NASA CR-143674

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 7: EOS SYSTEM DEFINITION REPORT

(NASA-CR-143674) EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY. REPORT NO. 7: EOS SYSTEM DEFINITION REPORT (Grumman Aerospace Corp.) 228 p HC $7.5 CSCL 22B G3/18
N75-15720
Unclas
09257

GRUMMAN

225

N75 15720

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 7: EOS SYSTEM DEFINITION REPORT

(NASA-CR-143674) EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY. REPORT NO. 7: EOS SYSTEM DEFINITION REPORT (Grumman Aerospace Corp.) 228 p HC $7.5 CSCL 22B G3/18

N75-15720

Unclas 09257

GRUMMAN

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 7: EOS SYSTEM DEFINITION REPORT

Prepared For

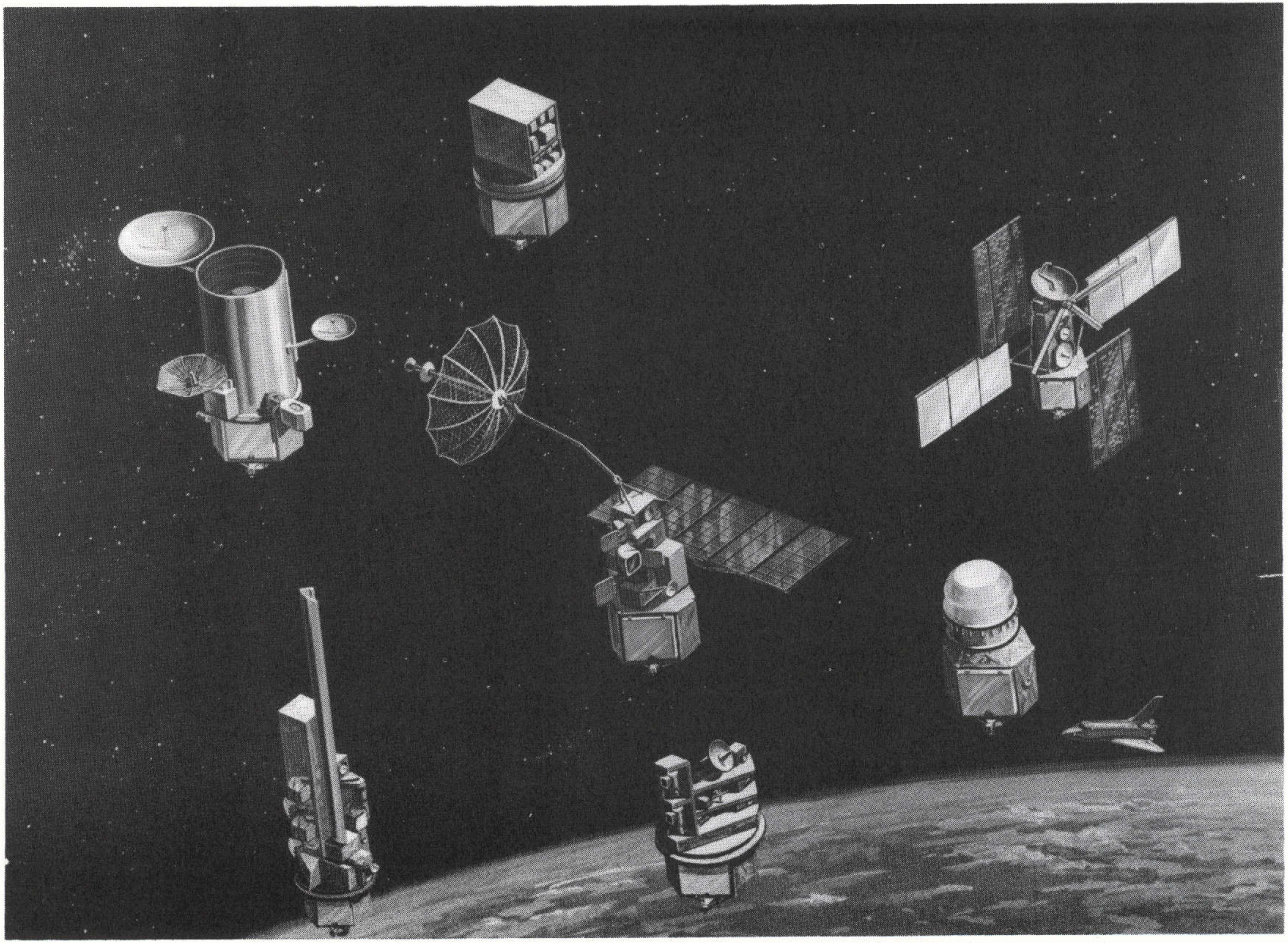
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND 20771

Prepared By

GRUMMAN AEROSPACE CORPORATION
BETHPAGE, NEW YORK 11714

NAS 5-20520

November 1974



CONTENTS

CONTENTS (Cont)

CONTENTS (Cont)

CONTENTS (Cont)

VOLUME 7 APPENDIX LIST*

------

*Appendixes are bound together in a separate volume

ILLUSTRATIONS

ILLUSTRATIONS (Cont)

ILLUSTRATIONS (Cont)

ILLUSTRATIONS (Cont)

TABLES

TABLES (Cont)

TABLES (Cont)

ABBREVIATIONS

AOP	Advance On-Board Processor
ACS	Attitude Control Subsystem
APDEL	Applications Program Development Laboratory
ATM	Apollo Telescope Mount
AASIR	Advanced Atmosphere Sounder and Imaging Radiometer
AP	Array Processor
AH	Amphere Hour
B&W	Black and White
BI	Bilinear
CCT	Computer Compatible Tapes
CRT	Cathode Ray Tube
CDP	Central Data Processing
CPF	Central Processing Facility
CTM	Compacted Thematic Mapper
C&DH	Communications & Data Handling
CMOS	Complimentary Metal Oxide Semi-conductor
CPS	Central Processing System
CC	Cubic Convolution
CONUS	Continental United States
CCD	Changed Coupled Device
CDPF	Central Data Processing Facility
DTC	Design-to-Cost
DOI	Department of the Interior
DT	Direct Transmission
DMS	Data Management System
DHG	Data Handling Groups
DTND	Decending Node Time of Day
EOS	Earth Observatory Satellite
EPS	Electrical Power Subsystem
ETC	Engineering Test Center
ERTS	Earth Resources Technology Satellite
EGRET	Explorer Gamma-Ray Experiment Telescope
FSS	Flight Support System
GSFC	Goddard Spaceflight Center
GDS	Ground Data System
GCP	Ground Control Points
GP	General Purpose
HRPI	High Resolution Pointable Imager
HDDT	High Density Digital Tape
IMS	Information Management System
IF	Intermediate Frequency
IR	Infra Red
IRU	Inertial Reference Unit
ISS	Information Services System
ITOS	Improved TIROS Operational Satellite
IMP	Instrument Mission Peculiars
LRM	Land Resources Management
LCGS	Low-Cost Ground Station
LUS	Local User System
LDEL	Local User System Diagnostic & Equipment Laboratory
L/V	Launch Vehicle
LEST	Large Earth Survey Telescope
MSS	Multi-Spectral Scanner
MISCON	Mission Control
MEMS	Module Exchange Mechanism System
MLI	Multi-Layer Insulation
MUX	Multiplexer (ing)
MIB	Minimum Impulse Bit

Preceding page blank

ABBREVIATIONS (Cont)

MMH	Mono Methyl Hydrazine
MEV	Million Electron Volts
NASCOM	NASA Communications
N_2H_4	Hydrazine
N_2O_4	Nitrogen Tetroxide
NN	Nearest Neighbor Interpolation
OBC	On-Board Checkout
OTS	Orbit Transfer Subsystem
OMS	Orbit Maneuvering Subsystem
PCC	Project Control Center
PDRM	Payload Deployment & Retrieval Mechanism
PGS	Primary Ground Station
PMI	Photomultiplier Tubes
PDS	Processor & Display Subsystem
QC	Quality Control
R&D	Research & Development
RF	Radio Frequency
RCS	Reaction Control Subsystem
SCPS	Support Computer Programming System
S/N	Signal to Noise
S/C	Spacecraft
SMM	Solar Maximum Mission
SRM	Solid Rocket Motor
SP	Special Purpose
STN	Satellite Tracking Network
SEASAT	Sea Satellite
SEOS	Synchronous Earth Observatory Satellite
SAR	Synthetic Aperture Radar
SP	Special Purpose
SCP	Storage Control Processor
TM	Thematic Mapper
T&T	Test & Integration
TDRSS	Tracking & Data Relay Satellite System
UTM	Universal Transverse Mercator
WBVTR	Wide Band Video Tape Recorder
WBS	Work Breakdown Structure
WBDHC	Wide Band Data Handling & Compaction

1 - EXECUTIVE SUMMARY

1.1 INTRODUCTION

The concept of earth remote sensing, both airborne and spaceborne, has been well demonstrated and has achieved wide acceptance. Although we have only started to exploit the social and economic benefits obtainable from remote sensing it is apparent that these benefits will have far reaching implications. Exploitation of remote sensing by the ultimate users, such as industry and local governments, will expand significantly in the next few years because of

- Better understanding of the data we are now getting
- Improved data processing techniques that give greater utility
- Cultivation of more varied uses for the data.

Current achievements in remote sensing clearly indicate that in four to five years a second generation capability will be needed to meet the more sophisticated demands for a higher resolution, higher data capacity, operational system. The basic objective of this study is to develop an operational remote sensing system for land resources management that meets these anticipated needs in 1979, and will accommodate follow-on missions through the 1980's.

NASA has recognized and clearly stated the need for reducing the cost of doing business in space. In line with this policy, three key guidelines were established for this EOS study:

- The basic EOS spacecraft will be a standard, modular "bus", useable for a broad range of earth orbiting missions in the 2000- to 6000-lb observatory class
- The standard spacecraft (Basic Spacecraft) will make maximum use of existing, "off-the-shelf" hardware
- The EOS observatory will be designed to interface with the Space Shuttle and will utilize it for maximum economic and operational benefits.

Adherence to the first of these guidelines will eliminate the costly and time-consuming task of developing a new dedicated spacecraft for every new class of satellite. A representative set of missions has been established by NASA/Goddard to provide driver requirements for the standard spacecraft (Table 1-1). Spacecraft modularity opens the door to reduced test costs as well as in-orbit module replacement by the Shuttle, if proven economically profitable. Use of existing hardware is now feasible with the wide variety of flight-proven, observatory-class hardware that is presently

Table 1-1 EOS Missions

MISSION	PURPOSE	INSTRUMENTS	LAUNCH DATE
A & A'	LAND RESOURCES MANAGEMENT	MULTI-SPECTRAL SCANNER, THEMATIC MAPPER	1979, 1980
B & B'	LAND RESOURCES MANAGEMENT	2 THEMATIC MAPPERS	1981, 1982
C	MARINE WATER RESOURCES AND POLLUTION	2 THEMATIC MAPPERS, HI RESOLUTION POINTABLE IMAGER, SYNTHETIC APERTURE RADAR	1980
D	OCEAN DYNAMICS	(SEASAT B)	1981
E	WEATHER OBSERVATION	(TIROS-O)	1982
F	TRANSIENT ENVIRONMENT PHENOMENA	(SEOS)	1981
G	OCEAN DYNAMICS	(SEASAT A)	1979
H	SOLAR MAXIMUM MISSION	(SMM)	1979

7T-55

available. Virtually all of the defined subsystem requirements can be met from this bank of qualified hardware, thereby saving considerable development cost and risk. The Space Shuttle will be the principal means of space transportation in the 1980's; potentially, it can reduce the cost of maintaining a long-term operational system through observatory retrieval or in-orbit resupply.

The EOS observatory and ground systems, defined in this report, meets these guideline objectives in a timely and cost effective manner. The program defined is for two spacecraft, EOS-A and A′, for land resource missions. The spacecraft design also has the capability to accommodate the long-term operational LRM missions as well as a wide variety of other follow-on missions.

1.2 PROGRAM/USER REQUIREMENTS

The success of the EOS program is primarily dependent on how well we reflect or anticipate the requirements of the user community. A number of surveys and symposia have been conducted by NASA to compile and order these requirements. In the Land Resources domain, these surveys and Grumman's direct user discussions have shown a reasonably consistent pattern of required improvements over present remote sensing systems: higher resolution, broader spectral coverage, faster data turnaround time, and digital data products in addition to current photographic products. These improved performance needs have been reflected in the EOS missions in terms of the following program requirements:

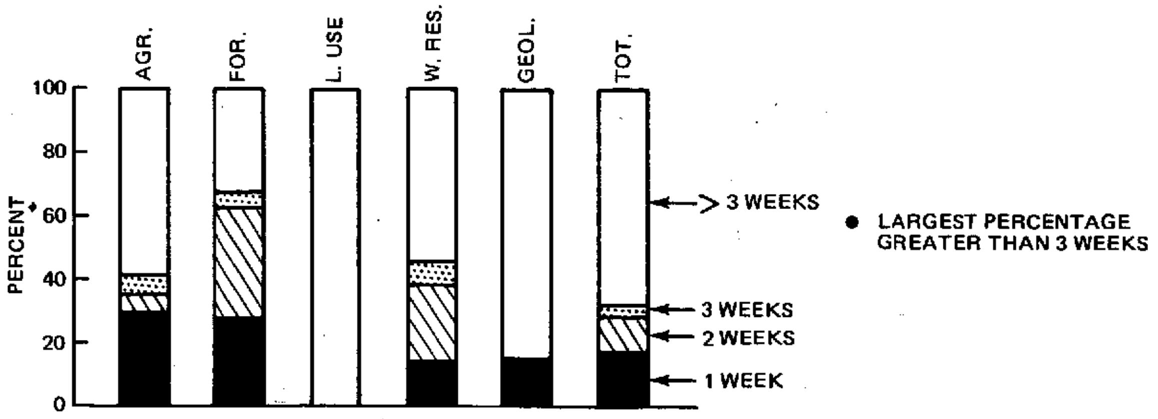
- Spatial Resolution: ±15m (1σ) for broad thematic mapping; 10 m for local imaging
- Spectral Bands: 7 bands covering the visual IR range
- Orbit: Sun-synchronous with descending node time of day between 9:30 a.m. and noon
- Mapping Revisit Cycle: 17 days maximum; 7 to 9 days as a design goal
- Data Products: Primarily digital (high-density digital tape, computer compatible tape) plus color and black and white photographs
- Data Turnaround Time: 24 to 48 hr
- Central Data Processing Throughput: 20 to 400 scenes per day (185 x 185-km, 7-band scenes)
- Number of Generic User Data Products: 10 to 100.

Since a very broad user community is involved, it is impossible to fully reflect everyone's needs in these requirements. It is, however, possible to maximize the number of users satisfied as a function of system cost and complexity. This requires understanding of the distribution of requirements among members of the user community and, to the greatest extent possible, the relative importance or priorities of these requirements. We have examined the user requirements distributions in consultation with Dr. Baumgardner of LARS at Purdue University. Figure 1-1 shows summary histograms of typical user requirement parameters based on a breakdown of specific applications under the general categories of Agriculture, Forestry, Geology, Land Use, and Water Resources. Distributions like these have been used in our study to evaluate the effectiveness of system performance capability (or requirements) as a function of cost. System effectiveness is a function of percentage of the User Community that is satisfied for each level of performance.

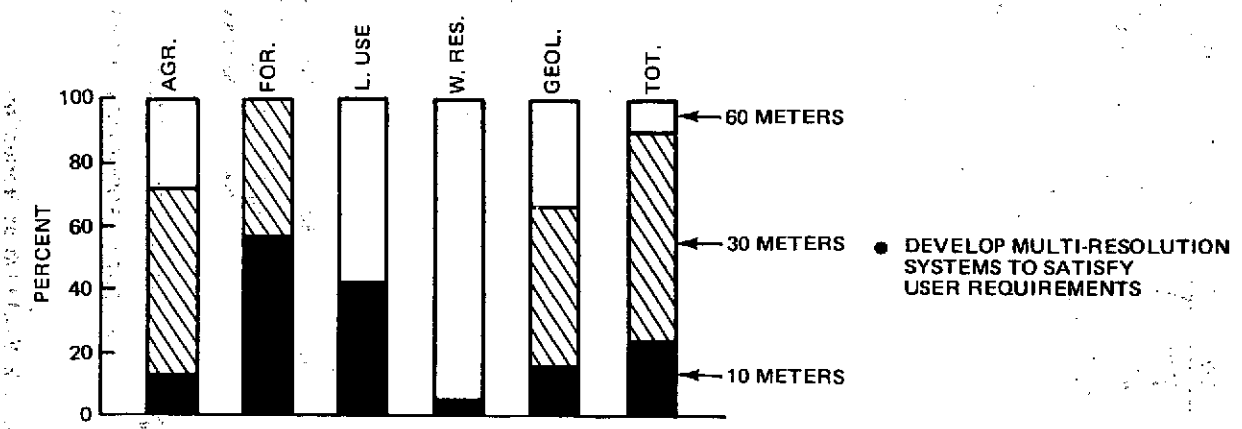
To evaluate the system data processing capacity we used International Data Acquisition as the measure of the effectiveness of system capability. We plotted world crop and rangeland distributions on maps such as the one illustrated in Fig. 1-2. It was then possible to overlay image scan swaths over these maps and determine the number of scenes required to map the world's wheat crop, for instance. The resulting number of (Thematic Mapper) scenes per day, as well as other data load sizing factors, are given in Table 1-2.

In general, the program effectiveness results support the program requirements discussed previously. Subsection 1.11 discusses our program effectiveness evaluation results.

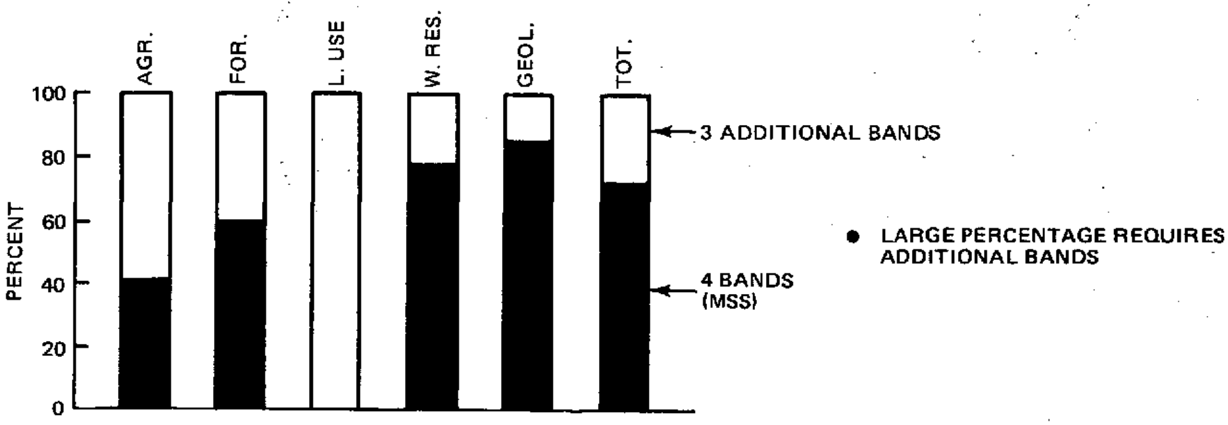
In addition to the technical requirements imposed on the EOS design, Grumman recommends that cost targets be established and that the



a. User Revisit Time Requirements



b. User Resolution Requirements



3-175

c. User Spectral Requirements

7-6

Fig. 1-1 User Requirements Distribution

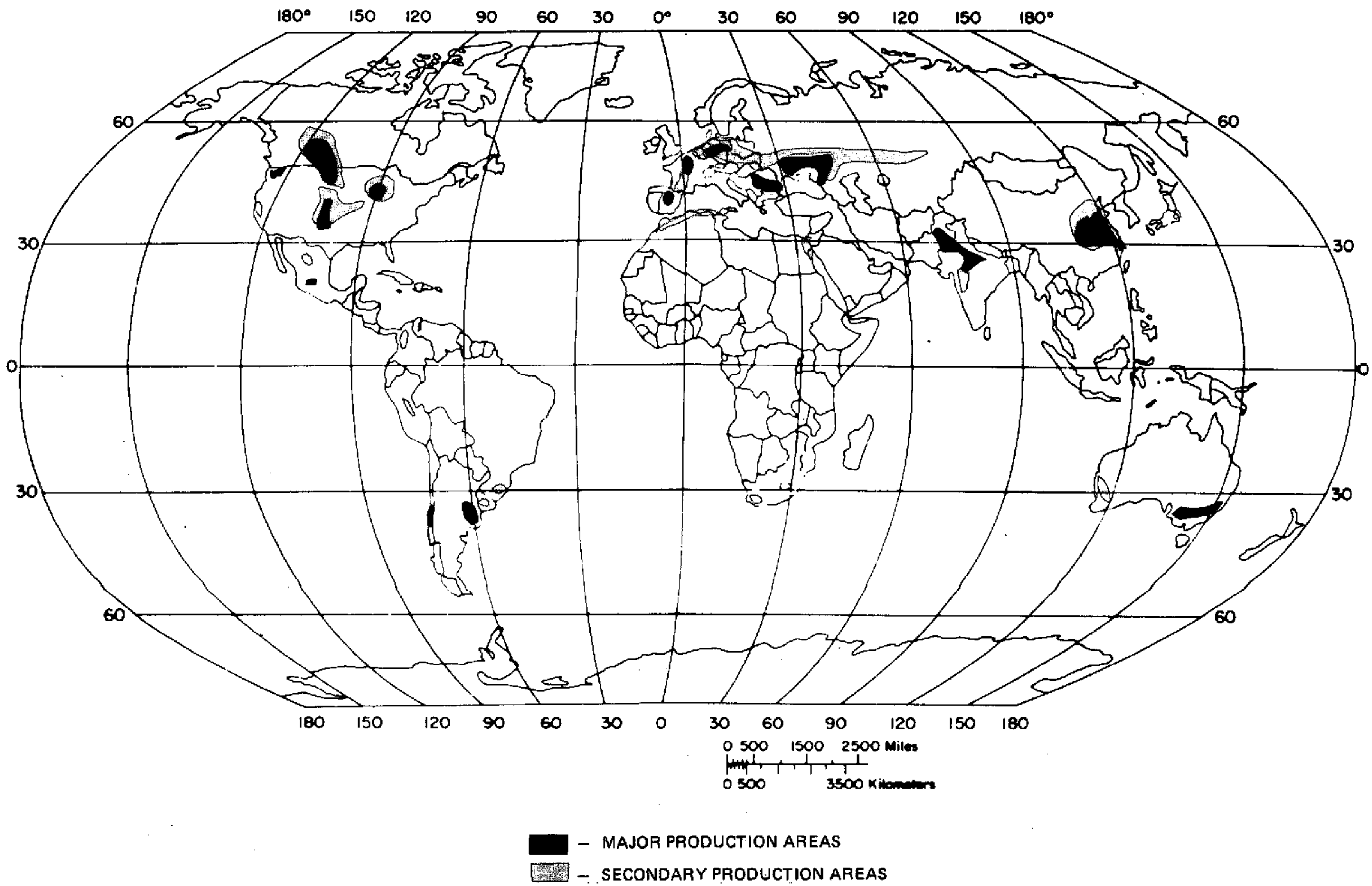


Fig. 1-2 World Wheat Production

Table 1-2 EOS Typical Agricultural Data Acquisition

Wheat (Triticum Vulgare)

MAJOR REGIONS	AVG SCENES/ DAY	DATA ACQUISITION TIME, MIN	CRITICAL MONITORING PERIOD	% OF MONITORING PERIOD WITH CLOUD COVER < 12.5%	> 75%
1. USSR SOUTHERN CENTRAL	13	5.9	MAY-AUG JUNE-SEPT	12	62
2. USA SOUTHERN NORTHERN	10	4.1	MARCH-JUNE JUNE-AUG	39 21	32 31
3. CHINA	7	3.2	MAY-AUG	19	61
4. CANADA	4	1.8	JUNE-SEPT	19	34
5. FRANCE	4	1.8	MAY-AUG	–	–
6. INDIA	4	1.8	NOV-MARCH	55	18
7. ITALY	3	1.4	MAY-AUG	41	32
8. TURKEY	3	1.4	MAY-AUG	–	–
9. AUSTRALIA	3	1.4	SEPT-DEC	30	40
10. ARGENTINA	3	1.4	SEPT-JAN	26	46

7T-30

system be "designed-to" meet these targets. We have two levels in mind: one at the program level for EOS-A and -A′, which constitutes the initial two-spacecraft program; and the second at the Basic Spacecraft level as a recurring cost for future missions. The cost targets and the design-to-cost approach are described in Subsections 1.10 and 1.12 of this summary.

1.3 EOS-A OBSERVATORY DESIGN

Our observatory design, illustrated in Fig. 1-3, is the result of detailed tradeoffs among flexibility, cost, and attainment of program requirements. An important feature of the observatory is that it will simultaneously perform an operational Land Resources mission using the well proven Multi-Spectral Scanner (MSS), and an R&D mission using the new Thematic Mapper (TM). The two functions are completely independent, including on-board data processing and communications. The R&D TM may, however, be used to back up or enhance the operational MSS. This combined operational/R&D approach minimizes program risk while offering large cost savings over separate missions. Key design characteristics are as follows:

- Instrument Complement – Five-band MSS (operational instrument) and seven-band TM (R&D instrument). A new TM has been defined which can provide
  - 30-m resolution
  - Easily expanded swath width from the initially specified 185 km up to 330 km (for a nine-day revisit cycle)
  - Output at 80-m resolution completely compatible with (and providing backup to) the operational MSS
  - An output covering a selectable 35-km swath for a local user (low-bandwidth, high-resolution data)
- Wide-Band Communications for Instrument Data – Ku-Band Tracking and Data Relay Satellite System (TDRSS) link for primary communications, X-Band direct to Satellite Tracking and Data Network (STDN) ground stations as a backup link. Band width is sized for a 240-Mbps data rate in both channels to allow for expansion to a higher data rate instrument complement. A 20-Mbps data link is also provided for communications to Low Cost Ground Stations (LCGS's) at X-Band. A 12.5-ft dish is used for the TDRSS link. This is the same antenna being developed for the TDRS spacecraft, which will save considerable development costs. Two X-Band

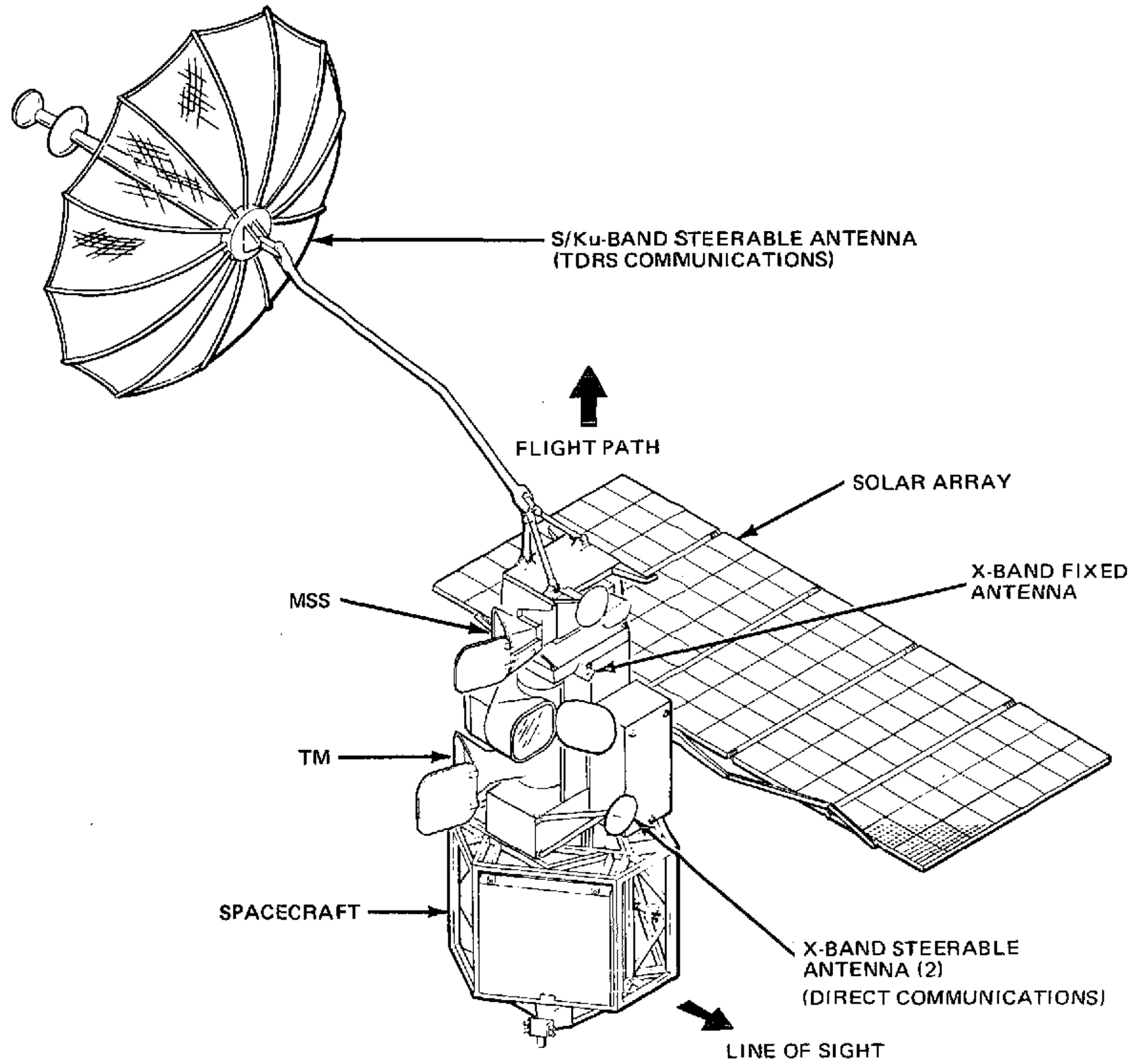


Fig. 1-3 EOS-A (TDRS) Flight Configuration

steerable antennas are used for the LCGS link. A low-gain fixed antenna may also be used for the latter two links. It has an advantage in the case of LCGS in that it provides a 500-km swath coverage. It can, therefore, potentially allow simultaneous communications with multiple local users

- Modularity - All standard spacecraft subsystems, as well as mission peculiar elements such as instruments, wide band communications, and antennas, are modular and easily replaceable on the ground (see Fig. 1-3). In addition, optional latch mechanisms have been designed which will allow in-orbit replacement using the Shuttle's Modular Exchange Mechanism (MEM)

- Orbit Characteristics - Orbital altitude will be in the 365 to 385 n mi range. The specific altitude selected will depend on final choice of swath width and revisit cycle. The orbit will be sun synchronous (near polar) with orbit time of day in the 9:30- to 11:30-a.m. range. The solar array is designed to allow prelaunch selection of sun angle anywhere in this range.

- Orbit Adjust Capability - Expendables are sized to maintain swath overlap at the equator of 20 km for two years

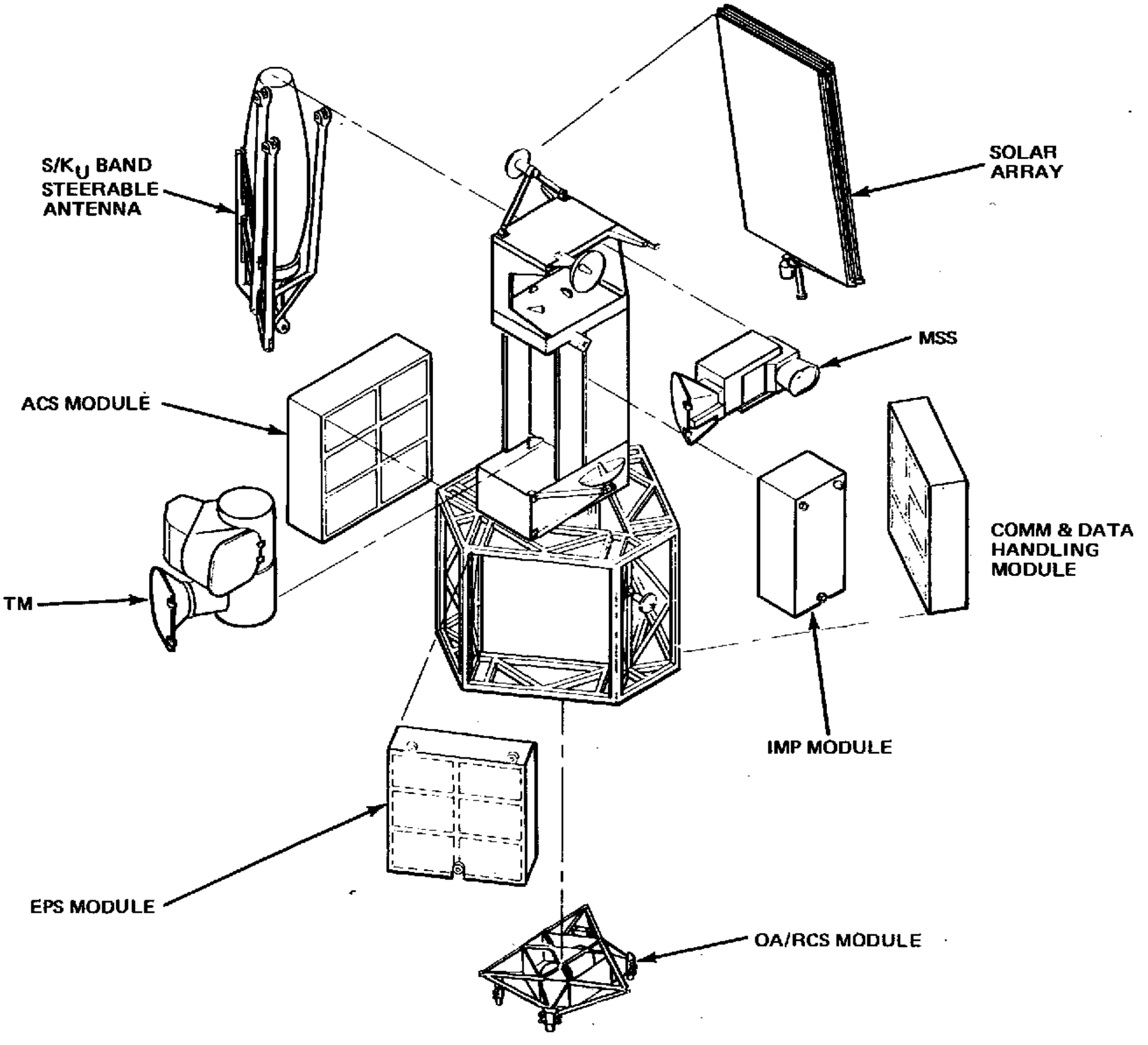
- Weight - Total observatory weight is 2401 lb, including a 202-lb contingency. With this total weight, the observatory can be launched into the required orbit with a Delta 2910. Use of a Delta 2910 saves about $4 million per launch over a Titan III B

- Design Life - The observatory has a Mean Mission Duration (MMD) of two years for normal operation, and a survival life of five years. The latter lifespan will allow for Space Shuttle revisit when it is available for polar orbit launches from WTR (1983 to 1984).

As seen in Fig. 1-4, the lower portion of the observatory is the standard spacecraft, or "bus". The next subsection treats this element of the observatory in more detail.

1.4 STANDARD SPACECRAFT

Since our standard spacecraft is a highly flexible "bus", it is useable for a wide range of earth orbiting missions as well as launch vehicles. The spacecraft design, illustrated in Fig. 1-5, utilizes 75 to 80% existing hardware, much of which has many years of flight experience on spacecraft such as OAO, OSO, and military satellites. Thus we are able to achieve an observatory-class spacecraft at low risk and low cost that will bridge the transition from the expendable launch vehicles of the 1970's to Space Shuttle in the 1980's.



7-4

Fig. 1-4 EOS-A (TDRS) Resupply Option

Key characteristics of the standard spacecraft are as follows:

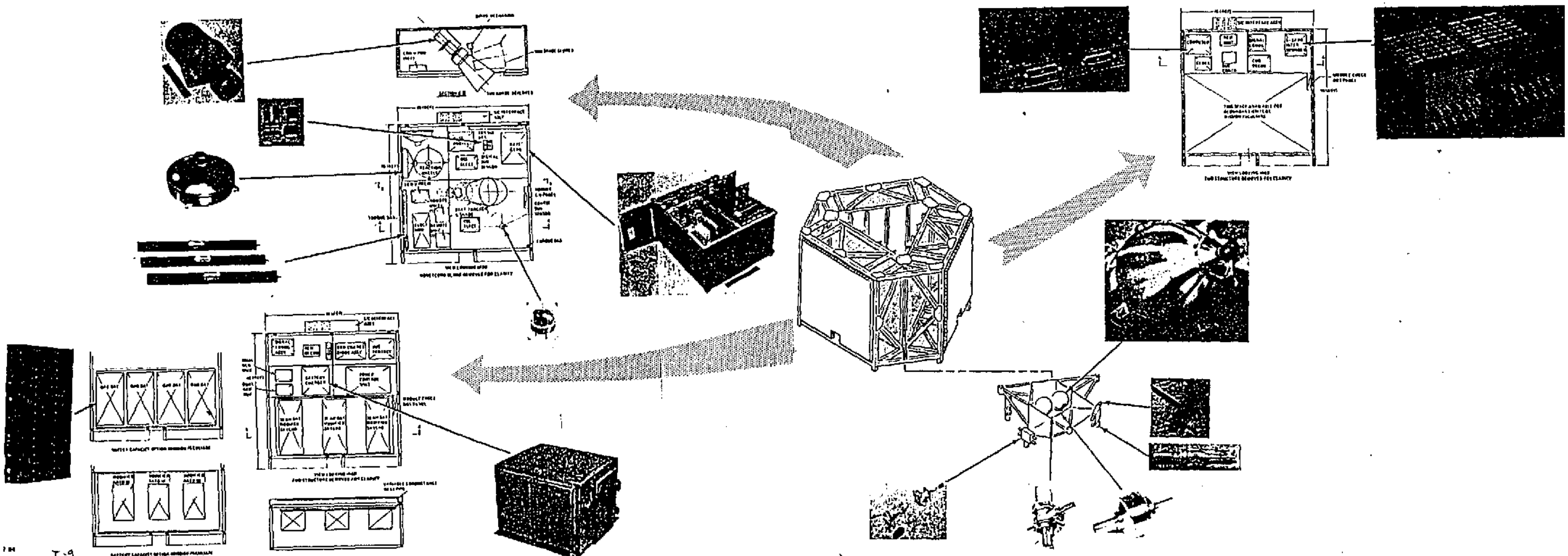
- Weight – 1361 lb, including a 146-lb contingency, for the Basic Spacecraft
- Payload Capability – Limited by the delivery system capability: i.e., for Delta - 1000 lb; for Titan III B - 4000 lb; for Shuttle - 24,000 lb (requires mission peculiar reaction wheel/torquer bar option). This gives a ratio of spacecraft to payload weight significantly greater than existing satellite systems
- Launch Vehicles – Can be launched on Delta 2910 or 3910, Atlas-F, any of the Titan class, and Space Shuttle. This allows selection of the most cost-effective launch vehicle for a particular payload and orbit
- Shuttle Utilization – For Shuttle deployment or retrieval, a segmented transition ring has been designed for easy addition to the spacecraft (27 lb added weight). Similarly, optional latch mechanisms are designed for in-orbit resupply of all subsystems using the MEM (52 lb added weight)
- Modular Subsystems – The attitude control subsystem (ACS), electrical power subsystem (EPS), communications and data handling (CDH), and orbit adjust/reaction control subsystems (OAS/RCS) each comprise a removable module. They are thermally and structurally independent, thus allowing the option of eliminating full-up spacecraft thermal acceptance tests. The ACS, EPS, and CDH modules are 48 x 18 in., and have built-in provisions for adding the latching mechanisms with negligible scar weight
- ACS Performance – Attitude pointing accuracy is 0.01 deg, stability is 10^{-6} deg/sec. This performance is with a built-in, fixed-head star tracker. The ACS will also accept pointing error signals directly from a payload sensor, such as a telescope, or a gimballed star tracker for greater accuracy. A combination of RCS thrusters, momentum wheels, and magnetic torquer bars are used for momentum exchange and removal
- Electrical Power – Available orbital average power is 1500 watts; peak power, up to 3500 watts. With the addition of battery chargers and batteries in conjunction with additional solar array area, this capability can be doubled. A rigid solar array is used in the basic design, but a flexible array can be utilized for weight or packaging advantages
- Orbit Adjust/RCS – Provisions for attitude control during initial stabilization and satisfying wheel unloading requirements is contained in the all hydrazine replaceable propulsion module. In addition, requirements for orbit adjust capability and orbit transfer stabilization can be met by mounting additional jets and tanks as required. Thruster size requirements from 0.1, 1, 5, and 75 lb have been defined for the various functions.

In summary, the standard spacecraft designed for EOS can handle most requirements for the earth orbiting missions in its basic form, and is further expandable for either added performance or redundancy with minor predesigned additions. Follow-on mission capability is discussed in more detail in the next subsection.

1.5 FOLLOW-ON MISSION ACCOMMODATION

The Grumman Basic Spacecraft design resulting from our EOS System Definition Study incorporates the concept of subsystem modularity. The modules which we have designed for this spacecraft "bus" can be used without geometrical variation on missions other than EOS-A. In fact, our studies indicate that a large number of diverse payloads can be captured by the Basic Spacecraft with just minor modification to the individual subsystem hardware complement.

Figure 1-6 shows the family of payloads investigated during the course of the study and the observatory configuration which resulted. The complement of follow-on missions investigated were SEASAT, SMM, EOS-C, TIROS-O, SEOS and EGRET. These encompass requirements for earth pointing (SEASAT, EOS-C, TIROS-O), solar pointing (SMM), geosynchronous earth pointing (SEOS) and inertial pointing (EGRET) spacecraft. As the figure shows, even though the individual missions requirements differed greatly a consistent geometrical arrangement for the Basic Spacecraft was maintained for all missions.



I-9
FOLDOUT 1

ORIGINAL PAGE IS
OF POOR QUALITY

I-9-a FOLDOUT FRAME 2

I-9-b

FOLDOUT FRAME 3

I-9-c

1-9

Fig. 1-5 Basic Spacecraft

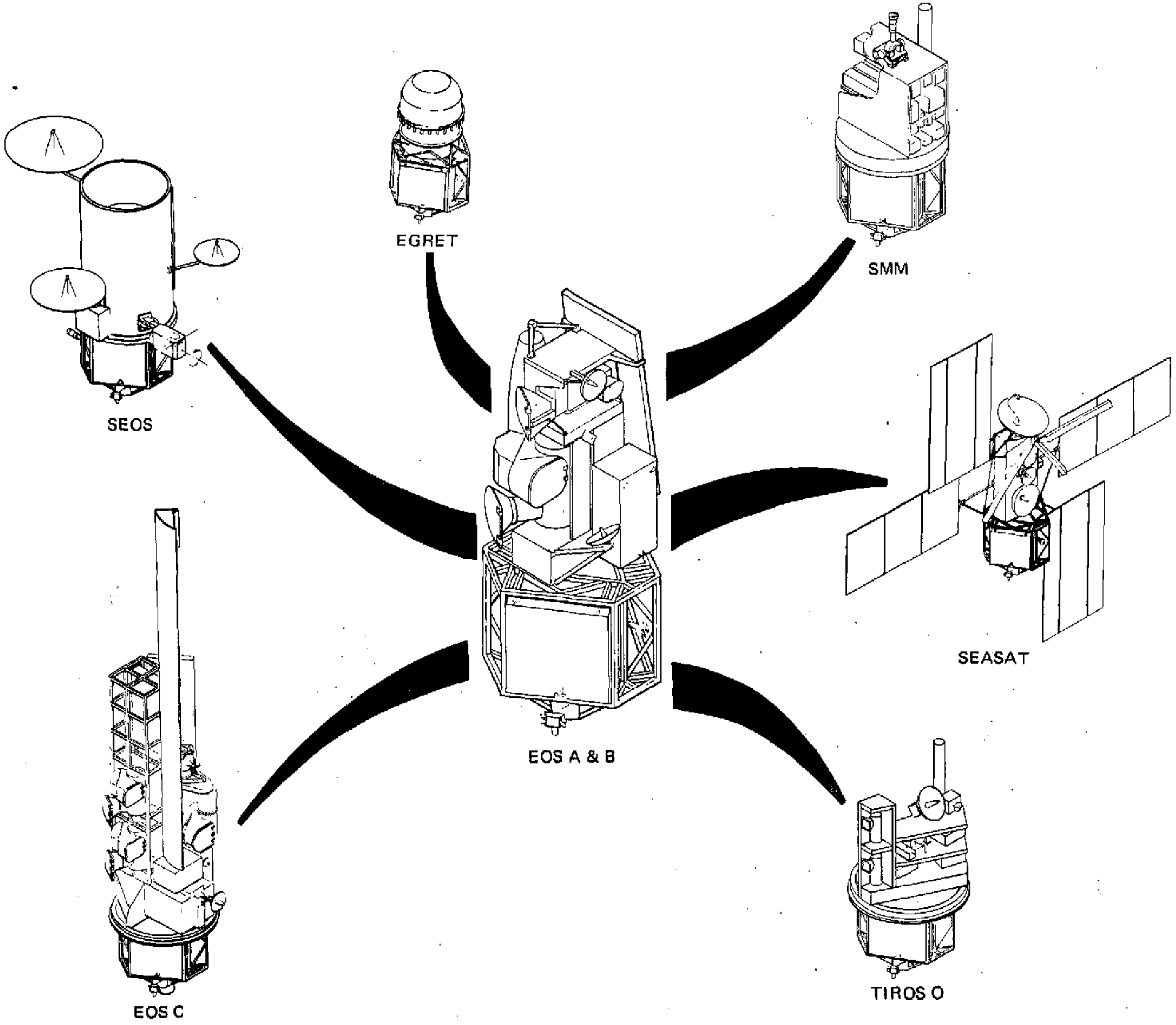


Fig. 1-6 Family of Standard Modular Spacecraft

Table 1-3 Standard Spacecraft Mission Requirements Summary

MISSION	ORBIT, N MI	L/V	PAYLOAD WT, LB	POWER, WATTS	DATA RATE, MBS	POINTING ACCURACY, DEG	POINTING STABILITY, DEG/SEC X 10^{-6}
EOS-A	365-385*	D2910	915	209	102	0.01	1
EOS-B	365-385*	D3910	1210	225	170	0.01	1
EOS-C	365-385*	T-III B	2340	450	350	0.01	1
EOS-D (SEASAT-B)	324 (90°)	D2910	973	450	90	0.2	10
EOS-E (TIROS-O)	450*	D3910	1037	268	1.1	0.01	2
EOS-F (SEOS)	19000 GEOSYNCH 0 LAT	T-III-C7/ TE364-4	2849	425	60	0.0016 (IN AUTO MODE)	0.46
SEASAT-A	432 108°	D3910	927	550	36	0.25	10
SMM	275-300 (28-33°)	D2910	1973	174	$5x10^{-3}$	.0003 (W/SUN SENSOR)	4.0**
EGRET	250 (28°)	D2910	2695	55	$3x10^{-3}$	0.1	–

7T-9

*SUN-SYNCHRONOUS
**ALSO REQUIRES 16 MIN SLEW IN 8 SEC

Table 1-4 Follow-On Mission Driver Requirements

SUBSYSTEM	REQUIREMENTS	FOLLOW-ON MISSION
• C&DH	– INCLUDE TAPE RECORDER – INTERFACE WITH 32 REMOTES – MEMORY EXPANDABLE TO 65K WORDS	– SEASAT – SMM – SMM/SEASAT
• EPS	– 600 W ORBIT AVERAGE NON-SUN SYNCHRONOUS RETROGRADE – 2KW PEAK WITH 25% DUTY CYCLE – TWO-AXIS ARRAY DRIVE	– SEASAT/EOS-C – SEASAT – SEASAT
• ACS	– POINTING 5.0 SEC – HOLDING .0017 SEC/SEC – SLEWING 16 MIN. IN 8 SEC	– SEOS/SMM – SEOS – SMM
• STRUCTURE	– SUPPORT OF AT LEAST 2500 LB OF PAYLOAD	– EOS-C
• THERMAL	– INCORPORATION OF VCHP & OSR INTO MODULE DESIGN*	– SEOS/SEASAT/SMM
• RCS/OA/OMS	– PROPELLANT FOR 100% WHEEL UNLOADING – THRUST VECTOR CONTROL CAPABILITY – ADDITIONAL KICK MOTORS & SUPPORT	– SEOS – TIROS-O/EOS-C – TIROS-O
• INST. DATA/W.B. COMM	– SUPPORT 300 MBPS – INCLUDE ONE OR MORE HI SPEED RECORDERS	– EOS C – EOS C

*VCHP = VARIABLE CONDUCTANCE HEAT PIPE
OSR = OPTICAL SOLAR REFLECTOR

7T-49

The major mission requirements for the application and science missions discussed above are given in Table 1-3. The table indicates that the requirements for most of the missions are within the range of the basic spacecraft capability. However, there are areas in which missions other than EOS-A are the design drivers. A summary of these driver requirements is given in Table 1-4. The table indicates (for each subsystem) the driver requirements beyond the EOS-A basic spacecraft capability, and which mission imposes these additions. Note that the final two requirements for the C&DH subsystem are the current expansion limits of the Basic Spacecraft.

While some of the indicated driver requirements significantly exceed the basic spacecraft capability, enough flexibility has been designed into the Basic Spacecraft so that all these requirements may be satisfied without compromising the subsystem module external geometric, electrical, or data interfaces. In fact, in most cases these follow-on driver requirements can be satisfied by incorporating production line changes in the module (e.g., the addition of a battery; use of larger size reaction wheels).

1.6 SPACE SHUTTLE UTILIZATION

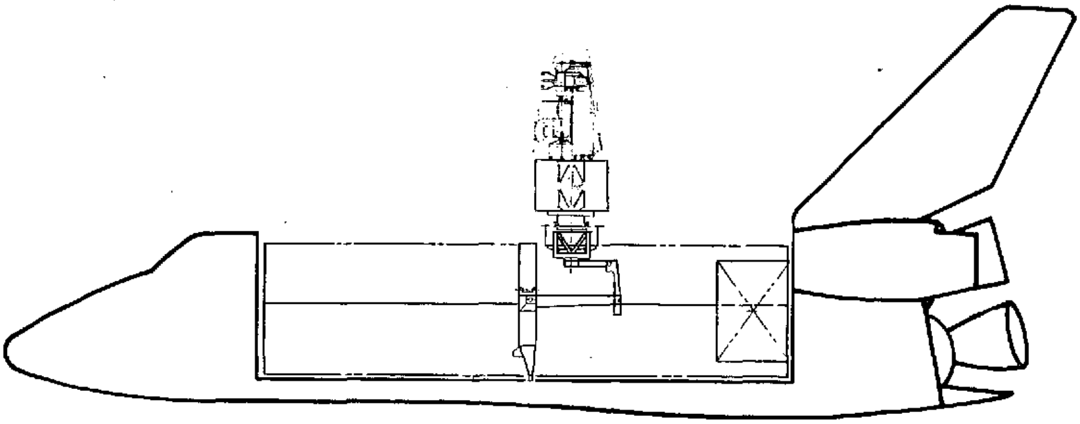
The most economically beneficial way of using Shuttle to maintain a long-term operational LRM mission is for In-orbit Resupply of EOS.

Three fundamental questions were addressed in the course of this study:

- What is the design impact associated with the Space Shuttle?
- Is the EOS compatible with Shuttle performance capabilities?
- What is the best mode relative to Shuttle use to obtain maximum cost and operational benefits (i.e., Delivery Only, Delivery plus Retrieve, or In-orbit Resupply)?

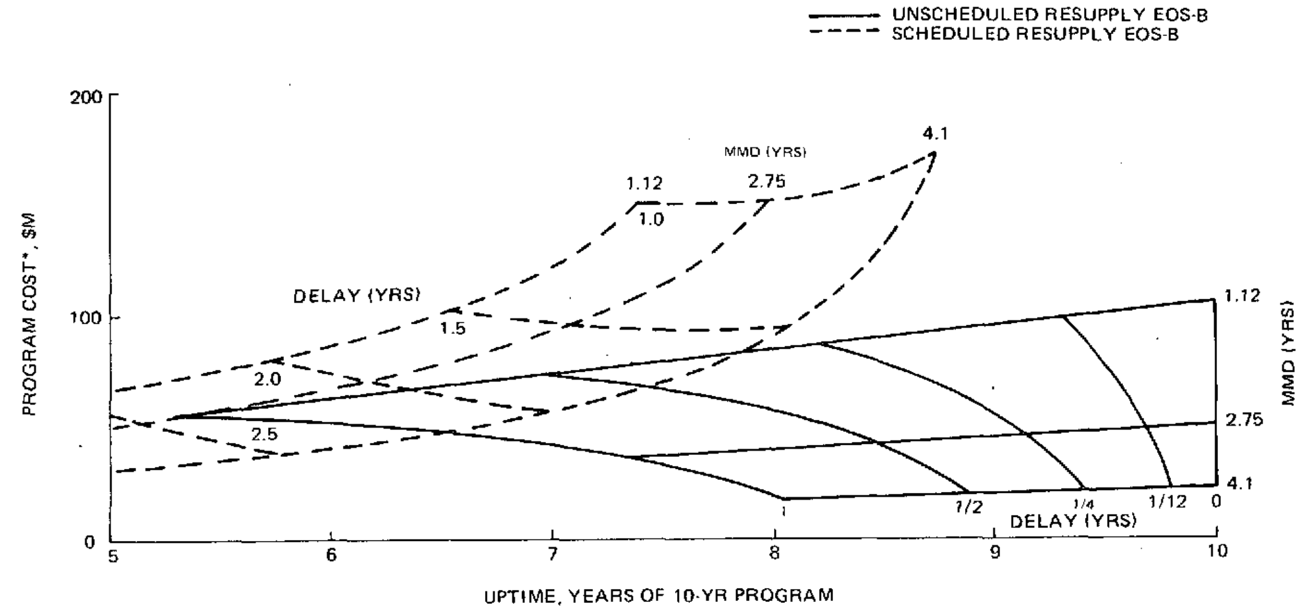
Design impact and Shuttle performance were investigated for EOS missions A through F (Table 1-1). Shuttle utilization benefits were studied for EOS-B (Fig. 1-7) and EOS-C, which represent two classes of long-term operational spacecraft. These investigations led to the following conclusions:

- Observatory weight impacts, exclusive of orbit transfer subsystem (OTS) considerations, are reasonable
  - 60 to 70 lb for Delivery Only
  - 70 to 80 lb for Deliver/Retrieve
  - 200 to 300 lb for In-orbit Resupply
- EOS program cost impact (non-recurring/recurring) to achieve Shuttle compatibility are minimal compared to total program cost for any projected Shuttle utilization mode.
  - \$0.4/\$0.5 million for Deliver Only
  - \$2.2/\$0.9 million for Deliver/Retrieve
  - \$4.4/\$1.3 million for In-orbit Resupply



7-9

Fig. 1-7 EOS Deployment

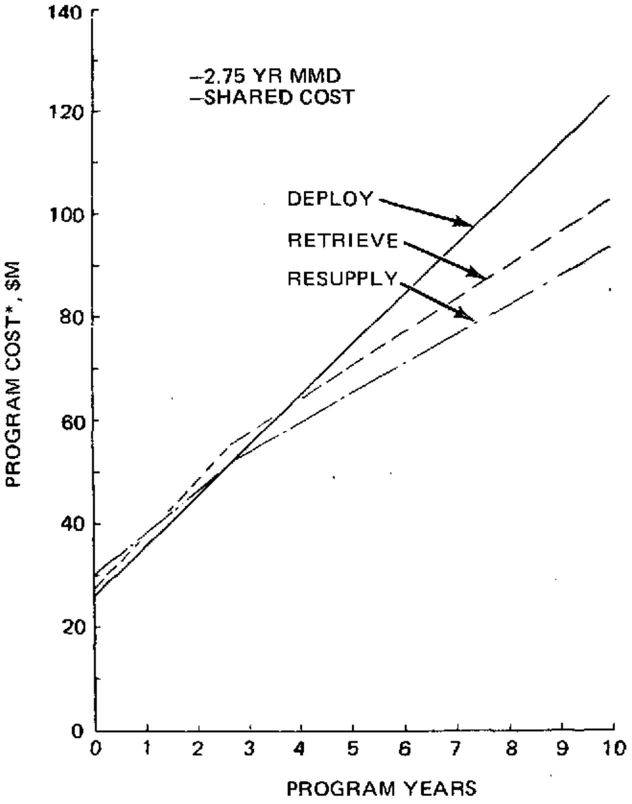


*Variable costs only; fixed costs not included

Fig. 1-8 Scheduled Vs Unscheduled Resupply

6-1
7-10

- Shuttle performance, in conjunction with the EOS OTS is adequate for all EOS mission concepts except SEOS, which requires a Tug
- All EOS configurations studied, including the necessary support and resupply equipment, meet Shuttle volume and center-of-gravity constraints
- High EOS subsystem and instrument redundancy is cost effective compared to total program costs in all Shuttle utilization modes (Fig. 1-8)
- For all EOS programs entailing on-orbit operating lifetimes in excess of two to three years, Resupply is the preferred Shuttle utilization mode. For shorter duration programs, Deliver Only is preferred (Fig. 1-9)
  - High-cost, high-weight payloads magnify the desirability of resupply for long-term operational programs
  - Resupply cost benefits can be greatly increased by reducing resupply system (i.e., MEM and module magazine) weight, assuming shared Shuttle transportation costs
  - Shuttle flights should be initiated on demand of a disabled spacecraft in all modes, rather than on a regularly scheduled basis (Fig. 1-8)



*Variable costs only; fixed costs not included

6-42
7-11

Fig. 1-9 Total Program Costs, EOS-B

– Proportional Shuttle transportation costs (multiple user) favor low Shuttle parking orbit plus EOS OTS.

In addition to the foregoing conclusions, our studies show that selection of a Shuttle operating orbit has a greater influence on EOS transportation costs. Direct Shuttle ascent to the required EOS mission orbit may eliminate the need for an OTS, but significantly increases operational costs.

1.7 CENTRAL DATA PROCESSING FACILITY

The guidelines used in our Central Data Processing Facility (CDPF) design study encompassed the volume of remote sensor data to be processed, and the quantity, type, and quality of the output products. These are listed in Tables 1-5 through 1-7. Our study has led to the following conclusions:

- The high data load, high quality requirements of the output data, and the fast turnaround demanded for the output products associated with the next generation remote sensing system can be met with currently known computer technology. However, the user community is still unsure of the processing algorithms required of the CDPF for maximum data user benefit. Thus, we are faced with a software and system problem rather than a hardware problem
- The CDPF should be configured with a capability to handle 20 TM scenes per day in the 1979 timeframe, and be capable of growth for the processing of 400 scenes/day in three to five years. This growth can and should be accomplished in a modular, add-on fashion
- Current data processing facilities are incapable of growing to meet the next generation requirements
- Due to the R&D nature of the processing algorithms, the CDPF should initially be software flexible
- While a large general purpose computer system offers software flexibility, any CDPF configuration of this nature becomes prohibitively expensive at high data volumes. A system consisting of distributed minicomputers offers flexibility for the initial CDPF at reasonable cost
- Special purpose hardware, while fast and relatively inexpensive, does not offer the flexibility required of the early CDPF
- Use of an associative array processor (AAP), such as the STARAN, offers flexibility, modularity for growth, and costs comparable to special purpose hardware. This approach appears eminently suitable for the CDPF
- The initial cost of the CDPF will be $10 to 12 million (in 1974 dollars)
- A Local User System (LUS), consisting of a network of low-cost local or regional receiving and data processing stations, should tie in either directly or indirectly with the CDPF. These distributed systems will promote greater efficiency in sharing the total data processing load as well as assure more expeditious data dissemination.

Table 1-5 Output/Input Product/Data Quantity

PRODUCT	DATA VOLUME	NUMBER OF DATA USERS	NUMBER OF FORMATS
HDDT (UNCORRECTED)	10^{10} – 10^{12} BITS/DAY	2 – 10	–
HDDT (CORRECTED)	10^{10} – 10^{12} BITS/DAY	2 – 10	–
CCT (CORRECTED)	10^{9} – 10^{10} BITS/DAY	10 – 100	1 – 1
BLACK&WHITE POS/NEG(1)	20 – 200 SCENES/DAY	5 – 50	1 – 3(3)
BLACK&WHITE PRINTS		5 – 10	1 – 3(3)
COLOR POS/NEG(2)	10 – 100 SCENES/DAY	2 – 20	1 – 3(3)
COLOR PRINTS		2 – 10	1 – 3(3)

(1) First generation product – 24mm (9.5 in.)

(2) Second generation product – 24mm (9.5 in.)

(3) Enlargement to standard map scales

(4) Processing considered as two 8-hr shifts per day

7T-47A

Table 1-6 Output Products

PRODUCT	GEOMETRICALLY UNCORRECTED	GEOMETRICALLY CORRECTED	REDUCED DATA OPTIONS
B/W FILM	✓		
COLOR FILM	✓		
HIGH DENSITY DIGITAL TAPE	✓	✓	
COMPUTER COMPATABLE TAPE		✓	✓
CUSTOM OUTPUT PRODUCTS	FILM PRODUCTS GEOMETRICALLY CORRECTED WITH CUSTOM GAMMA CAPABILITY		
SUBAREA ENLARGEMENTS	SPECIFIC MAP SCALES (e.g. 1:1,000,000 1:500,000)		
CUSTOM FILM	SPECIFIC FALSE COLOR		
CUSTOM DIGITAL PRODUCTS	CCT OUTPUTS WITH VARIOUS FORMATS (e.g., BAND INTERLEAVED, BAND SEQUENTIAL) AND SUBAREAS (e.g., PARTIAL SCENES)		

- Output product quality: as indicated in Table 1-7
- Output/input product/data quantity: as indicated in Table 1-5

7T-45A

Table 1-7 Output Product Quality

PRODUCT	GEOMETRICALLY UNCORRECTED(1)		GEOMETRICALLY CORRECTED(2)	
	TM	HRPI	TM	HRPI
• SWATH WIDTH, KM	185	48	185	48
• SPATIAL RESOLUTION				
– VISIBLE, M	30	10	30	10
– THERMAL, M	120	–	120	–
• LINEARITY (μ RAD) IFOV	0.2	0.2	0.2	0.2
• BAND TO BAND REGISTRATION (μ RAD) IFOV	0.1	0.3	0.1	0.3
• POSITION ACCURACY (W/O GCP),(3) M	± 450	± 450	± 170	± 170
• POSITION ACCURACY (WITH GCP,(3) M	–	–	± 15	± 15
• RELATIVE RADIOMETRIC ACCURACY				
– VISIBLE				
o TAPE, %	± 1.6	± 1.6	± 1.6	± 1.6
o FILM, %	± 5	± 5	± 5	± 5
– THERMAL				
o TAPE, K	± 1	–	± 1	–
o FILM, K	± 3	–	± 3	–

NOTES:

(1) Includes radiometric correction, earth-rotation correction, line-length adjustment, correction for earth curvature, and predicted emphemeris.

(2) Additionally includes use of best-fit ephemeris from measured data.

(3) GCP = ground control points.

7T-46A

Our CDPF design study has established the most cost effective configuration for performing:

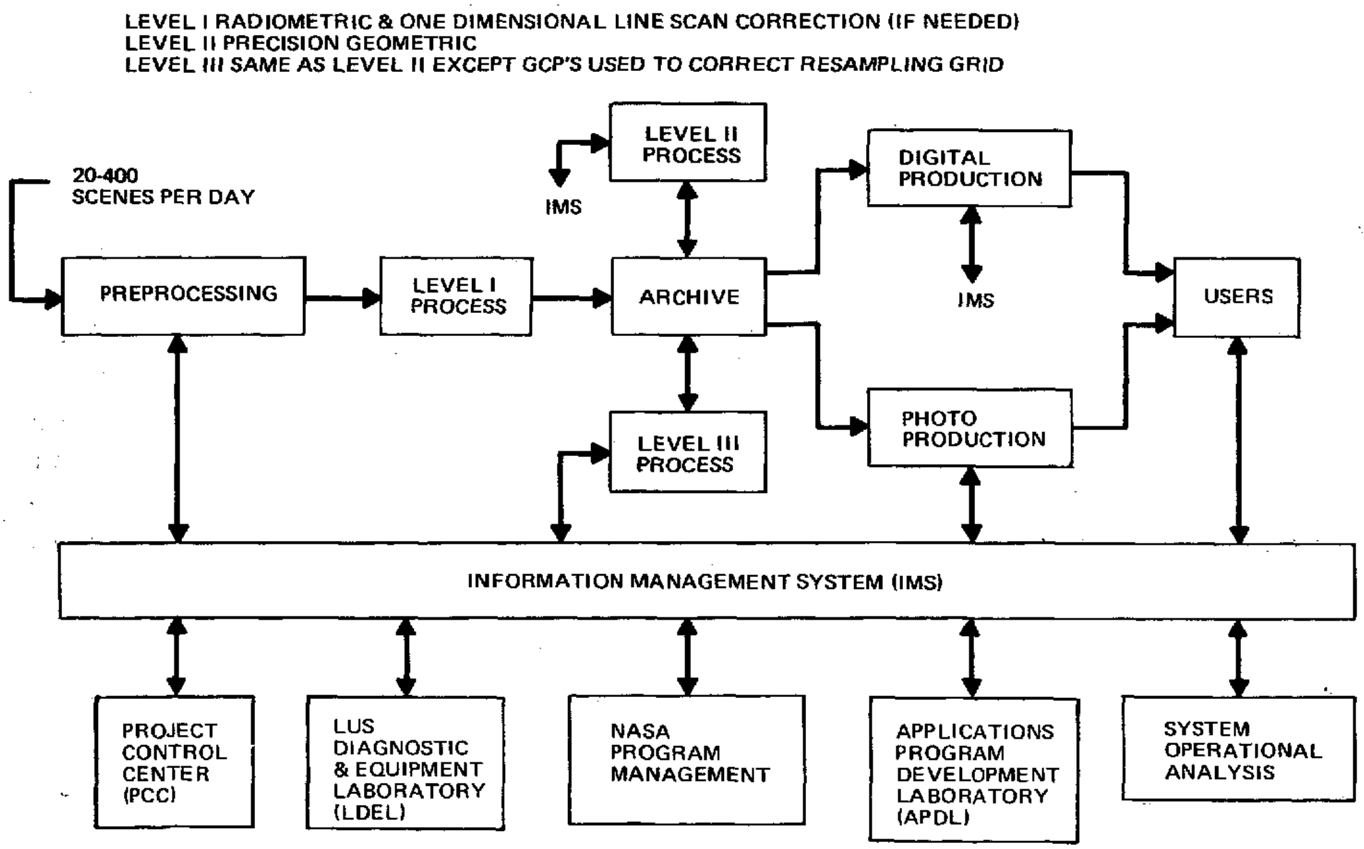
- Radiometric and geometric correction of raw data and
- Generation of specified output data product.

Figure 1-10 illustrates the general elements of such a facility. The interrelationships among the major CDPF functions are shown in Fig. 1-11.

PROCESSING - Three levels (stages) of processing have been considered:

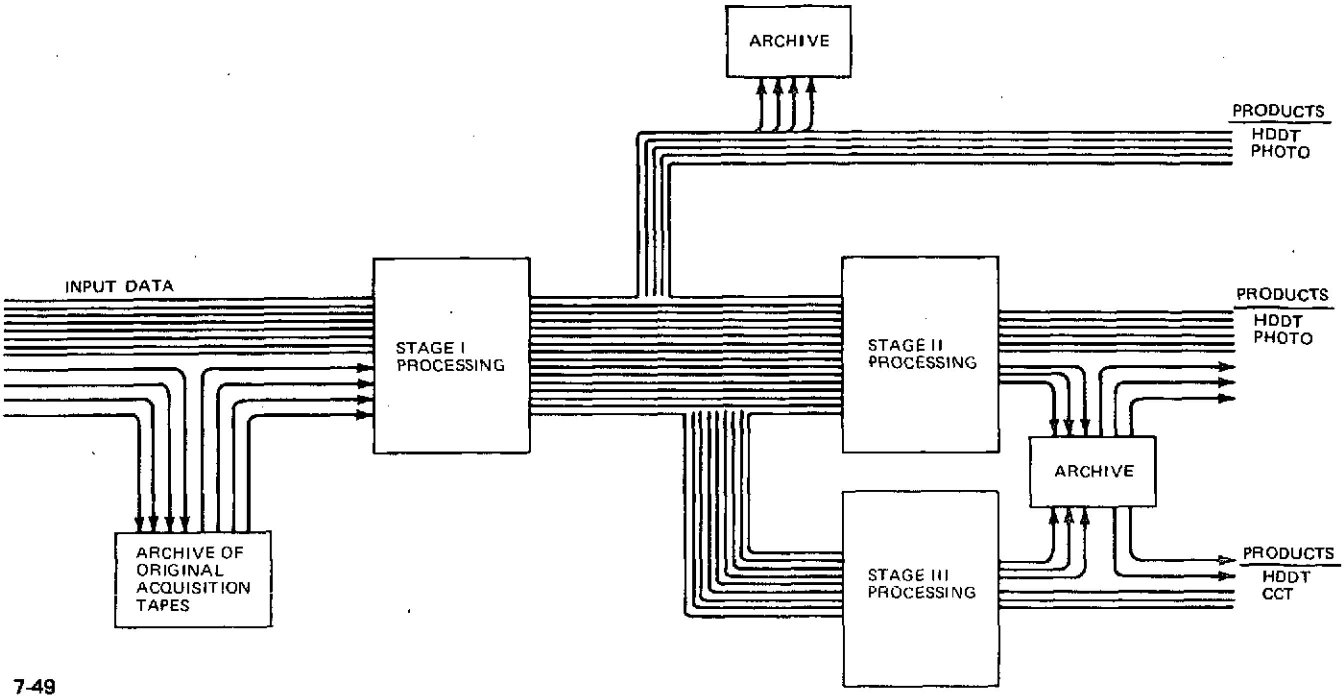
- Stage I - Calibration-type corrections using the calibration data provided with the image data. Included is radiometric correction plus any one-dimensional scan correction (line stretching) required by the particular scanner selected
- Stage II - Correction for earth curvature, earth rate, UTM projection, and two-dimensional sensor scan correction (e.g., correct for conical scan), using the best available estimates of attitude and ephemeris
- Stage III - Further refinement of the corrections made in Stage II by using GCP's to improve attitude and ephemeris data. Level III processing would be performed on a certain fraction of the data instead of Level II processing.

COST/THRUPUT/ALGORITHM/SCAN TECHNIQUE - 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, and with standard machines (e.g., minicomputers), costs increase linearly with scene load, Fig. 1-12.



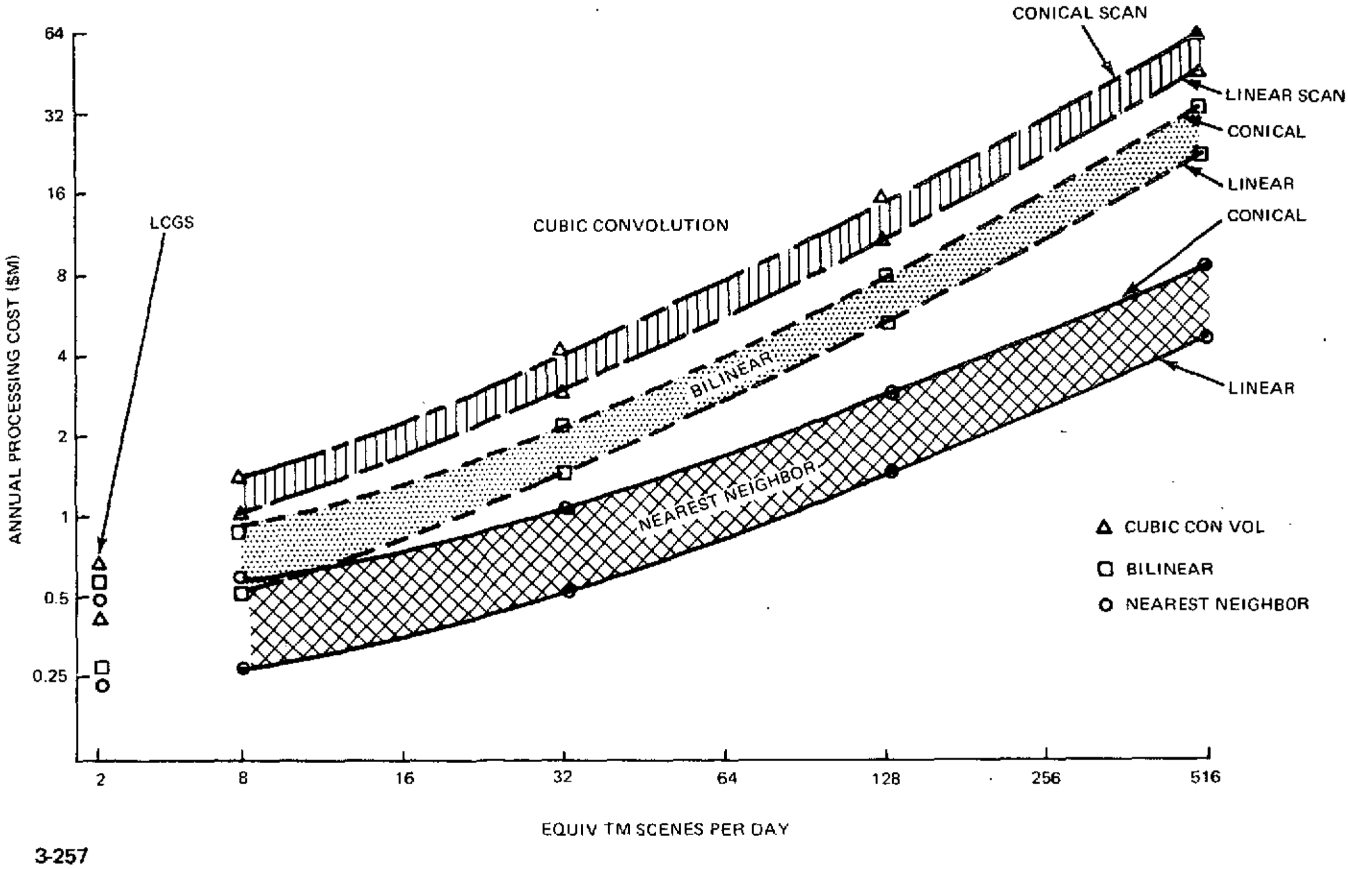
7-48
7-102

Fig. 1-10 General Structure of the Central Data Processing Facility



7-49
7-103

Fig. 1-11 Processing, Archival and Product Generation Functions Within the DMS



3-257
7-50

Fig. 1-12 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.

OUTPUT PRODUCTS - Figure 1-11 also shows the requirement for output (user) products at three points:

- Stage I: HDDT and Photo
- Stage II: HDDT and Photo
- Stage III: HDDT and CCT

Tables 1-5 through 1-7 relate the quantity and quality of these products. High density digital tape (HDDT) refers to any very high density tape (>10,000 bpi) which is not directly readable by a computer without special interface hardware. Computer compatible tape (CCT) refers to other magnetic tapes with density <10,000 bpi that are directly readable by computers. The photo products consist of black and white (B&W) film (positive and negative), B&W prints, color film (positive and negative) and color prints. The B&W and color film are to be 241 mm (9.5 in.). The required data load must be handled in a standard 16-hr day. This implies a 24-hr turnaround for most standing orders.

CDPF CONFIGURATION - Three concepts were considered: Use of a configuration

- Of multiple minicomputers
- Containing special-purpose (SP) digital hardware
- Centered around an AAP; specifically, STARAN.

The major cost drivers in all three approaches are the data handling/formatting/storage and the interpolation. The critical nature of the data handling/storage is driven by the enormous quantity of data in a TM scene, the processing speed requirements, and the fact that the output scan lines are tilted with respect to the input scan lines.

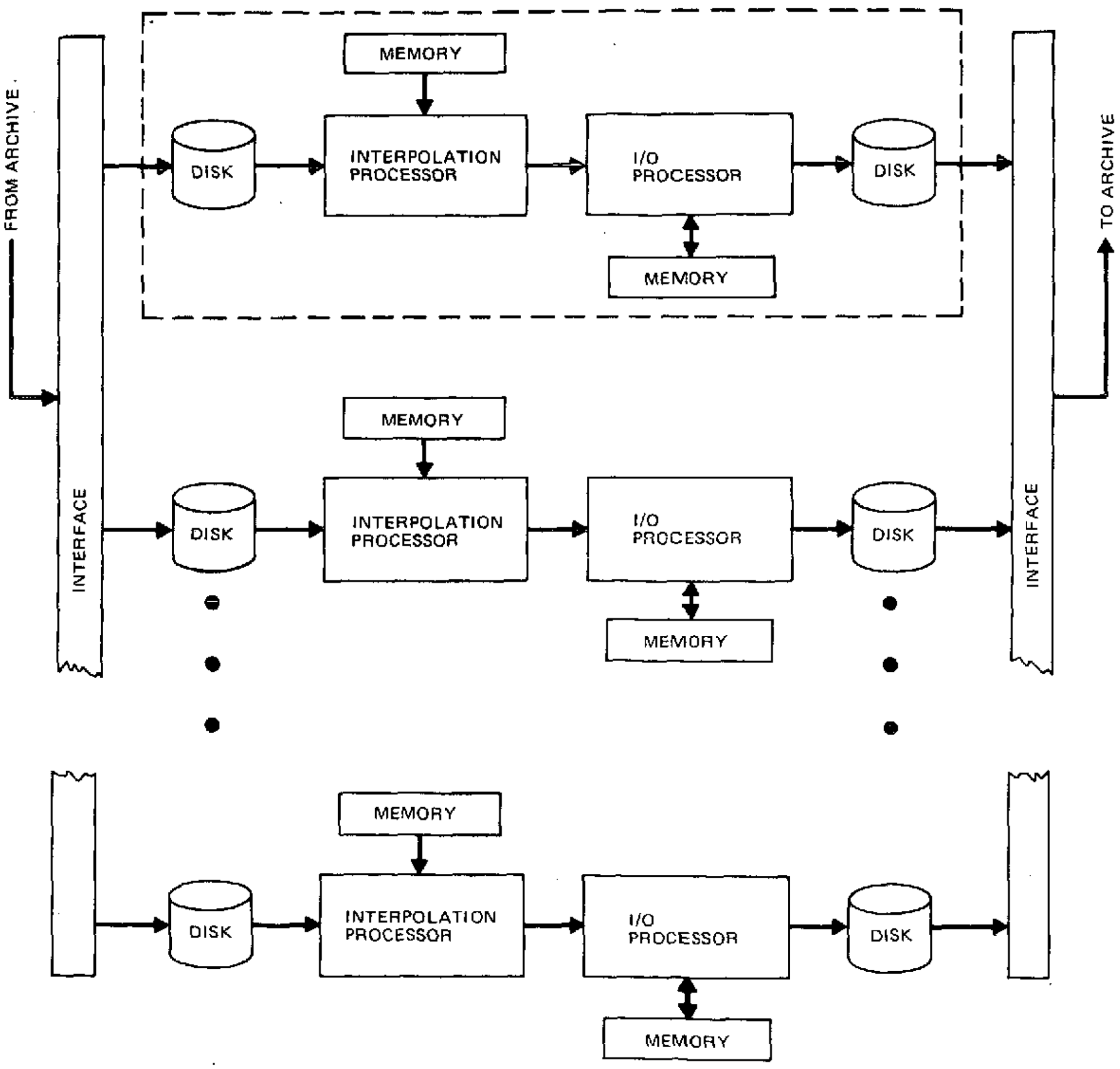
Figure 1-13, A through C show, respectively, the three alternative configuration concepts (Options A through C) for Level II/III processing. The basic module of Option A (minicomputer system) uses two processors, one to perform the interpolation and the other to handle the data. The basic module can process five TM scenes per day assuming bilinear interpolation. Four modules can process 20 TM scenes per day. To process 400 TM scenes per day using cubic convolution, 240 modules would be required.

Option B is the special purpose hardware configuration. Minicomputers will be used to implement the master process control and the grid computation, but the remainder of the system consists of hard-wired, special-purpose hardware. Interpolation algorithms are switch selectable and are limited to the three methods (nearest neighbor, bilinear, and cubic convolution) considered in the analysis.

The expansion of the special purpose hardware configuration from a minimum throughput version to a 400-scene-per-day system occurs in several stages. A basic single-thread module can handle 15 scenes per day. By doubling the disk, and then the image segment memory elements, the throughput of the module can be increased to 30, then 60, scenes per day. The next stage of expansion is to increase the number of modules. A total of seven modules is required to handle 400 scenes per day.

Option C is based on an unconventional general purpose processor, the Goodyear STARAN. The STARAN AAP is a general purpose computer with special architecture oriented toward the common manipulation of tabular data. The STARAN processor operates in a multiphased batch mode. It performs coordinate computation, interpolation, GCP location, and portions of the data handling computation for batches of output pixels. An 11-array system can handle 400 scenes per day, assuming cubic convolution interpolation. For the 20-per-day system, the usual minimum configuration of two arrays is recommended.

Table 1-8 shows a summary of the characteristics of the implementation options. Because Option C provides the flexibility of a general purpose system at a cost comparable to that of special purpose hardware, Option C is the recommended approach.



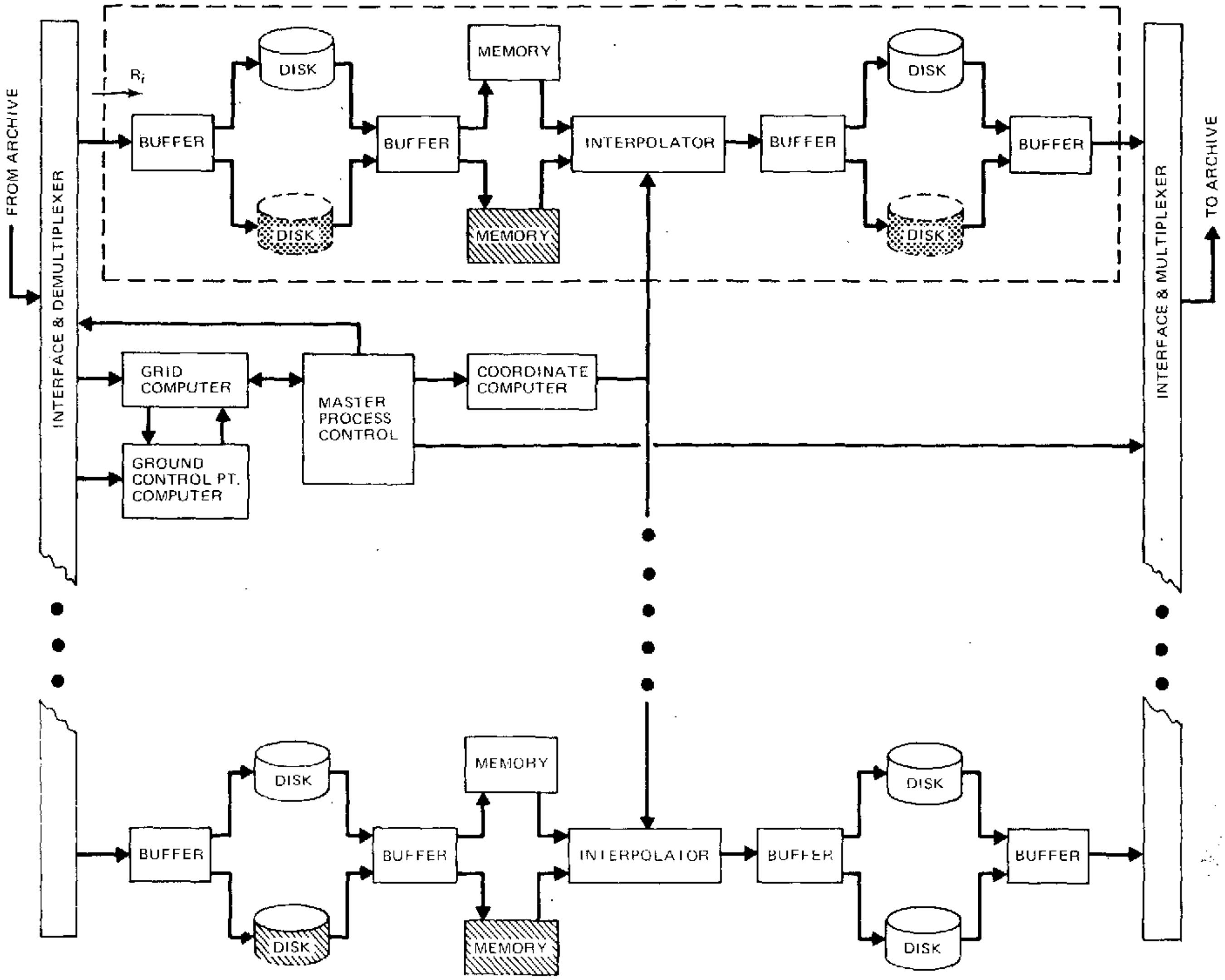
7-51, 7-58
7-104

- BASIC MODULE PROCESSES 5 SCENES/DAY
- FOUR BASIC MODULES PROCESS 4×10^{10} BITS (20 TM SCENES) PER DAY

A. Option A – Minicomputer Configuration

Fig. 1-13 Level II/III Processing Alternative Configuration Concepts (Part 1 of 3 Parts)

7



- HANDLES UP TO 12 ROTATION BETWEEN INPUT AND OUTPUT DATA
- PERFORMS RESAMPLING, INTERPOLATION, AND PRODUCES ONE OF 3 OUTPUT FORMATS

7-52, 7-59
7-105

- BASIC MODULE SIZED TO PROCESS:

  15 SCENES/DAY

  ADD EXTRA MEMORY PROCESSES 30 SCENES/DAY

  ADD EXTRA DISKS PROCESSES 60 SCENES/DAY

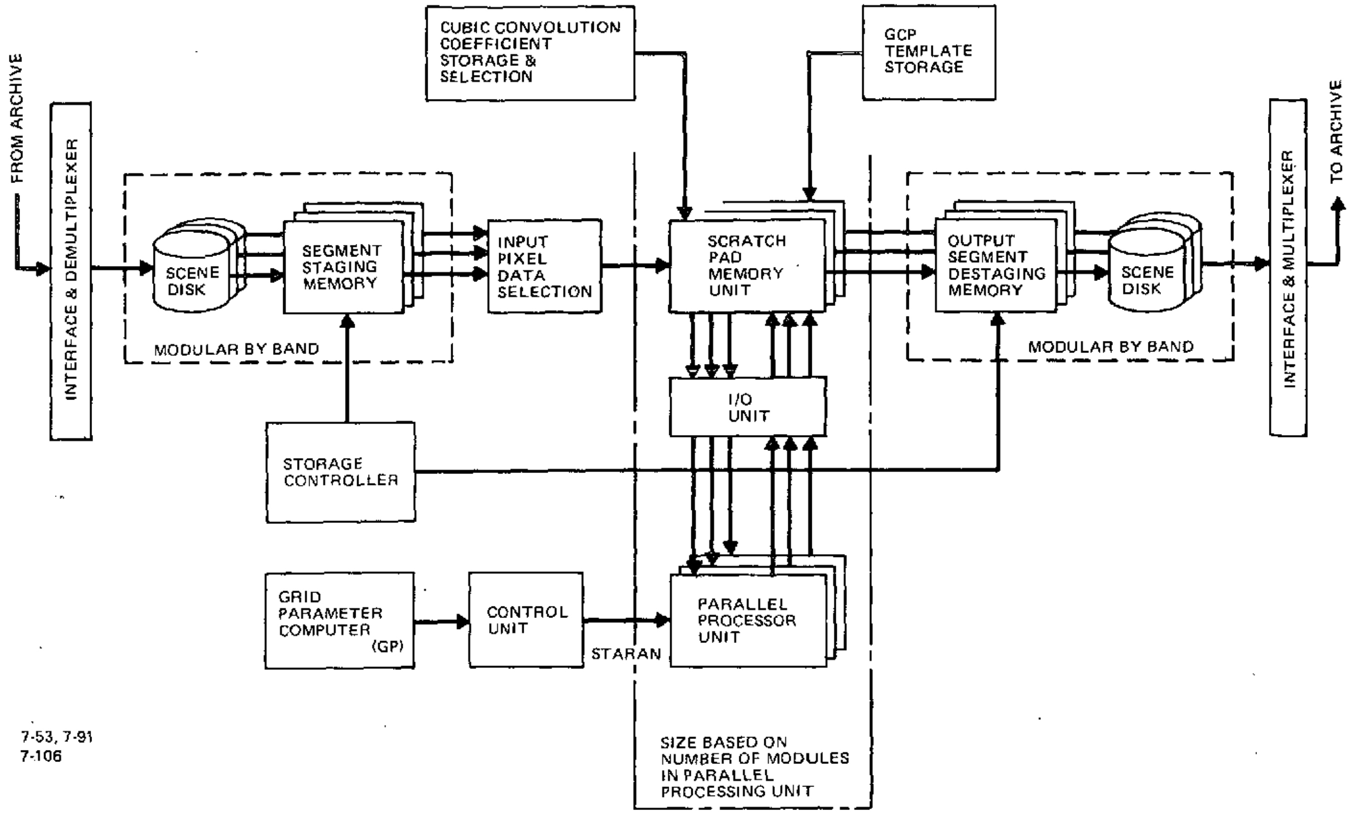
- SEVEN BASIC MODULES PROCESS APPROX 400 SCENES/DAY (8×10^{11} BITS/DAY)
- THROUGHPUT OF BASIC MODULE

  R_i 2.5 MBPS

  TIME TO PERFORM CUBIC CONVOLUTION ON EACH PIXEL 1.5 μS.

B. Option B — Special-Purpose Hardware

Fig. 1-13 Level II/III Processing Alternative Configuration Concepts (Part 2 of 3 Parts)



C. Option C – Associative Array Processor (AAP)

Fig. 1-13 Level II/III Processing Alternative Configuration Concepts (Part 3 of 3 Parts)

Table 1-8 Summary of CDPF Optional Configurations

	THROUGHPUT	EXPANDABILITY	FLEXIBILITY	RELIABILITY	RELATIVE COST
OPTION A MINI-COMPUTER	20 SCENES/DAY USING 4 MODULES (BILINEAR) INTERPOLATION)	5, 10, 15, 20 S/D ARE LOGICAL STEPS	MOST FLEXIBIE	@ 20 S/D HAVE FALL BACK CAPABILITY TO 75%, 50%, CAPACITY	–
OPTION B S.P. HARDWARE	FULLY PARALLELED SYSTEM CAN PROCESS 400 S/D USING CUBIC CONVOLUTION	EXPANDABLE IN STEPS, 15, 30, 60,400 S/D	VERY LITTLE FLEXIBILITY	SOME SINGLE-POINT SENSITIVITY; AT 400 S/D, CAN FALL BACK TO 6/7, 5/7, CAPACITY	10:1 CHEAPER THAN A AT 400 S/D
OPTION C STARAN	11 MODULES CAN PROCESS 400 S/D USING CUBIC CONVOLUTION	EXPANDABLE IN STEPS OF APPROXIMATELY 40 S/D	ALMOST AS FLEXIBLE AS OPTION A	SOME SINGLE-POINT SENSITIVITY; AT 400 S/D, CAN DE-GRADE IN STEPS OF 40 S/D	COMPARABLE TO COST OF OPTION B

7T-32
7T-51

1.8 INTERNATIONAL DATA ACQUISITION

The cost effectiveness of the TDRSS for International Data Acquisition was evaluated against

- Direct transmission (DT) to Regional Ground Stations and Primary Ground Stations
- Use of wide band video tape recorders (WBVTR) for the recording of data and playback when in contact with a STDN site (Table 1-9).

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.

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

Further evaluation in terms of scenes per day obtainable with TDRS as opposed to direct transmission and WBVTR indicate even greater relative cost effectiveness.

1.9 LOCAL USER SYSTEM/LOW-COST GROUND STATION

A systems viewpoint was taken with respect to a wide family of Local User Systems (LUS's) which includes the low-cost ground station concept. Centralized as well as local operations are necessary to assure system viability.

The basic cost conclusions (Table 1-10) are that minimum (basic) capability LCGS's can be provided for an equipment (hardware) cost, in quantities of 10 or more, of $125 thousand (1974 dollars), and that the enhanced processor and display subsystems, increasing the hardware cost to about $300 thousand in quantity, should provide as much local processing and analysis capabilities as most local area analysis specialists would need.

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

Table 1-9 International Data Acquisition System Cost Breakdown

(IN 1974 MILLIONS OF DOLLARS)

OPTION	EARTH TERMINAL	SPACECRAFT COSTS	$M/YEAR DATA PROCESSING & HANDLING COSTS	TOTAL COST (COST IMPACT TO EOS)**	
1. DT WITH SIX REGIONAL STATIONS	6	—	4.2	10.2	(0)
2. WBVTR (2TR's)	—	2	4.2	6.2	(2)
3. TDRSS	25 (BW PRICING)* 2.5 (BT PRICING)	3.0	4.2	32.2 7.7	(3) (1)
4. HYBRID 6 LCGS & WBVTR (1 TR)	0.6	1	0.4	2.0	(1)

* 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

3-259, 7T-10

Table 1-10 Low-Cost Ground Station Costs Vs Capability
(IN 1974 THOUSANDS OF DOLLARS)

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)	130
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)	223
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)	300

3-255
7T-11
7T-35

- Three cost targets: $130 thousand, 220 thousand, and 300 thousand (1974 dollars) for recurring (quantity 10 or more) hardware costs for LCGS LUS's that includes about $70 thousand 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.

Augmenting the low-cost ground stations are two concepts centralized within the CDPF. These are the Applications Program Development Laboratory (APDL) and the LUS Diagnostic and Equipment Laboratory (LDEL). The APDL concept permits centralized applications program development, eliminating the need for expensive development equipment (card readers, 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 analysis. Detected problems would be handled by local area maintenance personnel sent to a LUS as required. Each LUS would avail themselves of these CPF services over standard duplex telephone lines connected to the CPF.

An alternate means of data acquisition by the LUS to a direct RF downlink from the observatory was briefly explored. This consists of data transfer from the CPF to the LUS over highspeed telephone links. Preliminary studies showed this approach feasible and cost effective for a moderate number of LUS's (i.e., 35). An in-depth study of this approach, however, must be performed before a firm recommendation can be made.

1.10 COST SUMMARY

The EOS program, consisting initially of two spacecraft (EOS-A & A′), has been examined in detail for the most cost-effective design configuration. Program cost, in 1974 constant dollars, has been estimated at $162 million. Table 1-11 shows how this total could be split among major program elements; Table 1-12 provides an expected spread of the total over six fiscal years. Although the estimated cost has already incorporated several cost saving approaches (refer to Subsection 1.11), we recommend using a program target cost of $150 million in line with our Design-To-Cost (DTC) philosophy. Based on our DTC experience, we feel that this target is attainable without compromising major program objectives. "Cost Crunching," when applied only to non-fixed (launch vehicles and instruments) costs, would represent about 12%. If this approach were to be followed, the aplication of the groundrules of a DTC program, described in Subsection 1.12, would be required.

Since the basic spacecraft is considered standard for many missions, the costs for this spacecraft are broken out separately in Table 1-13. Of particular interest is the recurring cost, which is estimated in 1974 constant dollars at $6.2 million. Here again, we recommend a lower target cost of $5.5 million.

Table 1-11 EOS-A and A' Program Costs
(IN 1974 MILLIONS OF DOLLARS)

	NONRECURRING	RECURRING	TOTAL
• FIXED COSTS – INSTRUMENTS			$40.0
– TM (2)	(13.0)	(14.0)	
– MSS (2)	(1.0)	(12.0)	
– LAUNCH COSTS (2)	(0.250)	(17.0)	17.25
• OPERATIONAL SYS. COSTS			(23.47)
– MSS IMP (2)	(3.64)	(4.44)	
– GND DMS	(11.95)	(3.44)	
• R & D SYS COSTS			(32.06)
– TM IMP	(4.40)	(2.82)	
– GND DMS	(11.91)	(8.88)	
– NETWORK	(2.73)	(1.32)	
• SPACECRAFT			(39.87)
– BASIC SPACECRAFT (2)	(18.32)	(12.47)	
– M.P. SPACECRAFT (2)	(3.12)	(4.34)	
– SPARES & LOGISTICS	(0.41)	(1.21)	
• MISSION OPS	(4.73)	(4.90)	(9.63)
			TOTAL ($162.28)

3-226
7T-12

Table 1-12 EOS-A and A' Program Funding Summary
(IN 1974 MILLIONS OF DOLLARS)

	FY'77	FY'78	FY'79	FY'80	FY'81	FY'82	TOTAL
DATA MGT SYSTEM	$ 6.3	$14.9	$ 8.5	$ 4.5	$ 3.9	$2.1	$40.3
INSTRUMENTS	6.9	18.3	13.6	1.2	–	–	40.0
FLIGHT OPERATIONS	0.3	1.0	4.3	1.9	1.2	.9	9.6
LAUNCH SYSTEM	0.1	1.9	10.6	4.7			17.3
SPACECRAFT PROJECT	10.3	17.8	19.7	6.7	.4	.2	55.08
TOTAL PROGRAM	$23.9	$53.9	$56.7	$19.0	$ 5.5	$3.2	$162.28

3-227
7T-13

Table 1-13 Basic Spacecraft Cost
(IN 1974 MILLIONS OF DOLLARS)

	NONRECURRING	RECURRING
PROGRAM MGT	$ 1.58	$0.424
SYS ENGRG & INTERFACE	0.80	0.400
R&QA	0.72	0.320
I&T	0.29	0.240
DEVELOPMENT TEST	2.40	
GSE	2.31	
STRUCTURE, ADAPTER, ETC	1.80	0.597
EPS	1.11	0.730
SOLAR ARRAY & DRIVE	0.66	0.755
C&DH	2.93	1.138
ACS	2.37	1.160
RCS	0.57	0.471
O/B SOFTWARE	0.80	
SUBTOTAL	$18.32	$6.335

7T-14

1.11 PROGRAM/DESIGN EVALUATION

The design cost trades presented in Report No. 3 of this study were performed on an individual basis. Although the conclusions were, and still are, applicable for an individual trade area, a method must be applied to tie all tradeoff conclusions together from an overall design standpoint. The approach we have chosen to evaluate the total EOS design is to develop a system effectiveness model. The model relates system and hardware design and performance parameters to a single effectiveness Figure of Merit (FOM), which reflects top level program objectives. The effectiveness FOM we have chosen is " expected number of equivalent scenes per week" which expresses the expected system yield (in probabilistic terms) of a normalized data product. The normalized data product, or "equivalent scene," has a selected data quality (equiv- to a TM scene) and select mix of output products (CCT's, HDDT's, and B&W and color photographic images). All other system output design options are weighted relative to this normalized equivalent scene as a function of percentage of users satisfied (refer to Subsection 1.2).

Program design options were then evaluated in terms of the cost/performance (effectiveness) versus the resulting FOM (expected equivalent scenes) for a EOS-A and -A′ operational mission of two observatories, each with a two-year mission with one year of overlap. The results of this evaluation is shown in Fig. 1-14, which plots the total EOS-A and -A′mission observatory recurring plus operational cost per equivalent scene (cost effectiveness) produced during the operational missions. In examining this curve, the following general conclusions are apparent:

- The recommended EOS-A and -A′ program with a TM/MSS, 30-m resolution, and TDRS is a cost/performance effective approach within the constraints of using a conventional launch vehicle and the baselined TM 185-km swath width
- TDRS has a significant positive effect on program cost and performance effectiveness (Options 1 to 6 vs 6 to 13)
- The inclusion of provisions for Shuttle compatibility in the EOS design will permit a significant increase in performance at a very small cost increase when the Shuttle becomes operational (Option 1 vs 4 or 8 vs 7).

Note that the on-orbit resupply cost and performance effectiveness is not truly represented in this evaluation because its benefit is not realized for missions of less then 2.75 yr as described in our Shuttle utilization studies (Subsection 1.6). The recommended program indicated on the figure ruled out Shuttle utilization at this time since the initially defined EOS program did not include a 10-year operational system.

1.12 PROGRAM PLAN/MANAGEMENT APPROACH

Our recommended program plan and management approach is aimed at building a low cost EOS system without compromising top-level technical objectives.

The recommended program plan for EOS-A and -A′ is shown in Fig. 1-15. 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 effective utilization of personnel, GSE, and facilities while meeting EOS mission objectives
- Development and qualification of a Shuttle-compatible Basic Spacecraft which meets the requirements of EOS-A and -A′ as well as follow-on missions
- Design development and qualification completed prior to the start of the fabrication of flight hardware
- Static load qualification of the primary module and secondary structure by acceleration, including Shuttle crash-load demonstration
- Early structural qualification tests with component mass representations to define component environments prior to the start of component qualification tests
- Consolidation of all flight hardware environmental tests at the module level.

CONTENTS

ILLUSTRATIONS

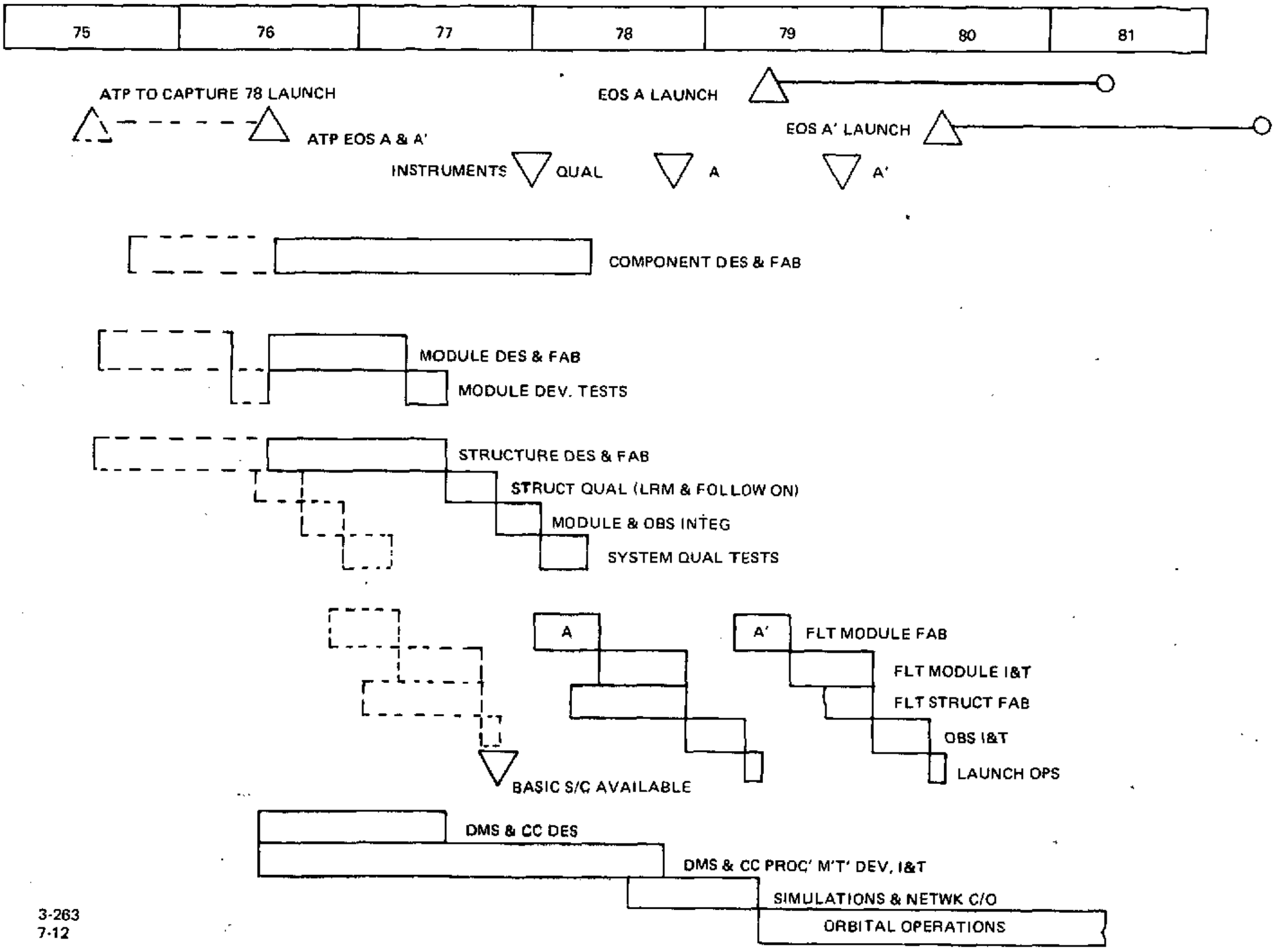


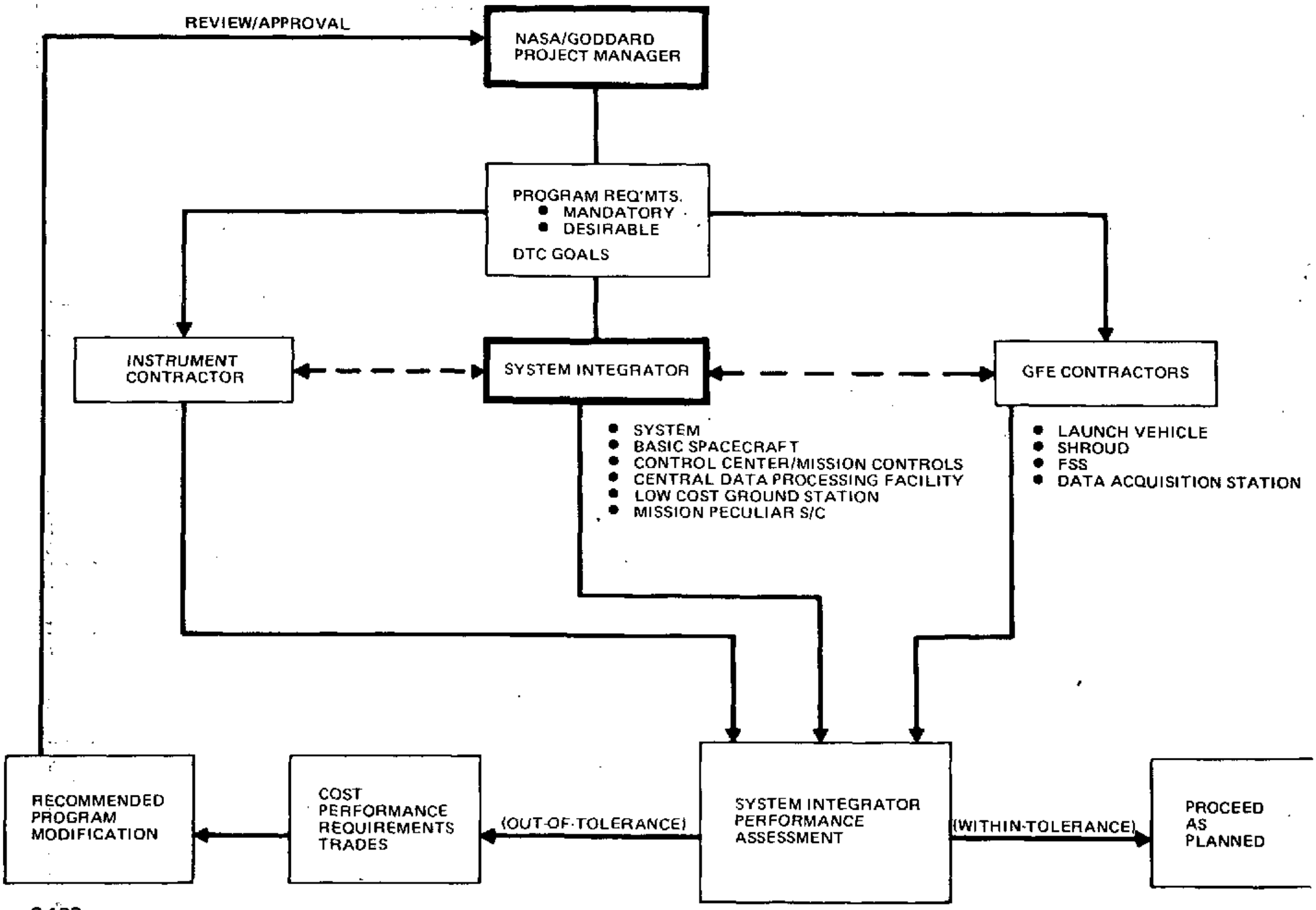
Fig. 1-15 EOS Program Schedule Summary

Inherent in the recommended program plan is a subplan that can be used to provide an acceptance tested Basic Spacecraft that is independent of a particular mission. This approach is illustrated by the schedule option shown in Fig. 1-14, which provides a Basic Spacecraft for a program with a 1978 launch.

The objectives of our recommended program management approach are to provide the management plan and controls necessary to design, develop, and integrate the EOS-A and -A′ program elements within specified program cost targets, and provide a low-cost standard spacecraft that will support future earth orbiting missions. To achieve these goals it is recommended that the EOS-A and -A′ program be conducted in a DTC environment with the specific cost targets defined in Subsection 1-10. To manage the program implemented in accordance with the DTC approach, we recommend a System Integration Team headed by a centralized program manager which we have designated as the System Integrator.

Our EOS System Definition Studies have established the DTC targets and program requirements for major spacecraft and ground system elements for the EOS-A and -A′ program. We have incorporated the DTC target into the EOS System Design Specifications. Each element contractor will be responsible for meeting the target set and further defining cost targets for each element of his Work Breakdown Structure (WBS). Designers will then have cost targets as design requirements and use data banks and productibility cost handbooks to select the detailed design which meets his cost and performance requirements. Where lower level WBS element cost and performance requirements cannot be met within cost targets, design cost tradeoffs of higher level requirements will be made by the element contractor to achieve overall element performance and cost targets. Figure 1-16 illustrates this activity flow.

The System Integrator shall be responsible for maintaining overall EOS-A and -A′ program costs within these targets. The scope of the System Integrator's tasks include schedule and technical performance as well as cost, and he has the overall responsibility under the direction of the GSFC Program Manager for all elements of the program. We recommend that the System Integrator, in his total program role, function through a working team concept comprised of personnel from NASA/Goddard, user groups, GFE contractors, and the instrument contractor. The direct communication provided by this team should bring overall management cost down through reduction of formal documentation, and provide the ability



3-198
4-1
7-13
7-14
7-15

Fig. 1-16 Design-To-Cost Activity Flow

to identify, jointly analyze, and resolve all interface problems in real time.

We also recommend that the instruments and DMS operations for the initial flight be procured by the Government and provided to the System Integrator as GFE. The System Integrator will manage the instrument contractors through the System Integration Team, and will resolve interfaces within the team or by an Interface Board with Goddard project management approval. The candidate instruments for the EOS program are in high-risk and low-risk categories. Since the TM and HRPI have a higher development risk, it is recommended that cost-type contracting be utilized. Instruments (such as the MSS and certain SEASAT instruments) that are of sufficiently low risk can be procured by either a firm fixed price contract or a fixed price incentive contract.

The overall contractual plan makes full use of a DTC philosophy, and presents a low-cost approach to the EOS-A and -A' execution phase. Cost savings expected from the above approaches are summarized in Table 1-14. The plan provides the structure to manage within program funding, and flexibility to manage within fiscal year funding. Also, an early selection of the System Integrator will assist in the instrument procurement as well as 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.

Table 1-14 Potential Cost Savings
(IN 1974 MILLIONS OF DOLLARS)

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

3-261, /T-24
7T-50

2 – SYSTEM REQUIREMENTS

2.1 PROGRAM/USER

The general design objective of the Earth Observatory Satellite Program is to provide a flexible, cost-effective "facility" for conducting a broad range of earth remote sensing missions. The facility will consist of a general purpose, or standard spacecraft capable of accommodating a wide variety of instruments, and all ground data acquisition and processing systems necessary to provide data directly to the users. Program functional elements for this facility are illustrated in Fig. 2-1.

The EOS program and user requirements, as well as many subsystem design requirements, have been imposed by the NASA/GSFC RFP for this study (PR No. 5-66203-202). Grumman has built around these requirements, adding to them and modifying them in some cases, as a result of our system trade studies. The complete requirements are provided in Report No. 5, "System Design and Specifications". Top level requirements are summarized as follows:

- The EOS system shall provide a basic capability to perform Land Resources Management (LRM) missions and shall be adaptable with minimum modification to support the following mission categories:
  - Earth observation
  - Solar observation
  - Stellar observation
  - Inertial pointing
- The Basic Spacecraft shall be modular and standardized for a broad range of missions in the foregoing categories
- The EOS system designed for LRM shall accommodate combined operational and R&D functions
- The EOS shall be designed to utilize the Space Shuttle for economic and operational benefits. Designs shall incorporate Shuttle deploy, retrieval, and in-orbit resupply
- The LRM mission instruments shall provide multi-spectral imaging of the earth's surface with spatial, spectral, and radiometric resolution as listed in Subsection 2.3
- Earth scanning revisit cycle for LRM missions shall be a maximum of 17 days. The design goal is 6 to 9 days
- Data turnaround for LRM mission shall be 24 to 48 hr
- Basic processed output products shall be digital and photographic
- Output products are required for up to 100 generic users
- Central processing throughput rate shall be capable of handling a minimum of 10^{10} bits/day and expandable to 10^{12} bits/day
- Provisions shall be made for international data acquisition via TDRSS
- The EOS System for LRM mission shall be designed to target cost.

2.2 OBSERVATORY REQUIREMENTS

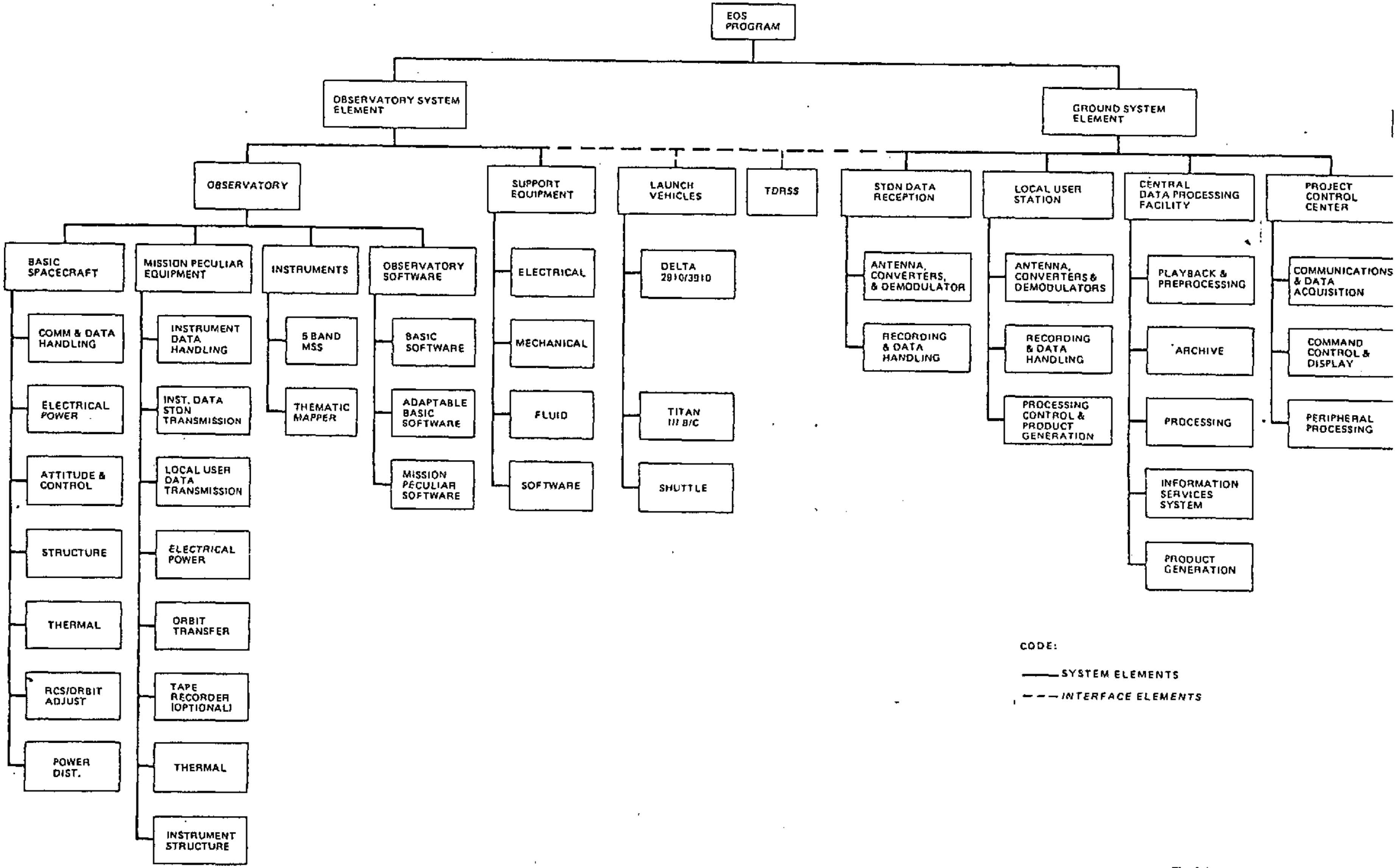
The observatory consists of the Basic Spacecraft and the Instrument/Mission Peculiar Equipment.

2.2.1 BASIC SPACECRAFT

2.2.1.1 COMMUNICATION AND DATA HANDLING SUBSYSTEM

The CDHS shall:

- Provide tracking, command and telemetry compatibility with STDN and TDRSS. Table 2-1 shows the STDN command direct link requirements
- Execute commands in both real and delayed time



(1)5-14
7-61

Fig. 2-1 Program Functional Elements

DOUT FRAME 2-2

OUT FRAME 2
2-2 - a

Table 2-1 C&DH STDN Direct Link

(Note: Minimum Design Margin 6dB Above Signal Level For 10^{-5} Bit Error Rate For:)

NO.	PARAMETER	REQUIREMENT
1	FREQUENCY	2025 TO 2120 MHz
2	GROUND ANTENNA SIZE	30 FT DISH
3	MIN ELEVATION ANGLE OF GROUND ANT.	5°
4	MIN GROUND TRANSMITTED POWER	500 W
5	ATMOSPHERE LOSS	0.6 d/B
6	POLARIZATION	RHCP
7	MAX SLANT RANGE	3040 km
8	UPLINK DATA RATE	2 Kbps
9	E/NO. REQUIRED	12 dB

T7-18

- Compress and store spacecraft data (recorder optional).

2.2.1.2 ELECTRICAL POWER SUBSYSTEM

The EPS shall:

- Provide for solar energy conversion, storage, and control of 28 ± 7-v power
- Power handling capability to 1500 w orbital average power
- Peak power to 3000 w.

2.2.1.3 ATTITUDE CONTROL SUBSYSTEM

The ACS shall provide:

- Accurately pointed, stable earth referenced platform with low jitter. The required ACS modes are summarized in Table 2-2.
- Inertial attitude hold for maximum solar power and for Shuttle retrieval operations
- For disturbance torques introduced by the instruments of $< 5 \times 10^{-3}$ ft-lb.

2.2.1.4 STRUCTURE SUBSYSTEM

The structure subsystem shall:

- Posses sufficient strength and rigidity to survive critical loading conditions that exist within the envelope of mission requirements
- Provide for support of a wide variety of instrument configurations
- Provide for modularity and inflight resupply of subsystems

Table 2-2 Summary of ACS Modes

NO.	MODE	PURPOSE
1	RATE CHANGE	NULL RATES AFTER BOOSTER SEPARATION. GENERATE ORBIT RATE ABOUT THE PITCH AXIS IN PREPARATION FOR THE EARTH-POINTING ATTITUDE HOLD MODE.
2	COARSE SUN ACQUISITION	ACQUIRE THE SUN FOR SOLAR POWER AND IN PREPARATION FOR FINE SUN ACQUISITION AND FOR SUBSEQUENT GUIDE STAR ACQUISITION.
3	FINE SUN ACQUISITION	POINT TOWARD THE SUN WITH INCREASED ACCURACY. UPDATE ATTITUDE IN PREPARATION FOR SUBSEQUENT GUIDE STAR ACQUISITION.
4	RATE HOLD	HOLD SELECTED RATE ABOUT SUNLINE FOR GUIDE STAR ACQUISITION (ALTERNATIVE: SLEW ABOUT SUNLINE TO ATTITUDE FOR GUIDE STAR ACQUISITION AFTER UPDATING USING DSS AND MAGNETOMETER). BACKUP FOR EARTH-POINTING. HOLD ORBIT RATE ABOUT PITCH AXIS PRIOR TO EARTH-POINTING ATTITUDE HOLD. BACKUP FOR DEPLOYMENT, RETRIEVAL, AND SERVICE OPERATIONS.
5	SLEW	CHANGE ATTITUDE FROM PRESENT ATTITUDE TO ANOTHER IN PREPARATION FOR NEXT EVENT, SUCH AS EARTH-POINTING.
6	EARTH-POINTING ACQUISITION HOLD	POINT THE INSTRUMENTS AT THE EARTH AND X AXIS IN THE DIRECTION OF FLIGHT TO PERFORM THE EOS MISSION.
7	INERTIAL-POINTING ATTITUDE HOLD	POINT THE INSTRUMENTS TOWARD A SELECTED POINT IN SPACE WITH THE ROLL ANGLE ABOUT THIS LINE IN SPACE CHOSEN FOR MAXIMUM SOLAR POWER. PERFORM A STELLAR MISSION. HOLD AN ATTITUDE SUITABLE FOR DEPLOYMENT, RETRIEVAL, OR SERVICING.
8	SURVIVAL	SURVIVE IN CASE OF FAILURES IN OTHER MODES. MAXIMUM SOLAR POWER IS OBTAINED. RETRIEVAL OR SERVICING CAN BE ACCOMPLISHED. SOLUTIONS TO FAILURES CAN BE WORKED OUT.

T7-19

- Provide interfaces between the spacecraft and launch vehicles (Delta, Titan and Space Shuttle), ground support equipment and launch pad handling equipment.

2.2.1.5 THERMAL SUBSYSTEM

The thermal subsystem design goal operating temperatures shall be 21 ± 11°C for the CDHS, ACS, EPS and 4.4 to 37.7°C for the OA/RCS. Primary approach for achieving temperature control shall be passive. The modules and structure shall be thermally independent of each other and the module shall be designed to dissipate all equipment heat into space.

2.2.1.6 ORBIT ADJUST/REACTION CONTROL SUBSYSTEMS

The OA/RCS shall provide:

- Propulsion power for translation and rotational maneuvers
- Desaturation of reaction wheels
- Correction of orbit injection errors and orbit adjustment due to orbit decay.

2.2.2 MISSION PECULIAR EQUIPMENT FOR LAND RESOURCES MANAGEMENT MISSION (EOS-A)

2.2.2.1 INSTRUMENTS FOR LAND RESOURCES MANAGEMENT

The LRM instruments (EOS-A) shall consist of two sensors, the Multi-Spectral Scanner (MSS) and the Thematic Mapper (TM).

2.2.2.2. COMMUNICATIONS AND DATA HANDLING SUBSYSTEM

The CDHS shall be capable of transmitting additional telemetry rates through TDRS. Table 2-3 is a partial list of the dual-feed S/Ku-band steerable antenna design requirements.

- Narrow band data rate: selectable, 16 and 32 Kbps, in addition to 8 Kbps, 4 Kbps, 2 Kbps and 1 Kbps
- Medium band data rate: 128 Kbps.

2.2.2.3 ELECTRICAL POWER SUBSYSTEM

The EPS shall be capable of providing to the observatory an orbital average power of 525 w for the two-year operational phase, a minimum of 200 w orbital average power shall be available for the instruments. Solar array drive requirements are contained in Table 2-4.

2.2.2.4 STRUCTURE SUBSYSTEM

The structure subsystem shall provide all structure required to support the instruments and mission peculiar equipment (TM, MSS, antennas, solar array, wideband communications, and data handling).

2.2.2.5 INSTRUMENT MISSION PECULIARS

The IMP's include the following subsystems:

- Wide-band data handling and on-board data compaction (Fig. 2-2 shows the subsystem interface)
- Primary relay (TDRS) wideband communications
- Primary direct (wide band) communications
- Local user (medium band) communications
- Tape recording system (optional).

The overall functions of the IMP's are to handle (process) the instrument data and format it appropriately; record it for later transmission (optional); transmit the instrument data to STDN Primary Ground Stations; also transmit certain data to Local User Stations (LUS); and transmit data via a relay satellite (TDRS). Specifically, the transmission system will be capable of transferring data at a composite total rate of 240 Mbps to primary stations via TDRSS, and 16 to 20 Mbps to local users. The baseline data sources for the primary links are the TM and the MSS; MSS data or compacted TM data will be sent over the LUS link.

The IMP's are required to fulfill the basic transmission functions at minimum total system cost, especially in the case of the LUS link, and should do so at a performance level adequate for an overall system error rate of 1 bit in 10^5. Therefore, the link transmission error rate has been specified in the range 10^{-6} to 10^{-5}, depending on the link.

Transmission should take place via frequency bands permissible for such missions, and at satellite power levels which do not exceed internationally

Table 2-3 C&DH Dual Feed - S/Ku-Band Steerable Antenna Design Requirements

PARAMETER	S-BAND	Ku-BAND HIGH GAIN	Ku-BAND LOW GAIN
1. FREQUENCY, GHz			
TRANSMIT	2.025 TO 2.120	14.6 TO 15.2	
RECEIVE	2.200 TO 2.300	13.6 TO 14.0	14.6 TO 15.2
2. ANTENNA TYPE	PARABOLIC DISH	PARABOLIC DISH	OPEN ENDED WAVEGUIDE
3. FEED TYPE	PRIME FOCAL POINT	CASSEGRAIN	N/A
4. POLARIZATION	RHCP	RHCP	RHCP
5. AXIAL RATIO, dB, MAX	1.5	1.5	1.5
6. INPUT VSWR AT ROTARY JOINT OUTPUT	1.4:1	1.5:1	1.5:1
7. SIDE AND BACK LOBE LEVELS, dB	$\leq$ 17.0	$\leq$ 17.0	N/A
8. ANTENNA DISH SIZE, FT	12.5	12.5	N/A
	FREQ. (2.25 GHz)	FREQ. (14.6 GHz)	FREQ. (14.6 GHz)
9. NET ANTENNA GAIN (dB) MEASURED AT THE ROTARY JOINT INPUT (INCLUDES ALL FEED ILLUMINATION AND TRANSMISSION LINE COMPONENT LOSSES)	35	51	0 (MINIMUM WITHIN 60° HPEW)
10. TRACKING CONFIGURATION	OPEN	CLOSED (PSEUDO MONOPULSE)	–
11. TRACKING ACCURACY, 3σ	–	0.17 DEGREES	–
12. POINTING ACCURACY, 3σ	–	0.05 DEGREES	–
13. GIMBAL STEP SIZE	–	0.02 DEGREES	–
14. SLEW RATE,			
VELOCITY	–	20 DEG/SEC MAXIMUM	–
ACCELERATION	–	60 DEG/SEC^2 MAXIMUM	–
15. SCAN ANGLE OFF-BORESIGHT, 2 AXIS (XY GIMBAL)	X (INNER) GIMBAL Y (OUTER) GIMBAL	90 DEGREES 110 DEGREES	

(1)T5-44, 7T-20

Table 2-4 Solar Array Drive Requirements

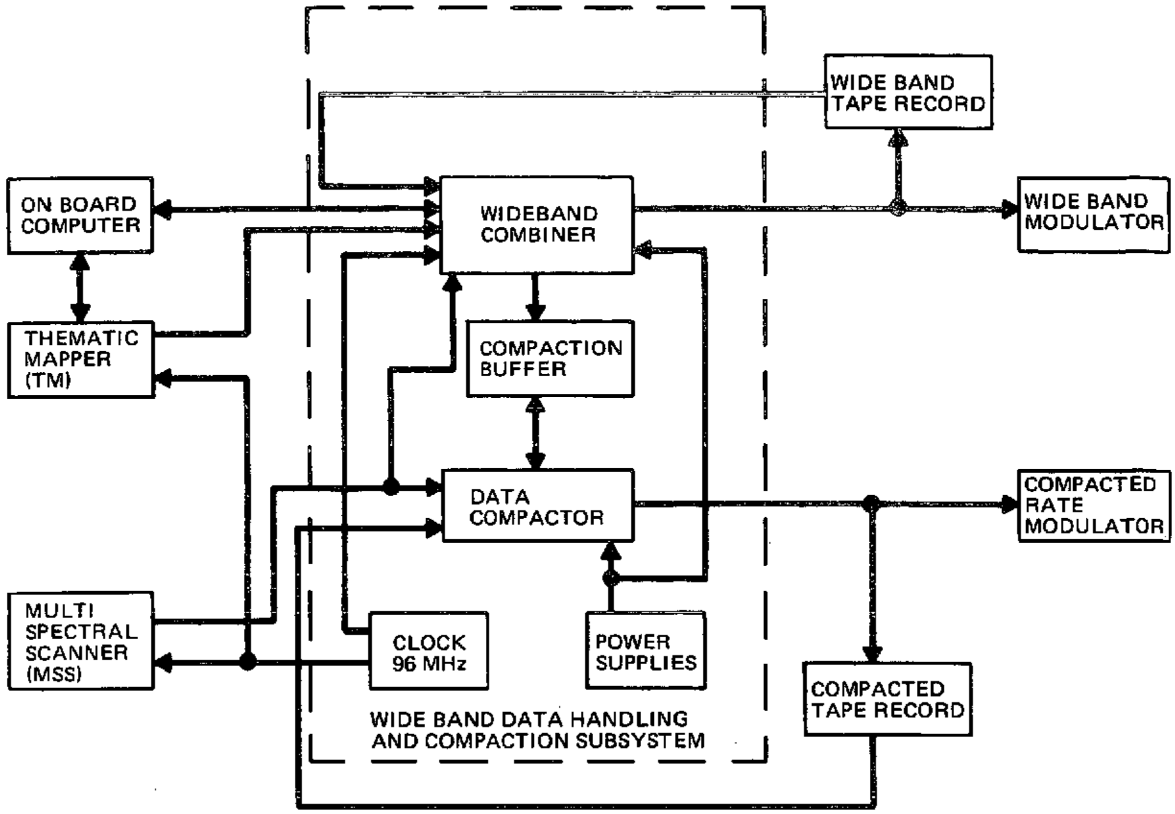
PARAMETER	REQUIREMENT
OPERATION	CONTINUOUS, BI-DIRECTIONAL
OPERATING VOLTAGE	28±7 VDC
TRACK RATE(1)	ORBIT DEPENDENT (3.8°/MIN NOMINAL
TRACK ACCURACY(1)	SPECIFIED IN EOS-SS-260(2)
FAST SLEW(1)	15°/MIN., NOMINAL
POSITION INDICATION	±1°
TORQUE(1)	2 TIMES TOTAL REFLECTED TORQUE AT OUTPUT SHAFT DUE TO FRICTION IN BEARINGS & SLIP RINGS MINIMUM
POWER TRANSMISSION(1)	50 A MAX; 125 VDC MAX
SIGNAL TRANSMISSION(1)	LIGHT/DARK SENSOR; TEMPERATURE & VOLTAGE FOR EACH SOLAR PANEL

NOTES

(1) REQUIREMENTS ARE MISSION PECULIAR – ORBIT AND/OR INSTRUMENT DEPENDENT.

(2) ACCURACY BASED ONLY ON TRACKING THE SUN WITHIN THE SPACECRAFT ORBIT PLANE & DOES NOT INCLUDE THE INCIDENT ANGLE VARIATIONS CAUSED BY OUT OF ORBIT PLANE MOVEMENT OF THE SUN.

(1)T5-46
7T-21



(1)5-47, 7-36

Fig. 2-2 C&DH Wide-Band Data Handling and Compaction Subsystem Interface

agreed-upon flux density limitations. The transmissions for local users should be able to serve stations located within 500 km of satellite nadir, while the primary direct link should function even when the satellite appears at a look angle of 2 deg above horizon (or higher) at a primary station.

2.2.3 SOFTWARE

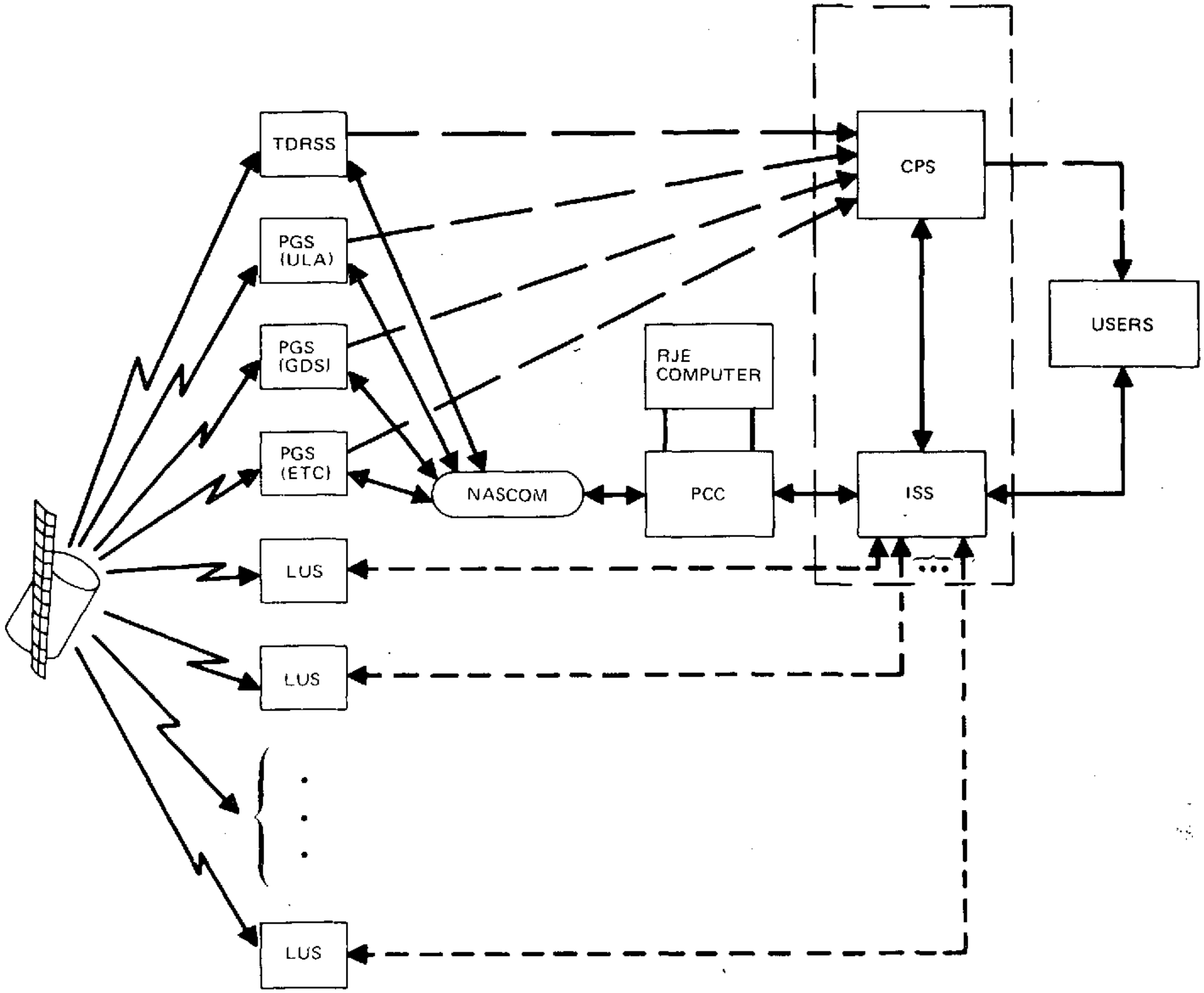
The observatory software shall be prepared in modules which may be assembled and verified independently before linking the software package with a specific spacecraft. Three classes of modules are:

- Basic software
- Adaptable software
- Mission peculiar software.

2.3 GROUND ELEMENT REQUIREMENTS

The ground system must provide a means of acquiring, recording, conveying, and processing EOS payload data to make them of use to EOS data users and a means of monitoring and controlling the observatory. Figure 2-3 depicts the EOS ground system. Major elements include:

- Primary Ground Station (PGS):
  - Number: 3 (ULA, GDS, ETC)
  - Error Rate: Pe $< 1 \times 10^{-6}$ with incident signal (carrier) power = -157.1 dBw with 9-meter antenna and -159 dBw with 12-meter antenna
  - Frequency: X-Band (8.025 to 8.4 GHz)
  - Data Recording: 240 Mbps
- Central Data Processing Facility (CDPF):
  - Initial Configuration Capability: Five scenes of TM data or equivalent, per day ($\approx 10^{10}$ bits/day)
  - Final Configuration Capability: 400 scenes of TM data per day ($\approx 10^{12}$ bits/day)
  - Output Products: As indicated in Table 2-5



SPACECRAFT PAYLOAD DATA
CONTROL DATA & INFORMATION
INFORMATION & SERVICES DATA

PGS–PRIMARY GROUND SYSTEM
LCS--LOCAL USER SYSTEM
(INCLUDES THE LOW COST GROUND STATION)

RJE– REMOTE JOB ENTRY
ISS--INFORMATION & SERVICES SYSTEM
CPS–CENTRAL PROCESSING SYSTEM
PCC–PROJECT CONTROL CENTER
EOS–EARTH OBSERVATORY SATELLITE
NASCOM–NASA COMMUNICATIONS
TDRSS–TRACKING AND DATA RELAY SATELLITE SYSTEM

(2)5-3
3-101
7-89
7-95

DATA COMMUNICATIONS, RECORDING, CONTROL, PROCESSING, AND INFORMATION SERVICES

Fig. 2-3 EOS Ground System Concept

Table 2-5 Output Products

PRODUCT	GEOMETRICALLY UNCORRECTED	GEOMETRICALLY CORRECTED	REDUCED DATA OPTIONS
B/W FILM	✓		
COLOR FILM	✓		
HIGH DENSITY DIGITAL TAPE	✓	✓	
COMPUTER COMPATABLE TAPE		✓	✓
CUSTOM OUTPUT PRODUCTS	FILM PRODUCTS GEOMETRICALLY CORRECTED WITH CUSTOM GAMMA CAPABILITY		
SUBAREA ENLARGEMENTS	SPECIFIC MAP SCALES (e.g. 1:1,000,000 1:500,000)		
CUSTOM FILM	SPECIFIC FALSE COLOR		
CUSTOM DIGITAL PRODUCTS	CCT OUTPUTS WITH VARIOUS FORMATS (e.g., BAND INTERLEAVED, BAND SEQUENTIAL) AND SUBAREAS (e.g., PARTIAL SCENES)		

- Output product quality: as indicated in Table 1-7
- 7T-45 Output/input product/data quantity: as indicated in Table 1-5

Table 2-6 Output Product Quality

PRODUCT	GEOMETRICALLY UNCORRECTED(1)		GEOMETRICALLY CORRECTED(2)	
	TM	HRPI	TM	HRPI
• SWATH WIDTH, KM	185	48	185	48
• SPATIAL RESOLUTION				
– VISIBLE, M	30	10	30	10
– THERMAL, M	120	–	120	–
• LINEARITY (μ RAD) IFOV	0.2	0.2	0.2	0.2
• BAND TO BAND REGISTRATION (μ RAD) IFOV	0.1	0.3	0.1	0.3
• POSITION ACCURACY (W/O GCP),(3) M	± 450	± 450	± 170	± 170
• POSITION ACCURACY (WITH GCP,(3) M	–	–	± 15	± 15
• RELATIVE RADIOMETRIC ACCURACY				
– VISIBLE				
o TAPE, %	± 1.6	± 1.6	± 1.6	± 1.6
o FILM, %	± 5	± 5	± 5	± 5
– THERMAL				
o TAPE, K	± 1	–	± 1	–
o FILM, K	± 3	–	± 3	–

NOTES:

(1) Includes radiometric correction, earth-rotation correction, line-length adjustment, correction for earth curvature, and predicted emphemeris.

7T-46 (2) Additionally includes use of best-fit ephemeris from measured data. (3) GCP = ground control points.

Table 2-7 Output/Input Product/Data Quantity

PRODUCT	DATA VOLUME	NUMBER OF DATA USERS	NUMBER OF FORMATS
HDDT (UNCORRECTED)	$10^{10} - 10^{12}$ BITS/DAY	2 – 10	–
HDDT (CORRECTED)	$10^{10} - 10^{12}$ BITS/DAY	2 – 10	–
CCT (CORRECTED)	$10^{9} - 10^{10}$ BITS/DAY	10 – 100	1 – 1
BLACK&WHITE POS/NEG(1)	20 – 200 SCENES/DAY	5 – 50	1 – 3(3)
BLACK&WHITE PRINTS		5 – 10	1 – 3(3)
COLOR POS/NEG(2)	10 – 100 SCENES/DAY	2 – 20	1 – 3(3)
COLOR PRINTS		2 – 10	1 – 3(3)

(1) First generation product – 24mm (9.5 in.)

(3) Enlargement to standard map scales

7T-47 (2) Second generation product – 24mm (9.5 in.)

(4) Processing considered as two 8-hr shifts per day

- Output Product Quality: As indicated in Table 2-6
- Output/Input Product/Data Quantity: As indicated in Table 2-7
- Basic Processing Technique: Digital
- Turnaround Time: 24 hr input to output in a 16-hr work day for data processing. Additional 24 hr for output product generation
- Information Services System: Information Management System for total control of CDPF, user interfaces, data prioritizing

- Project Control Center
  - Mission Planning: Coordination of user requests. Development of observatory contact messages
  - Mission Operations: Real-time monitoring and control of the observatory ephemeris update to CDPF
- Local User System
  - Data Acquisition: Direct from observatory
  - Data Recording: 15 to 20 Mbps
  - Frequency: X-Band (8.025 to 8.4 GHz)
  - Error Rate: $Pe < 1 \times 10^{-5}$
  - Data Type: Reduced TM resolution or coverage subset of data transmitted to the PGS.

3 – DESIGN/COST TRADE METHODOLOGY

3.1 PROGRAM EFFECTIVENESS

Figure 3-1, which appeared in our proposal, depicts our original approach to the problem of overall EOS System design. This approach included the evaluation of the many different and loosely constrained options by means of a system Figure of Merit (FOM), which would combine the results of the trade studies for each candidate, and numerically evaluate each.

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

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

The system synthesis task has been completed, providing weight and cost data for the spacecraft and DMS options. In addition, the design/cost tradeoffs shown have been completed and their individual results are contained in Report No. 3.

Figure 3-2 shows the interface between the program effectiveness model and the individual trade study areas. The trade studies have provided data and inputs for each option to the effectiveness model. The model then combined data from all trade areas to evaluate options on a programmatic basis. A further explanation of this process and its results are contained in Section 7 of this report.

3.2 PROGRAM OPTION DEVELOPMENT AND SCREENING

EOS candidate configurations and their associated mission models were developed to cover the spectrum of required mission capabilities and selected program cost budgets. This set of configurations was 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 used in the process were consistent throughout the study, but the approach was tailored to the EOS programmatic mission model received from GSFC during the course of the study. The approach taken for the initial missions depended heavily on ground-ruled inputs (such as sensors to be carried, number of spacecraft, flights in the mission model, and the booster used). The option development and screen process used is depicted in Fig. 3-3.

Since the EOS-B and -B' missions carry a complete complement of new instruments, and include the new capability of offset pointing of an instrument, the process shown in the figure has been modified for these missions. In this case, we restricted the sensor complement and possible boosters by ground rule, but the choice of booster and the orbit were decided by reconciling the sometimes conflicting requirements of user revisit, atmospheric drag, and Shuttle/booster payload-to-orbit capabilities.

The EOS missions downstream of EOS-B' have been grouped under the general category of "Follow-on missions". These missions were often incompletely defined, and even in those cases where there was no lack of definition, the impact of their missions on the EOS was unknown. Thus, the process shown in Fig. 3-3 was again modified when applied to these missions.

In all cases, however, EOS configuration options were developed which would cover the spectrum of requirements within cost. 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 cost, spacecraft/payload weight, and booster capabilities for programmatic options providing varying capability. The results of the system synthesis task for the EOS-A, -B, and -C missions are given in Fig. 3-4.

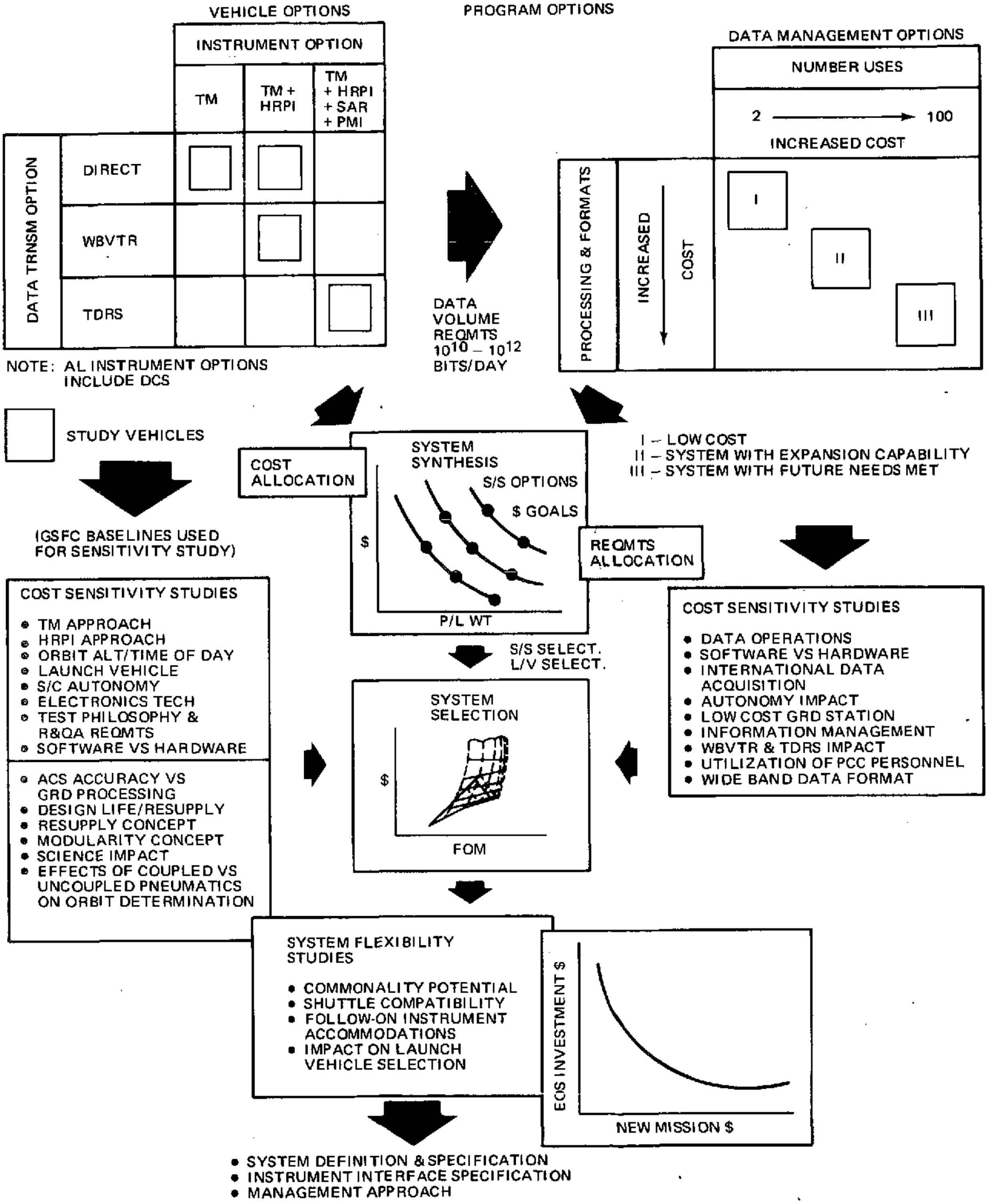


Fig. 3-1 System Design Approach. Cost is a Significant Factor in Our Design Approach

3-134
7-56

AVAILABILITY OF DATA PRODUCTS	QUANTITY OF DATA PRODUCTS	QUALITY OF DATA PRODUCTS	AVAILABILITY OF THROUGHPUT	QUANTITY OF THROUGHPUT	QUALITY OF THROUGHPUT	AVAILABILITY OF S/C SUPPORT	QUALITY OF S/C SUPPORT	AVAILABILITY OF INSTRUMENT OUTPUT	QUANTITY OF INSTRUMENT OUTPUT	QUALITY OF INSTRUMENT OUTPUT	EFFECTIVENESS OF LAUNCH VEHICLE	EFFECTIVENESS OF SYSTEMPARAMETERS	. . . ARE INPUT TO THIS SECTION OF THE EFFECTIVE-NESS MODEL / DATA AND OUTPUTS FROM THIS TRADEOFF OR DESIGN AREA . . .	
								•	•	•		•	ORBIT ALTITUDE	REPORT NO. 1 ORBIT/LAUNCH VEHICLE
						•	•				•		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	RPT #2
						•	•	•	•	•			INTERFACE SPECIFICATIONS	
							•	•		•	•		ORBITAL ALTITUDE/ORBIT TIME OF DAY	REPORT NO. 3 SYSTEM DESIGN/COST TRADEOFFS
						•		•			•		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	REPORT NO. 5 SYSTEM DESIGN & SPECIFICA-TIONS
						•	•	•	•	•			INSTRUMENT MISSION PECULIARS	
						•		•			•	•	SPACECRAFT DESIGN	
						•	•	•			•		SPACECRAFT MISSION PECULIARS	
•	•	•	•	•	•							•	DATA MANAGEMENT SYSTEM DESIGN	
						•		•			•		VEHICLE INTERFACES	
						•	•	•			•	•	SPACE SHUTTLE INTERFACES	RPT #6
						•	•	•			•	•	SPACE SHUTTLE UTILIZATION	
•	•	•	•	•	•	•	•	•	•	•	•	•	FLEXIBILITY COST/BENEFIT ANALYSIS	

Fig. 3-2 Interfaces Between System Effectiveness Model and Tradeoffs. FOM is Focal Point for all Tradeoffs.

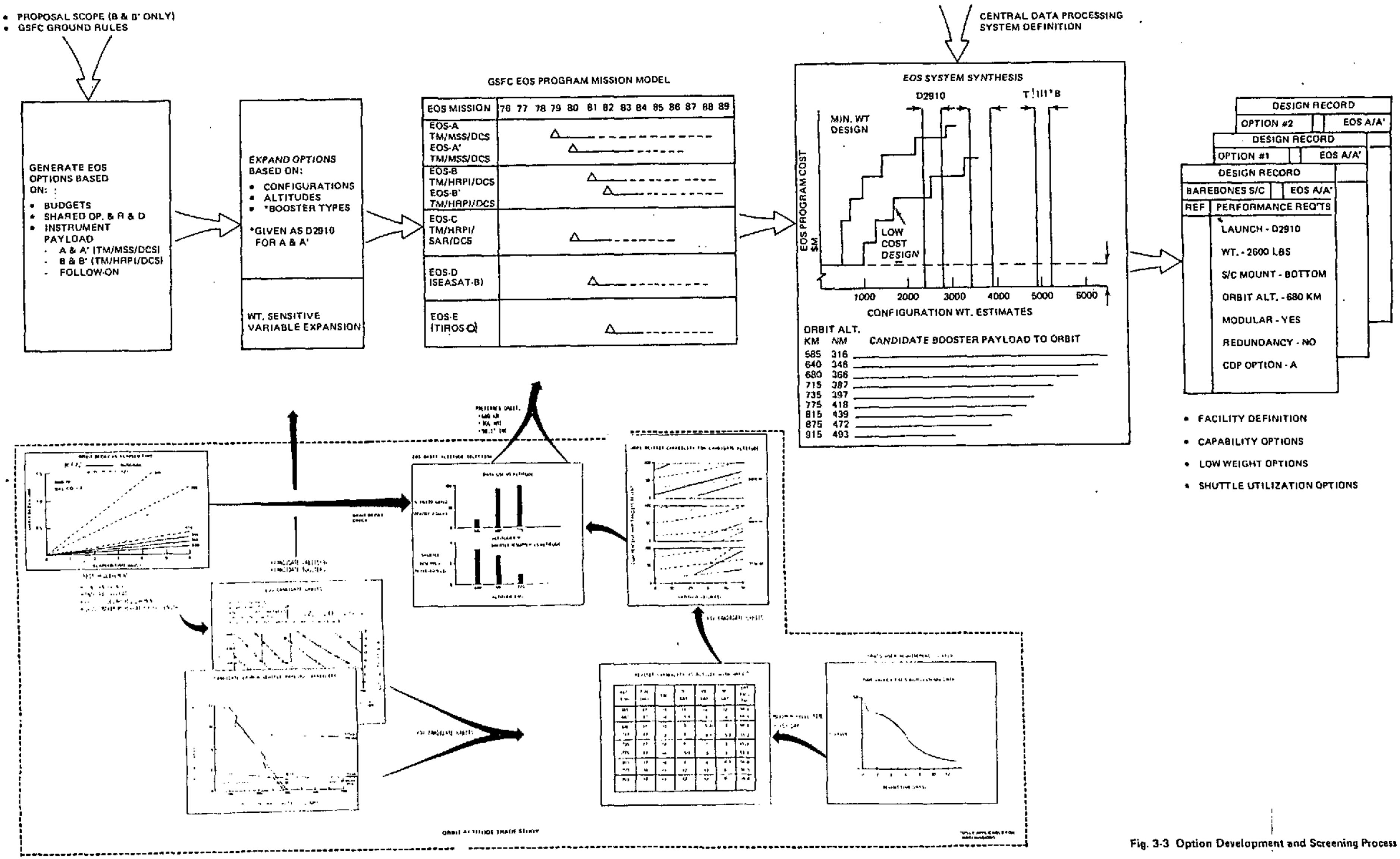
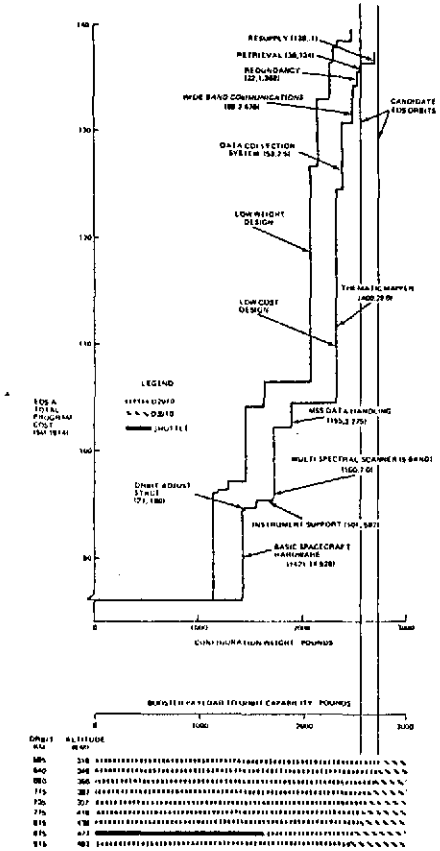
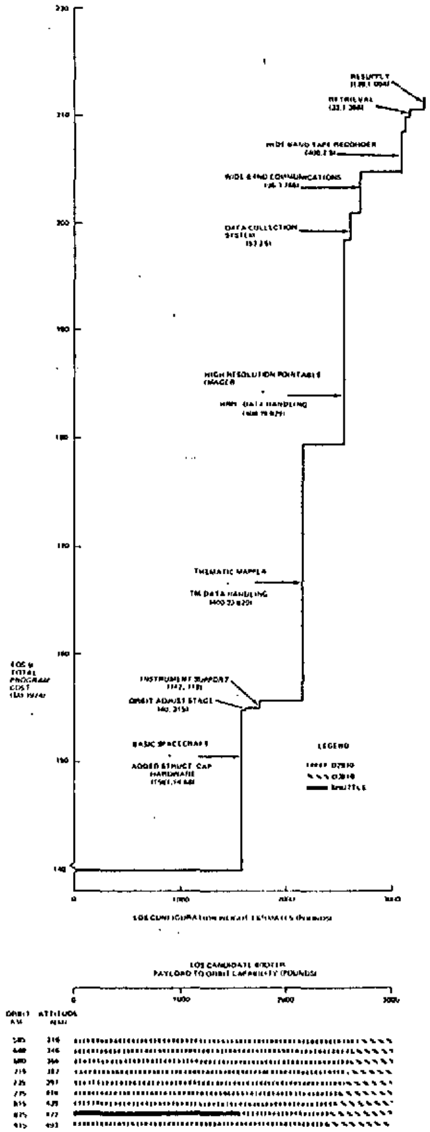


Fig. 3-3 Option Development and Screening Process

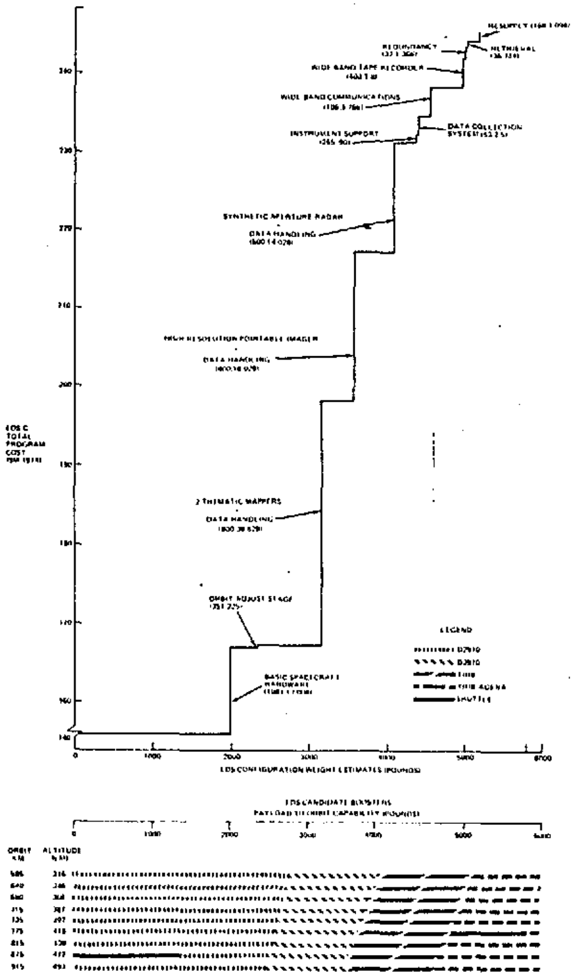
Preceding page blank ORIGINAL PAGE IS



3-130 EOS-A



3-131 EOS-B



3-132 EOS-C

Fig. 3-4 EOS-A, -B and -C System Synthesis

7-57 3-4

FOLDOUT FRAME

ORIGINAL PAGE IS

3-6

4 – TRADE RESULTS

4.1 ORBIT ALTITUDE SELECTION

The trade study provides the rationale for selecting an EOS mission altitude. The selected EOS Land Resources mission orbit should be sun synchronous with a minimum altitude having acceptable orbit decay, swath width, 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 TM swath width and the desired repeat cycle time (Fig. 4-1a). Our studies indicate that an altitude range of 365 to 385 n mi is best suited to these requirements.

A promising sun synchronous orbit for EOS missions A, B, and C is 366 n mi (678 km) (Fig. 4-1c) when using a TM with a 100-n mi (185-km) swath width. When using a HRPI with 30-deg offset pointing in CONUS viewing, 90% of a reference swath may be viewed again in three days. This orbit has a 17-day repeat cycle, and a 14-n mi swath overlap. The adjacent western swath overlap occurs in 3 days; the eastern in 14 days.

For a nine-day repeat cycle, an acceptable orbit within the recommended altitude is 382 n mi (708 km). This orbit is suited for a TM with a 178-n mi (330-km) swath width. It provides a TM swath overlap of 15 n mi; an adjacent swath overlap occurs in two days.

Figure 4-1d shows the orbit decay resulting from aero drag during the first six months for both a nominal and nominal +2σ atmosphere (Jacchia Model). Sideslip in the longitude of the orbit node for the initially 366-n mi altitude orbit appears in Fig. 4-1e. If corresponding swaths are permitted to accumulate, a nodal sideslip up to ± 20 n mi, this may take one and one-quarter to three months to achieve, depending on the severity of the atmospheric drag. Figure 4-1f shows the ΔV need for each orbit adjust: 0.3 fps for the nominal atmosphere and 0.8 fps for the nominal +2σ. At three-month intervals over two years, seven orbit adjusts are needed for a total of 2.1 fps. At one and one-quarter-month intervals, 19 adjusts require a total of 15.2 fps (Fig. 4-1f). 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 ΔV 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 n mi for the 100-n mi TM swath width.

4.2 LAUNCH VEHICLE SELECTIONS

The launch vehicle selection study determined the payload insertion capability of those launch vehicles that show promise as feasible EOS boosters.

The various EOS configurations, when taken with and without 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.

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 n mi, 98 deg inclination, a payload weight up to 2660 lb; 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 Delta 3910 whose maximum payload capability at 366 n mi is 3730 lb. The non-Shuttle EOS-C weight range is 4016 to 5130 lb and its suggested launch vehicle is the Titan IIIB (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 Shuttle, as may be seen by the bar chart (Fig. 4-2). A resupply mission, with payload range 5813 to 8684 lb, is also well within the Shuttle 9600-lb lift-circularize-and-rendezvous capability at the 366 n mi altitude.

4.3 INSTRUMENT APPROACH

This study has evaluated the competitive point designs provided for the proposed instruments, TM, HRPI, synthetic aperture radar, and passive multichannel microwave radiometer (Fig. 4-3 through 4-5). The results are:

- No single point design is considered optimum in the form proposed by the sellers
- The object plane scanner as a class offers significant growth potential relative to the EOS baseline without significant weight growth
- Spectral band selection by filtration techniques offers significantly better growth potential than does the spectrometer (dispersion) approach
- The reduction in preamplifier noise by cooling down to 200° K promises performance improvements for silicon detectors even in Band 1, making them highly competitive with photomultiplier tubes
- 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 signal-to-noise ratio requirements
- 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
- There are significant economies in obtaining the TM and HRPI from the same supplier due to a commonality factor possibly as high as 80%
- A new TM has been defined which can provide a 330-km swath at 27 m resolution, provide an output at 80 m completely compatible with, and providing a backup to, the operational MSS, and providing a pseudo-HRPI output covering a selectable 35-km swath at 30 m. Both the MSS backup and pseudo-HRPI signal would be compatible with the present DOI and planned low-cost ground stations
- Only six-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
- As the Land Resources Mission matures, the desirability of obtaining stereo coverage will increase, and an allowable drift up to ±50 n mi in the orbit prior to orbit adjust is preferred.

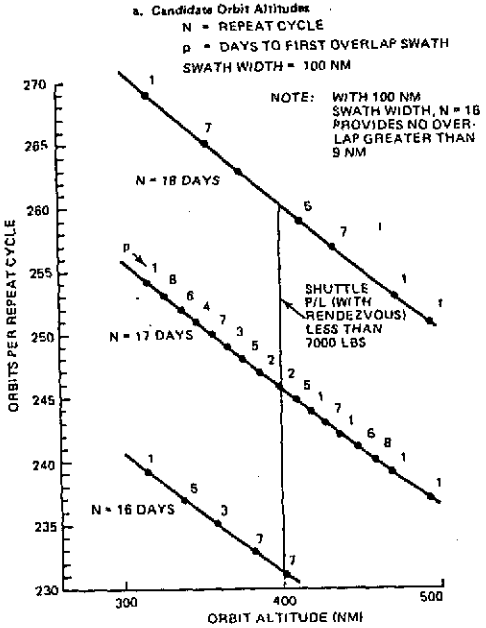
4.4 DATA OPERATIONS

The Central Data Processing Facility (CDPF) performs the data operations for the EOS program. Cost drivers that impact the facility include the daily data volume (throughput), the level of processing of these 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. To exercise the cost impact on the configuring and operation of a CDPF of these and other parameters, a cost/throughput model was constructed that 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 CDPF configurations (minicomputer systems and general purpose processor). Figure 4-6 illustrates the resulting cost-throughput relationships.

Exercising the cost/throughput model and analyzing the results lead 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 general purpose computer systems. The cost appears to behave roughly as a linear function of requirement parameters

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	6 (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)

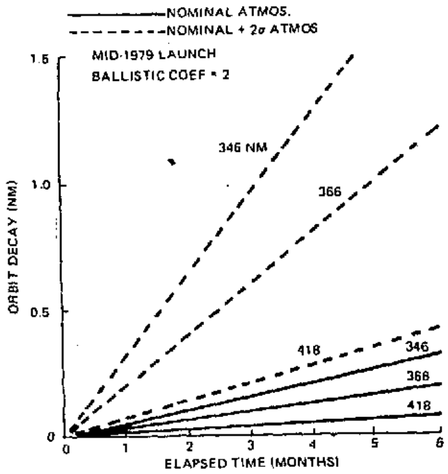
*NUMBER IN PARENTHESIS SHOWS DAYS TO ASSURE HRPI REVISIT TO AT LEAST 90% OF HRPI ACCESSIBLE TARGETS IF THAT OCCURS EARLIER THAN THE ASSURED ASSURED 100% DATE

+MAXIMUM HRPI OFFSET ANGLE = 30°

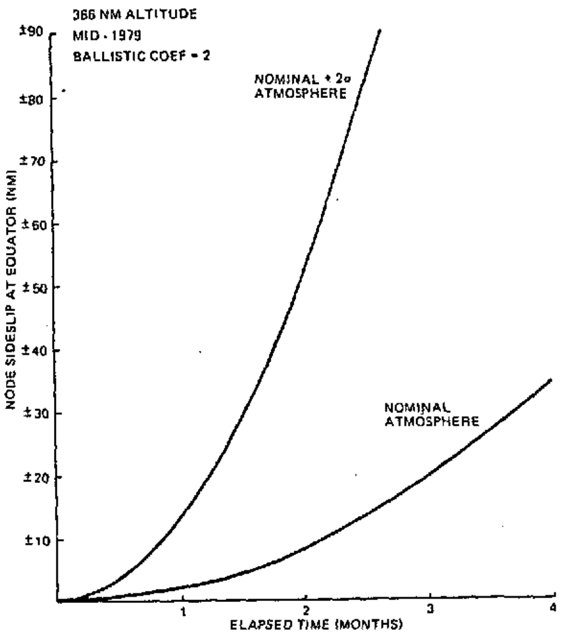
c. Orbit Altitude Selection Process

	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

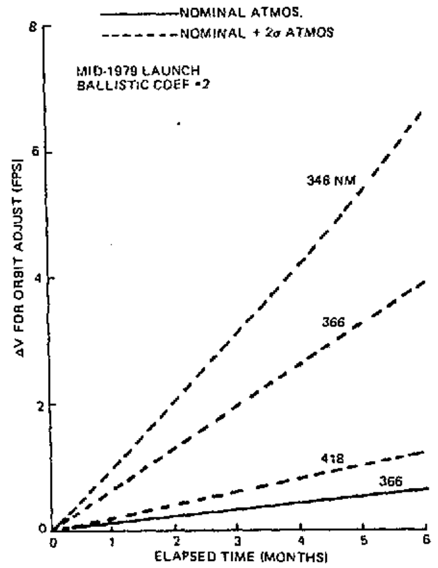


Fig. 4-1 Orbit Altitude Selection

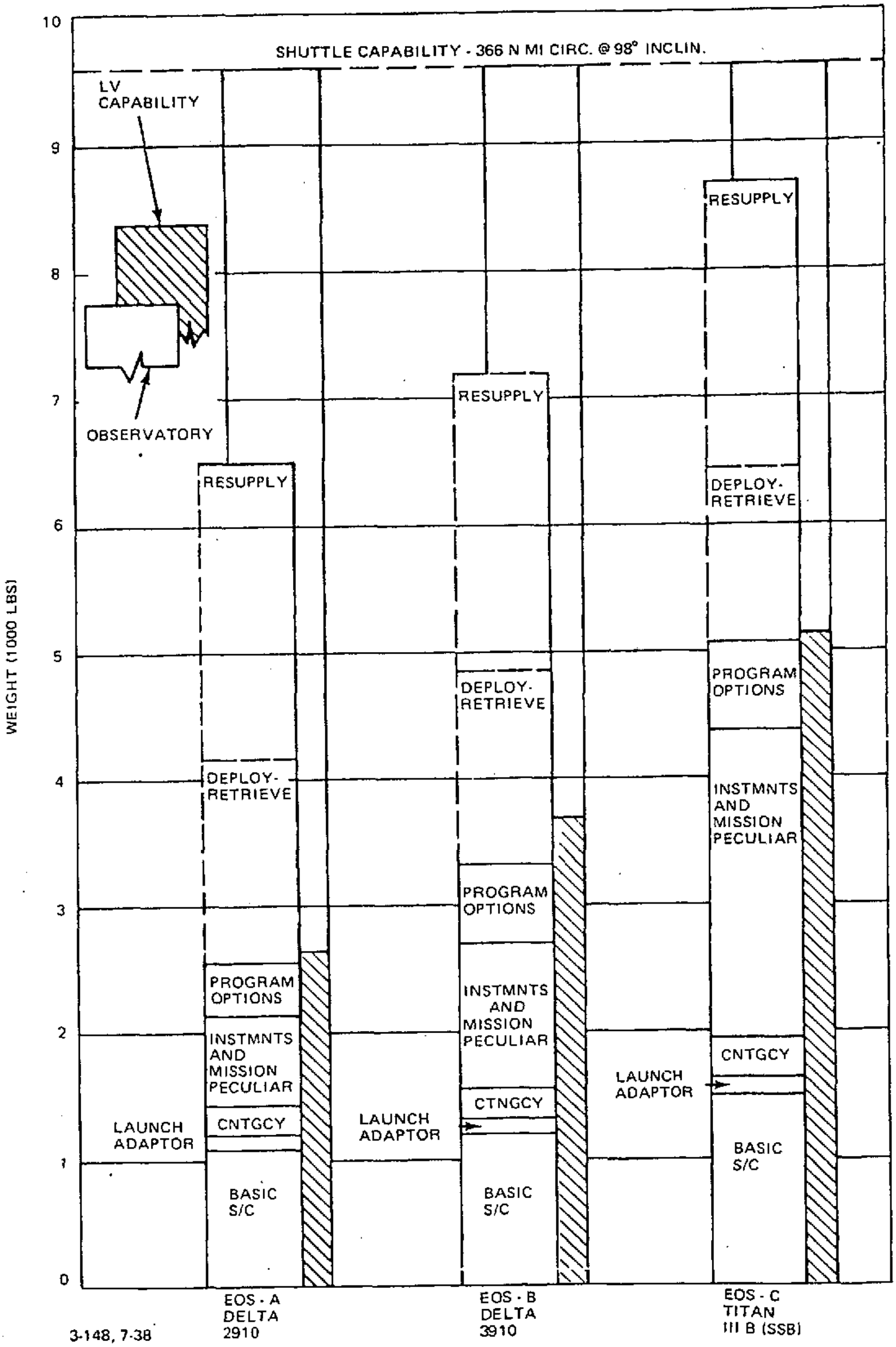
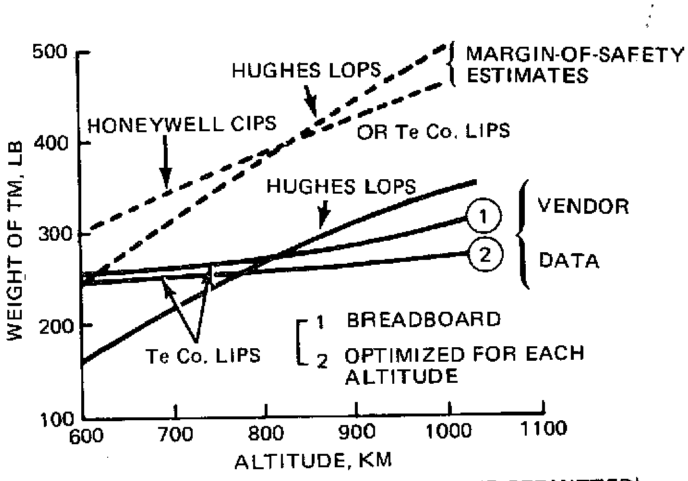


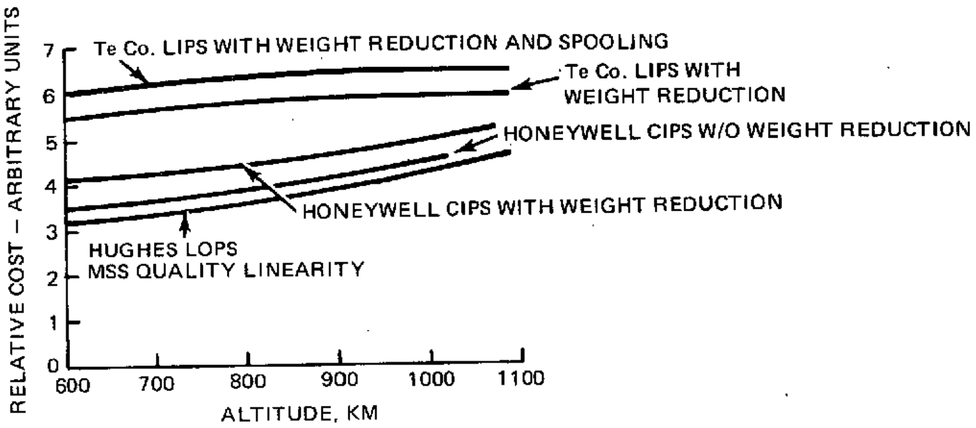
Fig. 4-2 Launch Vehicle Capability

7

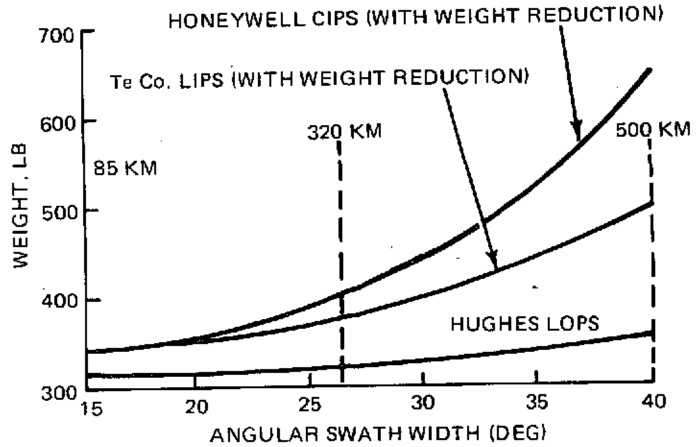


(NOTE: INSTRUMENTS SIZED FOR MINIMUM SNR PERMITTED)

7-40 **Fig. 4-3 Anticipated 185-Km TM Weight Vs Altitude**



7-41 **Fig. 4-4 Anticipated Intrinsic Cost-Difficulty Trends for 185-Km TM**



7-42 **Fig. 4-5 Anticipated Growth of TM Weight with Swath Width for 30-m Resolution at 680-Km Altitude**

- Studies subsequent to the data operations trades indicated economical hardware using special design processors or an array processor (Goodyear STARAN) provides lower costs above 20 scenes per day and to the maximum of 10^{12} bits per day
- The number of user 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 expendable 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.

In summary, the best trade between a flexible processor and an economical system indicates that the array image data correction processor should be implemented.

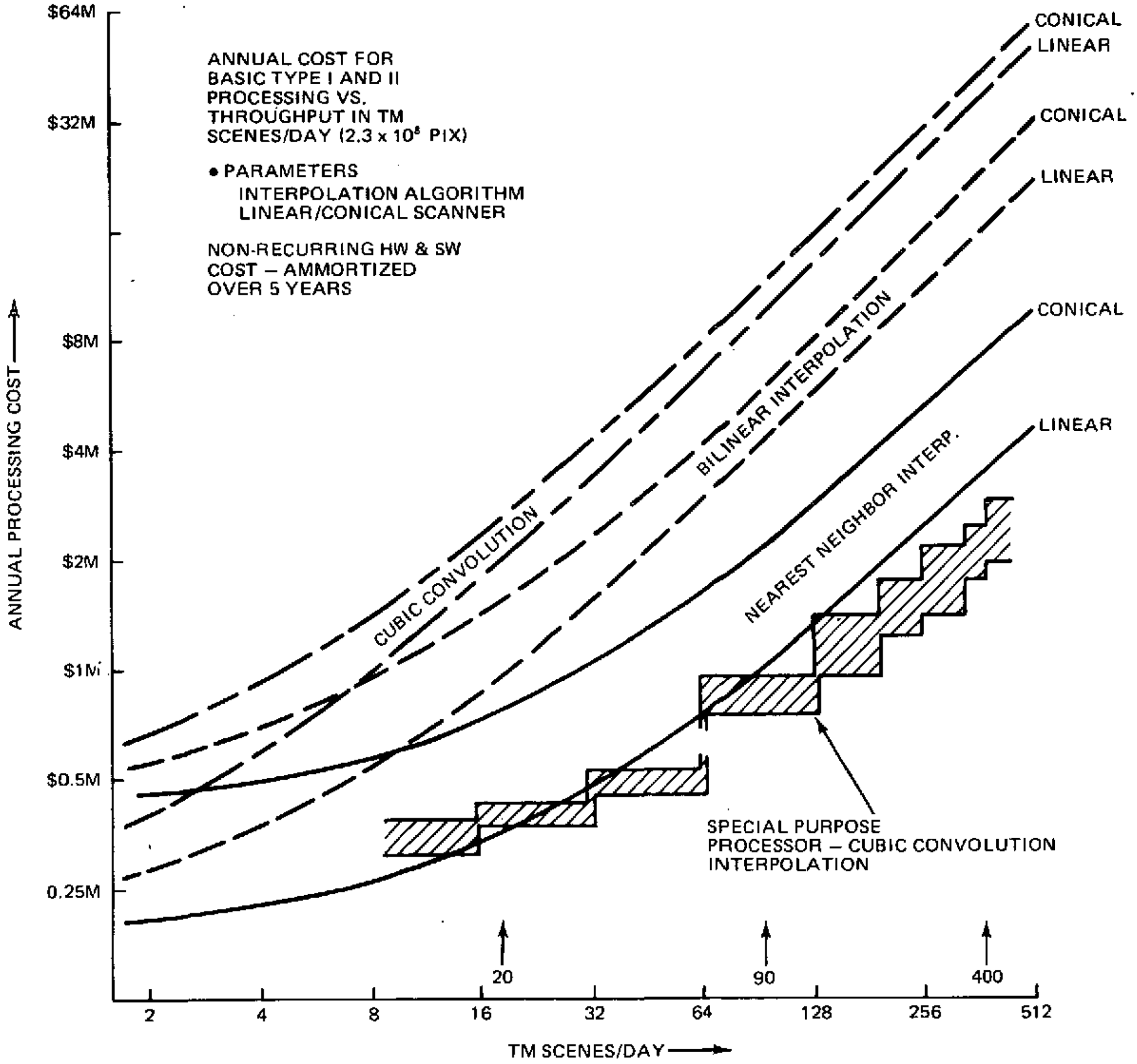


Fig. 4-6 Annual Processing Cost vs Throughput in TM Scenes per Day

4.5 ACS/CPF TRADEOFF

The purpose of this study has been to determine those ACS performance requirements that result in the lowest ACS/Central Processing Facility cost for a program of selected missions, and at the same time provide flexibility for meeting varying mission requirements.

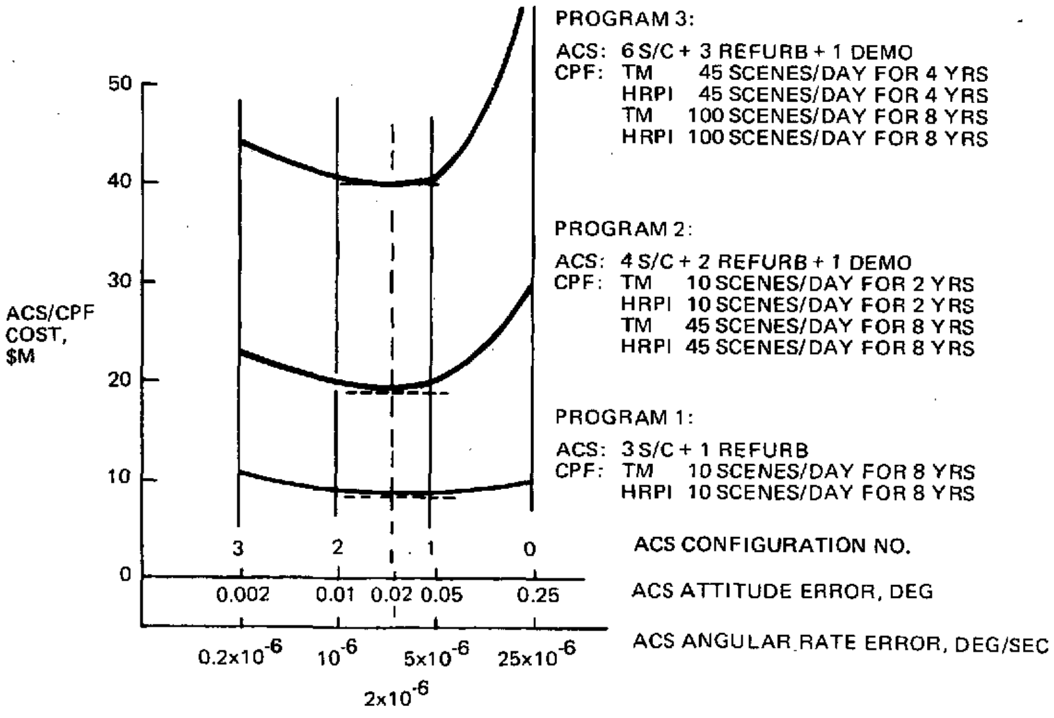
As shown in Fig. 4-7, the ACS/CPF cost is minimum for ACS configurations No. 1 (low cost) and No. 2 (baseline) for each of the three programs. These configurations have the following performance requirements:

Performance Error	ACS Configuration No. 1	ACS Configuration No. 2
ACS Pointing, deg	0.05	0.01
ACS Angular Rate, deg/sec (average over 30 minutes)	5×10^{-6}	10^{-6}

Since ACS configuration No. 2 has a performance which is five times better than that of ACS configuration No. 1, ACS configuration No. 2 is best on the basis of ACS/CPF cost and mission flexibility.

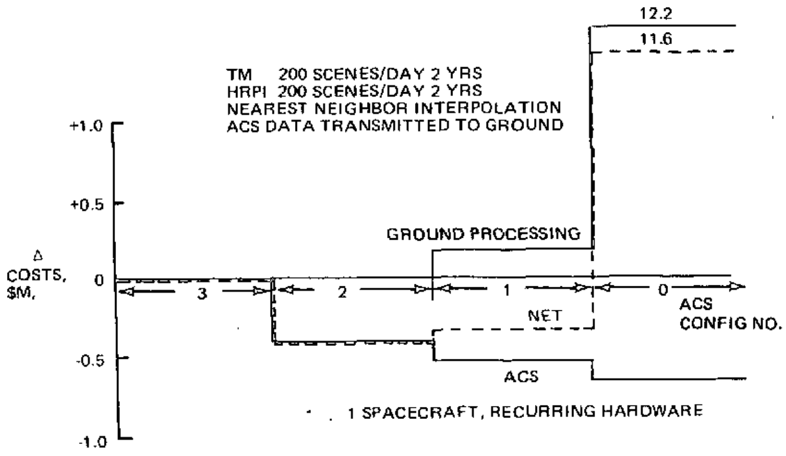
The results on a "per-spacecraft" basis are similar. As shown in Fig. 4-8, with decreasing ACS performance, the ΔACS cost goes down and the ΔCPF cost goes up. ACS configurations No. 3, 2, 1, and 0 have errors that are 0.2, 1, 5 and 25 times those of baseline at 0.01 deg and 10^{-6} deg/sec. The net ΔACS/CPF cost decreases in going from ACS configuration No. 3 to 2 remains approximately the same in going to ACS configuration No. 1, and increases sharply in going to ACS configuration No. 0. Thus, the net ΔACS/CPF cost is lowest for ACS configurations No. 1 and 2.

When the effects of increasing the number of scenes/day are examined, the results are again similar. As shown in Fig. 4-9, the recurring ACS hardware/manpower costs for one spacecraft are plotted at zero 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 No. 0 is cost competitive with ACS configurations No. 1, 2, and 3.

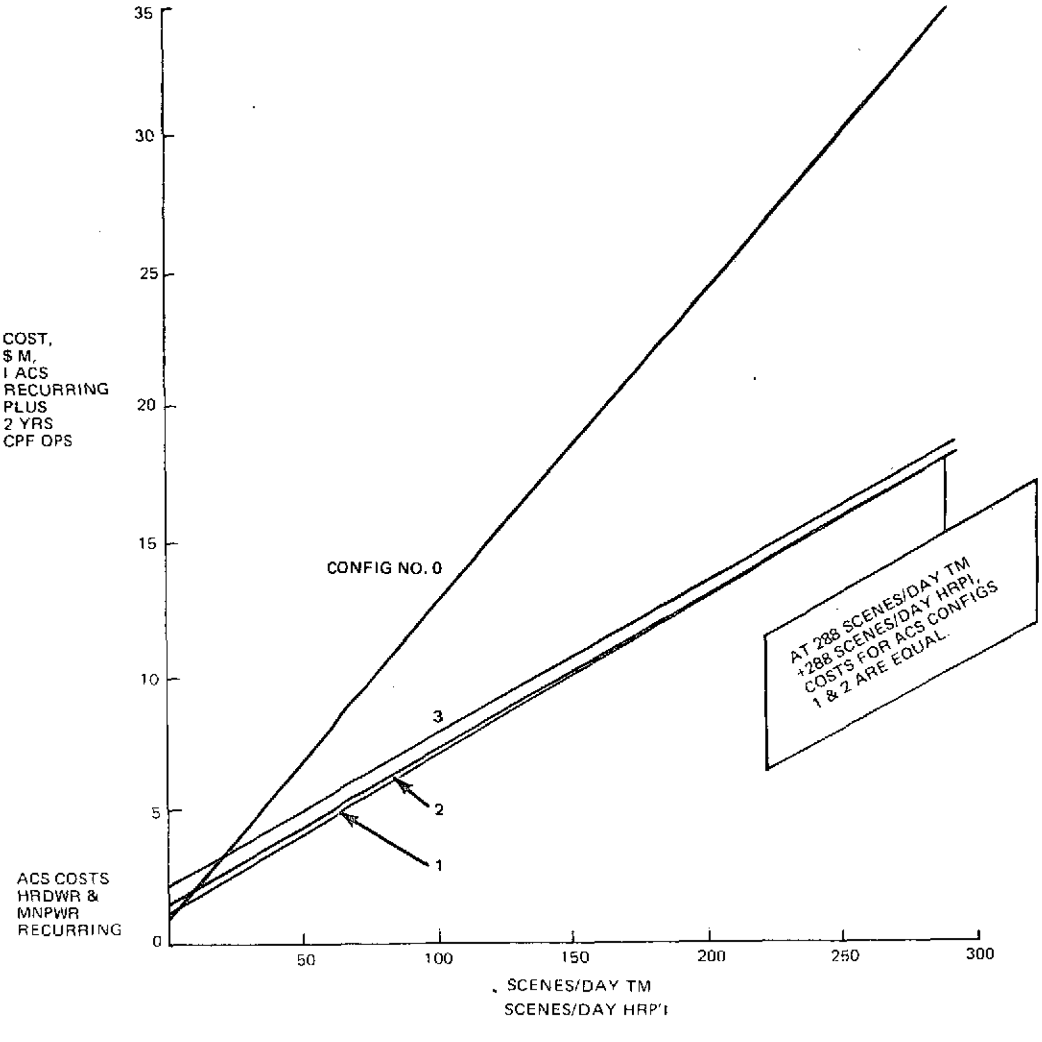


3-155
7-29

Fig. 4-7 ACS/CPF Cost Vs ACS Performance



3-156, 7-30 **Fig. 4-8 Delta Costs Vs ACS Configuration**



3-157
7-31

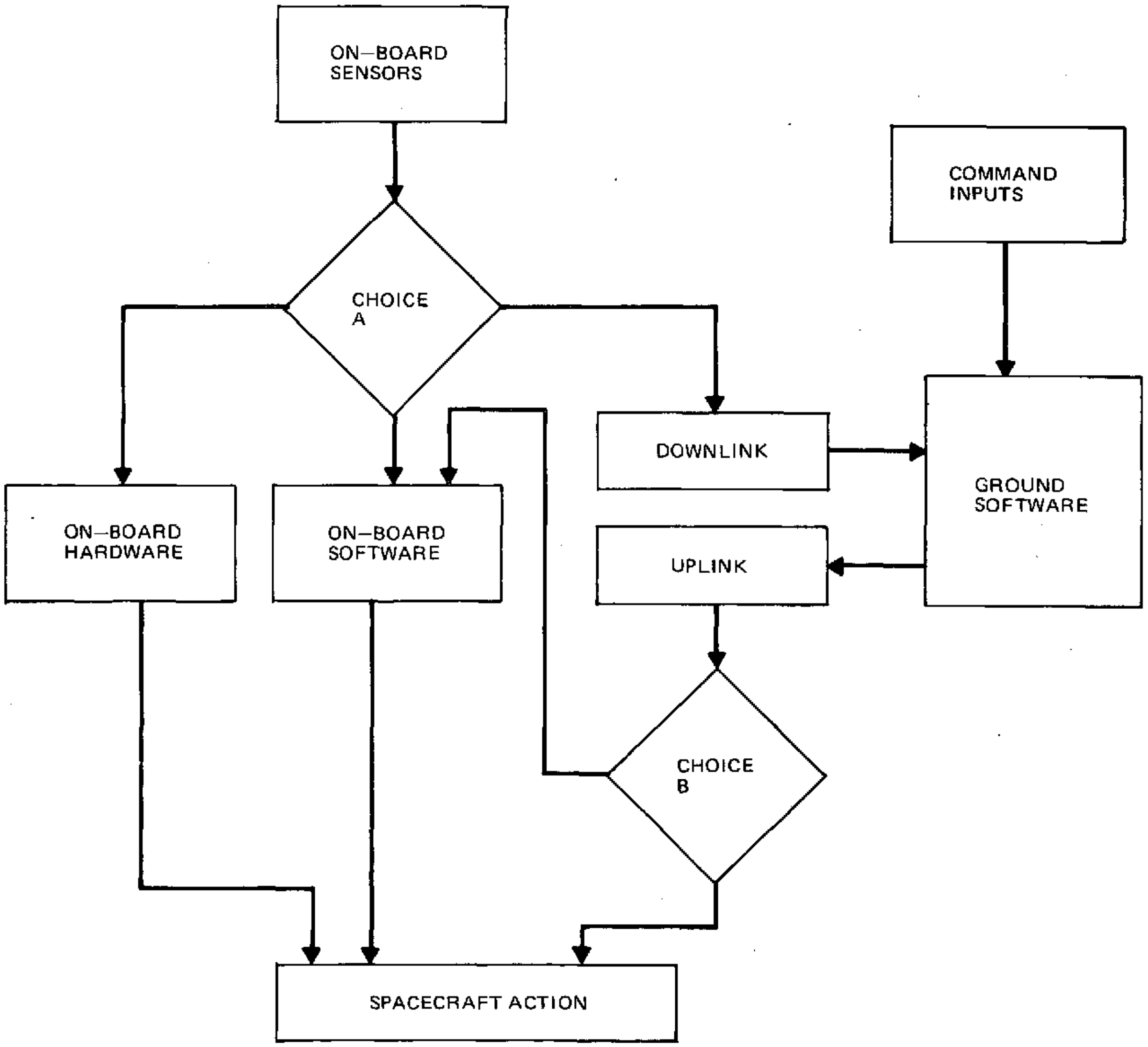
Fig. 4-9 Cost, ACS/CPF Vs Scenes per Day

When the number of scenes/day is higher than 20, ACS configuration No. 0 is not cost-competitive and ACS configurations No. 1 and 2 are lowest in cost, with ACS configuration No. 3 somewhat higher in cost.

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, ±0.01 deg; angular rate stability, $\pm 10^{-6}$ deg/sec over 30 min.

4.6 SPACECRAFT AUTONOMY/HARDWARE VS SOFTWARE

The trade study of spacecraft autonomy and of hardware versus software has involved a function-by-function resolution of the choices illustrated in Fig. 4-10. While each function of the spacecraft is a candidate for examination, care should be taken in the implementation of each autonomous function to assure that ground control is not inhibited and remains available as a backup. This study considers representative func-



3-158
7-39

Fig. 4-10 Autonomy/Hardware/Software Trades tions, 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.

Of all of 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.

4.7 INTERNATIONAL DATA ACQUISITION (IDA)

Methods for acquiring EOS data from areas other than CONUS have been defined during the current study with each having their own peculiar impact on the program. These options include

- Option 1: Direct transmission (DT) 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's). This configuration is primarily intended for use with an International Data Acquisition (IDA) mission involving relatively low data volume, such as wheat crop only.

The relative performance rating of each IDA option (less the hybrid), shown in Table 4-1, is 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 two-site (Alaska and NTTF) WBVTR configuration, the DT system, and finally, the single-site (Alaska) WBVTR system.

The costs of each of the three primary IDA options and hybrid system configuration are given in Table 4-2. The costs of the options have been derived from these data on dollars per minute of available data. The following conclusions have emerged:

- The WBVTR option is somewhat lower in cost than TDRSS. This is due to the $2 million per year allocated for data transmission between the TDRSS ground station and the processing center. If these costs are not included, TDRSS becomes the lowest cost option, assuming "bandwidth-time" (BT) pricing
- The hybrid configuration of DT and WBVTR is a low risk and low cost option for low volume data missions, such as single crop monitoring
- The DT option is not cost effective; it is outperformed by both TDRSS and WBVTR on high data volume (tilled land), and by the hybrid on low volume (wheat crop data)
- In conclusion, for the large data volume IDA requirements TDRSS is the most cost effective system if changes are on a bandwidth-time basis.

4.8 USER/SCIENCE AND ORBIT TIME OF DAY STUDIES

The purpose of this study has been to organize the user requirements for the spacecraft and instruments to provide guidelines for design evaluation. The conclusions are as follows:

- EOS spacecraft design should be flexible with respect to orbit time of day
- EOS data of 30-m resolution will satisfy 77% of the user applications. Capability of providing 10-m resolution is desirable to meet the requirements of the remaining 23% of applications
- The four MSS spectral bands will satisfy 72% of the user applications. The additional three bands provided by the TM are desirable to satisfy the remaining user applications
- Spectral bands specified for the TM are all useful. Relative priority of the seven bands are: MSS bands 1,2,3, and 4 are first priority; the thermal IR Band 7, (10.4 to 12.6μ) is second priority; signal-to-noise problems in Band 6 (2.08 to 2.35μ) may make this band of marginal value

Table 4-1 International Data Acquisition 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%)
DT	3 (53%)	3 (65%)	2 (91.5%)
WBVTR 1 SITE	4 (45.7%)	4 (56%)	4 (84%)

3-136
7T-53

Table 4-2 System Cost Breakdowns
(1974 Dollars in Millions)

OPTION	EARTH TERMINAL	SPACECRAFT COSTS +	$M/YEAR DATA PROCESSING & HANDLING COSTS	TOTAL COST (COST IMPACT TO EOS)**
1. DT WITH SIX REGIONAL STATIONS	6	–	4.2	10.2 (0)
2. WBVTR (2 TR'S)	–	2	4.2	6.2 (2)
3. TDRSS	* 25 (BW PRICING) 2.5 (BT PRICING)	1	4.2	30.2 (1) 7.7 (1)
4. HYBRID *** 6 LCGS & WBVTR (1 TR)	0.6	1	0.4	2.0 (1)

*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.

3-137
7T-54

- Radiometric corrections increase in complexity with wider scan angles. The variations in sun angle, atmospheric profiles, ground reflectivity, etc, over the field-of-view are discussed in Appendix D of Report No. 7
- 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
- Monitoring of world food production regions is a very visible application of EOS and warrants emphasis. Table 4-3 lists potential agricultural applications of EOS information systems.

4.9 UTILIZATION OF CONTROL CENTER PERSONNEL

The purpose of this study has been to define Mission Operations (MO) and Project Control Center (PCC) concepts and personnel utilization for EOS.

Figure 4-11 is a functional diagram of the PCC, and is shown for two activities – mission planning and real-time operations. Mission plan-

ORIGINAL PAGE IS OF POOR QUALITY

Table 4-3 Potential Agricultural Applications of EOS Information Systems

	A		B	C	D		E							F				G		H		I					J						
							SPECTRAL BANDS															PRODUCTS					DISTRIBUTION & USER AGENCIES						
							THEMATIC MAPPER							HRPI				SAR															
APPLICATIONS	HIGH SUN ANGLE	LOW SUN ANGLE	FREQUENCY WEEKS	RESOLUTION METERS	RAD. CORRECTIONS	GEOM. CORRECTIONS	0.5-0.6 u	0.6-0.7 u	0.7-0.8 u	0.8-1.1 u	1.55-1.75 u	2.08-2.35 u	10.04-12.64 u	0.5-0.6 u	0.6-0.7 u	0.7-0.8 u	0.8-1.1 u	"X" BAND	"Y" BAND	REGISTR. T.M. BANDS	SCENE GEOGR. LOCAT.	70 uu B&W	9x9 B&W	9x9 COLOR	CCT	MICRO FILM COMP. PRINTOUT	INTERIOR	AGRICULTURE	NOAA	USACE	ONR	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					RADIOMETRIC CORRECTION IS NEEDED WHEN DIFFERENT DATES ARE COMPARED
b. CULTIVATED VS. NON-CULTIVATED	P	A	26	60	N	N	X	1	X	1	X	X	1	X	1	X	1					+	–	–	+			X					
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

3-178, 7T-17

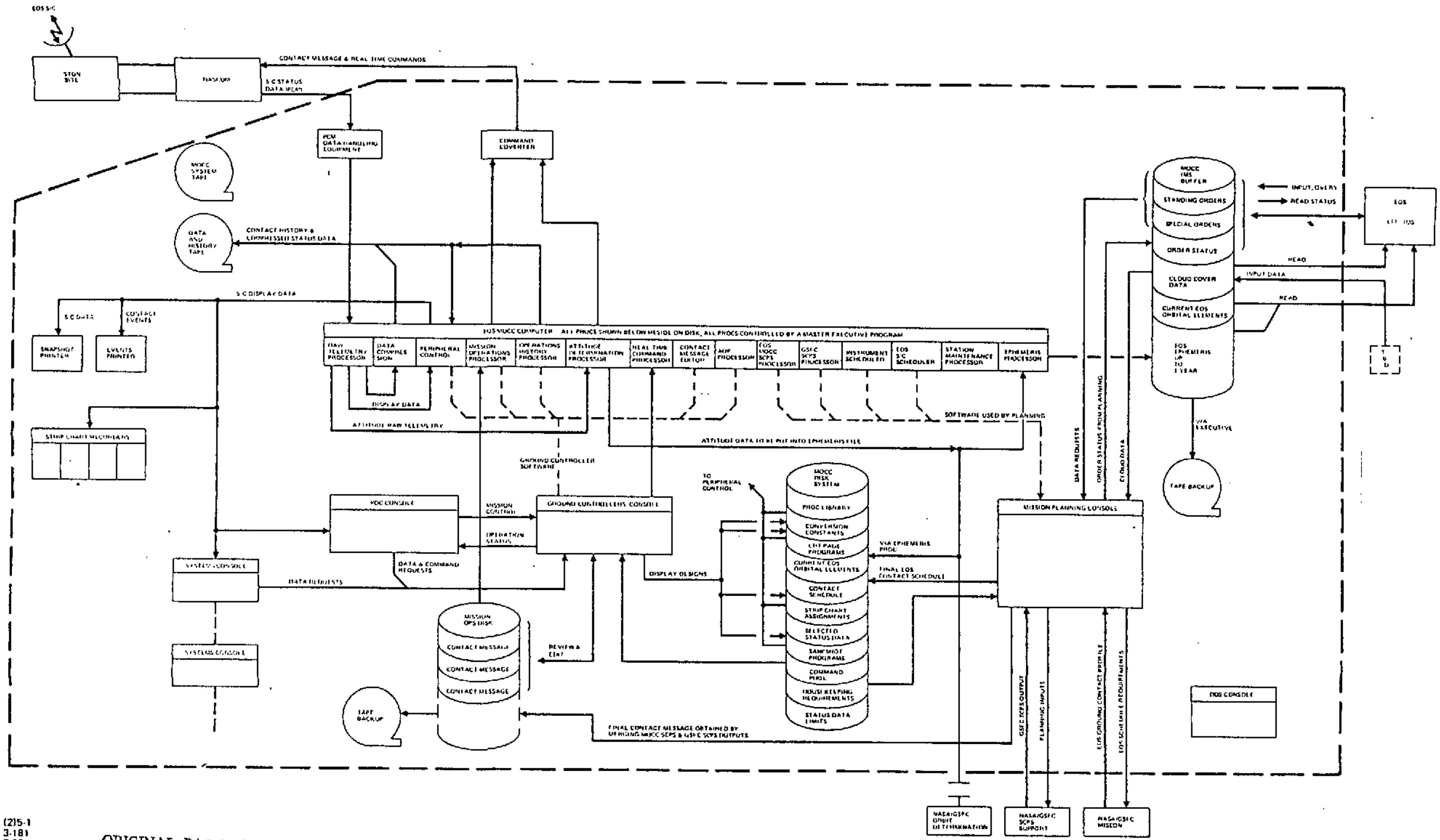


Fig. 4-11 EOS Project Control Center Functional Diagram

(2)5-1
3-181
7-32

ORIGINAL PAGE IS
OF POOR QUALITY

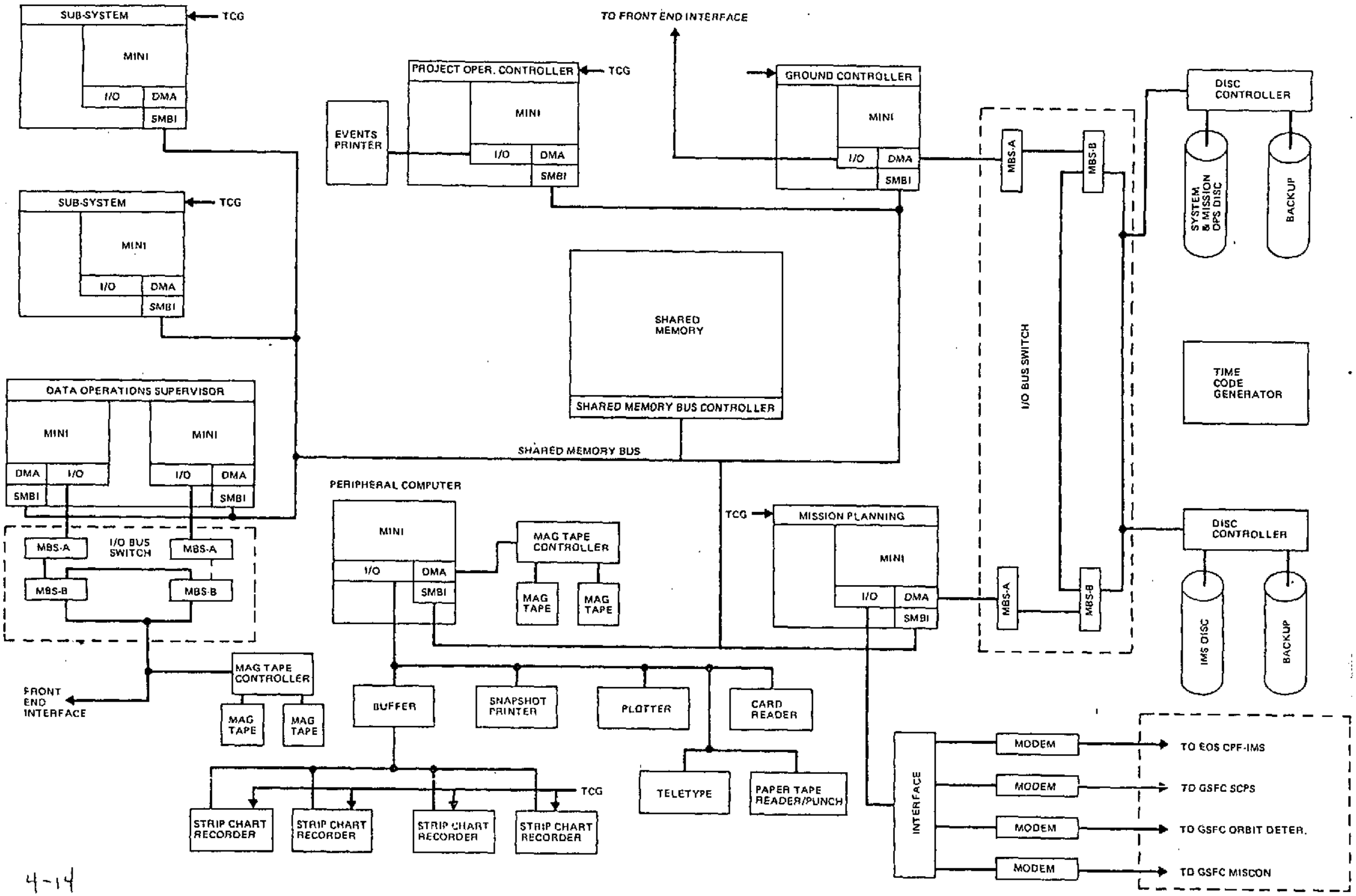


Fig. 4-12 PCC System Diagram—Grouped Mini

3-183
7-33

4-14

FOLDOUT FRAME /

ning 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 MO disk. Real-time operations involves the control and status determination of the observatory on a real-time basis. Housekeeping data enters the PCC via STDN and NASCOM, and is manipulated to drive the various displays and peripherals. Command generation is initiated by the Ground Controller and is relayed to the observatory via NASCOM and STDN.

Two configurations for the PCC were evolved – the grouped mini and the midi configurations. Figure 4-12 shows the grouped mini system. It consists of multiple minicomputers 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 console's function. Each computer (console) communicates with the other computers via a shared memory. Access to the shared memory is on a priority basis. Figure 4-13 shows the midi configuration. It is the conventional approach to a control center. Two midicomputers operate in a multiprocessing environment. The capability exists for either computer to sustain the activities of the PCC in some satisfactory, but reduced, mode in the event of failure of the other computer. 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.

The front end portion of the PCC (Fig. 4-14) will be the same for either of the configurations. The only unique part will be the front end interface unit that will interface with the computer per-

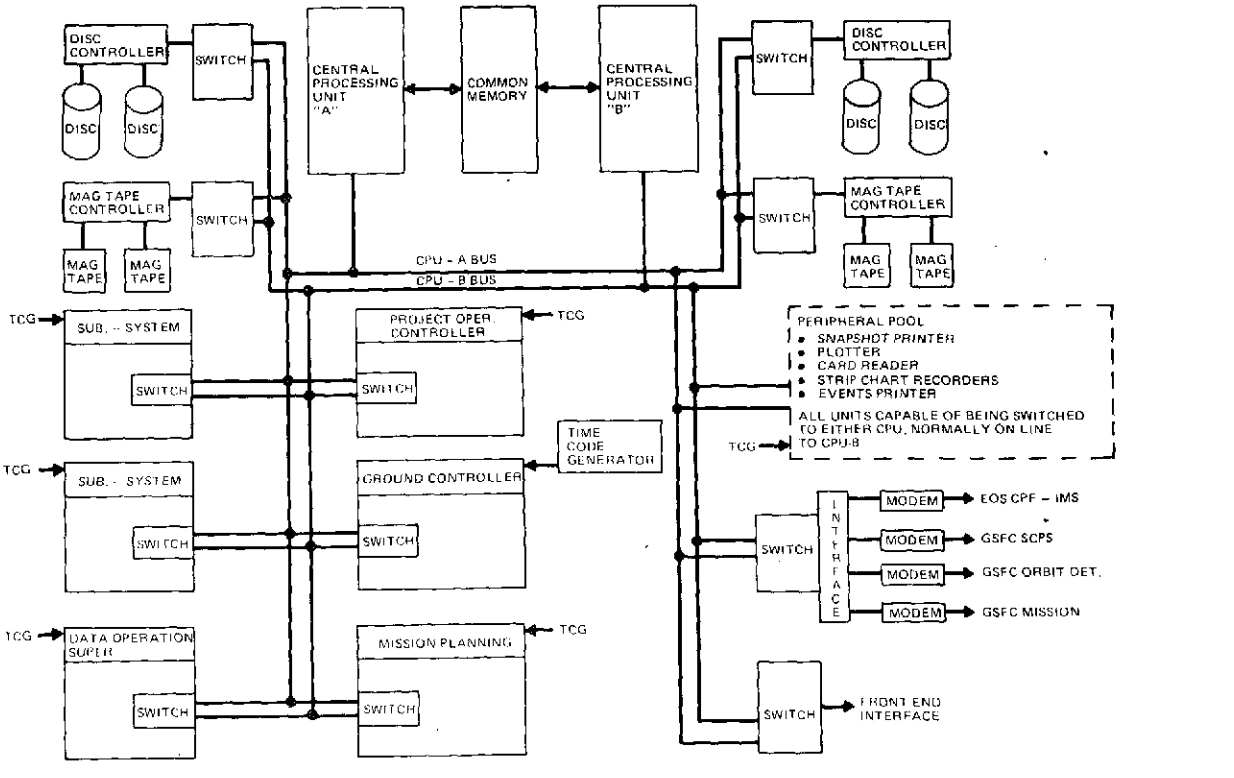


Fig. 4-13 PCC System Diagram-Central Midi

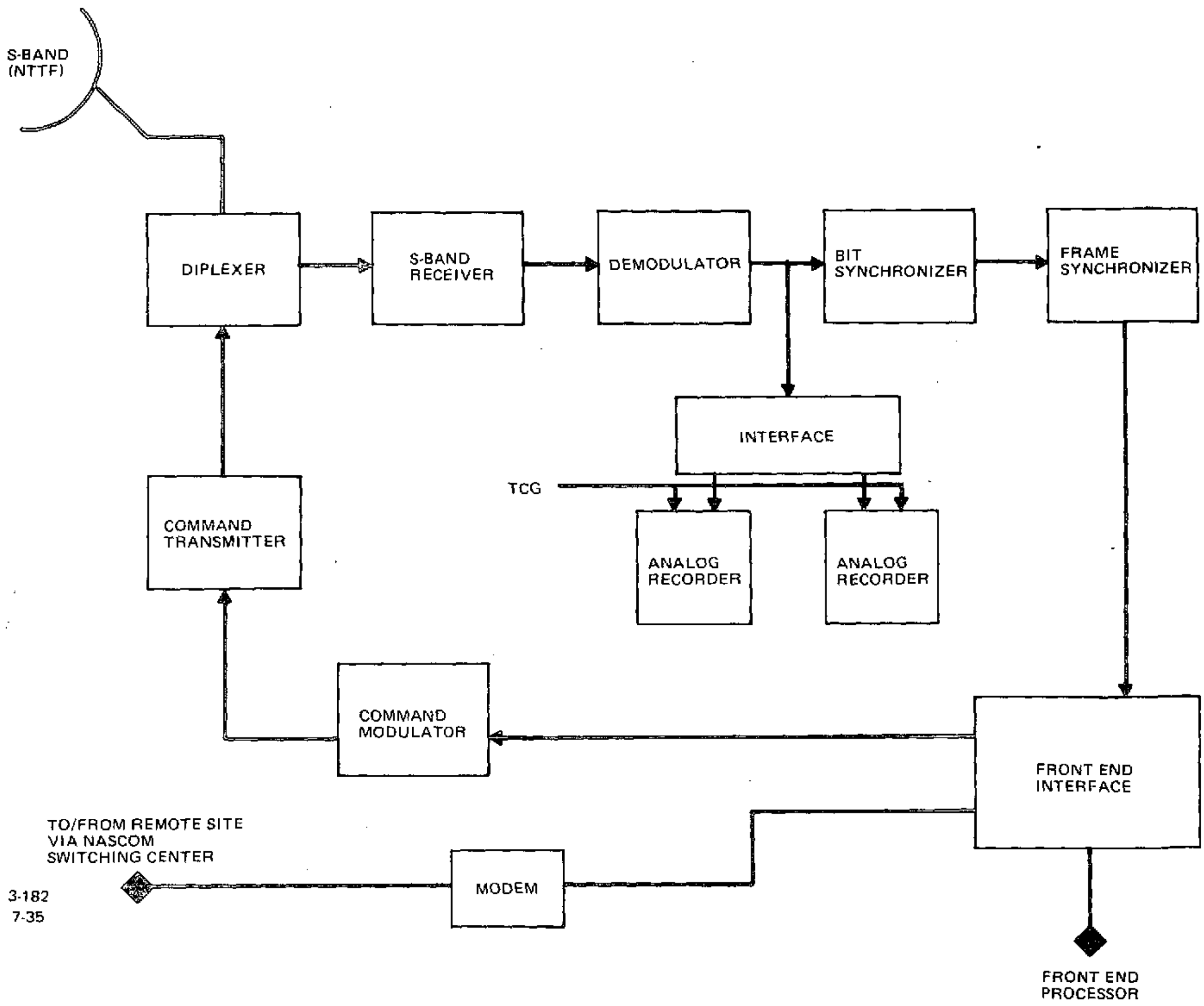


Fig. 4-14 EOS – PCC System Diagram Front End

forming the front end processing function.

The PCC staffing requirement is a function of three major factors:

(1) Number of spacecraft in orbit
(2) Level of advanced on-board processor (AOP) usage
(3) Management confidence in the overall operation.

To support a single spacecraft with minimal AOP involvement, a total of 56 people are required. This could be reduced to 39 people as use of the AOP is maximized. This staffing level assumes a four-team operation giving round-the-clock coverage in the PCC. Reduced coverage down to a two-team effort (a decision based on overall confidence in the operation) could allow these numbers to be reduced to 36 and 27, respectively. The addition of a second spacecraft to be handled in the PCC requires additional people. For a four-shift operation, the number of new people varies from 14 to 27, depending upon the level of AOP usage. For a two-team operation, these figures vary from 11 to 17, respectively.

Primary PCC activities will be mission planning, real-time operations, and mission analysis. PCC personnel will train in the T&I area as test conductors, and when necessary, PCC personnel will be off-loaded into the T&I area.

The baseline PCC 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.

4.10 WIDE BAND DATA FORMAT

It is possible to define optimum wide band digital (instrument) formats in the sense that data acquisition, processing, and user product generation are accomplished efficiently with a minimum time and equipment cost for reformatting and handling the wide band data. The wide band format study identified the constraints involved in selecting the best format for the data. These include:

- Constraints in the payload instruments
- Constraints at the ground stations
- Constraints imposed by the data users.

Four formats were identified for consideration. These were:

1. Pixel/Detector - Interleaved (P/DI) – The format arising out of the sampling of the detectors in the sensor
2. Pixel-Interleaved (PI) – Pixel 1 of band 1, pixel 1 of band 2, . . . pixel 1 of band N, pixel 1 of band 1, repeat for each line. (Probably best for analysis where all spectral bands are required)
3. Line-Sequential (LS) – Pixel 1 of band 1, pixel 2 of band 1, . . ., pixel M of band 1, pixel 1 of band 2, pixel M of band 2, . . ., pixel M of band N, repeat for each line
4. Band-Sequential (BS) – Pixel 1 of line 1, pixel 2 of line 1, . . ., pixel M of line 1, pixel 1 of line 2, . . ., pixel M of line 2, . . ., pixel M of line L, repeat for each band. (Probably best when only one spectral band is needed).

Tables 4-4 through 4-6 identify four format options for various elements of the CDPF. Table 4-7 shows the results of an evaluation of these options using a score range of 1 to 10. 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%) appear to be required in this format. The second-best option is to leave the data in the "natural" format (P/DI) throughout the processing and reformat only at the completion of all processing.

The evaluation in Table 4-7 has resulted from an initial treatment of the problem. 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.

4.11 MODULARITY LEVEL FOR STANDARD MODULES

The purpose of this study has been to assess the baseline and alternative modularity levels for the standard modules and determine the most economic approach for EOS-A.

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-in. 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 4-15 shows the preliminary distribution of equipment within the submodules. 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. Conclusions are that:

Table 4-4 Summary of Format/Processing Options

PROCESSING \ FORMAT	PIXEL/DETECTOR INTERLEAVED P/DI	PIXEL INTERLEAVED PI	LINE SEQUENTIAL LS	BAND SEQUENTIAL BS
TYPE I	BEST	← SECOND BEST	SECOND BEST →	VERY INEFFICIENT
TYPE II	← PROBABLY EQUALLY GOOD	PROBABLY EQUALLY GOOD →	SLIGHTLY LESS EFFICIENT	RELATIVELY INEFFICIENT
TYPE III	← THIRD BEST	THIRD BEST →	SECOND BEST	BEST

Table 4-5 Storage Requirements for Formatting

FROM \ TO	P/DI	PI	LS	BS
P/DI	–	ONE SWATH 3.8 x 10^6 BITS	ONE SWATH 3.8 x 10^6 BITS	ONE SCENE 2 x 10^9 BITS
PI	ONE SWATH	–	ONE LINE ALL BANDS 2.5 x 10^5 BITS	ONE SCENE 2 x 10^9 BITS
LS	ONE SWATH	ONE LINE ALL BANDS	–	ONE SCENE 2 x 10^9 BITS
BS	← ONE SCENE	ONE SCENE	ONE SCENE →	–

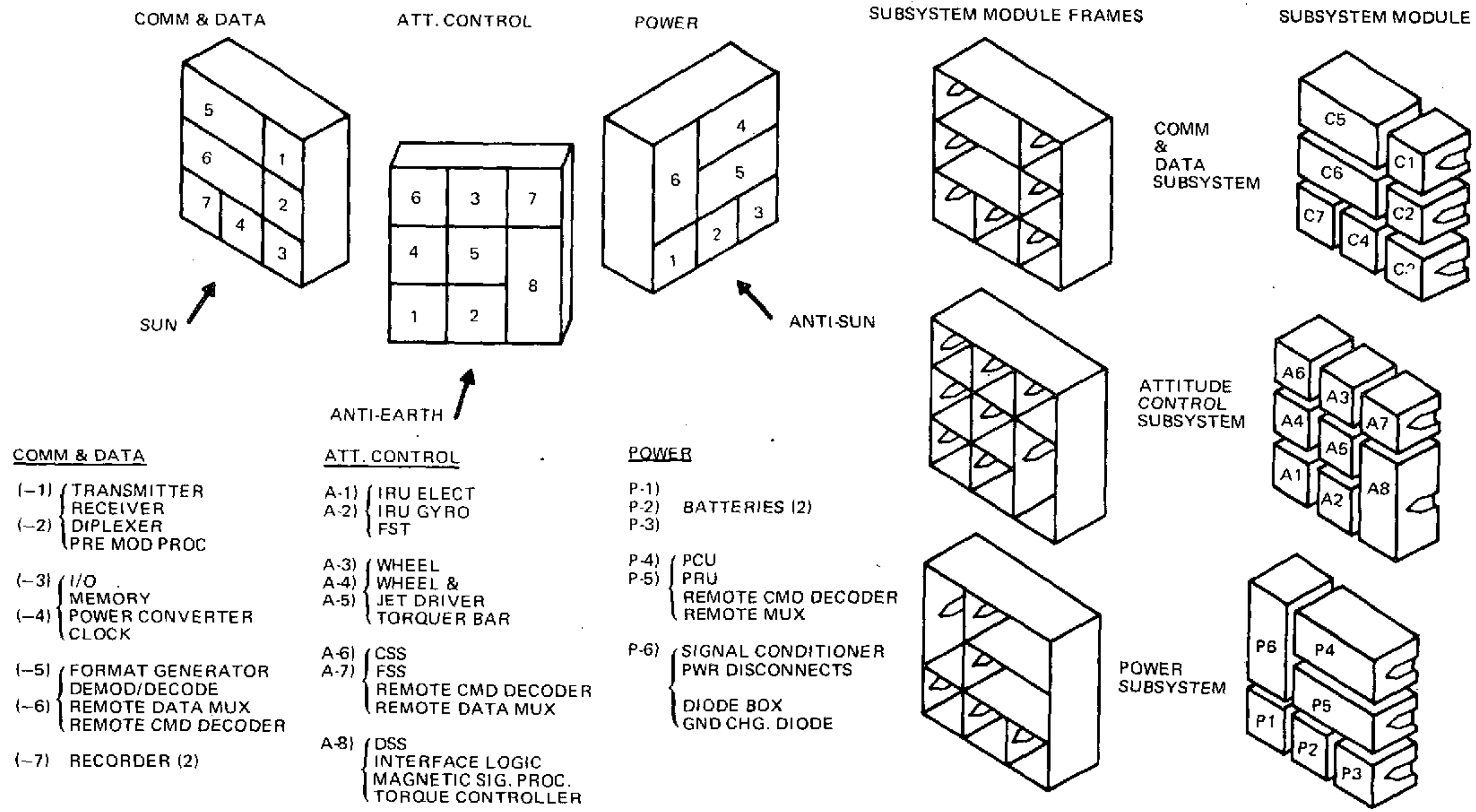
Table 4-6 Candidate Data Formats

OPTION	ACQUISITION INITIAL RECORDING	LEVEL I	ARCHIVE	LEVEL II	LEVEL III	USER PRODUCT (TAPE) GENERATION
A	P/DI	P/DI	P/DI	P/DI	P/DI	PI, LS, BS
B	P/DI	PI	PI	PI	PI	PI, LS, BS
C	P/DI	LS	LS	LS	LS	PI, LS, BS
D	PI	PI	PI	PI	PI	PI, LS, BS

Table 4-7 Format Evaluation Options

OPTION \ EVALUATION CRITERIA	INTERMEDIATE REFORMAT EFFORT	LEVEL I	LEVEL II	LEVEL 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

7T-44



3-194
7-25

Fig. 4-15 Subsystem Module Configuration – Modularity Approach

- Integrated subsystems have potential weight savings, but preclude on-orbit servicing
- Subsystem submodules have potential program cost savings (spares, refurbishment), but the weight penalty precludes launch on Delta 2910
- Baseline modularity level (subsystem modules) provides for both on-orbit service and Delta 2910 launch.

4.12 FOLLOW-ON MISSION ECONOMIC STUDY

This study determined the economic benefits in utilizing multi-purpose spacecraft to capture varying numbers of earth observation missions, and evaluated the cost impact of extending the baseline design to capture follow-on missions.

The study is basically a cost comparison of multiple-mission spacecraft against the corresponding single-mission spacecraft 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 where 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.

Results are presented in Table 4-8 for spacecraft of varying missions capabilities and the cost savings are indicated as percentages which accrue when multiple-mission capability spacecraft are used to fly the indicated missions. The percentages were derived by first computing the total cost of flying all missions (A thru E, SEASAT A, SEOS, SMM and EGRET) with a single-mission spacecraft. This cost (including both DDT&E and Production) was used as cost reference. The total cost for all missions was then computed for mixes of single-mission and various levels of multiple-mission capability spacecraft. The difference of total costs between the all single-mission case and the mixed case was the total cost saving for the mission model, and it was expressed as a percentage cost reduction from the reference single-mission case. The savings were computed in two cost categories: DDT&E alone, and for combined DDT&E and Production.

Table 4-8 indicates, for example, that flying the mission model with a multiple mission spacecraft, capable of capturing missions A to C and flying the remaining missions by single-mission spacecraft is 4% cheaper in DDT&E compared to flying all missions with single-mission spacecraft. When both DDT&E and Production costs are included the cost saving becomes 1%.

The Grumman baseline design concept was included as a comparison to the single-mission and multiple-mission mixes studied. The Grumman baseline extension approach was not to "build-in" subsystem performance to meet the most stringent mission in the set, but to capture additional missions by adding mission-peculiar subsystems performance capability as required. Conclusions are that:

- 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 cost savings compared to single-mission spacecraft approach were achieved through addition of performance capability to the Grumman basic spacecraft or using a multiple-mission spacecraft capable of capturing all the missions in the mission model.

4.13 MANAGEMENT APPROACH

This trade determined a practical, low-cost way of managing and controlling the EOS program.

Experience shows 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 applied to unit production costs, they have achieved similar results through a DTC approach for the total pro-

Table 4-8 Projected Cost Savings

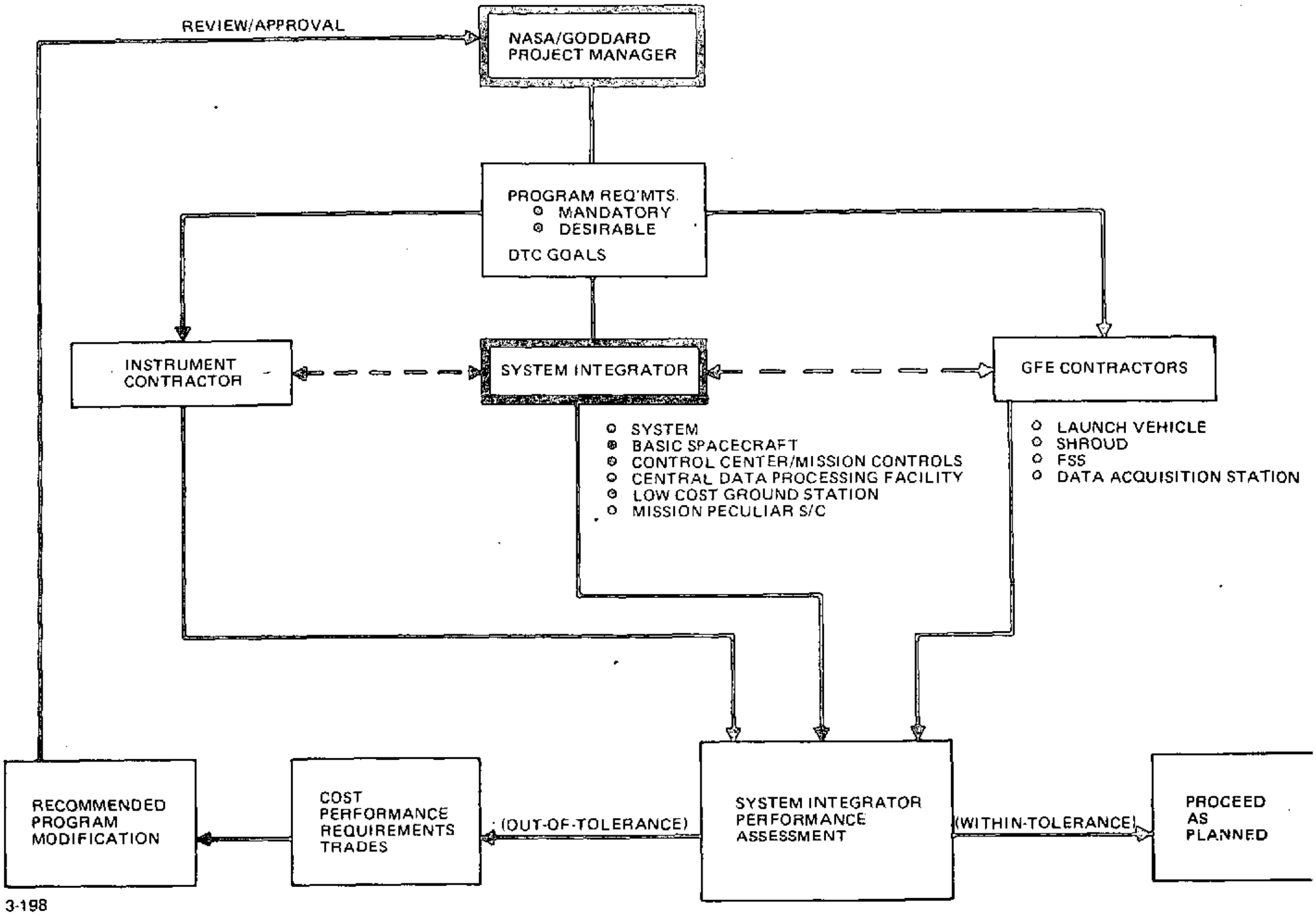
S/C MISSION CAPABILITY LEVEL	MISSION SYMBOL	MISSION NAME	$ SAVINGS IN DDT&E FOR ALL MISSIONS	% SAVINGS IN DDT/E AND PRODUCTION FOR ALL MISSIONS	REMARKS
A ONLY	A	EOS A			ALL
B ONLY	B	EOS B			SINGLE
C ONLY	C	EOS C			MISSION
D ONLY	D	SEASAT B			S/C
E ONLY	E	TIROS O	0	0	COST
F ONLY	F	SEOS			REFERENCE
G ONLY	G	SEASAT A			FOR FOLLOW-
H ONLY	H	SMM			ING CASES
I ONLY	I	EGRET			
A TO C	A				MULTIPLE
	B				MISSION
	C				S/C
D ONLY	D	SEE			
E ONLY	E	ABOVE	4%	1%	SINGLE
F ONLY	F				MISSION
G ONLY	G				S/C
H ONLY	H				
I ONLY	I				
A TO E	A				
	B				MULTIPLE
	C				MISSION
	D	SEE			S/C
	E	ABOVE	18%	10%	
F ONLY	F				
G ONLY	G				SINGLE
H ONLY	H				MISSION
I ONLY	I				S/C
A TO I	A				
	B				
	C				
	D	SEE			MULTIPLE
	E	ABOVE	42%	24%	MISSION
	F				S/C ONLY
	G				
	H				
	I				
GAC B/L	A				
GAC B/L EXT	B				
	C				
	D	SEE			S/C CAPABILITY
	E	ABOVE	35%	24%	EXTENDED
	F				AS REQD
	G				
	H				
	I				

3-195, 7T-7

gram. Since the EOS program has relatively low production volume, and development cost is a major fraction of program cost, the recommended program approach is DTC on a total program acquisition cost basis.

To manage the program implemented in accordance with a DTC approach, we recommend a centralized program manager which we have designated as the System Integrator, as shown in Fig.

4-16. This manager, responsible to the NASA/Goddard EOS Project Manager, is the system contractor for the EOS Basic Spacecraft, PCC (mission), Mission Peculiar Spacecraft Equipment, CDPF, and LUS. The instruments for the initial mission are procured by the Government and provided to the System Integrator. The System Integrator will manage the instrument contractors through the System Integration Team.



3-198
4-1
7-13
7-14
7-15

Fig. 4-16 Design-To-Cost Activity Flow

It is recommended that target costs be established for the Basic Spacecraft and for the EOS-A program. A DTC program should be implemented to achieve this cost. The proposed program requires:

- A System Integration Team concept with direct participation by NASA and associate contractor personnel
- Simplification of controls and documentation
- Direct purchase by NASA of the high technology instruments.

4.14 TEST PHILOSOPHY

The purpose of the test trade studies has been to define and evaluate the influences of the EOS design and system development approaches on the cost of Development, Qualification, Integration and Acceptance testing of the Spacecraft for the EOS Land Resources mission and follow-on missions. Table 4-9 is a listing of the test trade studies.

The significant areas of cost savings/impact identified by the test trade studies are:

- Savings of $500 thousand which represents 50% of the Environmental Acceptance test costs of "business as usual", at virtually no increase in risk, by combining all system and component environmental acceptance tests at the module level
- Modularity and follow-on mission qualification requirements add $125 thousand to the Qualification test cost; however, the flexibility and savings in total test costs provided by the modular Basic Spacecraft, over integrated, dedicated spacecrafts for each mission, more than offset the added $125 thousand in Qualification test costs imposed on the basic EOS program.

Table 4-9 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 DEVEL 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 $40K 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 $20 K 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

*COSTS/SAVINGS EXPRESSED IN 1947 DOLLARS

3-205(1), 7T-16(1)

Table 4-9 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

3-205(2), 7T-16(2)

4.15 SUMMARY – OPTICAL ATMOSPHERIC SCATTERING LIMITATIONS ON OFFSET POINTING PERFORMANCE OF THE EOS

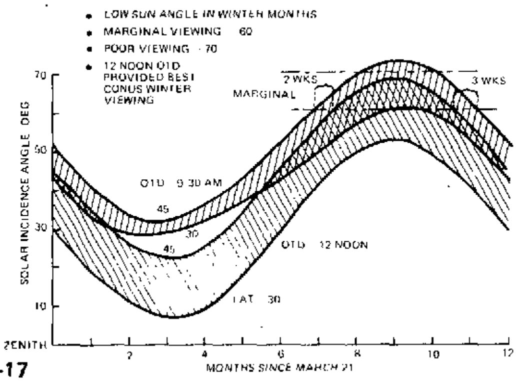
We have calculated the loss of contrast to be expected for the EOS sensors as a result of atmospheric scattering and absorption. Using an average mid-latitude summer model atmosphere and a typical dust aerosol profile, we utilized the Dave-Braslau layered atmospheric model for Rayleigh and Mie scattering to calculate the upward monochromatic light fluxes at the top of the atmosphere as a function of sun angle, viewing angle, and ground (diffuse) reflectivity. Apparent contrast levels of various ground targets and their backgrounds were then derived and compared to assess sensor contrast performance under a variety of conditions. Contrast performance of the sensors relates to their ability to perform boundary following functions and is related to, but does not directly determine, the ability of the sensor to perform target recognition functions.

The relation between sun angle range versus latitude and choice of orbit local time of day is illustrated in Fig. 4-17. Figure 4-18 indicates contrast level as a function of offset angle towards the sun for a ground target reflectivity of 20% versus a background reflectivity of 10% at 0.55 μm for four solar zenith angles. Together, Fig. 4-17 and 4-18 relate the orbit time-of-day to an effective contrast performance level based on average sun zenith angle. Although maximal contrast loss occurs as the sensor scans in the forward solar direction, all forward directions will experience contrast loss with increasing offset angle. In general, contrast levels drop with increasing offset angle and eventually limit the angle off vertical that can be tolerated, while still maintaining contrast fidelity. Broadly speaking, one should expect considerable variation in contrast performance over the FOV of wide-angle sensors.

The dependance of contrast-loss on ground target and background reflectivity is shown in Fig. 4-19 for forward solar scanning, a sun zenith angle of 60 deg, and for $\lambda = 0.55\ \mu$m. It is apparent that user applications for which target and background reflectivities are less than 30% are most sensitive to loss of contrast with offset angle.

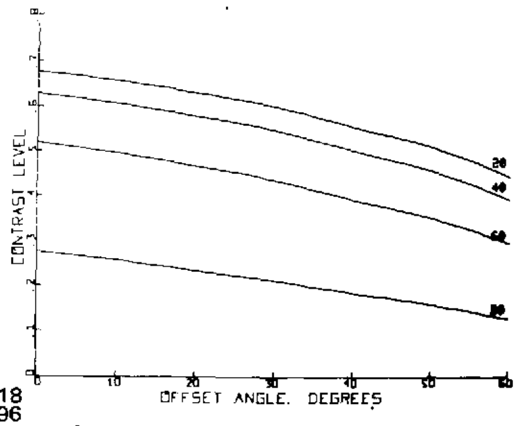
Figure 4-20 indicates the wavelength dependence of contrast loss (for the same reflectivities) and illustrates the fact that contrast performance penalties are the same throughout Bands 1 through 6.

For all cases it is apparent that there is some non-zero angle from the vertical at which a sensor may be canted, away from the sun, to realize optimal contrast performance.



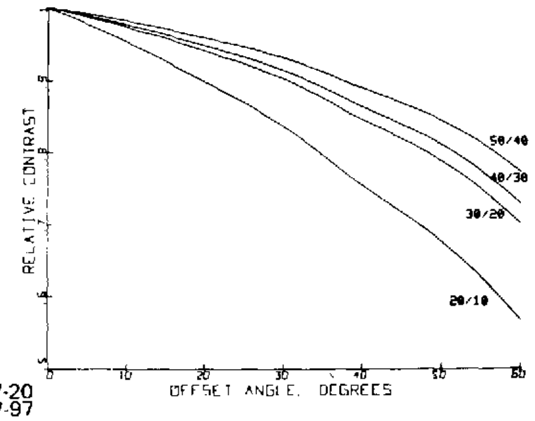
7-17

Fig. 4-17 Orbit Time of Day Vs Seasons



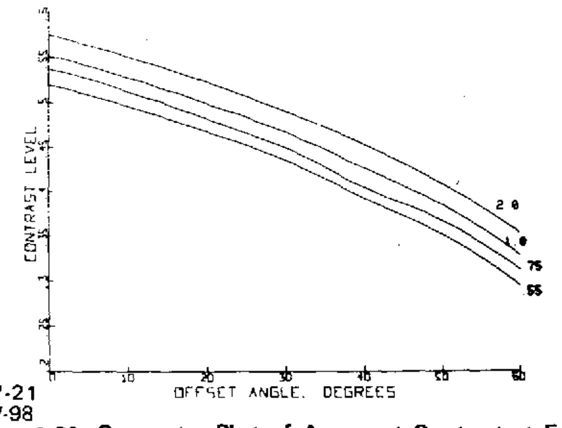
7-18
7-96

Fig. 4-18 Computer Plot of Apparent Contrast at Four Solar Zenith Angles, θ_o = 20, 40, 60, and 80 (For R_t = 20%, R_b = 10%, ϕ = 180°, λ= 0.55 μm)



7-20
7-97

Fig. 4-19 Computer Plot of Relative Contrast at Four Lambert (Diffuse) Target and Background Reflectivities, R_t and R_b = 50 and 40%, 40 and 30%, 30 and 20%, and 20 and 10%, Respectively (For λ = 0.55 μm, θ_0 = 60°, and ϕ = 180°)



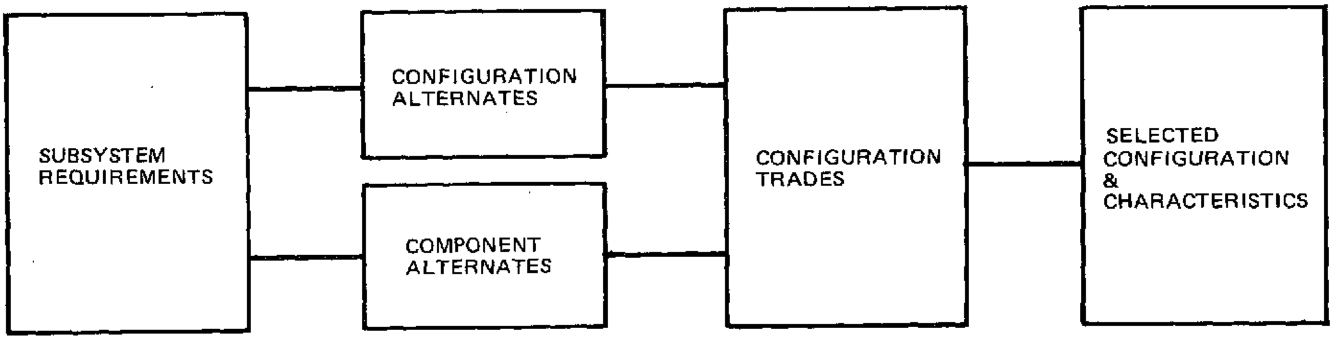
7-21
7-98

Fig. 4-20 Computer Plot of Apparent Contrast at Four Wavelengths, λ = 0.55, 0.75, 1.0, and 2.0 μm (For θ_0 = 40°, ϕ= 180°, R_t = 20%, R_b = 10%)

5 – OBSERVATORY DESIGN ALTERNATIVES

The selection process of a spacecraft configuration involved the task flow as shown in Fig. 5-1.

The input was the system requirements generated in the systems studies. The subsystem and vehicle alternate designs were evaluated and cost-effective, high-performance configurations selected.



7-60

Fig. 5-1 Configuration Selection Approach

5.1 OBSERVATORY SUBSYSTEMS

5.1.1 ATTITUDE CONTROL

(Refer to ACS Study Composite, Fig. 5-2.)

The attitude control subsystem (ACS) was designed to meet the basic requirements of Earth Pointing, Sun Pointing, Stellar, and Geosynchronous Earth Pointing missions. This range of missions reresults 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.

The basic requirements for the ACS are summarized in Fig. 5-2, Table a.

CANDIDATE CONFIGURATIONS – Three candidate ACS configurations were established as shown in Fig. 5-2a. Configuration No. I meets requirements lower than baseline (0.05-deg attitude accuracy and 5 x 10^{-6} deg/sec angular rate stability); No. II meets baseline requirements (0.01-deg attitude accuracy and 10^{-6} deg/sec angular rate stability); and No. III meets higher than baseline requirements (0.002-deg attitude accuracy and 0.2 x 10^{-6} deg/sec angular rate stability). These ACS configurations are also summarized in terms of components, cost, weight, and performance in Fig. a, inset Tables 1, 2, and 3. The components that change with configuration are the sensors (rate gyros, startrackers, and earth sensor) and the associated software in the CDH OBC. Each configuration has three different sizes of wheels and bars: size 1 for spacecraft up to approximately 8500 lb, size 2 for spacecraft between 8500 and 17,000 lb, 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 were space qualified or presently in development. The capability to handle Solar and Stellar missions in addition to the Earth Pointing missions is present in ACS Configurations No. II and III, but not in No. I.

The capability of the ACS Configuration No. I. exceeds that of the ERTS-A. This system is the least costly, complex, and versatile. ACS Configuration No. II 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 CDH on-board computer (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 No. III, a gimbaled star tracker, having high resolution and accuracy, is used to achieve the highest pointing accuracy.

The three ACS configurations were compared on a cost, weight, and performance basis in Fig. a, inset Table 4. The weights for all sizes remain below 600 lb. The recurring cost varies from $0.638 million (ACS Configuration No. I, size 1) to $1.370 million (ACS Configuration No. III, size 3).

SELECTED ACS CONFIGURATION – The selected configuration, Fig. 5-2b, can meet the requirements 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. The concept of providing the ACS control algorithms as a mission-peculiar software program to be processed in the OBC is viable.

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 b.

Modal operations are functionally described, including a list of the ACS sensors and actuators used in each mode, in Table c.

Packaging of the ACS components in the module is shown in Fig. c.

5.1.2 COMMUNICATIONS AND DATA HANDLING

The communications and data handling subsystem (CDHS) was designed to satisfy the EOS requirements (Report No. 3, Appendix C) 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 CDHS 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 is 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 CDHS, are located in their respective module. The subsystem is functionally separate and operates independently of the wide-band communications subsystem.

COMMUNICATIONS GROUP – (Refer to the CDHS (Communications Group) Study Composite, Fig. 5-3.) The communications group of the CDH module provides telemetry, tracking and command link compatibility with STDN, Shuttle Orbiter, TDRS (Option) and DOI (Option). Figure 5-3, Table a tabulates significant communication link requirements for these interfaces. The interface with STDN at S-Band is also shown in Table a.

Communications Group Configuration Alternatives – Seven alternative communication configurations were derived. They vary in capability and complexity from the single thread configuration of Fig. a, inset Fig. 1 (Configuration No. 1) to the sophisticated multimode Configuration No. 5 shown on inset Fig. 7. The basic parameters of these alternates are compared in Table b.

The primary difference between Configurations No. 1 and 2 is that Configuration No. 1 provides spherical antenna coverage on the uplink and hemispherical antenna coverage on the downlink, whereas Configuration No. 2 provides spherical coverage both uplink and downlink. Configurations No. 1 and 2 have dual redundant transponders.

In Configuration No. 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.

Configuration No. 4 is Configuration No. 2 plus a TDRS S-Band terminal. The terminal includes an S-Band transceiver package and a steerable antenna. Since wideband communications will have an interface to the TDRS S-Band at Ku-

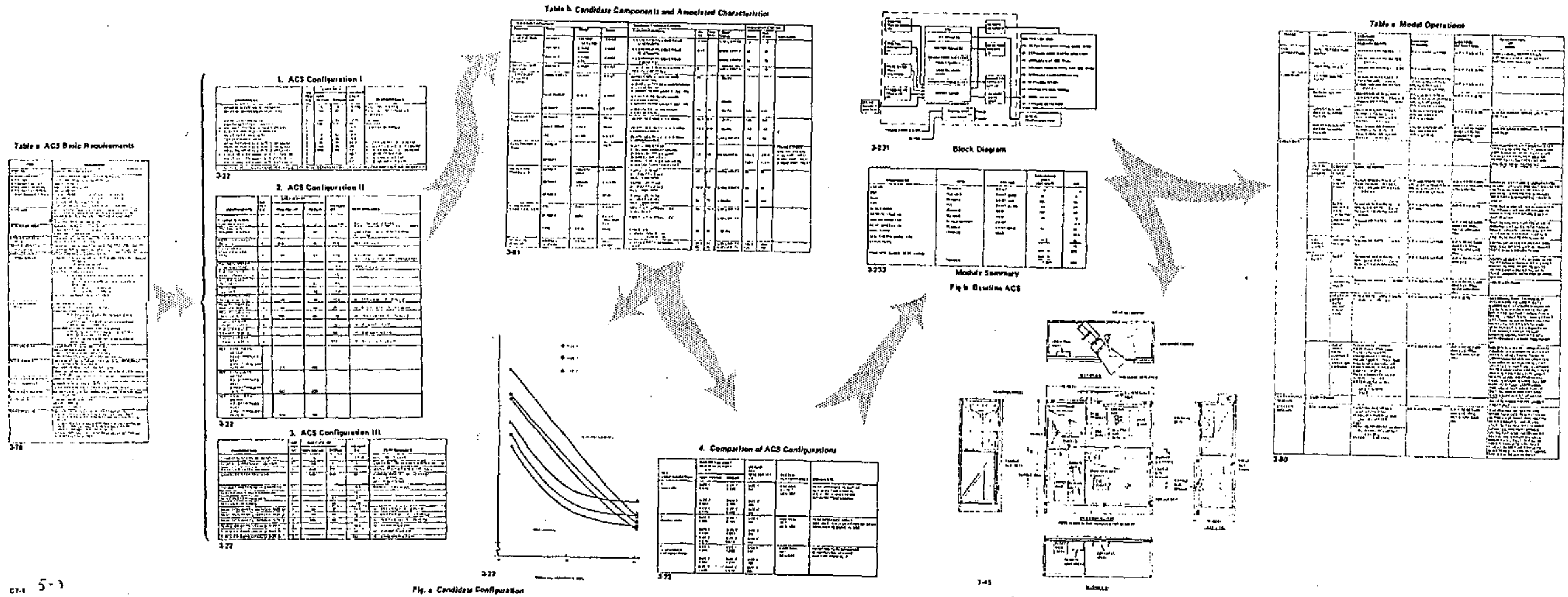


Fig. a Candidate Configuration

Fig. b Baseline ACS

Fig. c ACS Module Arrangement — EOS-A

Fig. 5-3 ACS Study Composite

FOLDOUT FRAME | ORIGINAL PAGE IS OF POOR QUALITY

5-3-a

ORIGINAL PAGE IS OF POOR QUALITY

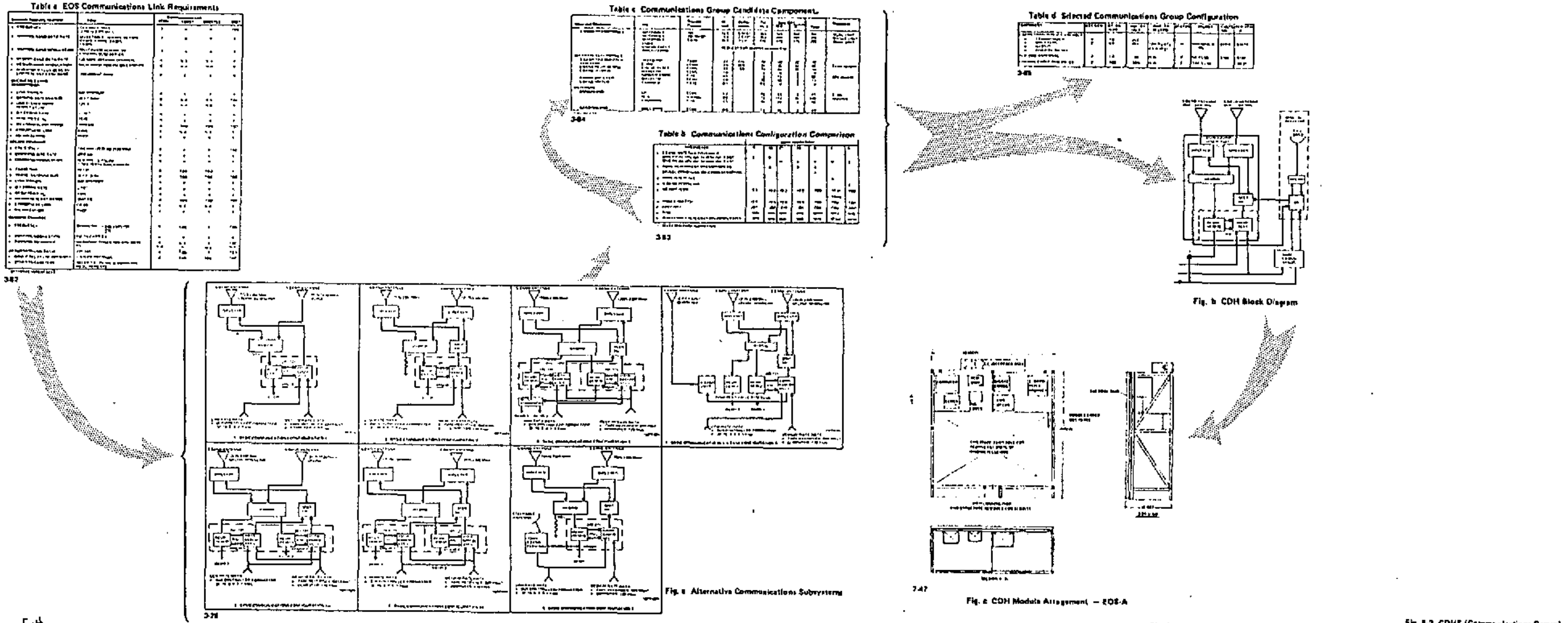


Fig. 5-3 CDHS (Communications Group) Study Composite

Fig. 5-3 CDHS (Communications Group)

ORIGINAL PAGE IS OF POOR QUALITY

Band, a dual frequency S/Ku-Band steerable antenna is being considered to satisfy both the narrowband 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 CDH module. The steerable S-Band antenna (7 to 11 ft diameter requirement) will probably be located on a boom to minimize vehicle blockage problems.

The last Configuration, No. 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.

Selected Communications Group Configuration – Analysis of alternate configurations, summarized in Table b, resulted in the selection of Configuration No. 2, Fig. a, Inset Fig. 3. This configuration utilizes a single S-Band transponder with an integrated hybrid, coaxial switch and two diplexers, along with two broadband S-Band shaped antennas.

The hardware was selected from available candidate components shown in Table c. The equipment list and cost allocation for the selected configuration is shown in Table d. Selection of this configuration satisfies the functional and performance/design requirements shown in Table a for a STDN S-Band interface and is considered a low-risk design because it uses space proven off-the-shelf components.

A communications group block diagram, and packaging of the components in the module, are shown in Fig. b and c, respectively.

DATA HANDLING GROUP – (Refer to the CDHS (Data Handling Group) Study Composite, Fig. 5-4.) The data handling group must acquire, process, record, format and route data/commands from and to the appropriate EOS subsystem (communications, ACS, electrical power, 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 and issue 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 data handling group requirements and their origin are outlined in Fig. 5-4, Table a.

Grumman software sizing estimates for command storage, spacecraft control, systems monitoring, etc, define 23.3 thousand 18-bit words including margin as required for storage in the computers main memory. Resolution to 30m 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 to 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.

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

While EOS is attached to the Orbiter, the Orbiter crew must be alerted and have the capability of monitoring any EOS parameters that will indicate a potentially hazardous condition. Nine to 12 EOS caution and warning functions have been identified by Grumman for EOS.

Data Handling Alternate Configurations – 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.

The NASA Standard Full Duplex System, with commands and addresses sharing a common bus, was selected and merged with the EOS baseline equipment characteristics. The decision to incorporate the NASA Standard data bus features into the baseline system was made so as to share nonrecurring development costs for this system. NASA Standard operating at a 1-Mbps rate, and using selfsynching Manchester II Bi-Phase L code, easily fulfills EOS requirements.

Since the full duplex system uses a common bus for both commands and addresses, a single central unit (controller/formatter), controlling the bus and issuing both addresses and commands, would reduce system complexity.

The selected baseline is shown in Fig. a. Configuration alternatives to the system are Configuration No. 1, which uses a remote unit that incorporates both a remote decoder and remote multiplexer (Mux). Configuration No. 1P is the same remote unit, except that it is power strobed with a 16-KHz square wave. Configuration No. 2 uses separate remote decoders and remote multiplexers, while Configuration No. 2P is the 16-KHz square wave power strobed version of Configuration No. 2 (the NASA EOS baseline for remotes).

DHG AOP Memory Alternatives – The advanced on-board processor (AOP) is available with three memory types: core, plated wire, and CMOS (complimentary metal oxide semiconductor). Core and plated wire are both considered to be acceptable memory types for EOS application while CMOS is conditionally acceptable.

The primary driver for memory selection is total program cost. Total program cost (including power costs) are shown in Figure b, inset Fig. 1. The selection of core memory for a single spacecraft requiring 24K memory saves $91 thousand over plated wire and $57 thousand over CMOS.

Selected Data Handling Configuration – The baseline single thread data handling group, Fig. c, is comprised of a 24K-word AOP with core memory, command decoder, buss controller/formatter unit, seven remote units (one located in the CDH module, the remaining six distributed throughout the spacecraft), a 4.096-MHz central clock and signal conditioning units, which condition high- and low-level signals from 0 to 5 vdc, and also contain D/A 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 nonrecurring costs. Assuming AOP procurement effort 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 85 KOPS, which is seven to eight times the current maximum requirement for EOS.

The AOP's capability to perform data compression is utilized on housekeeping data, thereby eliminating need for the optional tape recorder. This represents a savings of approximately $80 thousand, 8 w of power, and 14 lb of weight per spacecraft.

The selected full duplex data bus system, Configuration No. 1P, 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 CDH specification. Each unit also has 64 output channels for pulse commands plus four serial magnitude command outputs.

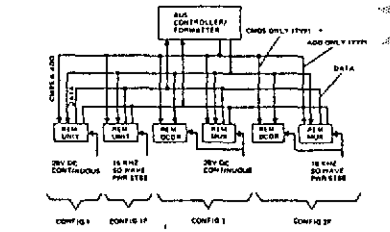
The controller/formatter also has dual receivers and transmitters which interface to the dual redundant busses. This unit can accept and interleave 50 commands per sec from the command decoder with 62.5 commands per sec from the AOP, and transmit these to the remote units. Telemetry output rates are command selectable

7

Table a Data Handling Subsystem Requirements

ITEM	REQUIREMENT	ORIGIN
COMPUTER		
MAIN MEMORY (K WORDS)	21.3	GAC SOFTWARE SIZING ESTIMATE
WORD SIZE (BITS)	18-24	NASA, GAC SIZING ESTIMATE
THRUPUT (KOPS)	8-12	GAC S/W SIZING ESTIMATE
LANGUAGE	ASSEMBLY	GAC MEMORY EFFICIENCY
CLOCK STABILITY	1 PART IN 10^? 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	8-12	GAC SIZING (LAUNCH VEHICLE DEPENDENT)
TAPE RECORDER (OPTIONAL)		
CAPACITY (BITS)	10^?	GAC
RECORD RATE(S) (KBPS)	32/16/8/4/2/1	NASA
REPRODUCE RATE (KBPS)	640	NASA
RECORD TIME (MINUTES)	110	GAC

3-86



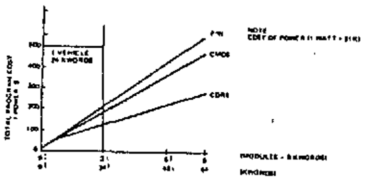
3-29 Fig. a EOS Data Bus System Configuration

5-7

FOLDOUT FRAME 1

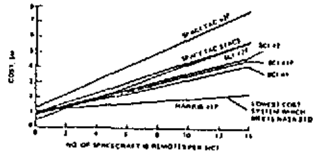
C7-3

3-27



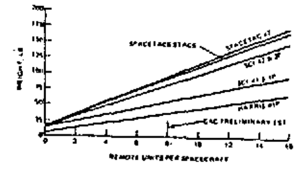
1. Computer Memory Type Total Program Cost

3-30



2. Total Program Cost (Non Recurring + Recurring) of Candidate Data Systems

3-31



3. Weight of Candidate Data Bus Systems

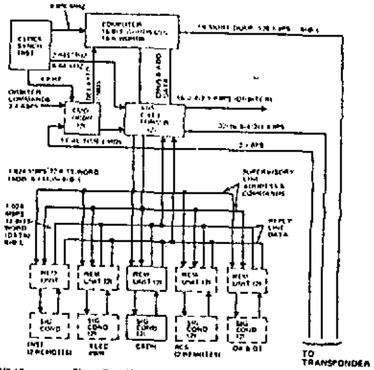
Fig. b Cost/Weight Comparisons

5-7-A

Table b Data Handling Summary

COMPONENT	MFG	STATUS	RECURRING COST (PER UNIT)	(%)
TRANSPONDER ASSY	MOTOROLA	[illegible]	[illegible]	26
A TRANSPONDER				
B DIPLEXERS				
C HYBRID				
D COAX SW				
ANTENNA	GE	[illegible]	[illegible]	[illegible]
COMPUTER	[illegible]	[illegible]	[illegible]	[illegible]
CONT. FORMATTER	HARRIS	NEW	[illegible]	[illegible]
REMOTE UNIT	HARRIS	NEW	[illegible]	[illegible]
CMD DECODER	HARRIS	NEW	[illegible]	[illegible]
SIG COND	GAC	[illegible]	[illegible]	[illegible]
CLOCK	[illegible]	NEW	[illegible]	[illegible]
WIRE, CONNECTOR	[illegible]	[illegible]	[illegible]	[illegible]
STRUCTURE			[illegible]	[illegible]
MODULE PRO TOTAL			[illegible]	100
PROGRAM, [illegible]			[illegible]	
TOTAL			[illegible]	[illegible]

3-237



1115-13 Fig. c Data Handling Block Diagram

Fig. 5-4 CONS (Data Handling Group) Study Composite

5-7

7

Table a Summary Load Analysis

LOAD	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]

3-87

Table b Major EPS Driving Requirements

[illegible]	[illegible]
[illegible]	[illegible]

3-88

Table c Comparison of Alternative EPS Configurations

[illegible]	ADVANTAGES	DISADVANTAGES
[illegible]	[illegible]	[illegible]

3-89

Table d EOS 22-Cell, Ni-Cd Battery Alternatives

BATTERY	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

3-90

Table e Alternative Solar Array/Battery Control Comparison

NAME	[illegible]	[illegible]	[illegible]	WEIGHT	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

3-91

C7-4

Table f Typical Solar Array Candidates Suitable for EOS

[illegible]	[illegible]	WEIGHT	POWER	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

3-92

Table g Selected EPS Components

[illegible]	[illegible]	WEIGHT, LB	
[illegible]	[illegible]	[illegible]	[illegible]

3-83

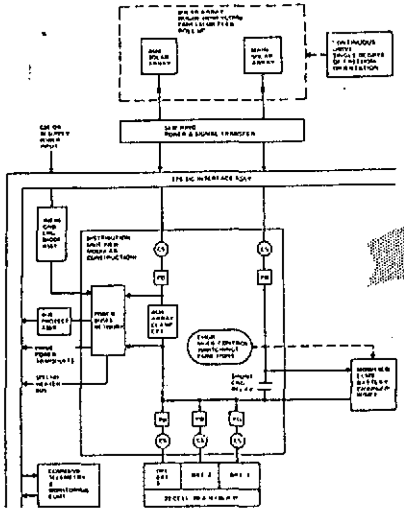
5-8

FOLDOUT FRAME /

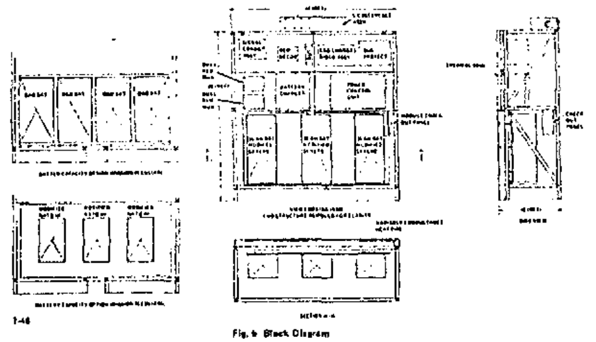
ORIGINAL PAGE IS OF POOR QUALITY

5-8-a

FOLDOUT FRAME 2



3-25 Fig. a EPS Functional Component Block Diagram



7-46

Fig. b Block Diagram

Fig. 5-8 EPS Study Composite

5-8

5-8-c

5-8-d

at 32/16/8/4/2/1 Kbps, and format consists of minor frames of 128 eight-bit words.

5.1.3 ELECTRICAL POWER
(Refer to the EPS Study Composite, Fig. 5-5.)

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

Figure 5-5, Table a defines known spacecraft/mission electrical power requirements. The basic spacecraft (exclusive of mission-peculiar payloads) is estimated to require approximately 300 w of orbital average power. EOS instruments and other associated payload equipment can range from an average of 150 w to over 350 w. Including missions other than EOS could result in an average payload power of up to 500 w. Therefore, the electrical power subsystem design load capability, based upon this tentative load analysis, should be in the range of 400- to 500-w orbital average. Maximum peak loads for the EOS are not expected to exceed approximately 2 kw.

Requirements which have a major impact on the electrical power subsystem design, configuration, performance weight and cost, are summarized in Table b.

General forms of EPS candidate 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 c summarizes some of the key advantages and disadvantages or various alternative configurations.

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.

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

- Battery alternatives (Refer to Table d)
- Solar array/battery control (Refer to Table e).

SOLAR ARRAY ALTERNATIVES – The power module solar array/battery charge equipment was selected to offer flexibility and latitude in defining a solar array that is optimized to particular mission requirements. The alternatives are shown in Table f.

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 ACS. The major determinant for this choice is the necessity to minimize resultant disturbance torques created by periodic solar array stops, starts, and reversals.

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. It is basically the same as that used for OAO.

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, down to just battery recharge
- Capability to maximum-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-amp/hr or 36-amp/hr batteries, thereby satisfying 40 to approximately 120-amp/hr capacity option requirement with only two or three batteries.

SELECTED EPS COMPONENTS – A detailed, functional/component block diagram of the selected EPS is shown in Fig. a. A summary of selected components are identified in Table g, and packaging of the EPS components in the module is shown in Figure b.

5.1.4 PROPULSION

(Refer to OTS/RCS/OAS Study Composite, Fig. 5-6.)

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 and restabilization, wheel unloading
- Orbit transfer, circularization, deorbit.

Figure 5-6, Table a shows the anticipated impulse requirements and fluid quantities, and Table b shows candidate propulsion components. Note that modularity will influence the design of each of the subsystems.

The propellant required to correct the L/V injection errors represents 97% of the total translational propellant on board the spacecraft, with the orbit-keep propellant representing 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.

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

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 n mi. Our studies selected an operational altitude of 366 n mi, eliminating the need for the OTS. However, propellant loading to transfer to and from a 493-n mi 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.

5.1.4.1 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 Fig. 5-6, Table a. 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 considera-

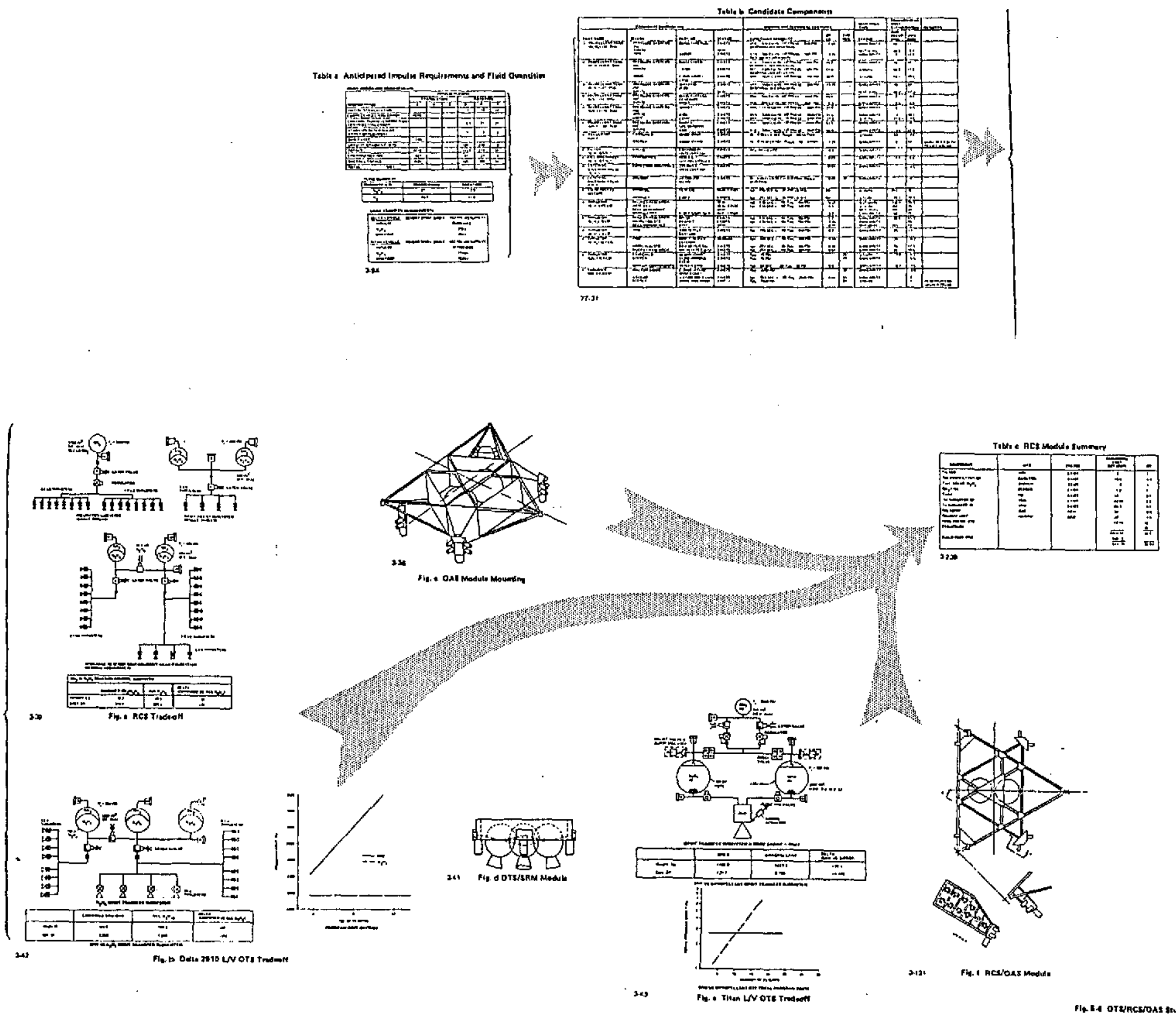


Fig. 5-6 OTS/RCS/OAS Study Composite

5-11-b

5-11

tion in the Delta 2910 spacecraft configuration, the lighter-weight N_2H_4 system was selected.

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 installed in a module mounted on the aft end of the spacecraft as shown in Fig. 5-6, c.

5.1.4.2 RCS CONFIGURATION TRADEOFF

Two alternatives were considered to fulfill the reaction control 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 RCS with the OAS should be considered. 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. 5-6, a.

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

The selected RCS 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 equipment is installed in a module mounted on the aft end of the spacecraft as shown in Fig. f. Table c provides a summary of the RCS Components.

5.1.4.3 OTS CONFIGURATION TRADEOFF

The primary means of providing orbit transfer capability, if required, is the use of solid rocket motors (SRM) shown in Fig. 5-6, d.

The alternatives studied were an OAS using 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-n mi orbit with the EOS being transferred to and from a 493-n mi 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-in. tanks are replaced by three 22-in. tanks. 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 Fig. 5-6, Table b.

Note 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 $1 million (see the Program Cost Savings curve associated with Fig. b).

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 for in the Boeing design. A bipropellant system using N_2H_4 and MMH and sized to the same total impulse as the four SRM-2's was assumed. A four-vehicle/four-flight program was also assumed. The results are shown in the table associated with Fig. e.

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 EOS. The Total Program Cost curve of Fig. e shows that a crossover point occurs in total program costs at the 10-to-11 flight point in the program.

5.1.5 SUBSYSTEM 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 con-

Preceding page blank

figuration was considered for the Titan arrangement.

Evaluations were conducted for the Land Resources mission. Worst-case minimum/maximum environment heat fluxes were used for each module. An altitude range of 300 to 500 n mi and descending node time of day (DNTD) range of 9:30 a.m. 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.

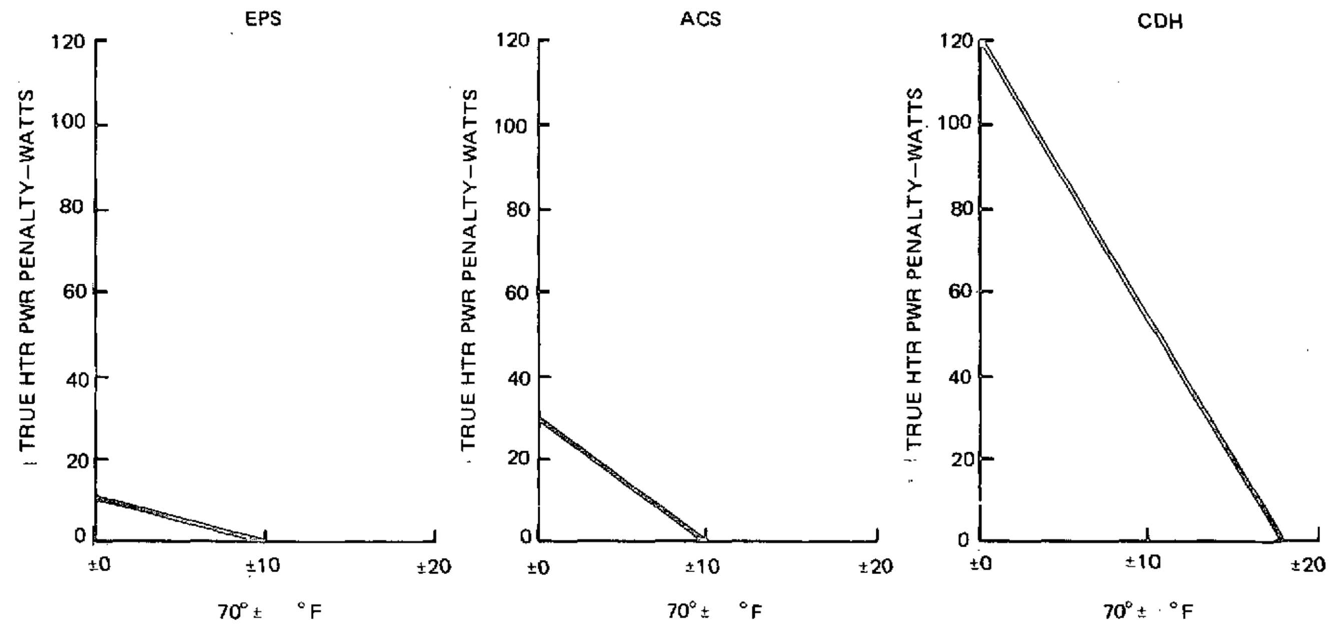
Analysis of the modules were considered on a lumped parameter, parametric basis. The ability to reject heat was studied as a function of alternate thermal options for each location. This technique established module location and feasibility of passive control, supplemented with heater power during low-power dissipating modes.

The cost per watt can vary between $0.75 and $1.75 thousand per watt, depending on the array selected. The savings in module acceptance test costs resulting from a narrow operating temperature range (± 10° F vs ± 50° F) can be as much as $16 thousand. The fundamental passive design cost tradeoff 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 tradeoffs are used to achieve the DTC targets.

Figures 5-7 and 5-8 show the design-cost tradeoffs conducted for the selected Delta module locations (apex toward nadir, Delta No. 1 Configuration). The evaluations were conducted for the LRM.

A hot-case heat rejection capability of 150 w 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° F. True heater power penalties in this case would be the power in excess of 150 w for each module.

The increase in power subsystem costs at $0.75 thousand per watt for a rigid array and $1.75 thousand per watt for a flexible array were then determined. The increase in module acceptance test costs ($20 thousand at ± 50° 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° F for EPS and ACS and ± 20° F for CDH). The results of a



3-44
7-62

Fig. 5-7 Passive Design For 150 Watts Hot-Case Heat Rejection similar evaluation, for the Titan configuration of modules (with the CDH module facing the earth), are quite close to the foregoing and yield the same conclusions.

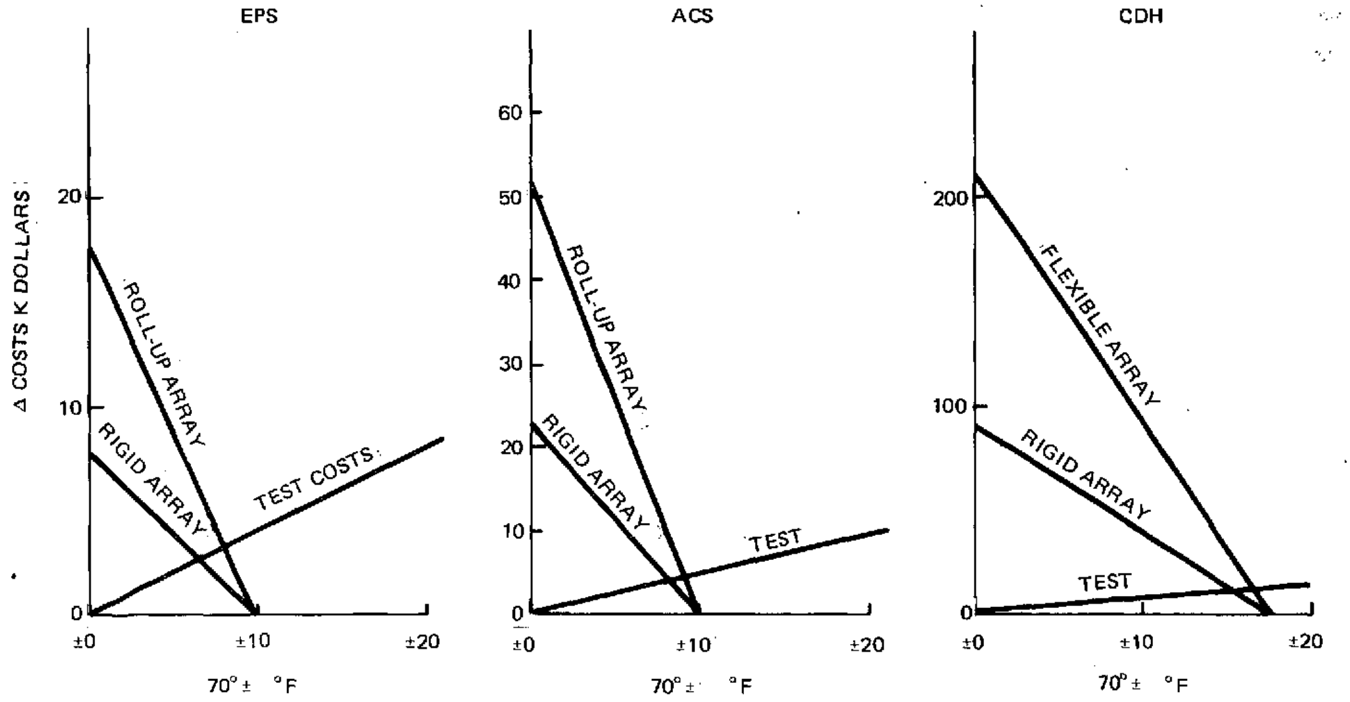
Local power loading within a module may require further trading off of active control costs to achieve these narrow operating temperature ranges. Designing for failure modes (such as a solar array hangup) would modify these results due to designing with different minimum/maximum heat fluxes. Future mission 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.

5.2 SPACECRAFT

5.2.1 SPACECRAFT THERMAL CONTROL SUMMARY – A matrix of structural concepts has been considered for both the Delta and Titan launch vehicles. The number of instruments gives additional mission peculiar complexity. 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 cost of thermal control hardware were obtained to support the design cost studies.



3-45
7-63

Fig. 5-8 Change in Subsystem and Acceptance Test Costs with Temperature

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

- Reductions in structure heater power from prestudy 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° F structure and insulation effectiveness of .05, the total structure heater power is 66 w. Using an insulation effectiveness of .02, which should be readily achievable, reduces the heater power to 28 w. Reducing the structure temperature to 40° F decreases the heater power requirements to the range of 15 to 38 w (range of insulation effectiveness). Although 100 w 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 w
- 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° F should only be a transient condition, during contact periods. A module support structure of 40° F is consistent with the minimum anticipated equipment operating temperatures. A 40° F OAS structure is consistent with minimum propellant temperature requirements.

DESIGN COST EVALUATION – The structure insulation design/cost trade study is shown in Fig. 5-9 and 5-10. Figure 5-9 shows the total structure heater power required for a Delta configuration spacecraft as a function of structure insulation effective emittance. Structure temperatures of 70° F and 40° F are plotted as parameters. The baseline design (70° F structure temperature and .05 effective emittance) requires 66 w of heater power.

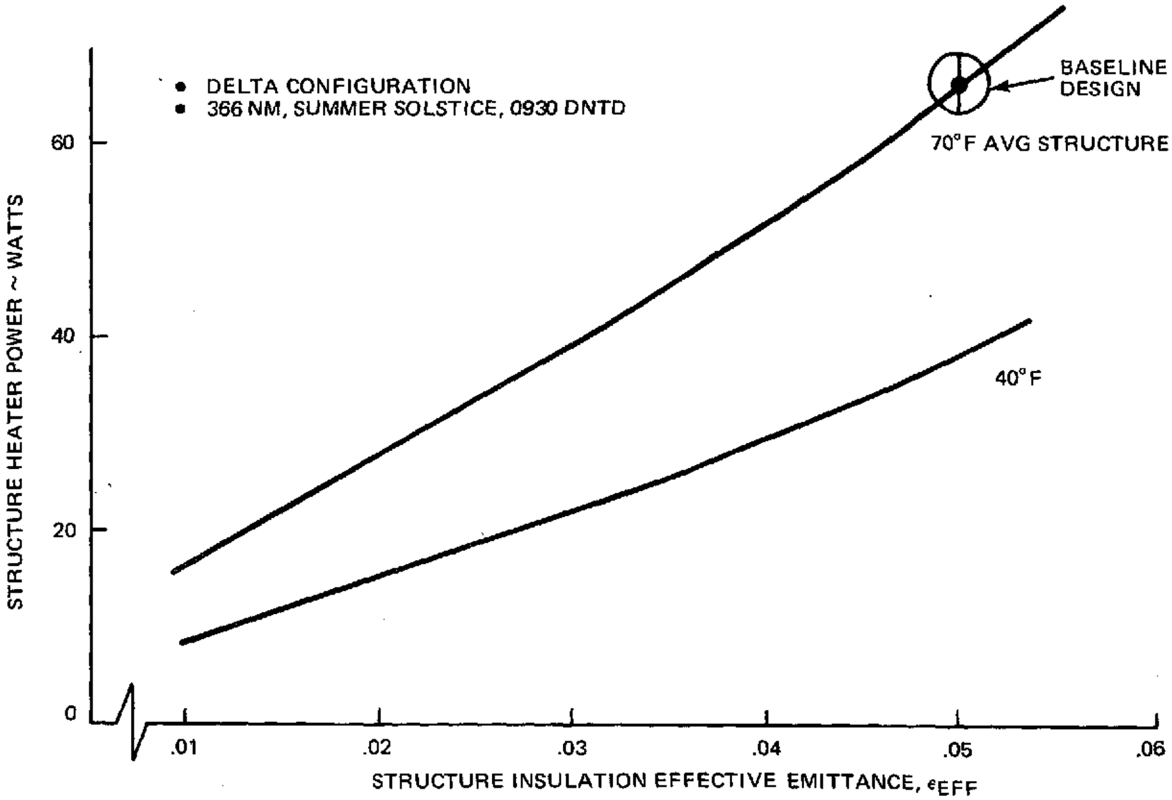
Figure 5-10 shows the cost of structure insulation as a function of effective emittance. Improved insulation performance is achieved by different design and insulation techniques. Figure 5-10 also shows the added power subsystem costs for structure heater power. Two extremes for solar array costs are plotted for a range of structure temperature and type of solar array. The cost of insulation plus solar array shown in Fig. 5-10 is 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° F structure temperature, the optimum structure insulation has an effective emittance of .015 to .02. The less costly honeycomb array and 40° 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.

5.2.2 STRUCTURE

(Refer to Basic Spacecraft Study Composite, Fig. 5-11.)

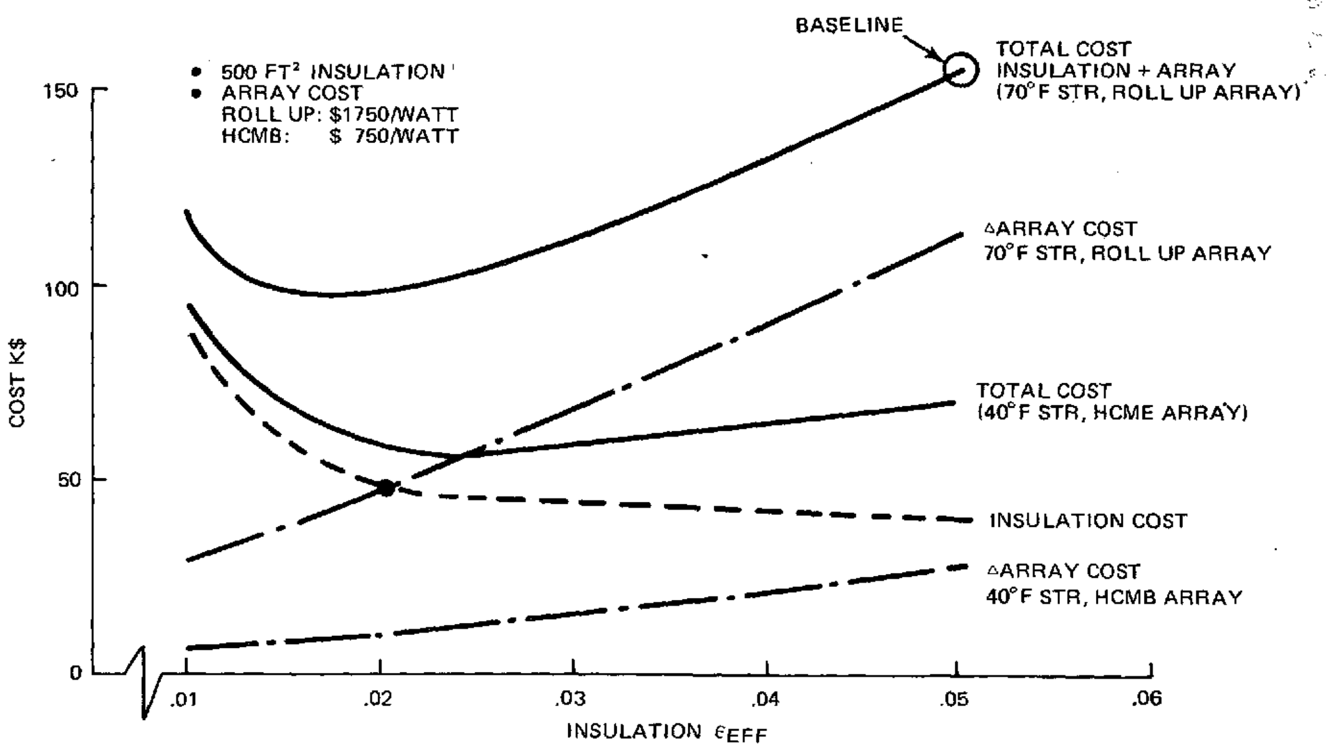
A Basic Spacecraft configuration compatible with Delta, Titan or Shuttle launch vehicles was designed to support a significant number of follow-on satellite missions. The general requirements for the structure subsystem were established to support this goal. These requirements were:

- One vehicle configuration shall support EOS –A, –A′, –B and –C missions (Fig. 5-11 Table a) 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 CDHS, 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



3-20
7-64

Fig. 5-9 Total Structure Heater Power



3-21
7-65

Fig. 5-10 Structure Insulation Plus Delta Array Costs

- The Basic Spacecraft configuration shall meet dynamic and static load requirements as defined in Tables b through e.

SPACECRAFT MOUNTING ON LAUNCH VEHICLES – The Basic Spacecraft is configured to be bottom-mounted when launched on a Delta vehicle, as shown in Fig. a. Provisions for transition ring mounting for launch or retrieval are inherent in the design and can be provided, if required. The bottom mount was selected because of the case of separation from the launch vehicle adapter with no significant spacecraft weight penalty.

When the Basic Spacecraft is launched by a Titan III vehicle, the clearance problem is reduced. The 86-in. OD of the spacecraft combined with the 110-in. ID of the shroud results in a 12-inch radial clearance, which significantly reduces the spacecraft extraction problem. For this installation, we recommend adding a 110-in. diameter ring to the Basic Spacecraft at its transition ring station and mounting it on an extended booster adapter.

The Space Shuttle launch and retrieval of the EOS requires a modified clamp-type separation mechanism at the Spacecraft 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 Manipulator System. The basic difference between the Grumman concept and the GSFC baseline transition ring assembly is that the Grumman concept supports six discrete mount fittings of the triangular Spacecraft configuration, whereas the GSFC concept has a continuous ring system. (This shown in Fig. 5-11, b.) Elimination of the continuous mounting ring results in a Spacecraft weight saving of 75 lb.

STRUCTURAL CONFIGURATION FOR TRIANGULAR VEHICLE – The structural configuration for the Basic Spacecraft is shown in Fig. b and c. 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 at three points. In this arrangement, primary structural loads are not induced in the subsystem 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 Spacecraft was configured in aluminum although other materials were investigated. Significant weight reduction and some recurring cost savings may be accomplished on the basic spacecraft structure bv substituting advanced composite materials for aluminum. The particular material investigated was a hybrid Graphite/Epoxy. 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.

It was concluded that, although cost of initial tooling for composites is high, cost of succeeding units is competitive with aluminum and saves 80 lb.

A structure was designed to make full use of the Titan launch vehicle volume and configuration advantages. This structure is shown in Fig. 5-11, d. 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.

5.2.3 RESUPPLY

The Grumman resupply concept provides servicing capability for all functional equipment and the EOS spacecraft. The replaceable assemblies are shown in Fig. 5-12.

LATCHING MECHANISM – The Grumman latch mechanism, shown in Fig. 5-13, consists of three hook-and-roller latches per module and utilizes a self-locking linkage. The latch hooks are config-

Table a Instrument Section Requirements

3-61

Table b Delta 2910 and Delta 3910 Load Factors

3-62

Table c Titan III B/NUS Load Factors

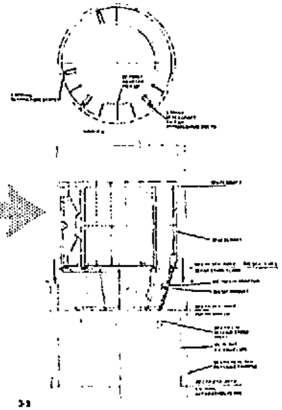
3-63

Table d Shuttle – Payload Bay Load Factors

3-64

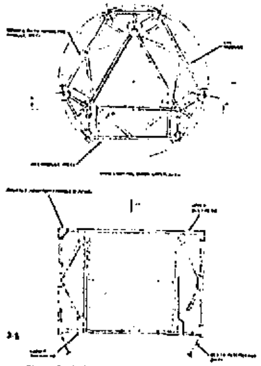
Table e Minimum Frequency Criteria

3-1



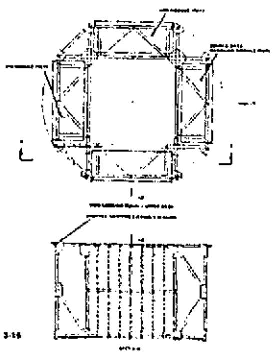
3-3

Fig. a Spacecraft/Launch Vehicle Adapter



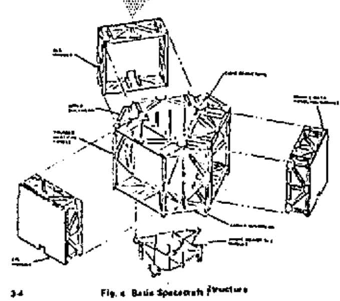
3-5

Fig. b Basic Spacecraft Structural Arrangement



3-16

Fig. d Titan Basic Spacecraft Structural Arrangement



3-4

Fig. c Basic Spacecraft Structure

Fig. 5-11 Basic Spacecraft Study Composite

Preceding page blank

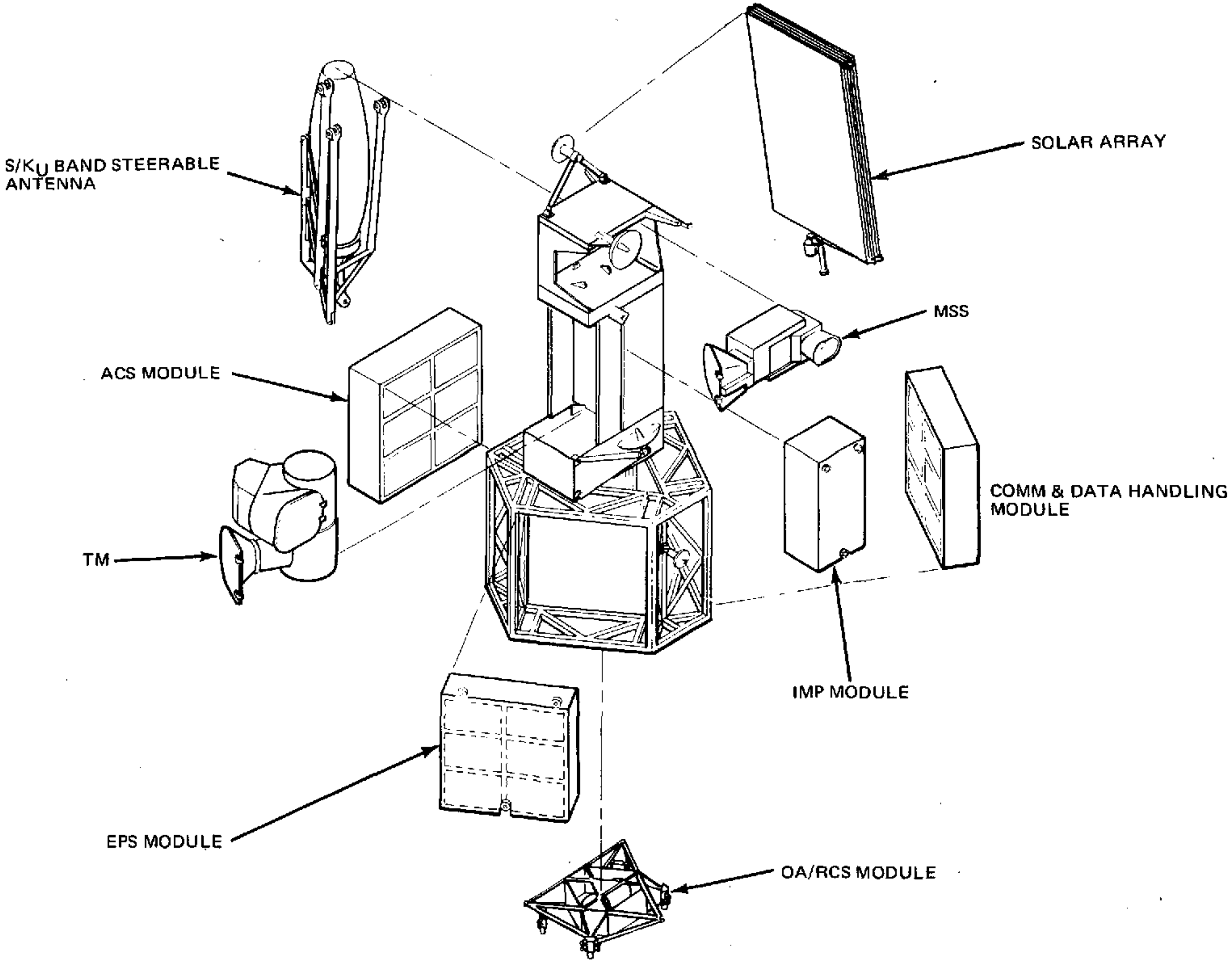


Fig. 5-12 EOS-A (TDRS) Resupply Option

7-4
7-101

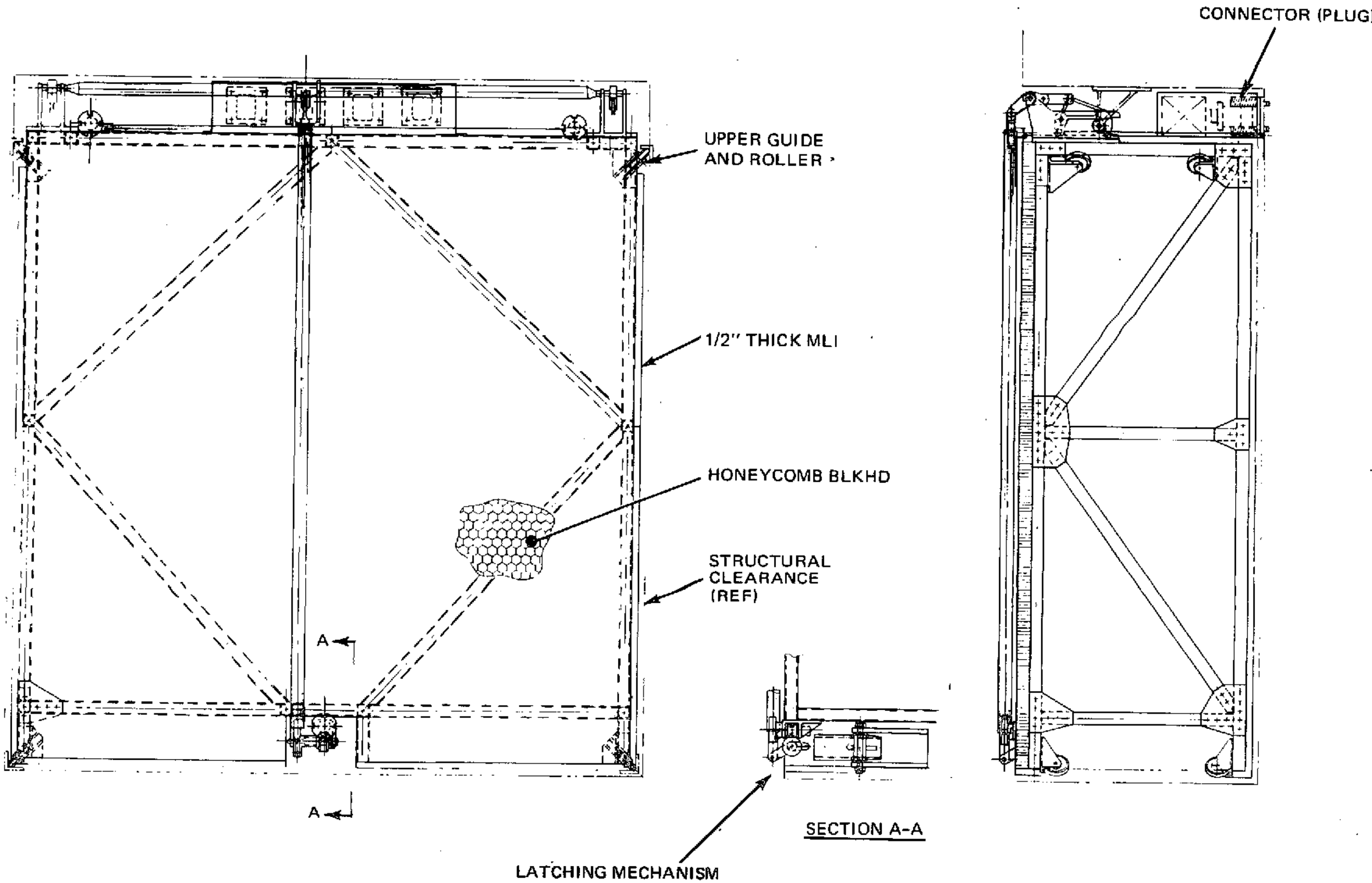


Fig. 5-13 Subsystem Module Structural Assembly Showing Latching Mechanism Detail

ured 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 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 MEM 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; needs only low actuator force
- Simple, reliable, and economical.

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:

Figures	Module
5-14	Subsystem Module
5-15	Thematic Mapper
5-16	RCS/OAS Module
5-17	Solar Array

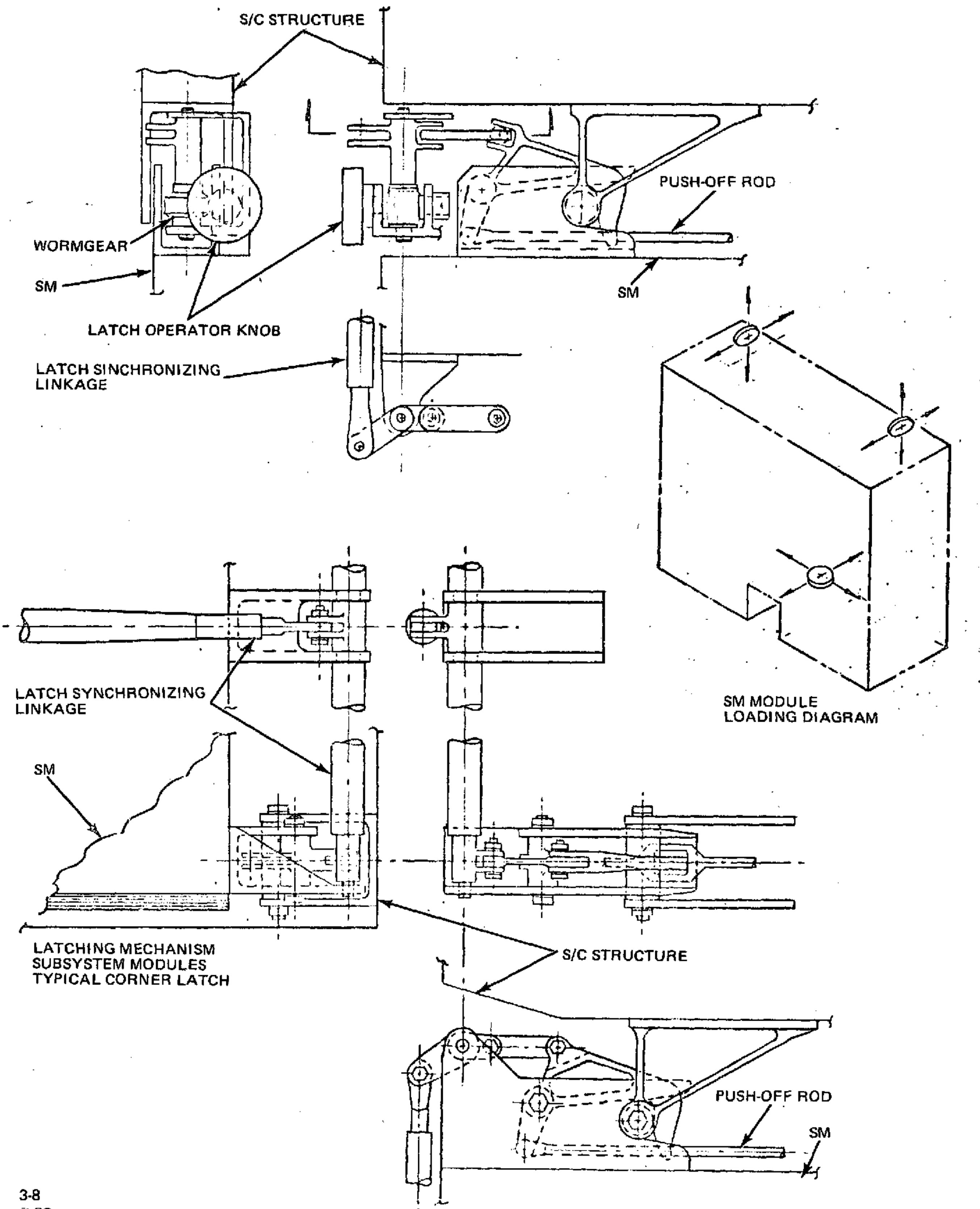
5.2.4 INSTRUMENT MISSION PECULIAR EQUIPMENT

5.2.4.1 WIDE BAND DATA HANDLING AND COMPACTION

The function of the Wide Band Data Handling and Compaction (WBDHC) subsystem is to convert, format, compact multiplex, and select multichannel data from the instruments and produce serial digital data streams at suitable rates for transmission to Primary and Low Cost ground stations, and to the TDRS satellite. The overall WBDHC subsystem block diagram is shown in Fig. 5-18. A speed buffer function is included to provide for a partial scene data compaction option. The compacted rate is constrained to be equal to the MSS rate. Factors contributing to the determination of the WBDHC configurations include:

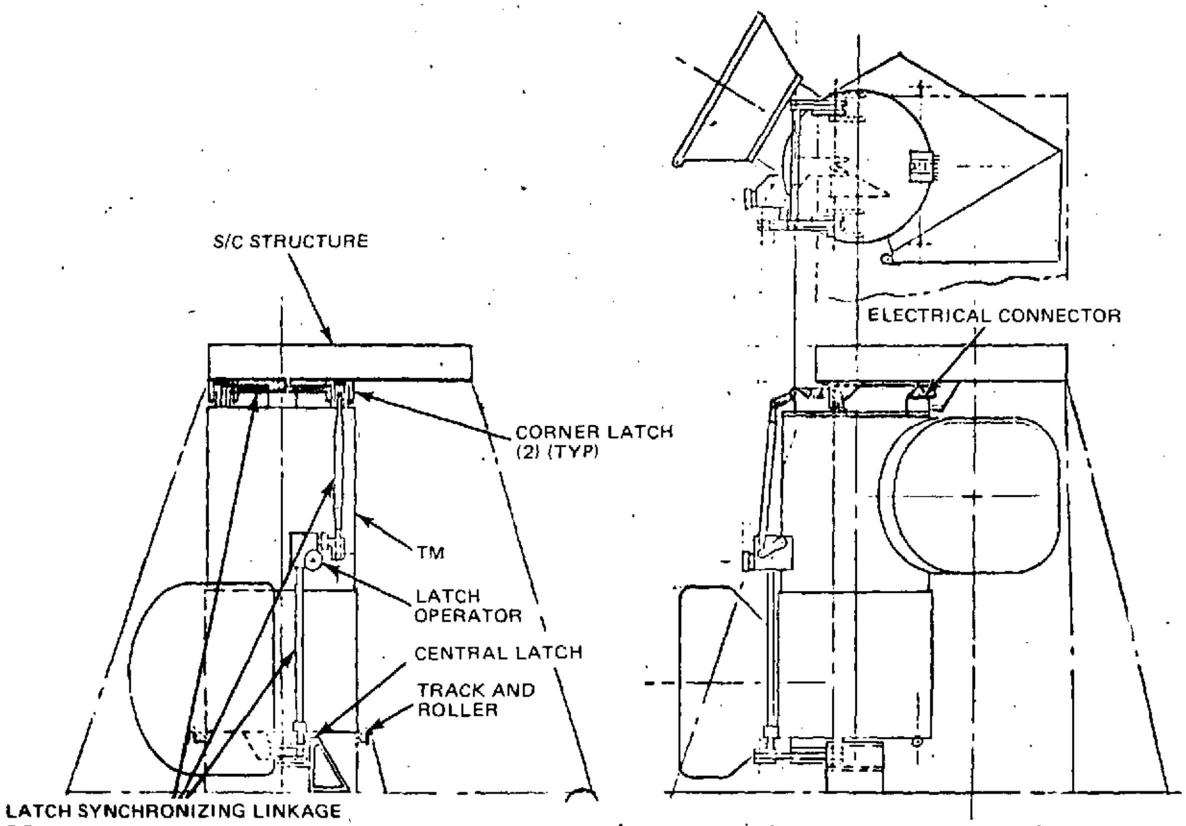
- Interface with instruments – digital or analog
- Data rate (if digital)
- Size of electronics package that can be placed inside or in contact with instrument
- Modular flexibility considerations (future instruments)
- Need for high capacity speed buffering in data compaction.

Figure 5-19 shows the instrument data handling and compaction alternatives considered. Figure 5-19, A applies to an analog interface. This is appropriate to the Harris MOMS concept. Hughes, Te-Gulton, and Honeywell have expressed a preference for a digital interface, as shown in B. The electronics associated with the instrument are in close proximity. A logical extension of this concept, shown in C, is to include the entire data handling function in the electronics delivered with the instrument. The disadvantage of this approach is in the high digital data rate interface to be reckoned with by the instrument vendor. A final concept, D, is to combine all the data handling and compaction functions in one unit in close proximity to the instrument. The low rate digital instrument interface approach was chosen as the best trade between risk and complexity.



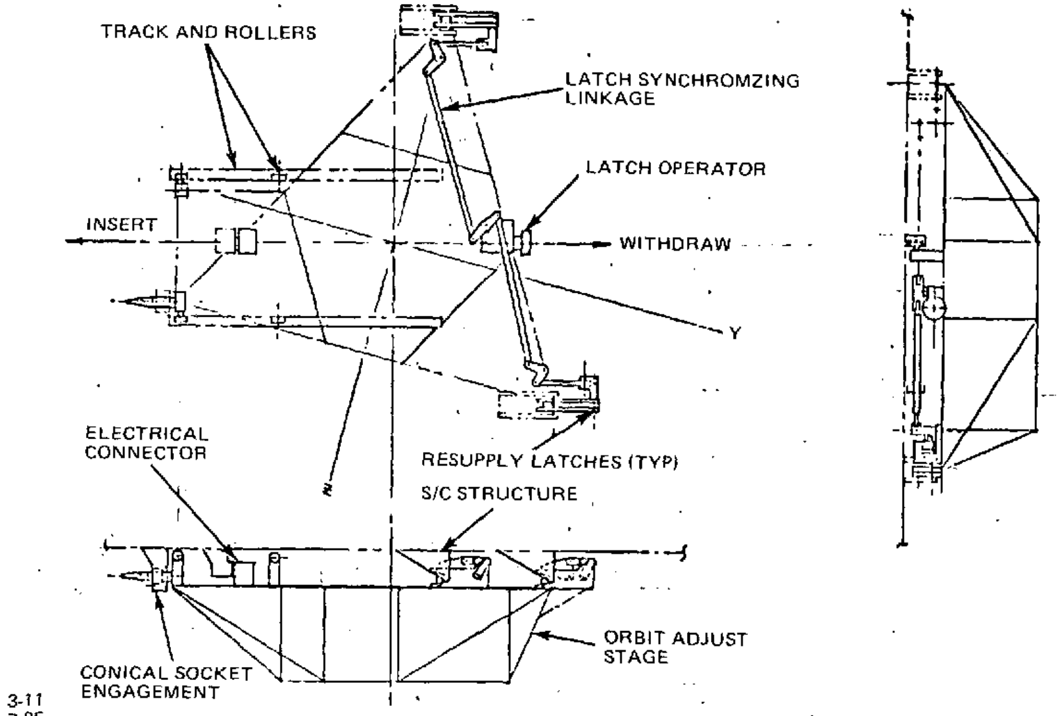
3-8
7-82

Fig. 5-14 Subsystem Module, Typical Central Latch



3-9
7-83

Fig. 5-15 Thematic Mapper Latching Concept



3-11
7-85

Fig. 5-16 RCS/OAS Latching Concept

ORIGINAL PAGE IS
OF POOR QUALITY

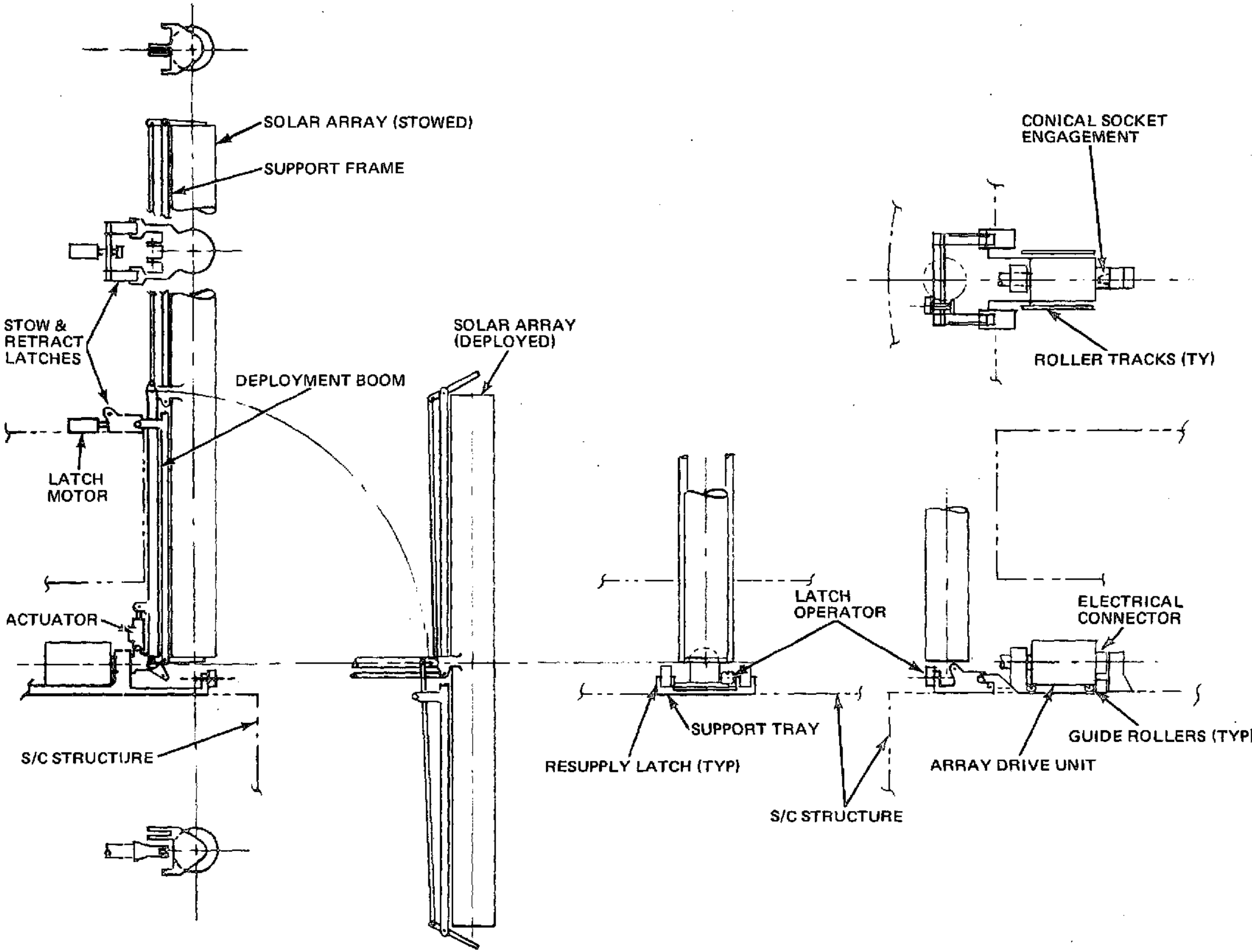


Fig. 5-17 Solar Array Mechanism for Stowage, Deployment, Retraction and Resupply

Design considerations included the effects of compacted data rate. From transmission system considerations, it is desirable to keep this rate below 20 Mbps to allow operation with modestly sized and low cost LUS's. Since the MSS rate is 16 Mbps, it would be appropriate to make the compacted TM rate the same value. Three types of compaction are possible: band selection, resolution reduction, or partial coverage. Options available within the 16-Mbps constraint are shown in Table 5-1. The partial coverage option requires a buffer memory, the size of which is dependent upon the number of bands involved.

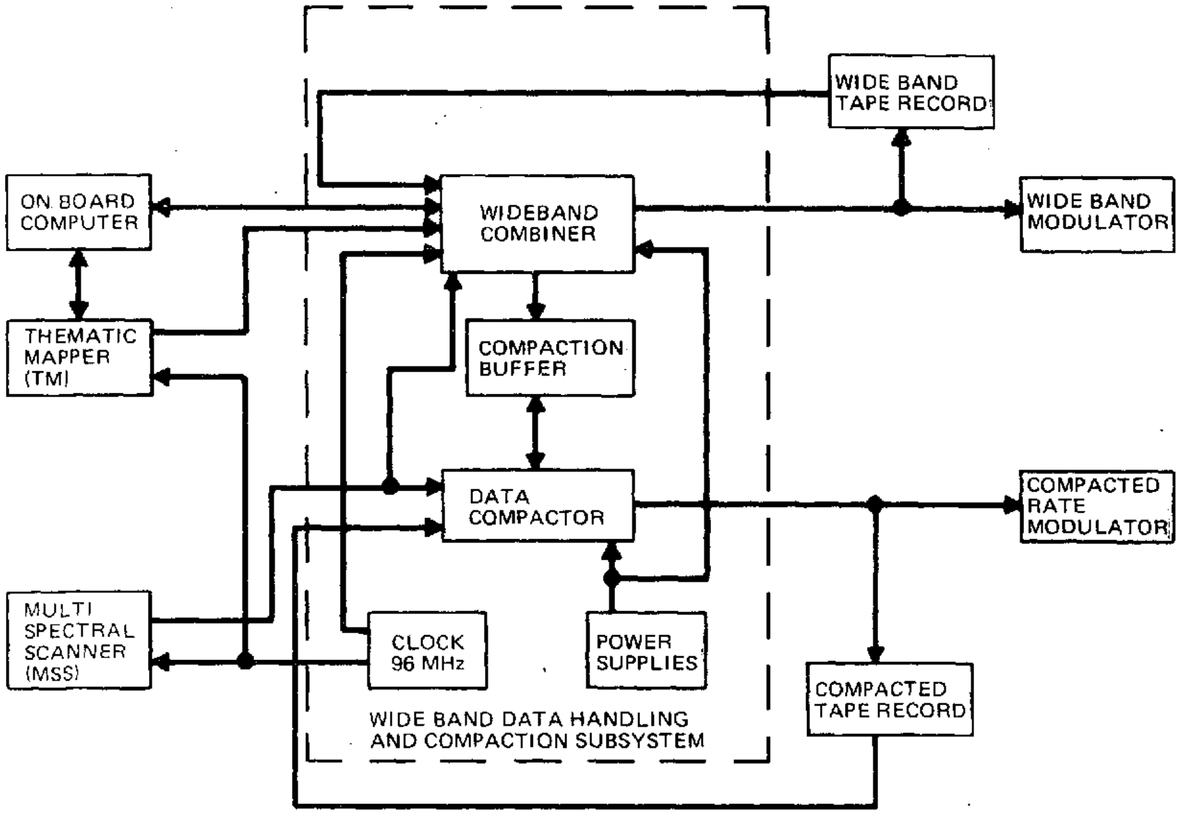
Cost, size, weight and power information are shown in Table 5-2. The range of values includes the uncertainty in the 0.5 to 1.0 x 10^6 bit speed buffer which will be required if partial scenes are desired from the data compaction. This speed buffer is considered a development risk at this time.

5.2.4.2 WIDE BAND COMMUNICATIONS

Wide band 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. The primary link has been sized at 240 Mbps, and is required to be received by STDN earth terminal sites. The LCGS link has been sized to handle a reduced data rate of 20 Mbps, and is to be received by small earth terminals. The key issue in the design of the small earth terminals is low cost.

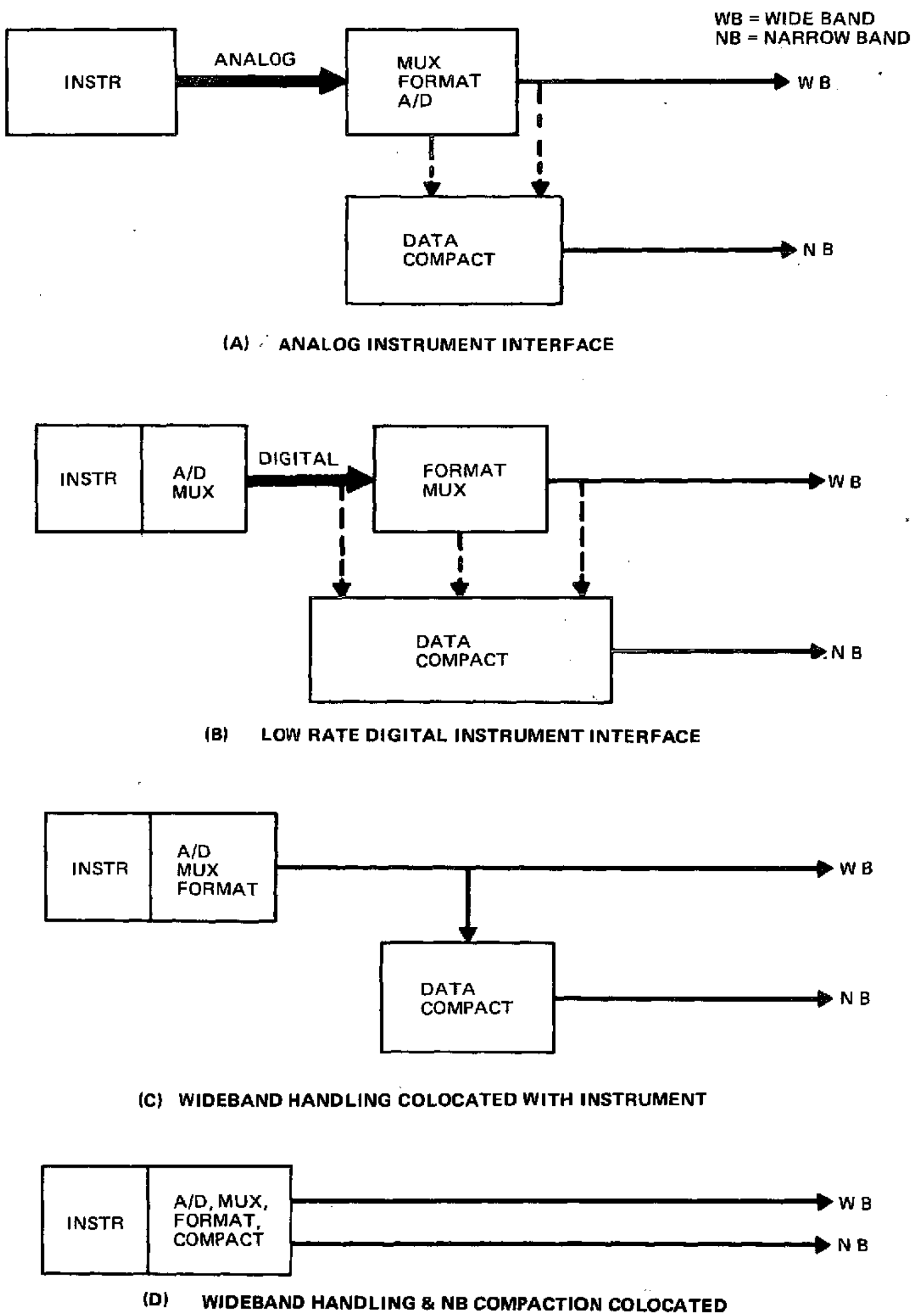
In addition to the direct communication link requirements, the EOS spacecraft may also be required to relay sensor data to earth through the proposed NASA TDRSS.

The TDRSS link and the two direct links represent the baseline EOS spacecraft wide band communication subsystem requirements. Alternative spacecraft subsystem designs were considered including the use of a wide band video tape recorder



(1)5-47
7-86

Fig. 5-18 Overall WBDHC Subsystem



7-87

Fig. 5-19 Basic Data Handling and Data Compaction Configuration

Table 5-1 Compaction Options

FORM	RESOLUTION	DATA RATE (APPROXIMATE)
ALL BANDS AT 1/6 RESOLUTION (2 RESOLUTION ELEMENTS IN DIRECTION OF SCAN BY 3 ELEMENTS IN S/C MOTION DIRECTION)	55 x 82.5 m	16 MBPS
ONE HIGH RESOLUTION BAND PLUS LOW RESOLUTION BAND	–	14.8 MBPS
ALL BANDS AT FULL RESOLUTIONS FOR AN 18 MILE SWATH	FULL	BUFFERED TO 16 MBPS
THREE BANDS AT FULL RESOLUTION FOR A 36 MILE SWATH	–	BUFFERED TO 16 MBPS
FOUR HIGH RESOLUTION BANDS AT 1/4 RESOLUTION, PLUS IR BAND	55 x 55 m	13.7

7T-39

Table 5-2 Data Handling and Compaction
Unit Cost, Size, Weight, and Power
(Costs in 1974 Dollars)

• COST	
– RECURRING	\$610K – \$2.3M
– NON-RECURRING	\$70K – 325 K (4 UNITS)
• SIZE	860 – 950 in³.
• WEIGHT	35–50 LB
• POWER CONSUMPTION	90 – 120 W

7T-40

(WBVTR) and MSS recorder to replace the TDRSS link requirement and alternative approaches for establishing the PGS and LCGS links.

5.2.4.3 ALTERNATIVE SUBSYSTEM CONFIGURATIONS

The wide band 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.

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 EIRP from the spacecraft is specified to be at least 61.3 dBW for the 12.5-ft 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.

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. This option employs three recorders: one wide band video tape recorder (WBVTR) for the TM, instrument output and two ERTS-type recorders for either the MSS or the compacted TM data. The WBVTR has a read-in and read-out rate or approximately 120 Mbps for periods of up to 15 min 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. 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 wide-band spacecraft communication subsystem are a 100-Mbps TM and 16-Mbps MSS link to PGS sites and a 20-Mbps link for compacted TM data to LCGS's. However, the PGS link subsystem has been sized to provide for 240-Mbps channels to accommodate future higher rate EOS missions.

There are two alternative approaches for establishing the primary and LCGS links. The first approach employs two narrow beam steerable antennas for both the primary and LCGS links; the second 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 No. 1; Fixed Antenna, Approach No. 2
- DC-to-DC converters
- RF switches
- Combiners or multicoupler.

KU–BAND DIRECT LINKS (OPTION) – An option with the X-Band wide band 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 (15 GHz). The subsystem component types are basically the same as those required for the X-Band subsystem configuration.

5.2.4.4 ALTERNATIVE SUBSYSTEM PERFORMANCE CONSIDERATIONS

Three frequency bands are potentially applicable for space transmission at the high bandwidths involved here: X-Band (specifically, 8.025 to 8.4 GHz), Ku-Band (specifically, 14.5 to 15.35 GHz), and K-Band (21.5 to 22 GHz). TDRSS is planned for Ku-Band operation. K-Band has not been pursued further since adequate link margins were found to be impossible. Power calculations for X- and Ku-Band operation are shown in Table 5-3.

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 5-3. In this latter case, a link margin of only 2.6 dB is realized under the worst-case loss conditions assumed in the calculations.

5.2.4.5 DIRECT LINK TRADES AND ISSUES

The two approaches for establishing the primary and LCGS links at X-Band are depicted in Fig. 5-20. Alternative No. 1 (Fig. 5-20,A) employs two narrow-beam steerable antennas with a 1-ft diameter and 28-dBi gain, each fed by a 4-w power amplifier. Due to ITU power flux density limits the LCGS link must be power controlled to 1 dBW as indicated in the power budget calculations of Table 5-3. Alternative No. 2 (Fig. 5-20,B) employs a narrow-beam steerable antenna 1 ft in diameter (28 dBi); a 4-w power amplifier for the PGS link; a fixed spacecraft antenna with a +30 deg beam width, and 7 dBi gain; and a 50-w 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 5-3.

The Ku-Band option for the direct links to PGS and LCGS sites demands higher spacecraft EIRP and ground station G/T ratios due to the larger propagation losses at this frequency. 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 S-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 dicate the most cost-effective approach depending on the number of local user stations in the system.

5.2.4.6 TDRSS/WBVTR OPTION

The baseline EOS wide-band communication subsystem includes a relay capability of wide band sensor data to earth via the TDRSS. An alternative to this configuration is to include on-board tape recorders of the wide band video, and ERTS-type for storage of data until the EOS is in view of a PGS location.

The power budget calculations for the TDRSS link and the X-Band direct link, which is the assumed communication link for the tape recorder

----

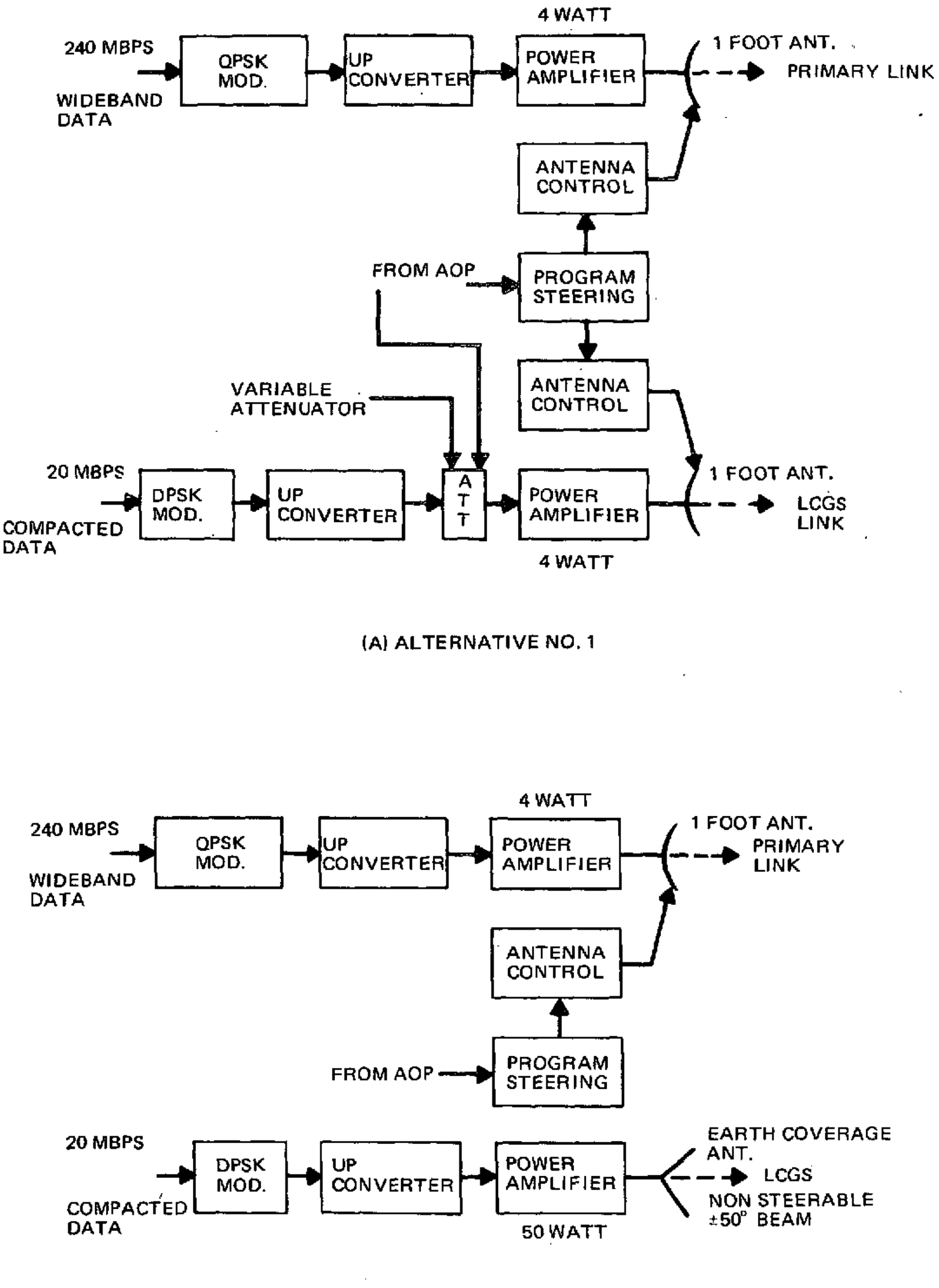
*The basic spacecraft approach differs from this Alternative No. 2 in that two identical narrow-beam antenna subsystems, in addition to the fixed antenna subsystem, are required. Here, only one narrow-beam antenna subsystem is considered because to evaluate Alternative No. 1 versus the baseline would 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.

Table 5-3 Signal Margins with EOS Wideband Links

PARAMETER		X-BAND (8.25 GHz)			KU-BAND (15 GHz)		
OPTION		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

3-268
7T-41



(A) ALTERNATIVE NO. 1

(B) ALTERNATIVE NO. 2

3-52
7-88

Fig. 5-20 X-Band Direct Link Communication Alternatives

option given in Table 5-3, 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 and the technical risks associated with the TDRSS data acquisition and tracking problem.

5.2.4.7 COST, WEIGHT, SIZE AND POWER CONSIDERATIONS

Table 5-4 presents the cost, weight, size, and power consumption data of the various alternatives. The cost data for each subsystem or alternative is broken down into nonrecurring and recurring costs per unit. Nonrecurring costs generally include design and development, qualification modeling and fabrication, test equipment and tooling, and qualification test costs. Recurring costs include production units costs, fabrication, assembly and installation and acceptance tests for production rate/quantity.

Whether to go WBVTR or TDRSS is implied by the data previously presented, which 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, WBVTR's have their own reliability problems; certainly some risk is attached to the 100-Mbps, 10^{11}-bit recorder itself. Favoring the WBVTR approach is the fact that the data are "delivered" to the PGS's 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 $2 million a year for domestic satellite, microwave relay, or leased.

By summing the appropriate combinations of subsystem data items found in Table 5-4, various wide band communication subsystem data elements can be determined. The direct link configurations (X- and Ku-Band) are compared and the results tabulated in Table 5-5. Using recurring costs, weight, and power consumption impacts as the primary basis of subsystem discrimination, the following conclusions can be drawn from these data:

- Any option involving the tape recorders in lieu of the TDRSS subsystem results in severe penalties in spacecraft weight and power requirements
- The TDRSS and Ku-Band direct-link configuration requires substantially more power than either of the X-Band direct-link configurations
- The first configuration, consisting of a TDRSS link in conjunction with the Alternative No. 1

Table 5-4 Summary of Subsystem/Option Data

SUBSYSTEM/OPTION	COSTS (1974 $K) NON-RECUR	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.

2-274
7T-42

Table 5-5 Summary of Total Subsystem Options

WIDEBAND COMMUNICATION SUBSYSTEM OPTIONS	RECURRING COSTS (1974 $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.

3-275
7T-43

(two steerable spacecraft antennas) X-band direct-link subsystem, requires less power than the Alternative No. 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-w TWTA demand in the Alternative No. 2 configuration

- Although nonrecurring costs and size were not used as the primary basis for subsystem option discrimination, the nonrecurring 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.

5.2.5 INSTRUMENT ACCOMMODATION

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

The combination of EOS-A instruments, TM and MSS, and TDRS results in the deployed configuration shown in Fig. 5-21. The drivers in this arrangement are the viewing requirements of each, including: radiator viewing to "Black Space", sensor viewing to the nadir, TDRS antenna pointing, and solar array sun tracking. The stored or launch configuration of the EOS-A is shown in Fig. 5-22 and 5-23.

Instrument section requirements and configuration investigated are shown in Table 5-6.

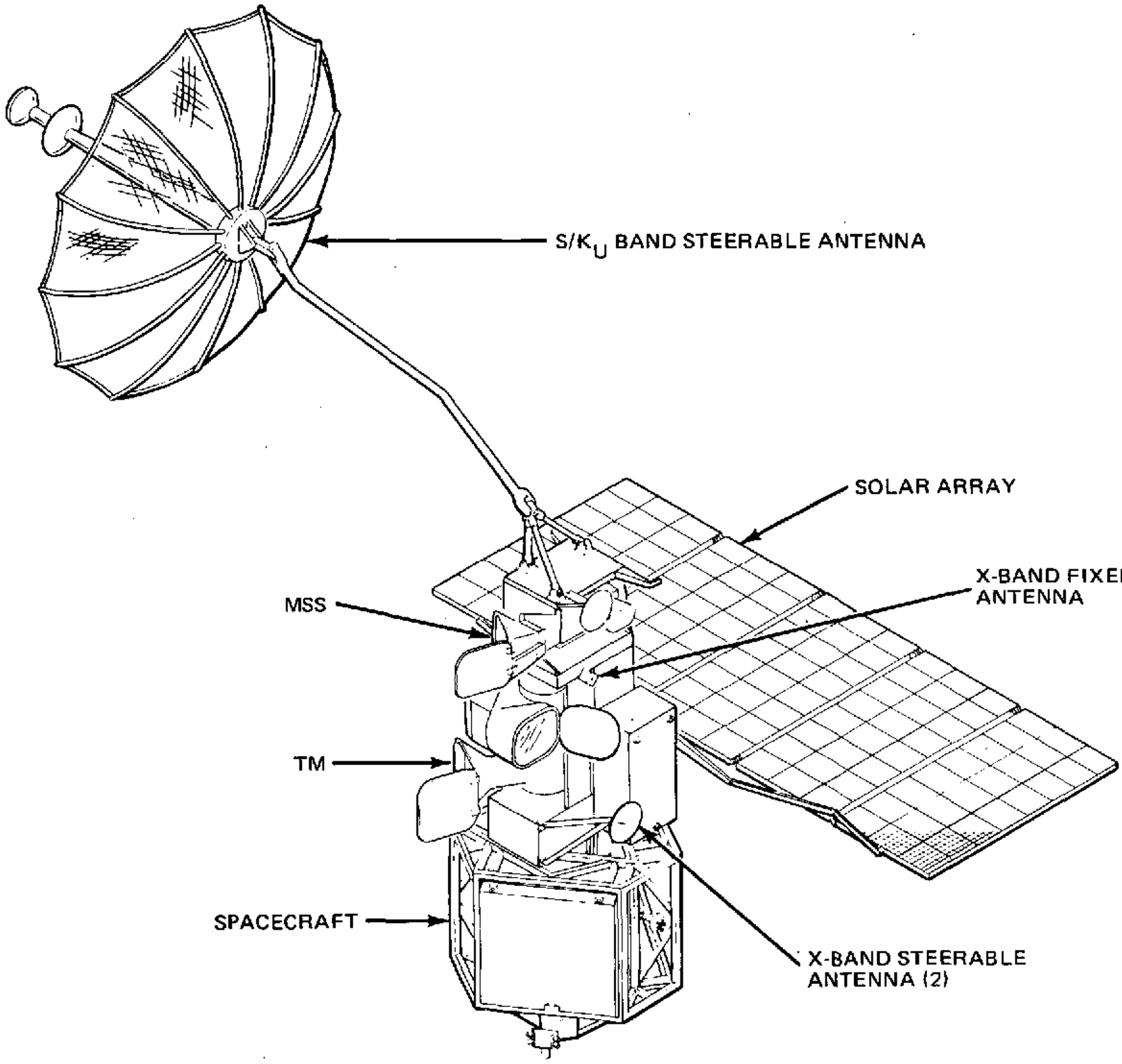
Other instrument configurations investigated showing the utilization of roll up arrays, tape recorder modules for the EOS-A and -B vehicles are shown in Fig. 5-24 and 5-25.

The resupply configuration for the instrument section is shown in Fig. 5-26.

5.2.6 EOS WEIGHT SUMMARY

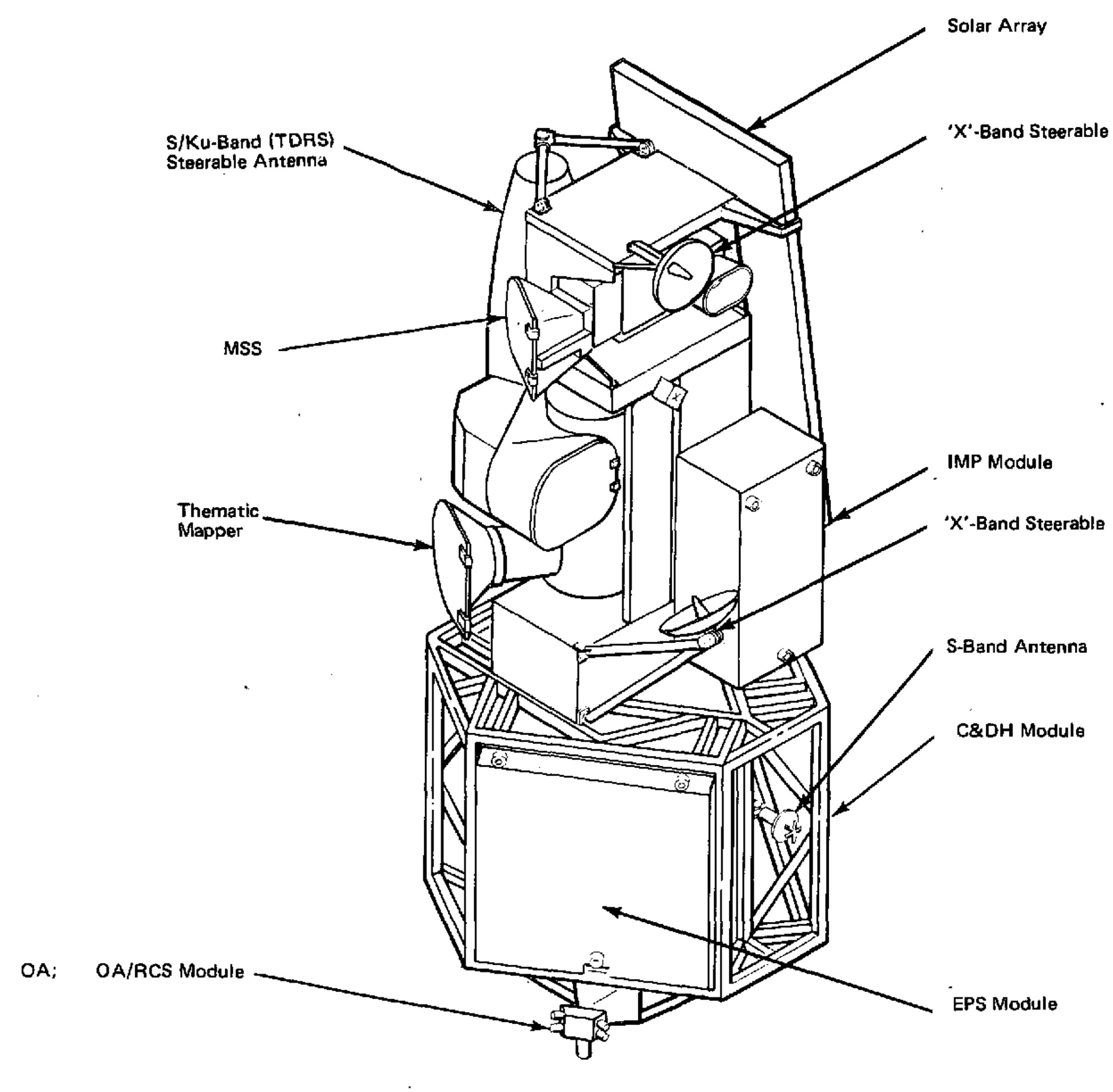
Spacecraft weights for EOS-A, -B, -C and the follow-on missions have been derived using the "barebones" spacecraft as a base. The build-up of these weights is shown in Table 5-7. The upper portion of the table depicts the weight impact to the basic spacecraft resulting from increased capability. Briefly, these may be described as follows:

- Shuttle Compatibility:
  - Deploy Penalty – Structural and C&W interfaces added to enable EOS launch and deployment by the Shuttle. The CDHS equipment redundancy and RCS pressure relief are added for fail-safe design required for Shuttle crew safety



7-3
7-100

Fig. 5-21 EOS-A (TDRS) Flight Configuration



7-93

Fig. 5-22 EOS-A (TDRS) Launch Configuration

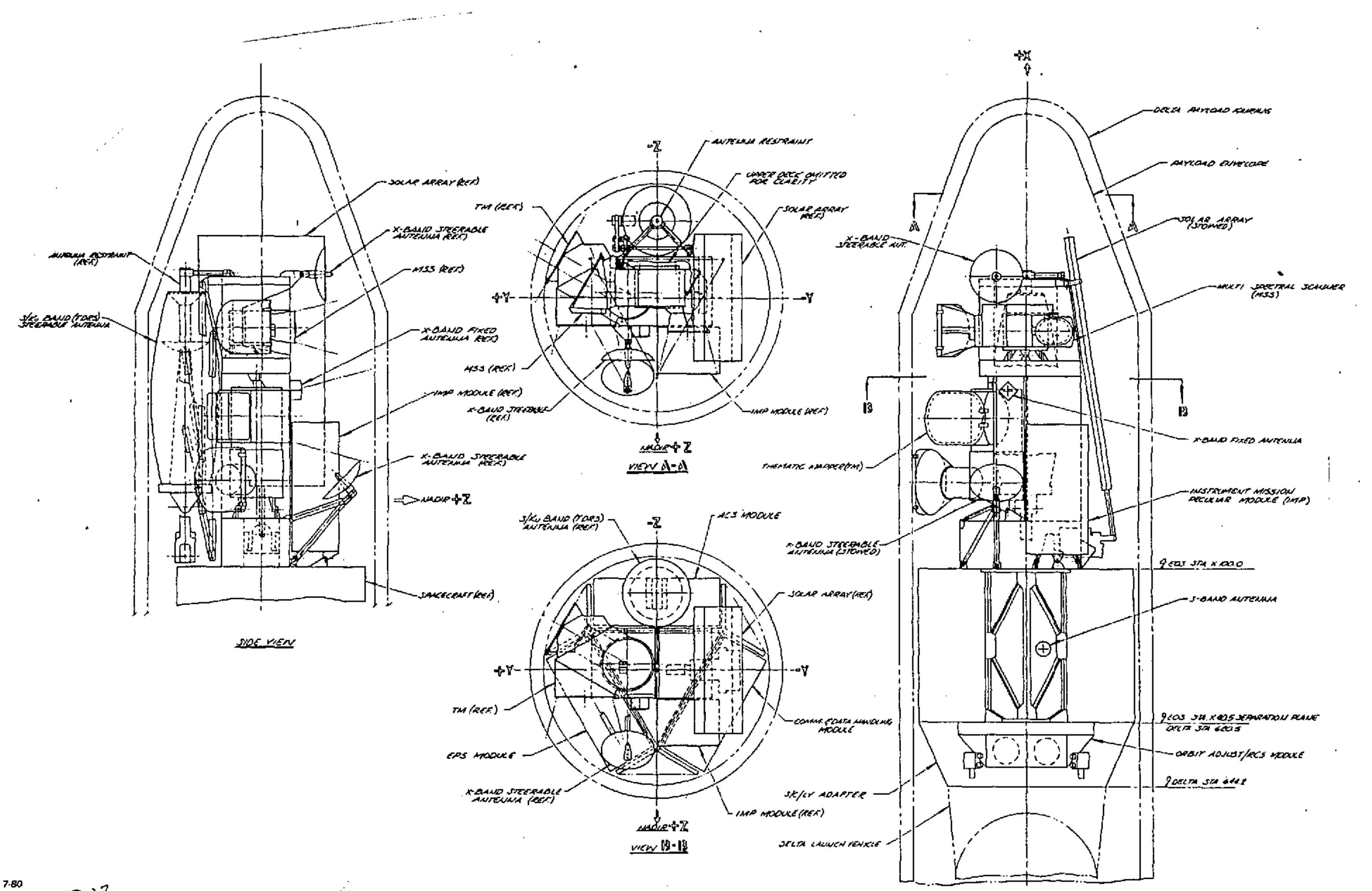


Fig. 5-23 EOS-A (TDRS) Configuration – Delta 2910 Launch Vehicle

7-80

5-37

FOLDOUT FRAME 1

ORIGINAL PAGE IS OF POOR QUALITY

Table 5-6. 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 (1) S/K_U BAND TDRS (12 FT)	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

3-61
7T-38

– Retrieve Penalty – Latches are added to lock the solar array(s) in the retracted position to allow the EOS to be returned in the Shuttle payload bay

– Resupply Penalty – Latches, tracks, rollers, and blind-mate connectors are added to allow on-orbit replacement of the solar array, RCS/OAS stage, and subsystem modules. Additional insulation provides thermal closure in the module bays

NOTE

Observe that the above three impacts are cumulative, i.e., spacecraft resupply capability cannot be achieved without deploy and retrieve capability.

- Two-year Service Life: Addition of a battery, where required, to reduce depth of discharge sufficiently to attain two-year battery life
- Increased Structural Capability: Structural weight increase reflects primary structure and launch adapter changes for the heavier EOS -C and F spacecraft.

The remainder of the table deals with the mission peculiar impact and the instrument weights. This mission peculiar impact is divided into two sections:

- Spacecraft Mission Peculiar: Changes to the basic spacecraft required by specific mission demands. Included here are additional batteries; solar array size changes; larger ACS reaction wheels and torquer bars; memory module; increased RCS capacity for orbit adjust and thrust vector control functions; and SRM kick stages
- Instrument Mission Peculiar: Items required only in direct support of the instrument payload. This includes support structure, resupply and stowage lock mechanisms, and thermal insulation, which are shown as instrument support; the TDRSS communications; and the wide band communications and data handling, which includes the MOMS and signal conditioning units.

The instrument group completes the build-up of launch weight, which is then compared to launch vehicle capability. For EOS-D, it was necessary to employ a lightweight roll-out solar array to meet the launch limit for the Delta 2910.

Additional, more detailed weight and mass properties data may be found in Appendix E of Report No. 7.

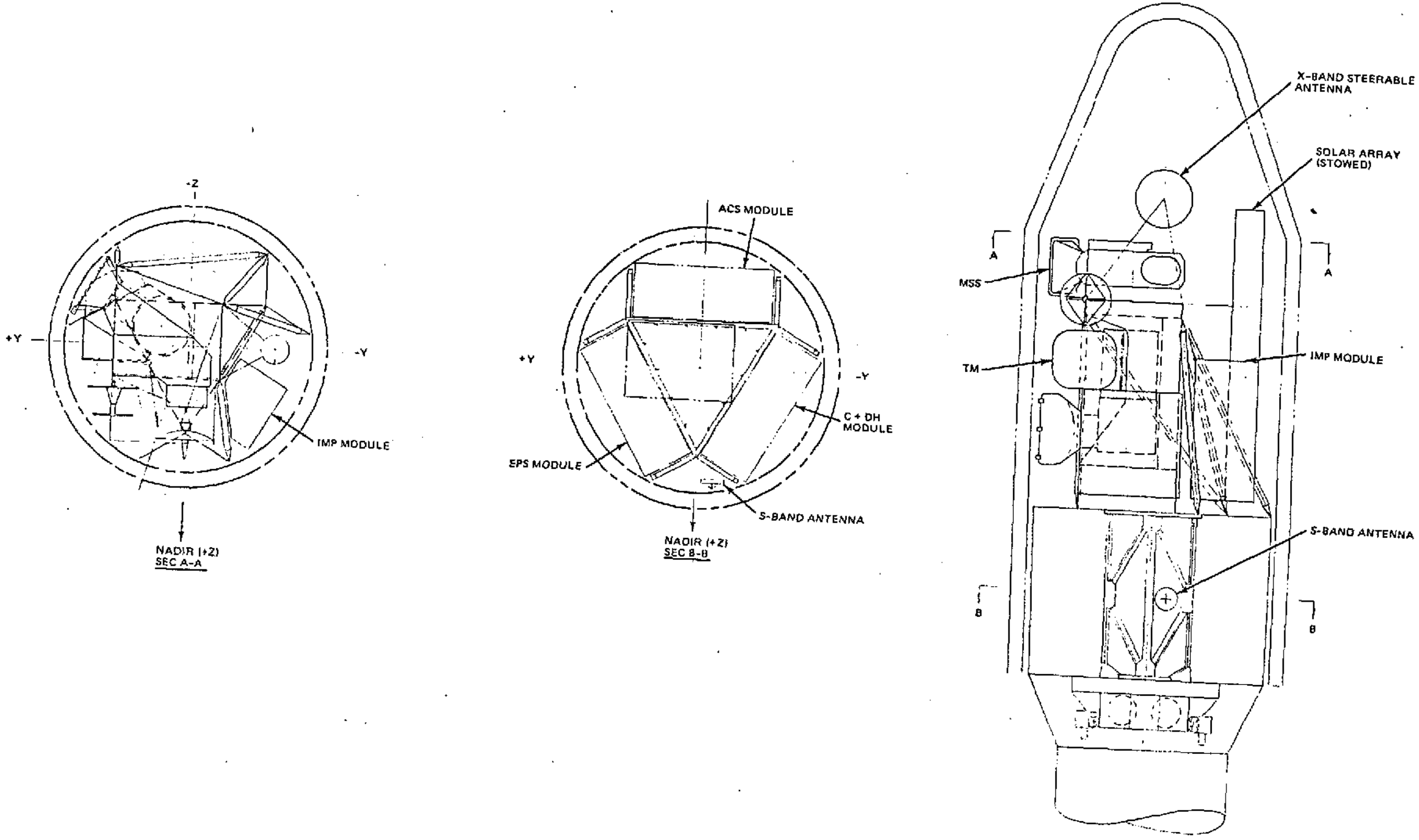
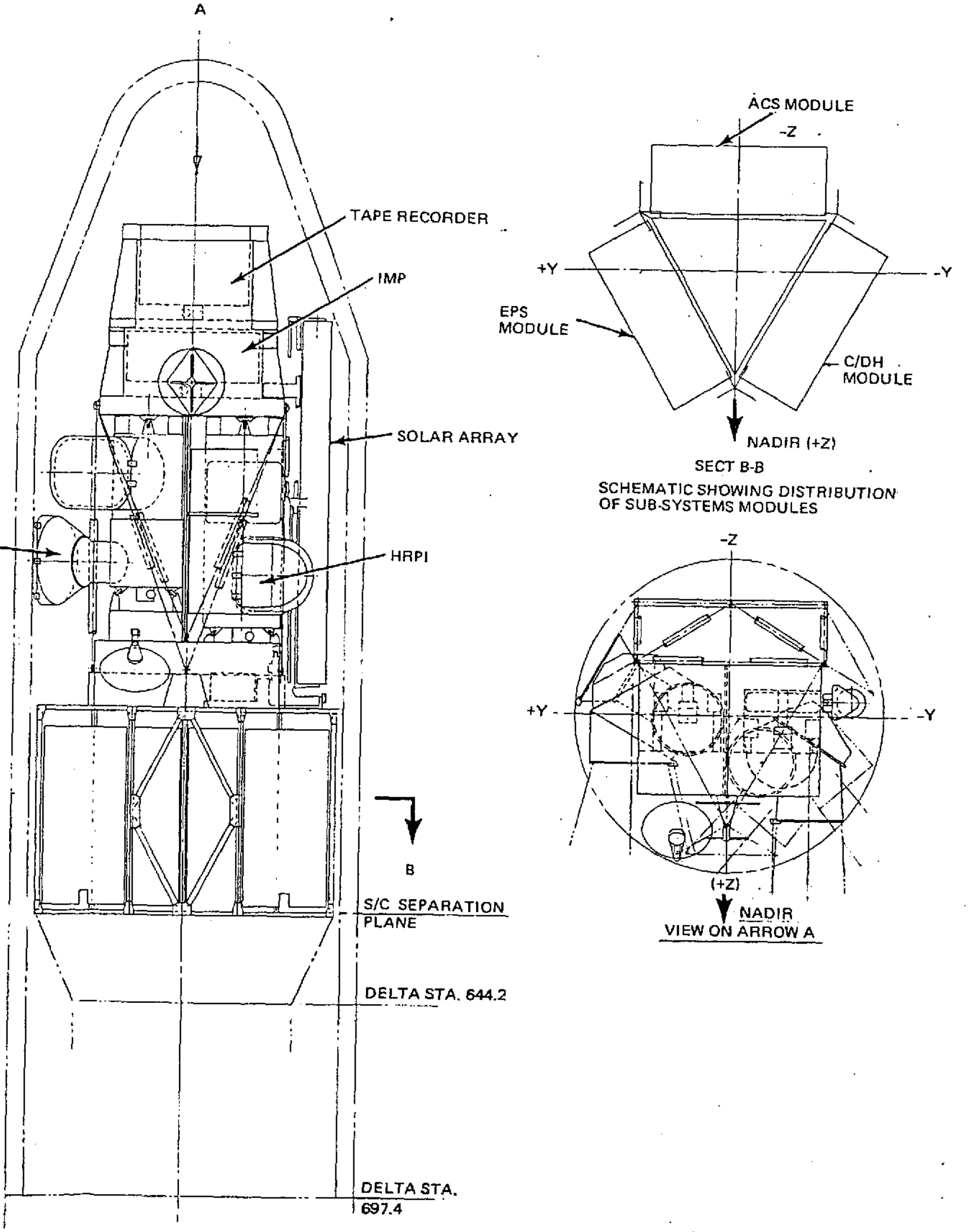


Fig. 5-24 EOS-A Instrument Configuration, Delta Launch Vehicle



3-18, 7-107

Fig. 5-25 EOS-B Instrument Configuration, Delta Launch Vehicle

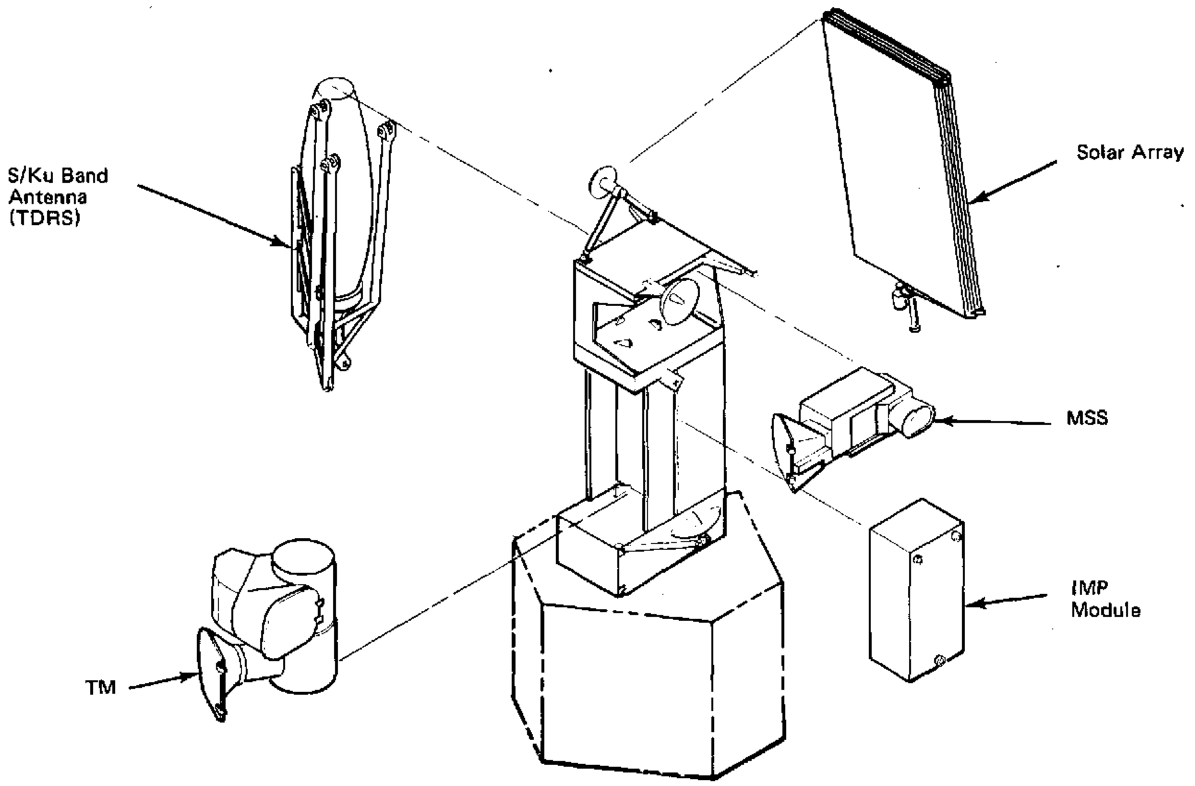


Fig. 5-26 EOS-A (TDRS) Instrument and Mission Peculiar Equipment

7-92

Table 5-7 EOS and Follow-on Mission Weight and Launch Vehicle Performance

ITEM DESCRIPTION	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)	1361	1361	1361	1361	1361	1361	1361	1361
– ORBITER DEPLOY PENALTY	67	67	67	67	67	64	67	67
– ORBITER RETRIEVAL PENALTY	1	1	1	2	1	1	2	2
– ORBITER RESUPPLY PENALTY	–	115	115	128	115	115	128	128
– 2-YEAR SERVICE LIFE (BATTERY)	–	–	32	32	–	–	32	–
– INCREASED STRUCTURAL CAPABILITY	–	–	60	–	–	80	–	–
– Δ CONTINGENCY	11	21	36	26	21	36	26	23
• BASIC SPACECRAFT	1440	1565	1672	1616	1565	1657	1616	1581
– SPACECRAFT MISSION PECULIAR	(47)	(47)	(629)	(200)	(718)	(59)	(175)	(59)
o THERMAL CONTROL	–	–	–	60	–	75	60	60
o SOLAR ARRAY	–	–	84	84	–	-60	61	-75
o ATTITUDE CONTROL	–	–	145	–	–	6	–	37
o COMM & DATA HANDLING	18	18	18	18	18	9	18	18
o ORBIT ADJUST/TRANSFER	27	27	340	27	682	41	27	27
o Δ CONTINGENCY	2	2	42	11	18	-12	9	-8
– INSTRUMENT MISSION PECULIAR	(354)	(425)	(742)	(431)	(428)	(344)	(431)	(467)
o INSTRUMENT SUPPORT (2)	136	189	445	235	198	214	235	231
o TDRSS COMMUNICATION	87	87	87	87	87	–	87	87
o WIDE BAND COMM & DATA HANDLING	88	96	112	46	88	88	46	88
o Δ CONTINGENCY	43	53	98	63	55	42	63	61
– INSTRUMENTS	(560)	(800)	(1700)	(706)	(770)	(2300)	(587)	(1431)
o MULTI-SPECTRAL SCANNER	160	–	–	–	–	–	–	–
o THEMATIC MAPPER	400	400	800	–	–	–	–	–
o HIGH-RESOLUTION POINTABLE IMAGER	–	400	400	–	–	–	–	–
o SYNTHETIC APERTURE RADAR	–	–	500	–	–	–	–	–
o SEASAT-B (OCEAN DYN & SEA ICE)	–	–	–	706	–	–	–	–
o TIROS-O (WEATHER & CLIMATE)	–	–	–	–	770	–	–	–
o SEOS (GEOSYNCHRONOUS EOS)	–	–	–	–	–	2300	–	–
o OTHER EXPERIMENTS	–	–	–	–	–	–	587	1431
• SUBTOTAL – SPACECRAFT	2401	2837	4743	2953	3481	4360	2809	3538
WEIGHT SAVING OPTIONS (3)	–	–	–	-133	–	–	–	–
• TOTAL SPACECRAFT WEIGHT - LB	2401	2837	4743	2820	3481	4360	2809	3538
• LAUNCH VEHICLE PAYLOAD CAPABILITY	2660	3730	5150	2825	3550	4700	3350	3900
o PAYLOAD MARGIN - LB	259	893	407	5	69	340	541	362
o LAUNCH VEHICLE (4)	D2910	D3910	TIIIB	D2910	D3910	TIIIC-7	D3910	D2910

NOTES: (1) BAREBONES SPACECRAFT WEIGHT INCLUDES 146 LB CONTINGENCY.
(2) INSTRUMENT SUPPORT WEIGHT INCLUDES RETRIEVAL STOWAGE LOCKS AND RESUPPLY MECHANISMS (EXCLUDING EOS-A) FOR IMP AND INSTRUMENTS
(3) WEIGHT SAVING OPTIONS EMPLOYED ARE:
a. ROLL-OUT SOLAR ARRAY (EOS-D) SAVINGS INCLUDE CONTINGENCY REDUCTION.
(4) TIIIB PAYLOAD LIMITS ARE FOR TITAN IIIB (SSB)/NUS.

1T-8
7T-2
A7T-2

6 – GROUND SYSTEM ELEMENT – DATA MANAGEMENT SYSTEM

6.1 INTRODUCTION

This section contains the major results of the EOS system definition study in the area of the Data Management System (DMS). The DMS encompasses all aspects of data handling, communication, storage, and processing from the output of the sensors in the EOS spacecraft, to the point where these data are used to generate the final user products – the digital tapes and hard-copy photographs. The overall function of the DMS is to convey the sensor data to the processing facility where it is converted into computer tapes and precise photo maps of the regions of the earth's surface that are viewed by the sensors in the spacecraft. Storage and archiving of both unprocessed and corrected sensor data occurs at the Central Data Processing Facility (CDPF), which is a major subsystem within the DMS. User requests for data are handled by the Information Services System (ISS), a component of the CDPF. The CDPF also contains facilities for correcting the raw sensor data and converting these data into precise latitude-longitude Universal Transverse Mercator (UTM) maps before producing the final output products. These products are then supplied to the users in several optional output formats.

Appendix A of Report No. 7 contains additional supporting and explanative data for the major subsystems of the DMS.

6.2 EOS DATA MANAGEMENT SYSTEM OVERVIEW

The EOS DMS consists of the system elements shown in Fig. 2-3, which are configured to support the EOS program by providing:

- Payload data acquisition and recording
- Data processing and product generation
- Spacecraft and data processing management and control
- All other support services to the data users.

Two types of data acquisition and processing configurations are included in Fig. 2-3. The primary or high-data-rate configuration is made up of Primary Ground Stations (ULA, GDS, and ETC) and the CDPF. Several secondary or LUS's are composed of low-cost receiving, recording, and processing and display subsystems that make up LCGS's.

The CDPF is composed of several subsystems that process payload data, produce data products, and provide for management and control and information and data retrieval services for the data users. Two subareas 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), which is a part of the ISS. Other services include packing and shipping of data products and a data-products scheduling and ordering capability.

6.3 INPUTS TO THE DATA MANAGEMENT SYSTEM

A general guideline for the DMS study has been the estimated range of 10^{10} to 10^{12} bits per day of image data which must be processed by the CDPF. This range has been further subdivided for the purpose of this study to define a minimum-capability, a baseline, and an expanded-capability system as shown in Table 6-1.

Table 6-1 Total Data to be Processed for the Three System Options

	MINIMUM	BASELINE	EXPANDED
THEMATIC MAPPER SCENES (27 M RESOLUTION)	20 SCENES*/DAY	90 SCENES/DAY	400 SCENES/DAY
TOTAL NUMBER OF BITS	4.22×10^{10} BITS/DAY	1.89×10^{11} BITS/DAY	8.44×10^{11} BITS/DAY

7T-200 *TM SCENE: 6912 X 6912 7-BIT PIXELS X 6-1/3 BANDS ≈ 2.11×10^{9} BITS/SCENE

The minimum system in Table 5-7 assumes that TM data is collected only over CONUS, and that a large fraction of these scenes (approximately 50%) are not processed because of excessive cloud cover. Even with these restrictions, however, the processing load for the minimum system is four-to-one greater than the 10^{10} bits/day lower bound. The increases in data load to 90 and to 400 scenes per day assumes that either a WBVTR or a relay satellite (TDRSS) is utilized to allow sensor data to be collected on a worldwide basis.

The data loads in Table 5-7 are based on a TM with six visible bands of 27 m resolution and one IR band with 81 m resolution. Simultaneous operation of an ERTS-type MSS, with four visible bands of 81 m resolution, would increase the total rates less than 10%.

The simultaneous operation of the HRPI, which is planned to have four visible bands at 10 m resolution, could almost double the data rates in Table 5-7. The assumption is made, therefore, that the scene load is divided equally among the two sensors for the TM scenes and 45 (48 m x 185 km) HRPI scenes per day with an input data load approximating 2 x 10^{11} bits/day.

The characteristics of the TM data is shown in Table 6-2. These parameters assume an orbital altitude of 705 km, and a spacecraft ground-track velocity (V_g) of 6.75 km per sec. The corresponding (approximate) MSS parameters are also shown for comparison.

The remaining description of the DMS is based on the assumption that the TM is the only sensor carried by the EOS spacecraft to be processed in the CDPF. The sizing and throughput estimates should, however, apply to combinations of the three sensors (TM, MSS, HRPI) as long as the total data loads correspond to the values in Table 5-7.

6.4 FUNCTIONAL REQUIREMENTS OF THE CENTRAL DATA PROCESSING FACILITY

The CDPF is shown in Fig. 6-1. The major functions of the CDPF are to:

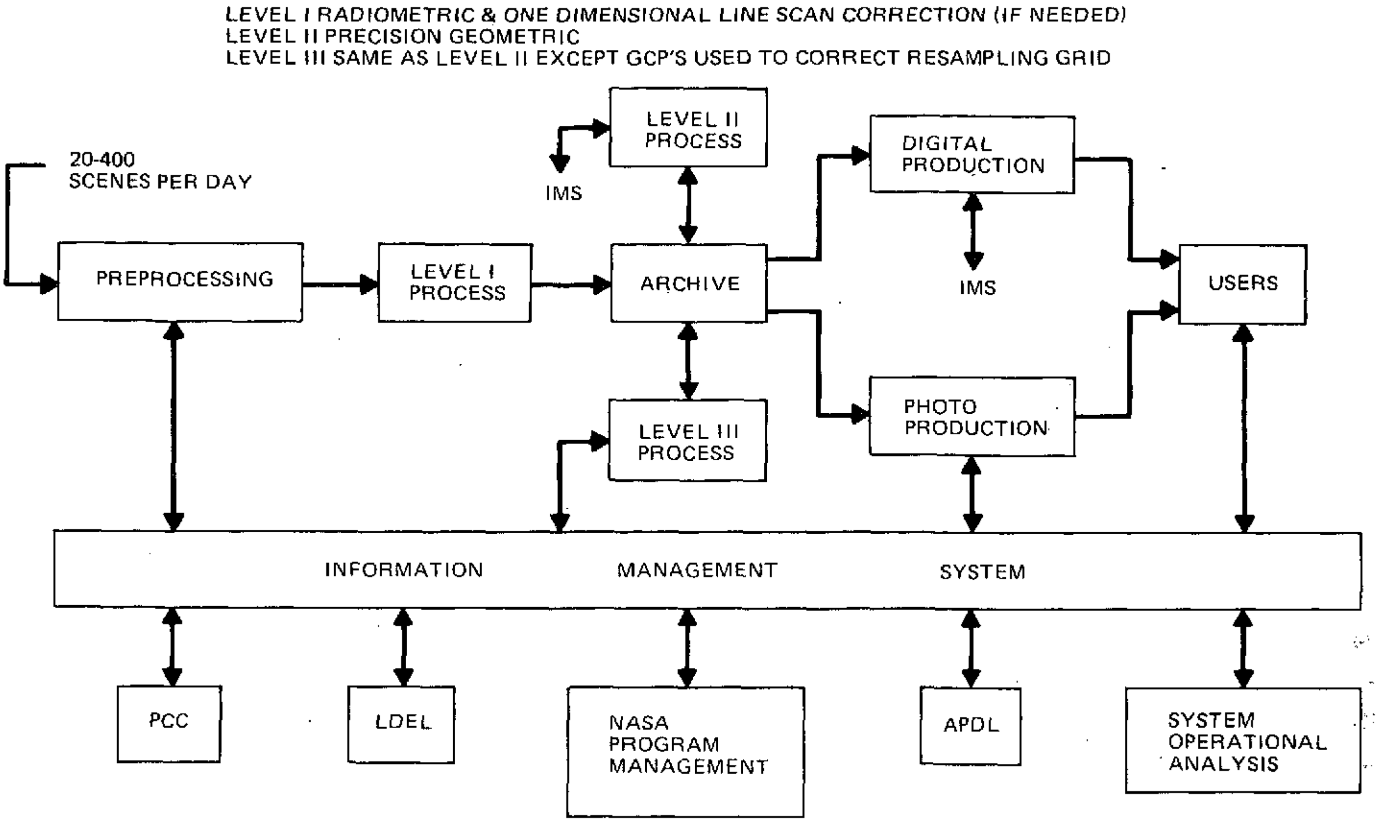
- Maintain an archive of raw and processed sensor data

Table 6-2 Assumed Parameters for Thematic Mapper (TM) and Multi-Spectral Scanner (MSS)

	NOMINAL SWATH WIDTH	185 KM	185 KM
PARAMETER	SENSOR	TM	MSS
1 RESOLUTION	METERS	27.0	81.0
2 LINES/SWATH		18.0	6.0
3 SCAN EFFICIENCY (%)		80.0	45.0
TOTAL SCAN PERIOD	MILLISEC	71.9	71.9
ACTIVE SCAN PERIOD	MILLISEC	57.5	32.3
4 ACTUAL SWATH WIDTH	KM	186.624	185.0
5 SAMPLES/PIXEL		1.0	1.5
6 BITS/SAMPLE		7.0	6.0
7 SAMPLING RATE (EACH DETECTOR OF HIGH RESOLUTION CHANNEL)		119.11 X 10^3	106.06 X 10^3
8 DATA RATE/BAND	MBPS	15.0	3.81
9 NUMBER OF BANDS		6 + IR	4 + IR
10 TOTAL DATA RATE	MBPS	95.0[a]	16.5[b]
11 PIXELS/LINE		6912[c]	3426
12 LINES/SCENE		6912[c]	2284
13 BITS/SCENE	BITS	2.11 X 10^9	2.03 X 10^8
14 PIXELS/SCENE		3.02 X 10^8	3.3 X 10^7
15 TIME TO COLLECT ONE SCENE	SECONDS	27.6	27.3

[a]BASED ON 6-1/3 BANDS
[b]BASED ON 4-1/3 BANDS
[c]SELECTED AS 18 X 3 X 2^7

7T-56



-48
7-102

Fig. 6-1 General Structure of the Central Data Processing Facility

- Perform correction processing on the raw data
- Generate the output products.

Only correction-type processing in the functions of the CDPF have been considered during this study. The interrelationships among the major CDPF functions are shown in Fig. 6-2. Generally, the three levels include:

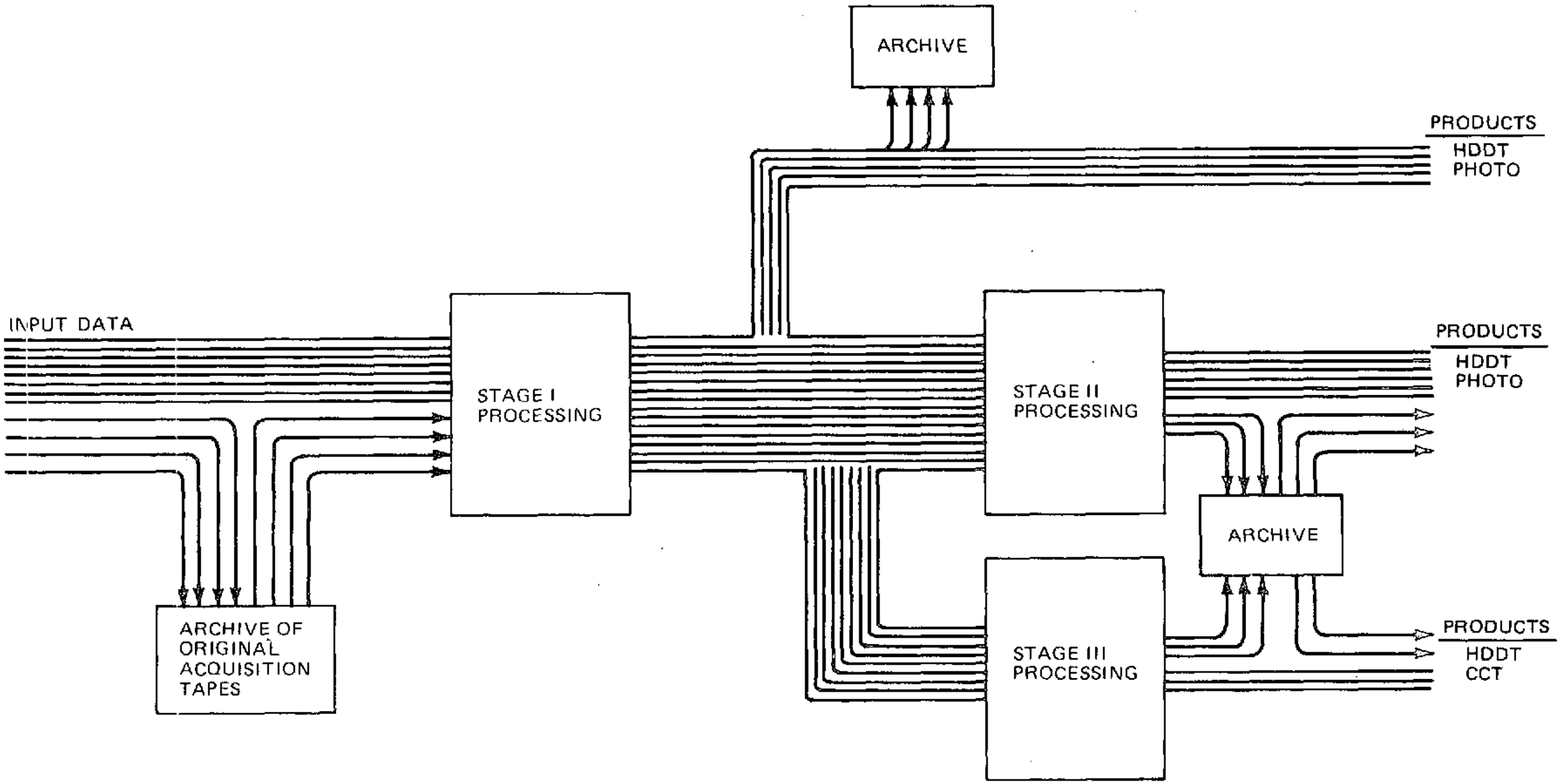
- Stage I - Calibration-type corrections using the calibration data provided with the image data. Included is radiometric correction plus any one dimensional scan correction (line stretching) required by the particular scanner selected
- Stage II – Correction for earth curvature, earth rate, UTM projection, and two-dimensional sensor scan correction (e.g., correct for conical scan), using the best available estimates of attitude and ephemeris
- Stage III – Further refinement of the corrections made in Stage II by using GCP's to improve attitude and ephemeris data. Level III processing would be performed on a certain fraction of the data instead of Level II processing.

Table 6-3 defines the percentages of the total input data which are to be processed and archived at each stage of the processing. These percentages are used to provide cost/performance breakpoints.

Table 6-3 Fraction of Data Processed and Archived

PRODUCT	% DATA PROCESSED	% ARCHIVED	PURGE INTERVAL OF ARCHIVE (MONTHS)
RAW INPUT DATA	0	100	
I	100	100	3, 6, 12
II	50, 100	50, 100	3, 6, 12
III	20, 50, 100	50, 100	3, 6, 12

7T-201



7-49
7-103

Fig. 6-2 Processing, Archival and Product Generation Functions Within DMS

6.5 OUTPUTS FROM THE CENTRAL DATA PROCESSING FACILITY

Figure 6-3 shows the requirement for output (user) products at three points:

- Stage I: HDDT and Photo
- Stage II: HDDT and Photo
- Stage III: HDDT and CCT

The HDDT refers to any very high density tape (>10,000 bpi) which is not directly readable without special interface hardware by a computer. The term CCT refers to other magnetic tapes with density <10,000 bpi that are directly readable by computers. The photo products consist of B&W film (positive and negative), B&W prints, color film (positive and negative) and color prints. The B&W and color film are to be 241 mm (9.5 in.). The B&W film is to be a first generation product; i.e., produced directly from the digital data through, for example, a laser beam recorder. The color film is to be a second generation product: i.e., produced from B&W film.

Not shown in Fig. 6-2 are the so-called custom products. Custom photo products include special gamma correction, special subarea enlargement to specific map scales and special false-color mixes. Custom digital products relate only to CCT and include partial scenes (subareas or areas with reduced swath width), special formats, and images produced at reduced resolution. Initially, as a first order approximation, it is assumed that these custom products require the same processing as required by other products identified in Fig. 6-2.

Table 6-4 shows the range of data products considered for this study.

The CDPF has been sized to handle the required data load in a standard 16-hour day. This implies a 24-hr turnaround for most standing orders. A 10% demand for retrospective orders for data previously archived is also included. These are part of the load defined in Table 6-4.

6.6 MAJOR DMS TRADEOFF AREAS

The major design tradeoff areas considered in this study include:

Table 6-4 Assumptions About Output Product Quantities

PRODUCT	NUMBER (EACH DIFFER-ENT)	AVG COPIES OF EACH	NO. USERS RE-CEIVING	NUMBER FORMATS
HDDT*	2, 20, 200	2	2-20	1
CCT (6250 BPI)	2, 10	1	10-50	5
CCT (1600 BPI)	1, 10	1	20-100	5
B&W FILM	20, 200	1	5-50	3
COLOR FILM	10, 100	1	2-20	3
PRINTS (B&W AND COLOR)	EXISTING ERTS PHOTOLAB		2-20	3

*DISTRIBUTED AMONG STAGES I, II AND III. ASSUME A MIX OF PACKING DENSITIES TO EQUAL TOTAL SPECIFIED. THE NUMBER OF HDDT'S SPECIFIED IS BASED ON PACKING ROUGHLY 10^{10} BITS PER HDDT

7T-57

- CDPF's which handle between 20 and 400 scenes of TM data per day
- The impact on the DMS of the number of data users and the number of different data formats required by these users
- Alternative configuration for the CDPF including minicomputer configurations, large-scale general purpose (GP) computers, or special-purpose (SP) digital hardware
- The concept of the CDPF, i.e., to remain as an R&D facility, to move to a prototype system or to plan at the outset for a full production facility
- An automated, semi-automated, or manual IMS
- Special versus conventional NASCOM communications
- New versus modified PGS's
- A range of modular designs for the LUS's
- Instrument options, primarily the relative complexity of processing data from the linear and conical scanners.

The following conclusions can be made relative to the the first three areas above.

- The throughput rate achievable in a GP processor configuration (minicomputer configuration or large-scale computer) is strongly dependent on the two-dimensional interpolation algorithm selected for use during Level II (III) processing. Throughput rate decreases almost 10:1 when a complex interpolation algorithm such as cubic convolution (CC) (approximation to sin x/x interpolation) is used rather than simple nearest-neighbor (NN) interpolation. Conversely, for a fixed throughput rate, it will cost almost 10 times more to process all data using CC rather than NN interpolation. This cost trend is shown in Fig. 4-6.

- Processing cost for GP configurations is a linear function of scene load. These trends are shown in Fig. 4-6 which apply only to Level I and Level II processing
- The number of data users does not, in itself, have a significant impact on CDPF cost. However, the number of users in conjunction with the copying or replication factor at the facility can have a significant effect on cost. When one moves from the situation of no copying, where only a master of each scene is produced at the CDPF, to the extreme case in Table 6-4 where all users each want a copy of all products, CDPF costs can easily double due to the costs of copying, handling, and expendables
- The cost of the CDPF is relatively insensitive to the number of different formats required by the data users. The decision to reformat the digital data, rather than supply it to users in the "natural" sensor format, does have a modest impact on CDPF cost. However, once the reformatting is undertaken, implying additional processing steps as well as storage, the flexibility of selecting one of several output formats is a relatively minor addition to complexity
- For an input scene load of 20 per day, a minicomputer configuration is less costly, by a factor of approximately 2:1, than a configuration that utilizes large-scale GP computers
- For scene loads beyond 20 per day (above 4×10^{10} bits per day), costs of either the minicomputer or the large-scale GP computer approach become excessive. At the higher scene loads, serious consideration must be given to SP hardware to perform the processing functions. Candidate functions include two-dimensional interpolation, the line stretching required under Level I processing for certain sensor options, and GCP location.

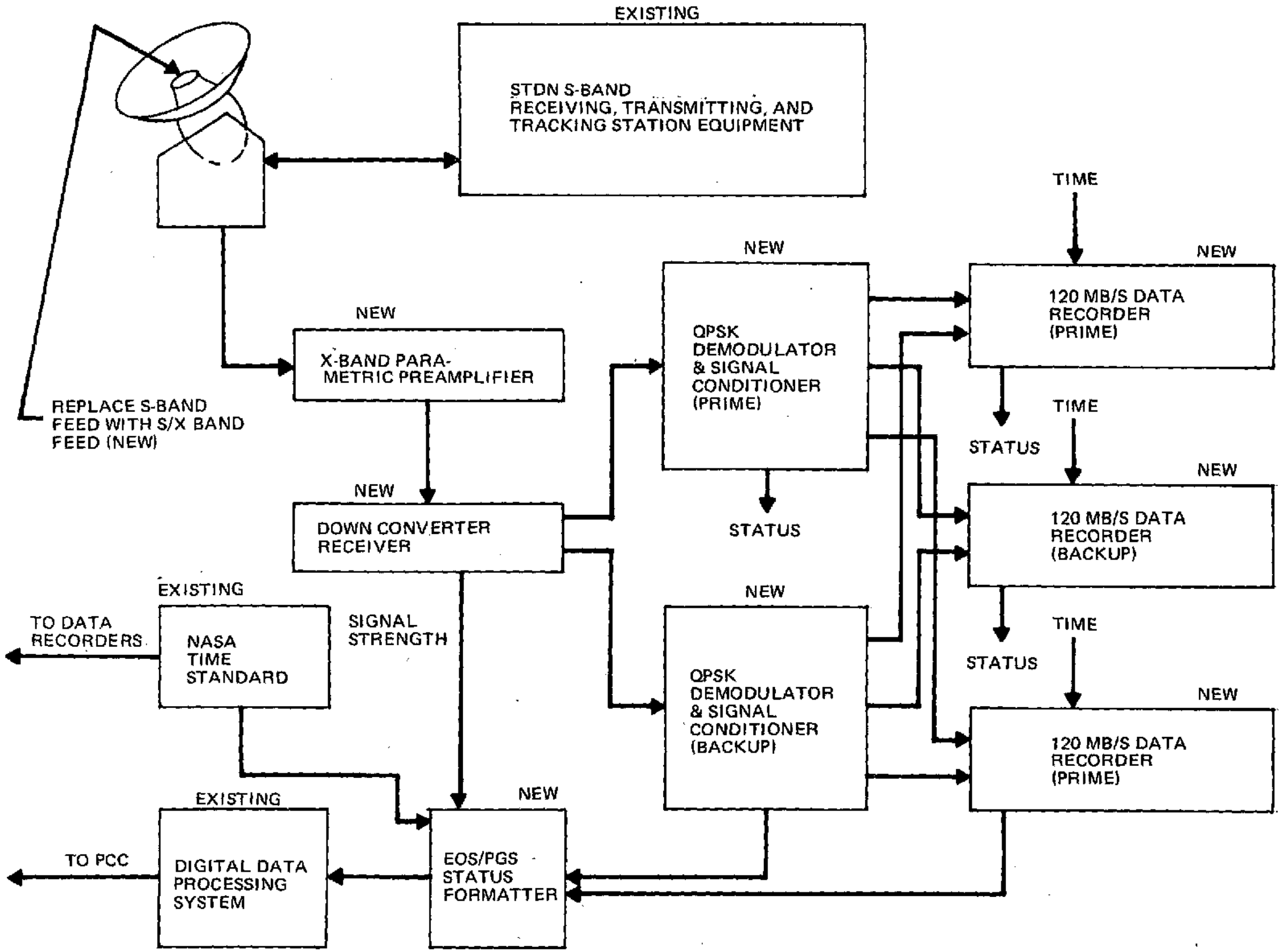
Concerning the fourth and fifth items listed, the following conclusions have been reached:

- The CDPF should be configured as an R&D system, with provision for expansion, rather than a prototype system or a full production facility
- The IMS should be semi-automated for the CDPF with provisions for conversion to a fully automated IMS in the production facility.

A further conclusion is that the planned NASCOM communications capability is adequate to hand to the EOS command, housekeeping, and tracking data needs for the prototype system. Modified STDN PGS's that acquire and record the EOS payload data are less expensive than developing new PGS's. Major conclusions concerning the primary ground stations tradeoff areas follow, and the changes are summarized in Fig. 6-3:

- A new dual S/X-Band feed installed in the existing 30/40-foot STDN reflectors instead of 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.
- A new uncooled parametric preamplifier instead of a new cooled preamplifier. The uncooled unit was selected because it yields adequate performance at minimum cost and maintenance
- A new receiver in lieu of modification to the existing site S-Band receivers. The new unit was selected because of design simplicity and installation, and increased reliability
- Suppressed-carrier QPSK modulation with digital encoding for ambiguity resolution instead of residual-carrier QPSK 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.

A modular LUS that can serve several user-application areas is relatively inexpensive as compared to regional stations. The LUS facility can



3-102
7-77

Fig. 6-3 Primary Ground Station Modifications to STDN Stations consist of a complete LCGS, or it can include only that subset of the LCG equipment needed to process and analyze the image data. Assuming an LUS population of between 10 and 100 terminals, centralized application program development and equipment diagnostic capability can reduce LUS maintenance costs and increase LUS utilization. No program development capability would be needed at the LUS terminals; these facilities could be operated by applications personnel rather than by computer operators and programmers.

The final major area concerns the options for the TM and the impact on the CDPF. The following conclusions apply:

- The required scan linearity (sweeps of the earth which traverse precise equal-angle-versus-time traces) can be met by the two focal-plane scanners being considered for EOS. The object plane scanner (the Hughes approach) is inherently less linear, and linearity can only be degraded by the requirement for high scan efficiency. The solution in the case of the Hughes approach is to calibrate the scanner and then correct the scan either in the instrument, in the spacecraft, or on the ground
- The exact means of correcting for non-linearities in the Hughes scanner has not been resolved at this time. In considering this scanner, it has been assumed for this study that the scan linearity correction must be done in the CDPF during Level I processing. This processing includes modeling the scanner on the ground, interpolating the original data samples, and resampling the data with an equal-angle clock (referred to hereafter as "line stretching"). The impact on the Level I processing is significant if the processing is done in a GP computer. It is strongly recommended that this function be performed with special-purpose hardware and that the necessary radiometric correction (required in all three scanner options) be performed at the same time
- Line stretching is a controversial operation since it requires interpolating the data twice – a one-dimensional interpolation during Level I processing followed by a two-dimensional interpolation during Level II (III) processing. The line stretching could be deferred until the final resampling; however, this deferral would complicate the Level II processing.

The conclusions concerning the Hughes scanner are mostly inapplicable if one of the two focal-plane scanners is chosen for the TM. The Honeywell conical scanner poses a new set of problems for the CDPF, as follows:

- The major impact of the conical scanner on the CDPF is the increased complexity of the coordinate computation algorithm which computes coordinates in the original input array (row and pixel number corresponding to rho and theta, respectively, for conical scanning) for the desired elements in the rectilinear (latitude/longitude) output array
- For the conical scan data the step of coordinate computation is no longer negligible compared to the interpolation step. This increase in the overall number of instruction/pixel is the main impact of the conical scanner – processing of the conical scan data will take longer, or will be more expensive for a fixed throughput rate
- A secondary impact of the conical scanner is in the area of storage. In resampling the original data over the small output blocks which make up the output scene, "extra" data must be stored to assure that all necessary output data can be obtained from the working block of input data. Although this is needed for the rectilinear scanner since the satellite ground track is inclined with respect to lines of longitude, even more data must be stored for the conical scan
- When SP hardware is considered for performing Level II processing, the coordinate computer is more complex for the conical scan data. Initial development cost would then be greater for the processor that handles conical scan data. The SP processor should be as capable of operating at the same throughput rate in processing the conical scan data as with rectilinear data
- Forward or backward facing scan does not make a difference in processing.

6.7 CONCLUSIONS CONCERNING CDP THROUGHPUT

The throughput requirements for CDPF can be examined parametrically by considering the number of scenes of TM data (one scene = 2 x 10^9 bits) that can be processed in one 16-hr day versus the number of machine operations performed on each pixel. The throughput rate is determined by the system I/O restrictions and the rate at which operations can be performed by the processor.

The requirements for input scene load from Table 5-7 can be converted to required input-output rate as shown in Table 6-5. These rates convert the total input data load to a continuous flow over 16 hr.

Table 6-5 Minimum I/O Rate for the Three Input Scene Loads

	MINIMUM SYSTEM	BASELINE SYSTEM	EXPANDED SYSTEM
TOTAL NUMBER BITS	4.22×10^{10}	1.89×10^{11}	8.44×10^{11}
MINIMUM I/O RATE (BITS/SECOND)	7.32×10^5	3.28×10^6	14.6×10^6

7T-202

If assumed I/O rates of the processor is 1, 10, and 100 Mbps, and assumed I/O time and processing time are additive, the relationship shown graphically in Fig. 6-4 is obtained. This set of curves show the total time required to process one TM scene versus the number of machine operations performed on each pixel. The right-hand scale denotes the throughput rate in scenes/day.

For the TM data taken with the linear scanner, the number of machine instructions required for Level I and II processing* using the three interpolation algorithms considered in this study are:

- Nearest neighbor interpolation - 11 instructions/pixel
- Bilinear interpolation - 28 instructions/pixel
- Cubic convolution - 63 instructions/pixel.

The following general conclusions can be drawn from Fig. 6-4:

- The minimum capability system (1 MIP, I/O rate 1 Mbps) is capable of processing approximately four TM scenes/day using bilinear interpolation. With a simpler interpolation algorithm (nearest-neighbor) the throughput rate doubles to eight scenes per day. Using cubic convolution, the rate is reduced to approximately two scenes per day. This configuration is typical of a minimum processing module using one or two minicomputers which process data from a single-port disc
- By retaining the I/O rate at 1 Mbps, throughput rate can approach 16 to 20 scenes per day by going to a (hypothetical) 10 MIPS processor. The system is highly I/O-limited for this situation, and throughput is relatively insensitive to the number of operations performed on each pixel
- Increasing the average I/O rate to 10 Mbps implies some paralleling, either processing the bands separately or reading simultaneously from multiple tracks (or surfaces) of a common scene disk. When matched with a (equivalent) 10 MIPS processor, the configuration can approach 30 to 90 scenes per day throughput rate, depending strongly on the interpolation algorithm used in the Level II processing
- To approach a throughput rate of 400 scenes per day, the equivalent I/O rate must be increased well beyond 10 Mbps and the equivalent processor rate must equal or exceed 100 MIPS.

6.8 CONCLUSIONS CONCERNING PRODUCT QUALITY

The basic objective of the CPS is to perform digital processing on the EOS image data to pro-

----
*If sequential similarity detection algorithm (SSDA) is used to find GCP's, then additional operations, when reduced to a per-pixel basis, are negligible so that the results also apply to Level I and III processing.

duce output products which are as nearly "perfect" as possible. The requirements on product quality affect the CPS in three areas:

- Correcting and preserving the radiometric accuracy of the image data
- Retaining a specified overall resolution in the image data
- Achieving specified relative and absolute geometric accuracies in the images.

A first step in the correction-type processing is to apply the sensor radiometric calibration data to correct the amplitude values of the data from each detector in each spectral band. After this correction for known differences in sensitivity, the CPS must retain a certain accuracy (eight-bit) in the computations performed on the picture elements. For the digital output products, the functions performed should have negligible effect on accuracy. Some degradation will occur, however, in producing photo products so that an additional error, possibly one percent of full scale amplitude, must be budgeted for the transfer from digital to photo product.

The resolution of the overall TM is specified in terms of an overall modulation transfer function (MTF) which takes into account the "smearing" (or filtering) of the true picture material as it is scanned in two dimensions by the sensor. Bandwidth restrictions in the electronics as well as certain steps performed in the digital processing also tend to reduce the MTF at high frequencies, and thereby reduce the clarity with which fine detail can be resolved in the images. Since inverse-MTF filtering (sometimes referred to as "edge sharpening") has been excluded as a CPS function for this study, the goal of the ground processing is to preserve the system MTF, within certain tolerances, as the processing steps are performed. The primary contributors to MTF degradation in the ground processing would be the one- or two-dimensional interpolation algorithms used during Level I and II (III) processing. General goals have been to limit this degradation in the region $f_s/8<f<f_s/2$ (f_s = sampling frequency) to less than 2 dB.

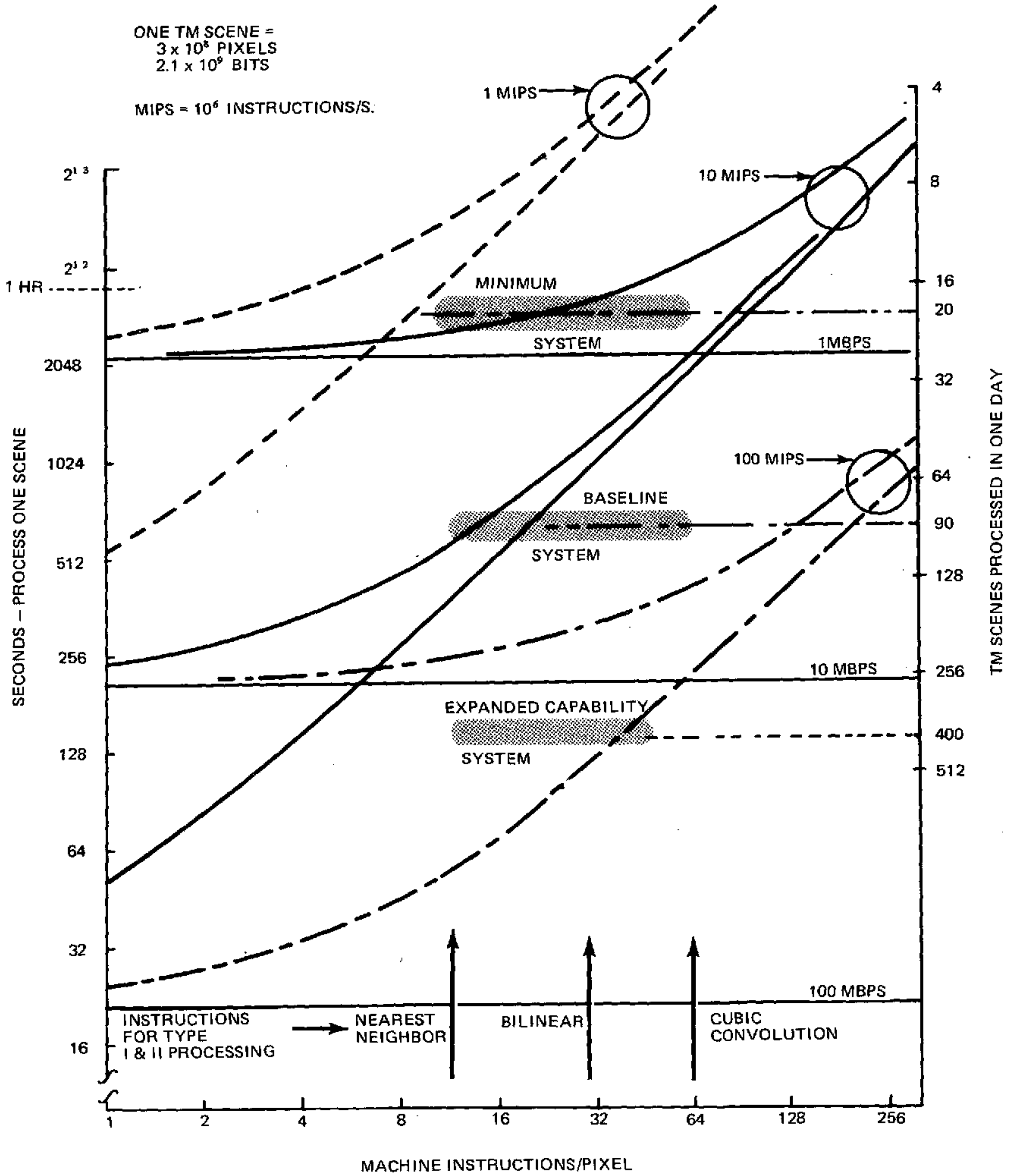
Simulation studies will be required to ascertain the degree to which both the radiometric accuracy and overall system resolution can be preserved through the overall system. These simulations are essential if a realistic apportionment of errors is to be made between the two major system components – the sensor and the ground processing.

Goals for both relative and absolute geometric accuracy of the TM images have been specified for two different situations. In the first, geometric location of individual picture elements are made using measured or predicted estimates of spacecraft ephemeris and attitude. Errors in these parameters will reflect directly into absolute errors in locating the pixels with respect to the desired map coordinates. In the second case, the image data itself is searched for ground control points, whose latitude and longitude are known precisely, and these points are used, in effect, to further refine the estimates of spacecraft position and pointing.

In addition to the absolute accuracy of the image data, a requirement exists for extremely precise relative positioning of the elements within any one scan line, and for precise alignment of successive groups of scan lines. To achieve the desired relative accuracy, the scanner must either be linear or any nonlinearity must be measureable so that corrections can be made. Also, attitude rates must be either extremely low, or again measureable, so that these errors can be corrected.

Given the factors discussed, certain conclusions can be drawn:

- It is essential that scanner nonlinearities be removed and that scan lines be rectified early in the processing if these steps are necessitated by the scanner choice
- The scan correction should be performed in special purpose hardware during Level I processing



7-78

Fig. 6-4 Throughput (Seconds/Scene and Scenes/Day) Instructions/Pixel

- o The automatic location of GCP's in the TM images is a necessary goal for the EOS system
- o Achieving absolute geometric positioning to one-half of a resolution element (15 m) is a desirable goal for the R&D EOS system, but it should not be specified as a firm requirement. Too many uncertainties still exist to justify such a requirement and the cost consequences of meeting it (or attempting to meet it) could be unreasonable.

6.9 IMPACT OF INSTRUMENT OPTIONS/ ORIENTATION

Three possible scanning approaches to the TM design have been retained as inputs to this study (object plane, conical, and focal). Several aspects of these approaches can have major impacts on the CDPF. Generally, these impacts can be summarized as shown in Table 6-6. Clearly, the third approach has the least impact on the ground processing. If this third option is not considered, a selection between the Hughes and Honeywell scanners, strictly on the basis of their impact on ground processing, would favor the Hughes approach by a clear (but not overwhelming) margin.

One additional factor that has a moderate impact on the ground processing is the orientation of the scan lines relative to lines of constant latitude on the earth. This orientation is the same for all three sensor types. The EOS orbit will be inclined at approximately 98 deg. As a result of this inclination, the scan lines will not be parallel to lines of latitude on the earth, but will be tilted with respect to these lines by an angle, θ_L. When blocks of scanner data are processed during Level II processing, some "extra" storage is required over and above that which would be required if the scan lines were parallel to lines of latitude. This extra storage is proportional to sin $(2\theta_L)$ and is required to assure that all data necessary to produce a single output block of picture elements is completely contained in the working array of original data.

If the direction of scan is perpendicular to the satellite velocity, θ_L will vary from 8 to 14 deg during a pass over CONUS. By rotating the scan direction 11 deg from perpendicular, the scan lines become more clearly parallel to lines of latitude and θ_L varies from +3 to -3 deg during a CONUS pass. Estimates are that the extra storage required for Level II processing can be reduced from 50 to 10% by this technique. Thus, to produce an output block of B pixels, an input block containing 1.5 B pixels is required for the unrotated scanner, but only 1.1 pixels if the scanning is rotated by approximately 11 deg.

Table 6-6 Summary of Impact of Scanner Options on Ground Processing

INSTRUMENT	INSTRUMENT IMPACT ON LEVEL I PROCESSING	INSTRUMENT IMPACT ON LEVEL II (III) PROCESSING
HUGHES OBJECT-PLANE SCANNER	– CORRECTION FOR SCAN NONLINEARITY REQUIRED – LINE RECTIFICATION REQUIRED – ALTERNATE SCAN-LINE REVERSAL REQUIRED (ALTERNATE SWEEPS ARE EAST-TO-WEST)	
HONEYWELL FOCAL-PLANE SCANNER (CONICAL SCAN)	– LINE RECTIFICATION MAY BE REQUIRED	– COORDINATE COMPUTATION DURING RESAMPLING (INTERPOLATION IS CONSIDERABLY MORE COMPLICATED THAN WITH RECTILINEAR SCAN). – THE LOCATION OF GROUND-CONTROL POINTS IN THE CONICAL SCAN DATA MAY BE MORE COMPLEX THAN IN RECTILINEAR DATA
TE-GULTON FOCAL-PLANE SCANNER	– LINE RECTIFICATION MAY BE REQUIRED	

7T-36

6.10 ALTERNATE CDP CONFIGURATIONS

Three concepts were considered:

(1) Use of a configuration of multiple minicomputers

(2) Use of a configuration containing special-purpose (SP) digital hardware

(3) Use of a configuration centered around an AAP; specifically, STARAN.

These concepts were focused primarily on the Level II/III processing, which is the largest task in the CDPF. The remaining CDPF elements are generally common to all three approaches.

Level I implementation uses conventional, general purpose minicomputers in the initial (20-scene-per-day) system, and incorporates the minicomputers in a configuration with special purpose hardware for the 400-scene-per-day system. The functions performed by the minicomputers in the 400-scene-per-day configuration include calibration table inversion, annotation data handling, and overall control.

The Level II/III processing system receives as input data the Level I output stored in the archive. The Level II/III output is stored in the archive to be accessed for the generation of products. The major functions that must be considered in any Level II/III processing system implementation are:

- Accept satellite ephemeris and attitude data and compute the parameters of a transformation from scanner coordinates (as rectified by the Level I process) to UTM coordinates
- Select, read from storage, and format the image data for processing
- Compute from the transformation parameters the location of each sample point to be calculated in the interpolation
- Interpolate the rectified image data to form the output image data
- Format and store the interpolated image segments
- Compute the locations of ground control points from the image data (Level II only)
- Perform overall control of the Level II/III process.

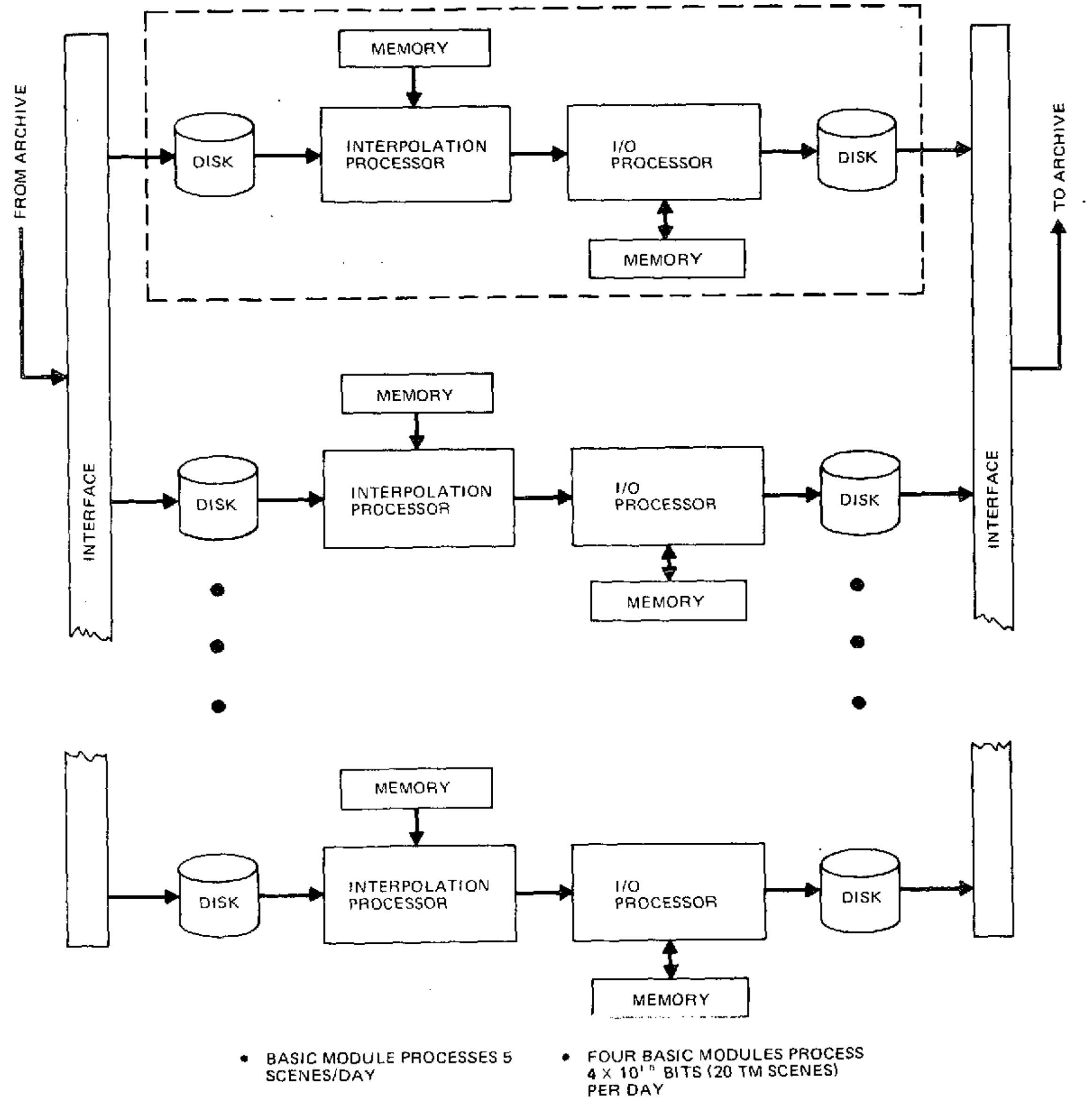
The major cost drivers in all three approaches are the data handling/formatting/storage and the interpolation. The critical nature of the data handling/storage is driven by the enormous quantity of data in a TM scene, the processing speed requirements, and the fact that the output scan lines are titled with respect to the input scan lines.

Figure 6-5, A through C shows, respectively, the three alternative configuration concepts (Options A through C) for Level II/III processing. The basic module of Option A (minicomputer system) uses two processors, one to perform the interpolation and the other to handle the data. The I/O processor is connected to a bank of memory large enough to contain an image segment. The basic module can process five TM scenes per day assuming bilinear interpolation. Four modules can process 20 TM scenes per day. To process 400 TM scenes per day using cubic convolution, 240 modules would be required.

Option B is the special purpose hardware configuration. Interpolation algorithms are switch selectable and are limited to the three methods (nearest neighbor, bilinear, and cubic convolution) considered in the analysis.

The expansion of the special purpose hardware configuration from a minimum throughput version to a 400-scene-per-day system occurs in several stages. A basic single-thread module can handle 15 scenes per day. However, this module can not maintain a continuous flow of data because of the necessity to wait while the various storage elements are handling data. By doubling the disk, and image segment memory elements, the system can operate its storage in a "ping-pong" fashion and thus maintain continuous data flow. These stages of improvement increase the throughput of the module to 30 and then 60 scenes per day. The next stage of expansion is to increase the number of modules. A total of seven modules is required to handle 400 scenes per day. This option provides a system having very little flexibility with respect to processing algorithm modifications or alternative applications of the system.

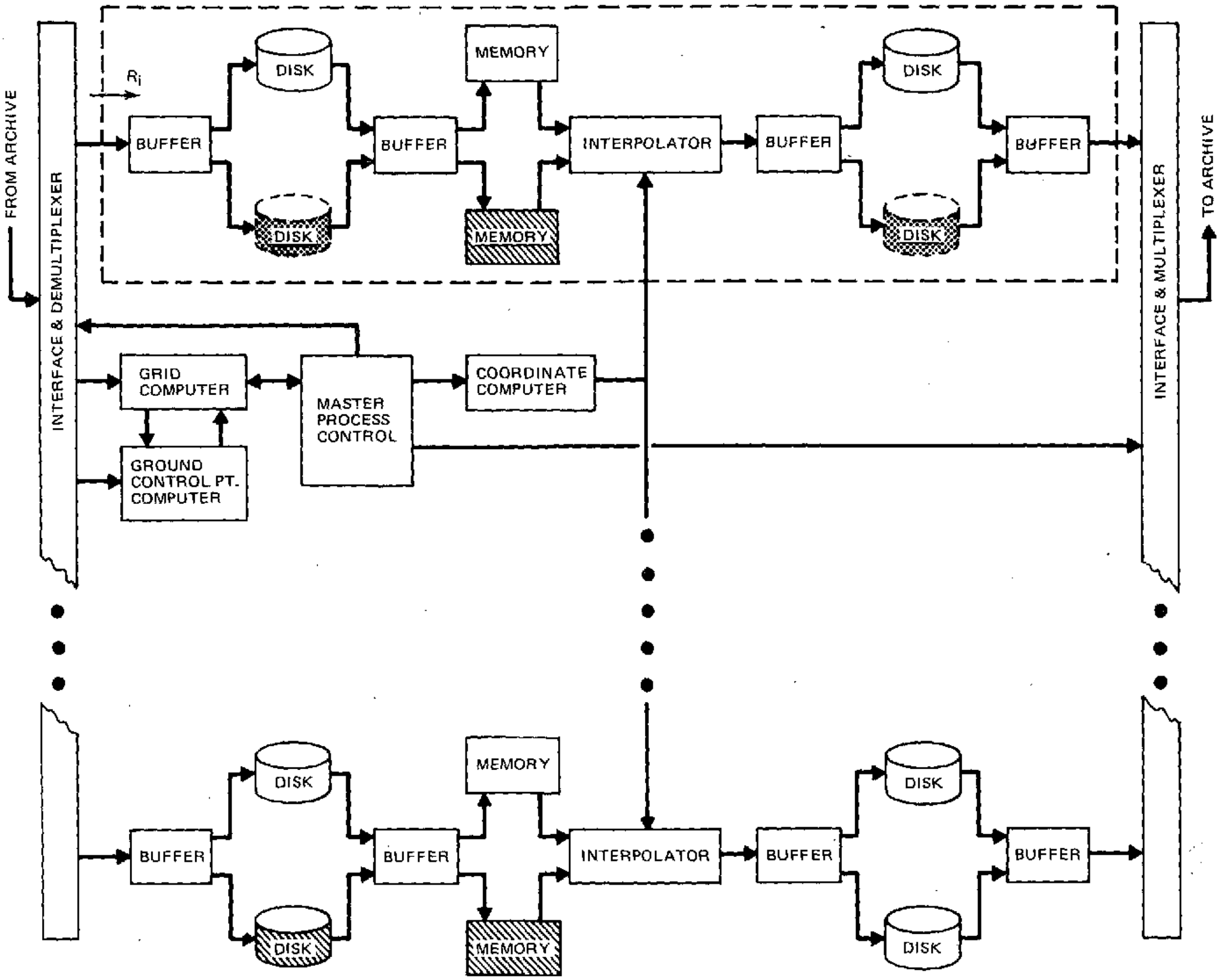
Option C is based on an unconventional general purpose processor, the Goodyear STARAN. The STARAN AAP is a general purpose computer with special architecture oriented toward the common manipulation of tabular data. The configuration uses a "ping-pong" scene disk system for input and output. The STARAN processor operates in a multiphased batch mode. It performs coordinate computation, interpolation, GCP location, and portions of the data handling computation for batches of output pixels sized according to the number of AP words in the configuration. The batches are processed in the AP and then returned to the scratchpad for either intermediate storage or output. The various data selection units



7-51, 7-58
7-104, A7-28

A. Option A – Minicomputer Configuration

Fig. 6-5 Level II/III Processing Alternative Configuration Concepts (Part 1 of 3 Parts)

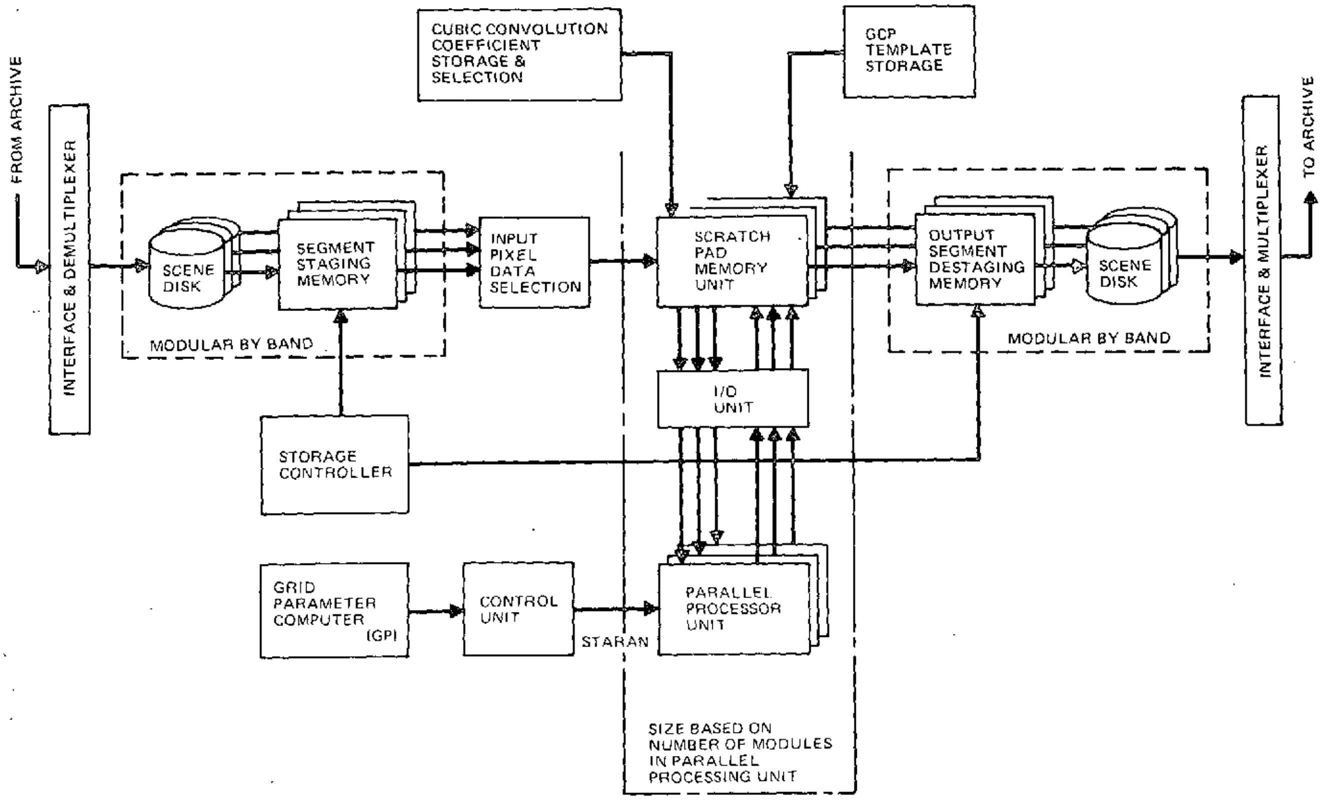


- HANDLES UP TO 12° ROTATION BETWEEN INPUT AND OUTPUT DATA
- PERFORMS RESAMPLING, INTERPOLATION, AND PRODUCES ONE OF 3 OUTPUT FORMATS
- BASIC MODULE SIZED TO PROCESS:
  - 15 SCENES/DAY
  - ADD EXTRA MEMORY PROCESSES 30 SCENES/DAY
  - ADD EXTRA DISKS PROCESSES 60 SCENES/DAY
- SEVEN BASIC MODULES PROCESS APPROX 400 SCENES/DAY (8×10^{11} BITS/DAY)
- THROUGHPUT OF BASIC MODULE
  - $R_i \approx 2.5$ MBPS
  - TIME TO PERFORM CUBIC CONVOLUTION ON EACH PIXEL $\approx 1.5\ \mu S$.

7-52, 7-59
7-105

B. Option B – Special-Purpose Hardware

Fig. 6-5 Level II/III Processing Alternative Configuration Concepts (Part 2 of 3 Parts)



7-53, 7-91
7-106, A7-40

C. Option C – Associative Array Processor (AAP)

Fig. 6-5 Level II/III Processing Alternative Configuration Concepts (Part 3 of 3 Parts)

surrounding the scratchpad are minicomputer or microcomputers that act as data handling controllers.

Two forms of modularity can be used in the configuration. The input and output memory systems are essentially modular by band. The processor, scratchpad, and associated data handling is modular according to the number of arrays in the STARAN. An 11-array system can handle 400 scenes per day, assuming cubic convolution interpolation. For the 20-per-day system, the usual minimum configuration of two arrays is recommended.

Table 6-7 shows a summary of the characteristics of the implementation options. Because Option C provides the flexibility of a general purpose system at a cost comparable to that of special purpose hardware, Option C is the recommended approach.

6.11 INFORMATION MANAGEMENT SYSTEM

The IMS will range from a fairly simple system suitable for providing basic service within an R&D configuration up to the highly automated system necessary in a full production configuration. The basic functions are summarized in Table 6-8.

The image catalog and data inventory function includes all activities involving the maintenance of directors for various purposes within the IMS. The system operates in two major roles, an internal role and an external role. In the external

Table 6-7 Summary of CDPF Optional Configurations

	THROUGHPUT	EXPANDABILITY	FLEXIBILITY	RELIABILITY	RELATIVE COST
OPTION A MINI-COMPUTER	20 SCENES/DAY USING 4 MODULES (BILINEAR) INTERPOLATION)	5, 10, 15, 20 S/D ARE LOGICAL STEPS	MOST FLEXIBLE	@ 20 S/D HAVE FULL BACK CAPABILITY TO 75%, 50% CAPACITY	–
OPTION B S.P. HARDWARE	FULLY PARALLELED - SYSTEM CAN PROCESS 400 S/D USING CUBIC CONVOLUTION	EXPANDABLE IN STEPS, 15, 30, 60,400 S/D	VERY LITTLE FLEXIBILITY	SOME SINGLE-POINT SENSITIVITY; AT 400 S/D, CAN FALL BACK TO 6/7, 5/7, CAPACITY	10:1 CHEAPER THAN A AT 400 S/D
OPTION C STARAN	11 MODULES CAN PROCESS 400 S/D USING CUBIC CONVOLUTION	EXPANDABLE IN STEPS OF APPROXIMATELY 40 S/D	ALMOST AS FLEXIBLE AS OPTION A	SOME SINGLE-POINT SENSITIVITY; AT 400 S/D, CAN DE-GRADE IN STEPS OF 40 S/D	COMPARABLE TO COST OF OPTION B

7T-32
7T-51

role, the function provides the user with a catalog of acquired images. This catalog is further augmented by image descriptors entered by the investigators making use of the various images. In its internal role, the function includes the maintenance of a comprehensive inventory of image originals and data products.

The variation of this function with system option is largely a matter of the extent to which data are included in the catalogs, and the degree of flexibility with which data can be retrieved using the query language.

The user ordering function includes a number of activities in the area of data acquisition and processing. In the data acquisition area, the system provides the user(s) with the capability of requesting specific data acquisition over specific areas on the ground. In the processing area, the

Table 6-8 Summary of IMS Functions

FUNCTIONS	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

7T-203

user ordering system represents the basic medium through which requests for specific data products are received. There are three basic classes of transactions taking place in the user ordering system. The first is a basic request for data previously acquired by the system or for observations defined by the user and known to be within the orbital capability of the satellite. These requests are known as "data" or "observation" requests. A second category of transaction is standing orders. In the case of data products, these represent generalized requests for classes of material by geographic area and quality that are placed prior to data acquisition and without specific knowledge of the specific identification of the data products requested. In the case of observations, these can be generalized requests made too far in advance of availability of specific orbital and coverage parameters. Alternatively, they can be requests for imagery known to be within the coverage capability of the satellite, but where the user is willing to make the request with sufficient generality and flexibility to permit the system to choose among the various possible alternative means of satisfying the request. A third category of transaction is a request for order status information.

The scheduling and control function includes two activities. The major activity is the organization of the product load for the various segments of the IMS, and the generation of appropriate instructions and work orders to efficiently accomplish the processing. In the simplest system, the product load is scheduled and organized on a daily basis and the output of the scheduling process consists of a set of printouts and tapes that provide, in a sequential format, the relevant information required for the day's operations. In the higher capability systems, the frequency of scheduling is increased and the linkage to the operator made more immediate to the extent that the scheduling is performed online at the time of the user request and is immediately available for operator or system action.

The second activity of the scheduling and control system is the monitoring of product quality. This activity is effectively performed through the use of standard data and work orders generated by the IMS. The standard data includes both an input data set and output comparison set for use in testing various stages of processing. The comparison can be made either manually or automatically as appropriate to the nature of the process and the capability level of the IMS.

The accounting, reporting, and historical data system performs two functions: budget control and cross tabulation of products. The accounting/reporting activity accumulates and summarizes detailed data on system utilization both for individual users and on an overall basis. Overall control of the system is exercised by the Project Manager by assigning each user some form of budget. The accounting system then functions in a manner similar to the accounting system used with a computer batch processing or time sharing system involving a large number of users. The variation in the accounting/reporting activity with system option is essentially a function of the reporting cycle. In the lower capability options, each user transaction is journalled and a monthly summary produced. In the higher capability system, the accounting data is kept online and a daily summary provided to the Project Manager. The users can request summaries of their own status and activity on the same basis as any other system transaction.

In the higher capability system, a second activity of the accounting system is the maintenance of a cross-tabulation of data product by users requesting them. This cross-tabulation is maintained in conjunction with the system directories and permits the individual user (on request) to determine the identity of all other users requesting imagery in a specific category (primarily geographic area). Software provisions permit any user to exclude transactions from being included in the cross-tabulation. The purpose of this activity is to permit users interested in specific imagery to obtain information by means beyond those provided by the image descriptor index.

The IMS will include the product routing and delivery functions of the CDPF system. In the lower capability options, product routing is limited to mail delivery only. Under these conditions the function of the IMS will be to print appropriate labels in conjunction with the work order and scheduling process. In the higher capability IMS options, there will be provisions for limited delivery of small, high-priority digital data orders directly to remote user terminals. This function will be performed in a manner analogous to the delivery of output files in a remote batch system.

Three options for the IMS configurations are summarized in Table 6-9, ranging from a relatively simple, to a relatively sophisticated, management system.

6.12 STORAGE AND ARCHIVE

The EOS "archive" could consist of simply a special room or vault which contains HDDT's of either the raw EOS data or that data which has received a certain level of processing. In this case, the archive would consume only floor space, which could become considerable if data storage is envisioned over a long period of time, but little else in the way of equipment. Retrieval from the archive would be essentially a manual operation with direction supplied by the IMS.

At the opposite extreme, a much more sophisticated archiving facility where manual operations are reduced to an absolute minimum can be configured. This highly automated system would be desirable in two situations: First, in an R&D type system, where maximum flexibility is essential to access portions of the total data base; second, in a full production system to maintain a high throughput rate, and to organize and keep track of the extermely large volume of image data, both processed and unprocessed. However, a highly automated archive facility tends to become expensive.

Given that the decision has been made to go to an automated archive, there are several digital archiving systems that will suffice. The Ampex TBM (terabit memory) was selected for detailed examination during this study since it can provide a basic system at moderate cost for an initial R&D CDP which can be expanded later to provide increased amounts of essentially "online" storage.

A minimum form of the TBM is shown in Fig. 6-6. Basic features of this configuration include:

- Record and playback simultaneously at 5.6 Mbps rates

Table 6-9 Summary of IMS Options

ITEM	IMS OPTION 1	IMS OPTION 2	IMS OPTION 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 OPERATOR REQUEST	DETAILED SCHEDULE OPTIMIZED FOR CURRENT LOCACTION OF DATA
• CATALOG LEVEL OF DETAIL	SIMPLE	SIMPLE	CURRENT DATA LOCATION
• SENSOR REQUEST LEAD TIME REQUIRED	HIGH	MEDIUM	LOW

7T-37

- Use of standard magnetic video tape (TBMTAPE)
- Tape capacity is 4.5×10^{10} bits (approximately 20 TM scenes)
- Two independent tape transport units (automatic switching when one unit is full)
- Average access time 15 sec, worst case 45 sec
- Independent read and write channels in data channel (DC) giving I/0 rate of 11.2 Mbps
- Uncorrectable error rate 1.5×10^{-11}.

Included in the configuration of Fig. 6-8 is a storage control processor (SCP) which maintains a master file directory of all data files. The SCP also manages internal work queues which are generated by requests from the IMS. The DC processor can transfer data either directly to the processing units, or indirectly to other processing units through a shared disk. The basic storage units can be expanded up to six-fold by adding additional transport drives and dual transport modules (TD and DTM in Fig. 6-6).

6.13 OVERALL CONCLUSIONS & FURTHER STUDY AREAS

Certain overall conclusions have emerged from this study. These follow as do recommendations for continued investigations.

- The overriding drivers of the DMS in EOS are the enormous quantity of data that can be supplied by the instruments, and the requirement that the data be corrected to near perfection before being delivered to the users. Conventional computers cannot perform the necessary corrections to one TM scene in two minutes, the time available at the highest throughput rate of 10^{12} bits/day
- The CDPF has lagged in the planning for EOS where the emphasis remains on the instruments and the spacecraft. If continued, this lag could result in either a severe backlog of unprocessed data at the CDPF, or compromises in the quality of data that is supplied to the users
- Instrument development for EOS has been directed toward achieving better radiometric accuracy and linear, near-perfect scanning, both highly desirable goals from the user standpoint. The critical importance of removing scan imperfections from the system must be re-emphasized since failure to meet these goals can impose a serious burden on the ground processing facility
- The goal of achieving $\pm 1/2$ pixel absolute geographic location accuracy of each TM pixel for EOS, although certainly desirable, must be reviewed carefully. At present, there is no guarantee that this tolerance can be met. The problems with geometric precision of the ERTS images are still being uncovered as users process the digital data more microscopically and assign significance to individual pixels. Considerable effort is being consumed in correcting and registering the ERTS images. This effort can only increase with the higher-resolution EOS data
- A giant step in data processing has been specified for EOS: all data shall be processed digitally to produce near-perfect products. Preparation for this step should be made through a logical sequence of design, prototype development, and concept validation stages. Such investigations, specifically into the feasibility of pipeline or array processors, should be undertaken immediately with ERTS data with a prime goal of developing concepts for the several orders of magnitude increase in throughput implied by EOS
- A comprehensive set of user requirements does not exist for EOS. Consequently, some difficulty has been encountered in sizing and estimating costs for systems that process 10^{10} to 10^{12} bits/day. The algorithms to be used in correction-type processing have not been standardized (each user seems to develop his own). Steps should be taken to assure standardization in the near future
- Correction-type processing is generally viewed as a necessary, but trivial, set of processing steps. This is understandable since the analysis-type processing is much more challenging. This view must change when faced with the data loads proposed for EOS; the correction-type processing (sometimes referred to as "preprocessing") can easily dominate the CDPF functions.

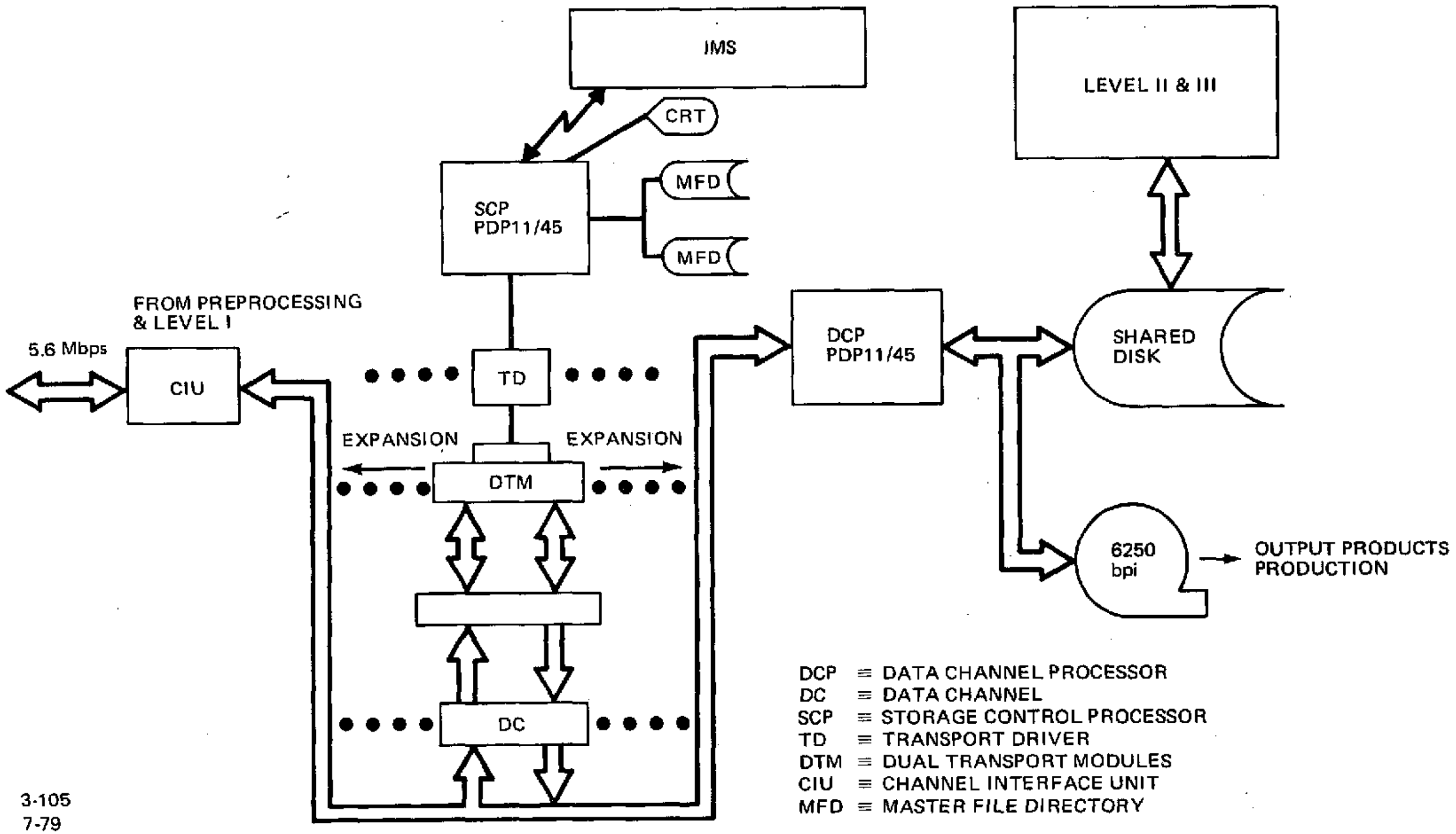


Fig. 6-6 Minimum Archive System for the R&D CDP Which Provides Ready Expansion

Given the current status of the CDPF for EOS, three related tasks are recommended:

- An error analysis of the TM images should be performed to realistically budget errors among the various subsystems of the DMS. This analysis must include the spacecraft, the ACS, the sensors, and the algorithms used in the CDPF. Simulated imagery should be generated to indicate the quality achievable in the overall EOS system
- The algorithms to be used in the CDPF should be standardized and agreement should be reached on a standard set of output products
- Prototype SP digital processors should be developed and tested to accomplish line stretching, two-demensional resampling/ interpolation (including image rotation), and GPC location. Modularity must be emphasized in these developments so that they are compatible with an expandable system.

6.14 LOCAL USER SYSTEM

A systems viewpoint has been taken with respect to the study and design of a wide family of conceptual EOS LUS's which includes the LCGS concept. The LUS's are complete ground stations and data processing systems that permit users to access the EOS data directly on a receive-only basis. Received data are a coverage subset of the data transmitted to PGS's. For example, high rate TM or HRPI data are compressed (reduced resolution or fewer bands) such that LCGS's receive data at about a 20 Mbps rate. However, MSS data can be received at full resolution. The purposes of the LUS are to:

- Display EOS image data in B&W or color visual images
- Produce copies of the visual images
- Format and selectively edit the image data
- Radiometrically and geometrically correct the image data
- Provide an analysis capability to support a multiplicity of applications.

The LUS terminal is composed of three subsystems: an RF/IF, data handling/recording, and processor and display subsystems. Major equipments within the subsystems and their interconnections are shown in Fig. 6-7.

A basic approach in developing the LCGS hardware and software configurations was to consider a range of user capabilities for the display, hardcopy, processing, and analyses of EOS payload data. The processor and display subsystem (PDS) could be composed of equipments to meet different local user needs and budgets at the time of system acquisition, but the PDS could be easily and inexpensively expanded later should the need arise. Table 6-10 lists capability and costs for various LCGS configurations.

The remaining subsystems, RF/IF and data handling/recording, are of single-capability design. They will properly interface any capability PDS.

Several methods of EOS payload data delivery are possible, but they are cost-effective only with respect to particular EOS system configurations. For example, dial-up 50 to 56 Kbps wide band common carrier lines should be available throughout CONUS during the EOS time frame. Computer-to-computer telecommunications could be efficiently used for delivery of selected centrally processed data to the LUS's. Additionally, CONUS DOMSAT communications could enable central to LUS data channels at 50 to 80 Mbps rates, depending on the selected EOS configuration. Thus, some user may require only the PDS configured to receive data from a centralized ground station. However, the current study concentrated on the direct-delivery method. EOS to user communications at data rates of about 20 Mbps were assumed as well as a CONUS LUS population of 50 to 150 units. The system concept could easily be expanded to include LUS's in foreign lands or contracted to just a few CONUS LUS's.

The systems concept provides modular hardware and software capabilities for the LUS's that would be complemented by centralized support capabilities. The centralized support

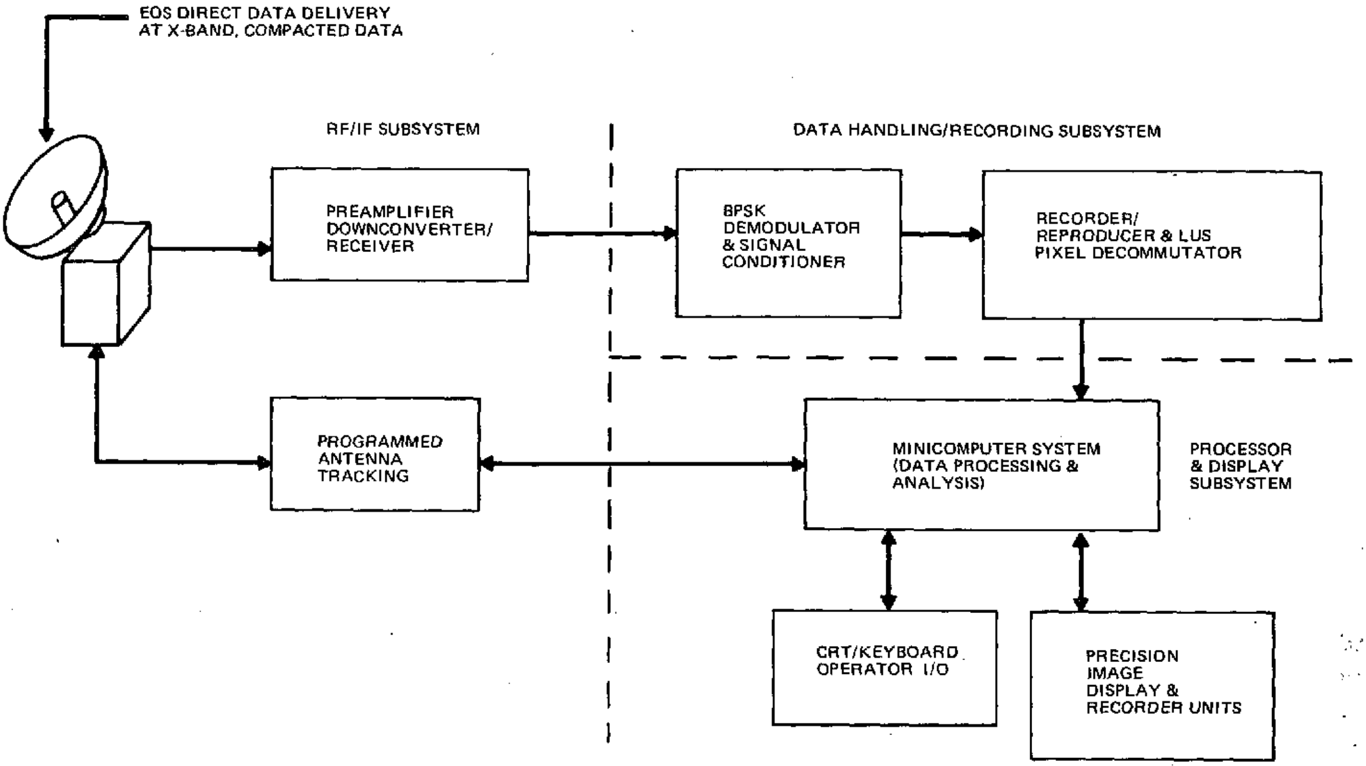


Fig. 6-7 Basic LUS Terminal Configuration

Table 6-10 Low-Cost Ground Station Hardware Cost Vs Capability 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)	130
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)	223
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)	300

3-255
7T-11
7T-35

elements are assumed to be located within the GSFC complex, and are co-located with (and within) the IMS, PCC and CPS.

The 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 LUS's. 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.

A basic assumption for the centralized/local system concept is that the LUS operators are primarly 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 LUS's is provided by the shared centralized system elements. This concept would only be cost-effective if there were many LUS's. The breakpoint for the cost-effectiveness has not been determined, but from experience it is estimated that it would be cost-effective if more than five to seven LUS's were implemented.

Adding or eliminating the APDL and LDEL elements does not affect the acquisition, display, and processing capabilities of the LUS. However, one centralized element 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 CCT's and picture products. Additionally, the operators would receive EOS orbit predictions and coverage time information from the IMS to point their local antennas and acquire the direct EOS-to-LUS data transmissions.

7 – DESIGN EVALUATION AND PREFERRED APPROACH

The design cost trades presented in Report No. 3 of this study were performed on an individual basis. Although the conclusions were, and still are, applicable for an individual trade area, a method must be applied to tie all tradeoff conclusions together from an overall design standpoint. The method employed must be capable of integrating sometimes diverse trade outputs into an evaluation of the total EOS design and provide an insight into which overall approach is preferred.

The approach we have chosen is to develop a system effectiveness model. This model relates system and hardware design and performance parameters to a single effectiveness Figure of Merit (FOM), which reflects top level program objectives. Thus we are able to:

- Evaluate the effectiveness of alternate designs in meeting program objectives
- Relate effectiveness of the alternate designs to cost.

The latter step is a clear representation of *system cost effectiveness.*

7.1 THE EOS FIGURE OF MERIT

The effectiveness FOM we have chosen is "expected number of equivalent scenes per week", which expresses the expected system yield (in probabilistic terms) of a normalized data product. The standard for this normalized data product is equivalent to a TM scene having the following characteristics:

- 185 x 185 km ground size
- 7 spectral bands of radiometric data
- ∓ 15 meter spatial resolution
- A composite digital output format which reflects the expected variations in both product mix and level of correction.

The foregoing defines a standard data product having a given *quality,* and permits all other data products to be related or "normalized" to this standard. Ignoring probability, the number, or *quantity* of such equivalent scenes that a system can deliver thus is a measure of that system's capability of effectiveness.

The next stage is to factor in system reliability and operational duty cycle, which is combined in the term *availability*. A system having a high yield of equivalent scenes, but very low reliability may not be as effective as a lower yield, highly reliable system. Similarly, a system that produces very high quality scenes, but at a 30-day revisit cycle, may not be as effective as a lower quality, 7-day revisit system.

Thus, the selected FOM combines *quality, quantity,* and *availability*. Some examples of the system and hardware parameters which contribute to these three factors in our model are:

- Instrument performance such as resolution and number of spectral bands
- Number of satellites in orbit
- Swath size
- CDP throughput
- Satellite mean mission duration
- Number of transmitted scenes (i.e., effect of WBVTR and TDRSS)
- Shuttle utilization mode.

To combine the many system parameters in the model it is, of course, necessary to weight them in accordance with how they relate to program objectives. Since program objectives are ultimately tied to user requirements, our method of weighting the parameters is to relate their range of performance to percentage of known user population satisfied. Although this approach is subject to some variability in terms of who the users are, and what satisfies them, our data in this area tracks very well with NASA surveys.

7.2 USER REQUIREMENTS WEIGHTING FACTORS

Information from different instruments provided in different formats has different values to the users. These values are estimated by the weighting factors given in Table 7-1. The factors were derived by analysis of the results of the extensive user requirements survey performed during the system definition study. They represent, in each user area, the percentage of the applications in the areas which would be served by the value of the parameter indicated in the table.

Table 7-2 shows the quantity, quality, and availability of the data output of each potential EOS instrument. The quantity of data is expressed by the number of equivalent TM scenes which the instrument can produce in one week, based upon its data rate (in bit per second) and the anticipated usage of the instrument. The quality of the data produced by each instrument is measured by the weighting factors. The ratio of two weighting factors expresses the relative value of one scene from each instrument to an "average user". Thus, weighting factors are chosen to be indicative of the economic value expected to be gained from utilizing the information provided by each instrument. The availability of an instrument is expressed by its uptime fraction. The expected number of equivalent scenes producible by an instrument is the product of these three numbers. This product is the primary measure of system effectiveness. It is multiplied by other factors expressing: the reliabilities of the launch vehicles (Table 7-3), the success of the subsystems at meeting design objectives (Table 7-4), the scores of orbit parameters relative to design goals (Table 7-5), and the ability of the data processing system to transform the data into desired products. The resulting number is the "expected number of equivalent scenes produced." The factors previously mentioned have the effect of reducing the expected number of equivalent scenes below that produced by the instruments, because the

Table 7-1 Instrument Weighting Factors

FACTORS AFFECTING AVAILABILITY OF DATA PRODUCTS	WEIGHTING FACTORS
1. TIMELINESS OF DELIVERY TO USERS	0.3
2. AVAILABILITY OF PRODUCT FORMATS	0.05
3. RESPONSIVENESS TO SPECIAL REQUESTS	0.2
4. RELIABILITY OF DELIVERY SYSTEM	0.4
5. FORMATS AVAILABLE THROUGH LUS	0.05

7T-204

Table 7-2 Instrument Data Output

INSTRUMENTS	UPTIME FRACTIONS	SCENES/ WEEK	WEIGHTING FACTORS
TM	.927	300	.35
HRPI	.846	250	.25
MSS-4	.935	200	.15
MSS-5	.935	300	.20
SAR	.895	20	.10
PMMR	.865	20	.10

7T-205

Table 7-3 Launch Vehicle Reliability

LAUNCH VEHICLE	RELIABILITY
DELTA	0.89
WEIGHT-CONSTRAINED TITAN	0.89
TITAN III B	0.96
TITAN III D	0.91
SHUTTLE	1.00

7T-206

Table 7-4 Subsystem Success at Meeting Design Objectives

SPACECRAFT SUBSYSTEMS	WEIGHTING FACTORS
1. ATTITUDE CONTROL	0.125
2. COMMUNICATION & DATA HANDLING	0.125
3. POWER	0.125
4. MISSION PECULIAR	0.125
5. PROPULSION/ORBIT ADJUST/REACTION CONTROL	0.125
6. WIDE BAND COMMUNICATION	0.125
7. TAPE RECORDER	0.125
8. SOLAR ARRAY	0.125

AVAILABILITY OF SYSTEMS = 0.95

7T-207

Table 7-5 Orbital Parameter Scores

ORBIT PARAMETERS	WEIGHTING FACTORS
1. REPEAT CYCLE	0.07
2. SWATH OVERLAP	0.07
3. ORBIT ADJUST FREQUENCY	0.07
4. SWATH SIDESLIP RATE	0.07
5. GROUND STATION COVERAGE	0.07
6. SUFFICIENT SHUTTLE PAYLOAD	0.08
7. TIME OF DAY	0.07
8. PERCENT OF LAND HRPI REVISITABLE	0.08
9. MAPPING COVERAGE	0.07
10. MINIMUM-COST BOOSTER	0.07
11. SUN ANGLE	0.07
12. TRACKING & DATA ACQUISITION	0.07
13. FULL EARTH COVERAGE TIME	0.07
14. TIME BETWEEN RESUPPLY VISITS	0.07

SATELLITE MMD = 2.75 YR

AVAILABILITY OF INSTRUMENTS = 1.00

7T-208

other systems are not perfect. The FOM then goes through two additional weightings to express its success at meeting specific mission and program objectives. Table 7-6 shows the LRM mission objectives identified in the Requirements Document (Report No. 3, Appendix C). Objectives 1 to 5 of Table 7-6 express the goal of providing data to the application areas named. Objectives 6 to 9 are requirements on the orbit chosen for the satellite. Objectives 10 to 18 are requirements on the satellite design. The design in question is scored for each objective, and the weighted sum of the scores is multiplied by the expected number of scenes. Table 7-7 shows how the scores of the applications areas mission objectives are computed. The analysis of user needs yielded information about the fraction of applications requiring a certain standard of information: frequency of observation, resolution, and spectral bands. The orbit and instrument complement of the satellite determine how many

Table 7-6 LRM Mission Objectives

LRM MISSION OBJECTIVES	WEIGHTING FACTORS
1. AGRICULTURAL APPLICATIONS	0.2
2. FORESTRY APPLICATIONS	0.05
3. LAND USE APPLICATIONS	0.1
4. WATER RESOURCES APPLICATIONS	0.1
5. GEOLOGY APPLICATIONS	0.05
6. SUN-SYNCHRONOUS ORBIT	0.02
7. 5% OVERLAP ORBIT	0.03
8. 17-DAY REPEAT CYCLE ORBIT	0.01
9. 3-DAY HRPI OFFSET REVISIT	0.06
10. SENSOR COMPLEMENT	0.05
11. MISSION LIFETIME	0.03
12. SATELLITE LIFETIME	0.03
13. DATA PRODUCT OPTIONS	0.05
14. PROCESSING LEVEL OPTIONS	0.04
15. DATA THROUGHPUT TIME	0.05
16. DATA THROUGHPUT QUANTITY	0.04
17. SPECTRAL BANDS	0.05
18. RESOLUTION	0.04

7T-204

Table 7-7 Mission Objectives Applications Areas

	FRACTION OF APPLICATIONS SATISFIED BY									
	FREQUENCY OF OBSERVATIONS No. OF WEEKS				RESOLUTION, METERS			AVAILABLE SPECTRAL BANDS		
AREA OF INTEREST	3	2 TO 3	1 TO 2	1	60	30	10	4	5	7
AGRICULTURE	.60	.65	.71	1.00	.29	.88	1.00	.40	.60	1.00
FORESTRY	.34	.38	.72	1.00	.0	.44	1.00	.60	.75	1.00
LAND USE	1.00	1.00	1.00	1.00	.60	.60	1.00	.34	.50	1.00
WATER RESOURCES	.56	.64	.87	1.00	.96	.96	1.00	.78	.85	1.00
GEOLOGY	.86	.86	.86	1.00	.35	.85	1.00	.85	.90	1.00
TOTAL(AVG)	.70	.72	.84	1.00	.11	.77	1.00	.72	.83	1.00

7T-210

applications are satisfied. Finally, Table 7-8 identified the major program objectives for EOS and estimates of their relative importance. A program concept is evaluated for each objective and the weighted sum of scores is multiplied by the expected number of equivalent scenes, giving the overall FOM.

The system effectiveness model, which is explained more fully in Appendix D of this report, combines the design parameters for each alternate with the value to the users (as measured by the weighting factors) of the products produced by this particular set of parameters (e.g., particular complement of instruments and particular swath width), and determines the value of the FOM which should be applied to the alternate. The resulting data for design alternates considered in this study and an application of this data to the identification of the design to be preferred is given in the following evaluation subsection.

7.3 PROGRAM EVALUATION

7.3.1 CONCLUSIONS

Program design options have been evaluated in terms of the cost/performance (effectiveness) of design options versus the resulting FOM (expected equivalent scenes) for a EOS-A and -A' operational mission of two observatories, each with a two-year mission with one year of overlap. The results of this evaluation are shown in Fig. 7-1, which plots the total EOS-A and -A' mission observatory recurring plus operational cost per equivalent scene (cost effectiveness) versus the total number of equivalent scenes (program effectiveness) produced during the operational missions. In examining this curve, the following conclusions are apparent:

- The recommended EOS-A and -A' program with a TM/MSS, 30-m resolution, and TDRS is a cost/performance effective approach within the constraints of using a conventional launch vehicle and the baselined TM 185-km swath width

Table 7-8 Major EOS Program Objectives

EOS PROGRAM OBJECTIVES	WEIGHTING FACTORS
1. ACCOMMODATE TM AND HRPI TOGETHER	0.1
2. ACCOMMODATE FOLLOW-ON INSTRUMENTS	0.025
3. REDUCED COSTS USING SHUTTLE	0.1
4. SHUTTLE REVISIT COMPATIBILITY	0.1
5. MODULAR DESIGN	0.1
6. RESUPPLYABLE	0.1
7. ACCOMMODATE FOLLOW-ON INSTRUMENTS	0.025
8. COMPATIBLE WITH EXPENDABLE BOOSTERS	0.08
9. MEET OPERATIONAL REQUIREMENTS D.O.I.	0.1
10. MAINTAIN CAPABILITY THROUGH 1989	0.05
11. LONG-LIFE SATELLITE	0.05
12. QUICK SYSTEM THROUGHPUT TIME	0.1
13. GOAL-NUMBER OF TM SCENES THROUGHPUT	0.08
14. INITIAL LAUNCH YEAR	0.07

7T-211

- TDRS has a significant positive effect on program cost and performance effectives (Options 1 to 6 vs 6 to 13)
- The inclusion of provisions for Shuttle compatibility in the EOS design will permit a significant increase in performance at a very small cost increase when the Shuttle becomes operational (Option 1 vs 4 or 8 vs 7)
- The TM with its 30-m resolution has a significant positive impact on program cost and performance effectiveness (Option 5 vs 6)
- Increase of the TM swath width to 330 km should be further studied since it produces a significant increase in effectiveness (Option 5 vs 4)

Note that the on-orbit resupply cost and performance effectiveness is not truly represented in this evaluation because its benefit is not realized for missions of less than 2.75 yr as described in our Shuttle utilization studies.

7.3.2 METHODOLOGY

The program design approach cost and performance effectiveness curve has been generated

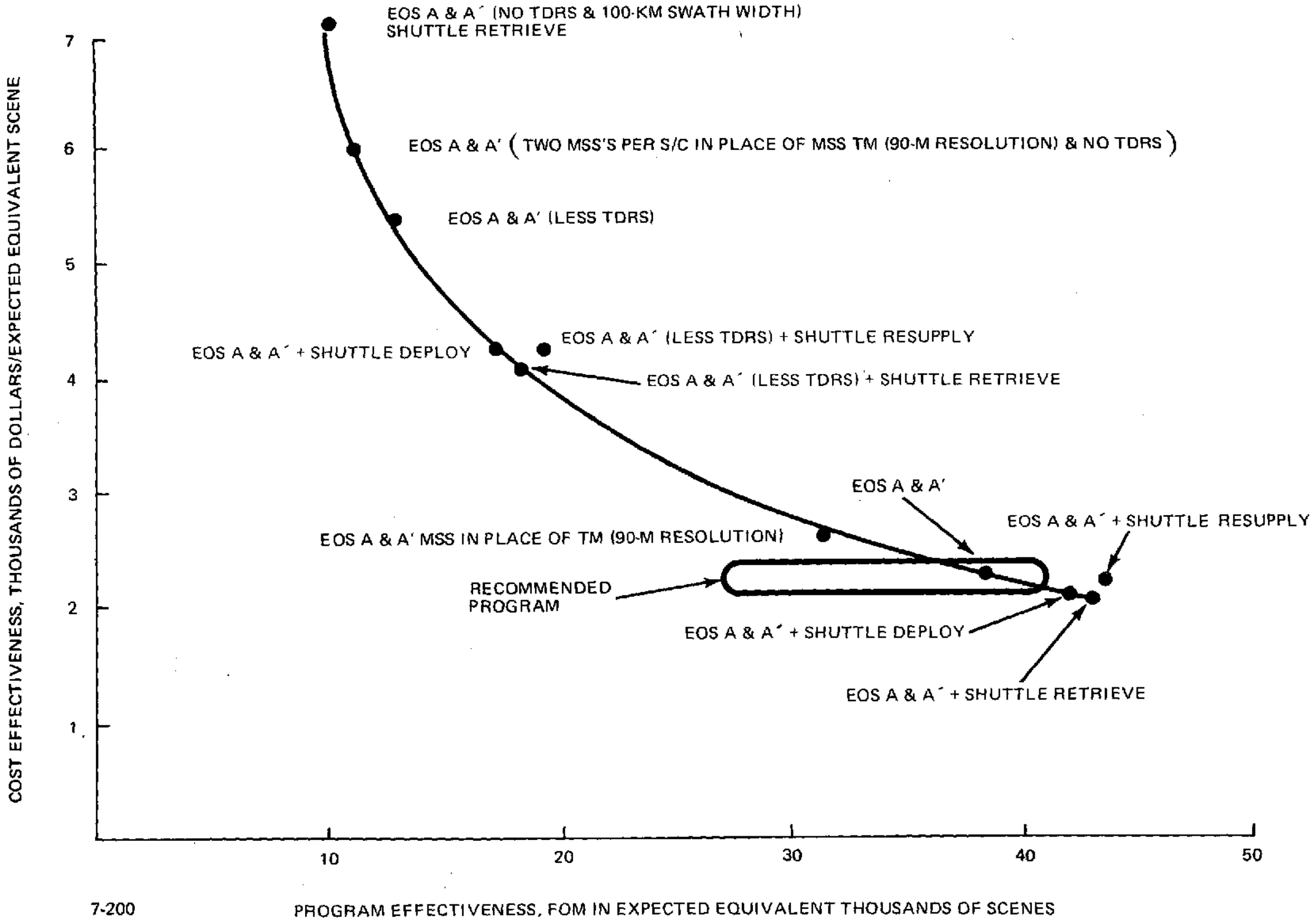


Fig. 7-1 Design Option Cost Effectiveness vs Program Effectiveness for EOS-A and -A' Observatory

Table 7-9 Design Option Figure of Merit and Costs

(1974 DOLLARS IN MILLIONS*)

DESIGN OPTION	FOM (FOUR MISSION YEARS)	*OBSERVATORY RECYCLING & OPERATIONAL COST (2 OBSERVATORIES/2 YEARS EACH
REFERENCE EOS OPTION • TM/MSS • DIRECT DATA • 185-KM SWATH WIDTH • 30-M RESOLUTION	13,204	72.520
DESIGN OPTIONS		
• TDRS	+ 25,456	14.500
• SPACE SHUTTLE DEPLOY	+ 3,924	1.086
• SPACE SHUTTLE RETRIEVE	+ 580	1.820
• SPACE SHUTTLE RESUPPLY	+ 436	2.526
• INCLUDING TM SWATH WIDTH TO 330 KM	+ 3,488	0.900
• REDVCED TM RESOLUTION TO 90 M	- 6,820	- 10.800
• REDVCED TM SWATH WIDTH TO 100 KM	- 7,128	- 0.300

by first, establishing the resulting equivalent scenes per year for each design option using the methodology described in Subsection 7.2. Next, the observatory recurring and operational cost of each design alternative, based on our EOS program design option cost (refer to Section 6), was identified. The results of these two steps are shown in Table 7-9.

Design alternatives were then synthesized to create each program design option, FOM, cost and cost/FOM (cost effectiveness). These parameters are shown in Table 7-10. The cost effectiveness versus program effectiveness (FOM) in cost per equivalent scenes versus number of equivalent scenes was plotted as represented in Fig. 7-1.

Table 7-10 Design Approach Performance Vs Cost

(1974 DOLLARS IN MILLIONS*)

	DESIGN ALTERNATIVES										
OPTION NO.	TDRS	DEPLOY	RETRIEVE	RESUPPLY	TM + MSS	90-M RES (2 MSS's)	330-KM SWATH WIDTH (TM)	100-KM SWATH WIDTH (TM)	EXPECTED EQUIVALENT SCENES FOR (2) 2-YR OBSERVATORIES WITH 1-YR OVERLAP	*OBSERVATORY SYS RECURRING COST	COST PER EQUIVALENT SCENES
1	X	X	X		X		X		46,688	90.226	1.93
2	X	X			X				42,620	88.106	2.06
3	X	X	X	X	X				43,636	92.452	2.11
4	X				X		X		42,183	87.920	2.08
5	X				X				38,695	87.020	2.24
6	X					X			31,875	76.220	2.39
7		X	X		X		X		21,232	76.326	3.59
8		X	X		X				17,744	75.426	4.25
9		X	X	X	X				18,180	77.952	4.28
10		X			X				17,164	73.606	4.28
11					X				13,240	72.520	5.47
12		X	X			X			10,924	64.626	5.91
13					X			X	10,616	75.126	7.07

7T-213

8 – PROGRAM COST COMPILATION

8.1 PROGRAM COST SUMMARY

The total estimated cost for the EOS-A and -A′ program in 1974 dollars is summarized in terms of the costs attributed to the R&D and Operational missions in Table 8-1. The $162.3 million includes $57.25 million of "fixed costs" for the launch vehicle, launch support, and instruments which were defined costs provided by NASA. The remaining $105.03 million are the "variable costs", or the costs based on our design trades and selected configurations (which were cost and performance optimized during our EOS system design definition studies).

A DTC target shown in Table 8-2 for the A- and -A′ program of $150 million is recommended as a reasonable goal. The $12.3 million delta between the identified program cost of $162.3 million and recommended target of $150 million applies only to the "variable cost" elements of the EOS-A and -A′ program since the estimates for the "fixed costs" were defined by NASA.

The EOS-A and -A′ program estimated at $162.3 million and targeted for $150 million 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
- PCC design, build and operations costs
- R&D and operational co-located ground DMS
- Network modifications
- Low-cost management approaches, including moderate simplification of test, documentation, and controls; use of a System Integration Team for project management, and a DTC approach.

Table 8-1 EOS-A and A′ Program Costs
(IN 1974 MILLIONS OF DOLLARS)

	NONRECURRING	RECURRING	TOTAL
• FIXED COSTS – INSTRUMENTS			$40.0
– TM (2)	(13.0)	(14.0)	
– MSS (2)	(1.0)	(12.0)	
– LAUNCH COSTS (2)	(0.250)	(17.0)	17.25
• OPERATIONAL SYS. COSTS			(23.47)
– MSS IMP (2)	(3.64)	(4.44)	
– GND DMS	(11.95)	(3.44)	
• R & D SYS COSTS			(32.06)
– TM IMP	(4.40)	(2.82)	
– GND DMS	(11.91)	(8.88)	
– NETWORK	(2.73)	(1.32)	
• SPACECRAFT			(39.87)
– BASIC SPACECRAFT (2)	(18.32)	(12.47)	
– M.P. SPACECRAFT (2)	(3.12)	(4.34)	
– SPARES & LOGISTICS	(0.41)	(1.21)	
• MISSION OPS	(4.73)	(4.90)	(9.63)
			TOTAL ($162.28)

3-226, 7T-12, 7T-25

Table 8-2 EOS-A and A' Program Cost Target
(IN 1974 MILLIONS OF DOLLARS)

	NON-RECURRING	RECURRING	TOTAL
• FIXED COSTS			40.0
– TM(2)	(13.0)	(14.0)	
– MSS (2)	(1.0)	(12.0)	
– LAUNCH COSTS (2)	(0.25)	(17.0)	17.25
• OPERATIONAL SYS COSTS			20.40
– MSS IMP (2)	(3.0)	(4.1)	
– GND DMS	(10.2)	(3.1)	
• R&D SYS COSTS			28.13
– TM IMP	(3.80)	(2.40)	
– GND DMS	(10.0)	(8.10)	
– NETWORK	(2.73)	(1.10)	
• SPACECRAFT			36.12
– BASIC SPACECRAFT (2)	(17.5)	(11.0)	
– M.P. SPACECRAFT (2)	(2.65)	(3.35)	
– SPARES & LOGISTICS	(0.41)	(1.21)	
• MISSION OPS	(4.20)	(3.90)	8.10
			TOTAL $150.00M

7T-27

Table 8-3 presents a representative distribution of costs incurred versus fiscal year in 1974 dollars, assuming a program start of mid-calendar '76; launch of EOS-A in April 1979, and EOS-A' in April 1980; and two years of on-orbit operations for each observatory.

8.2 DETAIL COSTS

The detail cost breakdown for the Basic Spacecraft EOS-A and -A' program and design cost options studies are presented in 1974 dollars in Table 8-4. The assumptions used in the definition of each program option are identified under each program and configuration option. The cost estimates and collection for each WBS element are based on the recommended program plans (Section 9) and the following ground rules:

- Structure Costs - Include the design, manufacturing, tooling, and wiring of the basic structure
- Module Costs - Include all engineering, manufacturing, tooling, test, quality control, and hardware procurement costs for the modules
- System Engineering and Integration - Includes the system analysis, systems integration, and instrument accommodations
- Integration and Test - Includes the engineering, manufacturing, and quality assurance for all activities required to integrate the modules into the Basic Spacecraft, integrate the mission peculiars, perform functional acceptance tests, and perform 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 nonrecurring costs)

Table 8-3 EOS-A and A' Program Funding Summary
(IN 1974 MILLIONS OF DOLLARS)

	FY'77	FY'78	FY'79	FY'80	FY'81	FY'82	TOTAL
DATA MGT SYSTEM	$ 6.3	$14.9	$ 8.5	$ 4.5	$ 3.9	$2.1	$40.3
INSTRUMENTS	6.9	18.3	13.6	1.2	–	–	40.0
FLIGHT OPERATIONS	0.3	1.0	4.3	1.9	1.2	.9	9.6
LAUNCH SYSTEM	0.1	1.9	10.6	4.7			17.3
SPACECRAFT PROJECT	10.3	17.8	19.7	6.7	.4	.2	55.08
TOTAL PROGRAM	$23.9	$53.9	$56.7	$19.0	$ 5.5	$3.2	$162.28

3-227, 7T-13, 7T-26

8-3

FOLDOUT FRAME 1

ORIGINAL PAGE IS
OF POOR QUALITY

8-3-a

FOLDOUT FRAME 2

Table 8-4 Detail Program Costs

8-3-b

8-3-c

8-3

- 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
- TM and MSS Instrument Data Handling - Cost of spacecraft instrument mission peculiars 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 AF range support costs.

The EOS-A and -A' program costs include DMS options A, B and C; TDRS; spacecraft autonomy; and low-cost management cost savings. All other options represent cost impacts on the Basic Spacecraft program with recurring costs for one spacecraft.

9 – FOLLOW-ON MISSION ACCOMMODATION

The EOS Basic Spacecraft has a broad enough capability to accommodate the instruments and their mission-peculiars for the SMM, SEASAT, SEOS, EOS-C, TIROS-O, and EGRET missions with no major changes to the Basic Spacecraft design. However, because of the diversity of orbit, on-orbit attitude, pointing requirements, slewing requirements, and instrument complements included in these missions, some changes must be expected in even a flexible spacecraft design. These changes are summarized in Table 9-1. Note that for all missions the basic subsystem configuration remains essentially intact.

The impact areas indicated in the table 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.

9.1 SEA SATELLITE (SEASAT)

SEASAT is a low-altitude, non-sun-synchronous, 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 retroreflector 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.

The SEASAT instrument characteristics are summarized in Fig. 9-1 which also shows the satellite installation in isometric form.

The basic EOS system functional integration diagram (incorporated in Report No. 3 as Fig. 3.1-2) can be modified to accommodate each mission. Figures 9-2 and 9-3 are presented to illustrate a modification specifically for the SEASAT mission. These illustrations subdivide the integrations diagram into spacecraft and ground-based portions.

9.2 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 1979). Its general mission objective is to make solar observations in all areas of the spectrum from visible to Gamma rays, and obtain data to supplement that 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.

The instrument payload of SMM is made up of X-ray and UV spectrometers, spectroheliographs (images), spectrographs, and a coronagraph as summarized in Fig. 9-4, which also shows the satellite installation in isometric form.

Closely integrated with the payload structure is a precision sun sensor which can furnish offset pointing guidance error signals within the photosphere of the sun to an accuracy of about 2 arc seconds. This is a small device with very little power consumption, but built to achieve optimum alignment and electronic stability.

9.3 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 United States. The SEOS will serve applications in the fields of:

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

Specific Application Areas - Some of the applications which the SEOS may serve are:

- o Earth Resources:
  - Detection and monitoring of water-suspended solid pollutant
  - Estarine dynamics and pollutant dispersal
  - Monitoring extent, distribution, and change of snow cover
  - Detecting and monitoring of fish location and movement
  - 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
- o 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
- o Warnings and Alerts:
  - Floods
  - Storms
  - Frosts and freezes
  - Fog.

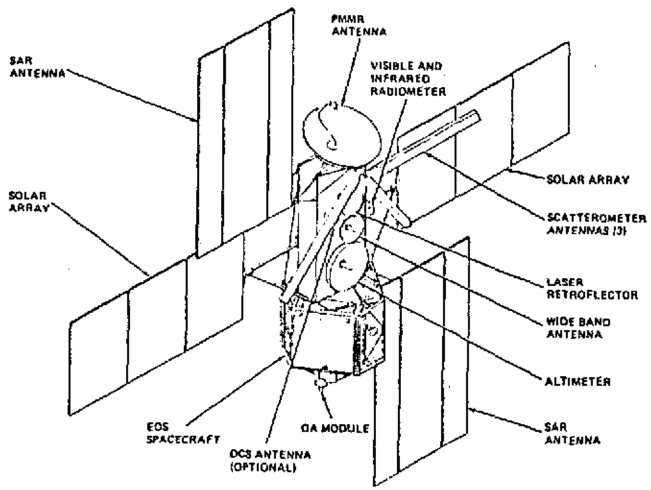
The prime instrument payload for the SEOS will be a multispectral Ritchey-Chrietien Cassegrainian telescope of approximately 1.5-m aperature. This telescope which is called the Large Earth Survey Telescope (LEST), will be used in conjunction with one or more of the following:

- o Advanced Atmospheric Sounder and Imaging Radiometer (AASIR)
- o Microwave sounder
- o Data collection system (DCS)
- o Framing camera.

Figure 9-5 summarizes the payload instrument characteristics, and shows an isometric view of the spacecraft installation.

9.4 EOS-C

EOS-C is a marine resource, hydrology and pollution-monitoring satellite devoted to the application of advanced electro-optical and microwave techniques for remote sensing of the foregoing resources, primarily those of CONUS, but possibly including other areas of recognized world economic importance. It makes use of optical and infrared mapping systems of refinement beyond the presently available capability



SEASAT FOLLOW-ON MISSION CHARACTERISTICS

Mission: SEASAT A
Orbit: 800 Km 108 inclination non-sun-sync, retrograde
Est. Spacecraft Wt: 2809

Initial Launch Date: 1978
Initial Booster: DELTA
Shuttle Utilization Mode: Retrieval/Resupply

INSTRUMENT	QTY	TOTAL WT, LB	POWER, W	VOL "LxWxH" IN. (M³)	APERTURE, IN. (M)	BANDS, GHz (μM)	RESOLUTION, M (DEG)	FOV, KM (DEG)	DATA RATE, MBPS	POINT ACCUR DEG (KNOWLEDGE)	POINT STAB. DEG/SEC	SPECIAL REQMTS	POSSIBLE SUPPLIERS	EST COST, $ 1974 REC	EST COST, $ 1974 NON-REC	REMARKS
1. ALTIMETER	1	100	125	(0.02)	40 (1)	BETW 17 AND 14	.1 TO .5 IN ALTITUDE 1600 TO 12,000 FOOTPRINT	1 AT NADIR	7	0.5	0.2		G.E. UTICA RCA WESTING-HOUSE HUGHES	TOTAL = 14M		BASED ON AAFE 2.5 KW TRANSMITTER
2. VIRR	1	22	8	19 x 7 x 9 (0.019)	5 x 17	(.6-.7) (10.5-125)	2200 TO 4200	EARTH 1/3 x 90 SPACE 1/3 x 20	12	0.5 (.1 TO .2) YAW 0.07	0.3		TI BARNES BLOCK HUGHES HONEYWELL PERKIN-ELMER			FROM GSFC PHASE A ITOS – RELATED, THERM. BOLOMETER (NO LOW TEMP.) S/C PROVIDES MOMENT. COMPENS.
3. PMMR OR MWR	1	110	65	12 x 12 x 12 (0.028)	50 OFF-SET PAR-ABOLA (1.25)	6.6 10.7 18 22 37	30K, 24K, 16K	35 OSCILL. CONICAL 45 OB-LIQUE FORWARD 1/2 TO 3 WIDE	4	1.0 (.1)	0.2		HUGHES AEROJET WESTING-HOUSE RCA G.E. AIL (C.H.)			MECH SCAN NIMBUS G-RELATED 17 GS OWELL T
4. SAR	1	ELECTRON 115 ANTENNA 150	180 TO 300	20 x 20 x 16 (0.108)	TWO 178 x 100 (4.5 x 2.5) PANELS	1.4 OR 1.7	25 TO 100	AZIMUTH 1 TO 2 TRANSV. DIRECT RANGE 6.5 WIDE 17 TO 20 FROM NADIR	1K TO 25K DEPENDS ON PRE-PROCES-SING ON BOARD	0.01 YAW 0.2 ROLL	0.2		WESTING-HOUSE GOODYEAR BALL PROS (ANTENNA) RAYTHEON MOTOROLA RCA			CCD PROCESSOR ON BOARD IS SMALL AND REDUCES DATA OUTPUT RATE
5. SCATTER-OMETER	1	ELECTRON 100 ANTENNA 80	107	16 x 16 x 15 (0.062)	FIVE STICKS 2.7 x 0.16 x 0.06)	14.5	25K TO 100K	FOUR FANS 45, 135, 225, 315 ONE FAN AT 90 0.5 WIDE AND 32 FAN FROM 25 OUT FROM NADIR	0.24	1 (.2)	0.7		RCA WESTING-HOUSE AEROJET RAYTHEON MOTOROLA HUGHES G.E.			FAN BEAMS NO SCANNING
6. LASER RETRORE-FLECTOR	1	20	–		20	LASER	–	ACCEPTS 60 OR 120 CONE					PERKIN-ELMER			PASSIVE REFLECTOR FOR LASER TRACKER

Fig. 9-1 SEASAT Follow-On Mission Characteristics

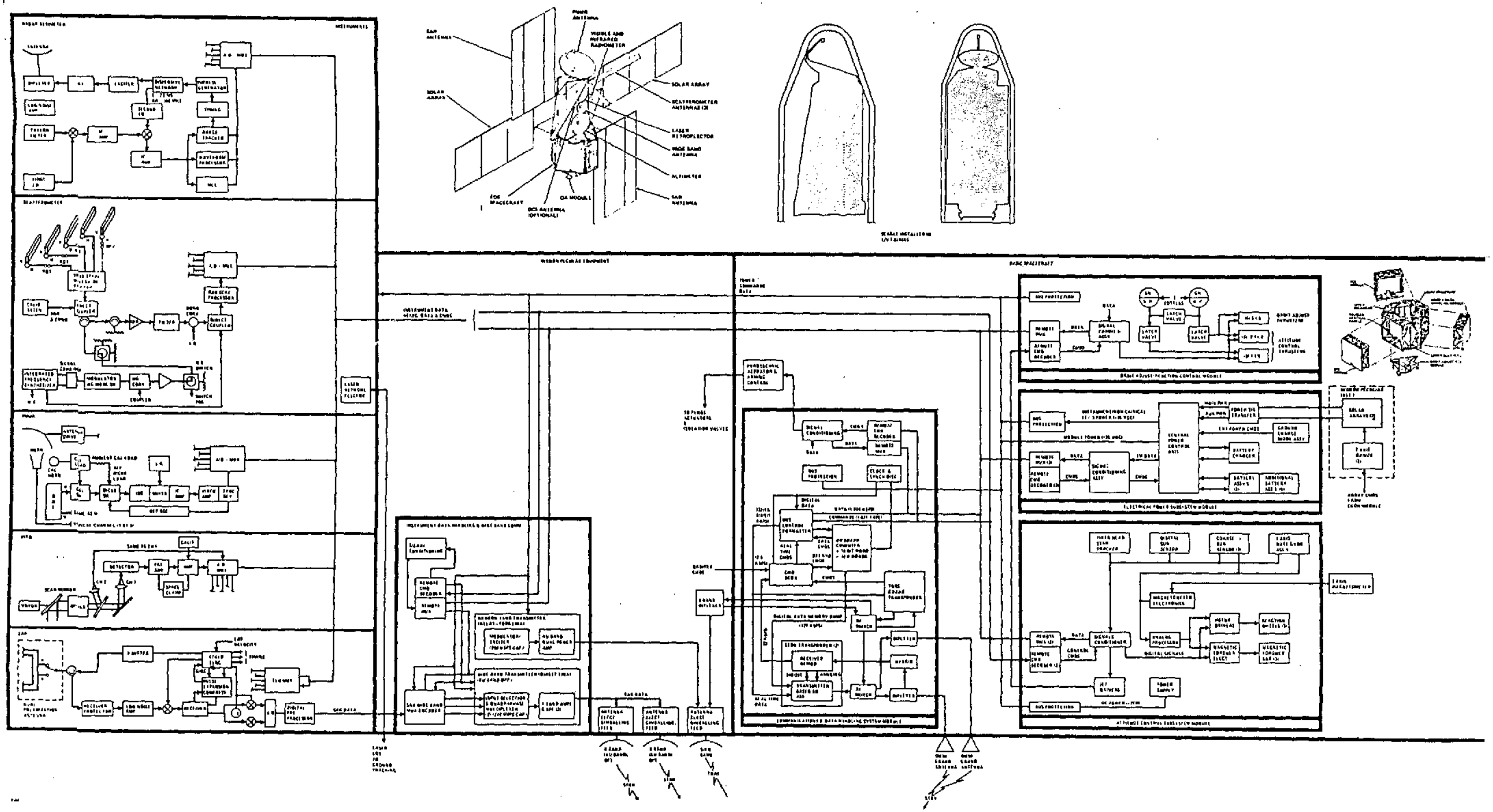


Fig. 9-2 SEASAT Observatory System Diagram

OUT FRAME

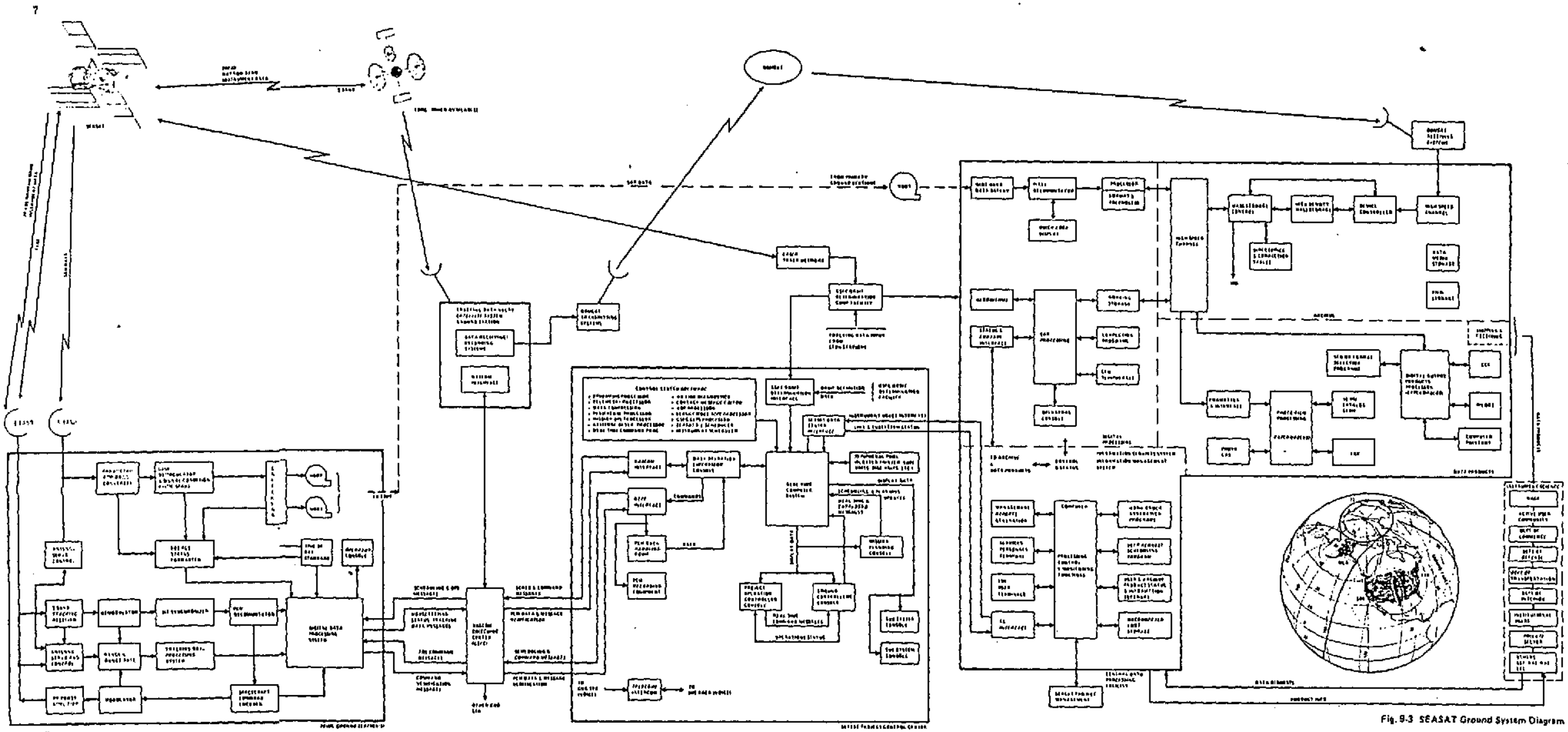


Fig. 9-3 SEASAT Ground System Diagram

9-5

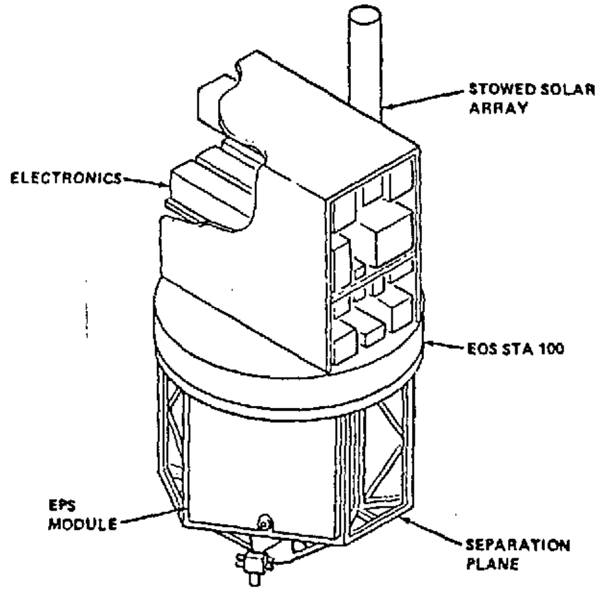
(2) 5-30, 7-43

FOLDOUT FRAME /

ORIGINAL PAGE IS OF P...

9-5-a

9-5-b



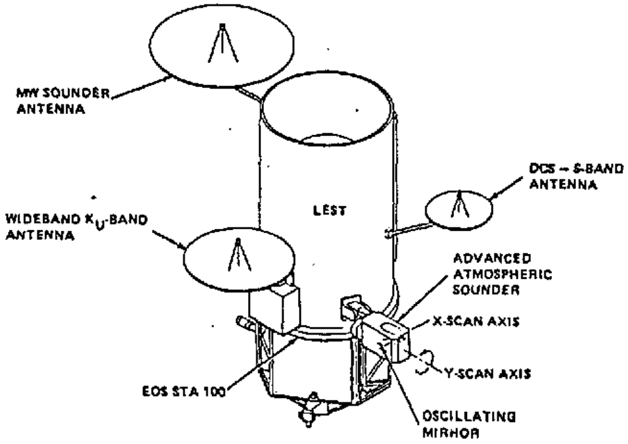
SMM FOLLOW–ON MISSION CHARACTERISTICS

Mission: SMM
Orbit: 275–300, Circular, 28–33° Incl.
Est. Spacecraft Wt.: 3538

Initial Launch Date: 1978
Initial Booster: DELTA 2910
Shuttle Utilization Mode: Retrieval–Redeployment

INSTRUMENT	QTY	SENSOR WT. LB + (ELECT. WT LB)	POWER, W	SENSOR VOL "LxWxH" IN (VOL, FT³)	APERTURE, IN. (m)	BANDS Å (KEV)	RESOLUTION, SEC	SENSOR UNOB-STRUCTED VIEW ANGLE (DEG)	DATA RATE KBPS	POINT ACCUR SEC	POINT STAB (10⁻³ DEG SEC)	SPECIAL REQMTS	POSSIBLE SUPPLIERS	ESTIMATED COST 1974 $K TOTAL	ESTIMATED COST 1974 $K NON-REC	REMARKS
1. UV MAGNETO-GRAPH	1	100 (35)	20	7 x 10 x 72 (1)		1100 - 2200	2-15	2	0.5	1	0.56			2000		
2. EUV SPECTRO-METER	1	100 (35)	20	10 x 10 x 72 (1)		20-700	2	2	1.0	1	1.12			3000		
3. HIGH-RESOLUTION X-RAY SPECTRO-METER	1	100 (35)	15	7 x 10 x 78 (1)		1-25	5-20	5	0.35	5	14			2000		
4. HARD X-RAY IMAGING	1	100 (35)	15	6 x 5 x 78 (1)		(10-60)	4	5	0.2	7	0.04			1300		
5. LOW & MEDIUM ENERGY X-RAY POLARIMETER	1	16 (20)	10	8 x 8 x 30 (0.5)		LOW 1-20 MEDIUM (10-50)	10-30 –	5	0.4	6	0.28 –			2500		
6. GAMMA RAY DETECTOR	1	200 (35)	12	(1)		(300-10,000)	–	20	0.5		–			2300		
7. HARD X-RAY SPECTROMETER	1	70 (35)	12	(1)		(10-500)	–	20	0.5		–			1700		NEEDS COOLING
8. SOLID STATE X-RAY DETECTOR	1	20 (20)	5	12 x 12 x 12 (0.5)		1-20	–	10	0.2		–			500		NEEDS COLD PLATE
9. CORONAGRAPH	1	100 (20)	10	5 x 12 x 72 (0.5)		3000 - 8000	–	20	0.5		–					
10. UV SPECTROMETER	1	110 (35)	20	8 x 12 x 72 (1)		500 1500	2	2	0.5	1	0.06			2000		
11. NEUTRON DETECTOR	1	205 (35)	15	10 x 20 x 36 (1)		(10⁴ - 10⁶)	–	20	0.2		–			2700		
12. Hα PHOTOMETER	1	20 (20)	10	4 x 4 x 36 (0.5)		6563	2	2	0.125	1	5.6			300		
13. FLARE FINDER	1	30 (20)	10	4 x 4 x 72 (0.5)		1-15	10	2	0.05	5	28			1000		
TOTAL		1171 + (380) 1551	174	(10.5)					4.98							

Fig. 9-4 SSM Follow-On Mission Characteristics



SEOS FOLLOW–ON MISSION CHARACTERISTICS

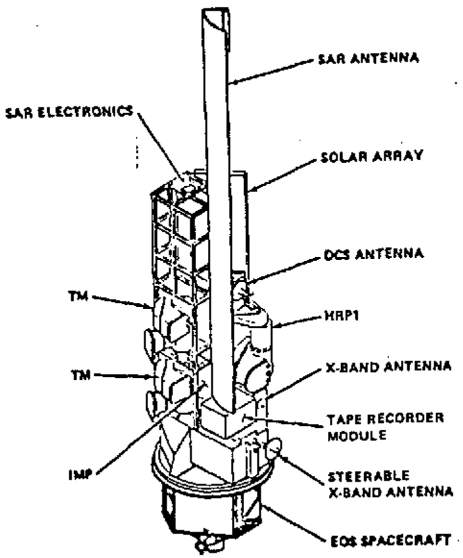
Mission: SEOS/EOS-F
Orbit: 19000 nm/0° – Geosynch
Est. Spacecraft Wt.: 4360

Initial Launch Date: 1981
Initial Booster: TITAN III-C7
Shuttle Utilization Mode: Deploy/Retrieve

INSTRUMENT	QTY	TOTAL WT LBS	POWER, W	VOL "LxWxH" IN. (m)	APERTURE IN. (m)	BANDS	RESOLUTION m (DEG)	SWATH km (DEG)	DATA RATE Mbps	POINT ACCUR DEG	POINT STAB. DEG/SEC x 10⁻⁴	SPECIAL REQMTS	POSSIBLE SUPPLIERS	ESTIMATED COST $ 1974 REC	ESTIMATED COST $ 1974 NON-REC	REMARKS
1. LARGE EARTH SURVEY TELESCOPE	1	2300	400	(1.5)	(1.6)	VISIBLE IR	100 800	3	50	0.0016	0.48	PASSIVE COOLER	ITEK, PERKIN ELMER, HUGHES			MUST HOLD 0.85 SEC FOR 2 SEC AND 2 SEC FOR 30 MIN.
2. MW SOUNDER	1				(2)	50GHz 180GHz	200K 50K		1							
3. FRAMING TV CAMERA	1					0.4...0.9 nm	316 45	1K x 1K 200 x 200	6							

7-75A

Fig. 9-5 SEOS Follow-On Mission Characteristics



EOS-C FOLLOW-ON MISSION CHARACTERISTICS

Mission: EOS-C
Orbit: 366 – 385 N MI – Sun Synch
Est. Spacecraft Wt.: 4743

Initial Launch Date: 1980
Initial Booster: TITAN III-B
Shuttle Utilization Mode: Resupply

INSTRUMENT	QTY	TOTAL WT, LB	POWER, W	VOL "LxWxH" IN. (FT)³	APERTURE, IN, (FT)²	BANDS μM	RESOLUTION M (DEG)	FOV KM (DEG)	DATA RATE MBPS	POINT ACCUR DEG	POINT STAB. DEG/ SECx10⁻⁶	SPECIAL REQMTS	POSSIBLE SUPPLIERS	ESTIMATED COST $ 1974 REC	NON-REC	REMARKS
1. THEMATIC MAPPER	2	800	100	47 x 40 x 42	16	1) 0.45-0.52 2) 0.52-0.60 3) 0.63-0.69 4) 0.8-0.95 5) 1.5-1.8 6) 10.4-12.6	30	185	230	0.01	10	PASSIVE COOLER	HUGHES, TI, HONEYWELL	$13 M	$14 M	
2. HIGH RESOLUTION POINTABLE IMAGER	1	350	70	52 x 42 x 39.5	16	1) 0.45-0.52 2) 0.52-0.60 3) 0.63-0.69 4) 0.8-0.95	10	40	90	0.01	10	±30° OFF-SET POINTING CAPABILITY	HUGHES, TI, HONEYWELL WESTINGHOUSE	$5 M	$10 M	ASSUMED TO BE PUSHBROOM SCANNER FOR THIS TIME FRAME
3. SYNTHETIC APERTURE RADAR	1	500	200	(12) + ANTENNA ANTENNA (50)-(90)	80	L-1.7GHz X-9.5GHz	30	50-75	90	0.01	10	1.2 KW PEAK POWER	GOODYEAR, WESTINGHOUSE	$5 M	$10 M	

Fig. 9-6 EOS-C Follow-On Mission Characteristics

in respect to ground resolution and radiometric fidelity. The high-resolution mapping radar adds a capability that is relatively free of atmospheric effects compared to the electro-optical, and enables dependable monitoring of resource phenomena even through clouds. For example, in the area of hydrology, which is of increasing economic importance, it becomes possible to bring all these techniques to bear upon a very continuous monitoring of water supplies, run-off, and down to the ultimate agricultural resource – soil moisture.

Orbital considerations are the same as those for EOS-A. The wider coverage which can be obtained from two TM's enchances the repeat cycle capability.

The electro-optical instrument payload consists of two TM's and one HRPI. The microwave instrument is the synthetic aperture radar. The performance goals for the TM represent an adjustment toward cost-effectiveness in the mission. The HRPI will supplement these data by a more intense coverage of certain areas of interest at higher resolution (e.g., 10 m), probably combined with lateral offset pointing. It is assumed that the EOS-C HRPI will be a "pushbroom" type scanner incorporating the best of the self-scanned array (either digital or CCD) technology available. This technology should be much improved in the next year or so, and make it possible to utilize what appears to be a most compact and efficient instrument.

The synthetic aperture radar (SAR) will be a relatively compact instrument electronically, but it will require a rather large antenna. The handling-deployment bulk and "figure" requirements for this antenna are severe, but optimism for the time frame has been engendered by new phased array technology which should make the antenna practical in these respects. It is anticipated that the use of L-Band will be increasingly emphasized, but it is probable that this SAR may be a combined X- and L-Band radar. It is also quite possible that the logistic burden of ground processing of SAR data will be lightened by the application of CCD processing in the required time frames.

Figure 9-6 summarizes the payload instrument characteristics, and shows the satellite installation in isometric form.

9.5 TIROS O

The TIROS O mission is intended to verify for operational use an advanced instrument payload. The payload will include remote sensing techniques from Nimbus and previous EOS follow-on flight experiments, as well as improved versions of those sensors carried by the previous N/ITOS vehicles. The TIROS O satellite may well be the first operational vehicle to be designed with the shuttle-exploitative modular design, so that in-orbit refurbishment of the payload can be effected and evaluated. The nominal orbital parameters are 905-n mi perigee, 915-n mi apogee, and inclination of 103 deg.

The payload instrument characteristics are summarized in Fig. 9-7, which also shows the spacecraft installation in isometric form.

9.6 EXPLORER GAMMA RAY EXPERIMENT TELESCOPE (EGRET)

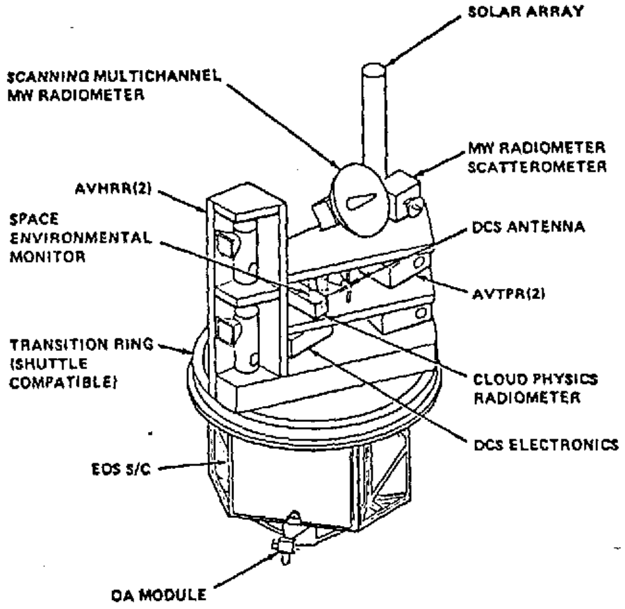
MISSION – The EGRET is a satellite that is compatible with Delta launch into a 250-n mi circular orbit with an inclination of about 28 deg, but could also be considered for launch to somewhat higher orbital altitude.

It is designed for mapping all interesting zones of the celestial sphere for high energy Gamma rays of cosmic origin. These Gamma rays may arise in the galactic interstellar gas from the decay of neutral mesons formed in the interactions of cosmic ray particles with the nuclei of the interstellar gas. Similarly, radiation of higher energy than 30 MeV, detected from non-galactic origins, and of relatively low intensity, may be measured and used for determining the consistency of existing cosmological models. Discrete, but probably weak sources above 100 MeV can be localized, and a spectral analysis of these sources used for their diagnosis. Intense short bursts of high-energy rays, such as predicted by the hydromagnetic theory of supernovae explosions, as well as bursts of lower energy rays, can be registered for analyses of great fundamental importance. Added to this is the recording of pulsations already reported from the crab nebula to a temporal accuracy suitable for diagnosis.

INSTRUMENT PAYLOAD – The experimental payload is described in an available May 1974 report by Robert Hofstadter, Co-Principal Investigator of Stanford University, and Carl E. Fichtel, Co-Principal Investigator of Goddard Space Flight Center, Associated with Hughes, Schilling, Crawford of Stanford; Kniffen, Hartman and Thompson of GSFC; M.K. Sommer of Max-Planck Institut Für Extraterrische Physik; and Favale and Schneid of Grumman. In the form described in this report, it is represented by a 2650 lb weight within an envelope approximately 65 in. in diameter and 88 in. in length. It consists of four mechanical assemblies as follows:

- Assembly A – Domes, including a plastic scintillator anti-coincidence dome; a 0.1-in. thick aluminum pressure vessel; and a thin optical light shield
- Assembly B – The Spark Chamber Telescope incorporating a stack of spark modules and tantalum pair production plates
- Assembly C – The Total Absorption Shower Counter, consisting of a large NaIT1 crystal and twelve five-in. PMTS for read-out
- Assembly D – The Bulkhead Pedestal, which is a structural interface for the experimental subassemblies.

The EGRET instrument characteristics are summarized in Fig. 9-8, which also shows the satellite installation in isometric form.



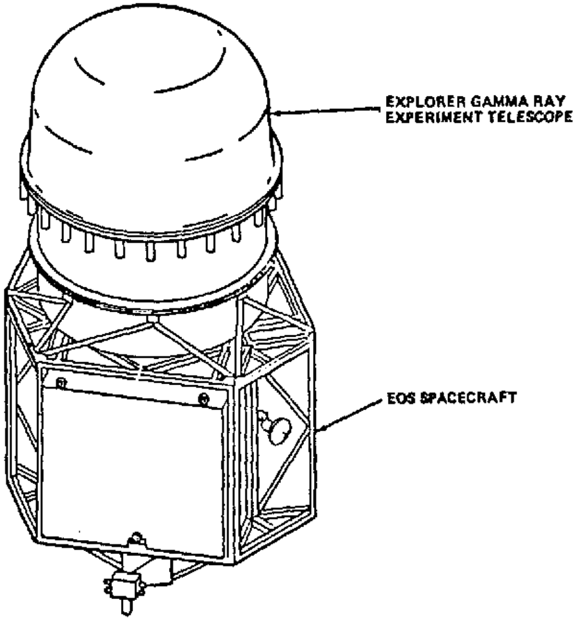
TIROS-O FOLLOW-ON MISSION CHARACTERISTICS

Mission: TIROS-O
ORBIT: 905 N. MI. - SUN SYNCH
EST. SPACECRAFT WT.: 3481

Initial Launch Date: 1982
Initial Booster: Delta 3910
Shuttle Utilization Mode: Retrieve/Resupply

INSTRUMENT	QTY	TOTAL WT LB	POWER,	VOL (FT³)	APERTURE, IN. (m)	BANDS µM	RESOLUTION M (DEG)	FOV KM (DEG)	DATA RATE MBPS	POINT ACCUR DEG.	POINT STAB. DEG SEC X 10⁻	SPECIAL REQMTS	POSSIBLE SUPPLIERS	ESTIMATED COST $ 1974 REC	NON-REC	REMARKS
1 ADVANCED VERY HIGH RESOLUTION RADIOMETER (AVHRR)	2	60	16	2.12		6-12.5 (4 BANDS)	1000	(125)	1			PASSIVE COOLER				ADVANCED VERSION OF EXISTING INST.
2 ADVANCED VERTICAL TEMPERATURE PROFILE RADIOMETER (AVTPR)	2	200	40	3.5		9-1.30 (12 BANDS)	1000	(61)	$4 \times 10^{[illegible]}$				BARNES			SAME
3 SCANNING MULTI-CHANNEL MW RADIOMETER ELECTRONICS SCANNER	1	115	85	1.0 3.0	39.36 (1)	6-53GHz (6 BANDS)	(140	(135)	$1.5 \times 10^{[illegible]}$							45° FWD LOOK ANGLE
4 MW RADIOMETER SCATTEROMETER ELECTRONICS ANTENNA	1	80 45	82	1.5	10.56		140	(60)	$1 \times 10^{[illegible]}$							30° FWD & 30° OFF + V LOOK ANGLE
5. CLOUD PHYSICS RADIOMETER	1	30	20	0.84		0.75-2.12 (5 BANDS)	140	(~75)	$3 \times 10^{[illegible]}$	0.01	2					
6 SPACE ENVIRONMENTAL MONITOR	1	25	6	0.73				(180)	1.2×10^{-4}							SUN POINTING
7 DATA COLLECTION SYSTEM	1	50	10	0.56					8×10^{-4}							

Fig. 9-7 TIROS O Follow-On Mission Characteristics



EGRET FOLLOW-ON MISSION CHARACTERISTICS

Mission: EGRET
Orbit: 250 N MI/28.5°
Est. Spacecraft Wt.: 4000

Initial Launch Date: 1979
Initial Booster: D7910
Shuttle Utilization Mode: Retrieve/Resupply

INSTRUMENT	QTY	TOTAL WT, LB	POWER, W	VOLUME "LxWxH" IN. (M)	APERTURE, IN. (M)	BANDS	RESOLUTION M (DEG)	FOV (DEG)	DATA RATE MBPS	POINT ACCUR, DEG	POINT STAB, DEG SEC×10	SPECIAL REQMTS	POSSIBLE SUPPLIERS	ESTIMATED COST $ 1974 REC	NON-REC	REMARKS
1. HIGH ENERGY GAMMA RAY TELESCOPE	1	2725	55	CYLINDER 2.20 M HEIGHT x 1.64 M DIAMETER	1.64 M	25 TO 30000 MeV	ANGULAR 1.4° @ 100 MeV 0.8° @ 200+ MeV 0.1° @ VERY HIGH ENERGIES ENERGY: • 10% 1σ FROM 70 TO 10^4 MeV • SOMEWHAT POORER FOR REST OF RANGE	±45° CONE	3×10^{-3}	0.1		• SOLAR ARRAY MUST BE PLACED SO AS TO MINIMIZE SECONDARY EMISSIONS • ACCESS MUST BE AVAILABLE TO PHOTOMULTIPLIER TUBES AND ELECTRONICS AFTER MATING TO S/C • TIMING ACCURACY ≤ 1 MSEC.	GSFC	TOTAL COST $5 M		VERY HIGH SENSITIVITY OF 1.8 x 10^7 CM^2 SR. SEC PER DAY > 312 TIMES OSO III > 26 TIMES SAS II > 12 TIMES COS -B

Fig. 9-8 EGRET Follow-On Mission Characteristics

Table 9-1 EOS Basic Spacecraft Impact for Follow-On Missions (Sheet 1 of 2)

MISSION	STRUCTURE			ATTITUDE CONTROL			COMM & DATA HANDLING			ELECTRICAL POWER			PROPULSION			THERMAL		
	IMPACT AREA	WEIGHT, LB	COST $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)
SEASAT-A LAUNCH 7/78 BOOSTER: DELTA 2910	• SYNTHETIC APERTURE RADAR A PROBLEM DUE TO VOLUME LIMITS OF DELTA SHROUD, BUT NOT DUE TO EOS SPACECRAFT. HOWEVER, THESE PROBLEMS MIGHT BE OVERCOME BY USE OF 2-DIMENSIONAL PHASED ARRAY APPROACH WHICH RELIES ON ELECTRONIC RATHER THAN MECHANICAL "FIGURE"; AND THEREFORE, TOLERATES FOLDING. • INCREASE IN ARRAY SIZE	 75	 300	CONTROL PROBLEMS POSSIBLE DUE TO ANTENNA MOTION DYNAMICS (WILL BE INVESTIGATED FURTHER)			NONE			NEW SOLAR ARRAY DRIVE ADD BATTERY	NEGLIG. 32	300 20				EPS MODULE 4VCHP & OSR C & DH MODULE 4 VCHP & OSR	30 30	50 50
SMM LAUNCH: 7/78 BOOSTER: DELTA 2910	• TRANSITION RING • REDUCE ARRAY SIZE	75 -55	30							ELIMINATE ARRAY DRIVE DUE TO SOLAR ATTITUDE REDUCE POWER REQUIREMENTS	SAVINGS IN STRUCTURE	100	NONE			EPS MODULE 4 VCHP & OSR C & DH MODULE 4 VCHD 4OSR	30 30 30	50 50 50
SEOS LAUNCH: 7/81 BOOSTER: T III C-2 (TE-364-4)	• TITAN ADAPTER • TRANSITION RING • GIMBALLED STAR TRACKER MOUNT ON MODULE • REDUCE ARRAY SIZE TO 60 FT² • ADDITIONAL N_2H_4 TANK	200 75 2 -60 +2.92 +2.92	100 30 10 NO CHANGE COST SAVING IN HARDWARE EFFECT - DESIGN INCREASE 13	GIMBALLED STAR TRACKER OPERATIONAL UPDATE OF GYROS TO MEET HOLDING REQTS	35 –	1,400 –	HIGH GAIN ANTENNA; MAY BE ELIMINATED BY DUAL FEED WITH WIDE BAND DATA OVER INSTRUMENT X-BAND ANTENNA ADDITIONAL COAXIAL SWITCH STAR TRACKER SOFTWARE & HARDWARE POSSIBLE	3.3 6	95 25	REDUCED POWER REQUIREMENTS COMPARED TO BASIC MODULE	ARRAY SAVINGS IN STRUCTURE	NONE	ADDITIONAL FUEL REQUIRED TO PERFORM WHEEL UNLOADING	11.5	NEGLIG.	EPS MODULE 4 VCHP ACS MODULE 4 VCHP C & DH MODULE 4 VCHP STRUCTURE HEATERS	25 26 25 3-8	24 24 24 23-53
EOS-C LAUNCH: 1980 BOOSTER: TITAN III-B	• ADD TRANSITION RING TO PICK-UP TITAN DIAMETER • RCS MODULE STRUCTURE	180 65	180 260							ADD BATTERY SOLAR ARRAY	40 100	20 180	ADD (4) SRM's; THREE REQD FOR DESIGN 1 FOR QUAL FIRING o ADD (3) TANKS o ADD (4) 75 # THRUSTERS ELIMINATE (4) 5 # THRUSTERS	1050 16	373.5 134			

7-66 (1)

Table 9-1 EOS Basic Spacecraft Impact for Follow-On Missions (Sheet 2 of 2)

MISSION	STRUCTURE			ATTITUDE CONTROL			COMM & DATA HANDLING			ELECTRICAL POWER			PROPULSION			THERMAL		
	IMPACT AREA	WEIGHT, LB	COST $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST, $(K)	IMPACT AREA	WEIGHT, LB	COST $(K)
TIROS-O LAUNCH 7/82 BOOSTER: T III B (SSB/NUS)	• TRANSITION RING	75	30	REQUIRES SRM CHANGE IN SOFTWARE TO FACILITATE SRM BURN	NONE	0	ADD PA MODULE	2	10	NEW SOLAR DRIVE	NEGLIG.	300	ADD 4 SRM's	1200	248	NONE		
	• ADDITION OF 2 N_2H_4 TANKS	1	6							DECREASE POWER REQUIREMENTS DUE TO DECREASED RADIATION, NET EFFECT ZERO			ADD 2 TANKS	6	26			
	• TITAN ADAPTER	200	100															
EGRET LAUNCH: 7/78 BOOSTER: D2910	• RING TRUSS FOR INSTRUMENT	50	18	LOWER ORBIT INCREASES ADDITIONAL AERODYNAMIC TORQUES, THUS INCREASING TOTAL DISTURBANCE TORQUES ON VEHICLE – REQUIRES LARGER REACTION WHEELS, PLUS LARGER MAGNETIC TORQUER BARS			NONE			DECREASE POWER REQUIREMENTS	NEGLIG.	NONE	ADDITIONAL PROPULSION FUEL (SEE ACS)	11	NEGLIG	EPS MODULE 4 VCHP & OSR	30	50
	• MODS FOR ADDITION OF ONE N_2H_4 TANK	.5	2.5										ADD 1 TANK	3	13	C&DH MODULE 4 VCHP & OSR	30	50
				3 WHEELS	30	30												
				3 BARS	120	88												

7-66 (2)

9-14

FOLDOUT FRAME /

FOLDOUT FRAME 9-14 - a

10 – SPACE SHUTTLE INTERFACES/UTILIZATION

The purpose of this phase of study has been to determine the impact and benefits associated with using the Space Shuttle for EOS delivery, retrieval, and on-orbit resupply.

EOS – Shuttle operational compatibility can be achieved in any of the three projected modes of Shuttle utilization with reasonable impact to program cost and observatory weight. Resupply is the preferred mode of Shuttle operations for EOS programs with an operational lifetime in excess of two to three years.

As shown in Table 10-1, EOS-B Shuttle compatibility can be achieved for Deliver and Retrieval with an Observatory weight penalty of about 70 lb. The Safety-of-Flight (SOF) and Flight Support System (FSS) interface requirements are virtually identical in these two modes. This penalty can be completely eliminated if the positioning platform can be eliminated (1433 lb) by using the Orbiter manipulator throughout the deployment or retrieval cycle. Introduction of on-orbit module replacement in Resupply, necessitates adding to the vehicle, latches, rollers, tracks, and signal/power connectors, resulting in a total penalty of 236 lb. These weight penalties do not compromise Shuttle capability to meet mission requirements, nor EOS installation within Orbiter payload volume and center-of-gravity envelopes.

The cost impact of EOS-B Shuttle compatibility, as shown in Table 10-2, is reasonable when compared to total program cost in each Shuttle utilization mode. Fundamental SOF items and FSS interfaces are included in the Delivery costs. The principal change for Retrieve is the addition of appendage retraction provisions while for Resupply, module replacement mechanization is the major driver. Costs of the FSS and the MEM have been identified, but are not included in cost impacts since they are considered general purpose equipment whose costs will be apportioned among all users.

The preferred mode of Shuttle utilization is dependent upon the desired observatory operating time on-station (i.e., program life). Figure 10-1 shows that for program durations of less than two to three years, there is little cost difference (less than $5 million) among utilization modes, with Deliver only slightly more economical than the other modes. As program life exceeds this threshold, however, Resupply become increasingly more beneficial, yielding a program savings at

Table 10-1 EOS-B/Shuttle Compatibility Observatory Weight Impact (EOS-B)

SUBSYSTEM	WEIGHT INCREMENT (LB) DELIVER	RETRIEVE	RESUPPLY	DRIVER
CDH	26	26	26	FAIL OPNL MONITOR/CNTL
EPS	10	10	55	MODULE CONNECTORS
ACS	0	0	0	
STRUCT MECH	(27)	(32)	(133)	VARYING INST COMPLEMENTS GIVE RANGE OF 133 LB (EOS-B) TO 187 LB (EOS-E)
– FSS	27	27	27	
– RETRACT	0	5	5	
– REPLACE	0	0	101	
THERMAL CNTL	0	0.4	18.4	THERMAL CLOSURES
PROPULSION	3.5	3.5	3.5	TANK PRESS CONTROL
TOTAL	67	72	236	MODULE REPLACEMENT

7T-28

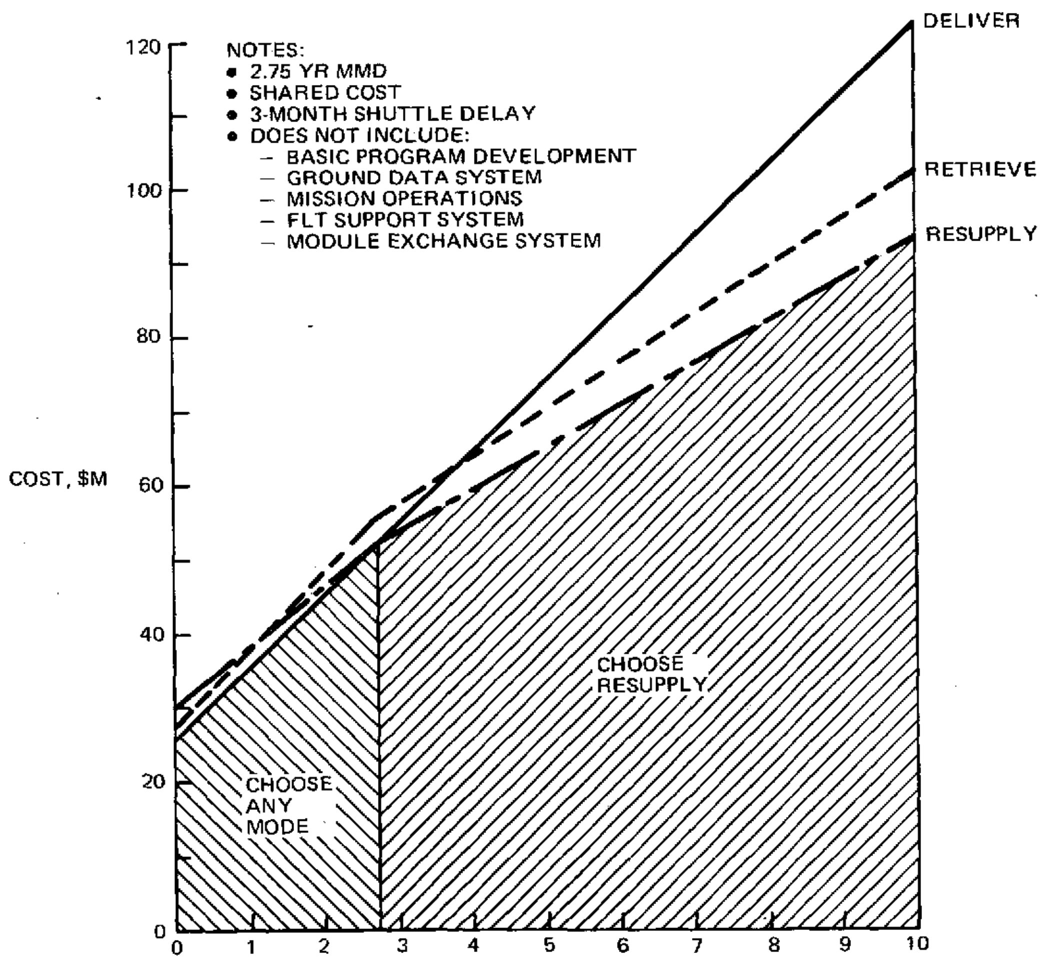
Table 10-2 EOS-B/Shuttle Compatibility Cost Impact (EOS-B)
(1974 DOLLARS IN THOUSANDS)

WBS ELEMENT	COST INCREMENT					
	DELIVER		RETRIEVE		RESUPPLY	
	NON-RECURRING	RECURRING	NON- RECURRING	RECURRING	NON-RECURRING	RECURRING
PROGRAM MGT – CONTRACTOR	23	23	111	91	185	90
SYSTEM ENGRG AND INTEGRATION	100	20	200	40	300	100
RELIABILITY AND QUALITY ASSURANCE	80	40	100	40	160	40
DEVELOPMENT TEST	0	–	50	–	516	–
GND SUPPORT EQUIP. – OBSERVATORY	44	–	44	–	44	–
STRUCTURE	62	30	150	30	521	160
POWER	34	22	34	30	76	49
SOLAR ARRAY DRIVE	0	0	186	36	231	36
COMM AND DATA HANDLING	90	373	90	373	90	373
INSTRUMENT SUPPORT STRUCT	0	0	416	235	620	380
ORBIT ADJUST SUBSYSTEM	70	35	170	35	70	35
CONTROL CENTER OPERATIONS	0	–	128	–	200	–
OBSERVATORY TOTAL	503	543	1,579	910	3,013	1,263
FLIGHT SUPPORT SYSTEM	4,856	2,042	4,856	2,042	4,900	2,100
MODULE EXCHANGE SYSTEM					10,000	2,500

7T-29

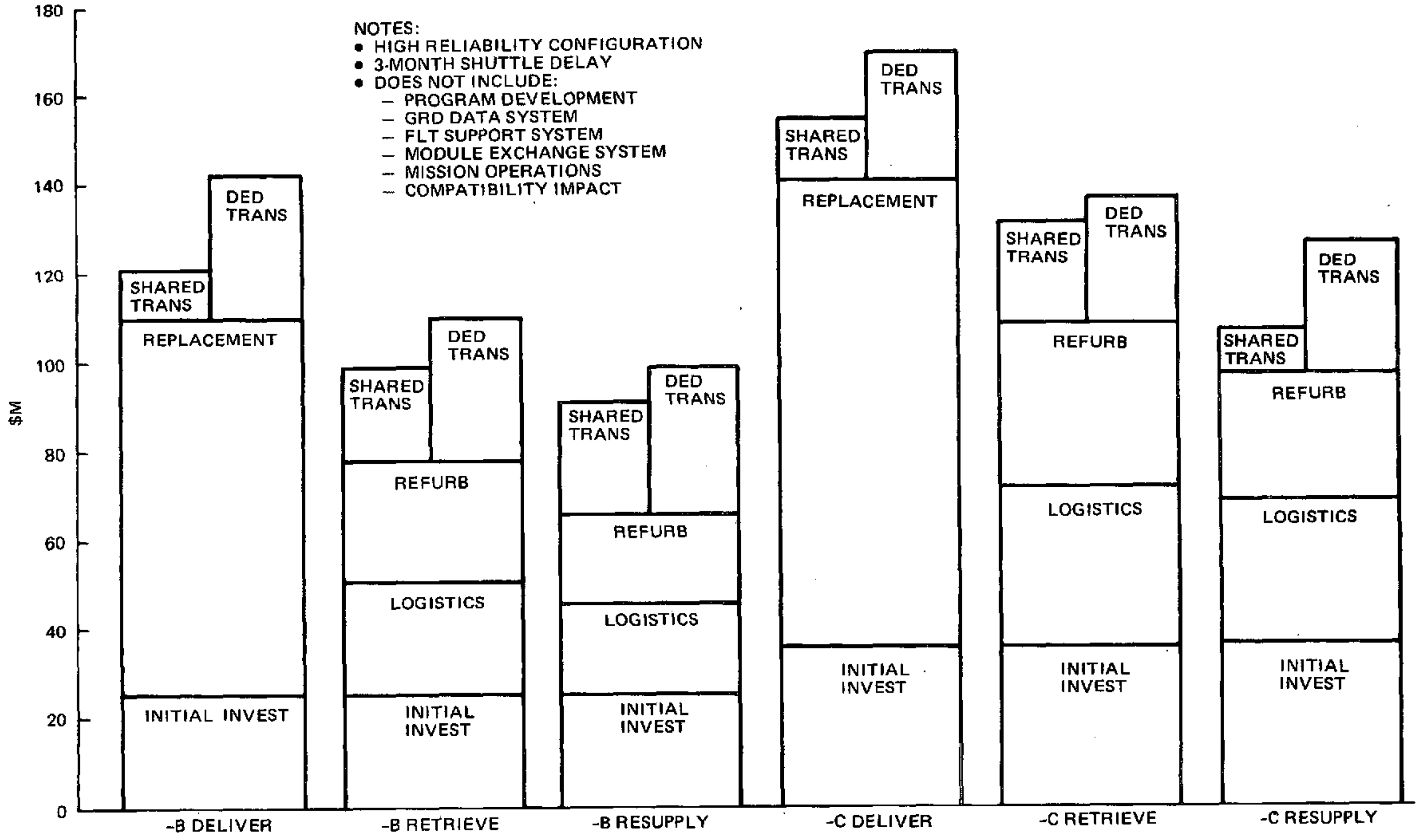
the end of 10 yr of approximately $10 million over Retrieve and $30 million over Deliver. While Resupply is the preferred mode for long duration programs, if the necessary capabilities cannot be implemented on time, the Retrieve mode should be incorporated to take advantage of its cost saving potential.

The relative merits of the three utilization modes are insensitive to the weight and cost of observatory instrument and mission peculiar equipment complements. As shown in Fig. 10-2, an observatory configured for the EOS-C mission, with a front-end complement significantly more costly than the EOS-B mission configuration, demonstrates the same relationship among modes for a 10-yr program. In any mode, Shuttle flights should be initiated only on demand, rather than on a regularly scheduled basis to take full advantage of prolonged observatory operation beyond its design life and reduce transportation and recycle costs. Similarly, high observatory redundancy to extend design life (to a maximum of approximately four years) is cost effective. If Shuttle flights can be shared with other payloads, program costs can be further reduced ($10 to 20 million) as shown in Fig. 10-2. Sharing transportation costs favors a combination of low Shuttle parking orbit (150 to 200 n mi) and EOS orbit transfer capability. The desirability of Resupply can be further enhanced by reducing the weight, and attendant EOS-chargeable transportation costs of the Resupply mechanisms.



7-1

Fig. 10-1 Shuttle Utilization Costs (EOS-B)



6-8
7-2

Fig. 10-2 Shuttle Utilization Mode Cost Comparison, 10-Year Program

11 – RECOMMENDED PROGRAM PLAN AND MANAGEMENT APPROACH

11.1 PROGRAM PLAN

The recommended program plan for EOS-A and -A' is shown in Fig. 11-1. 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
- Development and qualification of a Shuttle-compatible standard spacecraft (Basic Spacecraft) which meets the requirements of EOS-A and -A' as well as follow-on missions
- Design development and qualification completed prior to the start of the fabrication of flight hardware
- Static load qualification of the primary module and secondary structure by acceleration to include Shuttle crash-load demonstration
- Early structural qualification tests with component mass representations to define component environments prior to the start of component qualification tests
- Consolidation of all flight hardware environmental tests at the module level.

Inherent in the recommended program plan is a subplan which can be used to provide an acceptance tested Basic Spacecraft, that is independent of a particular mission. This approach is illustrated by the schedule option shown in Fig. 11-1, which provides a Basic Spacecraft which meets the requirements for a program with a 1978 launch. The cost savings realized by this approach in design, development, and test areas is illustrated in Fig. 11-2, which compares the average non-recurring costs of using our Basic Spacecraft for the EOS mission only; the EOS and SEASAT mission; and the EOS, SEASAT, and SMM missions.

11.2 MANAGEMENT APPROACH

The objectives of our recommended program management approach are to provide the management plan and controls necessary to design, develop, and integrate the EOS-A and -A' program elements within specified program cost targets, and provide a low-cost standard spacecraft which will support future earth orbiting missions. To achieve these goals it is recommended that the EOS-A and -A' program be conducted in a Design-to-Cost (DTC) environment with the specific cost targets defined in Section 8 for each program element. To manage the program implemented in accordance with the DTC approach, we recommend a System Integration Team headed by a centralized program manager which we have designated as the System Integrator.

Our EOS System Definition Studies have established the DTC targets (refer to section 8) and program requirements for major spacecraft and gronnd system elements for the EOS-A and -A' program. We have incorporated the DTC target into the EOS System Design Specifications. Each element contractor will be responsible for meeting the target set and further defining cost targets for each element of his Work Breakdown Structure (WBS). Designers will then have cost targets as design requirements and use data bands and productibility cost handbooks to select the detailed design which meets his cost and performance requirements. Where lower level WBS element cost and performance requirements cannot be met within cost targets, design cost tradeoffs of higher level requirements will be made

8-11

FOLDOUT FRAME 1

11-2

ORIGINAL PAGE IS OF POOR QUALITY

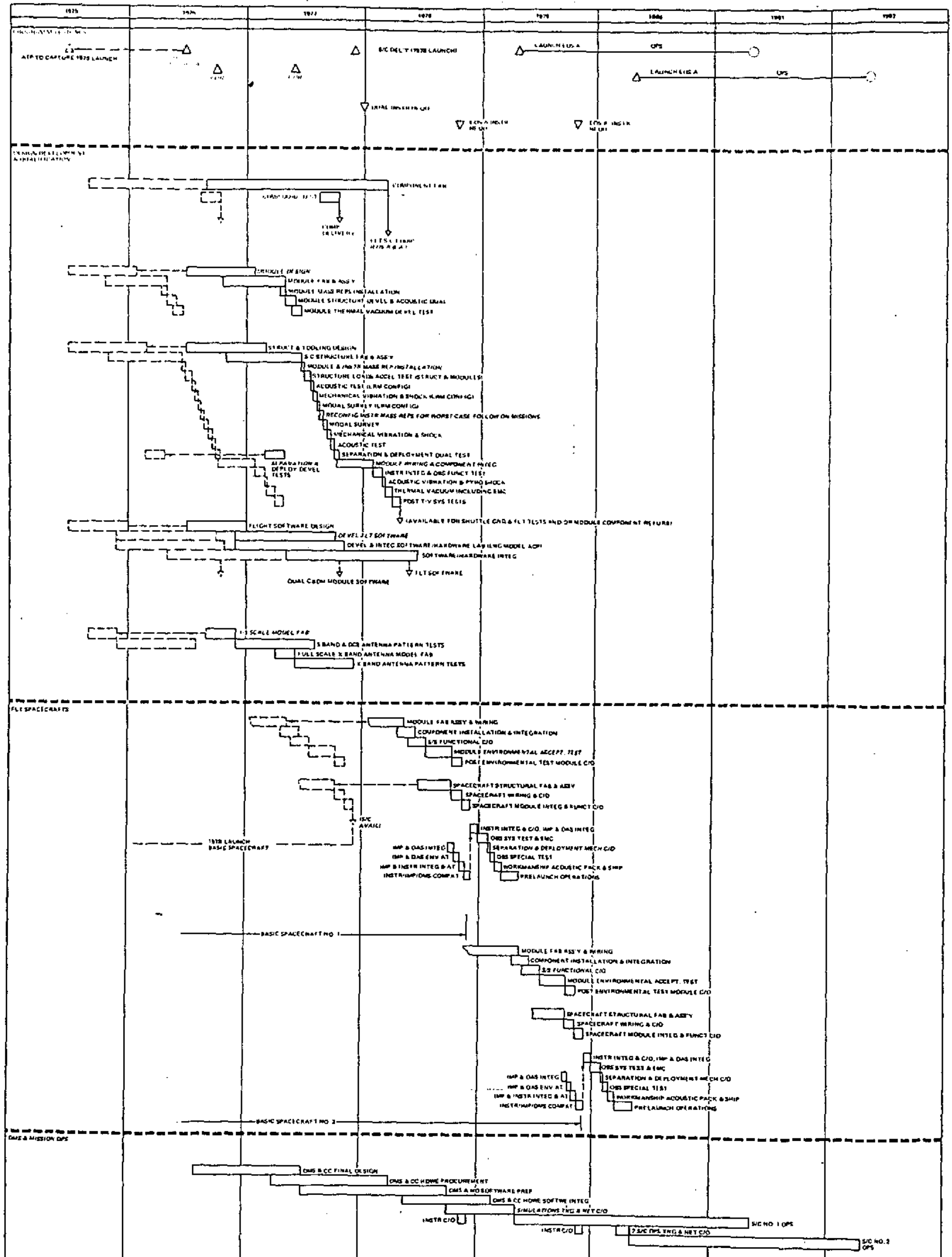
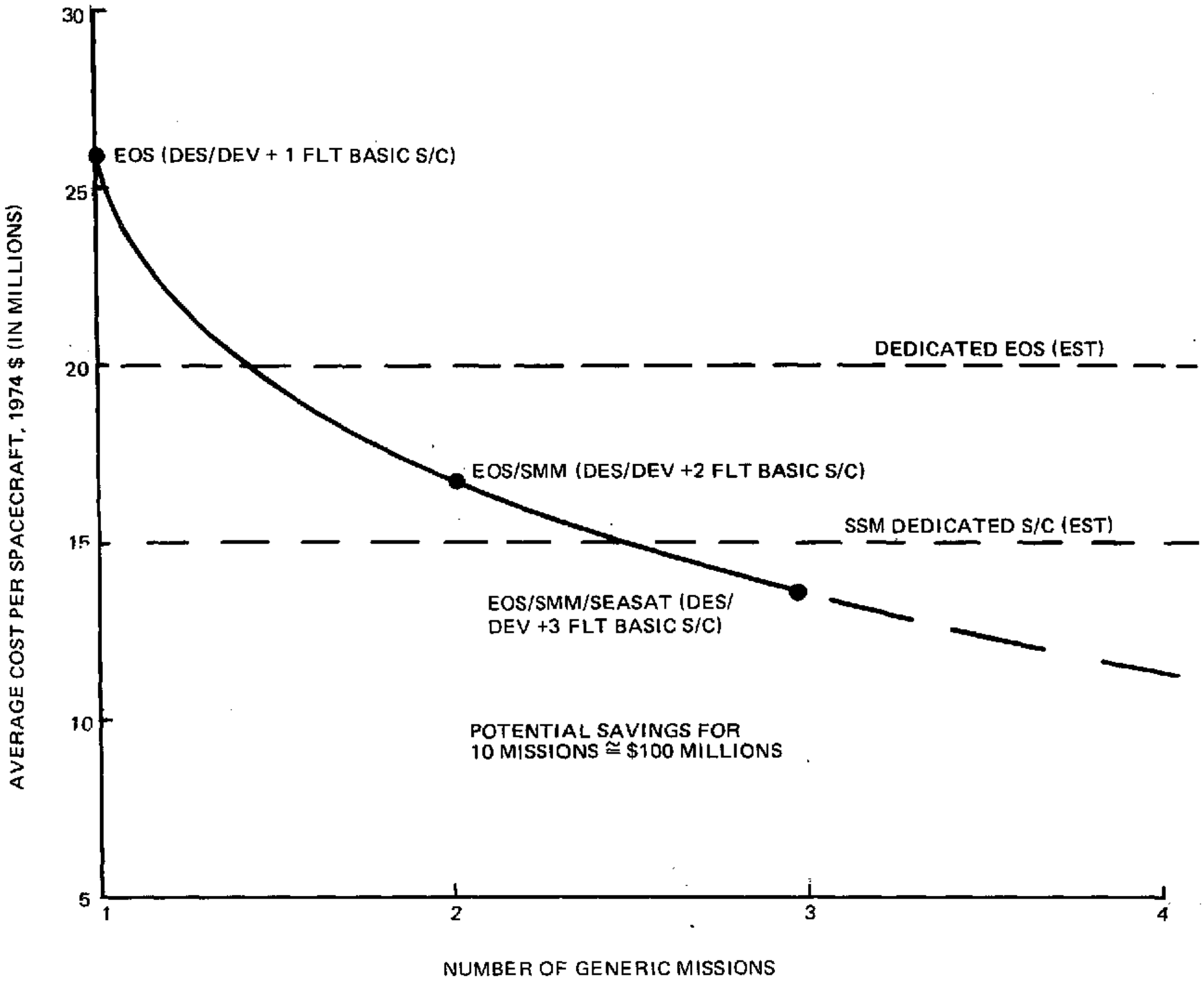


Fig. 11-1 Program Master Schedu

11-2-a

FOLDOUT FRAME



7-27

Fig. 11-2 Spacecraft Average Cost – EOS/SMM/SEASAT

by the element contractor to achieve overall element performance and cost targets.

The System Integrator shall be responsible for maintaining overall EOS-A and -A' program costs within these targets as illustrated by Fig. 4-16. The scope of the System Integrator's tasks include schedule and technical performance as well as cost, and he has the overall responsibility under the direction of the GSFC Program Manager for all elements of the program. We recommend that the System Integrator, in his total program role, function through a working team concept comprised of personnel from NASA/Goddard, user groups, GFE contractors and the instrument contractor. The direct communication provided by this team should bring overall management cost down through reduction of formal documentation, and provide the ability to identify, jointly analyze, and resolve all interface problems in real time. The System Integration Team recommended for the EOS-A and -A' and the typical distribution of team members for these as well as follow-on misiions, are shown in Fig. 11-3 and Table 11-1. Note that once the Basic Spacecraft contractor does not have to be the EOS-A and -A' Program manager, the Basic Spacecraft contractor does not have to be the System Integration contractor. Therefore, future Basic Spacecraft would be provided in the same manner as would a launch vehicle or any other program element.

Table 11-1 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
3-199, 4T-2, 7T-23

Our recommended procurement plan, in Fig. 11-4, is consistent with the System Integrator concept and a DTC program approach. Our recommended plan for the EOS-A and -A' phase is also planned to permit the introduction of multiple procurements of the Basic Spacecraft and the LCGS, and provides alternate methods for future procurement.

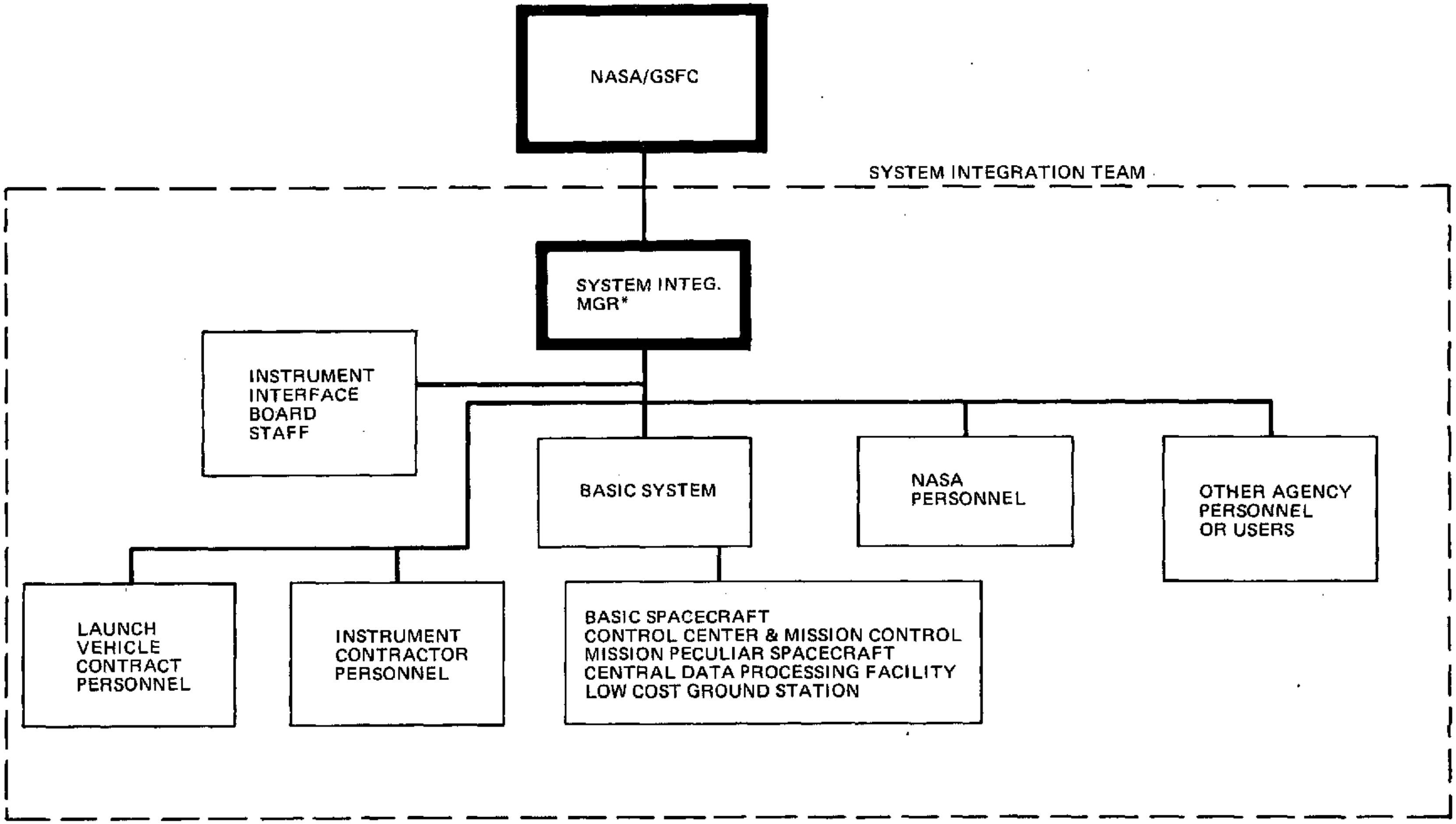
The instruments and DMS operations for the initial flights are procured by the Government and provided to the System Integrator as GFE. The System Integrator will manage the instrument contractors through the System Integration Team, and will resolve interfaces within the team or by an Interface Board with Goddard project management approvals. The candidate instruments for the EOS program are in high-risk and low-risk categories. Since the TM and HRPI have a higher development risk, it is recommended that cost-type contracting be utilized. Instruments (such as the MSS and certain SEASAT instruments) that are of sufficiently low risk can be procured by either a firm fixed price contract or a fixed price incentive contract.

The launch vehicle, shroud, FSS, MEMS and modifications to the DAS are to be procured under the normal Government procurement practices.

The System Integrator is the prime contractor for the EOS-A and -A' mission, including the Basic Spacecraft, PCC, mission peculiar spacecraft, CDPF, and LCGS. We recommend 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.

The competition for the EOS-A and -A' execution phase will be a management and technical competition. A cost-type contract should be used for this procurement. In accordance with the objectives of a DTC program, and as required by the System Integrator responsibility to manage within the DTC goals, cost tradeoffs will be a continuous requirement. The Basic Spacecraft, modules and the LCGS may be procured by fixed price contracts following their development. For follow-on missions, the Basic Spacecraft or selected modules can be procured by the Government and supplied to a System Integrator GFE, or a procurement package including drawings and specifications can be provided GFE. The LCGS can be procured by the Government for use by the users (Option A), or the procurement package could be provided for the use of the user (Option B)

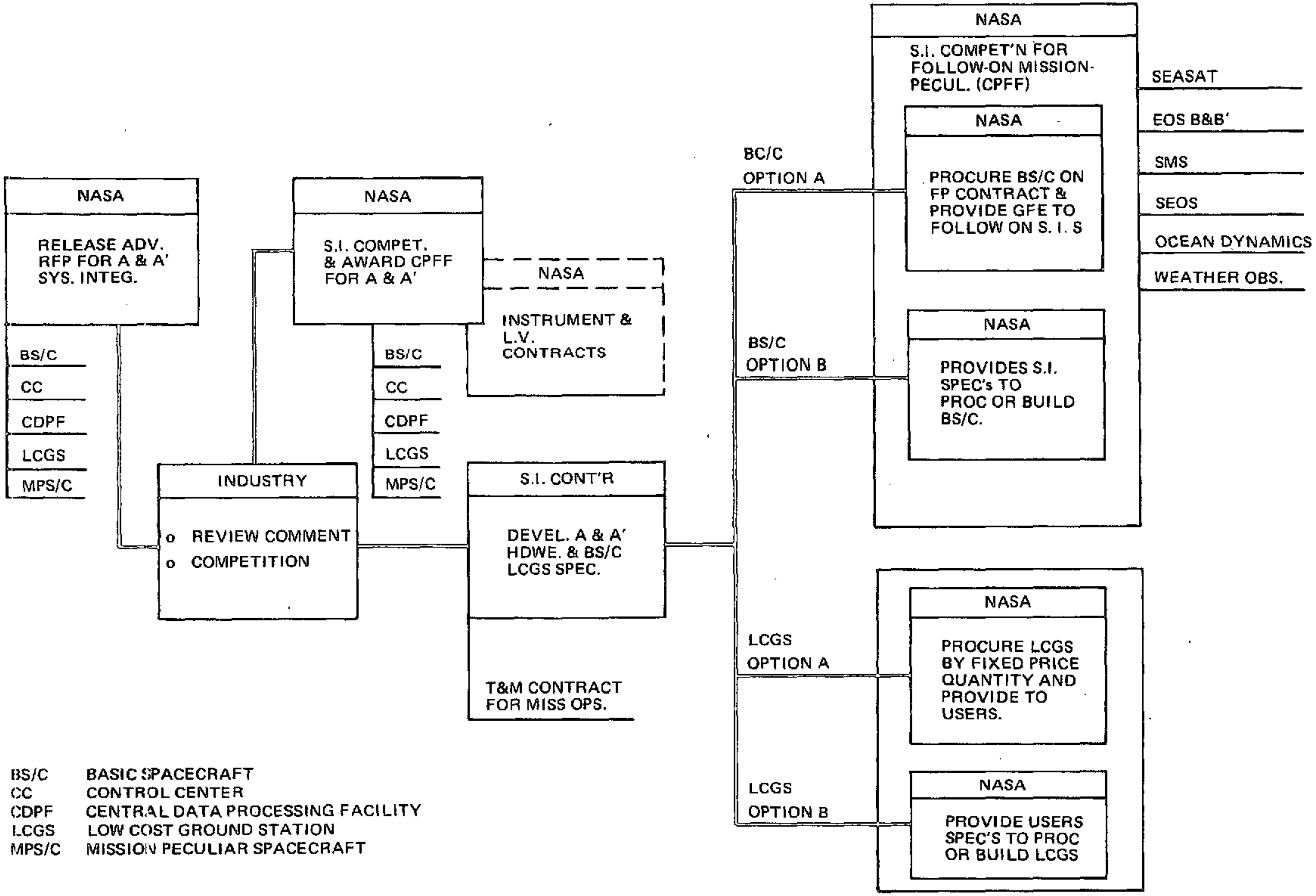
The DMS operations (including the Mission Control, data processing operations and support)



* BASIC SYSTEM CONTRACTOR FOR EOS A, A'. FOR FOLLOW-ON MISSIONS MAY BE OTHER CONTRACTOR OR AGENCY.

3-200
4-2
7-28

Fig. 11-3 System Integration Team



BS/C BASIC SPACECRAFT
CC CONTROL CENTER
CDPF CENTRAL DATA PROCESSING FACILITY
LCGS LOW COST GROUND STATION
MPS/C MISSION PECULIAR SPACECRAFT

Fig. 11-4 EOS Program Development

should be contracted on a time-and-material or labor-type basis.

The overall contractual plan makes full use of a DTC philosophy, and presents a low-cost approach to the EOS-A and -A' execution phase. Cost savings expected from the above approaches are summarized in Table 11-2. The plan provides the structure to manage within program funding, and flexibility to manage within fiscal year funding. Also, an early selection of the System Integrator will assist in the instrument procurement as well as 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.

Table 11-2 Potential Cost Savings
(IN 1974 MILLIONS OF DOLLARS)

MANAGEMENT APPROACH	POTENTIAL COST SAVING (EOS A AND A')
• DESIGN TO TARGET COST FOR BASIC SPACECRAFT AND INITIAL DMS	12.2
• SYSTEM INTEGRATION TEAM CONCEPT	1.0
• SIMPLIFIED CONTROLS AND DOCUMENTATION	1.25
• SIMPLIFIED TEST	1.8
• GFE INSTRUMENTS	12.4
• DIRECT PROCUREMENT-OPERATIONS DATA PROCESSING	3.2
	TOTAL 31.85

3-261, 7T-24
7T-50