ground Station may be procured in a similar manner. The Low Cost Ground Station may be procured by the Government for use by the users or the procurement package could be provided for the use of the user.

The DMS operations including the Mission Control, Data Processing operations and support should be contracted by a time and material or labor type basis. Each contract should be by individual contracts rather than by the System Integrator for maximum direct procurement, as a Design-to-Cost is not of significant value during this phase of the contract.

This contractual plan makes full use of a Design-to-Cost philosophy and presents a low cost approach to the EOS A and A' execution phase. It provides the structure to manage within the program funding and the flexibility to manage within fiscal year funding. Also, an early selection of the System Integrator will assist in the Instrument procurement and assist in optimum planning for the Basic Spacecraft. The development of a Basic Spacecraft will also enhance future space programs by providing standard spacecraft hardware for low cost space programs.



BS/C	BASIC SPACECRAFT
CC	CONTROL CENTER
CDPF	CENTRAL DATA PROCESSING FACILITY
LCGS	LOW COST GROUND STATION
MPS/C	MISSION PECULIAR SPACECRAFT

Fig. 6.17-3 EOS Program Development

CONTRACTING TECHNIQUES BETWEEN THE PRIME CONTRACTOR AND SUBCONTRACTORS

Subcontracting will be conducted within the Design-to-Cost goals and desirable requirements identified.

In that the seller assumes the highest degree of risk under fixed price contracting, flexible type contracting should be minimized. Maximum use of "off the shelf" components would appear to complement this basic policy. In areas where development or modification could make use of fixed price contracting counter productive in terms of program interests, flexible type contracting would be used on a selective basis. Where practical, individual subcontracts would be segmented to isolate the areas of uncertainty that lend themselves to flexible pricing and if necessary delay contracting of later phases until definition is sufficiently clear to permit firm pricing.

In concert with the policy of maximum use of fixed price subcontracting, the specifications should convey maximum responsibility to the subcontractor. Maximum responsibility is conveyed by a minimum of detail. One extreme would be to procure equipment "suitable for the use intended" by defining "the use intended". Conversely as you add specific design detail and other restrictive requirements, you assume responsibility for the effect of this detail and create a scope envelope that is more subject to contractual change. However, all critical performance, test and interface parameters will be clearly defined.

Experience has shown that seller's responsibility for the successful performance of his equipment can be effectively extended through installation and checkout in the end article. This is in contrast to the traditional method of basing acceptance upon inspection and test at source or incoming inspection at destination. Selected sellers would be contractually obligated to provide personnel and equipment to participate and shepherd their flight hardware through spacecraft installation and checkout.

Another method of providing selected sellers motivation throughout their subcontract performance will be to provide sellers an opportunity to earn additional fee based upon the performance of their equipment in orbit.

Early and continuing emphasis will be applied to produceability. Initial design reviews will incorporate specific attention to this discipline to insure cost effectiveness and uniform repeatability of the product. Manufacturing methods and processes will receive early attention to provide confidence prior to production release.

It is recommended that documentation requirements for each procurement be "tailored", taking into account the specific use of which it will be put, quality and extent of existing data and formats currently employed by the seller that may differ from specification requirements. Emphasis shall be placed upon the practical needs of the potential users and not on rigid conformance to uniform standard requirements.

Responsibility for monitoring a seller's adherence to the quality assurance provisions of a Grumman subcontract should rest with Grumman. Specifically, acceptance of an endproduct at a seller's plant should be at the discretion of the Grumman quality assurance representative.

If under the subcontract clause of the prime contract, the Government reserves the right of prior approval in the placement of specific types of subcontracts, there should be a time limit established for this approval cycle. This will permit more precise scheduling of the procurement plan and will prevent delays that could ultimately effect equipment delivery. A time period of ten (10) days, after which in the absence of disapproval Grumman would be authorized to proceed, would appear reasonable.

Pooling procurement of critical components that are common to several subcontractors equipment has been found beneficial from a cost, schedule and quality viewpoint. In such cases, Grumman and the Government can benefit from the lower cost from larger volume procurement and maintain greater control over uniform quality of the parts.

As part of the evaluation process of all seller proposals a risk analysis will be prepared. This analysis will identify specific areas of risk in schedule, cost and technical performance. In cases of competitive procurement this analysis will become part of the selection criteria. In addition the analysis will provide the basis for planning the procurements in such a way as to minimize program impact.

In order to minimize total program cost by maximum use of available Government supplies and services, the Government will be considered a potential supplier in areas such as special test equipment, residual flight hardware, engineering services and test facilities. Program requirements in these areas will be by the System Integrator team to insure taking advantage of opportunities that may exist.

SUBCONTRACTING TECHNIQUES FOR THE MODULES

For the design and development of the modules, a comparison between development by the prime contractor or by the prominent supplier of the module components indicates a 15% savings if the module is developed by the prime contractor. This is based upon data on the EOS, Attitude Control System Module. In the production phase of the modules, no significant difference in cost is indicated.

6.18 TEST PHILOSOPHY

6.18.1 SUMMARY

The total EOS Development, Qualification, Integration and acceptance test program is shown in the master program schedule. The approach shown is based on the basic EOS test requirements identified in Section 6.18.2 and the trade studies documented in Section 6.18.3, which examine the alternative approaches to satisfying both the basic LRM mission and common spacecraft test requirements.

The recommended approach features:

- Combining all System and Component environmental acceptance tests at the module level representing a cost savings of 500K or 50% per S/C of environmental acceptance test costs over the conventional, component and system environmental test approach, at virtually no program cost, schedule or technical risk.
- Qualification of the basic spacecraft structure and modules for follow-on as well as the basic missions, to provide a level of design confidence which permits NASA to take advantage of the cost benefits of a multi buy spacecraft procurement plan. Based on subcontractors estimates for a 30 unit buy versus a 5 unit buy this could represent a 20% cost savings in component costs alone.
- Separate component and module qualification tests to insure that component qualification levels are adequate to cover follow-on mission environments.
- Verification of the flight instrument and ground processing system compatibility and functional performance, independent from the basic spacecraft flow, providing both a low cost approach to demonstrating the EOS mission peculiar hardware and software flight readiness as well as maximum I & T schedule contingency and flexibility.
- Utilizing the high percentage of developed subsystem avionics hardware to permit elimination of a costly bench or laboratory avionics development spacecraft.
- Making the systems qualifications spacecraft available for performing Shuttle ground and flight EOS on orbit resupply demonstration, and/or refurbishment for flight.

Integration and Test

- Basic Spacecraft

The Integration and Test program for a typical set of EOS flight hardware is shown in Figure 6.18-1. The components are functionally tested at the subcontractor and shipped to the prime contractor for integration into the modules. Once the modules are integrated and functionally checked the modules are subjected to either acoustic or mechanical vibration, 2 days of thermal cycling and a 6 day Thermal Vacuum test to verify component and module work manship. The current plan calls for serial testing of the modules using the same Test and Integration (T&I) station and unique software interfacing through the module remote decoders and multiplexers. Individual hardwired GSE is used for module power and monitoring of hardlines during test. The non-mission unique on board software will be developed, debugged, and qualified in the software development lab. Integration of the flight software with the flight computer will be accomplished during integration of the C & D H module. Initial software required for the ACS processing during ACS module tests will be simulated from the T & I station.

Upon completion of the module level tests the subsystem modules will be integrated together, and functionally checked as a system on the flight spacecraft back-end. At this point we have a basic spacecraft ready for integration of the mission peculiar Instruments, Instrument module, mission software and Orbit Adjust/Reaction control system module.

- Mission Peculiars

The Instrument, IMP and Wideband antenna will be tested as a system, at GSFC, with the DMS Primary Ground Station to demonstrate the flight and ground system compatbility, prior to integration of the Instrument with basic spacecraft. Observatory/Control Center and Network compatibility will be demonstrated via RF through the WTR ground station.

Integration of the mission peculiars could directly follow the basic spacecraft buildup as shown in Figure6.18-1 or be downstream with the spacecraft held in controlled environment storage facilities. In the typical flow and schedule shown, the next step would be to integrate the mission peculiar hardware and software, and perform observatory level systems performance tests, including EMC. The separation and solar array deployment mechanisms are then tested. The solar array size and potentially its deployment techniques is mission peculiar, therefore performed at this point in the schedule. The separation test, however, could be performed earlier, but is performed here as a matter of convenience. Prior to shipment of the observatory of WTR for launch, workmanship acoustic test of the integrated observatory is shown. The observatory, in flight configuration with the exception of pyrotechnics, will be transported to WTR in the horizontal position. WTR prelaunch operations are scheduled for 5-6 weeks as a target since no extraordinary tasks or special tests are required at the launch site. Shuttle, Titan and Delta launch vehicle flows a all permit observatory integration with the launch vehicle about 7-10 days prior to launch.

Development and Qualification test

The recommended Development and Qualification test program for the EOS flight hardware is shown in Figure 6.18-2. Since it is a design goal to accommodate all missions with the same basic spacecraft, the recommended test program considers the basic EOS Land Resources Mission configuration qualification as well as qualification of the basic spacecraft for follow on missions.

The recommended plan provides for module, thermal development and structural qualification testing at the module level. The thermal development tests are required on at least two of the subsystem modules to verify the thermal analysis model. Acoustic and mechanical vibration qualification tests are performed at the module level for two reasons; to determine component environments seen during module tests in order to evaluate module level acceptance test methods, and to qualify the basic module structure.

Spacecraft

The same modules used for module level tests with the component mass reps installed will be used for vehicle level tests. The LRM OAS and IMP module configuration will be used for Observatory Structural Qualification. OAS/RCS IMP modules for follow on missions will be qualified where required at the module level. The Program Requirements (Volume 3, Appendix D) calls for vehicle static tests to verify structural loads, however, based on the trades studies documented in section 3 of this volume, a vehicle level acceleration is recommended as a more cost effective approach to verifying the module and S/C static strength. An early acoustic and mechanical vibration test is recommended for both the LRM and follow on configuration for two reasons;

1. to provide early verification of component environmental levels to preclude downstream requal
2. to permit an early evaluation of the total spacecraft environments vs the design environment for the various configurations.

The Observatory model survey is required for the cantilevered and free-free configuration for both the LRM and follow on configuration.

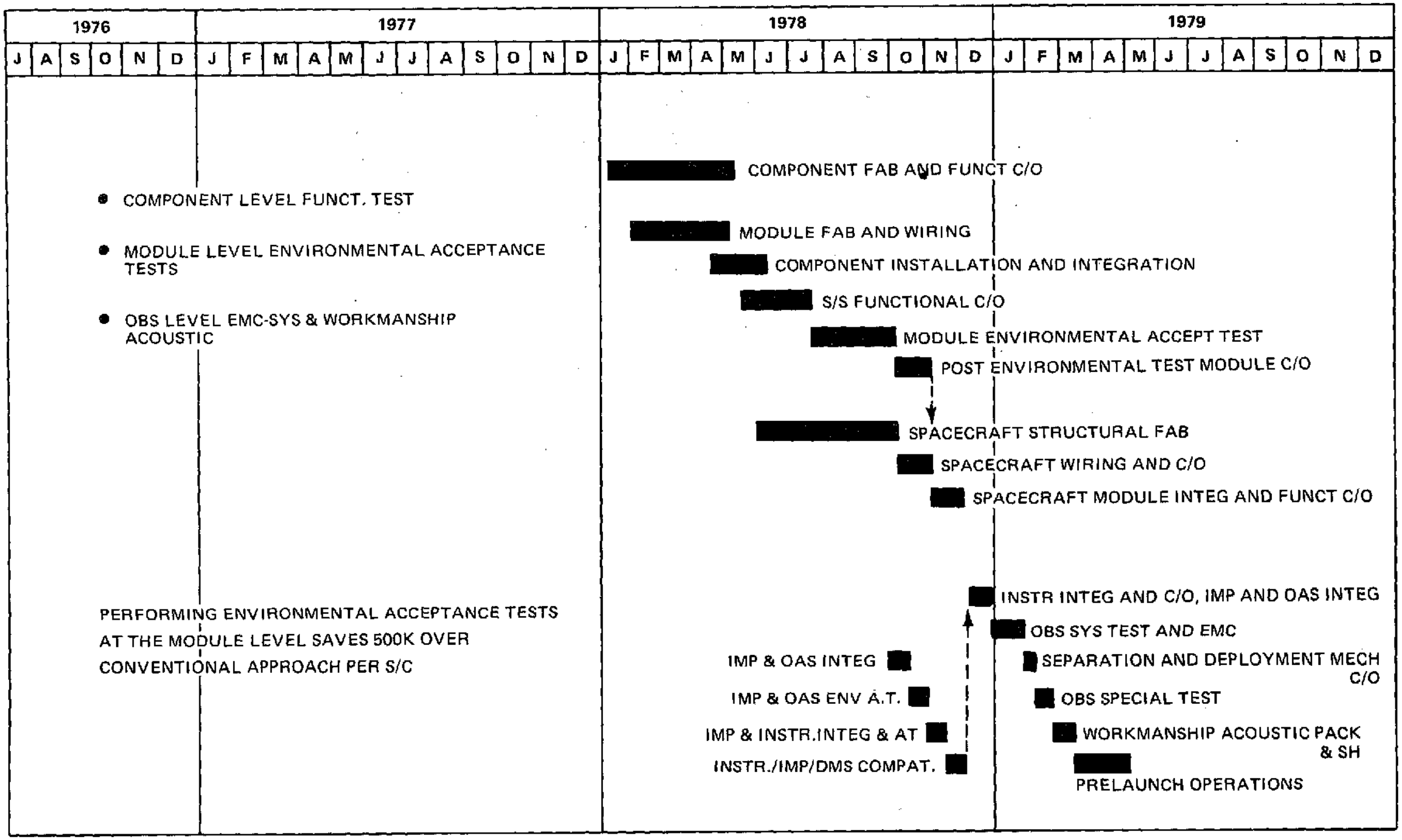


Fig. 6.18-1 EOS Flight Hardware Test Program and Flow

- Mechanisms

Qualification of the separation, and deployment mechanism as well as the basic LRM solar panel structural design can be qualified during the observatory level qualification tests. Earlier in the program off line development tests of the mechanisms will be required to verify the basic design approach.

- Antenna Patterns

S Band X Band and DCS antenna pattern tests are scheduled early after contract go ahead to finalize antenna locations, and verify link analysis for the LRM Configuration. These models should be maintained, to verify follow-on mission antenna patterns.

- Observatory Systems

Following the structural qualification the vehicle is reconfigured for Observatory Systems Qualification, including a qualification model LRM instrument consisting of EMC, Acoustic, Thermal Vacuum, RFI and pyrotechnic shock test. Vendor qualification components where available or a flight component are required for Observatory level system qualification. Three and one half months of S/S module integration is allowed for in the schedule to cover first time integration of the subsystems. Since parallel integration of the modules can be performed by time sharing the T & I station, this should be adequate to debug the basic spacecraft subsystems and test procedures.

Completion of System qualification tests permit the qual components to be refurbished for flight or utilization of the entire qual vehicle for the demonstration of shuttle resupply concepts in both ground and shuttle flight test.

- Instrument Interfaces

A Spacecraft simulator and the mission peculiar IMP module will be required to support instrument manufacturers for functional interface verification during instrument qualification and acceptance tests.

Ground Systems and Instruments

Figure 6.18-3 shows the Ground System Hardware and software flow for both the DMS and mission operations control center.

- Instrument Data Acquisition

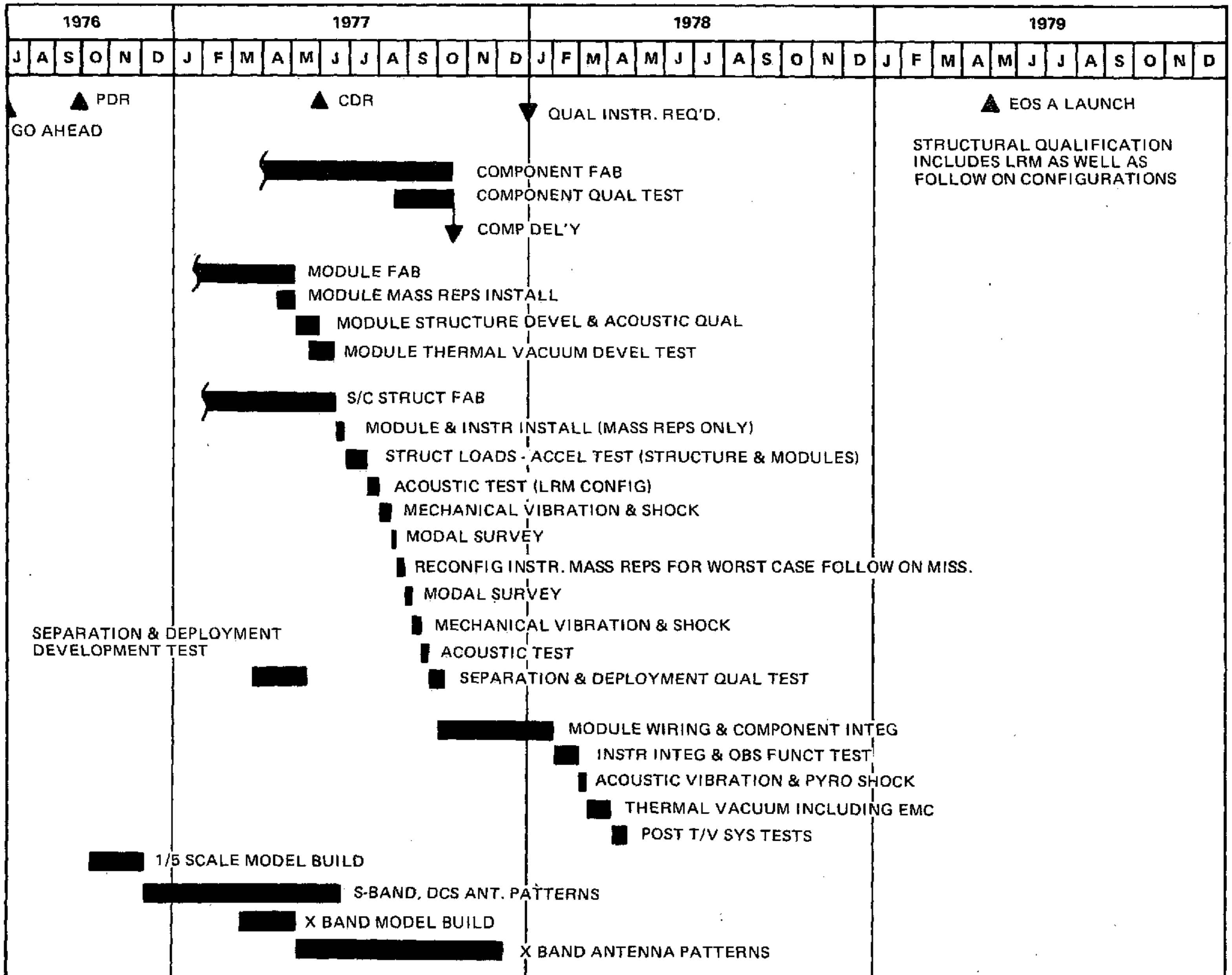


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Fig. 6.18-2 Flight Hardware Development and Qualification Test Summary

Instrument Data Acquisition during Observatory Qualification and acceptance tests will be accomplished via a low cost Ground Station front end which will be part of the T & I station. Observatory test tapes will be provided to validate Low Cost Ground Station field site installations. Primary ground station-observatory compatibility tests can be demonstrated by testing the Instrument - IMP and spacecraft simulator setup at GSFC prior to integration of the instrument with the flight spacecraft, or by shipping the entire observatory to GSFC, prior to delivery for WTR for launch. The cycling of the Intrument, IMP and spacecraft simulator through GSFC was choosen since the total Instrument data train including RF is checked, at minimum cost without tying up the entire T & I crew and observatory. During observatory systems tests Instrument performance will be evaluated by internal instrument calibration monitoring, at the T & I station, for both wide band full data rate operations and the Low Cost Ground Station lower data rate.

- Control Center

Control center check out and personnel simulations training is recommended to be accomplished by software simulation because it provides maximum flexibility in schedule, and follow on mission reconfiguration, at minimum cost.

6.18.2 Study Alternatives

The following Trade Study Report summarizes the EOS program verification requirements, and the alternatives and recommended approach to satisfying each test requirement. The significant trade studies performed in support of the selection of the recommended approach are documented in subsection 6.18.3 of this Volume.

The first column of the following summary identifies the verification requirement. The second column is broken down into five categories.

1. ANAL - Indicates analysis can satisfy part or all of requirement.
2. TEST - Indicates a test is required to satisfy the requirement.
3. OPT - Indicates a significant cost saving option to "business as usual" is permitted because of the unique EOS design.
4. DES - Indicates that there are varying design approaches which influence test requirements.
5. A.T. - Indicates that for this requirement there is a functional or environmental acceptance test requirement, to verify workmanship of the flight hardware in addition to development and qualification test requirement.

The next 3 columns indicate the alternatives for satisfying the requirement and the last column indicates the recommended alternative.

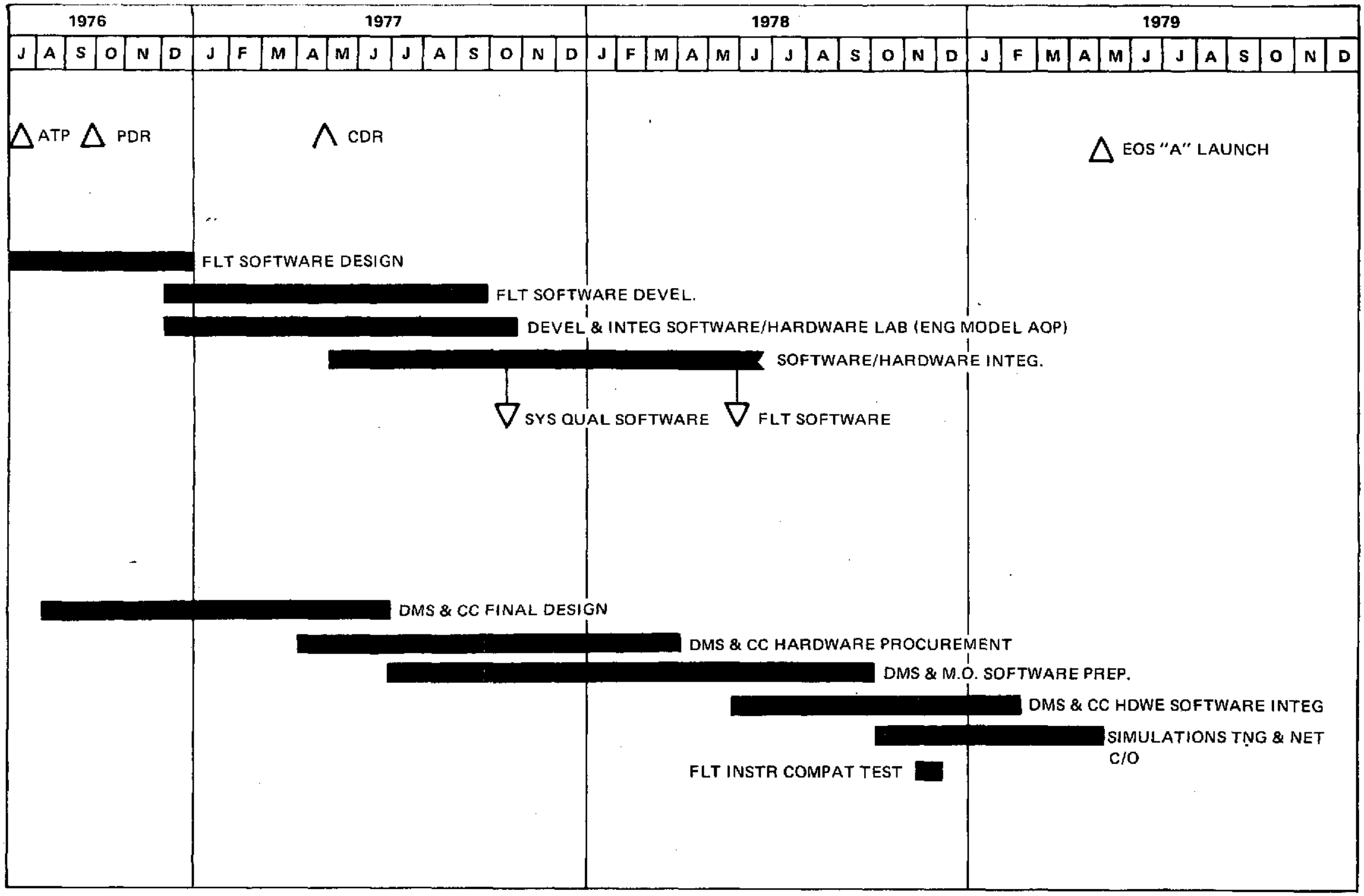

Fig. 6.18-3 EOS Flight and Ground Software and Ground Systems Integ and Test

06/68-9

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7/1.7.3.14

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
1.0 STRUCTURAL									
1.1 SPACECRAFT									
VERIF. STRUCT STATIC DESIGN LOADS		X				STRUCT QUAL TEST (STATIC LOAD)	STRUCT QUAL TEST (CENTRIFUGE)		1-LOWEST TOTAL PROG COST
VERIF STRUCT DYNAMIC DESIGN LOADS (MECH VIB) & SHOCK		X				STRUCT QUAL TEST MASS REPS	STRUCT QUAL TEST OPERAT. EQUIP.	ACCOUSTIC TEST IN LIEU OF MECH.	1- " "
VERIFY STRUCT NAT. FREQUENCIES (3 COND)		X				STRUCT MODAL SURVEY CENTER-LEVER, (2) FREE-FREE			1-
VERIFY STRUCT ACOUSTIC DESIGN LOADS		X				STRUCT. QUAL TEST WITH MASS REPS.	STRUCT. QUAL TEST OPERATING EQUIP.		1 & 2 LOWEST COST RISK
1.2 INSTRUMENT									
VERIFY INSTR. STRUCT. STATIC DESIGN LOADS		X	1			TEST SEPARATELY	TEST AS PART OF SPACECRAFT		2 LRM 1 FOLLOW ON
VERIFY INSTR. STRUCT DYNAMIC DESIGN LOADS MECH VIB & SHOCK		X	1			TEST SEPARATELY	TEST AS PART OF SPACECRAFT		2 LRM 1 FOLLOW ON
VERIFY INSTR STRUCT NAT. FREQUENCIES		X	1			TEST SEPARATELY	TEST AS PART OF SPACECRAFT		2 LRM 1 FOLLOW ON
VERIFY INSTR STRUCT ACOUSTIC DESIGN LOADS		X	1			TEST SEPARATELY	TEST AS PART OF SPACECRAFT		2 LRM QUAL 1 FOLLOW ON QUAL
1.3 MODULE									
VERIFY MODULE STRUCT STATIC LOADS	X	X	1			ANALYSIS AND/OR OVER DESIGN	STRUCT QUAL TEST(CENTRIFUGE OR STATIC LOAD)	STRUCT QUAL ON CENTRIFUGE WITH VEH.	3 LOWEST COST
VERIFY MODULE STRUCT DYNAMIC LOADS MECH VIB & SHOCK		X	1	X		STRUCT QUAL TEST MASS REPS	STRUCT QUAL TEST OPERAT. EQUIP	STRUCT QUAL WITH VEHICLE & VEH CONFIG OPTIONS	1 & 3 LOWEST TOTAL PROG COST

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1 LOW COST MGMT OPTION

GRUMMAN

TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7/ (cont)

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
1.3 MODULE (cont'd)									
VERIFY MODULE STRUCT NAT. FREQUENCIES		X	1			VEHICLE LEVEL MODAL SURVEY			1
VERIFY MODULE STRUCT ACOUSTIC DESIGN LOADS		X	1			STRUCT TEST MASS REPTS	STRUCT QUAL TEST OPERAT. EQUIP.	STRUCT QUAL TEST WITH VEH & VEH CONFIG. OPTIONS	1 & 3 QUAL LOWEST TOTAL PROG COST
1.4 OAS/RCS									
VERIFY OAS STRUCT STATIC LOADS	X	X	1	X		STRUCT QUAL TEST (STATIC LOAD) OR (CENTRIFUGE)	STRUCT QUAL TEST WITH VEHICLE & VEHICLE TEST OPTS	ANALYSIS AND/OR OVERDESIGN	2 - LRM QUAL 1 - FOLLOW ON MISSIONS
VERIFY OAS STRUCT DYNAMIC LOADS MECH VIB & SHOCK		X	1	X		C MODULE QUAL TEST OFF VEH.	QUAL ON VEHICLE		2 - LRM QUAL 1 - FOLLOW ON MISSIONS
VERIFY OAS STRUCT NAT. FREQUENCIES		X	1	X		VEHICLE MODAL SURVEY			1
VERIFY OAS STRUCT ACOUSTIC DESIGN LOADS		X	1	X		C MODULE QUAL TEST OFF VEH	QUAL ON VEH		2 - LRM QUAL 1 - FOLLOW ON VEH.
1.5 ORBIT TRANSFER									
VERIFY ORB XFER STAGE STRUCT STATIC DES.	X	X				TEST AS MODULE	TEST WITH VEHICLE (CENTRIFUGE) OR (STATIC LOAD)	ANALYSIS AND OR OVER DESIGN	NOT APPLICABLE: WILL BE PART OF OAS/RCS MODULE
VERIFY ORB XFER STAGE STRUCT DYNAMIC DES. MECH VIB & SHOCK	1	X				TEST AS MODULE	TEST WITH VEHICLE		
VERIFY ORB XFER STAGE STRUCT NAT FREQ.		X				TEST WITH VEH			
VERIFY ORB XFER STAGE STRUCT ACOUSTIC LOADS		X				TEST AS MODULE	TEST WITH VEHICLE		

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1 LOW COST MGMT OPTION

GRUMMAN

TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7 (continued)

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL.	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
1.6. SOLAR ARRAY									
VERIFY SOLAR ARRAY STATIC DES. LOADS (2 COND)	X	X		X		SEPARATE TEST STATIC LOAD OR CENTRIFUGE	TEST ON VEHICLE WITH VEH. OPTIONS OF NON CELLED ARRAY	TEST ON VEHICLE WITH VEHICLE OPTIONS OF CELLED ARRAY	2 - PRELIM. - DEPENDS ON ARRAY
VERIFY SOLAR ARRAY DYNAMIC DESIGN LOADS MECH. VIB&SHOCK		X		X		SEPARATE TEST	TEST ON VEHICLE WITH VEH. OPTIONS OF NON CELLED ARRAY	TEST ON VEHICLE WITH VEHICLE OPTIONS OF CELLED ARRAY	2 " "
VERIFY SOLAR ARRAY NAT FREQUENCY		X		X		SEPARATE TEST	TEST ON VEHICLE WITH VEH. OPTIONS OF NON CELLED ARRAY	TEST ON VEHICLE WITH VEHICLE OPTIONS OF CELLED ARRAY	2 " "
VERIFY SOLAR ARRAY ACOUSTIC DESIGN LOADS		X		X		SEPARATE TEST	TEST ON VEHICLE WITH VEH. OPTIONS OF NON CELLED ARRAY	TEST ON VEHICLE WITH VEHICLE OF CELLED ARRAY	2 " "
2.0 THERMAL									
2.1 SPACECRAFT									
VERIFY SPACECRAFT THERMAL BALANCE	X	X	1	X		FULL UP OBSERVAT.QUAL. TEST	ELEM. + S/C QUAL. TEST	MODELS + S/C QUAL TEST	2 - LOW RISK
2.2 MODULE									
VERIFY MODULE	X	X	1	X		TEST ON FULL VEHICLE QUAL	QUAL TEST ON VEHICLE + MODEL	SEPARATE QUAL TEST	2 - LOW RISK
2.3 VERIFY INSTR. THERMAL BALANCE	X	X	1	X		TEST ON FULL VEHICLE QUAL	QUAL TEST ON VEHICLE + MODEL	SEPARATE QUAL TEST	3 - LOW RISK
2.4 VERIFY OAS/OTS THERMAL DESIGN	X	X	1	X		TEST ON FULL VEHICLE QUAL	QUAL TEST ON VEHICLE + MODEL	SEPARATE QUAL TEST	2 - " "
3.0 MECH. SYSTEMS									
3.1 SPACECRAFT									
VERIFY SPACECRAFT S/A DEPLOYMENT DES		X		X		SEPARATE TEST	TEST ON QUAL. VEH.		1 & 2 - PRELIM DEPENDS ON DESIGN
VERIFY SOLAR ARRAY ORIENTATION DES.		X		X		SEPARATE TEST	TEST ON QUAL. VEH.		1 & 2 - " "
VERIFY SPACECRAFT SEPARATION DESIGN		X		X		SEPARATE TEST	TEST ON QUAL. VEH.		1 & 2 - " "
VERIFY ANTENNA POINTING SYS. DESIGN		X		X		SEPARATE TEST	TEST ON QUAL. VEH.		2 -

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1 Low Cost Management Option

GRUMMAN

TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7. continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
3.2 PYROTECHNIC DEVICES									
VERIFY PYRO LATCH MECHANISM DESIGN		X				ELEMENT TEST			1
VERIFY PYROTECHNIC CARTRIDGE DESIGN		X				ELEMENT TEST			1
4.0 GND&FLT SYSTEMS & FUNCTIONAL PERF									
4.1 EOS SYSTEM									
VERIFY INST DATA ACQUISITION & TRANSMISSION			1		X	QUAL OBS.	FLT IMP + INSTR AT PRI GND STA SITE		2 - LOW COST & RISK
VERIFY FLT & GND INSTR. DATA HANDLING COMPATIBILITY			1		X	QUAL OBS	"		2 " "
VERIFY CONTROL CENTER/SPACE CRAFT COMPATIBILITY					X	FLT OBS.	QUAL OBS		1 - SCHEDULE
VERIFY SPACECRAFT/GROUND NETWORK COMPATIBILITY					X	FLT OBS	QUAL OBS		1 - SCHEDULE
VERIFY DCS PERF FROM LOCAL XMITTER THROUGH TO GPS	X				X	FLT OBS	QUAL OBS	SIMULATION TAPES + ANAL.	3 - LOW COST 1 - WITH T&I STA ONLY

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1-LOW COST MANAGEMENT OPTION

GRUMMAN

TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPS	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.2 SPACECRAFT									
VERIFY INSTRUMENT/ SPACECRAFT FUNCT. COMPATIBILITY		X			X	VEH.	SIMULATOR + VEH.	BENCH S/C	1 - QUAL & FLT VEH (LRM) 1 - FLT VEH (FOLLOW ON)
VERIFY INTEGRATED S/C SYS PERF UNDER NOM & OF NOM ENV.		X	1		X	VIB & T/V AT VEH. LEVEL + COMP LEVEL	VIB & T/V AT MODULE + COMP LEVEL	VIB & T/V AT MODULE LEVEL ONLY	1 - QUAL - LOW RISK 3 - ACCEPT - LOW COST
VERIFY SPACECRAFT STABILIZATION AND CONTROL MODES	X	X			X	FUNCT. PERF. ON QUAL VEH. + ANAL	MODULE DYNAMIC TESTS + VEH FUNCT. + ANAL	FLT VEH FUNC. PERF. + ANAL	1 - QUAL 3 - ACCEPTANCE
VERIFY ERS POWER BUDGETS VS S/C LOADS	X	X			X	PERF MISSION PROFILE ON VEH WITH SIM. S.A. INPUTS	BENCH S/C	ANALYSIS	1 - QUAL & ACCEPTANCE
VERIFY TLM & CND CAPABILITY TO ALL S/C SYSTEMS INCL PYRO CKT FUNCT.		X			X	VEH	BENCH S/C		1 - " "
VERIFY N.B. COMM. SYS PWR MARGAINS & FREQUENCIES		X			X	VEH	BENCH S/C	MODULE	1 - " "
VERIFY SPACECRAFT N.B. & W.B. ANTENNA PATTERNS		X				SCALE MODEL ANTENNA PATTERN			1
VERIFY W B COMM. SYS. PWR MARGAINS & FREQUENCIES		X				VEH	BENCH S/C	MODULE	1
VERIFY SPACECRAFT SYSTEM EMC IN ALL OPERATING MODES		X			X	FULL MATRIX + INDUCED ON VEH.	MATRIX ONLY ON VEH.		2 - PREL SELECTION FOR QUAL& A.T.
VERIFY RFI COM-PATIBILITY OF S/C AND LAUNCH SYSTEM		X			X	QUAL VEH. WITH SIM RFI	SHIP QUAL VEH TO WTR.	FLT VEH.	1 & 3 LOW COST
VERIFY OPB SOFTWRE PERF. FOR ALL NOMINAL AND OFF NOMINAL OPERATING MODES		X			X	SOFTWARE INTEG. LAB	MODULE LEVEL TEST	QUAL VEH. TEST	1, 2, & 3

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1 LOW COST MANAGEMENT OPTION

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.2 SPACECRAFT (cont.)									
VERIFY S/C-L/V FUNCTIONAL & MECHANICAL COMPATIBILITY		X				QUAL. VEHICLE	INTERFACE SIMULATORS WITH FLT. OR QUAL INTERSTATE	FLT. VEHICLE	2 - LOW COST
VERIFY SPACECRAFT WRING HARNESS CONTINUITY		X			X	HAND C/O	AUTOMATED C/O		2 - BASED ON CONTINUED BUILD LOWEST COST & LEAST RISK.
VERIFY S SYS. PERF FOR BUS VOLT EXTREMES, ARRAY INPUT LIMITS AND MAX AND MIN LOADS		X			X	VEHICLE	BENCH S/C	MODULE	1 & 3 QUAL & FLIGHT
VERIFY SPACECRAFT ACS SENSOR - INSTR. ALIGNMENT MAINTENANCE		X				S/C PRE & POST ENV. TEST ALIGNMENT CHKS.	PRE ENV. TEST ALIGNMENT WITH INSTRUMENTED S/C TO VERIFY NO ALIGNMENT SHIFT DURING TEST		1 - QUAL TBD FOR ACCEPT.
VERIFY SPACECRAFT MAGNETIC MOMENT	X	X				ANALYSIS	MAGNETIC SURVEY		1 - NOT A SIGNIF. INFLUENCE ON CONTROL SYS FOR EOS
4.2.1. EPS MODULE & SOLAR ARRAY									
VERIFY ALL POWER MODULE COMMAND FUNCTIONS THROUGH REMOTE DECODERS		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY POWER MODULE TELEMETRY OUTPUTS FROM REMOTE MUX		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY POWER MODULE OUTPUT PWR TO C&DH, ACS INSTR. & SIGNAL CONDIT. MODULE FOR BUS VOLT. EXTREMES & LOAD LIMITS		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY PRU/PCU PERF. REQMT. FOR ARRAY INPUT VOLT EXTREMES		X			X	MODULE LEVEL	VEHICLE LEVEL		"

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NOTE A: PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL.

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.2.1 EPS MODULE & SOLAR ARRAY (continued)									
VERIFY BATTER, VOLT TEMP, CURRENT UNDER TYPICAL ORBITAL MISSION PROFILES		X	1		X	MODULE LEVEL ENV. TEST	VEHICLE LEVEL ENV TEST	AMBIENT TESTS AT EITHER MOD OR S/C LEVEL	2 - QUAL 1 & 3 - ACCEPT.
VERIFY EPS MODULE PERF. UNDER NOM & OFF NOMINAL ORBITAL ENV.		X	1		X	MODULE LEVEL ENV. TEST	VEHICLE LEVEL ENV. TEST		2 QUAL 1 ACCEPTANCE
VERIFY SOLAR ARRAY OUTPUTS FOR EXP. ORBITAL ENV.	X	X		X	X	SEPARATE ARRAY OR PANEL T/V TESTS FLT. OR QUAL.	FLASHER-SOLAR SIM. FLT. OR QUAL.	ARRAY ON VEHICLE FLT. OR QUAL.	1 & 2 QUAL AND FLT - PRELIM. 2 - LOW COST & RISK
VERIFY EPS MOD WIRING HARNESSES					X	HAND C/O	AUTOMATED C/O		2
4.2.2 ACS MODULE									
VERIFY MODULE COMMAND & SENSOR INPUTS THROUGH REMOTE DECODERS		X			X	MODULE	QUAL. VEH. LEVEL		SEE NOTE A
VERIFY MODULE TLM OUTPUTS THROUGH REMOTE MUX		X			X	MODULE	QUAL. VEH. LEVEL		" "
VERIFY THE ACS MODULES & SVN SENSOR ANALOG PROCESSOR JET CONTROL LOOP GAINS	X	X			X	COMP. ANAL ONLY	HYBRID SIM	ANALYSIS + MODULE OR VEH. LEVEL TEST	3
VERIFY THE ACS MODULE & SVN SENSOR ANALOG PROCESSOR REACTION WHEEL CONTROL LOOP GAINS	X	X			X	COMP ANAL ONLY	HYBRID SIM	ANALYSIS + MODULE OR VEH LEVEL TEST	3

1 - LOW COST MGMT OPTION

NOTE A PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS

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FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.2.2 ACS MODULE (cont)									
VERIFY THE ACS MODULE & SUN SENSOR ANALOG PROCESSOR JET FUNCT. PERF. & POLARITY		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY THE ACS MODULE & SUN SENSOR ANALOG PROCESSOR		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY THE ACS MODULE & GYRO ANALOG PROCESSOR REACTION WHEEL LOOP GAINS	X	X			X	COMP. ANAL ONLY	HYBRID SIM	ANALYSIS & MODULE OR VEH. LEVEL TEST	3
VERIFY THE ACS MODULE & GYRO ANALOG PROCESSOR JET LOOP GAINS	X	X			X	COMP. ANAL ONLY	HYBRID SIM	ANALYSIS + MOD OR VEHICLE LEVEL TEST	3
VERIFY THE ACS MODULE & GYRO ANALOG PROCESSOR MOMENTUM WHEEL POLARITY & FUNCT. PERF.		X			X	MODULE TEST	VEHICLE TEST		SEE NOTE A
VERIFY THE ACS MODULE & GYRO ANALOG PROCESSOR JET POLARITY & FUNCT. PERF.		X			X	MODULE TEST	VEHICLE TEST		SEE NOTE A
VERIFY THE ACS MODULE & MUS ANALOG PROCESSOR LOOP GAINS	X	X			X	COMP ANAL ONLY	HYBRID SIM	ANALYSIS + MOD OR VEHICLE LEVEL TEST	3
VERIFY THE ACS MODULE & MUS ANALOG PROCESSOR FUNCT. PERF. & POLARITY		X			X	MODULE TEST	VEHICLE TEST		SEE NOTE A
VERIFY THE ACS MODULE & SUN SENSOR OPC JET CONTROL LOOP GAINS	X	X			X	COMP. ANAL. ONLY	HYBRID SIM	ANALYSIS + MOD OR VEH. LEVEL TEST	3

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NOTE A PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS						COMPARISON MATRIX OF DESIGN APPROACHES			SELECTION
4.2.2 ACS Module (con)	ANAL	TEST	OPT	DES	A.T.	1	2	3	
VERIFY THE ACS MODULE & SUN SENSOR OBC REACTION WHEEL CONTROL LOOP GAINS	X	X			X	COMP. ANAL. ONLY	HYBRID SIM	ANALYSIS & MOD OR VEHICLE LEVEL TEST	3
VERIFY THE ACS MODULE & SUN SENSOR OBC JET POLARITY & FUNCT. PERF					X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY THE ACS MODULE & SUN SENSOR OBC REACTION WHEEL CONTROL LOOP POLARITY & FUNCT. PERF.					X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY THE ACS MODULE & GYRO OBC REACTION WHEEL LOOP GAINS	X	X			X	COMP ANAL ONLY	HYBRID SIM	ANALYSIS + MOD OR VEHICLE LEVEL TEST	3
VERIFY THE ACS MODULE & GYRO OBC REACTION WHEEL FUNCT. PERF. & POLARITY		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY THE ACS MODULE & GYRO OBC JET LOOP GAINS	X	X			X	COMP. ANAL. ONLY	HYBRID SIM	ANALYSIS + MOD OR VEHICLE LEVEL TEST	3
VERIFY THE ACS MODULE & GYRO OBC JET LOOP FUNCT. PERF. & POLARITY		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY THE ACS MODULE & MVS OBC LOOP GAIN	X	X			X	COMP. SIM ONLY	HYBRID SIM	ANALYSIS + MOD. OR VEHICLE LEVEL TEST	3

NOTE A PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.2.2 ACS MODULE (cont.)									
VERIFY THE ACS MODULE & MUS OBC OBC FUNCT. PERF. & POLARITY		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY THE ACS MODULE DSS OBC ATTITUDE DETERMINATION & GYRO UPDATE	X	X			X	COMP SIM ONLY	HYBRID SIM	ANALYSIS + MOD OR VEHICLE LEVEL	3
VERIFY THE ACS MODULE DSS OBC ATTITUDE CONTROL MODE	X	X			X	COMP SIM ONLY	HYBRID SIM	ANALYSIS + MOD OR VEHICLE LEVEL TEST	3
VERIFY THE ACS MODULE FHT OBC ATTITUDE DETERMINATION & GYRO UPDATE FUNCT.	X	X			X	ANALYSIS + MODULE TEST	ANALYSIS + VEHICLE TEST		SEE NOTE A
VERIFY THE ACS MODULE FHT OBC ATTITUDE CONTROL MODE	X	X			X	COMP SIM ONLY	HYBRID SIM	ANALYSIS + MOD OR VEHICLE LEVEL TEST	3
VERIFY ACS MODULE PERF. FOR OFF NOM ATTITUDE & RATES	X				X	COMP SIM	HYBRID SIM	" "	3
VERIFY ACS MODULE PERF FOR INPUT BUS VOLTAGE EXTREMES		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY ACS MODULE PERF FOR NOM & OFF NOMINAL ENV. CONDITIONS		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY ACS MODULE MAGNETOMETER ATTITUDE DETERMINATION & GYRO UPDATE FUNCT.	X	X			X	ANALYSIS + MODULE TEST	ANALYSIS + VEHICLE TEST		SEE NOTE A

NOTE A PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATIONS & ACCEPTANCE TEST REQMTS

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.2.2 ACS MODULE (cont)									
VERIFY THE ACS SENSOR ALIGNMENT MAINTENANCE THROUGHOUT MISSION ENVIRONMENT		X			X	MODULE LEVEL	VEHICLE LEVEL		1 & 2 QUAL TBD ACCEPTANCE
VERIFY ACS MODULE WIRING HARNESS		X			X	HAND RING OUT	AUTOMATED C/O		2 - LOW COST
4.3 COMM & D.H MODULE									
4.3.1 COMM-S BAND									
VERIFY S-BAND VSWR		X			X	VEHICLE LEVEL			SEE NOTE A
VERIFY S-BAND TRANSMITTER PWR LEVELS & FREQ.		X			X	MODULE LEVEL	VEHICLE LEVEL		
VERIFY S-BAND TRANSMITTER MOD & TRANSMISSION OF 20 KBS DATA		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY S-BAND TRANSMITTER MOD & TRANSMISSION OF 500 KBS DATA		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY ALL 3 BAND REDUNDANT LOOPS		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY S-BAND RANGING	X	X			X	MODULE LEVEL + ANALYSIS	VEHICLE LEVEL + ANALYSIS		"
VERIFY S-BAND RECEIPT & DEMOD OF S/C COMMANDS		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY S-BAND RECEIVER SIGNAL THRESHOLD LEVELS		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY S-BAND SVB SYSTEM PERF. FOR BUS INPUT VOLT. EXTREMES		X			X	MODULE LEVEL	VEHICLE LEVEL		"

NOTE A PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATION & ACCEPTANCE TEST REQMT.

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST OPT DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.3.1 COMM S-BAND (continued)							
VERIFY S-BAND SUB SYSTEM PERF FOR NOM & OFF NOM ENV. OPER CONDITIONING	X	X	X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY S-BAND DECODER INTERFACES		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY S-BAND MUX INTERFACES		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY S-BAND INTERNAL TLM FUNCTIONS		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY S-BAND INTERNAL COMMAND FUNCTIONS		X	X	MODULE LEVEL	VEHICLE LEVEL		"
4.3.2 D.H. S. BAND							
VERIFY D.H. SYS. CLOCK ABSOLUTE FREQUENCY		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY CLOCK FREQ. STABILITY	X	X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY OBC LOAD & DUMP CAPABILITY & ACCURACE		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY OBC - FORMAT GEN. I/O		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY OBC CMD DECODER I/O		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY TAPE RECORDER LOAD/DUMP CAPABILITY & BIT ERROR RATES		X	X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY COMMAND DECODER OUTPUTS TO REMOTE DECODER		X	X	MODULE LEVEL	VEHICLE LEVEL		"

NOTE A PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMT.

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.3.2 D.H. S-BAND (Continued)									
VERIFY FORMAT GENERATOR INPUT OUTPUT TO REMOTE MUX		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY REMOTE DECODER SCA INTERFACE		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY REMOTE MUX SCA INTERFACE		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY SCA CAUB. CURVES		X			X	VEHICLE LEVEL			"
VERIFY D.H. INTERNAL TLM POINTS		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY D.H. INTERNAL CMD FUNCTIONS		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY D.H. PERF. FOR BUS INPUT EXTREMES		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY D.H. PERF. FOR MISSION NOM & OFF NOM. COND.		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY D.H. FORMAT GENERATOR 500 KB/S OUTPUT FORMAT & ERROR RATE		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY D.H. FORMAT GENERATOR 20 KB/S OUTPUT DATA FORMAT AND ERROR RATES		X			X	MODULE LEVEL	VEHICLE LEVEL		"
VERIFY COMM. & D.H. MODULE WIRING HARNESSES						HAND RING OUT	AUTOMATED C/O		2 - LOW COST

NOTE A PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7. continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.3.3 OBC SOFTWARE									
VERIFY OBC OPERATIONAL SOFTWARE/ HARDWARE FUNCTIONAL INTEG. & PERFORMANCE FOR NOM AND OFF NOM INPUT CONDITIONS		X			X	SOFTWARE LAB	MODULE LEVEL	VEHICLE LEVEL	1, 2, & 3
VERIFY OBC SUBROUTINE SOFTWARE/ HARDWARE FUNCTIONAL INTEG & PERFORMANCE FOR NOM & OFF NOM INPUT CONDITIONS		X			X	SOFTWARE LAB	MODULE LEVEL	VEHICLE LEVEL	1, 2, & 3
* 4.5 OAS/OTS/RCS									
VERIFY OAS PROOF & SYSTEM LEAK INTEGRITY		X		X	X	MODULE LEVEL	VEHICLE LEVEL		1 - FACTORY 3 - LAUNCH SITE
VERIFY OAS COMMAND & THRUSTER RESPONSE		X		X	X	VEHICLE LEVEL	MODULE LEVEL		SEE NOTE A
VERIFY RCS LEAK INTEGRITY		X		X	X	VEHICLE LEVEL	MODULE LEVEL		1 - FACTORY 3 - LAUNCH SITE
VERIFY RCS COMMAND & THRUSTER RESPONSE		X		X	X	VEHICLE LEVEL	MODULE LEVEL		SEE NOTE A
VERIFY OTS COMMAND AT SRM INTERFACE		X		X	X	VEHICLE LEVEL	MODULE LEVEL		SEE NOTE A
VERIFY OTS WEIGHT ECG		X		X	X	SRM			1 - AT LAUNCH SITE
VERIFY OTS COMMAND & THRUSTER RESPONSE	X	X		X	X	SRM BY ANAL ELECT INTERFACE BY TEST			SEE NOTE A
VERIFY OAS THRUSTER THRUST LEVELS	X	X		X	X	ANALYSIS + COMP. TEST			1

NOTE A: PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS						COMPARISON MATRIX OF DESIGN APPROACHES			SELECTION
4.5 OAS/OTS/RCS (cont)	ANAL	TEST	OPT	DES	A.T.	1	2	3	
VERIFY OAS THRUSTER THRUST VECTORS	X				X	ANAL + ALIGNMENT CHECK			1
VERIFY RCS THRUSTER THRUST LEVELS	X			X	X	ANAL + COMP TEST			1
VERIFY RCS THRUSTER THRUST VECTORS	X				X	ANAL + ALIGNMENT CHECK			1
VERIFY OTS THRUST LEVEL	X			X		ANAL OF SRM BURN TIME			1
VERIFY OTS THRUST VECTORS	X				X	ANAL + ALIGNMENT			1
VERIFY OAS FLOW RATES	X	X		X	X	MODULE TEST	VEHICLE TEST	COMP TEST	1 or 3 DESIGN DEPENDENT
VERIFY RCS FLOW RATES	X	X		X	X	MODULE TEST	VEHICLE TEST	COMP TEST	1 or 3 DESIGN DEPENDENT
VERIFY OAS PERF. FOR NOM & OFF NOM MISSION ENV.		X			X	MODULE TEST	VEHICLE TEST		2 - QUAL (LRM), 1 QUAL FOLLOW ON 1 - A.T.
VERIFY RCS PERF. FOR NOM & OFF NOM MISSION ENV.		X			X	MODULE TEST	VEHICLE TEST		2 - QUAL (LRM), 1 QUAL FOLLOW ON 1 - A.T.
VERIFY OTS PERF. FOR NOM & OFF NOM MISSION ENV.	X	X			X	ANALYSIS			1
VERIFY OTS/OAS/RCS PROPELLANT USAGE VS. EXP. LIFE	X					ANALYSIS			1
4.6 INSTR. & W.B. DATA									
4.6.1 W.B. DATA									
VERIFY INSTR. A TO D THROUGHPUT CONVERSION DATA FORMAT & TIMING		X			X	MODULE LEVEL IN HOUSE	MODULE AT INSTR. MANUF.	VEHICLE LEVEL	1 & 3 MODULE TO BE PROVIDED TO INSTR. MANUF. FOR A.T.

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

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FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.6 INSTR & W B. (continued)									
4.6.1 (continued)									
VERIFY INSTR. D TO D THROUGHPUT DATA FORMAT & TIMING		X			X	MODULE LEVEL IN HOUSE	MODULE AT INSTR. MANUF.		1 & 3 MODULE TO BE PROVIDED TO INSTR. MANUFACTURER
VERIFY INSTR. D TO D, COMBINER STORAGE AND TIMING FUNCTIONS FOR DATA COMPACTION		X			X	MODULE LEVEL IN HOUSE	MODULE AT INSTR. MANUF.	VEHICLE LEVEL	1 & 3 "
VERIFY W. B. DATA COMMAND FUNCTIONS		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY W. B. DATA S-BAND TLM FUNCTIONS		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
VERIFY WBRTR DATA & STORAGE DATA DUMP & BIT ERROR		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
*VERIFY TDRS KU BAND MODULATION, XMITTER FREQ. & PWR LEVELS	X	X			X	SIMULATION OF EOS WIDEBAND SYS. TDRS	ANALYSIS OF INTERFACE WITH BOTH SIDE VERIF. INDEPENDENTLY BY TEST		TBD
*VERIFY KU BAND XMITTER ANTENNA VSWR		X				VEHICLE LEVEL			1
VERIFY KU BAND ANTENNA POINTING CONTROL		X				VEHICLE LEVEL			1
VERIFY MEDIUM BAND XMITTER MODULATION XMITTER POWER & FREQ. LEVELS		X				VEHICLE LEVEL	MODULE LEVEL		SEE NOTE A
VERIFY MEDIUM BAND XMITTER VSWR BETWEEN XMITTER & BOTH ANTENNAE		X				VEHICLE LEVEL			1

NOTE A: PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.6.1 W. B. DATA (continued)									
VERIFY BOTH W. B. ANTENNAE POINTING CONT.		X			X	VEHICLE LEVEL			1
VERIFY THE W. B. DATA HANDLING PERF. FOR MISSION NOM & OFF NOM OPER. CONDITIONS		X			X	MODULE LEVEL	VEHICLE LEVEL		2 - QUAL 1 - A.T.
VERIFY THE W. B. DATA HANDLING PERF. FOR INPUT BUS VOLT EXTREMES		X			X	MODULE LEVEL	VEHICLE LEVEL		SEE NOTE A
4.6.2 INSTR.									
VERIFY INSTR. OPTICAL PERF. AND DETECTOR OUTPUTS		X			X	INSTR. TESTS	VEHICLE TESTS		1 - PREL DEPENDS ON INSTR. SELF CALIB OR GSE CAL TECHNIQUE
VERIFY INSTR. SCAN RATES & MECH. FUNCTION		X			X	INSTR. TESTS	VEHICLE TEST		1 - "
VERIFY INSTR. SENSITIVITY & THRESHOLDS		X			X	INSTR. TESTS	VEHICLE TESTS.		1 - "
VERIFY INSTR. COMMAND FUNCT.		X			X	S/C SIMULATOR FOR INSTR. TEST	VEHICLE TEST		1 & 2
VERIFY INSTR. S-BAND TLM FUNCTIONS		X			X	INSTR. TEST	VEHICLE TEST		SEE NOTE A
VERIFY INSTR. PERF. FOR MISSION NOM & OFF NOM OPER. CONDIT.		X	1		X	INSTR. TEST	VEHICLE TEST		1 - QUAL + LRM SYS QUAL 1 - ACCEPT.
VERIFY INSTR. PERF. FOR INPUT BUS VOLT EXTREMES		X			X	INSTR. TEST	VEHICLE TEST		SEE NOTE A

NOTE A: PART OF STANDARD FUNCTIONAL TEST AT MODULAR AND VEHICLE LEVEL

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1. Low Cost Management Option

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.6.3 D.C S									
VERIFY DCS RECUR SENSITIVITY		X			X	VEHICLE TEST	MODULE TEST		SEE NOTE A
VERIFY DCS REC'R DEMOD FORMATS, TIMING		X			X	VEHICLE TEST WITH TEST GND TRANSMITTER	MODULE TEST		"
VERIFY DCS MVX ONTO DOWNLINK W.B. DATA STREAM		X			X	VEHICLE TEST	MODULE TEST		"
VERIFY DCS THROUGH-PUT SYS ERROR		X			X	VEHICLE TEST	MODULE TEST		"
VERIFY DCS THROUGH-PUT PERF. FOR MISSION NOM & OFF NOMINAL OPER. CONDT.		X	1		X	VEHICLE TEST	MODULE TEST		1 - QUAL 2 - ACCEPTANCE TEST
VERIFY DCS THROUGH-PUT PERF. FOR IN-PUT BUS VOLT EXTREMES		X			X	VEHICLE TEST	MODULE TEST		SEE NOTE A
4.7 DMS & M.O.									
4.7.1 PRIMARY G&D STA - CONTROL CENTER									
VERIFY RECEIPT, DEMODULATION, DECOMMUTATION AND PROCESSING OF S/C S-BAND STATUS DATA		X			X	VEHICLE GND SYS TEST			1 - CONTROL CENTER NETWORK S/C COMMAND AND TLM LIN DEMO
VERIFY ANTENNA TRACKING COMMANDS & PATTERNS	X	X			X	ANAL & VISUAL CHECKS			1
VERIFY PRL GND STATION DOWN LINK O/P TO CENT. PROC. STA.		X			X	WITH VEHICLE PROVIDING DATA	SIM S/C DATA		1 & 2 CONTROL CENTER NETWK + PRI GND STA - S/C CMD & TLM LINK DEMO

1 - LOW COST MGNT OPTION

NOTE A: PART OF STANDARD FUNCTIONAL TEST AT MODULE AND VEHICLE LEVEL

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7. continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.7 DMS & M.O. (cont)									
4.7.1 (continued)									
VERIFY PRL GND STA S/C TRACKING & EPHEMERIS DETERMINATION	X	X			X	ANALYSIS & SIM S/C DATA			1 -
VERIFY PRL GND STA PRN RANGING & TRACKING DATA O/P TO C C.	X	X			X	ANALYSIS + SIM S/C DATA			1 -
VERIFY OPERATOR MESSAGE DISPLAY AT PRL GND STA.		X			X	SIM S/C DATA	WITH VEHICLE PROVIDING DATA		1 & 2 CONTROL CONTER, NETWK + PRI GND STA S/C CMD & TLM LINK DEMO
VERIFY PRL GND STA CC S-BAND COMMAND & TLM INTERFACES		X			X	WITH VEHICLE PROVIDING DATA	TEST TAPE		1 & 2 "
VERIFY PRI GND STA MODULATION & TRANSMISSION OF S/C COMMANDS		X			X	VEHICLE TEST			1 & 2 "
VERIFY PRL GND STA UPLINK & DOWNLINK PROCESSING FOR EXTREMES OF S/C DATA LIMITS		X			X	VEHICLE TEST + TEST TAPE	COMPUTER SIMUL.		2 - COMPUTER SIMUL M.O. TRAINING)
VERIFY PRI GND STA EMERGENCY CONTINGENCY OPERATIONS		X			X	TEST TAPE	SIMULATIONS TRAINING WITH COMP. MODEL OF S/C FUNCTIONAL OPER	QUAL. TEST VEH. USED AS TNG TEST BED	2 "
VERIFY PRI STA CONT. CTR PERSONNEL READINESS TO SUPPORT ORB. OPS.		X			X	SIMULATIONS TNG WITH COMP. MODEL OF S/C FUNCTIONAL OPER	QUAL TEST VEH. USED AS TNG TEST BED.		1 "

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION & QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.7 DMS & M.O. (cont)									
4.7.2 CENTRAL DATA PROCESSING									
VERIFY PRI GND STA INTERFACES		X	I		X	TEST TAPES	VEHICLE TEST	MODULE + INSTR, TEST AT PRI GND STA SITE	1, 2, & 3 SEE NOTE B
VERIFY WIDEBAND DATA QUICK LOOK DISPLAY		X	I		X	TEST TAPES	VEHICLE TEST	"	1 & 3 SEE NOTE B
VERIFY DATA STORAGE FUNCTIONAL STORAGE AND REPLAY CAPABILITY		X	I		X	TEST TAPES	VEHICLE TEST	"	1 & 3 SEE NOTE B
VERIFY CONVERSION OF WIDEBAND DATA TO DATA PRODUCTS		X	I		X	TEST TAPES	VEHICLE TEST	"	1 & 3 SEE NOTE B
VERIFY ALL PROCESS ALGORITHMS	X	X			X	TEST TAPES + COMP SIM.		———	1
VERIFY GEOMETRIC CORRECTION SOFTWARE	X	X			X	TEST TAPES + COMP SIM.		———	1
VERIFY OPERATORS CONSOLE CONTROL & DISPLAY FUNCTIONS		X	I		X	TEST TAPES	VEHICLE TEST	"	1 & 3 SEE NOTE B
VERIFY CDP-CC INTERFACES		X	I		X	TEST TAPES	VEHICLE TEST	"	1 & 3 SEE NOTE B
4.7.3 CONTROL CENTER									
VERIFY S-BAND DOWNLINK PROCESSING DISPLAY AND ANALYSIS SOFTWARE & HARDWARE FUNCTIONAL PERF.	X	X			X	TEST TAPE + COMP. SIM TNG.	VEHICLE TEST		1 & 2

I. LOW COST MANAGEMENT OPTION

NOTE B: THE FLIGHT IMP MODULE, INSTRUMENTS, AND ANTENNA WILL BE TESTED AT GSFC PRIOR TO INTEG WITH FLIGHT S/C

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7. continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
4.7.3 Control Center (continued)									
VERIFY S-BAND COMMAND GENERATION SOFTWARE & HARDWARE OUTPUT FORMATS	X				X	VEHICLE TEST	SIMULATION COMPUTER		1 & 2
VERIFY S/C -CC EMERGENCY OPERATIONS PROCED.		X			X	TEST TAPE	SIMULATIONS TNG WITH COMP SIMUL. OF S/C FUNCT.	QUAL. TEST VEHICLE USED AS TNG TEST BED	1 & 2
VERIFY CC-HARDWARE SOFTWARE PERF. FOR NOM S/V DATA AND COMMAND PROCESSING		X			X	TEST TAPE	VEHICLE TEST		1 & 2
VERIFY CC-REMOTE SITE S/V UP AND DOWN LINK DATA TRANSMISSION		X			X	TEST TAPE	VEHICLE NETWORK TEST		1 & 2
VERIFY CC-SV-PRI GND STA UP & DOWNLINK DATA HANDLING		X			X	VEHICLE TEST	TEST TAPE		1 & 2
VERIFY CC-CDP INTERFACES		X			X	VEHICLE TEST	TEST TAPE		1 & 2
4.7.4 LCGS									
VERIFY LCGS ACQUISITION OF S/C WIDEBAND RF & DEMOD.	X	X			X	S/C SIMULATOR (RF HARDWARE) & DATA TAPE	SET UP LCGS AT VEH. TEST SITE-SEND DATA TAPE TO REMOTE SITE	ANALYSIS	2 - PART OF T&I STATION FOR VEHICLE C/O
VERIFY LCGS DECODE PROCESSING & DATA PRODUCTS O/P		X			X	TEST TAPE	SET UP LCGS AT VEH. TEST SITE-SEND DATA TAPE TO REMOTE SITES		2 "

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TRADE STUDY REPORT SUMMARY

NOMENCLATURE: DESIGN VERIFICATION AND QUALIFICATION & ACCEPTANCE TEST REQMTS.

WBS NUMBER: 1.7.7 continued

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	ANAL	TEST	OPT	DES	A.T.	COMPARISON MATRIX OF DESIGN APPROACHES 1	2	3	SELECTION
5.0 SHUTTLE INTERFACES									
VERIFY S/C - SHUTTLE STRUCTURAL LAUNCH INTERFACE		X				INTERFACE SIMULATION	QUAL - DEMO S/C -GND TEST		2 - PREL RECOMMENDATION
VERIFY S/C-SHUTTLE ELECTRICAL/ ELECTRONIC INTERFACES		X				INTERFACE SIMULATION	QUAL - DEMO S/C -GND TEST		2 - " "
VERIFY SHUTTLE - S/C DEPLOYMENT FUNCT.					X	SIMULATION GND TEST + ORB-TEST WITH DEMO-S/C	S/C SIMULATED MASS & STRUC. IF DEPLOY, ONLY DES FOR GND & ORB TEST	SCALE MODEL WITH MECH. FUNCTIONS TO MATE WITH SHUTTLE SIMULATOR	1 - " "
VERIFY SHUTTLE-S/C RETRIEVAL FUNCT.		X				SIMULATION GND TEST & ORB TEST WITH DEMO S/C			1 - " "
VERIFY SHUTTLE-S/C ON ORBIT RESUPPLY MANIPULATOR FUNCT. INTERFACES		X				SIMULATION GND TEST + ORB TEST USING QUAL OR DEMO S/C	SEPARATE GND SIMULATION TEST ARTICLE + ORB TEST WITH DEMO S/C		1 - " "
VERIFY SHUTTLE-S/C ON ORBIT RESUPPLY FUNCTIONAL PERF.		X				SIMULATION GND TEST + ORB TEST	SEPARATE SIMULATION TEST ARTICLE + ORB TEST WITH DEMO S/C	SCALE MODEL WITH MECH. FUNCTION TO MATE WITH SHUTTLE SIMULATOR	1 - " "

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6.18.3 TRADES STUDIES

6.18.3.1 INTRODUCTION

The basic EOS program test requirements and alternate implementation approaches based on the modular design, and baseline subsystems are identified in section 2 of the Management Approach Volume. These test requirements were examined and for each test category; development, qualification and flight observatory integration and test, and an assessment of the cost impact of the test alternatives was made to identify the major test program cost drivers. In addition, the impact on test cost and test ability of design alternatives was considered. Table 6.18-1 summarizes the results of this assessment, and the studies performed of the test requirements identified as the more significant cost drivers.

6.18.3.2 INTEGRATION AND TEST

The major test cost consideration in Integration and Test (I&T) is the flexibility of combining and/or segregating tests because of the modular observatory design.

Study of the I&T program options provided by the modular design indicated that significant cost savings is achieved without technical or cost risks by performing all environmental acceptance tests at the S/S module, and instrument level of assembly. This approach was studied in detail; the results are summarized in Table 6.18-2 found in subsection 6.18.3.2.2 of this report.

The flight hardware non environmental Integration and Test efforts associated with the modular design were assessed, and it is concluded that the modular approach provides a greater flexibility in subsystem and observatory buildup, then the conventional integrated spacecraft design. Some of the advantages of a modular design vs. an integrated design from the I&T viewpoint in addition to the environmental test cost savings are;

- Parallel integration of subsystems
- More effective subsystem integration tests because of remote multiplexers and decoders permitting full functional checkout of each subsystem on an individual basis
- Option for substitution of a complete subsystem while a subsystem anomolies are investigated off line from the flight vehicle flow.

Table 6.18-1 EOS Spacecraft Test Philosophy Trade Summary (Sheet 1 of 2)

TEST FUNCTION	TEST/REQUIREMENT	DESIGN TEST IMPACTS	COST INFLUENCE/STATUS
THERMAL DESIGN DEVEL TEST	MODULE LEVEL THERMAL MODULE TESTS/VERIFY THERMAL ANALYSIS	70° SPACECRAFT WOULD SHORTEN TEST TIME FROM 8 DAYS TO 4/MODULE	TEST COST SAVINGS UP TO 80K-SAVINGS NOT INCLUDED IN BASELINE COSTING. DECISION PENDING TOTAL S/C THERMAL DESIGN COST TRADE
VIBRATION & ACOUSTIC DEVEL TESTS	MODULE LEVEL ACOUSTIC AND MECH VIB TEST-TO ESTABLISH ACCEPTANCE TEST APPROACH WHICH WILL EFFECTIVELY WORKMANSHIP SCREEN COMPONENTS AT MODULE LEVEL	ADDED LEVEL TEST TO THE PROGRAM	TEST COST INCLUDED SINCE "MODULE TEST ONLY" IS BASELINED. (COST INCLUDED IN 75K SHOWN FOR QUAL)
AVIONICS DEVEL TESTS	SOFTWARE DEVEL TEST	QUANTITY OF SOFTWARE DEVEL TESTS DEPENDENT ON LEVEL OF USE TO PERFORM OBS FUNCTIONS FOR BOTH ORBITAL AND GND SOFTWARE	SOFTWARE DEVEL TEST PROGRAM NOT SIGNIFICANTLY COST IMPACTED BY ADDED SOFTWARE FUNCTIONS
	HARDWARE DEVEL TEST	OTHER THAN ANTENNA PATTERNS & INSTR DEV TEST NO SUBSYSTEM OR VEHICLE LEVEL AVIONICS DEVEL TESTS HAVE BEEN IDENTIFIED. COMPONENT SELECTION TRADES WILL CONSIDER COMPONENT LEVEL DEVEL TEST FIRST TIME. INTEG IS IN QUAL MODULE	S/C & MODULE LEVEL DEVEL TEST PROGRAM COSTS NOT AFFECTED BY DESIGN APPROACH. COMPONENT LEVEL DEVEL TEST COSTS TREATED IN SELECTION STUDIES.
SOLAR ARRAY & DEPLOYMENT MECH. DEV. TESTS	DEPLOYMENT & DRIVE DEVEL TEST - TO VERIFY DEPLOYMENT MECH.	RIGID ARRAY DEVEL TEST REQD. THEREFORE, REQUIRING A FIXTURE OR VEHICLE TIME FOR TEST. FLEXIBLE ARRAY DEVEL WOULD NOT REQUIRE VEHICLE FOR DEVEL	RIGID ARRAY TEST COST INCLUDED APPROX. 40 K IN TEST COST ASSUME USING QUAL. HARDWARE.
SHUTTLE INTERFACES DEVEL TESTS	VERIFY RESUPPLY, LAUNCH & RETRIEVAL INTERFACES	DEPENDENT ON SHUTTLE UTILIZ STUDIES. POTENTIALLY SOME SMALL OFFLINE DEVEL TESTS FOR LATCHES. USE QUAL SPACECRAFT FOR FLT DEMO & GROUND INTERFACE TEST	SEE SHUTTLE UTILIZATION TRADE STUDY
STRUCTURAL MODAL SURVEY	CANTELEVER & FREE MODAL SURVEY TO VERIFY THE BASIC STRUCTURE FREQUENCIES FOR BOTH LAUNCH VEHICLE AND S/C CONTROL SYSTEM ANALYSIS	TEST COST NOT IMPACTED BY DESIGN ALTERNATIVES HOWEVER, FOR COMMON S/C APPROACH A MODAL SURVEY FOR FOLLOW-ON CONFIG REQUIRED	FOLLOW ON CONFIG. MODAL SURVEY ADDS ABOUT 20K TO TEST PROG COST
STRUCTURAL QUALIFICATION	STATIC-LOAD	MODULAR DESIGN REQUIRES STATIC LOAD TEST QUAL OF MODULES IN ADDITION TO THE PRIMARY STRUCTURE	ADDS ABOUT 75K TO QUAL PROGRAM MODULE VIBRATION & ACOUSTIC TESTS
	ACOUSTIC, SINE & SHOCK TEST	ADDITION OF MODULE STRUCTURE AND STRUCTURAL INTERFACES IN ADDITION TO PRIMARY STRUCTURE & ALSO ADDED TEST TO QUAL FOLLOW ON CONFIG	FOLLOW-ON CONFIG ADDS ABOUT 30K TO STRUCT QUAL TEST COSTS
SEPARATION SYSTEM QUALIFICATION	QUANTITATIVE SEPARATION TEST WITH RATES AND TIP OF ANGLES MEASURED	MAY BE SOMEWHAT MORE DIFFICULT WITH EXTRACTION REQUIRED FOR TRANSITION RING MOUNT	NO SIGNIFICANT COST IMPACT

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Table 6.18-1 EOS Spacecraft Test Philosophy Trade Summary (Sheet 2 of 2)

TEST FUNCTION	TEST/REQUIREMENT	DESIGN TEST IMPACTS	COST INFLUENCE/ STATUS
SOLAR ARRAY DEPLOYMENT MECHANISM & ARRAY QUALIFICATION	QUAL DEPLOYMENT MECHANISM	SAME COMMENT AS FOR DEVEL TEST	ADDED COST OF FOLDUP ARRAY FIXTURE INCURRED AGAINST DEVEL TEST
SYSTEM THERMAL VACUUM, SINE, ACOUSTIC AND SHOCK QUALIFICATION	SYS LEVEL QUAL OBS TEST PROGRAM	MODULAR DESIGN PERMITS QUALIFICATION TESTING TO BE ACCOMPLISHED AT THE MODULE LEVEL OR SYSTEM LEVEL	HIGH RISK OF ONLY QUAL AT MODULE LEVEL NOT CONSIDERED ACCEPTABLE
COMPONENT QUALIFICATION	COMPONENT QUALIFICATION TEST PROGRAM	COMPONENT QUAL COULD BE CONDUCTED AT MODULE LEVEL	SAME AS ABOVE
FLIGHT OBSERVATORY ENVIRONMENTAL ACCEPTANCE TESTS	ACOUSTIC AND THERMAL VACUUM TEST TO VERIFY WORKMANSHIP	MODULAR DESIGN PERMITS OPTION OF TESTING AT SUBSYSTEM LEVEL WITH ONLY A FINAL WORKMANSHIP ACOUSTIC AT THE VEHICLE LEVEL	SEE COMPONENT ACCEP TEST BELOW
COMPONENT ENVIRONMENT A ACCEPTANCE TEST	THERMAL VACUUM & VIBRATION TEST TO VERIFY COMPONENT WORKMANSHIP	MODULAR DESIGN PERMITS OPTION OF PERFORMING ON A SUBSYSTEM BASIS IN THE MODULES	TOTAL PER SPACECRAFT COST SAVINGS FOR PERFORMING ENVIRONMENTAL ACCEPTANCE TEST AT THE "MODULE LEVEL ONLY" IS APPROX. 500K IF BOTH VEHICLE AND COMPONENT ENVIRONMENTAL ACCEPTANCE TESTS ARE ELIMINATED. APPROACH USED IN BASELINE

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6.18.3.2.1 INPUTS TO DESIGN STUDIES

- Thermal design approach

A design consideration which influences I&T cost is the thermal design study of a 70°F thermal control system. It is expected, based on the thermal mass of 400 to 500 lb module, that a hot cold soak T/V test requires about 5 1/2 to 6 days to run. If the thermal design was established to maintain the module components at 70°F the hot cold transition times for acceptance test of the module would be reduced to the time required to move $\pm 10^{\circ}$ from ambient and functionally test the module in vacuum. This is estimated to reduce the T/V test time/module from 6 to 4 days. Based on the T/V acceptance test costs per day derived under ANALYSIS in section 6.18.3.2.2 of this report, the total cost savings per spacecraft set of modules, 3 S/S + IMP, would be 4/Kday/module or (4 days) (4K) = 16K/module. This number has been input to the thermal design 70° spacecraft trade study for consideration in the higher level spacecraft thermal design/ cost trade.

- One vs Two Spacecraft System

One of the EOS system level trade studies is the cost of a One versus Two Spacecraft System to achieve a given mission. In support of this study the cost influence of the spacing between launches was examined. Figure 6.18-4 shows the increase in the total cost of mission operations GSE, test and supporting manpower, as a function of time between launches, for simultaneous launch through a two year interval between launches. As would be expected the further apart the launches the more expensive the Test and Mission Operations costs.

- Over design to eliminate test

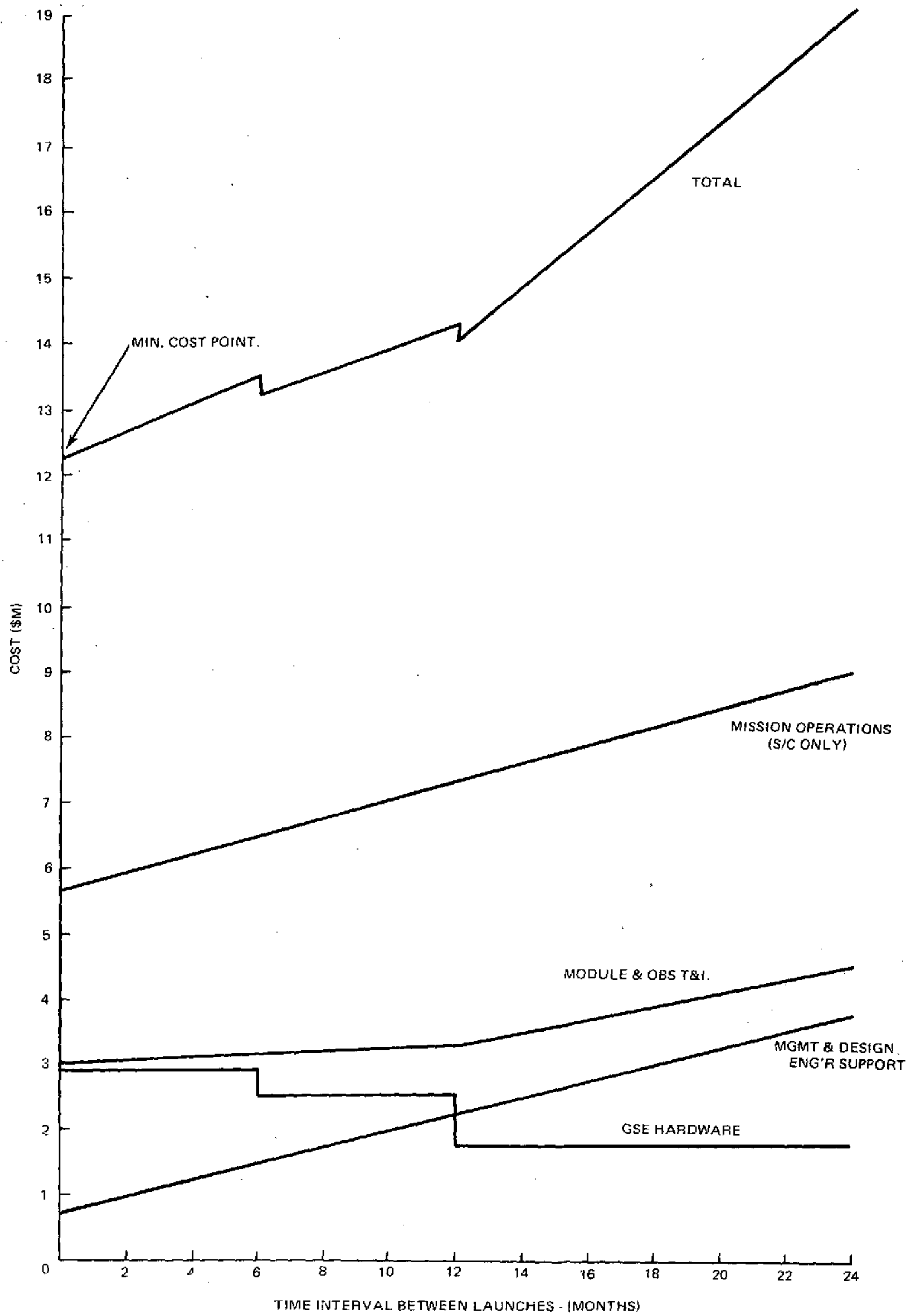
The cost savings which could be achieved by overdesign of the spacecraft and module structure to eliminate testing was studied. It was concluded that the only test which could be eliminated by overdesign of the spacecraft is the structural static load qualification test of the spacecraft and module structure; however, it is not a cost effective approach from a weight penalty viewpoint. Dynamic structural qualification tests are critical to definition of component design levels and Launch Vehicle interfaces, and analysis is not as simple as for static loads. Therefore the dynamic structural qualification tests cannot be eliminated.

The test cost savings by structural overdesign is a one time program costs savings of approximately 40K. The critical loads for the spacecraft and module structure are the dynamic loads and not static loads therefore a simple determination of the weight increase required to provide a 200% safety factor is not practical. However, when the tight weight margin for EOS A & A' for the Delta 2910 launch is considered, coupled with the fact that the initial weight penalty would be carried forever by the basic spacecraft, it was concluded that the potential long term weight handicap of overdesign may cost much more than any initial test cost savings. Therefore over design to eliminate test is not recommended for the EOS spacecraft.

6.18.3.2.2 LEVEL OF TEST FOR ENVIRONMENTAL ACCEPTANCE TEST

The objective of this study is to identify the most cost effective level of spacecraft system integration to conduct vibration and Thermal Vacuum acceptance tests.

SUMMARY AND CONCLUSIONS - Modular subsystem design permits the option of conducting combined subsystem component environmental tests at the module level. Inherent in this approach is the obvious cost savings; however, the cost savings must be traded off against the total program cost and schedule risk of a single in line test to uncover workmanship defects on 10-20 components.



3-206 **Fig. 6.18-4 Test and Mission Operation Costs Vs Time Between Launches for a Two-Spacecraft System**

The environmental acceptance test alternatives which were studied for this study are:

- Typical Integrated Spacecraft Component and observatory tests
- Alt 1. Component and Module Level Tests
- Alt 2. Module Level only

Table 6.18-2 shows the approach to studying these alternatives and summarized the costs and risk involved with each. The conclusions that are made from this study are as follows:

- Conducting all component and observatory acceptance tests at the module level represents a 50% cost savings in environmental acceptance test costs at virtually no cost risk.
- The same type work arounds used in a typical integrated spacecraft test program can be used to maintain program schedules in the event of component failures during environmental test.
- Module tests should provide the same level of confidence in component performance as component tests however components can be subjected to a more severe vibration workmanship screen at the component level.
- Subcontractors meeting delivery dates is more critical for a module level environmental test approach, therefore, the state of the design and program milestones must be considered.

In conclusion the module level environmental acceptance test is a realistic low cost alternative to business as usual and is recommended for the EOS design.

GROUND RULES AND ASSUMPTIONS -

- The basic vehicle consists of 3 subsystem modules, an Instrument Mission Peculiar Module and an OAS/RCS Module, Single Instrument and a foldup solar array.
- No redundancy was included in the subsystem modules. (Note if redundancy was considered, cost savings of modular vs component tests is further realized because of the added number of components).
- Only recurring acceptance test costs were used.
- Average technician/engineering costs of 35K/man year were used.
- An acoustic level workmanship test will be performed at the observatory level prior to delivery of the spacecraft to the launch site.
- Module level Thermal Vacuum tests consist of a 2 day thermal cycle test followed by a 24 hour hot and 24 hour cold soak at vacuum with the test profile shown in Fig. 6.18-5.

Table 6.18-2 Alternative Level of Acceptance Test Study

STUDY APPROACH	IDENTIFIED ALTERNATIVES	ESTABLISHED COST OF EACH ALTERNATIVE	ESTABLISHED COST OF TEST FAILURE	ASSESSED PROG COST RISK	ASSESSED TECH RISK	ASSESSED SCHEDULE RISK	ASSESSED SUBCONTRACTOR COST IMPACT	RECOMMENDED APPROACH
	COMPONENT & MODULE LEVEL ENVIRONMENTAL ACCEPTANCE TESTS	995K	9.3K/COMP FAILURE 60.8K/SYS LEVEL FAILURE	EACH FAILURE ADDS COST/ FAILURE TO PROG COST	LOW-MEDIUM	LOW	NONE	NO
STUDY OUTPUTS	COMPONENT & SYSTEM LEVEL ENVIRONMENTAL ACCEPTANCE	993.4K	9.3K/COMP FAILURE 25.3K/ MODULE FAILURE	EACH FAILURE ADDS COST PER FAILURE TO PROG COST	LOW	LOW	NONE	NO
	MODULE LEVEL ONLY ENVIRONMENTAL ACCEPTANCE TESTS	501K	25.3K/MODULE FAILURE	EACH FAILURE ADDS COST PER FAILURE TO PROG COST HOWEVER IT WOULD TAKE 19.4 MODULE FAILURES FOR THIS APPROACH TO REACH MIN COST OF OTHER	MEDIUM-COMP MAY SEE LESS SEVERE WORKMANSHIP SCREEN DURING MOD VIB QUAL IDENTIFY IF A REAL PROB.	LOW-ASSUMING QUAL OR FOLLOW ON PRODUCT COMPONENTS AVAIL FOR SUBSTITUTE	ABOUT ½ M/M PER SUBCONTRACTOR TO ASSURE HIS COMPONENT TESTED PROPERLY	YES - POTENTIAL 50% COST SAVING AT VIRTUALLY NO TECH OR COST RISE

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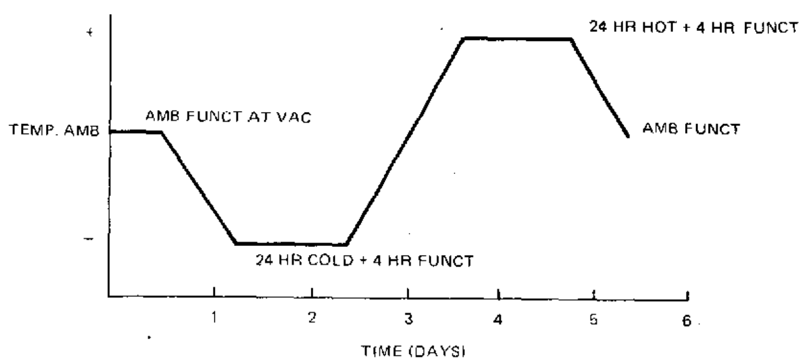

3-209 **Fig. 6.18-5 Module T/V Test Profile**

- Module vibration consists of either a 3 axis mechanical vibration or an acoustic test.

ANALYSIS

Cost Derivation -

- Component level tests

Table 6.18-3 shows the recurring environmental acceptance test costs for the basic subsystem components. Subcontractor costs shown include both acceptance environmental test cost and pre, during and post environmental test functional C/O. These costs were established from a combination of subcontractor quotes and estimates based on the similarity of components to those for which we obtained subcontractor quotes. In addition to the subcontractor costs, prime contractor manpower was added to each of the component costs for test plan, test report, procedure review, approvals, and test witnessing by engineering and/or Q.C. personnel. Complexity factors and experience where more than 1 unit/spacecraft is required is factored into the prime contractors manpower costs.

A summary of the component environmental acceptance test costs used for this study is shown below:

EPS	105.4K	OAS/RCS	96.5K
ACS	188.3K	S/A	45 K
C&DH	155.2K	S/A Drive	4.5 K
IMP	62 K		
		TOTAL	661.9 K

- Module Level tests

Figure 6.18-6 and -7 show the derivation of test costs for a typical module level vibration and thermal vacuum test. OAS/RCS and IMP module level test costs were factored based on their relative complexity to the S/S modules. The environmental acceptance test costs per module based on figures 1.1.3 and 4, used for this study are as follows:

EPS	68K	OAS/RCS	35K
ACS	68K	IMP	40K
C&DH	68K	INSTR.	90K

- Observatory level tests

Figures 6.18-8 and 6.18-9 show the recurring acoustic and thermal test costs for observatory level environmental acceptance tests. It should be noted that for the observatory transition times for ambient to cold to hot dictate that the Thermal Vac. tests take 8 days. The observatory level environmental test costs are summarized follows:

T/V	=	8 Man years	X 35K/	=	280K
		2K/day x 8 days		=	16
VIB	=	2.1 man years	X 35K	=	71.5K
		X 35K			366.5K

Table 6.18-3 S/S Module Component Level Acceptance Test Cost (Recurring)

MODULE/EQUIPMENT	QTY PER S/C	SUBCONTR TEST COST	TEST REPORT ($K)	PRIME CONT'R MANPOWER ($K)	TOTAL ($K)	FUNCT ONLY ($K)
EPS MODULE						
BATTERY	1	8	1	8	17.0	1.2
BUS PROTECT ASSY	1	5	.8	5	10.8	1
GND CHG DIODE ASSY	1	5	.8	4	9.8	1
S/C INTERFACE ASSY	1	5	.8	4	9.8	1
SIGNAL COND ASSY	1	6	1	6	13.0	1.2
DISTRIBUTION BOX	1	6	1	6	13.0	1.2
BATTERY CHARGER	1	6	1	8	15.0	1.2
REMOTE DECODER	2	5	1.5	2	8.5	1
DUAL REMOTE MUX	2	5	1.5	2	8.5	1
TOTAL EPS					105.4	9.8
ACS MODULE						
COARSE SUN SENSOR	2	3	.8	2	5.8	.5
DIG. SUN SENSOR	1	6	1	6	13.0	1.2
RATE GYRO ASSY	1	5	1	8	14.0	1
FIXED HEAD TKR	2	20	2	8	30.0	3
MAGNETOMETER SYS	1	8	1	8	17.0	1.5
ANALOG PROCESSOR	1	6	1	7	14.0	1.2
REACTION WHEEL	3	39	2.5	10	51.5	6
JET DRIVER	1	5	.8	5	10.8	1
REMOTE DECODER	2	5	1.5	2	8.5	1
REMOTE MUX	2	5	1.5	2	8.5	1
SIG COND ASSY	2	8	1.2	6	15.2	1.5
TOTAL ACS					188.3	18.9
COMM & D.H. MODULE						
TLM/CMD ANT	2	4	1	3	8	3
HYBRID JUNCT.	1	1.5	.6	3	5.1	.2
RF SWS	1	1.5	.6	3	5.1	.2
TRANSPONDER	1	10	1.2	8	19.2	2
DIPLEXER	2	3	1	3	7	.8
COMPUTER	1	10	2	12	24	2
TAPE RECORDER	1	8	1.5	7	16.5	2
CLOCK	2	4	.8	3	7.8	1
CONTROLLER FORMATER	2	8	1	8	17	1.8
REMOTE DECODER	2	5	1.5	2	8.5	1
DUAL REMOTE MUX	1	3	1	4	8	.8
SIGNAL COND	1	5	.8	6	11.8	1
COMMAND DECODER	2	8	1.2	8	17.2	1.5
TOTAL C & DH					155.2	17.3
OAS/RCS						
FILTER	2	1	.5	2	3.5	–
SOLENOID VALUE	3	7.5		3	10.5	1
N2H4 FILL DISCONNECT	1	–				
GN FILL DISCONNECT	2	–				
TANK	2	–				
THRUSTERS						
–0.1#	8	20	1.5	6	27.5	8
–1.0#	8	20	1.5	6	27.5	8
–5.0#	4	10	1.5	6	27.5	
TOTAL OAS/RCS					96.5	17
IMP. (ASSUME 40% C & DH)					62	8
S/A & DRIVE					55	–
INSTR.		90	–	–	–	

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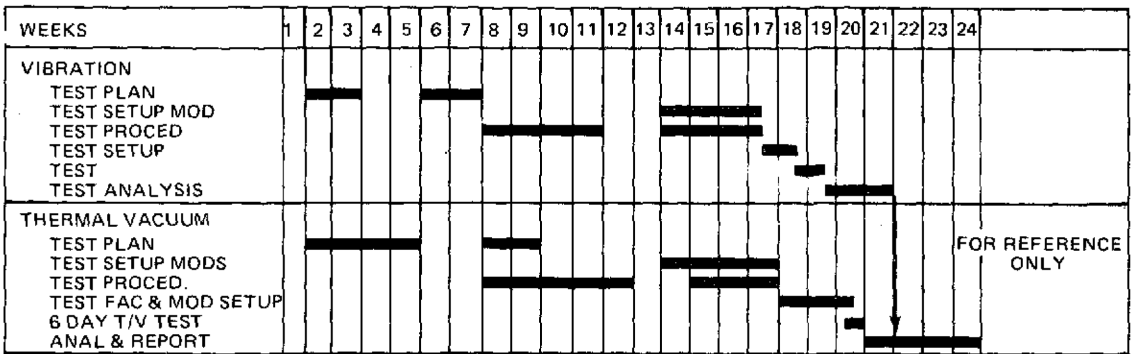

VIBRATION	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	
LOADS & DYN		1								1							½	½	½	1					
ENV. TEST & OPS		1	1						1		1			1		1	1	1	½	1	1				
INSTR.																1		1	½						
SUBSYS. ENG'R		1							1		1							½	¼	2					
GSE ENG'R											1				1			½	¼						
SOFTWARE																			¼						
DOCUMENTATION				1	1				1							1	1								
Q.A.											1							½	½						
SYS & STRUCTURAL			½								1							½							
ELECT TECHS															1			1	½						
MECH TECHS																		1	½						
ENV. TEST TECHS																	1	1	1						
INSTR TECHS																	½	1	½						
TOTAL		3	1½	1	1				3	1	5			1	2	3	4	8½	5¼	4	1				44¼
REC									1		1				1	2	3	7½		4	1				25¾
NON REC.		3	1½	1	1				2	1	4			1	1	1	1	1							18½
															1	2	2	3							8

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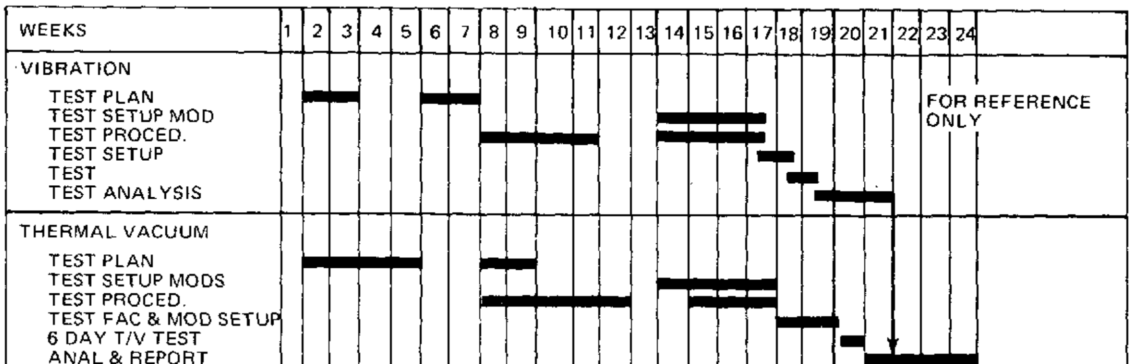
Fig. 6.18-6 Module Level Vibration Test Cost (Recurring)



THERMAL VAC	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	
THERMAL		1	1	1				1	1	1	1					1			1	1	2	1	1		
LOADS & DYN																									
ENV. TEST & OPS		1		1				1	1		1	1		1	1		1	1	1	3	½		1	1	
INSTR.										½				1		½		1	1	1	½				
SUBSYS ENG'R		1	1	1				1	1	1	1			1	1	1			1	1	½		1	1	
GSE ENG'R				1						1					1				1	1					
SOFTWARE											1					1	1		½	½					
DOCUMENTATION			1	1						1	1	1					1						1	1	
Q.A.																1		1	1	1					
SYS & STRUCTURAL			1	1					1		1				1					½		1	1		
ELECT TECHS																		1	1	3					
MECH TECHS																			1						
ENV. TEST TECHS																	1	2	2	3	1				
INSTR TECHS.																	1	1	2	3					
TOTAL		3	4	6				3	4	4½	6	2		3	4	4½	5	7	12½	18	4½	2	5	3	101
REC								1		1		1		1		1	3	6	12½	18	4½	2	5	3	59
NON. REC.		3	4	6				2	4	3½	6	1		2	4	3½	2	1							42
RECURR/3														1		½	2	4	3		1				11½

REMARKS: 1 - T/V CHAMBER EXPENDABLES FOR GAC - 7FT X 7 FTCHAMBER = $1K/DAY

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Fig. 6.18-7 Module Level Thermal Vacuum Test Costs (Recurring)

MONTHS	1	2	3	4	5	
TEST PLANNING & PROCED						
LDS & DYN ENGR	¼ M/M	½ M/M	½ M/M	½ M/M		
FACILITY ENGR (VIB TEST ENGR)	½ M/M	¾ M/M	1½ M/M			
TEST ENGR	½ M/M	2 M/M	3 M/M	2 M/M		
TEST & TEST PREPS						
TEST ENGR				3½ M/M		
TEST TECH				2 M/M		
FACILITY ENGR				1¼ M/M		
FACILITY TECHS & MOVE				4 M/M		
POST TEST						
ANALYSIS				1½ M/M	3 M/M	
TOTAL MAN MONTHS	1¼	3¼	5	14¾	3	27¾ M/M

3-212 **Fig. 6.18-8 Observatory Level Acoustic Test Recurring Cost**

MONTHS	1	2	3	4	5	6	
• TEST PREPARATION – #4 S/C (OAO-C)							
o THERMAL	4 M/W	12 M/W	8 M/W	8 M/W			
o POWER PROFILE	2 M/W	2 M/W	4 M/W	2 M/W			
o TEST PLAN PREP'S	12 M/W						
o INSTR-S/C (DESIGN & INSTALL)	6 M/W	5 M/W	4 M/W	4 M/W			
o TEST PROCEDURE PREP & C/O	8 M/W	12 M/W	16 M/W	16 M/W			
o FACILITY PREP.		2 M/W	4 M/W	12 M/W			
o INSTR-FACILITY	2 M/W	2 M/W	6 M/W	8 M/W			
o S/C PREP. (HTR SKIN & STE)	2 M/W	2 M/W	4 M/W	32 M/W			
• S/C & GSE INSTALL IN FACILITY & CO							
o S/C – INCLD'G – DRY RUN					24 M/W		
o GSE MOVE-VALIDATE					6 M/W		
• TEST RUNNING (24 HR) (# IN $\frac{\text{EQ MEN}}{\text{DAY}}$)					8 DAYS		
o TEST MGMT & ENG'R – S/C & GSE					30/DAY		
o TECHNICIANS & DATA AIDS					18/DAY		
o FACILITY & DATA ACQ ENG'R					9/DAY		
o FACILITY & DATA ACQ TECHS					9/DAY		
o Q.C/Q.E					6/DAY		
• POST TEST							
o S/C MOVE & DECONFIGURE						8 M/W	
o FACILITY TEAR DOWN						12 M/W	
o ANALYSIS & REPORTS						32 M/W	
TOTAL MAN MONTHS	6	9.5	11.5	20.5	35.5	13	=96M/M

REMARKS:
1-GAC 19 FT x 26 FT T/V Chamber expendable costs = $2 K/Day

3-213 **Fig. 6.18-9 Recurring Observatory Level Acceptance Thermal Vacuum Costs**

Cost and Cost Risk for Each Alternative vs Test Failures - The cost of a failure discovered during component, module and observatory level of test is dependent on the availability of spares, the nature of the failure and where in the test sequence the failure occurred. On the component level a failure in Thermal Vacuum, in the worst case, would require a repair and complete retest with a minimal change of impacting the program schedule. A component failure in module Thermal Vacuum would be more costly in terms of retest, and schedule; however the failed component could be tested on the component level and replaced without an entire module retest. A component failure on the observatory level would have a greater schedule impact than either component level or module level tests. However, the retest could be conducted at the component level. The likelihood of an observatory test failure at the component level is less because in this approach components would be tested at the component level prior to vehicle tests.

In order to evaluate the cost risk of each test approach we must first establish the cost of a failure at each level of test and then determine the cost impact of one, two and three or more failures for each alternative to establish the overall cost risk of each approach. The ground rules for establishing the cost of failures are as follows:

- All failures are assumed to be simple component workmanship failures which do not wipe out other components.
- All failures are assumed to be found in Thermal Vacuum tests.
- A VIB and T/V retest of the repaired component for each failure will always be conducted at the component level only.
- A failure during component test will terminate the test
- A failure during a module or observatory test will extend the test 4 days to permit substitution of the qual or work-around unit. Observatory level workmanship acoustic check will verify component mounting integrity.
- Component level test failure costs

$$\text{Avg. Comp. Test Cost} = \frac{\text{Total Cost of S/S Comp Tests}}{\text{\# Comp.}}$$

$$\text{Avg. Comp. Test Cost} = \frac{448.9\text{K}}{48} = 9.3\text{K}$$

Cost to program of avg. test cost failure = 9.3K

- Module level test failure costs

Cost to program of a Module Test failure	= 4 days T/V testing and Comp retest + 1K day expendable cost
Avg. for 4 days of vehicle T/V testing	=72 men x 4 day =14.5 M/M
	=14.5 M/M x $\frac{3k}{MM}$ = 43.5K
Cost to program of an observatory test failure	=43.5K + 9.3 K + 8K = 60.8K

- Vehicle Level Test Failure Cost

Cost to program of an observatory test failure	= 4 days observatory T/V testing + comp. retest cost + T/V expendables
Avg. for 4 days of vehicle T/V testing	= 72 men x 4 day = 14.5 M/M
Cost to program of an observatory test failure	= 43.5K + 9.3K + 8K = 60.8K

- Cost Risk

The cost to the program of each type of test failure based on primarily retest costs are as follows:

Component level test failure + 9.3K/failure

Module level test failure = 25.3K/failure

Observatory level test failure= 60.8K/failure

For the purpose of this study total program schedule slip and T&I personnel not directly involved in testing are not costed in the above numbers.

The probability of failure of the component level test should be the same as for the module only test approach and the probability of component failure at the observatory level should be less because in this approach all components were tested at the component level. However, the likelihood of a test induced failure is greater at the observatory level. Therefore, in evaluating the cost risk of each approach the probability of failure could be considered approximately equal.

Since testing at the module level only is significantly less costly than the other two alternatives, let us assume that there are 0 failures for the other two alternates. This establishes the worst case for the module test only approach, in terms of the number of allowable failures, before the cost of this alternative reaches the minimum cost of the other two approaches.

Assuming that there are 0 failures:

Typical integrated spacecraft Environmental Acceptance Test cost with component and observatory level tests = 993K

Alt 1. Implements Environmental Acceptance Tests at the Component and Module level = 995K

Alt 2. Module only environmental Acceptance test = 501K

Therefore the cost savings, comparing all failure free conditions is:

993K - 501K = 492 K

Number of failures before cost of Alt 2 reaches typical integ. or Alt 1	$\frac{\text{(Cost Savings)}}{\text{(Cost per Module) failure}}$	$\frac{492K}{25.3K} = 19.4$ Failures

Therefore, we can tolerate 20 component failures during a module level only test approach, before its cost is as high as the minimum cost of a typical integrated S/C or the module plus component environmental acceptance test program. In reality it should be assumed that all of the failures found at the module level would be found at the component level in the other approaches. This assumption further increases their costs and makes the module only approach even more attractive.

In conclusion the cost risk of failures is acceptable for the module only test approach and in fact is not significantly more than the other alternatives studied.

Technical Considerations

There are two classes of technical considerations, the subjective, based on emotion and the real, based on data.

One subjective argument is that the typical integrated spacecraft approach and the component plus module test approach provide more confidence in hardware performance because of the increased exposure to launch and orbital environments in respect to the module only test approach. However, if one keeps in mind that the basic objective of the environmental acceptance test is to screen workmanship defects, the module tests could simply run longer to achieve equivalent test time. A second subjective argument for vehicle level environmental tests is that the vehicle is tested in the configuration and environment in which it must function to perform the program mission. The typical acceptance Thermal Vacuum test is generally performed with many compromises to flight configuration, heater skins, special test breakout boxes installed, no solar arrays and in the case of OAO without the flight batteries. One more level of compromise is to environmentally test the vehicle in several pieces with the same total system type test run on the observatory in an ambient environment. Even acceptance vibration is compromised when the observatory is reconfigured for T/V. In any approach a final workmanship acoustic would revalidate the final flight assembly of the observatory and should be retained in the module test only approach.

In addition to the subjective arguments described above there are some technical shortcomings to the module test approach which must be considered. Vibration tests at the component level provide a more severe environment than at the module level for screening workmanship defects. Typical vibration environment induced on a component during an acceptance test is shown in Figure 6.18-10. Also shown is the typical environment induced on a component during an observatory level vibration test.

The basic reason for the difference is the attentuation and/or amplification of the frequencies by the vehicle structure located between the input and the component. It is anticipated that the components mounted within the module would see the same type of excitation as shown for vehicle level test, with somewhat better control, due to the smaller module size. How much, or how significant is the difference is unknown and somewhat subjective, and will be evaluated during module qual. tests.

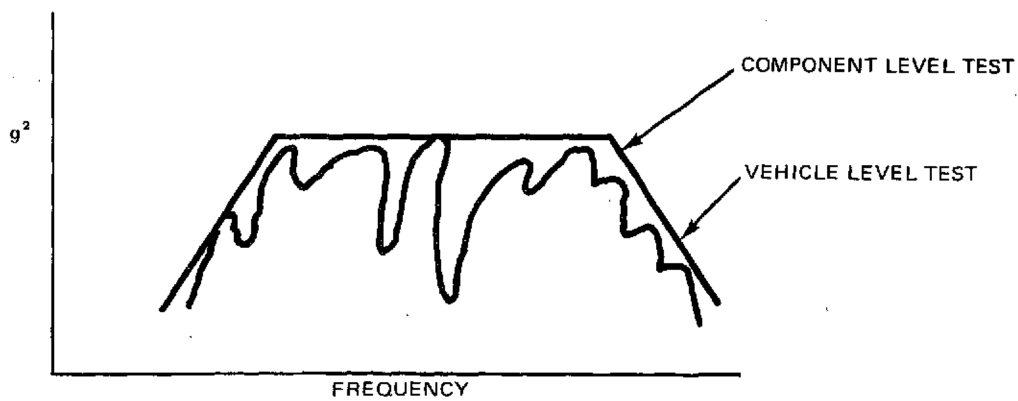

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Fig. 6.18-10 Typical Vibration Environment

Selected components could be subjected to component level vibration tests if qualification tests uncovered inadequate screening levels.

Thermal Vacuum

Thermal vacuum testing at the module level on the other hand would present no significant differences in environment as far as the components are concerned for the purposes of workmanship screening. However, as discussed elsewhere of this report, the addition of thermal cycling is added to further insure screening of all component workmanship defects at the module level.

The basic design is modular and the modules have independent thermal control as does the instrument, once the basic design is verified in observatory level qual tests an adequate baseline will be established for simulation and verification of structure thermal interfaces for acceptance tests. Therefore, verification module thermal controls workmanship also present no problem at the module level.

Schedule Considerations

In addition to the impact of a failure during test discussed under cost considerations, component deliveries have a more significant impact on the total vehicle schedule for the module level test approach. The offset of the cost savings (500K) to the schedule risk is easily seen by assuming 40% of the 500K is set aside for overtime or parallel testing using adiitional men. Converting 40% of 500K to manpower it provides at 35K/manyear 68. 8 manmonths of contingency.

If we ignore for a minute the cost savings in examining the schedule risk, it is apparent that the potential risk is highest in a one or two spacecraft program for the module level test approach and where most of the components are new build. In the module test only case, slippage of a component delivery coupled with the in line workmanship screen of all components at once for the first time, presents a definite schedule consideration which is not present in the other two approaches studied. Where a multi module buy of components is planned for many spacecraft, this problem can be eliminated by planning production rates and gaining experience in production times to provide component backups.

There are more positive schedule considerations for the module only test approaches:

- The option of serial or parallel buildup of modules

- Substitution of an entire qualification module to allow the observatory level I&T flow to continue while awaiting a component delivery
- S/S trouble shooting or component replacement can be performed off line to the basic I&T flow

In summary the schedule risk of the module only test approach is not significant when one considers the relative cost savings and the options for parallel scheduling. In addition the planned EOS components further reduce risk because most of these have been built before and realistic delivery dates established.

Contractual Considerations

In order to compensate for the loss of control over the component tests, it is assumed that a cost penalty of paying for their review of the module test plans, procedures and setup as well as some Q.C. serveillance during test. At the present time, it is estimated that this penalty is about 1/2 mm or 2K/component type, or less than 75K for the first S/C. It is expected that recurring costs would be much less and would eventually approach 0.

6.18.3.3 Qualification Test

Qualification requirements vs. design approaches were assessed and the following areas of impact were identified:

- Addition of the module structure as an element to be qualified in the modular design approach generates an additional 125K in Qualification Test costs
- Component qualification could be accomplished at the module level, however, cost savings is offset by a much higher schedule and cost risk.
- An objective of the qualification tests for the basic spacecraft should be to qualify for follow-on configurations as well as the LRM.

6.18.3.3.1 MODULE STRUCTURAL QUALIFICATION

The module structures present additional elements to be qualified in the modular spacecraft design which are not present in the integrated spacecraft design. If we examine the potential methods of qualification of the module structure shown in Table 6.18-4. Each of the test methods for satisfying the module structural qualification requirements will satisfy the basic test requirements, however the recommended test program shown in Table 6.18-3 is more cost effective from the total qualification program viewpoint. The

Table 6.18-4 Alternative Methods of Qualifying the Module Structure

TEST REQUIREMENT	METHOD 1	METHOD 2	RECOMMENDED METHOD
VERIFY MODULE STATIC LAUNCH LOADS	SEPARATE MODULE LEVEL ACCEL. TEST	S/C OR OBS LEVEL ACCEL. TEST (MASS REPS OF COMPONENTS)	METHOD 2 BECAUSE IT IS LESS COSTLY
VERIFY MODULE MECHANICAL VIBRATION LAUNCH LOADS	SEPARATE MODULE LEVEL VIBRATION TESTS	S/C OR OBS LEVEL VIBRATION TEST (MASS REPS OF COMPONENTS)	METHOD 1 IS REQUIRED TO ACHIEVE OVER ALL EOS PROGRAM COST SAVINGS AND OBJECTIVES
VERIFY MODULE ACOUSTIC LAUNCH LOADS	SEPARATE MODULE LEVEL ACOUSTIC TESTS	S/C OR OBS LEVEL ACOUSTIC TEST (MASS REPS OF COMPONENTS)	METHOD 1 IS REQUIRED TO ACHIEVE OVERALL EOS PROGRAM COST SAVINGS & OBJECTIVES
VERIFY MODULE PYROTECHNIC & LAUNCH VEHICLE SHOCK LOADS	SEPARATE MODULE LEVEL SHOCK TESTS	S/C OR OBS LEVEL SHOCK TEST (MASS REPS OF COMPONENTS)	METHOD 2 BECAUSE IT CAN BE IMPLEMENTED AT MINIMAL COST VEHICLE LEVEL

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merits of each of the test alternatives shown are discussed below:

- Verification of module static launch loads - This test is most cost effectively run at the observatory level.

The acceleration loads induced on the module can be achieved with equal fidelity on the module and/or observatory level, therefore no technical risk is incurred by either approach. In addition, the module structural and mechanical interfaces with the primary structure, are tested at the vehicle level.

The cost of a module acceleration test, excluding the instrumentation which would be the same for each method, is about $2500 per module. Since the basic spacecraft contains, 3 subsystem modules, the IMP module, and an RCS/OAS module, the module acceleration tests would run 12.5K for a A/C set. The vehicle acceleration test run on the GSFC LPS would cost 15K with or without modules based on a recent estimate by GSFC for the USAF/GAF Earth Limb Measurement Satellite program. If we elected to perform the vehicle acceleration loads test statically, using a typical wiffle tree approach it would cost at least as much as the centrifuge approach and does not allow the module structure to be tested in the same test, because of the complex component to module load paths. The cost of mass reps to perform static tests of the module is not a factor since the mass reps are required for OBS modal survey in any event. The same level of instrumentation and personnel support would be required for the modular spacecraft acceleration as would be required for the integrated spacecraft acceleration and therefore, is not a factor in this trade. Therefore, the added cost to the program for acceleration tests for the modules, is only the cost of the time required to physically install the modules in the vehicle. This cost is negligible if the basic spacecraft acceleration load structural test is performed dynamically, which is the recommended test approach.

- Verification of the module mechanical and acoustic launch loads. - The verification of the mechanical and acoustic loads could be accomplished in an observatory level qual test. However the approach selected for acceptance tests had added the requirement to accomplish this at the module level. Module level tests are required for two reasons:

1. To select the best technical approach between acoustic and mechanical tests providing adequate vibration levels to screen component workmanship defects at the module level.
2. It is a design and qualification goal to have a set of modules which are qualified to cover as many follow-on missions as possible. To achieve this the ideal case would be to have the components, and module structure insensitive dynamically to the change instrument configuration. Module level vibration and acoustic tests will baseline a set of component environments and module transmissibility numbers early in the program which can be used to compare the module and component vibration environment inputs to these elements measured in observatory level tests.

- Verification of module pyrotechnic and launch vehicle shock loads. - This is simply accomplished as part of the vehicle level tests and does not require additional consideration because of modular design.

Therefore, the total impact of the addition of the module requirements to test program is the cost of the additional structural acoustic and vibration qualification tests. This added cost is generated by both the overall program goal of universal modules and the environmental acceptance test philosophy of module only test. It is estimated that delta this cost to the program averages approximately 15K/module which impacts total program non-recurring cost by 75K.

6.18.3.3.2 Qualification Level of Test

Modular design also offers the same flexibility for combining levels of qualification testing as described for the Integration and Test environmental acceptance tests. Our present recommendation is to perform component and system level environmental qualification tests for the first modular spacecraft mission.

Components could be qualified at the module level either for selected environments or for the total component qualification test program. System Qualification tests could also be conducted in segments as described for the acceptance test program. Since the cost of component qualification tests is significantly higher than for acceptance and the level of prime contractor participation is proportionately higher, the potential cost savings would be significantly higher than in the component acceptance test program. However, this cost savings must be traded off against the risks to the program for combining component qualification at the module level and/or eliminating system level qualification tests.

The first question is the potential cost savings for each approach. Table 6.18-5 summarizes the cost for environmental qualification tests for each test level.

Table 6.18-5 Environmental Qualification Test Costs

TEST LEVEL	TEST COSTS INCLUDING FUNCTIONAL			
	VIB & SHOCK	T/V	ACCEL	TOTAL
COMPONENT*	–	–	–	700K
MODULE (5)	200K	400K	100K	700K
OBSERVATORY	150K	500K	150K	800K

3-216 *20K/COMPONENT x 35 COMPONENTS REQUIRING QUAL.

The observatory and module test costs were derived by adding the recurring and non-recurring costs derived for acceptance tests documented in section 1.1.2 and adding in time for fixture design. The component level test costs were derived by counting the number of discrete components, and assessing the number requiring qual or delta qual at 35 components x 20K/component qual test.

Therefore, the total cost of each test approach adding in 150K for the instrument and 60K for functional test of the qualification units prior to delivery, is as follows:

Module only	910K
Components + Module	1.55M
Components + Vehicle	1.65M

As suspected, the cost of conducting of the module qualification tests at the module level only is significantly cheaper.

Now let us consider the cost risk in terms of cost per failure, specifically in light of the high percentage of unqualified components (63%).

Using the same logic as for acceptance test failures discussed in subsection 6.18.3.2.2 and assuming 20K for a unit retest, the cost per qual test failure is as follows:

Component Level	=	20K/Failure
Module Level	=	25K/Failure
Observatory Level	=	80K/Failure

by the same process used for the analysis of the acceptance test program. We can say that the $\frac{\text{(Observatory Level \$) - (Module Level \$)}}{\text{(The \$ / Module Failure)}}$

The number of failures for which the cost of the module approach reaches the minimum cost of the component-observatory level approach.

$$= \frac{1.65\text{ M} - .910\text{ M}}{.025} = 29.6 \text{ Failures}$$

If one neglected risk this would be assumed to conclude that the module level qualification program is the best approach. However, if we consider that 63% of the module components require qualification and it is obvious that the risks inherent in the module level only qualification approach outweighs its cost savings as follows:

- The failure of any component in module qualification is a cost of 25K + the schedule delay incurred because presumably there is no substitute component which permits the module test to continue. If the schedule slip/day is only 5 I&T men idle and no total program slip, 1 man week/day would be added to the program cost. If the component could be analyzed, repaired, retested and be reinstalled in the module with only 1 week schedule slip, 4K more/failure must be incurred driving the number of allowed failures down to $\frac{1.65\text{M} - .910\text{M}}{29\text{K}} = 25.5$

- The probability of failures in qualification test is much higher than the 10% or 5.6 failures used for the analysis of modular acceptance test. For qualification tests, the probability of failure should be raised to 30%, therefore, 30% x 56 boxes - 16.8 failures. This leaves a total of (8.7)(29K) = 250K for catastrophic type schedule slips (the more likely case) where the problem cannot be repaired in one week.

- The module only approach presents a much higher potential for total program slip since each failure is directly in line with the program schedule and substitute components would not exist.

- Subcontractor participation in qualification tests conducted at the module level would be much higher because it is the first environmental tests for this component. Whether he was involved at his contractual insistence to verify the prime's test plan, or because the prime wanted to insure rapid recovery in the event of a failure, it would probably cost about 1 manmonth/component in the subcontractors engineering Q.C. cost, or 4K/box x35 boxes= 140K more. This would be directly off the cost savings reducing the contingency to 110K.

In the comparison of the component-module approach to the component-system level approach costs, about 100K is saved by the component-module approach. ($100K).

In summary, it is concluded that the cost risk of placing all the emphasis on the module qual test only approach is not acceptable and the small cost savings of the other alternative, the component-module test approach, with no system level qualification test, is not worth the loss of design confidence. Therefore, for the first modular spacecraft mission where a high percentage of the components require qualification, it is recommended that the qualification test be conducted at the component and system level.

6.19 RELIABILITY AND QUALITY ASSURANCE

The Reliability tasks listed in NHB 5300.4 (1A), "Reliability Program Provisions for Aeronautical and Space System Contractors," have been reviewed to identify those tasks which can be performed in a more cost effective manner without altering program risks. The three major areas investigated are:

Reliability Program Management

Reliability Engineering

Test and Reliability Evaluation

It was determined that significant savings can be achieved in the areas of (a) Reliability Program Control, Progress Reporting and Evaluation, (b) Reliability Predictions, and (c) Problem/Failure Reporting and Correction for non launch critical GSE. If all the recommended EOS alternative approaches are selected, an estimated 7200 man hour savings can be realized without altering program risk. The detailed discussion of this study is presented in Report IV, Management Approach.

SYSTEM COSTING

7 - SYSTEM COSTING

This section presents the costs for the EOS program options and summarizes the methods used to collect these costs.

7.1 PROGRAM OPTION COSTS

Figure 7-1 presents the total cost in 1974 dollars for the basic EOS spacecraft and modules, the program options studied, and the cost of adding the spacecraft configuration options, redundancy and Shuttle utilization functions. The cost areas listed vertically are the elements of the EOS WBS and the horizontal columns represent the program options broken down into non-recurring and recurring cost for each program option.

7.2 COSTING GUIDELINES AND ASSUMPTIONS

The assumptions used in the definition of each program option are identified in Figure 7-1, under each program and configuration option. The costing groundrules and assumptions used for each WBS element are based on the Program Master Schedule shown in the Management Approach volume, the EOS WBS shown in Figure 7-2, and the WBS Dictionary found in the appendices to this volume. A summary of the WBS element costing groundrules used in estimating and collection of the cost of options are as follows:

- Structure costs - Include the design, manufacturing, tooling and wiring of the basic structure.
- Modules costs - Include all engineering, manufacturing, tooling, test, QC and hardware procurement costs for the modules.
- System Engineering and Integration - Includes the systems analysis, systems integration and instrument accommodations.
- Integration and Test - Includes the engineering, manufacturing and QC for all activities required to integrate the modules into the basic spacecraft, integrate the mission peculiars, perform functional acceptance tests and launch operations.
- Development and Qualification Test - Includes all the spacecraft, module and observatory development and qualification tests, excluding component qualification which is costed under module non-recurring costs.
- Environmental Test - Covers the workmanship acoustic test on the flight observatory prior to shipment to the launch site.
- GSE S/C and GSE mission - Includes all T&I software, electrical, mechanical and fluid GSE design and manufacturing.

- MSS, TM, DCS - Cost of basic instruments, costs used are as follows:

	Non-Recurring	Recurring
MSS	$1.0 M	$6 M
TM	13 M	7 M
HRPI	10 M	5 M
DCS	$2 M	$0.500K

- TM, MSS, Instrument Data Handling - Cost of spacecraft Instrument Mission Peculiars (IMP) including module design, test and hardware.
- Control Center Operations - Includes the mission operations software, mission planning and mission operations.
- Control Center - Includes the hardware design and fabrication.
- Data Processing Operations - Personnel support for operation of the central data processing facility.
- Central Data Processing - Includes all management, engineering, procurement, manufacturing, facilities and integration and test costs required to provide a Ground Data Processing facility for the mission instruments.
- Launch System - Launch system costs include the fairing, launch vehicle, launch services, and A.F. range supports costs. These costs were obtained from the respective launch vehicle contractors and the JSC published cost for a Shuttle Launch and are as follows:

	Non Recurring	L/V Recurring	Fairing & Adapter	Launch Services
Delta 2910	$0.25M	$ 4.5 M	$1.0	$3.0 M
Delta 3910	0.5 M	5.3 M	1.0	3.0 M
Titan IIIB	3 M	8.3 M	1.47 M	3.37 M
Shuttle	$2M	$10 M	TBD	TBD

7.3 COST METHODOLOGY

Cost Methodology is described in two subsections. The first, Section 7.3.1, covers the estimating flow from requirements definition to cost reporting. The second, Section 7.3.2, describes the EOS Data Bank and its use in the cost estimating process.

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7.3.1 COST ESTIMATING FLOW

The flow chart, Figure 7-3, covers program definition and cost estimating. The content of this flow chart is described with forms and instructions which were used during the study and are presented as a series of figures and instruction sheets as follows:

- Performance Requirements (Fig. 7-4): Relates configuration to mission model - gives summary configuration data and mission program planning data.
- Mission Model/Master Schedule: Portrays mission and program milestones and tasks
- Equipment List (Fig. 7-5): Describes subsystem equipment at component level

Figure 7-6 represents the cost data bank inputs. Three basic formats are required as inputs to this estimate core, represented by Figures 7-6(a), (b) and (c).

- Component Selection Sheet (Fig. 7-6a): Identifies components and quantities for each equipment subsystem Module of the configuration option.
- EOS Procurement Cost Data (Fig. 7-6b): This format is used by the Procurement Department to document all estimates for procured equipment. Data for these sheets will originate from:-
  - In-house estimates presently available from other programs such as ELMS or GPS.
  - Seller response to informal request or formal ITQ.
- GAC Task Description/Manpower Sheets (Fig. 7-6c): Covers task description and manpower associated with each WBS item at the input level. References hardware and software cost to be included with task manpower in a single WBS item.

These three cost estimate formats provide input to the computer program which compiles cost for each WBS at Level 5 (a few elements are costed at Level 4).

7.3.2 EOS DATA BANK

The cost data bank stored in the computer was initialized for EOS using the cost estimating data provided by the EOS Cost Procurement Data Sheet shown in Figure 7-6(b). Once the basic data bank was established, it provided a simple method of compiling program hardware costs based on component selections for each option studied.

The flow of data bank inputs and outputs are portrayed in Figure 7-7. Each block of the flow chart which refers to formats is referenced to a numbered example of the specific format as follows:

- Input Formats: Formats 1 and 2 are input formats (Ref. Fig. 7-5 and 7-6(b) used to generate data bank information for candidate subsystem components. Format 5 (Ref. Fig. 7.6(a)) is used to select from the data bank, components desired for a specific configuration.
- Output Formats: Format 3 (Ref. Fig. 7-8 for sample) lists all data in the data bank. Format 6 (Ref. Fig. 7-9) is the output resulting from component selection. It is the basis for procurement cost input to the Cost Summary, Figure 7-1, elements of the WBS.

Data bank outputs coupled with manpower estimates were used to evaluate the total program costs for each option studied. Costs of selected configurations are those presented in Figure 7-1. This data bank will be maintained for use throughout the EOS study.

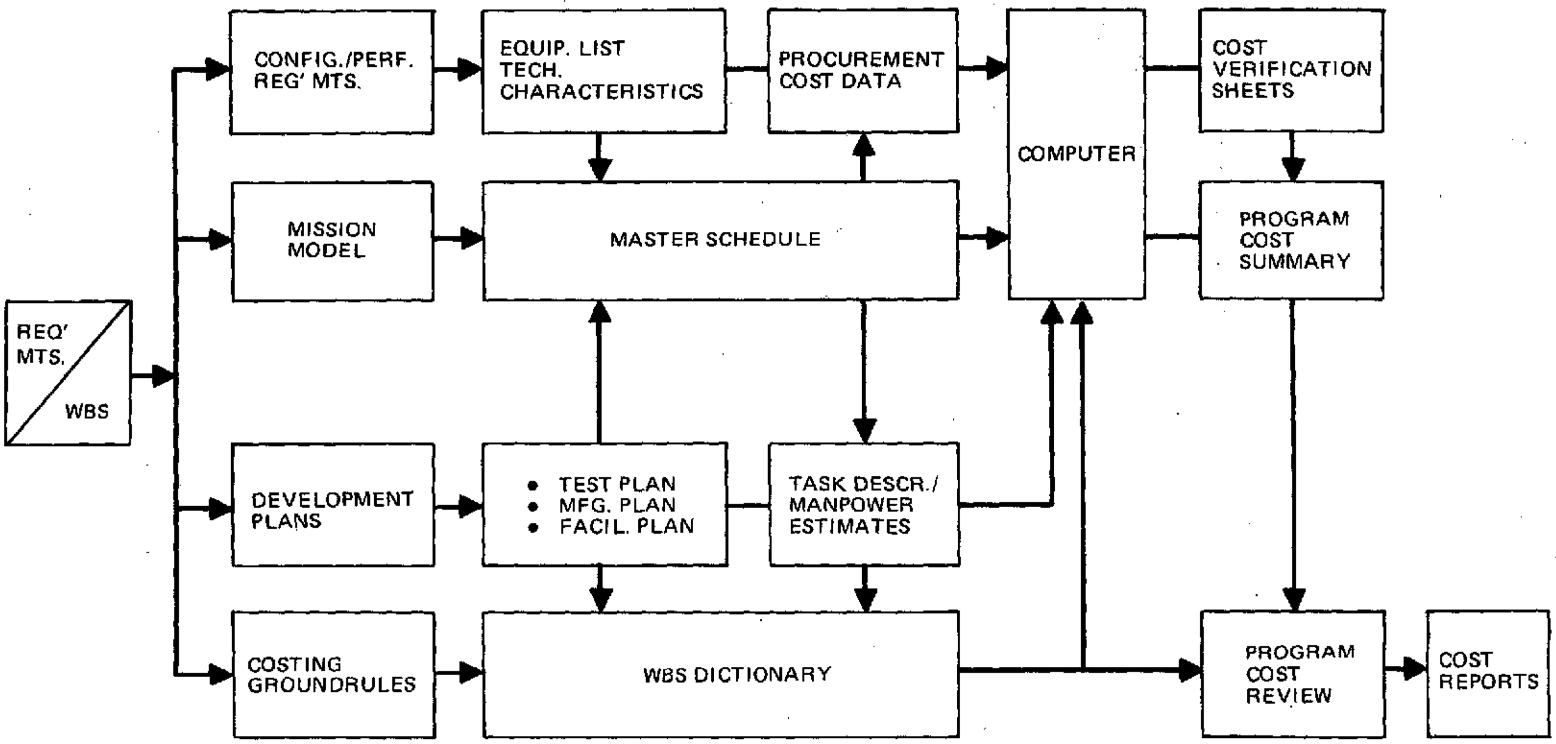

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Fig. 7-3 Cost Estimating Flow

DESIGN RECORD – PERFORMANCE REQUIREMENTS

NOMENCLATURE	SHEET	HARDWARE/SOFTWARE IDENTIFICATION NO. / SPECIFICATION NUMBER/WBS NUMBER
DELTA BASELINE CONFIGURATION	A	Config. PDDC509-101 Mission Model A, A[1]

REFERENCES	PERFORMANCE REQUIREMENTS	
	Launch Vehicle	Delta 2910
	Allocated Wt., S/C*	2660 LBS
	Instruments Compliment (LRM)	MSS, TM, DCS
	Follow-on Instruments	None
	Shroud	86 in Dia 237 in high max.
	S/C Mounting	Base
	Modularity	Modular design with three standard modules
	Resupply Definition	On-orbit, Shuttle, Retrieve & Resupply
	Kick Stage	None
	Data Transmission	(A) Direcr or (B) WBTR (MSS)
	Orbit Time of Day	Pre-launch Selectable 9:30 AM - 2:30 PM
	Program Cost Target	$175 Million
	Launch Site	WTR
	Launch Data (1st S/C)	CY1979
	Orbit	(366.1 NM) 678.0 KM 98.09°
	No. of Flt. S/C	2
	Shuttle Demo Flt S/C	No
	Life	2 yrs, Consumables 5 yrs, Survival 5 yrs

PREPARED BY	GROUP NUMBER & NAME	DATE	CHANGE LETTER
R. Papsco	EOS Systems Eng.	5/15/74	B
			REVISION DATE 6/19/74
APPROVED BY J. Marino			PAGE 2 OF 2

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Fig. 7-4 Design Record, Performance Requirements

WBS NO.

CONFIGURATION E O S EQUIPMENT LIST

SUBSYSTEM/COMPONENT DATA HANDLING

COMPONENT NAME	DEVELOPMENT STATUS			PHYSICAL CHARACTERISTICS							TECHNICAL CHARACTERISTICS	RELIABILITY		PROGRAMMATIC		
	STATE-OF-THE ART RATING	MONTHS TO QUAL	OTHER PROGRAM USAGE	MATERIALS	WEIGHT (LBS)	LENGTH (INCHES)	WIDTH (INCHES)	HEIGHT (INCHES)	VOLUME (CUBIC INCHES)	POWER (WATTS)		EQUIVALENT OPERATING HOURS	MTBF (HOURS)	UNITS PER VEHICLE	EQUIVALENT UNITS FOR S/S TEST	EQUIVALENT UNITS FOR MAJOR GND TGT
COMPUTER	4		ERTS-B		34	13	7	9		38	16K			1		
TAPE RECORDER	5	12	AF/503		14.5	9	13	5.4	550	8	HEALTH, STATUS, RECOG.		12K	1	N/A	1
CLOCK	4		OSO-I		4	16.5	5.8	3.2		4	± 1 PART IN 10^6/DAY (STAB)			1		
FORMAT GENERATOR	4		OSO-I		3	14.6	5.8	1.4		2				2		
REMOTE DECODER	2		NEW		1	6	4	2		0.1	64 CHANNELS			6		
DUAL REMOTE MUX	2		NEW		1	6	4	2		0.1	64 CHANNELS			3		
SIGNAL CONDITIONER	4		NEW		16	13.75	8	10		23.8				1		
SENSORS	4	8			0.2	N/A	N/A	N/A	N/A	0.1			1M	100	N/A	10
COMMAND DECODER	3		NEW		12	12	3	6		3				2		
PRECISION CLOCK	4		NEW								± PART IN 10^9/DAY MISSION PECULIAR			2		

Fig. 7-5 Equipment List

COMPONENT SELECTION SHEET

Program Option No. ________ Date ______

WBS No. ________ Subsystem ________ Cog. Engr. ______

Code	Name of Component	Quantity		Complexity Factor	REMARKS
		Non-Recurring	Recurring		

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Fig. 7-6(a) Component Selection Sheet

EOS PROCUREMENT COST DATA

ITEM DESCRIPTION/MODEL Remote Multiplex System (Config. #3) *   DATE 7/3/74

WBS 1.7.3.1/52-01 S   PREPARED BY R.B. Conklin

PROGRAM EOS.   COG. ENGR. T. Newman

SELLER S.C.I.   COST/YEAR $ 1974

	Development $ (Excl.Hrdwr.)	Hardware Qty*	Hardware Cost	Total Cost	Average Unit Cost
SUBSYS/COMPONENT:					
Design					
Vendor Test					
GAC Test Hdwr.					
Modification/Design					
Restart					
Sub-Total	866,000			866,000	
Production					235,000 **
SUPPORT:					
GSE					
Spares					
Operations					
Total Support					
GRAND TOTAL					

*Enter test units, production units, GSE units or sets; use equivalent unit approach for bits and pieces, for example: 1.5 test units for a complete test unit and test spares.

DATA SOURCE/NOTES Sellers budgetary estimate per TWX - 7/2/74.
* Combined remote units - not power stacked.
** Price per S/C

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Fig. 7-6(b) EOS Procurement Cost Data

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GAC - TASK DESCRIPTION/MANPOWER

TASK NAME____________________ DATE____________________

WBS____________________ PREPARED BY____________________

PROGRAM OPTION____________________ COG. ENGR.____________________

TASK OBJECTIVE

HARDWARE	SOFTWARE

	SUB-TASK REQUIREMENTS & MANPOWER BY FISCAL YEAR								
SUB-TASK	INPUT	OUTPUT	1976	1977	1978	1979	1980	1981	1982
1.									
2.									
3.									
4.									
5.									

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Fig. 7-6(c) Grumman Task Description/Manpower Sheet

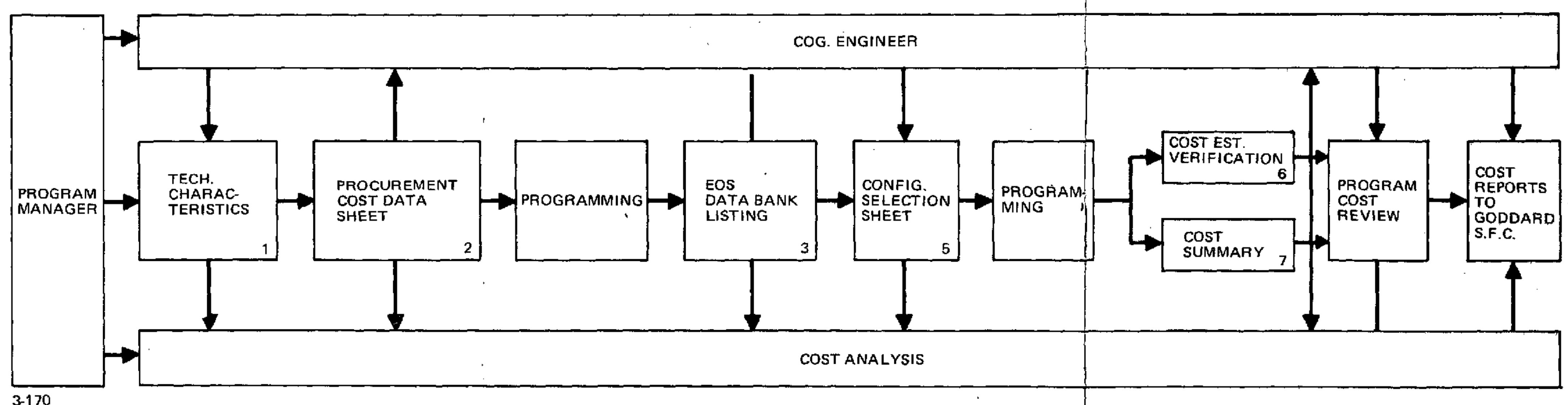

Fig. 7-7 EOS Data Bank/Subsystem Cost Flow

FOLDOUT FRAME 1

FOLDOUT FRAME 2

EOS COMPONENT PROCUREMENT COST

WBS NO. 1.7.3.1 SUB SYSTEM: BASIC C&DH MODULE COG ENG: J.BERNSTEIN/T.NEWMAN

DATE 9/06/74

COMPONENT CODE

2 DIGIT GENERIC CODE
2 DIGIT SELLER I.D.
1 LETTER EST.SOURCE G = GRUMMAN,C = CSC,S = SELLER
1 LETTER COST REVISION LEVEL A - Z,A = 1ST,B = 2ND,...

CODE	COMPONENT	SELLER	MODEL	NR COST	UNIT COST	GRUMMAN COST CODE	FLIGHT EXP YEARS	REMARKS
1101G	BUS CONTRL/FORMATTER	OSO-I		0.0	44.0	004	0.0	3LB-2WATTS-OSO-I-2PV
1102S	BUS CONTRL/FORMATTER	HARRIS		0.0	17.0	004	0.0	BUS CONTRL/FORMATTER
1201G	REMOTE DECODER	OSO-I		0.0	102.0	004	0.0	1LB-.1WATT-64CHNLS-NEW-6PV
1301G	DUAL REMOTE MUX	OSO-I		0.0	62.0	004	0.0	1LB-.1WATT-64CHNLS-NEW-3PV
1401G	COMMAND DECODER	OSO-I		0.0	92.0	004	0.0	12LB-3WATTS.
1402GA	COMMAND DECODER	SPACETAC		180.0	29.0	004	0.0	COMMAND DECODER
1403S	COMMAND DECODER	HARRIS		100.0	92.0	004	0.0	COMMAND DECODER
1501GA	SIGNAL CONDITIONER	SPACETAC		33.0	37.0	004	0.0	ELMS
1502GA	SIGNAL CONDITIONER	SPACETAC		134.0	27.0	004	0.0	SIGNAL CONDITIONER
1504S	SIGNAL CONDITIONER	HARRIS ELECTRONICS		393.0	42.[illegible]	004	0.0	SIGNAL CONDITIONER
1505S	SIGNAL CONDITIONER	ITHACO		217.4	45.2	004	0.0	SIGNAL CONDITIONER
1601G	SENSORS FUEL,TEMP	TBD		0.0	0.0	004	0.0	ELMS
1602G	SENSORS FUEL,TEMP	COX		0.0	1.0	004	0.0	SENSORS FUEL,TEMP
1701G	SPACECRAFT INTERF	TBD		0.0	2.7	004	0.0	SPACECRAFT INTERF
1801G	WIRING HARNESS	TBD		0.0	2.2	004	0.0	WIRING HARNESS
1901G	BUS PROTECT ASSY	TBD		0.0	0.5	004	0.0	BUS PROTECT ASSY
2001GA	CLOCK	GULTON IND		4.0	19.0	004	0.0	ELMS
2101G	MEMORY MODULE	TBD		0.0	25.0	004	0.0	MEMORY MODULE
2201S	REMOTE UNIT	TBD		0.0	10.0	004	0.0	REMOTE UNIT

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Fig. 7-8 Format 3 Sample

OPTION: A / A'

EOS COST ESTIMATE VERIFICATION

WBS NO. 1. 7. 3. 3   SUBSYSTEM: BASIC ACS MODULE   COG ENG J.ZEMAN/G.ZETKOV

DATE 7/17/74

CODE	COMPONENT	SELLER	MODEL	NON-RECURRING COMPLEX/WRAP	NON-RECURRING QTY	NON-RECURRING COST	RECURRING COMPLEX/WRAP	RECURRING QTY	RECURRING COST	TOTAL COST
	*** PROCUREMENT COST ***									
0101G	COARSE SUN SENSOR	BENDIX	1771	1.0	0	5.00	1.0	2	4.00	9.00
0201S	DIGITAL SUN SENSOR	ADCOLE	1594	1.0	0	0.00	1.0	1	42.00	42.00
0303S	GYRO	BENDIX NAV &CONT.DIV		1.0	0	650.00	1.0	1	235.00	885.00
0503GA	STAR TRACKER FIXT HD	ITT		0.3	0	40.50	1.0	1	43.00	83.50
0901G	SPACECRAFT INTERF	TBD		1.0	0	0.00	1.0	1	2.70	2.70
1001G	WIRING HARNESS	TBD		1.0	0	0.00	1.0	1	4.50	4.50
1101G	BUS PROTECTION ASSY	TBD		1.0	0	0.00	1.0	1	0.80	0.80
1601G	THERMAL SENSOR&CNTRL			1.0	0	0.00	1.0	4	6.00	6.00
1701G	THERMAL BLANKET			1.0	0	0.00	1.0	1	0.40	0.40
1801G	RAW MATERIAL			1.0	0	0.00	1.0	1	0.10	0.10
1901G	REACTION WHEELS			1.0	0	10.00	1.0	3	90.00	100.00
2001S	ACS SUBSYSTEM	ITHACO		1.0	0	370.40	1.0	1	193.20	563.60
2101S	REMOTE UNIT	HARRIS		1.0	0	0.00	1.0	2	20.00	20.00
	PROCUREMENT TOTAL					1075.90			641.70	1717.60
	*** TOTAL IN/HOUSE***					0.00			0.00	0.00
	*** GRAND TOTAL ***					1075.90			641.70	1717.60

Fig. 7-9 Format 6 Sample