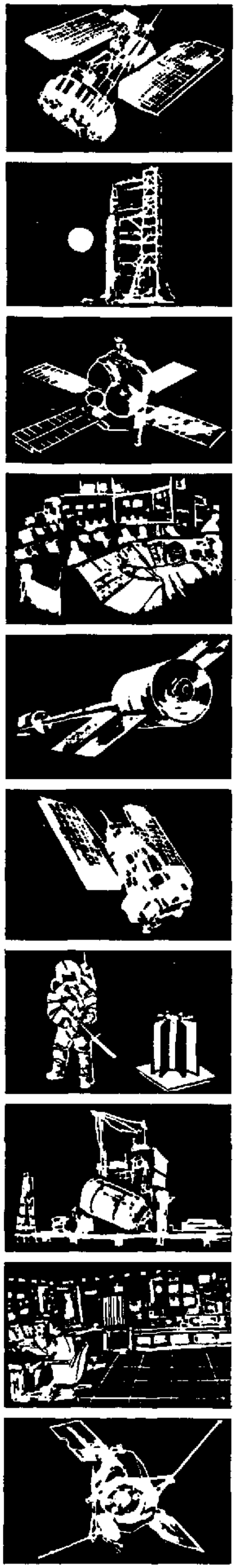

SPACE DIVISION

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15 July 1974

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

Report No. 2

INSTRUMENT CONSTRAINTS AND INTERFACES

EARTH OBSERVATORY SATELLITE

GENERAL ELECTRIC

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INSTRUMENT CONSTRAINTS AND INTERFACES

Prepared for:
GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland 20771
Under
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GENERAL ELECTRIC
SPACE DIVISION
Valley Forge Space Center
P. O. Box 8661 • Philadelphia, Penna. 19101

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SECTION 1.0
SUMMARY

1.1 SCOPE/BACKGROUND

The Instrument Constraints and Interface Specifications Report has been focused primarily at a Land Use Classification mission utilizing a 7 band Thematic Mapper and a 4 Band High Resolution Pointable Imager. Extrapolations from this baseline to the myriad of mission objectives, instrument complements and instrument contractor's design interations that are being proposed within the remote sensing community have been documented where apropos. The Thematic Mapper is a 7 band nadir-looking radiometer with 30 meter ground IFOV, 185 KM swath, with one of three candidate scanning techniques. The HRPI is a 4 band radiometer with equal to or less than $\pm 45^{0}$ offset pointing, 10 m ground IFOV, 45 KM swath, with one of four candidate scan techniques.

To provide maximum utility from this instrument study, the mission and performance for the instruments as defined in specification EOS-410-02 were first reviewed and expanded to reflect the instrument as part of the total remote sensing system. The candidate instrument designs were then extrapolated to this expanded baseline, and critiqued both in terms of their adaptation into the overall system and on a comparative basis relative to the alternate instrument design approaches. Also included as an appendix is a preliminary EOS Interface Handbook, which briefly describes the mission and system, specifies the spacecraft interfaces to potential instrument contractors and describes the instrument interface data required by the system integration contractor.

The variation in both the depth of design detail and in baseline design requirements and assumptions among the candidate instrument contractors dictated a top level instrument concept evaluation. Detail design features (where available) add to design credibility and were influential in the evaluation, but have not been critiqued in this report.

1.2 KEY CONSIDERATIONS

The following paragraphs describe the major factors which scoped the instrument evaluation effort. These key considerations provide the basis for the criteria used in the subsequently described instrument evaluation and recommendations.

Central Ground Station. Various instrument scan techniques exist. Each scan technique must be evaluated relative to both the cost impact of processing its data within the central data processing facility and its compatibility with current and future types of resources management information system and analysis techniques.

Low Cost Ground Stations. A major factor in candidate instrument evaluation is the cost of processing required to provide data of acceptable quality for many low cost user stations. The ability of an instrument to operate with relatively simple ground receiving and processing must be weighed heavily. In addition, given that there is the potential for many low cost ground stations, then on-board processing becomes cost effective. Hence, the compatibility of the instruments with the various on-board processing techniques must also be evaluated.

System Performance. The remote sensing data user community has progressed to include sophisticated technical disciplinaries who are concerned with the utility of the available data. Subjective, fuzzy definitions of data quality are no longer appropriate. The "standard" performance parameters of resolution, geometric and radiometric accuracies by which data quality is judged must be defined and specified to a level such that the user may determine the utility of the data for the type of information extraction that he requires.

Wideband System. A limiting system element that constrains overall system performance is the wideband data system. An upper limit of 240 mbps (375 MHz bandwidth) has been used in synthesizing system designs.

Spacecraft Interface. The instruments are being developed to fly in the pre-shuttle era where the advantages of less emphasis on weight and volume are unfortunately not yet available. The use of a Delta launch vehicle minimizes the cost of the overall system but requires light weight, small volume and relatively low power instruments.

Design Flexibility. The fluid state of definition of early EOS missions dictates that the basic instrument designs be flexible enough to accommodate changes in mission requirements (such as swath width or operating altitude) both early or fairly late in the design cycle. These changes must be accommodated without major cost or schedule impact.

Design Risk. Both the high development cost and lack of shuttle retrieve capability during the early flights of the instruments require that the standard aerospace philosophy of minimum risk be followed. Risk in any of the areas of design, development, manufacture, test, launch environment, or lifetime must be given serious consideration in determining the acceptability of a particular instrument design.

1.3 CONCLUSIONS/RECOMMENDATIONS

THEMATIC MAPPER

Geometric Accuracy. Geometric Accuracy will be difficult but nevertheless possible to achieve. Additional study by instrument contractors is required to verify their ability to meet the detailed error budgets. Scanners with multiple scanning mirrors need to provide an indication of which mirror is being used to aid data processing.

Radiometric Accuracy. Radiometric Accuracy appears realistic but also requires additional study and verification by instrument contractors against detailed error budgets.

In Flight Calibration. The following is recommended:

a. Each instrument should be capable of using the sun for an absolute calibration source on command.

b. Delete electronic calibration if such a calibration introduces any additional noise sources.

c. Incorporate DC restoration in the instrument to preserve dynamic range and reduce auxiliary data requirements and processing load on the ground, particularly at the low-cost ground stations.

Band-to-Band Misregistration. When in the "raw" instrument data stream misregistration should be constrained to a range of several hundred pixels maximum in the cross-track direction only. Band 7 should lead the scan, since it requires fewer (larger) pixels storage to register to the other bands. Further, the ratio of the size of the detectors in band 7 should be an integral multiple of the size of the detectors times the number of detectors per band in the other bands to simplify data processing.

Noise. Dynamic performance analysis indicates that the white noise power spectral density is the critical parameter to consider in relation to data extraction utility. Current S/N specifications constrain only the integral of the noise power spectrum from DC to some band limit frequency. No change is proposed except to make the noise power spectrum available and to specify the S/N at a signal frequency equivalent to $\frac{1}{FOV}$ instead of the current practice of $\frac{1}{2\ FOV}$.

Spectral Separation Technique. Multilayer interference filters and spatial separation is preferred to prism monochromators for spectral band determination. The interference filters provide significant flexibility in design and provide better optical efficiency. Band-to-band registration, which is generally used to justify the prism monochromators, can be met anyway, using tight alignments and good design in the spacecraft and/or provided for easily and cheaply in ground processing.

Analog/Digital Conversion and Multiplexing. In lieu of 100 analog signals being routed to a remote digitizer and the resulting fragmented specifications and test program, the sampling, A/D conversion and submuxing should be performed within the instrument. The input data to the spacecraft multiplexer (which adds auxiliary data and generates the composite digital data stream) should be digital and ordered on a per band basis. Auxiliary data added to the composite video will be generated and inserted by the system contractor to facilitate ground processing.

Band 6. No clear-cut answer has been found for the Band 6 controversy. Therefore, it is recommended that an assessment be made by each instrument contractor of the costs savings resulting from deletion of this band or from relaxing performance requirements. A small cost saving would suggest keeping the band to experimentally determine the utility of the data. A large cost saving would justify its deletion.

Maximum/Minimum Radiances. More detailed analysis of available data is required prior to the selection of maximum and minimum radiances. A "best estimate" is included which differs somewhat from the values specified by GSFC.

NUMBER OF DETECTORS PER BAND

The number of detectors per band must be set with an integer relationship between bands 1 thru 6 and band 7 to aid in data registration. Reduced resolution (compacted data) must be an integral divisor of the total number of detectors in a band.

HIGH RESOLUTION POINTABLE IMAGER

Data Rate. The pushbroom HRPI has an inherent 100% scan efficiency and, even assuming as high as 10% overhead for addition of ancillary data, provides the lowest data rate of all HRPI concepts.

Calibration Complexity. The large number of elements in the pushbroom scanner considerably increases the calibration complexity. Due to responsivity and offset variation from element-to-element a transfer characteristic is required for each element. The Honeywell image plane scanner requires 80 calibration curves, while the Te scanner require 50. Although the Hughes CCD array has 18 x 15 elements per band, it requires only 18 calibration curves per band because the charge transfer process averages detector variations over the 15 elements in the scan direction. Also, the CCD array design minimizes the calibration problem because the image is always on axis; hence, uniform illumination as well as knowledge of the irradiance distribution over a wide field angle is not required. The system impact of 18 versus 50 or 80 calibration curves is not especially significant. The impact of 19,200 calibration curves is, however, and reflects itself in more costly processing on the ground system.

Offset Pointing. The offset pointing technique used in the linear image plane scanning HRPI provides the largest pointing angle without vignetting, has the least size/weight impact on the instrument design, and is the most straightforward to implement. The pushbroom HRPI offset pointing for large angles (30-45°) requires either a large mirror or the whole instrument to be pointed. Conversely, the pushbroom HRPI provides a simple solution to scan-to-scan overlap that all mechanical sensors have at large offset angles. From a data quality point of view, offset angles for any approach should not exceed 35°.

Size Reduction. The Westinghouse pushbroom scanner design greatly exceeds the S/N requirements. The instrument size (and weight) can be reduced considerably while still exceeding the S/N goals. This size reduction will alleviate the offset pointing problem such that the instrument can be oriented with the optical axis normal to the velocity vector to eliminate image rotation, and a pointing mirror can be used.

SNR. In comparing mechanical scanners, the object plane scanner provides the greatest clear aperture for a given instrument size. This is reflected in either a smaller instrument for a given SNR requirement or higher SNR.

CCD Technology. The CCD array focal plane with time delay integration has the advantage of increasing the integration time per element. However, some of the characteristics such as crosstalk between elements, blooming at high irradiance levels, MTF and dynamic range must be evaluated to determine the applicability of this technology to high resolution and high radiometric accuracy instruments.

Alignment of Chips. In the pushbroom HRPI the alignment of chips into an array and alignment of arrays in the four spectral bands with respect to each other is critical. Initial alignment and maintenance of this alignment through the launch and temperature environments needs to be investigated.

Error Budget. In the Hughes design the number of elements per band is constrained to 18 in order to minimize mapping errors at an offset pointing angle of 40^{o}. If the error budgets were limited to within the thematic mapper swath ($\pm$ 7^{o}), the number of elements would not be constrained, and the time delay integration afforded by the CCD technology would not be required to meet the performance requirements.

COMBINED TM/HRPI INSTRUMENT

Because of the significant limitations in mission flexibility and adaptability imposed by placing the high resolution FOV within the low resolution FOV, it is not recommended that the combined TM/HRPI be considered for EOS missions.

PROGRAMMATIC

Instrument Evaluation Results. The non-uniformity of specifications and resulting point design parameters for the candidate instruments, while an impediment at times, does not prevent a valid assessment of the various designs and their impact on the overall EOS system.

Detector Technology. Many different detectors and operating temperatures are under consideration by the instrument contractors. Cooling is a question that still needs to be resolved. Very little parametric noise data seems available. The detector and pre-amplifier technology efforts in process at the various instrument contractors seems to be uncoordinated and fragmented. It would seem appropriate for NASA to lead this technology area with sufficient money and direction to provide a consistent set of data available to all. CCD technology and its application to remote sensing should be pursued.

Cost Models. All of the instrument contractors are still working to the criteria of near diffraction limit performance. As we change our ground rules for shuttle launched instruments where weight and size becomes less of a problem and where dynamic performance response becomes the design criteria, optics will become large low-figure light gatherers. It is recommended that additional instrument cost modeling and cost trades be performed to determine the appropriate direction for instrument designs in the shuttle era which will minimize total system costs.

1.4 SUMMARY INSTRUMENT EVALUATION

The relative rankings of the three Thematic Mapper approaches are summarized in Table 1-1, with respect to their overall system impact. The basis for the rankings is summarized in Section 2.4 of this report. It must be emphasized that all instrument approaches are feasible from a development standpoint, in terms of their abilities to be integrated into the system and their capability to serve as the primary data source for the Land Resources Management mission.

From Table 1-1, it is obvious that the object plane scanner and the linear image plane scanner generally exceed the conical image plan scanner in the various evaluation categories. In addition, the overall cost impact on the system is greater for the conical scanner (Reference Volume III). For these reasons, the object plane and linear image plane scanners are preferred with not a great deal to choose between them. The final decision must be made considering the instrument manufacturers cost to complete and their ability to deliver on schedule.

Table 1-2 is a corresponding summary evaluation of the HRPI designs. Detailed evaluauation of the HRPI designs was somewhat more cursory than for the TM approaches because of the timing and design detail of the HRPI study results. Once again, the object plane and linear image plane scanners rank significantly higher than the image plane conical scanner or either version of the pushbroom arrays. But also, any version is feasible and can perform the intended mission. In terms of overall system cost impacts, the object plane scanner and linear image plane scanners are the lowest with a plus delta of 185K for the pushbroom arrays and a plus delta of $350K for the conical scan (Reference Volume III). The final decision here must also be made considering manufacturers cost to complete and schedule. One other factor must be considered and that is the desirability of developing a new (pushbroom array) technology area in parallel with the scanners. This factor was not and could not be considered in the evaluation.

Table 1-1. Summary Evaluation of Thematic Mapper Approaches

Criteria	Image Plane Conical Scan	Object Plane Scan	Image Plane Linear Scan
Performance			
Signal to noise	-*	+	-
Inherent Geometric Accuracy	+	-*	++
Overall Radiometric Accuracy	0	0	0
Spacecraft Interfacing			
Envelope and Weight	-*	+	0
Power	-*	0	0
Data Handling	0	-*	0
Design Detail	0*	+	0*
Central Data Processing			
Cost	-	0	0
Complexity	-	0	+
Local User Stations			
Cost	0	0	+
Complexity	0	0	+
Design Flexibility			
Change Swath	-	+	0
Change Altitude	-	0	0
Change Spectral Bands	-*	0	0
Accommodate on-board correction	-	0*	+
Design Risk			
Inherent Design	-	0	-
Experience	+	+	0

- Marginal performance

0 Acceptable performance

\+ Significantly better than other approaches

* can be improved significantly

Table 1-2. **Summary Evaluation of High Resolution Pointable** Imager Approaches

Criteria	Pushbroom Array Linear	Pushbroom Array Staggard	Image Plane Conical Scan	Object Plane Scan	Image Plane Linear Scan
Performance					
Signal to noise	+	++	-	0	-
Inherent Geometric Acc.	+	0	-	-*	++
Overall Radiometric Acc.	-	0	0	-	0
Spacecraft Interfacing					
Envelope and Weight	-*	-*	-*	+	-
Power	0	0	-*	0	0
Data Handling	+	+	0	-*	0
Design Detail	0*	0*	0*	+	0*
Central Data Processing					
Cost	-	-	-	0	0
Complexity	0	-	-	0	+
Local User Stations					
Cost	-	-	0(1)	0	+
Complexity	0	-	0	0	+
Design Flexibility					
Change Swath	0	0	-	+	0
Change Altitude	+	0	-	0	0
Change Spectral Bands	-	-	0	0	0
Accommodate On-board Correction	-	-	-	0*	+
Design Risk					
Inherent Design	0	0	-	0	-
Experience	0	0	+	+	0

- Marginal performance

0 Acceptable performance

\+ Significantly better than other approaches

* can be improved significantly

(1) Conical format storage

1.5 5 BAND MSS MISSIONS

Limiting the mission altitude to the 496 n mile design altitude of the ERTS MSS would impost stringent design requirements on the spacecraft, launch vehicle and TM which may not be warranted. Additional system analysis needs to be performed to optimize the altitude for an EOS mission with a MSS/TM complement. A primary initial step is evaluating the effects of changing the orbital altitude of the MSS.

The three major items to be considered are:

- o ground resolution - the current MSS has some excess signal to noise and can accommodate improved resolution.
- o data rate - data rate increases to about 18 mbps can be accommodated by the MSS without major redesign or performance degradation. Data rate changes, however, may have serious cost implications to the many users who have equipment designed to operate at the lower rates.
- o scan mirror mechanics - all approaches involve some changes to the scan mirror mechanism. Some are more complex than others with associated cost and risk.

Three options have been identified to permit the MSS to operate at a lower altitude:

1. degrade resolution to about 83 meters but maintain the current 15 mbps data rate (no mux redesign) and only make slight modifications to the optics and scan mirror mechanisms. This is the least expensive and least risky of the three options.
2. maintain the present MSS ground resolution of 78 meters (increasing the instrument IFOV), make some modifications to the mux to operate at 18 mbps and modify the scan mirror mechanism. This is more costly and involves slightly more risk than the first option.
3. take advantage of the lower altitude to improve resolution to approximately 60 meters. This option has as yet many unexplored consequences. A very top level look at the parameters involved is summarized in the following paragraphs without specific endorsement since significantly more work is required prior to reaching firm conclusions. It appears to be the most costly and risky of the three options but buys a noteworthy improvement in resolution performance.

MSS Resolution Improvement. The Instantaneous Field of View (IFOV or pixel) of the MSS on the ground is related to altitude by the formula

$$IFOV_G = \left(\frac{W}{f}\right) h$$

where w = fiber optics size (stop or exit pupil)

= 2.79×10^{-3} inches for present MSS

f = telescope focal length

= 32.5 inches for present MSS

h = orbital altitude

Figure 1-1 indicates ground IFOV as a function of altitude. The relationship could be affected by changing f or changing the fiber optics. Hughes has investigated both options and concluded that changing the optics is slightly cheaper and less risky than changing the fiber optic design.

Swath. The present MSS scan mirror images data $\pm$ 5.78^{o} from NADIR. This too, can be changed slightly but would require a redesign of the scan mirror and scan monitor system. If the scan mirror excursion is held fixed, then the resulting ground swath as a function of orbital altitude is shown in Figure 1-2 where: ground swath = 2 h tan 5.78^{o}. Also shown is the number of days required to obtain full contiguous (10% sidelap) synoptic coverage for that swath.

Stripe Geometry. The present MSS mirror cycle time is 73.42 m seconds. Figure 1-3 indicates the subsatellite distance travelled in this period and the strip width of six detectors per present MSS design as a function of altitude. Significant underlap and overlap exists unless the mirror period is changed as a function of altitude. Also shown in Figure 1-3 is the $\pm$0.5 IFOV underlap/overlap lines which indicate that $\pm$10 nm about the 496 nm nominal ERTS altitude can be accommodated without appreciable performance degradation. This indicated that the 506 nm, 9 day repeat orbit can be accommodated with the present design. Figure 1-4 indicates the mirror period required for zero overlap/underlap as a function of altitude. An evaluation should be performed to determine the hardware limits and problems associated with changing the mirror frequency. The mirror structure has a natural frequency around 2 Hz and since the scan frequency increases with decreasing altitude, no problems are expected.

Sample Rate. The present MSS samples at 1/IFOV in-track and 1.4/IFOV cross-track. If we assume that the number of pixels per stripe remains constant, then the data rate changes as shown in Figure 1-5 as a function of altitude and cross-track oversampling. Three required studies become obvious:

1) What are the effects of changing the 1.4 times oversampling in the cross-track direction on system performance and ground processing.
2) What are the data rate change impacts on the mux, demux, wide band data handling system and ground tape recorder.
3) What are the spacecraft and ground processing impacts of on-board data compaction to increase the effective scan efficiency above 46%.

S/N. As data rate increases, the instrument signal to noise decreases. A system level study of the effects on output product utility needs to be performed.

Band-to-Band Registration. The band-to-band registration is nearly an integer number of pixels at design altitude. If the focal plane configuration of the instrument is not adjusted as a function of altitude, the band to band registration errors will increase (but not by more than $\pm$ .5 pixel).

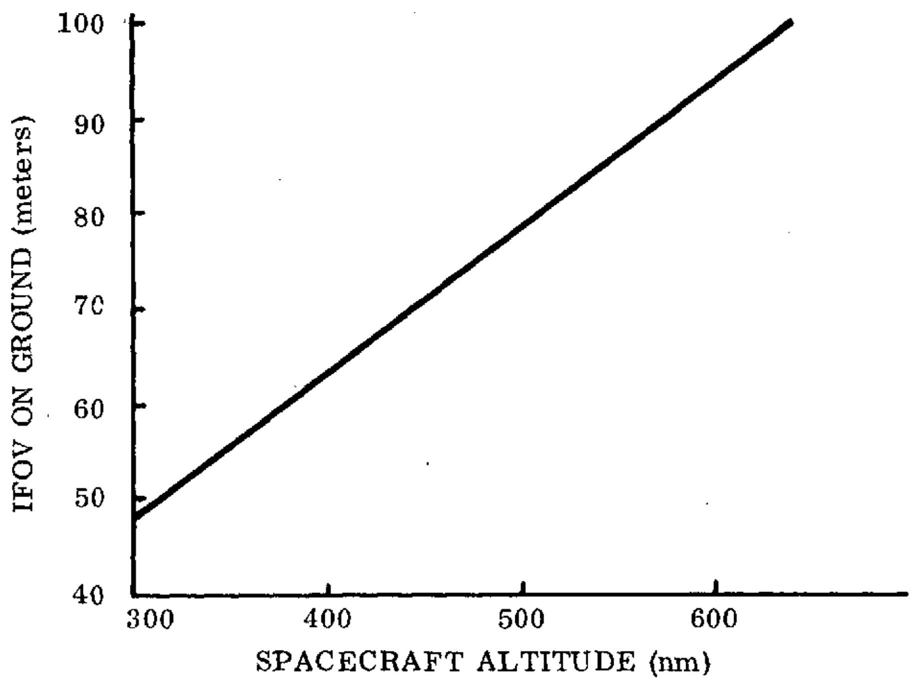


Figure 1-1. IFOV vs. Altitude for ERTS-1 MSS

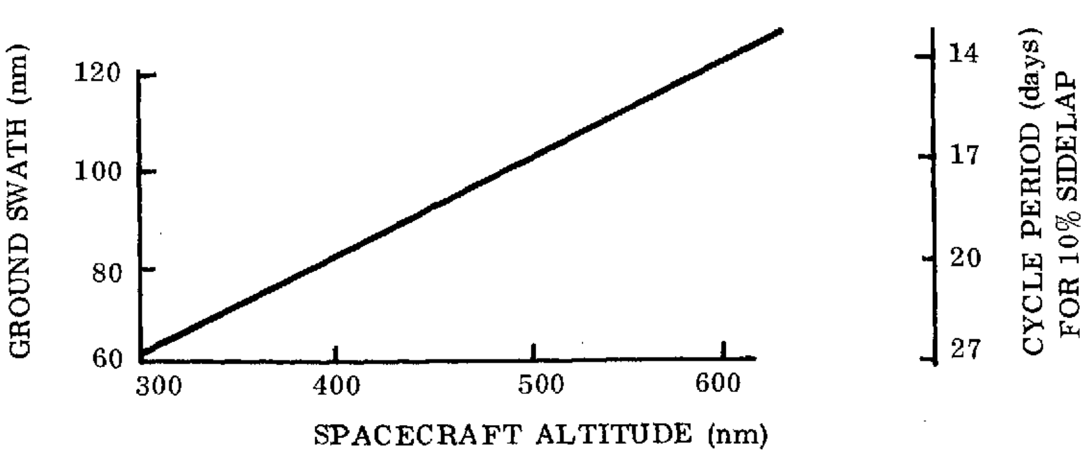


Figure 1-2. Swath Width vs. Altitude for ERTS-1 MSS

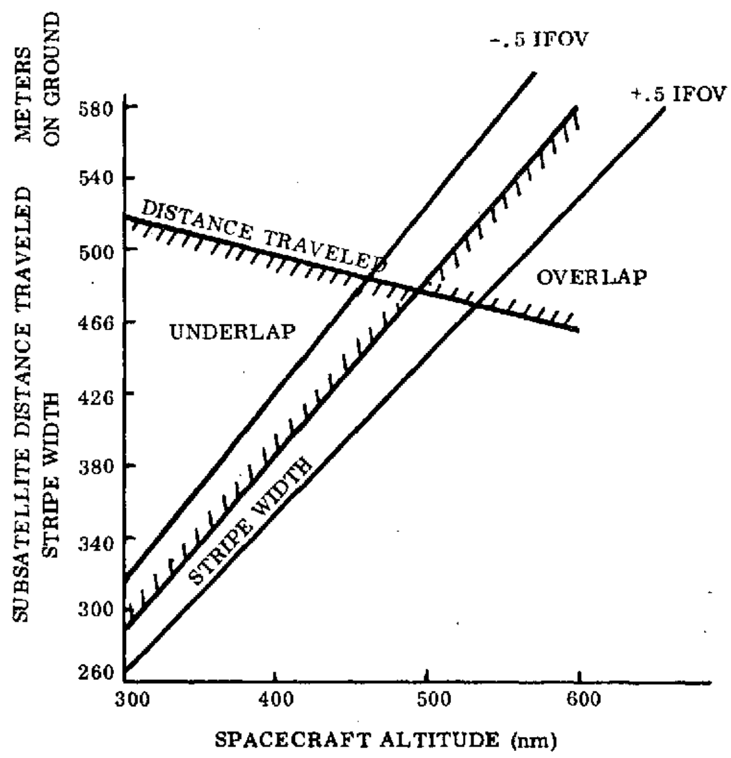


Figure 1-3. Underlap/Overlap as Function of Altitude for Scan Period of 73.42 m Seconds

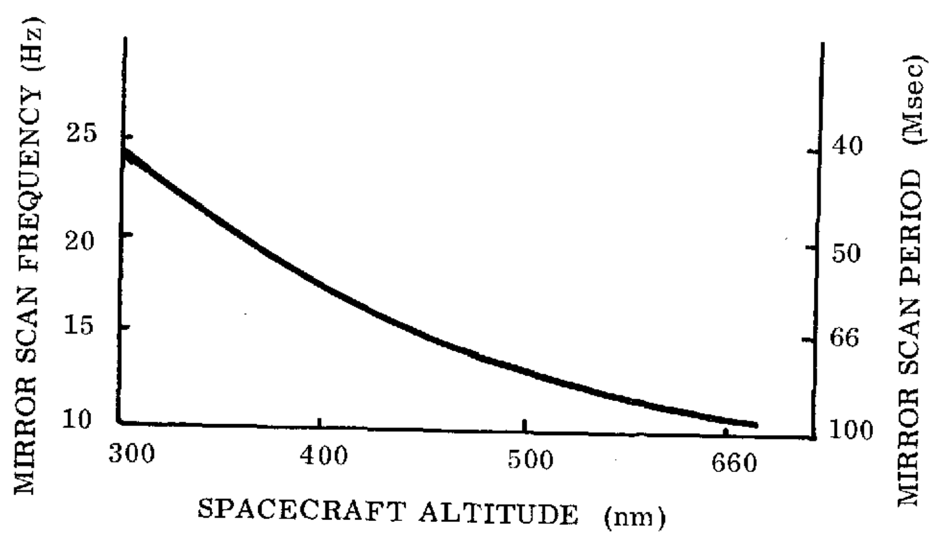


Figure 1-4. Mirror Frequency and Period vs. Altitude for Zero Underlap/Overlap

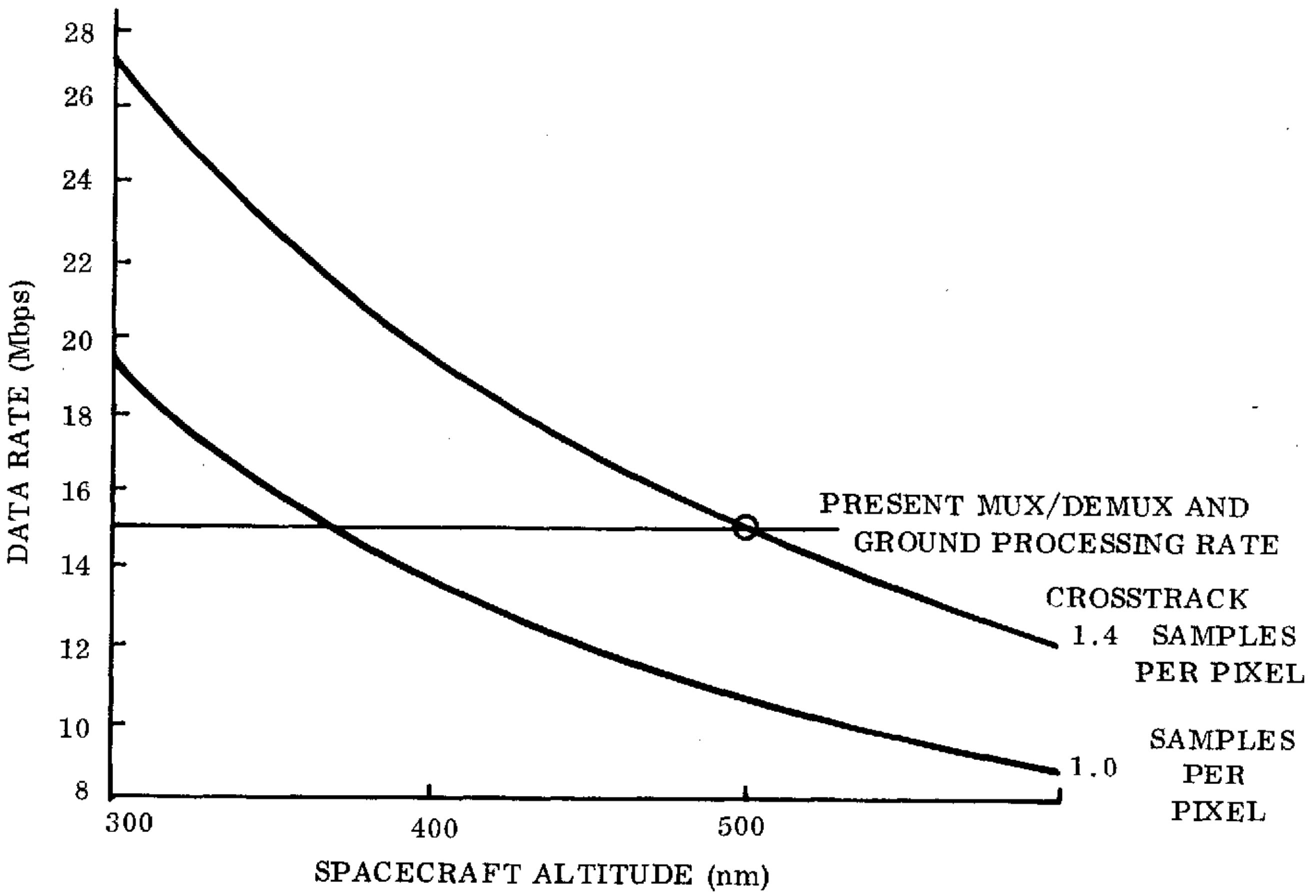


Figure 1-5. Data Rate vs. Altitude

SECTION 2.0

THEMATIC MAPPER

2.1 POINT DESIGN STUDIES AND UPDATE COMPARISONS

Table 2-1 indicates the non-uniformity of specifications and resulting design parameters that were addressed while accessing the point design studies of the candidate instrument contractors. These conditions exist because:

1. The studies were contracted differently (i.e., existing contract extensions, new work statements, etc.)
2. Requirements were slightly time phased.
3. Contractors applied varying amounts of anticipation of requirements and design constraints.
4. The candidate contractors worked to varying depths of design detail.

GE's approach to the study, as detailed in subsequent paragraphs, was to generate a mission and performance baseline set of requirements, extrapolate the updated instrument designs to the common baseline and then perform evaluation tradeoffs.

2.2 EOS INSTRUMENT REQUIREMENTS/SPECS

2.2.1 MISSION REQUIREMENTS (HRPI AND THEMATIC MAPPER)

One of the major overall objectives of the EOS-A mission is to support R&D effort in the applications area of Land Resources Management. This mission will develop advanced instruments and processing systems which can provide multi-spectral imagery of the land surface of the Earth at significantly improved spatial, spectral and temporal resolutions than is available from either the Earth Resources Technology Satellite or the projected Department of the Interior's Earth Survey Operational System. It thus will permit studies of the direction in which operational land use inventory and Earth resource management programs should proceed. Initial flight test of the instruments and applications research with the data will be in 1979. Key to the satisfaction of this broad mission

Table 2-1. Summary of Instrument Contractors Point Design Parameters

	Thematic Mapper		
	Original	Update	
Altitude, Km			
Honeywell	914	900	
Hughes	914	717	
Te-Gulton	914	715	
Descending Node Time			
Honeywell	9:30	9:30	
Hughes	9:30	9:30	
Te-Gulton	9:30	11:30	
Angular IFOV, μrad			
Honeywell	30	33	
Hughes	30	30	
Te-Gulton	30	35	
Clear Aperture Area, cm^2			
Honeywell	950	950	
Hughes	1450	990	
Te-Gulton	525	625	
F-Number			Bands
Honeywell	5.6	5.6	1-7
Hughes	4.33	6.0	1-4
	2.2	2.5	5-7
Te-Gulton (difficult to define)	26	10	1-4
	11	10	5-6
	1.5	10	7
Number of Detectors/Band (Bands 1-6 / Band 7)			
Honeywell	16/4	16/4	
Hughes	14/4	16/4	
Te-Gulton	15/3	15/3	

Table 2-1. Summary of Instrument Contractors Point Design Parameters (Continued)

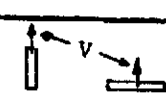
	Thematic Mapper Original	Thematic Mapper Update	
Detector Types			
Honeywell	PMT Si Hg Cd Te	Si PIN Hg Cd Te	
Hughes	PMT Si In Sb Hg Cd Te	Si PIN In Sb Hg Cd Te	
Te-Gulton	Si In Sb Hg Cd Te	Si In Sb Hg Cd Te	
Dwell Time, μ sec			Bands
Honeywell	8.64 34.5	10.0 40.0	1-6 7
Hughes	7.1 7.5 25.0	4.4 4.4 17.6	1-3 4-6 7
Te-Gulton	9.2 46.0	8.0 40.0	1-6 7
Data Rate, Kbps			
Honeywell	63.0	67.0	
Hughes	83.3	153.3	
Te-Gulton	69.0	81.7	
Size Envelope, in.			
Honeywell	84L x 36D	72L x 36D	
Hughes	83 x 44 x 25	a) 67 x 36 x 20 b) 42 x 36 x 36	
Te-Gulton	84 x 36 x 38	84 x 36 x 38	
Weight, lbs			
Honeywell	450	600	
Hughes	401	a&b) 320	
Te-Gulton	598	598	
Power, watts			
Honeywell	180 + 50 heat	180 + 50 heat	
Hughes	55 + 0 heat	45 + 0 heat	
Te-Gulton	110 + 10 heat	110 + 10 heat	
Orientation		Either	 L = V L⊥V L = long dimension V = velocity vector
Honeywell	L=V		
Hughes	L⊥V	a) L⊥V b) L=V	
Te-Gulton	L⊥V	L⊥V	

Table 2-1. Summary of Instrument Contractors Point Design Parameters (Continued)

Signal to Noise Ratio

Band	Minimum Radiance watt/cm^2 - x 10^{-5}		SNR TM		HRC TM		HAC TM		Te-Gulton TM	
	Original	Update	Original	Update	Original	Update	Original	Update	Original*	Update
1	11	22	12	10	11	11.5	17.4	12.3	14.5	8.7
2	9	19	9	7	8	12.5	13.5	13.8	14.5	11.8
3	7	16	5	5	5	13.8	9.6	13.6	13.2	13.1
4	11	30	3	5	3.8	15.6	6.1	14.8	11.9	17.5
5	2	8	4	5	5	23.0	5.2	3.8	7.0	11.5
6	1	3	3	5	.55	2.0	2.7	1.9	3.7	6.5
7	$300^{\circ}K$	$300^{\circ}K$	$.5^{\circ}K$	$.5^{\circ}K$	$.34^{\circ}K$	$.3^{\circ}K$	$.55^{\circ}K$	$.97^{\circ}K$	$.3^{\circ}K$	$.3^{\circ}K$

3% ≥5%

Based on Hovis's
pre-ERTS Radiance numbers.

* SNR calculated on basis of up-date minimum radiance values

objective is the development of sensors and other spacecraft systems to acquire spectral measurements and images suitable for generating thematic maps of the Earth's surface. Two instruments have been selected to support these early EOS missions: a seven band Thematic Mapper and a four band High Resolution Pointable Imager. These instruments must be designed consistant with the following mission requirements:

Global Coverage. EOS-A will image the bulk of the world's land masses and near-costal zones. This coverage requirement can be translated into a maximum average imaging time of about 15 minutes per orbit.

Altitude. The spacecraft will operate in a 774 km/altitude, (418 n. m.) sun-synchronous orbit. This orbit permits full global coverage with the TM and at the same time provides the opportunity for access to any point on the globe using a HRPI with offset pointing capability of $\pm$ 30 degrees from nadir. The 774 km orbit exactly repeats its coverage every 17 days.

Swath Width. The swath width projected on the ground from the 774 km orbit will be 185.3 km for the TM. The swath width at nadir for the HRPI will be 46.3 km.

Offset Pointing. The HRPI will incorporate an off nadir pointing capability in the cross track direction (perpendicular to the orbital plane) of $\pm$ 30 degrees in 1° steps with 0.05° repeatability. There is no requirement for offset pointing capability in the TM.

Descending Node. The exact descending node time for the satellite has not yet been determined but will be between 10 am and noon.

Spectral Bands. To satisfy the mission objective of providing improved spectral resolution, seven spectral bands are required in the TM as follows:

0.5 - 0.6 μ_m 0.6 - 0.7 μ_m 0.7 - 0.8 μ_m 0.8 - 1.1 μ_m	μ_m	Visible and near IR to continue R&D effort with improved spatial resolution in these bands.
1.55 - 1.75 μ_m 2.1 - 2.35 μ_m	μ_m	Provide space based measurements in the IR portion of the spectrum to permit new R&D effort particularly in the agricultural area.
10.4 - 12.6 μ_m		To continue R&D in the thermal IR region with improved spatial resolution.

Four spectral bands are required for the HRPI instrument to permit an evaluation of both high resolution sampling techniques and the advantages of improved temporal resolution via off nadir pointing in a well understood spectral region. The bands are:

0.5 - 0.6 μm
0.6 - 0.7 μm
0.7 - 0.8 μm
0.8 - 1.1 μm

Resolution. Consistant with overall mission objectives of providing significantly improved spatial resolution, the instruments will provide data with the following:

TM -	Bands 1-6	30 meters
	Band 7 (Thermal IR)	120 meters

2.2.2 PERFORMANCE REQUIREMENTS

The GSFC Specification for an Earth Observation Scanning-Radiometer Experiment (SSR) is quite thorough. If the point design studies had been generated to this set of requirements, there would be less of an evaluation problem. The only topics not covered were spacecraft interface items such as uncompensated angular momentum, magnetic dipole restrictions, etc., which are discussed in Appendix A, The EOS Handbook. The objective of this section is to describe the updating of the TM requirements to a common baseline mission and to further delineate the instrument's portion of the total system error budget. In several areas, there are insufficient data to determine precise final requirements (i.e., minimum and maximum radiance, minimum acceptable S/N). In these areas a best estimate has been made and, where appropriate, additional tasks that are beyond the scope of the present study have been suggested. For instance, no systematic study of MSS scene radiance of temporal synoptic coverage has been found. A histogram of MSS output levels for a large sample of scene content over seasonal and all-weather conditions could be used to set more realistic minimum and maximum radiance levels.

The following sections are by design much more tutorial than required for a specification, but it seemed appropriate to keep supporting documentation and assumptions together with the baseline requirements recommendations.

2.2.2.1 Sun Angle

Figure 2-1 indicates the daily solar altitude as a function of latitude for a 1200 hours descending node. A 30° sun angle has been selected to set minimum radiances.

2.2.2.2 Minimum and Maximum Radiance

The MSS gain settings were based on work that Dr. Warren Hovis performed in late 1970 to set maximum orbital radiance values for the ERTS-1 9:30 orbit for mid-range sun elevation angles. Extrapolation of these original numbers to other sun elevation angles have resulted in the SSR spec numbers shown in Table 2-2. Table 2-2 also also includes data from Te, E.L. Krinov[(1)] based on 400 natural objects and the AFCRL data handbook. Krinov and the AFCRL handbook indicate a very good match with NASA EOS baseline data except in Bands 4 and 5.

The original minimum radiance selected for the EOS point studies was 3% of the maximum scene radiance. This minimum radiance was increased by NASA to the numbers shown in Table 2-3 prior to the EOS TM point study reports being issued. Krinov and AFCRL data is shown for 3 values of reflectance. Te and Westinghouse data is included for comparison.

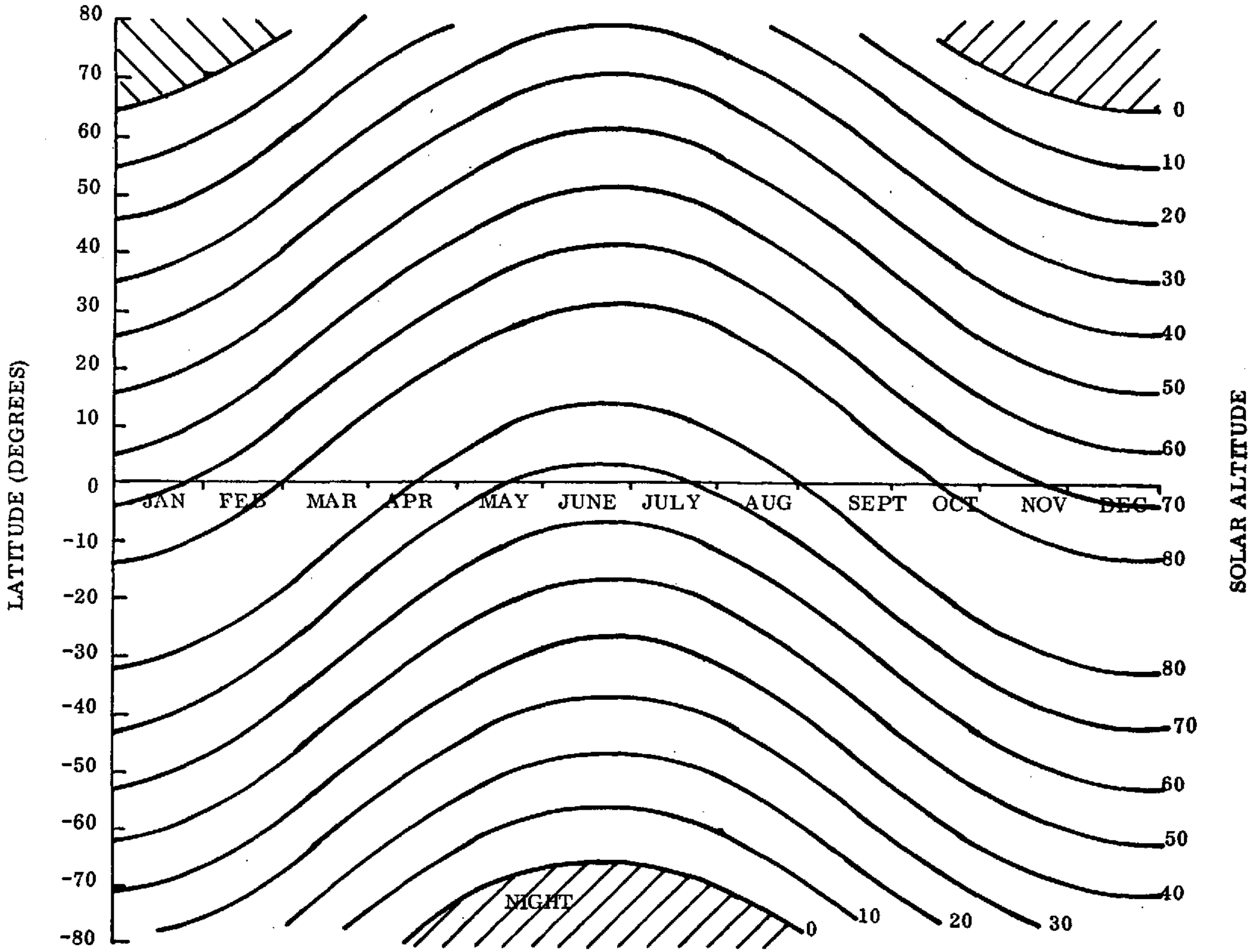


Figure 2-1. Constant Solar Angle Contours for Descending Node at 1200 Hours

Table 2-2. Maximum Radiance Estimates
(watts/cm^2 - STER x 10^{-5})

Band	NASA EOS Spec	Dr. Hovis (ERTS)	Te	Krinov $\alpha=90^o$	AFCRL $\alpha=90^o$
1	363	248	502	320	378
2	297	200	409	318	364
3	231	176	335	270	293
4	363	460	633	563	568
5	66	n/a	86		108
6	33	n/a	48		50
Sun Elevation	?	$45^o - 53^o$	90^o	90^o	90^o
ρ	?	.5 - .8	.8 - .9	1	1
Atmospheric Attenuation	?	Yes	No	Yes	Yes
Sky Radiance	?	Yes	No	Yes	No

Recent ERIM user data indicates that the minimum reflectance required is:

Bands	ρ min
1-4	.02
5	.06
6	.03

Therefore, the EOS minimum reflectances appear to be slightly high. Figure 2-3 indicates how scene radiance increases as a function of ground reflectance for each of the first four spectral bands.

The close agreement of the KRINOV and AFCRL data allows one to select minimum radiance close to the ϵ = .03 values (30^o sun elevation angle) and between the EOS original and revised requirements. Although recommended values for minimum and maximum radiances are presented at the end of this section, more detailed analysis of available data is recommended prior to finalizing these values.

Table 2-3. Minimum Radiance Estimates

$(\text{watts/cm}^2 \Omega - 10^{-5})$

Band	EOS		Te	Westinghouse	Krinov				AFCRL		
	Orig	Rev			Sky	ρ=.03	ρ=.05	ρ=.10	ρ=.03	ρ=.05	ρ=.10
1	11	22	21	12	17.5	18.3	21.0	27.8	18.9	21.8	29.9
2	9	19	17	10	10.1	12.3	15.0	21.8	13.0	16.2	24.4
3	7	16	14	8	6.4	8.9	11.3	17.3	9.0	11.7	18.1
4	11	30	27	16	7.8	14.9	19.0	34.6	14.0	19.1	31.9
5	2	8	4	-	(2) 4.2	-	-	-	5.0	6.2	8.9
6	1	3	2	-	(2) 2.2	-	-	-	2.5	3.0	4.3
Sun Elevation			50°	6°	30°	30°	30°	30°	30°	30°	30°
ρ			.05	.20	-	.03	.05	.10	.03	.05	.10
Atmospheric Attenuation		(1) No?	No	No	-	Yes	Yes	Yes	Yes	Yes	Yes
Sky Radiance		(1) No?	No	No	Yes	Yes	Yes	Yes	Yes	Yes	Yes

(1) No? - Assumption based on relative values of GSFC vs. Te vs. Westinghouse for bands 1-4 being nearly constant

$$\frac{\text{GSFC}}{\text{Te}} = 1.1 \qquad \frac{\text{GSFC}}{\text{Westinghouse}} \doteq 1.7$$

GSFC bands 5 and 6 must be a different reflectance than bands 1-4.

(2) Data from "Spectral Radiance of Sky and Terrain at Wavelengths between 1-20 Microns", JOSA, Dec. 1960.

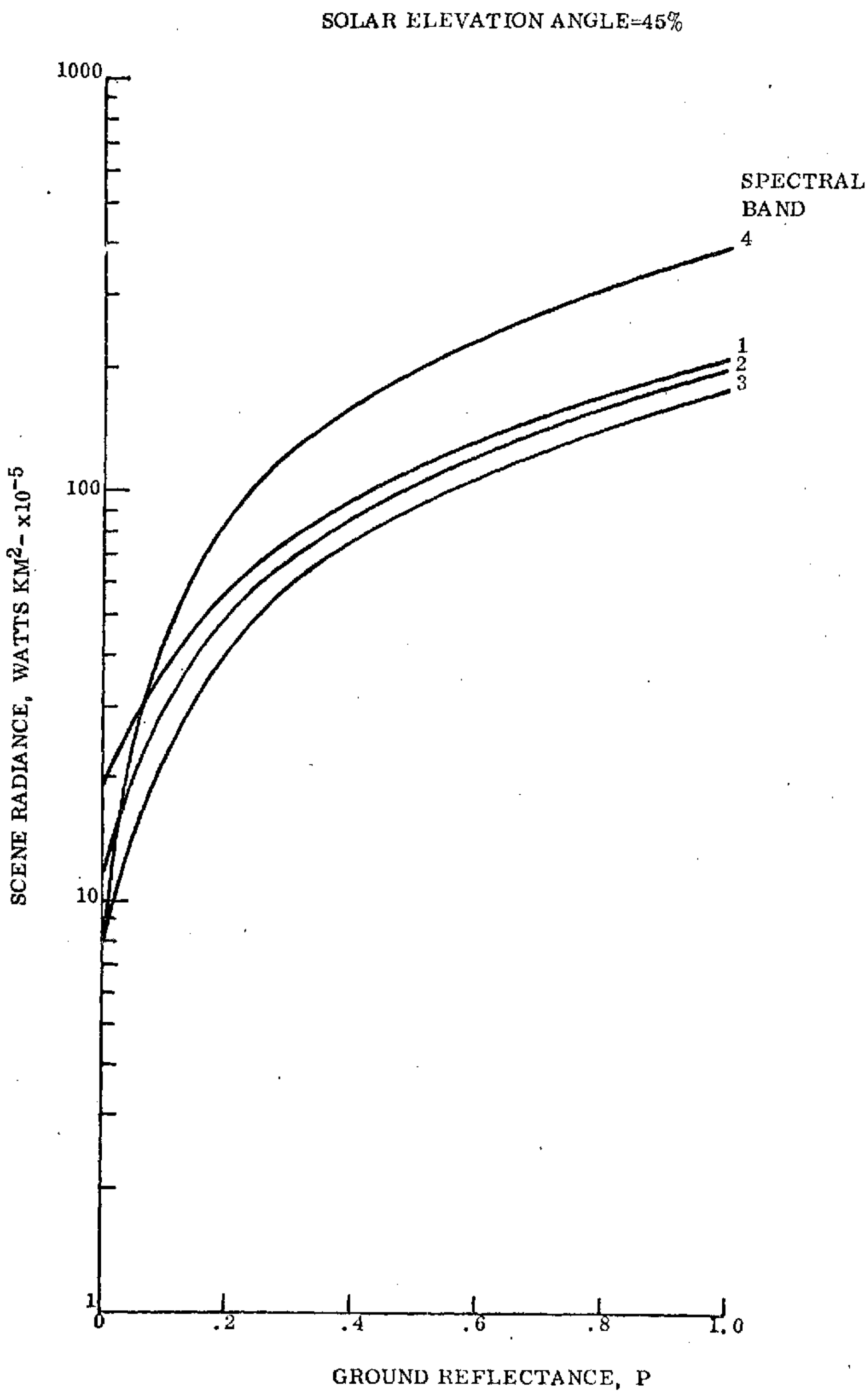


Figure 2-2. KRINOV Data

Band 7 Considerations. Recent ERIM User Requirements indicates a need for thermal data from $250^{\circ}K$ to $340^{\circ}K$ with $0.25^{\circ}K$ resolution at maximum radiance and $1.0^{\circ}K$ at minimum radiance. The NASA specification for EOS is $220^{\circ}K$ to $320^{\circ}K$ with $0.5^{\circ}K$ resolution at $300^{\circ}K$. The key driving factor for the EOS baseline is the quantization level available. A brief study concluded that it is not realistic to have different quantization levels in the seven spectral bands as this is difficult and expensive to implement. However, using double precision processing or some integral number times the quantization level of the first six bands is a possibility to expand Band 7 dynamic range. Two selectable ranges is another possibility, but there is the problem of DC restoring and calibration to two separate minimum radiances.

Figure 2-3 indicates Band 7 radiance (hence detector voltage) vs. temperature. To obtain $0.5^{\circ}K$ resolution at $300^{\circ}K$, a one-level change (for 7 bit quantization) must be representative of a radiance change of 1.44×10^{-5} watts/cm^2 ster. For 128 levels, the total radiance range that can be accommodated is $128 \times 1.44 \times 10^{-5} = 185 \times 10^{-5}$ watts/cm^2 - ster. A single range from $243^{\circ}K$ to 320° with 1.44×10^{-5} watts/cm^2 - ster. radiance steps is recommended. This will provide a Band 7 temperature resolution as shown in Figure 2-4.

S/N and Dynamic Range. The upper part of Table 2-4 indicates the dynamic range resulting from the minimum and maximum radiance levels and S/N specifications issued by GSFC to the instrument contractors. The lower part of the table is a recommended revision to these specifications based on system performance analysis conducted as part of this study. The S/N at minimum radiance are the original NASA specified values. These have been maintained since they tend to equalize the usable signal-to-noise ratio (i.e., after it is reduced by typical contrast ratio at the top of the atmosphere as shown in Table 2-5 across the various spectral bands. The recommended minimum/maximum radiances do contain compromises in Bands 1, 2, 4 and 7 as indicated in the table. However, these compromises appear warranted in that the system can be designed with a reasonable

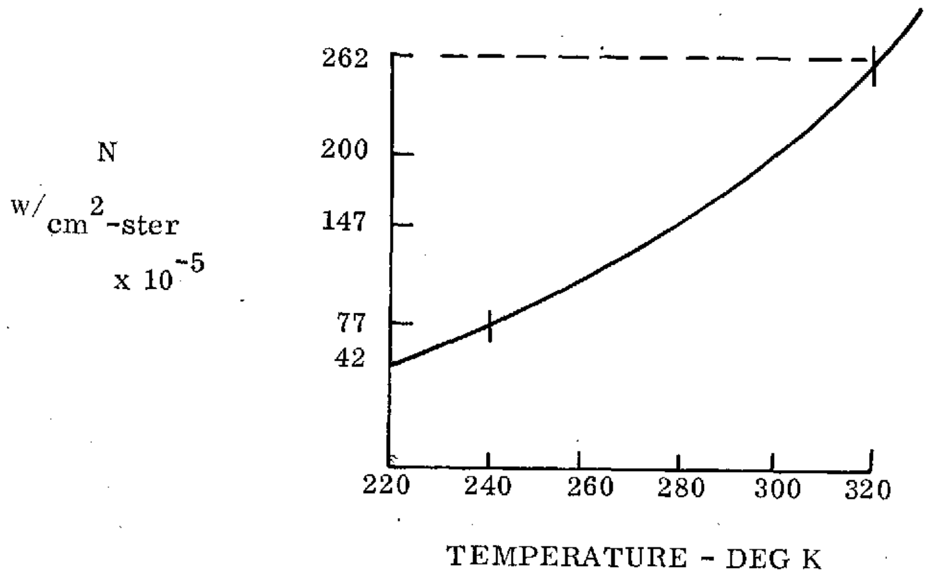


Figure 2-3. Band 7 - Radiance vs. Temperature

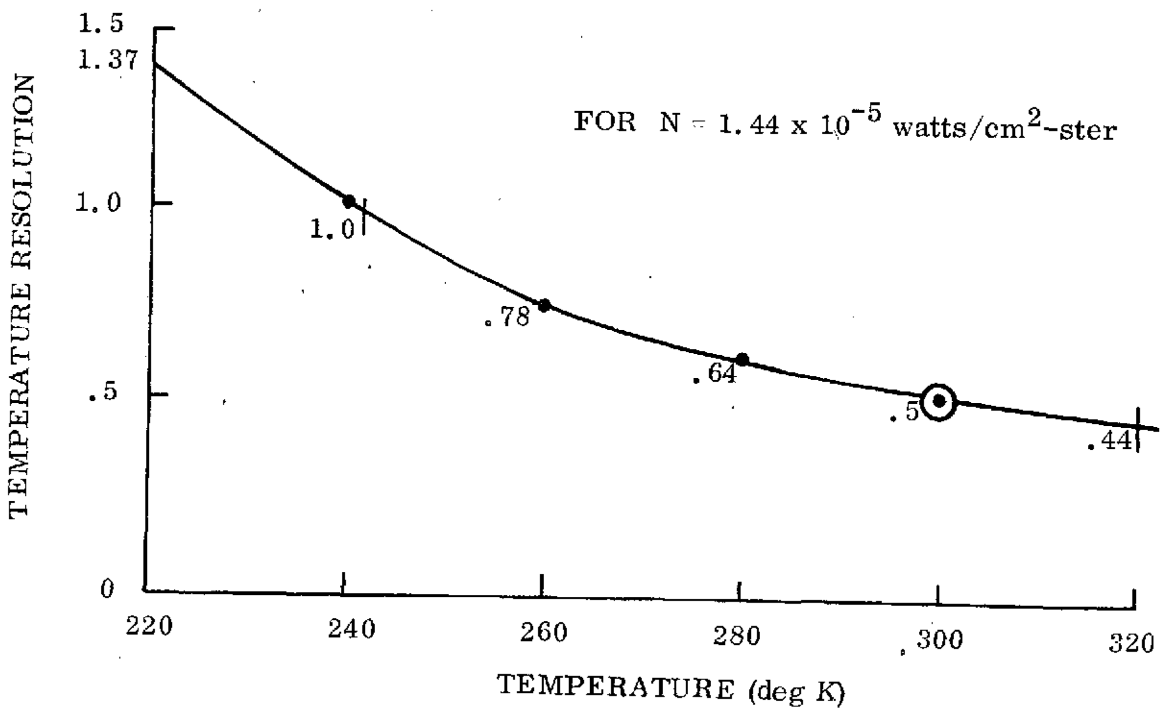


Figure 2-4. Band 7 Temperature Resolution as a Function of Temperature

Table 2-4. Original and Proposed Performance Baseline

Band No.	Spectral Interval	S/N at Minimum Radiance	Maximum Radiance $X10^{-5}$ watts/cm^2-Ster	Maximum Radiance $X10^{-5}$ watts/cm^2-Ster	Dynamic Range
NASA EOS SYSTEM PERFORMANCE					
1	.5 - .6	10	22	363	165
2	.6 - .7	7	19	297	109
3	.7 - .8	5	16	231	72
4	.8 - 1.1	5	30	363	61
5	1.55 - 1.75	5	8	66	41
6	2.1 - 2.35	5	3	33	55
7	10.4 - 12.6	NE $\Delta T = .5^\circ K$ at $300^\circ K$	42 ($220^\circ K$)	304 ($320^\circ K$)	184
RECOMMENDED EOS SYSTEM PERFORMANCE BASELINE					
1	Same as above	10	22 H	320	128
2		7	16 H	292	128
3		5	9	230 L	128
4		5	14	358 L	128
5		5	5	100	100
6		5	3	50	83
7		NE $\Delta T = .5^\circ K$ at $300^\circ K$	77 ($243^\circ K$)	262 ($320^\circ K$) L	128

H = too high for reflectance of $\rho = .02$
L = too low for max radiance required

quantization level (7 bits) and without the design and operational complications of multiple gain modes or range compression.

References:

(1) Krinov, E. L., "Spectral Reflectance Properties of Natural Formations," Laboratoria Aerometochov, AKAD, MAVK, SSSR, Moscow 1947.

(2) "Spectral Radiance of Sky and Terrain at Wavelengths Between 1 and 20 Microns. Sky Measurement," Bell, Eisner, Young & Oetsen, Journal of Optical Society of America, Volume 50, No. 12, December 1960.

(3) Interim Oral Presentation, "Multispectral Data Application Evaluation," Contract NAS 9-13386, June 1974.

Table 2-5. SNR Reduction Caused By Scene Contrast & Instrument MTF

Band	$\Delta\lambda$ (μm)	Minimum Radiance (Watts/cm^2 Ω)	Spec SNR (1)	Contr. Ratio (2)	$(SNR)_C$ (3)	MTF @ f_c (4)	$(SNR)f_c$ (5)	Minimum Radiance (Watts/cm^2 Ω)	$(SNR)f_c$
1	.5 - .6	22	10	1.89	4.7	.60	2.8	363	46
2	.6 - .7	19	7	2.70	4.4	.60	2.6	297	41
3	.7 - .8	16	5	6.75	4.2	.59	2.5	231	36
4	.8 - 1.1	30	5	15.9	4.7	.58	2.7	363	33
5	1.55-1.75	8	5	2.78	3.2	.55	1.8	66	15
6	2.10-2.35	3	5	2.30	2.9	.54	1.5	33	17
7	10.4-12.6	300°K	0.5°K	6.0	0.6°K	.66	0.9°K		

NOTES:

(1) Spec. SNR is defined for low spatial frequency (unity MTF) and infinite contrast ratio.

(2) Contrast ratio are typical values for given spectral bands at the top of the atmosphere assuming an infinite contrast ratio at the ground

$$C = \frac{\text{Ground Radiance + Sky Radiance}}{\text{Sky Radiance}}$$

These are optimistic values since ground contrast will never be infinite.

(3) $(SNR)_c$ is signal-to-noise ratio as reduced by typical contrast ratio.

(4) MTF is typical values for total instrument at the critical frequency

$$f_c = 1/2 \text{ IFOV}$$

(5) $(SNR)_{fc}$ is signal-to-noise ratio as reduced by typical contrast and MTF. This value is analog output of sensor and does not include sampling effect on MTF.

2.2.2.3 Geometric Accuracy

Geometric accuracy is the most critical system output product performance parameter. The baseline TM geometric mapping error budget required to meet these system accuracies, is shown in Table 2-6. This represents a large apportionment of the total system requirement. Based on preliminary discussions and analysis by the instrument contractor, this budget is expected to be difficult, but possible to achieve.

Table 2-6. Geometric Mapping Error Budget Baseline

Line Scanner	TM
Start of Scan Stability	3 μrad
*Along Scan Positional Accuracy (repeatability along entire scan including optical distortions)	4 μrad
Across Scan Non-linearity (⊥1 to scan line)	4 μrad
Detector Position	
Placement (to a specific location)	.1 IFOV
Knowledge	0.5 IFOV
Number of Detectors in Bands 1-6	16
Number of Detectors in Band 7	3

* Variations from this accuracy which are linear are acceptable.

2.2.2.4 Radiometric Error Budget

Table 2-7 is the TM Radiometric Error Budget which has been derived from the total system accuracy requirements.

The maximum allowable errors are specified at both minimum and maximum radiance and can be linearly interpolated for radiance inputs between these values. The ratio of maximum to minimum radiance is assumed to be 33:1. Both fixed and temporal errors are specified. This budget appears realistic, but verification by the instrument contractor is still required.

Table 2-7. Instrument Radiometric Error Budget

Temporal Errors

1. Calibration Source Sability - long-term stability in on-board calibration source spectral radiance (range of 8% at minimum radiance to 0.5% at maximum radiance).
2. Transfer Characteristic Gain or Responsibility ($\frac{\delta R}{R}$) - variation in sensor gain or responsivity between calibration updates for each linear segment (0.3% gain variation).
3. Transfer Characteristic Offset ($\frac{\delta DK}{V_{DK}}$) - variation in dark current or voltage between calibration updates.

$$\frac{\delta DK}{} = k_{DK} \frac{V_{min}}{V_{DK}} - 1$$

$k_{DK} = .05$ = allowable radiance error at minimum radiance input due to offset variation

Note: Above limits on transfer characteristic errors can be adjusted; however, errors should not exceed the following:

Maximum Radiance

$$\sqrt{\left(\frac{\delta L}{V}\right)^2 + \left(\frac{\delta R}{R}\right)^2 + \left(\frac{\delta DK / DK}{(V_{max} / V_{DK}) - 1}\right)^2} \leq 0.0005$$

Minimum Radiance

$$\sqrt{\left(\frac{\delta L}{V}\right)^2 + \left(\frac{\delta R}{R}\right)^2 + \left(\frac{\delta DK / V_{DK}}{(V_{min} / V_{DK})^{-1}}\right)^2} \leq 0.08$$

Fixed Errors

1. Transfer Characteristic Linearity ($\frac{\delta L}{V}$) - Deviation of approximate linear characteristic from exact transfer characteristic over a given linear segment (range of 5% at minimum radiance to 0.3% at maximum radiance). Required number of linear segments used in the approximation is selected based upon this requirement.
2. Spectral: Band-to-Band - inaccuracy in knowledge of spectral band definition and calibration source spectral radiance (1.6% at any radiance input).
3. Spatial: Across field-of-view - error in knowledge of irradiance at instrument focal plane as a function of field angle (0.8% at any radiance input).

Note: Calibration source uniformity should be optimized to minimize error due to shift of irradiance at focal plane.

2.2.3 INTERFACE REQUIREMENTS

The instrument interface requirements are defined in Appendix A.

2.3 EXTRAPOLATED INSTRUMENT DESIGNS

The thematic mapper mission parameters defined below were used as a baseline for instrument design and performance comparison. The updated instrument point designs for each of the contractors (as defined in Section 2.1) have been extrapolated to meet the baseline performance requirements. In this extrapolation, instrument parameters such as clear aperture, optical efficiency, scan efficiency, detector responsivity, detector area, and detector and preamplifier noise per unit bandwidth were used as defined in the updated point designs. A summary of the optics and detector parameters is given in Table 2-8.

Normalized Mission Parameters - Thematic Mapper

Given	
Altitude - KM	775
Swath Width - KM	185.3
IFOV - μrad Band 1 to 6	35
Band 7	140
Detectors/strip - Band 1 to 6	16
Band 7	4

Table 2-8. Instrument Optical/Detector Parameters

Band #	Optical Efficiency			Detector Responsivity (amps/watt) or D* (cm $Hz^{1/2}$/watt)			Clear Aperture (cm^2)			Detector Area ($cm^2 \times 10^{-5}$)		
	G-Te	HAC	HRC	G-Te	HAC	HRC	G-Te	HAC	HRC	G-Te	HAC	HRC
1	.310	.71	.343	.27	.27	.26	625	990	950	7.6	5.2	TBD
2	.289	.71	.336	.35	.34	.34				7.6	5.2	
3	.328	.71	.332	.42	.39	.44				7.6	5.2	
4	.328	.71	.346	.25	.24	.26				7.6	5.2	
5	.420	.56	.400	.62	.65	8×10^{11}				7.6	1.0	1.1x2.1 mil
6	.451	.56	.414	.81	.89	2×10^{11}				7.6	1.0	1.1x2.7 mil
7	.338	.54	.540	2×10^{10}	1.5×10^{10}	2×10^{10}				5.6	15.9	4x4 mil

A set of basic performance equations were developed for the mechanical scanner and were used for extrapolating the performance of each of the instruments to the common baseline requirements. These expressions are in general as shown in the GSFC Working Group Report No. NASA SP-335, "Advanced Scanners and Imaging Systems for Earth Observations". Some modifications have been made to include the effect of FET voltage noise in the expression for photodiode noise. These equations are given in Table 2-9.

The extrapolated instrument scan parameters as derived for the baseline are shown in Table 2-10. Using the minimum radiance levels as defined by the GSFC Specifications, the minimum power incident on a detector element, the S/N, and the noise equivalent power for each spectral band and each instrument were determined. These results are shown in Table 2-11.

Based upon these results some general observations and conclusions can be made.

1. The Hughes S/N values are higher in Bands 1-4, where each of the contractors are using Silicon detectors. This is due to the larger clear aperture and higher optical transmission efficiency obtained in the Hughes instrument. In general, an object plane scanner will provide a more efficient optical system than an image plane scanner for a given instrument size. The Honeywell Spectral Separation Technique, which provides inherent spectral band registration for bands 1-6 has a poor optical efficiency due to relay optic and fiber optic losses.
2. The NEP of the Te Silicon detectors is better because the detectors and preamps are cooled to 200°K. This compensates partically for the poorer Te optical efficiency. Cooling essentially reduces dark current noise, detector shot current noise, FET current noise, and FET load resistor thermal noise. FET voltage noise and detector responsivity are unaffected by cooling.
3. Te has a lower scan efficiency which reflects in a higher signal frequency and data rate (approximately 20% higher). The higher signal frequency increases the noise bandwidth and results in a S/N penalty.

Table 2-9. Basic Design Equations

1. Signal Current (Photomultiplier, PIN Photodiode)

$$I_s = (N_1' - N_2')\ F\ (X,\alpha)\ \alpha^2\ A_o\ \tau_o\ R\ \tau_D$$

- N_1' = apparent radiance associated with target, watts/cm^2-Ster
- N_2' = apparent radiance associated with background, watts/cm^2-Ster
- $F(X,\alpha)$ = Spatial frequency response
- α = instantaneous field of view, radians
- A_o = Collecting aperture area, cm^2
- τ_o = Optical efficiency
- R = mean detector responsivity over the spectral bandpass, amps/watt
- τ_D = detector element collection efficiency

2. Noise Current

Photomultiplier (Photoelectric/Shot Noise)

$$I_n = \left[2\ N_2'\ \alpha^2\ A_o\ \tau_o\ R\ \tau_D\ K_\sigma\ \epsilon\ \Delta f_n\right]^{1/2}$$

- K_σ = electron multiplier relative noise
- ϵ = electronic charge, 1.6×10^{-19}
- Δf_n = noise bandwidth, Hz

PIN Photodiode

$$I_n = \left[(i_{n_s}^{\ 2} + i_{n_d}^{\ 2} + i_{n_p}^{\ 2} + 4\ KT/R_L)\ \Delta f_n + (V_A\ (2\pi\Delta f_s)\ (C_i + C_f)^2 \Delta f_n\right]^{1/2}$$

- i_{n_s} = signal shot current noise, amps/$Hz^{1/2}$
- i_{n_d} = dark current noise, amps/$Hz^{1/2}$
- i_{n_p} = preamp noise current, amps/$Hz^{1/2}$
- $4\ KT/R_L$ = Load resistor thermal noise current
- V_A = FET voltage noise density
- Δf_s = signal bandwidth, Hz
- C_i = input capacitance
- C_f = feedband capacitance

3. Signal Power (Photoconductor)

$$P = (N_1' - N_2')\ F\ (X,\alpha)\alpha^2\ A_o\ \tau_o\ \tau_D$$

4. Noise Equivalent Power (Photoconductor)

$$NEP = \frac{\sqrt{A_d\ f_n}}{D^*}\ NF$$

- A_d = detector area, cm^2
- D^* = detector detectivity, cm $Hz^{1/2}$/watt
- NF = preamp noise figure

Detector Area

$$A_d = \alpha^2\ D_o^{\ 2}\ (f/n_o)^2\ \tau_D$$

- D_o = optic diamter, cm

5. Signal Bandwidth

$$\Delta f_s = \frac{(V/h)\ \theta}{2K_s\ \eta\ \alpha^2}$$

- V = satellite velocity, Km/sec
- h = satellite altitude, Km
- θ = cross-track swath width, radians
- K_s = scan efficiency
- η = Number of Detectors per band

Noise Bandwidth

$$\Delta f_n = K_f\ \Delta f_s$$

- K_f = ratio of filter effective noise bandwidth to information bandwidth.

Table 2-11. Extrapolated Instrument Scan Parameters

	HAC	HRC	TE
Cone Angle	N/A	16.38^{o}	N/A
Strip Width - m	434	471	434
Strip Length - Km	185.3	190.8	185.3
Ground IFOV - m - Bands 1 to 6	27.1	29.5	27.1
- m - Band 7	108.5	117.9	108.5
Total Time/strip - m sec	65.2	70.8	65.2
Scan Efficiency - %	80	80	73.7
Active Time/strip - m sec	52.2	56.7	48.0
Number of Pixels/line - Bands 1-6	6840	6744	6840
Band 7	1710	1686	1710
Max. dwell time - Bands 1 to 6	7.6	8.4	7.0
per element Band 7	32.8	33.6	28.1
Signal frequency per detector			
Bands 1 to 6 KHz	65.8	59.6	71.4
Band 7 - KHz	16.5	14.9	17.8
(1 sample per pixel)			
Data Rate (7 bit Quantization, Mbps)	89.4	80.9	97.0

4. The performance comparison in Bands 5 and 6 is inconclusive. The Te and Hughes Instruments use photovoltaic indium antimonde detectors cooled to 100^{o}K while Honeywell uses mercury cadmium telluride cooled to 190^{o}K. The variation in the detector NEP from Band 5 to Band 6 in the Honeywell Instrument and between the Te and Hughes Instruments in both bands is inconsistent and cannot be explained. However, it can be concluded that the performance in Band 6 for any instrument is marginal at best.

5. All instruments use HgCdTe detectors cooled to 100^{o}K for Band 7. The variation in NEP is due to the differences in assumed value of detectivity, detector area, and noise bandwidth. All instruments exceed the requirements of 0.5^{o}K for minimum detectable temperature difference. As discussed in Section 2.2.2.2 the heat will be limited to 0.5^{o}K by quantization and dynamic range limitations rather than instrument performance.

Table 2-11. Instrument Performance Parameters

Band #	P_{min}(wattsx10^{-12})			Signal-to-Noise			NEP(wattsx10^{-12})		
	G-Te	HAC	HRC	G-Te	HAC	HRC	G-Te	HAC	HRC
1	52.2	188	87.6	13.6	22.8	15.1	3.84	8.25	5.80
2	41.9	163	74.0	14.2	25.6	16.4	2.95	6.37	4.51
3	40.0	137	61.7	15.8	25.2	18.1	2.53	5.44	3.41
4	75.2	258	118	16.3	27.4	20.5	4.61	9.42	5.76
5	25.7	54.2	37.1	13.8	7.0	25.4	1.86	7.74	1.46
6	10.3	20.3	14.6	7.8	3.5	2.9	1.32	5.80	5.03
7	300°K	300°K	300°K	.42°K	.36°K	.22°K	50	108	62

2.4 EVALUATION OF EXTRAPOLATED DESIGNS

2.4.1 PERFORMANCE DEVIATIONS

The instrument designs as extrapolated to the baseline requirements nearly meet or exceed the specified S/N in Bands 1 thru 5 as shown in Table 2-12. The significant exception is Band 6 where all but TE fail to meet the performance specification. The minimum radiance values used in the evaluation of the extrapolated designs were those specified by GSFC. Using those recommended in Table 2-3 would result in somewhat lower S/N performance in Bands 2 thru 6. All designs exceed specified performance in Band 7.

Table 2-12. Signal-Noise Ratio Performance Deviation

	Minimum Radiance (Watts/cm^2 Ster)	S/N			S/N Spec
		Hughes	Honeywell	Te-Gulton	
1	22	22.8	15.1	13.6	10
2	16	21.6	13.8	12.0	7
3	9	14.1	10.2	8.9	5
4	14	12.8	9.6	7.6	5
5	5	4.4	15.9	8.6	5
6	3	3.5	2.9	7.8	5

The foregoing considers only deviations in S/N performance, where performance extrapolations could conveniently be made and instrument contractors had generated data in response to the NASA spec. Time did not permit response from all instrument contractors to the additional geometric and radiometric performance parameters developed during this study. Hughes has indicated that all radiometric parameters can be met with the exception of the 0.5% stability for the on-board calibration source at maximum radiance. Complete responses from all instrument contractors are needed to determine where overall performance can reasonably be met and whether any reallocation of error budgets is appropriate.

2.4.2 IMPLEMENTATION COST DELTAS

2.4.2.1 Spacecraft Interfacing

Mechanical. The relative design detail of the candidates deters a one-on-one evaluation between contractors. Hughes concerned themselves with physical design parameters and produced a fairly optimized package for size and weight in some detail. Te generated a parametric study and can produce an instrument from about 2/3 size to full size, depending on system requirements (montly S/N at a particular altitude and ground resolution). All meet the original EOS specifications on size (84" x 36" Dia.) and weight (600 pounds).

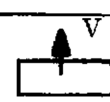
For equivalent performance, the object plane scanner should always be more compact and lighter since it uses a full telescope and only has to image the instantaneous field of view throughout the instrument. The image plane scanners (both conical and linear) have to size and weight scale around a fixed scan wheel size since scan wheel redesigns represent major efforts.

The implementation cost deltas for the alternate thematic mapper designs are a function of the instrument size, orientation and weight which impact the launch vehicle selection for EOS. These critical parameters for the alternate designs are summarized in Table 2-13. Both the Honeywell and Te versions can be size and weight optimized to make them more compatible with a Delta launched mission.

Table 2-13. TM Critical Parameters (Mechanical IF)

Design	Size	Weight	Orientation
Honeywell	72" L x 36" D	600	Either
Hughes (a)	67" x 36" x 20"	330	$L \perp V$
(b)	42" x 36" x 36"	320	$L \parallel V$
Te Gulton	84" x 36" x 38"	598	$L \perp V$

$L \parallel V$



$L \perp V$

V = Velocity Vector
L = Long Diversion

A summary of the critical parameters for the alternate launch vehicles under consideration for EOS is included in Table 2-14. These parameters include launch vehicle cost, total allowable payload instrument weight, shroud diameter and instrument budget, and shroud cylindrical length and TM budget.

Table 2-14. Launch Vehicle Critical Parameters (Mechanical I/F)

Launch Vehicle	Cost (M$)	Allowable Payload Inst Wt (Lbs)	Cross Track Length Total In.	Cross Track Length TM Budget	In Track Length Inst. Total In.	In Track Length TM Budget
Delta 2910	6.0M	675 Lbs	86	74	108	42
Delta 3910	8.0M	1000 Lbs	86	74	108 (1)	42 (1)
Titan IIIB NUS	12.2M	1300 Lbs	106	74+	144+	72+

(1) Elongated shroud may be considered for Delta 3910 (increase of 36" appears feasible).

If either envelope or weight of the candidate instrument forces selection of the next larger launch vehicle, the cost impact is either $2M or $6.2M. If the 675 pound payload weight is distributed roughly between TM and HRPI, then each instrument should target for 320 pounds, 72" x 36" x 36", instrument perpendicular to spacecraft velocity vector. Since all TM's can orient perpendicular, the HRPI could orient parallel to velocity vector if required.

Instrument contractor responses to the size and weight reduction question indicate that each could build an instrument to fit the delta requirements, but since the resulting performance changes were generated to non-uniform ground rules, the results have not been fully evaluated.

A comparison of the TM critical parameters to the launch vehicle critical parameters allows a classical evaluation of the relative impacts of the point design alternate TM concepts.

a) Impact of Honeywell TM. The unoptimized weight of the Honeywell TM of 600 lbs precludes launching a two-instrument payload (TM and HRPI) on Delta 2910.

The two-instrument payload could be launch on Delta 3910 by limiting the HRPI weight to approximately 400 lbs or eliminating a wide band tape recorder. The cost for the two-instrument capability for EOS-B, using the Honeywell TM, therefore becomes the $2M cost difference between the Delta 2910 and 3910 launch vehicles.

b) Impact of Hughes TM. Two Hughes TM designs are shown in Table 2-13, each weighing 330 lbs. The 42" x 36" x 36" size instrument is preferred for a Delta spacecraft launch since it is more easily packaged within the 86" shroud envelope diameter. This Hughes TM is the only current design compatible with a TM and HRPI Delta 2910 launch and, only if a lightweight HRPI (330 lbs) is assumed.

c) Impact of Te-Gulton TM. All three critical parameters of the current unoptimized Te-Gulton TM design defined in Table 2-13 cause a problem in incorporating the instrument on a Delta launch. The most severe restrictions are the size and orientation requirement that makes it impossible to mount the instrument in the 86 inch diamter Delta shroud. Therefore, the cost penalty of using the baseline Te-Gulton TM is the delta cost between the Delta 2910 and Titan IIIB NUS, or $6.2M.

d) Summary of Cost Impacts. The cost impacts on launch vehicle selection are summarized in Table 2-15.

Table 2-15. Alternate TM Impact on Launch Vehicle Selection

Alternate TM	Launch Vehicle Required	Launch Vehicle Cost - M$	Relative Cost Impact
Honeywell	Delta 3910 (Marginal weight)	8.0	+2.0M
Hughes	Delta 2910 (very marginal weight)	6.0	----
	Delta 3910 (ample weight margin)	8.0	+2.0M
Te-Gulton	Titan IIIB NUS (ample weight margin)	12.2	+6.2M

The table shows that a lighter weight, more compact unit is preferred since it allows the use of the lower cost Delta launch vehicle which, in turn, provides a cost savings of from 2.0 to 6.2M dollars.

Electrical. The command, telemetry, clock and timecode hardware interfaces with the instruments are not affected by the candidate design approaches. The data bus distribution technique and the sizing of the spacecraft command and telemetry systems permit variation in servicing requirements without impact. This is not true, however, with power.

The Westinghouse HRPI and Te-Gulton TM were arbitrarily chosen in sizing the EOS power system. Variation in input power requirements have a direct effect on this sizing in the area of solar array and battery capacity. Comparison of the various TM designs are as follows:

TM	Avg. Pwr.	Power	Array Size	Amp-Hr/Battery
Te-Gulton	37 W	Reference	Reference	Reference
Hughes	45 W	+ 8	+ 2 Ft^2	0
Honeywell	200 W	+ 163	+ 35 Ft^2	+6

There is no significant impact of any design except the Honeywell baseline approach which has not been optimized for minimum power. If the 200W average power is an actual requirement for the Honeywell design, an extra solar array panel would be required at a cost of 4K/ft^2. Also, the boost converter and battery capacity would have to be increased (relatively small cost impact of less than $10K).

2.4.2.2 Central Data Processing Facility: Instrument Implementation Cost

The delta cost impact of the various instrument approaches on the ground system is summarized in Table 2-16, for the Thematic Mapper. These delta costs assume that the ground system is initially designed to accommodate the particular instrument. The modularized ground system design approach resulted in a hardware configuration which is fairly insensitive to the differences in the candidate instruments.

The major cost impact parameters for the thematic mapper are the scan philosophy and the band-to-band misregistration. The delta implementation costs due to these parameters are shown in Table 2-16, with the TE Scanner selected as a baseline. The Hughes oscillating mirror scanning approach has a +$40K cost impact due to the back and forth scan cycle. The Honeywell Conical scan has a +$300K impact due to the additional storage required to linearize the data. The Honeywell TM also has a $40K cost increase to achieve band-to-band registration due to the offset of band 7 perpendicular to the scan direction, rather than along the scan direction. Parameters such as scan linearity, data formats, radiometric banding, relative radiometric accuracy, etc., do not impact ground system cost providing the TM instrument manufacturers meet the previously discussed design specifications for accuracy and stability.

Table 2-16. Impact of Thematic Mapper Instruments on Ground System Cost

	Scan Technique	Band-to-Band Registration	Total
Hughes	+$40K	0	+$40K
Honeywell	+$300K	+$40K	+$340K
Te-Gulton	Reference	Reference	Reference

2.4.2.3 Low Cost Readout Stations: Instrument Implementation Cost

The EOS-A spacecraft will transmit a portion of the instrument data directly to users at many Low Cost Readout Stations. This data will be derived from the TM or HRPI, processed by the spacecraft compactor and then transmitted. Just as in the CDPF, the various instrument approaches have impact on the cost of the LCRS. Table 2-17 summarizes the cost impacts for the TM. It is assumed that the LCRS has been initially designed to accommodate a particular instrument implementation approach.

The major cost impact parameters for the TM are scan philosophy and band-to-band misregistration. The delta costs to these parameters are given in Table 2-17 assuming the Te scanner as a reference. The Hughes scanner has +$25K cost delta due to the storage requirement needed to correct for the back and forth scan. The Honeywell data will not be linearized (i.e., it will be corrected but remain in conical format) in the LCRS hence there is no cost impact for the Honeywell scanner. There is, however, a $10K cost increase to achieve band-to-band registration in the conical scanner since band 7 is offset from the other 6 bands.

Other parameters such as scan linearity, data format, radiometric banding and relative radiometric accuracy do not impact LCRS cost.

Table 2-17. Impact of TM Instrument Approach on LCRS

Approach	Scan Technique	B-B Misregistration
Te	Reference	Reference
Honeywell	0	+$10K
Hughes	+ $25K	0

2.4.3 DESIGN FLEXIBILITY

The instrument systems requirements are, of course, quite dynamic and change within limited bounds as a function of the present mission model. The thematic mapper is the next generation contiguous synoptic coverage instrument and should be flexible enough to meet various mission scenarios. Several parameters have been selected as key to the instruments flexibility for various missions. Small deviations to certain of these parameters (numbers of spectral bands, increased S/N, and decreased IFOV) seemed equally feasible in each candidate design although relative cost, size, weight and performance impact may change slightly.

2.4.3.1 Swath Increase

Increasing the swath width to decrease access time affects each scan technique differently. Note that all designs will require additional ground processing to correct errors at the swath extremes.

The object plane scanner can accommodate increased swath rather easily. The scan mirror size will grow somewhat to fill the aperture, and the scan drives will require some redesign.

The linear image plane scanner will require a larger pointing mirror and primary. The roof wheel design will be maintained, but scan efficiency will be reduced and noise bandwidth will be increased. The number of detectors required will probably double so that the power required will increase significantly.

A conical image plane scanner is also amenable to increased swath. Two techniques are available: (1) either increase the cone angle (longer path length, more atmosphere); or (2) have fewer but longer arc segments and resultant increased geometric corrections. Both methods will require larger optics than the present system.

2.4.3.2 Altitude Changes

Spacecraft altitude is one of the most critical instrument design drivers and the TM may have to be capable of accommodating several alternatives for various missions. For example, a TM and MSS instrument combination might be in a 500 nmi orbit (to accommodate the MSS) while a TM and HRPI combination could be in a 418 nmi orbit to provide HRPI access time. For a fixed detector IFOV, altitude changes require scan rate adjustments, swath width changes, and intra-detector spacing changes in all instrument design approaches. In addition to all of the above, the conical scanner requires cone angle adjustment and resultant optics changes.

2.4.3.3 Offset Pointing

The capability to offset point the TM for the TM and HRPI mission has been deleted, but for a future mission scenario that continues the MSS on early flights and has a TM onboard as the experimental next-generation contiguous synoptic coverage instrument, offset pointing capability may be revived as a requirement because of the more frequent access possible and the cloud avoidance capability.

The TE design can most easily accommodate offset pointing, requiring only a modest growth in the pointing mirror. The Hughes design is next in line, but their present baseline is that the entire instrument must be rotated as a unit for offset angles of approximately $\pm 30^{\circ}$. The Honeywell design also requires rotating the instrument, but the image geometry becomes very unwieldy.

2.4.3.4 Orientation

Accommodating any of the proposed instrument orientations has not proven to be a problem for the modularized instrument bays, but it is worth noting that the conical scanner can easily accommodate orientation either parallel or perpendicular to the velocity vector which may prove helpful when other payload complements are considered. Hughes has two baseline designs that have opposite orientations.

2.4.4 DESIGN RISKS

The variation in depth of design detail among the contractors would result in apparent inconsistencies in discussing design risks, thus, the following discussion deals with the basic design approaches. Inconsequential instrument design and documentation inconsistencies and/or inappropriate assumptions are not discussed.

2.4.4.1 Conical Scanner.

The major area of concern is the thermal design and dependence on heat pipes and radiant coolers. Cooler freezeovers and emissivity changes drastically changes the thermal control capability, as does heat pipe deterioration. The point study design for the conical scanner requires 100 to 200% more power during operating periods than the other candidates and requires it for full orbital operation. However, this design was the only candidate that didn't include a command to adjust instrument focus in the event of ambient temperature change or optical shifts. In addition, the design includes three focal planes which makes focus adjust difficult.

2.4.4.2 Image Plane Scanner

The major design risk with this scanner is the two-way scan stability and repeatability. This should be resolved during the breadboard program.

2.4.4.3 Linear Scanner

Detector cooling is used to obtain low noise. Very little information is available yet on the problem of cooling. The design should not be optimized on cooled detectors until cooling feasibility is established since the increased aperture required if uncooled detectors must be used will constitute a major design change.

A second design risk is the tight manufacturing and alignment tolerances. This is not unique to the linear scanner of course, but tolerances seem to be more stringent in the roof wheel design. Launch loads and deflection sets are not yet included in the error budgets. The 20 mil nominal free-space gap between roof apices is a constant cause for concern during ground testing.

2.5 INSTRUMENT DESIGN RECOMMENDATIONS

2.5.1 SPECTRAL SEPARATION TECHNIQUE

Multilayer interference filters and spatial separation is preferred to prism monochromators for spectral band determination. The interference filters provide a great deal of flexibility in design. They can provide high peak transmission with high roll-off transmission curves. They provide the capability of overlapping channels if required as user requirements begin to dictate bands based on extraction signatures. Packaging is simple and fewer optical elements are generally needed. The prism approach also pays a penalty in optical efficiency. The band-to-band registration that is obtained as a result can be accomplished via tight alignments and good design and/or easily and cheaply provided in ground processing.

2.5.2 IN-FLIGHT CALIBRATION

The radiometric requirements in Section 2.2 are aimed toward providing calibrated data to the user. To this end, the following is recommended:

a) Each instrument should be capable of using the sun for an absolute calibration source on command.

b) Deletion of the electronic calibration if such a calibration introduces any additional noise sources. Operationally, an electronic calibration is not mandatory. Its utilization is mainly in instrument troubleshooting.

e) DC restoration in the instrument. This preserves dynamic range and reduces auxiliary data requirements and processing load on the ground, particularly at the low-cost ground stations.

2.5.3 A/D AND SUB-MUX

In lieu of 100 analog signals being routed to a remote digitizer and the resulting fragmented specifications and test program, the sampling, A/D conversion and submuxing should be performed within the instrument. The input to the spacecraft multiplexer (which adds auxiliary data and generates the serial data stream) should be serial digital data per band.

2.5.4 NOISE

Dynamic performance analysis indicates that the white noise power spectral density is the critical parameter to consider in relation to data extraction utility. Current S/N specifications constrain only the integral of the noise power spectrum from DC to some band limit frequency. No change is proposed except to make the noise power spectrum available and to specify the S/N at a signal frequency equivalent to $\frac{1}{\text{IFOV}}$ instead of the current practice of $\frac{1}{2\ \text{IFOV}}$.

2.5.5 BAND-TO-BAND REGISTRATION

Band-to-band registration is not much of a correction problem if it is constrained to a range of several hundred pixels maximum in the cross-track direction only. Simple design practices can aid in keeping these offsets low. Band 7 should lead the scan, since it requires fewer pixels storage to register to the other bands. Further, the ratio of the size of the detectors in band 7 should be an integral multiple of the size of the detectors times the number of detectors per band in the other band to simplify data processing.

2.6 PROGRAM RECOMMENDATIONS

Several areas have been identified for additional study at the systems and/or instrument level. Those studies listed as system studies are general in nature and do not necessarily require an instrument contractor.

2.6.1 SYSTEM STUDIES

Detectors. Table 2-18 indicates the detectors and operating temperatures that have been considered by the instrument contractors from time to time. The present baseline for all contractors is silicon in the first four bands. To cool or not to cool, that is the question that still needs resolving. Very little parametric noise data seems to be available. The detector and preamplifier technology efforts in process at the various instrument contractors seems to be uncoordinated and fragmented. It would seem appropriate for NASA to lead this technology area with sufficient money and direction to provide a consistent set of data available to all.

Cost Models. All of the instrument contractors are still working to the criteria of near diffraction limit performance. As we change our ground rules for shuttle launched instruments where weight and size become less of a problem and where dynamic performance response becomes the design criteria, optics will become large low-figure light gatherers. It is recommended that additional instrument cost modeling and cost trades be performed to determine the appropriate direction for instrument designs in the shuttle era which will minimize total system costs.

2.6.2 INSTRUMENT STUDIES

The majority of the instrument studies recommendations are contained in Sections 2.4 and 2.5. Two additional general recommendations are included here:

Band 6. No clear-cut answer has been found for the Band 6 controversy. Available user requirements indicate a need for both Band 5 and 6 data in over half the disciplines. Only the land use classification tasks indicate that either Band 5 or Band 6 is sufficient as the discriminator. The performance prediction for Band 6 data is so poor, however, that its utility is suspect. It seems unlikely that hard user data on acceptable performance will be available in the near future, so the Band 6 controversy should be solved by something more expedient. It is recommended that an assessment be made by each instrument contractor of the costs savings resulting from deletion of this band. A small cost saving would suggest flying the band to experimentall determine its utility. A large cost saving would justify its deletion.

<u>CCD Technology</u>. There should be a continued investigation of CCD technology and its application to remote sensing.

Table 2-18. Detectors

	Band									
	1		2		3		4	5	6	7
HAC	PMT		PMT		PMT		Si	InSb	InSb	HgCdTe
	Si	CCD	Si	CCD	Si	CCD	CCD	$110^{\circ}K$	$110^{\circ}K$	$110^{\circ}K$
HRC	PMT	Si	PMT	Si	PMT	Si	Si	HgCdTe (PC/PV) $190^{\circ}K$	HgCdTe (PC/PV) $190^{\circ}K$	HgCdTe $110^{\circ}K$
G-TE	Si $200^{\circ}K$		Si $200^{\circ}K$		Si $200^{\circ}K$		Si $200^{\circ}K$	InSb $100^{\circ}K$	InSb $100^{\circ}K$	HgCdTe $100^{\circ}K$

SECTION 3.0

HIGH RESOLUTION POINTABLE IMAGER

3.1 POINT DESIGN STUDIES COMPARISON

Four point design studies, with subsequent updates and modifications, were performed for the definition of a High Resolution Pointable Imager (HRPI). The primary difference between the design approaches is the scan technique utilized.

- o Pushbroom Scanner
- o Object Plane Mechanical Scanner
- o Image Plane Conical Scanner
- o Image Plane Linear Scanner

The point designs for the Pushbroom Scanner and Mechanical Scanners were performed at different times and based upon somewhat different requirements. The point design requirements are summarized in Table 3-1. The following paragraphs provide a brief description of each approach with a discussion of the key features. In addition, the basic instrument components which represent significant differences in the approaches are described.

3.1.1 PUSHBROOM SCANNER

The pushbroom scanner consists of integrated self-scanned solid state silicon photodiode arrays comprised of 4800 elements in each of four spectral bands. The radiometer operates in the "pushbroom" mode with cross-track resolution determined by the element size and spacing and along track imaging provided by the spacecraft motion and time sampling of the elements. The center-to-center element spacing in the array is equivalent to 10 meters on the ground and the swath width covered is 48 km. The instantaneous field of view ($\sim 3^{o}$) is pointable (cross-track) in 1^{o} steps $\pm 10^{o}$ and repeatable to within $\pm$ 1 arc second. Subsequent changes in HRPI requirements increased the pointing angle to $\pm 40^{o}$ and relaxed the repeatability to $\pm$ 0.1 degree. The impact of these changes on the instrument design has not been evaluated at this time.

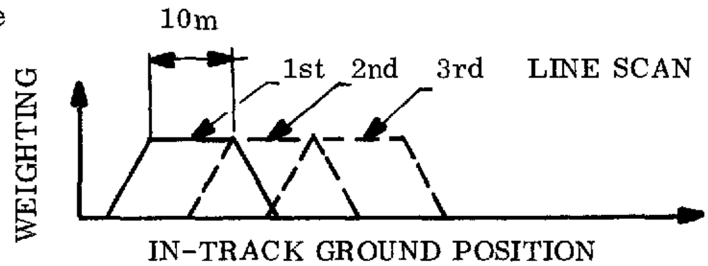
Table 3-1. High Resolution Pointable Imager Point Design Requirements

	PUSHBROOM				MECHANICAL SCANNER			
Altitude, km	914				717			
Swath Width, km	48				46 max, 20 min.			
Ground IFOV, m	10				10			
Offset Pointing	$\pm 10^{o}$				$\pm 40^{o}$			
Weight, kg (lb)	272 (600)				227 (500)			
Size, cm (in)	213 (84) length x 91 (36) dia.				100 (40) x 125 (49) x 130 (51) 90 (36) x 90(36) x 180 (72)			
Power, watts	100 orbital avg.				100 + 50 for heat			
Registration	----				25% IFOV			
Scan Accuracy	< 50% IFOV				30% IFOV			
Radiometric Accuracy	< 5%				< 2% full scale			
Spectral Bands	.5-.6	.6-.7	.7-.8	.8-1.1	.5-.6	.6-.7	.7-.8	.8-1.1
Min. Radiance, mw/cm^2/ ster	.12	.10	.08	.16	.22	.19	.16	.30
Min. SNR	----	----	----	----	5	5	5	5

The optical system consists of a pointing mirror, a telescope, and a prism assembly which provides inherent spatial registration of the four spectral bands. The telescope is a catadioptric form consisting of a spherical primary and secondary mirror and several refractive correction lenses.

The point design study was based upon the utilization of a two-row staggered element configuration for the array with 96 elements per chip. This configuration minimizes the optic focal length for a given element size, thereby maximizing the S/N. The ground geometry for such an array is as shown in Figure 3-1. More recent designs include an array with 128 elements per chip with the same configuration and a non-staggered (in-line) chip with the same element spacing. The advantage of the in-line geometry is that the pixels are not staggered relieving some ground processing and the susceptibility to spacecraft attitude changes. The disadvantages are that at each chip edge 2 or 3 elements will be missing, and the along scan cross-talk, at longer wavelengths, will be greater.

With the staggered array configuration, the information required to form one scan line is contained in three scan data sets, since the coverage in the in-track direction is a weighted function as shown in the figure.



A complete array for one spectral band is comprised of five groups of 10 chips per group and 96 elements per chip. An element is comprised of a photodiode, a gain stage and readout and reset switches.

The data outputs from the 96 elements on a chip are multiplexed onto four data lines (Figure 3-2). Two data lines are provided for each row of the two-row staggered array. The 10 chip outputs are additionally multiplexed into group outputs on four lines, resulting in 20 lines per array or a total of 80 data lines. The readout is not sequential and is dependent upon array configuration. However, it should be noted that data taken during a readout cycle contains data from every other scan element cross-track and along track and hence data from the n^{th}, n + 1, and n+2 cycle are required to reproduce a single complete cross-track scan line.

Signal processing on each of the 80 data lines consists of:

(1) current to voltage conversion;
(2) offset subtraction;
(3) gain change amplifier;
(4) sample and hold

Each of the 80 data outputs is in a "sample and hold" signal format.

Some of the critical array characteristics which affect radiometric performance and calibration requirements include:

(a) Element-to-element responsivity variation on a chip may be 7-10% standard deviation from the mean with max/min ratio approximately 1.4:1 (end elements as much as 3:1). Thus, each of the 19,200 elements requires a calibration curve. The change in responsivity is approximately 10% for a temperature change from 25°C to 0°C.

(b) Element response linearity is approximately 1-2% of the full scale over an input irradiance range of 600:1 to 1000:1. Thus, calibration at two points, a dark input and a convenient light input should be sufficient.

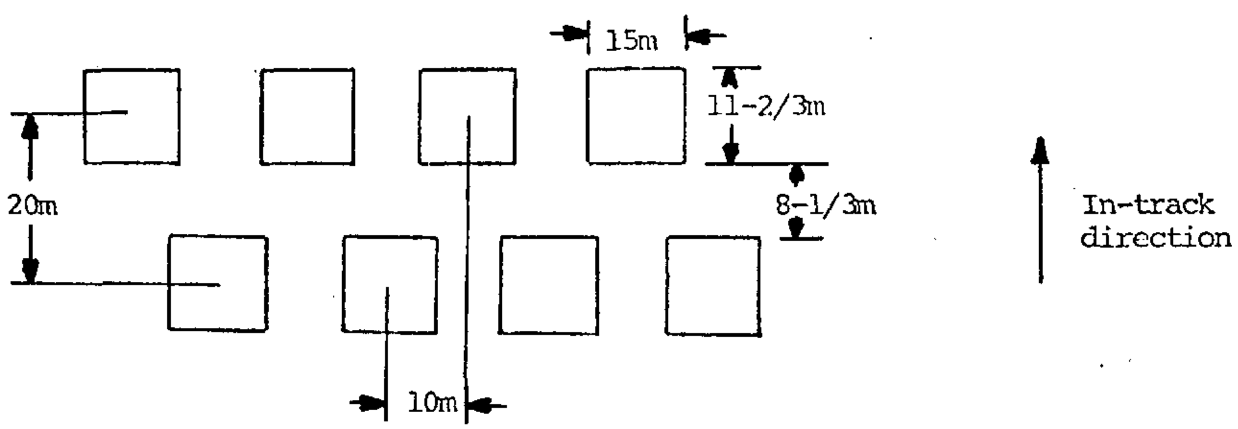


Figure 3-1. Pushbroom HRPI Array Configuration Ground Coverage Geometry

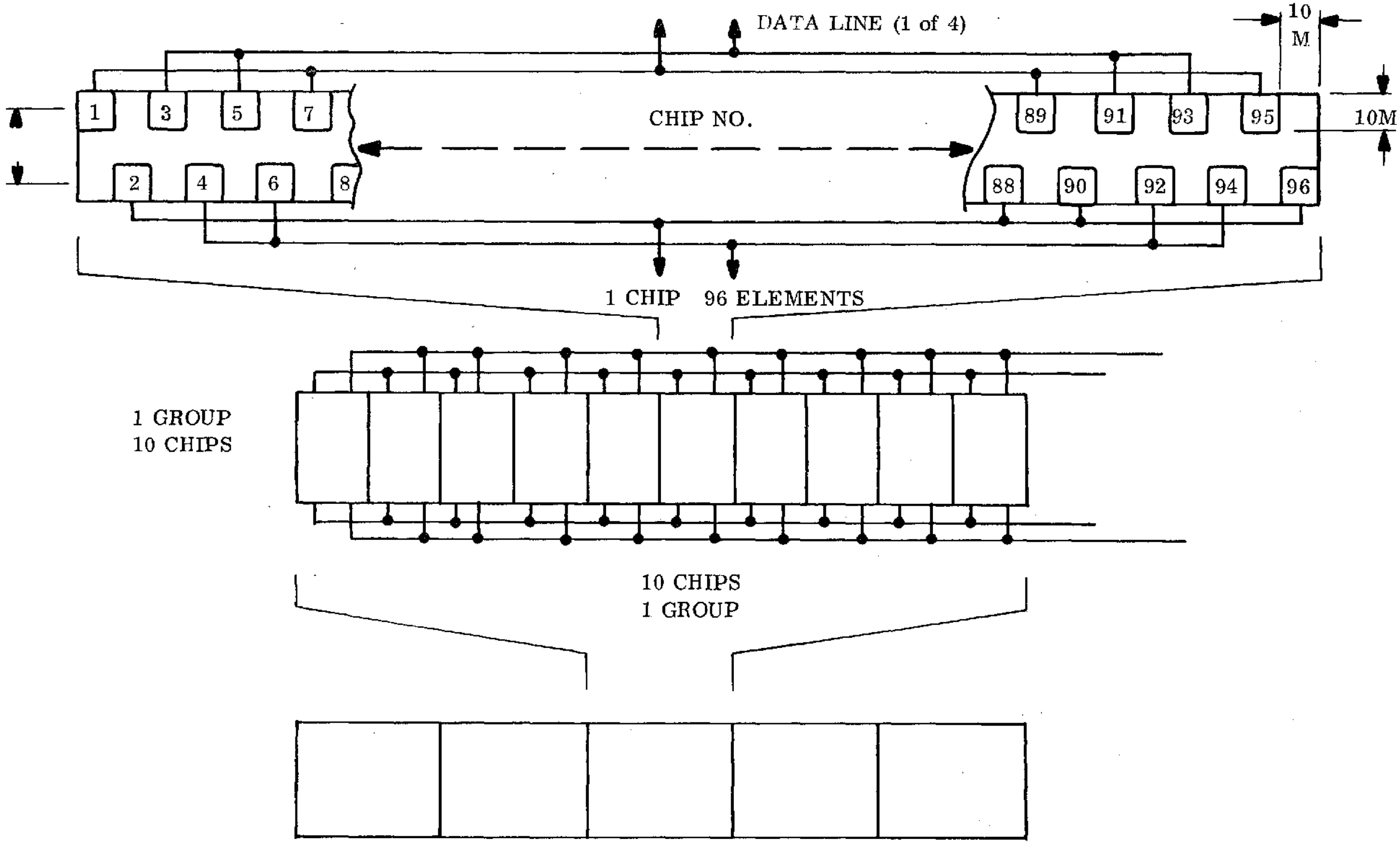


Figure 3-2. Typical Array Configuration

(c) The dark current offset is high (could be equal to the signal current) and can be reduced by a factor of two for every $10^{\circ}C$ temperature reduction. This is the reason for maintaining the array temperature at $0^{\circ}C$ with a maximum variation of $\pm 0.5^{\circ}C$.

(d) Element response stability is such that in-flight calibration update of once per orbit is anticipated. This assumes $\pm 0.5^{\circ}C$ temperature stability is maintained.

3.1.2 OBJECT PLANE MECHANICAL SCANNER

The object plane scanner HRPI point design is a derivative of the Thematic Mapper. The significant departures in the requirements, which have the greatest design impact, are the increased resolution (approximately a factor of three) and the offset pointing. The increased resolution essentially reduces the power incident on the detector thereby reducing S/N. In order to compensate for this, the number of detectors was increased and charge coupled silicon arrays employed with time delay integration along the scan to increase the dwell time or integration time per IFOV.

The offset pointing requirement necessitated a mechanical design which requires rotation of the instrument.

The instrument is rigidly supported at each end on preloaded bearings. Roll motion about the bearing axis is prevented by a spring loaded detent mechanism. Offset pointing is accomplished by a solenoid withdrawal of the detent and by a gearhead drive motor which engages a gear segment attached to the telescope housing. Redundant solenoid and drive motor are engaged through separate cam driver systems. The pointing drives, detent mechanisms, and instrument bearings are mounted on an aluminum honeycomb base which can be positioned and attached to the spacecraft throughout a range of roll angles with respect to nadir.

The focal plane contains a two-chip, charge coupled silicon array which detects, samples, and amplifies signals for all four spectral bands. Detectors are arrayed along scan as well as across scan and charge integration is obtained by time-delayed integration along scan and matching the delay to the scan rate. The charge-coupled device is useful for this purpose as the time delay integration can take place at the charge packet level before amplification.

A sketch of the layout is shown in Figure 3-3. Each spectral band uses two sets of time delay integration (TDI) illuminated registers that are interleaved by the scanning action. The concept uses three phase clocking in the scan, or TDI, coordinate so that charge can be transferred alternately in either scan direction by appropriate phase switching. Two phase clocking is used in the readout coordinate to minimize the clock rate. Readout of a sample from each line takes place (for all lines) during the part of the three-phase clock cycle when charge is not being transferred into the readout column. Gates at the ends of the readout columns are operated to alternate the two functions, fat zero insertion and readout, depending on the direction of scan.

The CCD inherently provides one sample per IFOV at its output. For the point design, the sample interval and IFOV size is taken as equal to the along-track line width. A more detailed tradeoff of the following parameters could lead to a different optimum for the along-scan IFOV:

1. MTF balance between the two axes
2. Clock frequency capability of the CCD
3. Alias content of the signal

The CCD may have need of protection against the phenomenon of blooming, wherein high signal from bright features spills charge over to other areas. Protective circuits can be provided as part of the CCD, at some expense in space and layout complication.

3.1.3 IMAGE PLANE CONICAL SCANNER

The Conical Scanning HRPI point design differs from the Thematic Mapper in several key areas. In order to achieve the $\pm 40^{\circ}$ offset pointing angle, the instrument orientation is constrained with its long axis parallel to the flight velocity vector. Offset pointing is achieved by rotating the entire scanner optics about the optical axis which eliminates

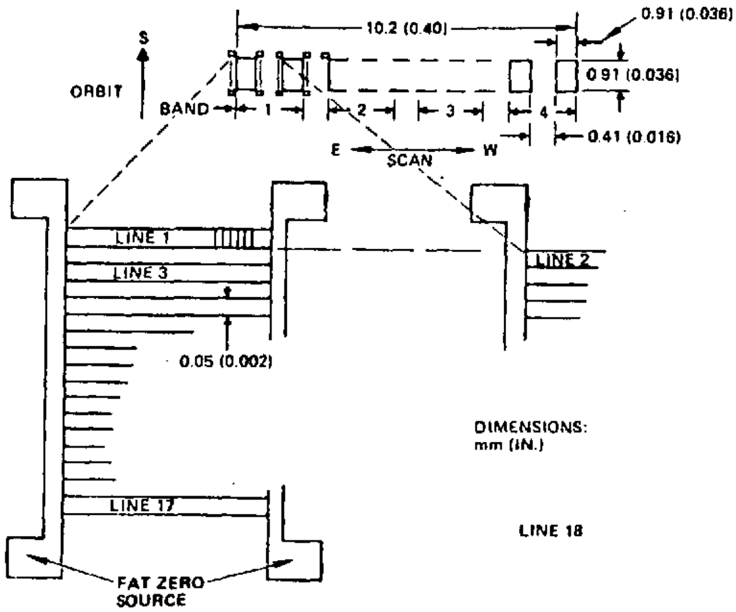


Figure 3-3. HRPI Image Plane Design - Object Plane Scanner

optical image rotation. The primary mirror is not rotated, which simplifies the pointing bearings design but places an extra burden on the design of the pointing optics support structure and bearings if focus is to be maintained under all conditions of pointing angle and environmental change. Reduction of the pointing angle requirements to $\pm 30^{o}$ would simplify the pointing mechanism design to an extrapolation of the TM point design with the addition of a servo controlled pivoted pointing mirror.

The HRPI design uses four uncooled 80-element arrays of photovoltaic planar diffused silicon photodiode detectors to achieve the required S/N performance. Spectral separation is achieved via four optical bandpass filters. No spectrometer is used. The four arrays are mounted side-by-side so that each array views different IFOV's. The choice of 80 elements per array was based upon S/N and off-axis optics MTF considerations.

The swath width of 46 km is obtained using a scan wheel of 6-inch nominal diameter which is one-third the size of the TM wheel. This allows the use of an exceptionally rigid wheel on a sturdy mount which minimizes scan wheel jitter geometric errors.

3.1.4 IMAGE PLANE LINEAR SCANNER

The general configuration for the linear scanning HRPI is identical to the TM point design. Offset pointing is obtained with the pointing mirror with angles up to $\pm 45^{\circ}$ achievable. The scan wheel diameter is smaller than the TM (28" vs. 34") and the wheel rotation rate is slower (.21 rps vs. 1.0 rps), but there are more roofs (64 vs. 18) and the scan efficiency is lower (66% vs. 74%). The clear aperture area was increased to 803 cm^2 from 625 cm^2 in order to meet the S/N requirements.

The detector configuration consists of four 50-element linear arrays spatially separated. The detectors are diffuse junction PIN silicon photodiodes cooled with their FET amplifiers to about $200^{\circ}K$.

The scan rate can be adjusted to match the scene advance rate. Stripe-to-stripe underlap (causing gaps in information) or overlap (causing redundant information) occurring with variations in satellite altitude and offset pointing angle can be eliminated. The utility of this feature for offset pointing compensation is questionable since the ground station can always reject the redundant information and the high-speed transmission link requires a constant information rate.

3.1.5 HRPI POINT DESIGN PARAMETRIC COMPARISON

A summary of the point design parameters for each of the instruments is given in Table 3.1-2.

3.2 <u>INSTRUMENT REQUIREMENTS - HRPI</u>

3.2.1 MISSION REQUIREMENTS

Section 2.2.1 delineated most of the HRPI mission requirements and rationale. Discussion of the general requirements are not repeated; only the HRPI unique aspects are considered.

Coverage and Operations. HRPI should be capable of operating simultaneously and independently of any other instrument payload. Average HRPI on time is up to 15 minutes per orbit, with about 75% being useful imaging time and 25% slewing time. Data would be sent normally during slew, but not processed at the receiving station. An auxiliary signal inserted into the composite video should indicate beginning and end of slew.

Altitude. 775 km.

Orbit. 17 day repeat, interlace factor of 6.

Swath Width. 46.3 km.

Offset Pointing. $\pm 30^{\circ}$ in $1^{\circ} \pm .05^{\circ}$ steps.

Slew Rate. 1 to 2 seconds/degree (to be set by uncompensated momentum specification).

Descending Node. 10-12 AM.

Performance. The HRPI radiometric and geometric performance is to be specified within the FOV of the TM ($\pm 6.8^{\circ}$ from nadir).

Resolution. The nominal resolution is specified as 10 meters at nadir for all bands.

3.2.2 PERFORMANCE REQUIREMENTS - HRPI

In Section 2.2.2, the performance requirements for the TM's were detailed and some care and forethought was given to being generic enough so as not to encroach upon the design flexibility and nuances of a particular scan technique. For the HRPI the specification problem becomes even more difficult due to the addition of the pushbroom array to the three basic scanners. The following performance parameters therefore, are segregated between scanning HRPI (ScHRPI) and pushbroom HRPI where appropriate.

Table 3-2. HRPI Point Design Parameters

Scan Technique	IFOV (u rad)	Swath Width (km)	Alt. (km)	Offset Pointing	Clear Aperture (cm^2)	No. Detectors per Band	Data Rate 7 bits (mbps)	SNR Band 1	SNR Band 2	SNR Band 3	SNR Band 4	Size (in)	Power (watts)	Wt. (lb) & Orientation
Pushbroom Array	10.9	48	914	$\pm 10^{\circ}$	1370	4800	87.4	79	70	60	60	27x72	100 +23 PT +21 HT	553 L⊥V
Object Plane Scan	14	40.0	717	$\pm 40^{\circ}$	1035	18x15	86.1	8	9	9	10	52x39x 42	81+10 PT	330 L=V
Conical Image Plane Scan	14	46.0	716	$\pm 40^{\circ}$	973	80	112.0	8.9	9.8	9.4	9.5	36x72	180+80PT +50 HT	600 L=V
Linear Image Plane Scan	14	46.3	715	$\pm 45^{\circ}$	803	50	132.4	5.4	5.4	6.1	6.9	36x38 x84	100 +54 HT	598 L⊥V

3.2.2.1 Sun Angle

Same as for TM (see Paragraph 2.2.2.1).

3.2.2.2 Minimum and Maximum Radiances

Same as TM bands 1-4.

3.2.2.3 Geometric Accuracy

The HRPI geometric mapping error budget given in Table 3-3 is applicable within a 13.6° angle centered at nadir. This represents a large apportionment of the total system error budget.

Table 3-3. Geometric Mapping Error Budget Baseline-HRPI

Scanning HRPI	
Start of Scan Stability	2 μ rad
* Along Scan Positional Accuracy (repeatability along entire scan including optical distortions)	1.3 μ rad
Across Scan Non-linearity (⊥ to scan line)	4 μ rad
Detector Position	
Placement (to a specific location)	.1 IFOV
Knowledge	0.05 IFOV
*Variations from this accuracy which are linear, are acceptable.	
Pushbroom HRPI - Relative Geometry	
Detector Position - two dimensional	
Placement	0.1 IFOV
Knowledge	0.05 IFOV
Sampling Time Error	100 μ sec
Optical Distortions of LOS over Array	
Control (band to band)	1.3 μ rad
Knowledge	2 μ rad
HRPI Offset Pointing (Array and Scanner)	
Increment steps and repeatability	$1^{\circ} \pm .05$
Stability once locked	4 μ rad
Slew rate - nominal	1°-2°/sec

3.3 EXTRAPOLATED INSTRUMENT DESIGNS

The HRPI mission and performance requirements as defined in Table 3-4 were utilized as a baseline for instrument design and performance comparison.

Table 3-4. Baseline System Performance Parameters, High Resolution Pointable Imager

Altitude	775 Km
Swath Width	46 Km
Ground Resolution	10 m
Angular IFOV	12.9 μ rad
Offset Pointing	$\pm 30^{\circ}$

The instrument point designs for each of the contractors have been extrapolated to meet the baseline performance requirements. In this extrapolation, instrument parameters such as clear aperture, optical efficiency, scan efficiency, detector responsivity, detector area, detector and amplifier noise per unit bandwidth, and the number of detectors per band were used as defined in the point designs.

The same basic performance equations developed for the Thematic Mapper evaluation are used for extrapolating the mechanically scanned HRPI instrument performance.

The signal-to-noise expression used for the pushbroom array is given by

$$\frac{S}{N} = \frac{(N_1 - N_2)\, F(x, \alpha)\, \alpha^2\, A_o\, \tau_o\, \tau_D\, t_I}{NES\, A_\alpha}$$

where t_I = integration time, sec

$$t_I = \frac{\alpha h}{V}$$

NES = Noise equivalent signal, joules/cm^2

The other parameters are as previously defined. Using the minimum radiance values as defined by the GSFC spec, the SNR for each spectral band and each instrument was determined. These results are shown in Table 3-5.

Table 3-5. Baseline Instrument Performance Parameters High Resolution Pointable Imager

Scanning Approach	Clear Aperture (cm^2)	No. Detectors Per Band	Scan Efficiency	fs (KHz)	Data Rate 7 bit (Mbps)	Dwell Time (μ sec)	SNR Band 1	SNR Band 2	SNR Band 3	SNR Band 4
Pushbroom	1370	4600	1.00	76.5	94.3 (10% OH)	1500	109	95	82	68
Object Plane	1035	18 x 15	.85	100	100.8	75	6.2	6.9	6.9	7.7
Conical Image Plane	973	80	.80	24.6	110.3	20	7.9	8.7	8.4	8.5
Linear Image Plane	803	50	.66	46.4	129.8	10.8	4.8	4.8	5.5	6.2

3.4 EVALUATION OF EXTRAPOLATED DESIGNS

3.4.1 PUSHBROOM VS. SCANNING HRPI

An evaluation of the extrapolated candidate designs leads to the following general observations and conclusions:

1) The pushbroom HRPI has a much higher SNR because of the longer dwell or integration time per element. This advantage can be traded for reduced instrument size and weight by reducing the optic aperture diameter. For example, by reducing the clear aperture area to 850 cm^2 (equivalent to f/3 optic @ 775 km altitude), the instrument weight would be reduced to approximately 350 lbs with a loss in SNR by a factor of 0.62. A further reduction in clear aperture area to 480 cm^2 (equivalent to f/4 optic @ 775 km altitude), reduces the SNR by an additional factor of 0.56 with an instrument weight of 260 lbs. With this aperture size (approximately 11" diameter), the SNR is still greater than the mechanically scanned HRPI. (Band 1=38, Band 2=35, Band 3=29, Band 4=24).

2) The pushbroom HRPI has an inherent 100% scan efficiency. Assuming a 10% overhead for addition of ancillary data to the data stream, the data rate is still lower than the mechanically scanned HRPI. The linear image plane design with its lower scan efficiency has a significantly higher ($\approx$35%) data rate.

3) In comparing mechanical scanners, the object plane scanner provides the greatest clear aperture for a given instrument size. This is reflected in either a smaller instrument for a given SNR requirement or higher SNR. The linear image plane design, even with cooling the detectors and preamps to 200^{o}K in order to reduce the detector and FET noise current, provides a marginal SNR. This is principally due to the lower scan efficiency, which results in higher noise bandwidth, and smaller clear aperture area. The Hughes design with the time delay integration CCD array increases the dwell time per element which effectively improves the SNR.

4) The large number of elements in the pushbroom scanner considerably increases the calibration complexity. Due to responsivity and offset variation from element-to-element a transfer characteristic is required for each element. Although the CCD array has 18 x 15 elements per band it requires only 18 calibration curves per

band because the charge transfer process averages detector variations over the 15 elements in the scan direction. Also the CCD array design minimizes the calibration problem because the image is always on axis, therefore, uniform illumination as well as knowledge of the irradiance distribution over a wide field angle is not required.

5) The offset pointing technique used in the linear image plane scanning HRPI provides the largest pointing angle without vignetting, has the least size/weight impact on the instrument design, and is the most straightforward to implement. The pushbroom HRPI offset pointing for large angles (30-45^{o}) has a problem unless the whole instrument is pointed. With the optical axis parallel to the spacecraft velocity vector, image rotation occurs, and with the optical axis normal to the velocity vector, the pointing mirror gets large and the overall instrument size is increased considerably. This problem is somewhat alleviated if the instrument size is reduced as discussed in Item (1).

6) The CCD array focal plane with time delay integration has the advantage of increasing the integration time per element. However, some of the characteristics such as crosstalk between elements, blooming at high irradiance levels, MTF and dynamic range need to be determined to evaluate the applicability of this technology to a high resolution and high radiometric accuracy instrument.

7) In the pushbroom HRPI the alignment of chips into an array and alignment of arrays in the four spectral bands with respect to each other is critical. Initial alignment and maintaining this alignment through the launch environment and temperature environment needs to be investigated.

3.4.2 GEOMETRIC PERFORMANCE

3.4.2.1 Pushbroom Scanning

Figure 3-4 shows the orbit geometry associated with "pushbroom"scanning." Coverage in the vehicle in-track direction is obtained by the vehicle orbit rate that sweeps out swaths that are essentially parallel to the vehicle sub-satellite ground track. The cross-track coverage is dictated by the sensor field-of-view. (Number of detectors in the linear array.) Figure 3-5 shows a typical scan pattern that results when a staggered array is used. Satellite motion, sensor viewing and earth curvature and rotational effects combine to cause the sampling grids to become "distorted." One of the main considerations in using a "pushbroom scanning" technique, is the registration of the data obtained from the two halfs of the staggered array.

<u>In-Track Misregistration.</u> In-track misregistration, ΔPx, is caused by the sampling rate not being in sync with the projected distance between the two halfs of the staggered array and the ground velocity. For an array that has a separation of two elements the ideal sampling rate for zero in-track misregistration is:

$$T_s = \left(\frac{D}{2V_g}\right) \text{ seconds}$$

where

D - projected distance on the ground between the two halfs of the staggard array

V_g - vehicle ground velocity

For a system in which the sampling rate is fixed, any variation in D or V_g will cause the data from the two halfs of the staggard array to be misregistered in the vehicle in-track direction. These variations can be caused by altitude changes from nominal, or the design value and/or use of the sensor at non-zero offset pointing angles. The effect of altitude variations (calculated about a nominal of 775 KM) on in-track registration amount to only 0.03 meters/km for nadir viewing. The effects of offset pointing are more severe. As the offset angle is increased from zero degress (nadir looking) the distance D increases due to the increase slant range to the earth surface, while the ground velocity decreases due to the curved earth. The effect of the increasing slant range is by far the most significant. Figure 3-6 shows the in-track misregistration for the "center" of the HRPI field of view and at the edges of the swath. An offset angle of 5^{0} corresponds to the edge of the swath covered by the TM.

<u>Cross-Track Misregistration</u>. Cross-track misregistration, ΔPy, is caused by the vehicle ground velocity not being perpendicular to the linear array. With a delay of two cycles required to complete a row of data, any motion along the linear array will cause the elements to not fill the gaps and to be displaced from their desired location.

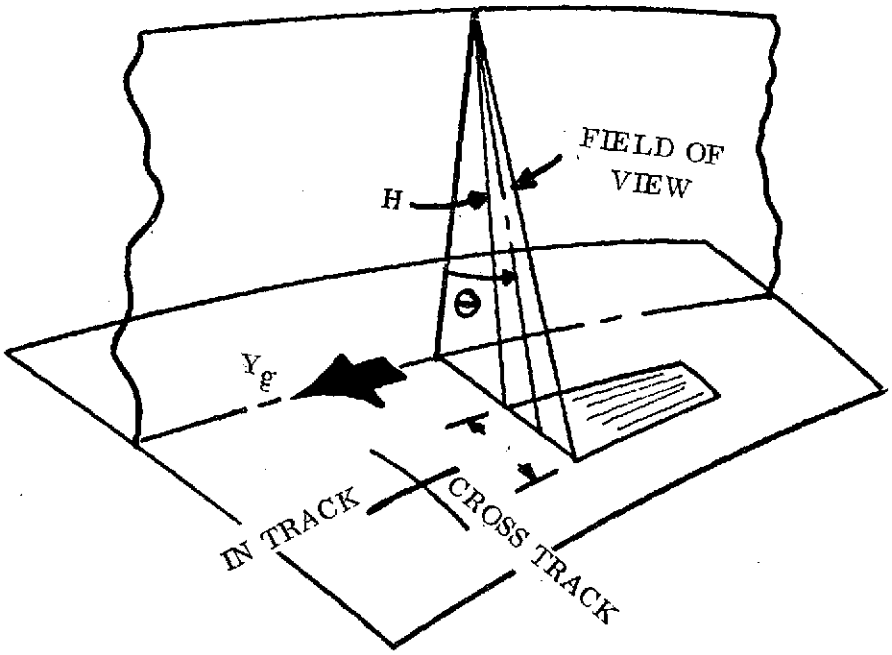


Figure 3-4. Scan Geometry

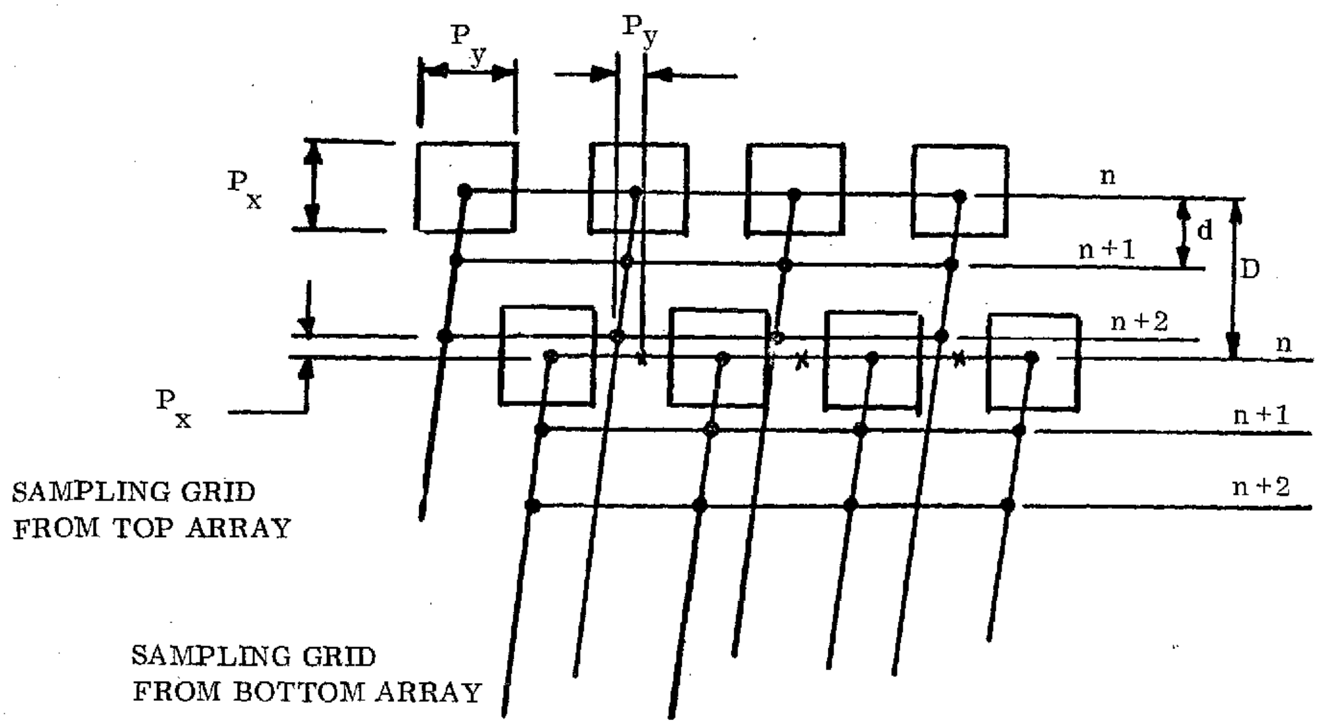


Figure 3-5. Scan Pattern

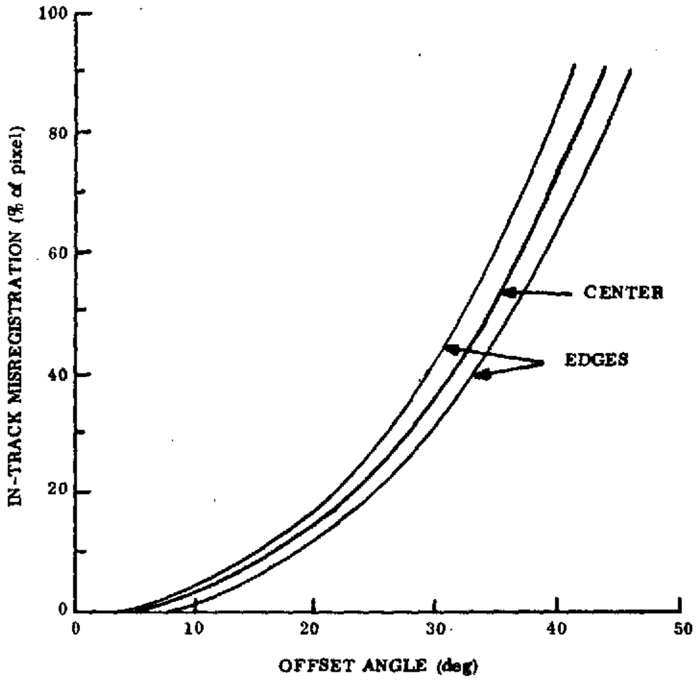


Figure 3-6. In-Track Misregistration for Various Off-Nadir Look Angles

Table 3-7 shows the effect of earth rotation on cross-track misregistration. Note that the vehicle yaw angle can be used to compensate for this effect. However, for a fixed yaw angle, it will be optimum at only one latitude. One can reduce the maximum cross-track misregistration by selecting the optimum yaw angle for a latitude of 60^{o}.

Lineal Resolution. Figure 3-7 shows the normalized linear resolution in the along track and cross track directions as a function of offset viewing angle. This can grossly be interpreted as the amount of streach between the centers of adjacent picture elements with respect to a nadir view. Bounds are also drawn for the edges of the HRPI swath.

3.4.2.2 Scanning HRPI's

The geometric effects of offset pointing for the scanning HRPI design vary as a function of the scan technique. Both the object plane and linear image plane scanners will exhibit some misregistration errors with increasing offset angle. These will be less than for the

Table 3-6. Cross-Track Misregistration (Earth Rotation)

	ΔPy meters	
Latitude	$\theta = 0^o$	$\theta = 2^o$
90^o	0.00	-0.70
80^o	0.24	-0.46
70^o	0.48	-0.22
60^o	0.70	0.00
50^o	0.90	+0.20
40^o	1.07	+0.37
30^o	1.21	+0.51
20^o	1.32	+0.62
10^o	1.38	+0.68
0^o	1.40	+0.70

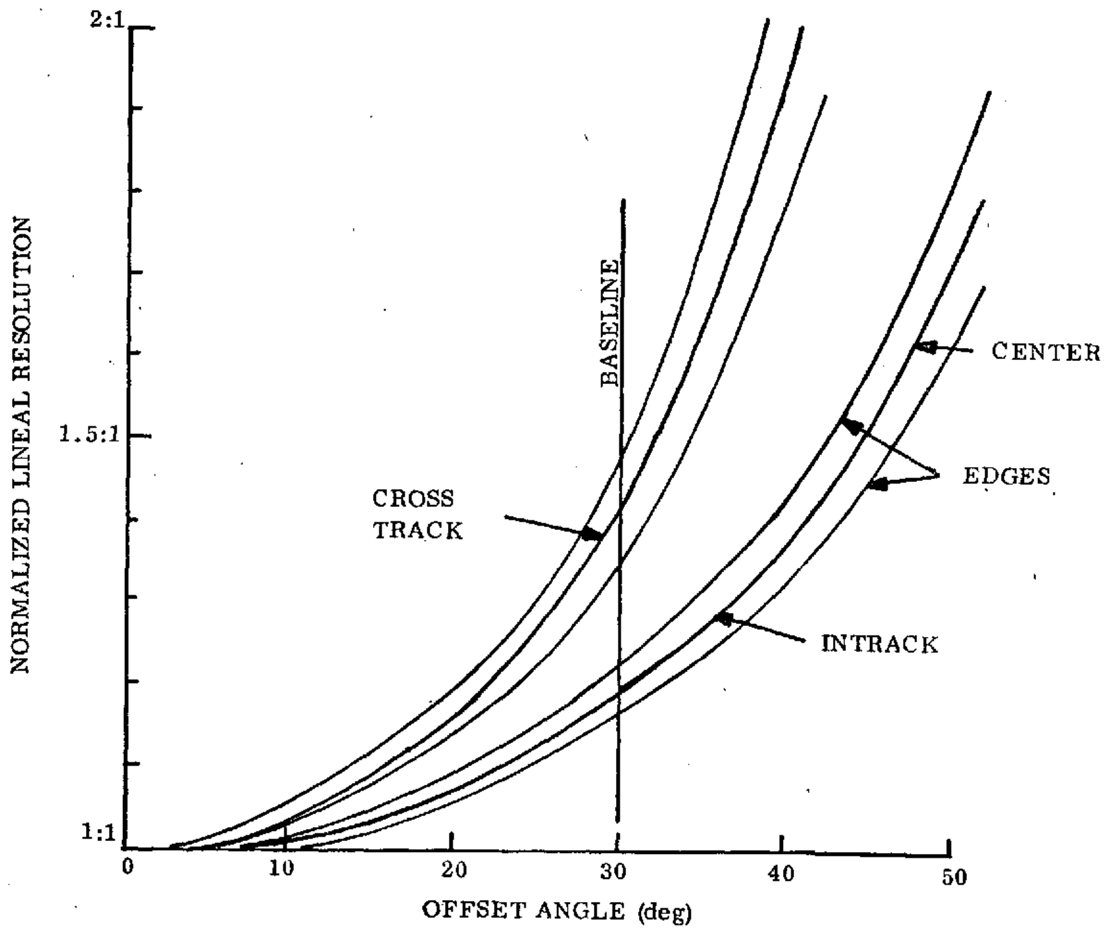


Figure 3-7. Cross and Along Track Linear Resolution

two array pushbroom instrument. The effect on the conical image plane scanning HRPI will be minimized by the spectral separation technique used.

All three scanning HRPI designs will suffer from changes in linear resolution with offset pointing. The effect will be approximately the same as those given in Figure 3-6 for the pushbroom HRPI.

3.4.3 IMPLEMENTATION COST DELTAS

3.4.3.1 <u>Spacecraft Interfacing</u>

<u>Mechanical</u>. The relative design detail of the candidate instruments prevents a classical interface evaluation among competitors. Only Hughes concerned themselves with physical design parameters and produced a fairly optimized package for size and weight, but their design would require CCD technology development work to prove performance feasibility. The Westinghouse design could be reduced in size and weight since they have significant performance margin. Both Te and Honeywell have very little design latitude since they have fairly marginal performance.

The implementation cost deltas for the alternate HRPI designs are a function of the instrument size, orientation and weight which impact the launch vehicle selection for EOS-A. These critical parameters for the alternate HRPI baseline design are summarized in Table 3.7.

Table 3-7. HRPI Critical Parameters (Mechanical I/F)

Design	Size (In.)	Weight (lb)	Orientation
Honeywell	72 L x 36 D	600	L$\parallel$V
Hughes	52 L x 43 x 39	345	L$\parallel$V
Te-Gulton	84 x 36 x 38	598	L$\perp$V
Westinghouse	72.5 x 27 D	553	L$\perp$V

L$\parallel$V L $\perp$ V

L = long dimension
V = velocity vector

A summary of the critical parameters for the alternate launch vehicles under consideration for EOS is included in Table 3.8. These parameters include launch vehicle cost, allowable payload instrument weight, shroud diameter and shroud cylindrical length.

Table 3-8. Launch Vehicle Critical Parameters (Mechanical I/F)

Launch Vehicle	Cost (M$)	Allowable Payload Inst. Wt. (Lbs)	Cross Track Length		In Track Length	
			Total	HRPI Budget	Inst. Total	HRPI Budget
Delta 2910	6.0	675	86"	74"	108"	64"
Delta 3910	8.0	1000	86"	74"	108"	64"*
Titan IIIB NUS	12.2	1300	106"	74"+	144"+	72"+

* Elongated shroud may be considered for Delta 3910 (increase of 36" appears feasible).

Comparing the HRPI critical parameters to the launch vehicle critical parameters allows an evaluation of the relative impacts of the alternate HRPI concepts.

Impact of Honeywell HRPI. The weight of the unoptimized Honeywell HRPI of 600 lbs precludes launching a two instrument (TM & HRPI) payload on Delta 2910. The two instrument payload could be launched on Delta 3910 by limiting the TM weight to approximately 400 lbs or eliminating a wideband tape recorder which has been assumed added for a Delta 3910 launch. The cost for the two instrument capability for EOS-A, using the Honeywell HRPI @ 600 lbs becomes, therefore, the difference between the Delta 2910 and 3910 launch vehicles, or 2M$.

Impact of Hughes HRPI. The Hughes HRPI with a weight of 345 lbs. and envelope of 52"L x 43" D x 39" W is marginal for a two instrument payload on Delta 2910 from both a weight and volume consideration. The Hughes HRPI must be used in conjunction with a small lightweight TM if they are both to be launched on the Delta 2910.

Impact of Te-Gulton HRPI. The unoptimized Te-Gulton HRPI is not compatible with a Delta launch since its size and required orientation preclude mounting the instrument in the Delta 86" diameter shroud. The cost penalty for flying the Te-Gulton HRPI as defined in the point study baseline is the difference between the Delta 2910 and the Titan IIIB NUS, or $6.2M.

Impact of Westinghouse HRPI. The weight of the unoptimized Westinghouse HRPI of 553 lbs restricts it to the Delta 3910 by limiting the TM weight to approximately 430 lbs or eliminating a wide band tape recorder which has been assumed added for a Delta 3910 launch. The cost of two instrument capability for EOS-A, using the Westinghouse HRPI is the difference between the Delta 2910 and 3910 launch vehicles, or $2M.

Summary of Cost Impacts. The cost impacts on launch vehicle selection (the major cost impacts in mechanical interface area) are summarized in Table 3-9. The table shows that a lighter weight, more compact unit is preferred since it allows the use of the lower cost Delta launch vehicle giving a cost savings of from 2.0 to 6.2M dollars.

Table 3-9. Alternate HRPI Impact on Launch Vehicle Selection

Alternate HRPI	Launch Vehicle Req'd	Launch Vehicle Cost M$	Relative Cost Impact M$
Honeywell	Delta 3910 (marginal wt)	8.0	+2.0
Hughes	Delta 2910 (very marginal weight)	6.0	Reference
	Delta 3910 (ample weight margin)	8.0	+2.0
Te-Gulton	Titan IIIB NUS (ample weight margin)	12.2	+6.2
Westinghouse	Delta 3910 (marginal weight)	8.0	+2.0

Electrical. The command, telemetry, clock/timecode hardware interfaces with the instruments are insensitive to any particular design approach. The data bus distribution techniques and the sizing of the spacecraft command and telemetry systems permit variation in servicing requirements without impact. This is not true, however, with power.

The westinghouse HRPI and Te-Gulton TM were arbitrarily chosen in sizing the EOS-A power system. Variation in input power requirements have a direct effect on this sizing in the area of solar array and battery capacity. A comparison of the various designs is shown below:

HRPI	Average Power	Delta Power	Delta Array	Delta Amp HR/Battery
Westinghouse	29 w	Reference	Reference	Reference
Te-Gulton	25 w	-4	$-1\ ft^2$	0
Hughes	8 w	-21	$-4.5\ ft^2$	0
Honeywell	200 w	+171	$+37\ ft^2$	+6

The only significant impact is the Honeywell baseline approach which has not been optimized for minimum power. If the 200 w average power is an actual requirement for the Honeywell design, an extra solar array panel would be required at a cost of ~\$4 K/ft^2. Also, the boost converter and battery capacity would have to be increased (relatively small cost impact of less than \$10K).

3.4.3.2 Central Data Processing Facility: Instrument Implementation Costs

The cost impact of the various instrument approaches on the ground system is summarized in Table 3-10 for the High Resolution Pointable Imager. These costs assume that the ground system is initially designed to accommodate a particular instrument. The modularized ground system design approach resulted in a hardware configuration which is fairly insensitive to the differences in the candidate instruments.

All instruments except the Hughes are band-to-band registered. The data format cost impact includes the various scanning approaches. The staggered pushbroom array is most expensive because of the need to buffer extra lines of data to fill the gaps and complete a line. Linearity includes the cost impact of removing the non-integral pixel spacing in the Westinghouse pushbroom arrays, as well as straightening the Honeywell conical scan. The radiometric accuracy (both relative and banding) is more expensive for the HRPI's due to the large number of detectors requiring correction.

Table 3-10. Impact of HRPI Instruments on Ground System Cost

	Band-to-Band Registration	Data Format	Linearity	Radiometric Banding	Radiometric Accuracy	Total
Westinghouse Stagger Array	Reference	Reference	Reference	Reference	Reference	Reference
Westinghouse Linear Array	0	-60K	0	0	0	-60K
Hughes	+20K	-30K	-45K	-65K	-15K	-135K
Honeywell	0	-60K	+255K	-65K	-15K	+115K
Te-Gulton	0	-60K	-45K	-65K	-15K	-185K

3.4.3.3 Low Cost Readout Stations: Instrument Implementation Cost

The EOS-A spacecraft will transmit a portion of the instrument data directly to users at many Low Cost Readout Stations. This data will be derived from the TM or HRPI, processed by the spacecraft compactor and then transmitted. Just as in the CDPF, the various instrument approaches have impact on the cost of the LCRS. Table 3-11 summarizes these cost impacts. It is assumed that the LCRS has been initially designed to accommodate a particular instrument implementation approach.

The major cost impact parameters are scan philosophy and band-to-band misregistration. Other parameters such as scan linearity, data format, radiometric banding and relative radiometric accuracy do not impact LCRS cost. The Westinghouse staggered array configuration is used as the reference.

All instruments are band-to-band registered except the Hughes resulting in a cost delta of +$10K. In data formatting, the staggered array is the most expensive due to the need to buffer extra lines to fill the gaps in the data. The slightly higher Hughes data formatting cost results from the need to provide storage to compensate for the back and forth scan. The $30K lineary cost differences result from the y-corrections needed to remove the non-integral pixel spacing effects in the Westinghouse array. The $10K banding cost difference is due to the need to calibrate a much larger number of detectors in the arrays as compared to the scanners.

Table 3-11. Impact of HRPI Instrument Approach on LCRS

Approach	B-B Misregistration	Data Formatting	Linearity	Banding	Total
Westinghouse Staggered Array	Reference	Reference	Reference	Reference	Reference
Westinghouse Linear Array	0	-$35K	0	0	-$35K
Te	0	-$35K	-$30K	-$10K	-$75K
Honeywell	0	-$35K	-$30K	-$10K	-$75K
Hughes	+$10K	-$25K	-$30K	-$10K	-$55K

3.4.4 DESIGN FLEXIBILITY

The capability of each of the instrument designs to adapt to a change in requirements as specified by various mission scenarios was evaluated and is summarized in the following paragraphs.

3.4.4.1 Pushbroom Scanner

In theory, the pushbroom scanner has the greatest flexibility in accommodating parameter changes such as swath width, altitude and IFOV. These parameters can be changed by increasing or decreasing the number of detectors in an array, the optic focal length and the readout timing rate. Practical considerations such as off-axis optical aberrations and complexity due to increased number of detectors places limits on this flexibility.

The spectral band is limited to $0.5 - 1.1\ \mu m$ and the addition of another spectral band within this range requires the addition of a considerable number of detectors (~4800). Spectral separation techniques are limited to spectrometry prism approaches. Inherent spatial registration of the spectral bands is necessary to prevent a large memory cost in the CDPF for multiple line storage to achieve band-to-band registration. Spectrometry prism techniques are generally more complex and have poorer optical transmission efficiency than spatial separation techniques.

Offset pointing angle is limited by the pointing mirror size with the velocity vector perpendicular to the optic axis. However, a smaller instrument size, which can be obtained due to the significant S/N advantage, can alleviate this limitation.

3.4.4.2 Object Plane Scanner

The object plane scanner can accommodate increased swath width with changes in scan rate and number of detectors per band. The scan mirror size will need to be increased to fill the aperture, and the scan drives will require some redesign. Since the S/N performance is marginal, this will be a significant factor in considering increased swath width.

Hughes has designs which can accommodate either orientation with respect to the spacecraft and, since in their present baseline the entire instrument is rotated for offset pointing, a change in this requirement is easily accommodated.

3.4.4.3 Conical Scanner

A Conical Scanner is less amenable to accommodating an increased swath. In addition to scan rate and detector number changes, the scan wheel diameter may need to be increased, and the cone angle increased resulting in a longer slant path. For pointing angles greater than $\pm 30^{o}$, the whole instrument is rotated and the orientation is constrained to having the optical axis parallel to the velocity vector.

3.4.4.4 Linear Scanner

Due to marginal S/N performance, the Te design is least amenable to an increase in swath width and/or altitude or a decrease in IFOV. Any changes in these parameters would necessitate an increase in aperture size which in turn reduces the scan efficiency. The present data rate is already higher than the other designs. On board data compaction (spooling) could reduce this negative feature significantly.

The Te design can most easily accommodate changes in offset pointing angle and the image geometry effects are minimum compared to the other approaches. The instrument orientation is constrained to having the optical axis perpendicular to the velocity vector.

3.4.5 DESIGN RISKS

3.4.5.1 Pushbroom Scanner

The critical design risk in the Pushbroom Scanner is the fabrication, reliability and alignment of the photodiode arrays. Individual elements can be damaged in the assembly of the chips into an array. Providing the necessary chip alignment for obtaining geometric accuracy is also critical. Maintaining the array linearity and the alignment of the four arrays relative to each other through the launch and thermal environment is difficult.

In order to minimize the dark current variation, the array temperature stability must be maintained to within $\pm 0.5^{\circ}C$ between calibration updates. A tight thermal control system is required to achieve this. Present generation chips have included heaters and temperature sensors for improved thermal control of the array.

Offset pointing with the staggered array configuration causes misregistration of the pixels which must be corrected. Also, the staggered configuration requires that two lines of data be read out and stored in order to obtain a complete line of ground data. This could be a design risk with regard to on-board data processing or at best adds a processing load to the central ground station data processor. A linear configuration alleviates these problems but two or three end elements are lost per chip-to-chip interface causing a loss of about 100 pixels per line for each spectral band. Another disadvantage of the linear array is the additional crosstalk between elements (particularly at longer wavelengths) which, in turn, affects the system MTF and radiometric accuracy.

3.4.5.2 Object Plane Scanner

The Hughes design utilizes CCD detector arrays with time delay integration in order to achieve the required SNR. A better definition of the CCD array performance in such parameters as MTF, crosstalk, blooming at high irradiance levels and dynamic range is required to evaluate the capability of this technology to meet the EOS geometric and radiometric accuracy requirements.

The Hughes offset pointing design requires the rotation of a large mass so that the pointing change rate (40° in 1 minute) is constrained by uncompensated momentum limitations set by the spacecraft.

3.4.5.3 Conical Scanner

The Honeywell offset pointing approach is to rotate the instrument with the exception of the primary mirror. This reduces the rotational weight and simplifies the design of the pointing bearings and their support. However, this places demands on the pointing optics support structure and bearings to assure that focus is preserved for all pointing angles and environmental conditions.

3.4.5.4 Linear Scanner

The Te roof wheel scanner requires 64 roofs which need to be aligned with respect to each other within extremely tight tolerances. The alignment needs to be maintained while the scanner is rotating and through the thermal environment.

The SNR performance of the Te design is marginal. Increasing the aperture size reduces the scan efficiency which places an increased burden on the wideband data handling system. In order to improve performance, the detectors and preamplifiers are cooled to 200°K, which necessitates the use of a radiative cooler and adds relay optics for providing an image at the cooler surface.

3.4.6 DESIGN RECOMMENDATIONS

The Westinghouse pushbroom scanner design greatly exceeds the S/N requirements. The instrument size (and weight) should be reduced considerably and will still exceed the S/N goals. This size reduction will alleviate the offset pointing problem such that the instrument can be oriented with the optical axis normal to the velocity vector to eliminate image rotation, and a pointing mirror can be used.

In the Hughes design the number of elements per band is constrained to 18 in order to minimize mapping errors at an offset pointing angle of 40°. It is recommended that the error budgets be limited to within the Thematic Mapper swath ($\pm$ 7°) resulting in the number of elements not being constrained. The time delay integration afforded by the CCD technology then may not be required to meet the performance requirements.

SECTION 4.0

ALTERNATE INSTRUMENT APPROACHES

Two alternatives to the separate TM and HRPI instruments have been considered. These are discussed in the following sections.

4.1 COMBINED THEMATIC MAPPER/HRPI

An approach to the combination of the Thematic Mapper and HRPI into one instrument with common optics was considered in a design study by Perkin-Elmer. The major thrust of the study was to evaluate the feasability of utilizing a common optical system for the two focal plane configurations associated with the TM and HRPI. The operational and design guidelines for the combined instrument are given in Tables 4-1 and 4-2 respectively. The bulk of the study report consists of optical tradeoff information and recommendations indicating the limitations of combined instrument optical configurations.

The proposed configuration utilizes a wide field of view Schmidt optical system with a linear image plane scanner for the TM and a pushbroom array for the HRPI. The TM and HRPI fields of view are separated in the image plane and displaced in the spacecraft velocity direction by 1.3°. A general discussion is given on detectors but performance is not compared and no recommendations made as to preferred approach.

The study concludes that it is technically feasible to combine the instrument with the constraint that the HRPI field of view is a fixed portion of the TM field of view. To achieve higher access rates to specific targets, the entire instrument is pointed off-nadir. This inhibits gathering contiguous synoptic data while also gathering higher resolution sampled data unless the targeting requirements for the two types of data happen to be within the instrument field of view. The combined instrument pays a substantial penalty in flexibility as various mission models are considered.

Table 4-1. Combined Instrument Operation

1.	Instrument capabilities operate simultaneously.
2.	Instrument capabilities maintain independence.
3.	HRPI will remain within TM FOV.
4.	Ground swath widths are fixed at baseline values
5.	Whole instrument may be offsetpointed (body pointing or diagonal mirror).
6.	Spacecraft may not be pointed.
7.	Inter-instrument spectral redundancy is to remain.
8.	Inherent reliability competative with use of individual instruments.

Table 4-2. Combined TM/HRPI Instrument Design Guidelines

1. Corrected Field of View:	equal to largest instrument combined; 11.5° = TM
2. Image Quality:	equivalent to best instrument combined; 10 μR = HRPI; 30 μR = TM
3. Aperture Size:	determined by TM or HRPI at 16 to 24 inches
4. Focal Plane Accessibility:	external; to permit multiple instrument focal plane structures without vignetting
5. Image Geometry:	flat or curved depending on scanner type
6. Spectral Range:	0.45 μ to 12.5 μ, or 0.45μ to 2.5μ without thermal IR elimination of 2 μm band allows refractive correctors
7. Size/Weight:	not to greatly exceed the largest of the instruments (36 x 84 in. cylinder, 600 lbs weight of TM or HRPI)

4.2 Te DUAL-MODE SCANNING SPECTRO-RADIOMETER (DMS) EVALUATION

The Te proposal for a Dual Mode Scanner (Report 142-73, dated April 1974) has been reviewed as summarized below.

Mission Objectives. A primary goal of the TM is contiguous synoptic coverage of land masses with the smallest access time possible. Therefore, as indicated in paragraph 4.1 any combined instrument must usually be pointed at Nadir with the high resolution sub-swath contained within the full field of the instrument. The DMS has a 139 Km ($\pm 5.56^{o}$) swath so that access time is increased. The orbit would have to be changed from the 715 Km altitude proposed so that suitable frame sidelap would be possible.

Instrument Performance. The instrument S/N in the high resolution mode ranges from 3.6 to 4.6 in Bands 1-4. The resulting penalty for such a low S/N is that the dynamic performance of the high resolution mode is essentially no better than the lower resolution mode. If the S/N could be improved by a factor of 2 as suggested, this would alleviate the problem somewhat, but the FET amplifier improvement requires development work.

Manufacturability. The alignment complexity that would result from the need to align 42 roofs, 2 IMC's and channel optics is formidable. Also increased electronic complexity is required due to the 65% scan efficiency. A minimum of about 0.25 M bits of data storage is required for "spooling."

Because of the significant limitations on mission flexibility imposed by the need for the high resolution FOV to be within the TM FOV, GE recommends that the combined TM/HRPI instrument not be considered for EOS missions.

SECTION 5.0

INSTRUMENT CONTRACTUAL REQUIREMENTS

Volume 4, "Low Cost Management Approach" recommends a contractual arrangement for EOS in which NASA/GSFC awards a prime contract to a Systems Integration Contractor and contracts to a number of Associate Contractors who provide the payload instrumentation for the various missions. The Systems Integration Contractor would be responsible for design, development and production of the basic spacecraft, the mission peculiar section and integration of the instruments into the spacecraft. In addition, the systems contractor would be responsible for design, development and production/installation of the Operational Control Center and Data Processing Facility at the NASA location. The Systems contractor would also be responsible for overall systems engineering.

The Instrument Contractor would be responsible for the design, development, production and test of his equipment and would also be responsible for ascertaining that his equipment is indeed compatible with the spacecraft and the data handling system.

General Electric recommends that NASA place the Instrument Contracts directly with the instrument suppliers for the following reasons:

- The most effective use of existing GSFC expertise will take place if NASA procures the instruments directly. Since NASA has already initiated contracts with these suppliers it appears advantageous to continue with this mode of operation.
- This contractual method enables GSFC to directly tradeoff cost, schedule, and performance between the spacecraft contractor and the experiment contractors.
- The direct contract between NASA and the instrument suppliers will avoid the payment of "double G&A" by the government, thereby effecting a substantial cost savings in the program.

Since the interface between the spacecraft ground system and the instruments is probably the most critical and complex one in the entire program, the relationship between the Systems Integration Contractor and the Instrument Contractor should be specifically deliniated in the contracts and incentives provided to foster close working relationships. The following points are recommended to be incorporated in the instrument contracts:

- o The Systems and the Instrument Contractors must mutually agree (with NASA concurrance) on a milestone schedule concerning the design, development and delivery of the hardware to the Systems Contractor for integration into the spacecraft. An award fee arrangement should be specified for meeting these critical milestones with hardware that successfully demonstrates its capability to meet a pre-arranged set of instrument/ system requirements. Milestones should include system level demonstrations, box and delivery of the instruments to the prime contractors. This arrangement should motivate the Instrument Contractors to provide quality hardware on schedule and continue to be motivated to assist the systems contractor during final systems integration and checkout.
- o It is recommended that the Systems Contractor should be a member of the Instrument Contractor's Change Control Board. In this manner, the Systems Contractor will be able to advise NASA as to the effect of the change on the total spacecraft and ground data processing systems. Many times a seemingly innocuous change in an instrument could have far reaching effects on the system downstream thereby seriously affecting schedules and costs.
- o The Systems contractor should be present at Instrument design reviews in order to assure that compatibility does indeed exist between the instruments and the spacecraft. The Systems Contractor, because of his knowledge of the total scope of the program is in an excellent position to advise NASA as to the adequacy of the design.
- o The Systems contractor should review all materials and processes used in the manufacture of the instruments to avoid such things as outgassing problems, magnetic materials problems, etc. Reviews of this nature, early in the program, will reduce the likelihood of problems later on when they can seriously effect costs and schedule.
- o The Systems contractor should receive copies of instrument failure reports, test data packages, contract change notices, requirements documents and convenience change agreements in order to stay abreast of the instrument developer's progress and problems. Once again, the systems contractor can advise NASA as to the seriousness of these various matters in terms of total mission success.

- The Systems Contractor should participate in the early design of Bench Test Equipment, Handling Equipment, Targets, Facility Requirements, Cooling and Ground Support Equipment. Since the equipment is generally sent to or duplicated at the Systems Contractor's house, early familiarity with its operational requirements would save time and dollars later in the program and possibly prevent unrealistic facility requirements.

In general, GE recommends a close alliance among NASA, the Instrument Contractor and the Systems Contractor to promote a free interchange of information and ideas which will benefit all in the program. This does not require any exceptional contract terms and conditions to be placed upon the instrument suppliers.

APPENDIX A

EARTH OBSERVATORY SYSTEM

(EOS-A)

HANDBOOK

SECTION 1
INTRODUCTION

The purpose of this handbook is to provide a concise description of the EOS-A system for use by potential instrument suppliers, to identify the interface between the instruments and the system, and to facilitate information flow early in the program between the prime contractor and the instrument suppliers. Specifically, the handbook:

- o Provides both a broad system overview and a specific mission description;
- o Defines the organization, philosophy and interfaces of the modularized spacecraft and ground systems;
- o Defines the payload characteristic information required by the system.

The general description of the EOS system is included to show the inter-relationship of the payload to the system and to provide a basis of understanding of the particular interfaces. The description places particular emphasis upon the spacecraft since the basic interface is with the spacecraft system. The system interface definition provides the requirements for the design of the payload side of the interface. Information on payload characteristics is required to achieve a compatible system. Because much of this information affects the spacecraft design, the information is required early in a particular mission cycle.

The handbook is separated into three sections. Section 2 provides an overall system description and a specific description of the EOS-A mission.

Section 3 defines the spacecraft systems for the selected mission. Each subsection describes the functioning of a spacecraft subsystem and defines the spacecraft side of the payload interfaces. In addition, the payload characteristics required to achieve a compatible system are identified.

Section 4 defines the test program for the spacecraft. It also includes a general description of the two major segments of the ground system, the Operations Control Center from which the on-orbit operations of the spacecraft are controlled, and the Data Processing Facilities where instrument data is converted to output products for investigators.

SECTION 2

MISSION AND SYSTEM DESCRIPTION

2.1 EOS PROGRAM

2.1.1 OVERALL MISSION

EOS is the next generation system for R&D and operational applications missions in low Earth orbit. The overall objective of the EOS Program is to provide an economical, multi-purpose, modular spacecraft system to support observations missions through the 1980's in the areas of:

- o Earth and ocean survey
- o Pollution detection and monitoring
- o Weather and climate predictions

As the system developed it became clear that the multi-purpose spacecraft approach has the capability to support other missions also including solar pointing such as the Solar Maximum Satellite and geosynchronous missions typified by the Synchronous Earth Observatory Satellite (SEOS).

The key feature of this advanced remote sensing system is the development of a modular observatory system, the spaceborne elements of which are compatible with varying levels of launch vehicle capability. The system can currently utilize the Delta and Titan launch vehicles and will utilize the space Shuttle when that system becomes available in the early 1980's. The ability to match the launch vehicle to the required spacecraft weight and altitude for a given mission greatly enhances the economies which can be realized with the multi-purpose spacecraft approach.

To achieve a low-cost system design, a modular building block approach has been adopted. This concept utilizes a set of generalized subsystems in such a way that a variety of missions can be supported. By standardizing the mechanical configurations and electrical interfaces of the subsystem modules, and by designing each of them to be structurally and thermally independent entities, they can be clustered to support a mission-unique instrument system without the need for subsystem redesign.

The modularity concept has been extended to provide for eventual on-orbit replacement of elements using the space Shuttle in the 1980's. On-orbit service will be used both for periodic maintenance of the spacecraft as well as replacement in case of failures. In addition, the spacecraft is retrievable by the Shuttle for refurbishment on the ground. This further extends the economic benefits of the EOS system by permitting repeated reuse of spacecraft elements.

The EOS system offers the following capabilities:

- o The ability to launch the observatory with either the Delta, Titan or Space Shuttle.
- o The ability to completely reconfigure the spacecraft for different payloads without major redesign.
- o The potential to perform in-orbit resupply and/or retrieval when the Space Shuttle becomes available.

2.1.2 EOS-A: THE LAND RESOURCE MANAGEMENT MISSION

The initial mission in the EOS series is a Land Resource Management Development Mission. This mission will develop advanced instruments and processing systems which can provide multispectral imagery of the land surface of the earth at significantly improved spatial, spectral, and temporal resolutions than are available from either the Earth Resources Technology Satellite or from the projected Department of the Interior Earth Resource Survey Operational System. It thus will permit studies of the direction in which the operational land use inventory and earth resource management programs should proceed. Initial flight test of the instruments and applications research with the data will be in 1979. The specific EOS-A mission objectives are to:

- o Develop sensor and other spacecraft systems to acquire spectral measurements and images suitable for generating thematic maps of the earth's surface.
- o Operate these systems to generate a data base from which land use information such as crop or timber acreages or volumes, courses and amounts of actual or potential water run-off and the nature and extent of stresses on the environment will be extracted.
- o Demonstrate the application of this extracted information to the management of resources such as food and water, the assessment and prediction of hazards such as floods, and the planning and regulation of land use such as strip mining and urbanization.

To accomplish these mission objectives it is necessary to:

- o Develop space-borne sensors for the measurement of parameters, as required by earth observations discipline objectives, with increased performance and in new spectral regions not achievable by present sensors.
- o Evolve spacecraft systems and subsystems which will permit earth observations with greater accuracy, coverage, spatial resolution and continuity than existing systems by avoiding spacecraft constraints on sensor performance.
- o Develop improved information processing, extraction, display and distribution systems so that the applicability of the observations may be enhanced.
- o Achieve these objectives with sufficient economy and flexibility to permit the operational use of any hardware or other system components with little or no redevelopment.
- o Use the space transportation system's resupply and retrieval capability to sustain and refresh this remote sensing capability through the 1980's, thereby providing an efficient means for demonstrating the viability of improvements prior to committing to operational use.

To achieve its broad objectives, EOS-A provides for the repetitive acquisition of high resolution multispectral data of the earth's surface on a global basis. Two sensor systems have been selected for this purpose: a seven-channel Thematic Mapper (TM), and a four-channel High Resolution Pointable Imager (HRPI). In addition, the observatory will be utilized as a relay system to gather data from remote, widely distributed, earth-based sensor platforms equipped by individual investigators. The data acquired by the total EOS-A system will thus permit quantitative measurements to be made of earth-surface characteristics on a spectral, spatial, and temporal basis.

Systematic, repeating earth coverage under nearly constant observation conditions is provided for maximum utility of the multispectral data collected by EOS-A. The Observatory operates in a circular, sun synchronous, near-polar orbit at an altitude of 418 nautical miles. It circles the earth every 100 minutes, completing 14 orbits per day and views the entire earth every 17 days. The orbit has been selected and will be trimmed so that the satellite ground trace repeats its earth coverage at the same local time every 17-day period within 10 nautical miles. A typical one-day ground coverage trace is shown in Figure 2-1 for the daylight portion of each orbital revolution.

To make maximum utilization of the high resolution, off-nadir pointing capability of the HRPI instrument, a day-to-day orbit pattern has been adopted which permits access to any point on the earth within three days. This day-to-day orbit pattern is shown in Figure 2-2. The two outer orbits (Orbit 1 Day 1, Orbit 2 Day 1) represent the ground traces of adjacent orbits on a single day. On the second day a ground trace (Orbit 1 Day 2) falls approximately one-third the way between the two. On the third day, a ground trace (Orbit 1 Day 3) falls two-thirds of the way between the first two. The HRPI off-nadir pointing capability is equal to one-third the distance between the two swaths on the first day; hence potential access anywhere on the earth is provided every three days. On the fourth day the pattern begins again.

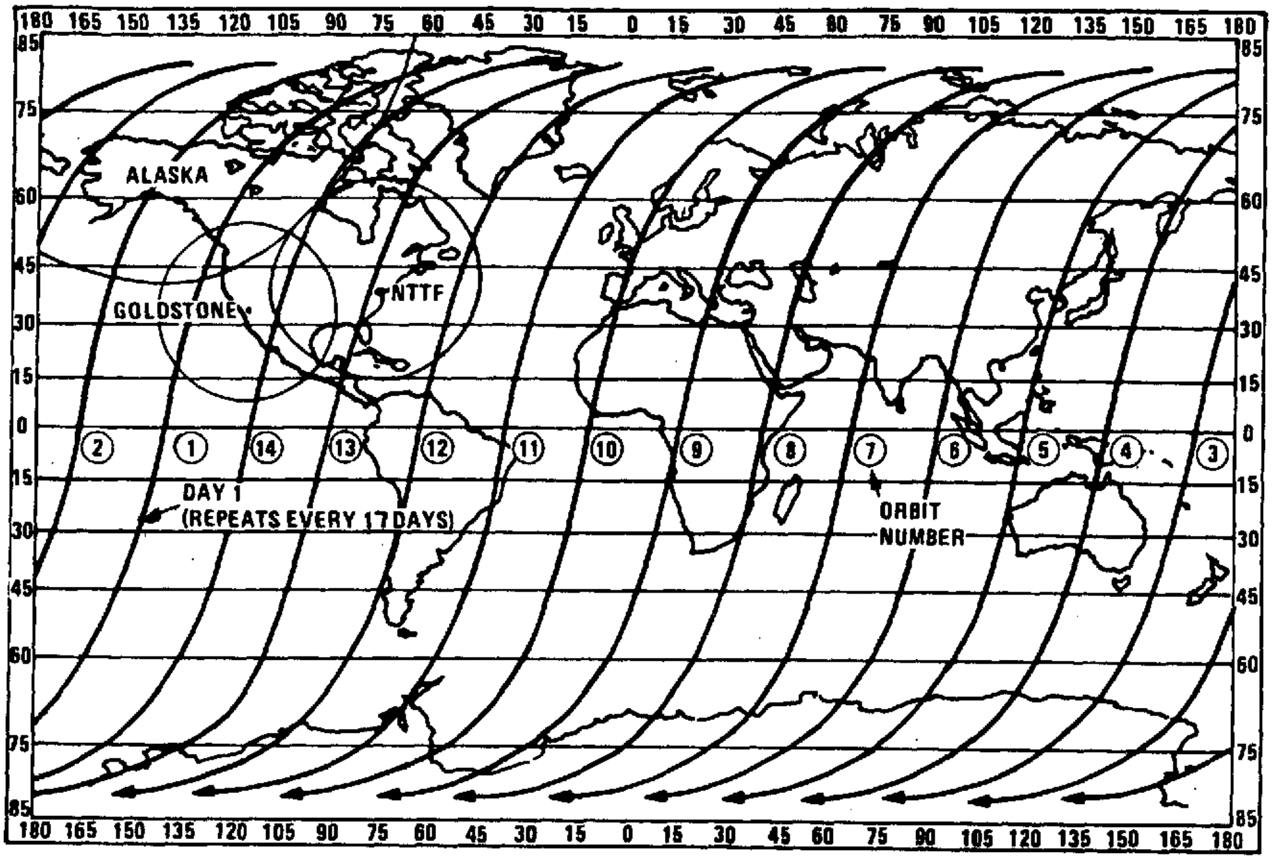


Figure 2-1. Typical EOS-A Daily Ground Trace
(Daylight Passes Only)

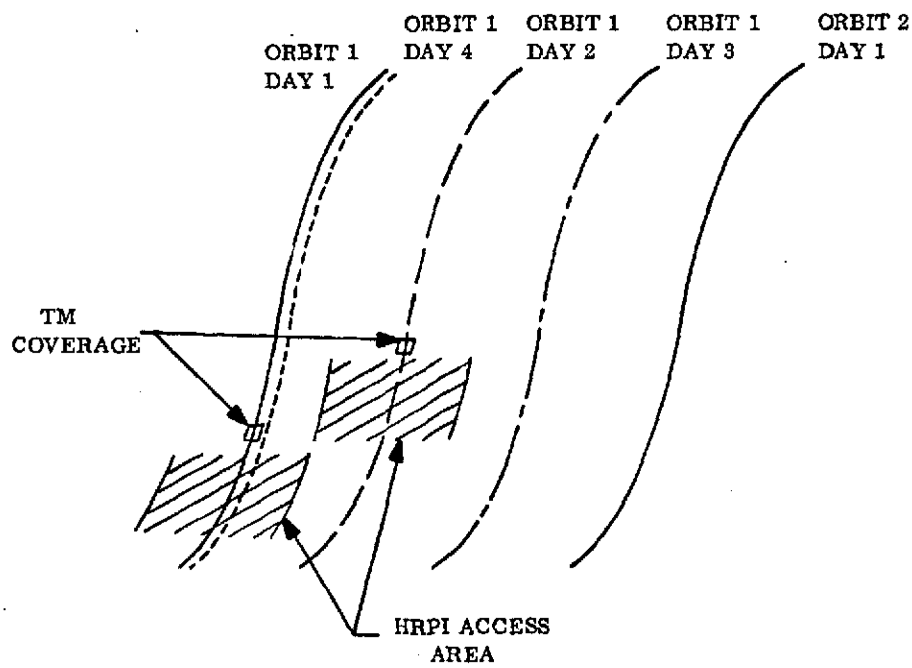


Figure 2-2. EOS-A Orbit 3-Day HRPI Access

C-2

2.2 EOS SYSTEM

2.2.1 OVERVIEW

The overall system applicable to all EOS missions is illustrated in Figure 2-3. The Observatory carries an instrument payload and all necessary communications equipment for return of the payload data. The spacecraft "housekeeping" telemetry, tracking and command data are compatible with stations from NASA's Space Tracking and Data Network (STDN). Wideband payload data is received at Fairbanks, Alaska, Goldstone, California, and the GSFC Network Test and Training Facility at Greenbelt, Maryland.

The Operations Control Center (OCC) is the focal point of all mission orbital operations. Here the overall system is scheduled, spacecraft commands are originated and orbital operations are monitored and evaluated. The OCC operates 24 hours per day and its activities are geared to the operations timeline dictated by the 100 minute spacecraft orbit and the network coverage capability. Telemetry and command data transfer between the OCC and remote ground sites is accomplished by NASA Communications (NASCOM). The primary receiving stations in Alaska, Goldstone, California, and the NTTF at Goddard provide contact with the spacecraft on 12 or 13 of the 14 orbits each day.

The Central Data Processing Facility (CDPF) accepts payload data in the form of magnetic tapes received in realtime at the NTTF station or by mail from Alaska and Goldstone. The CDPF then performs the required correction and annotation of the data and prepares output products for users in the form of computer compatible tapes and color and black and white imagery. The CDPF includes a storage and retrieval system for all data and provides for the delivery of data products and services to investigators and other data users.

2.2.2 GENERAL PURPOSE SPACECRAFT

The elements of the Observatory system include the general purpose spacecraft subsystems and the mission-peculiar equipment which together comprise the spacecraft Observatory. The Observatory configuration is shown in Figure 2-4. The attitude

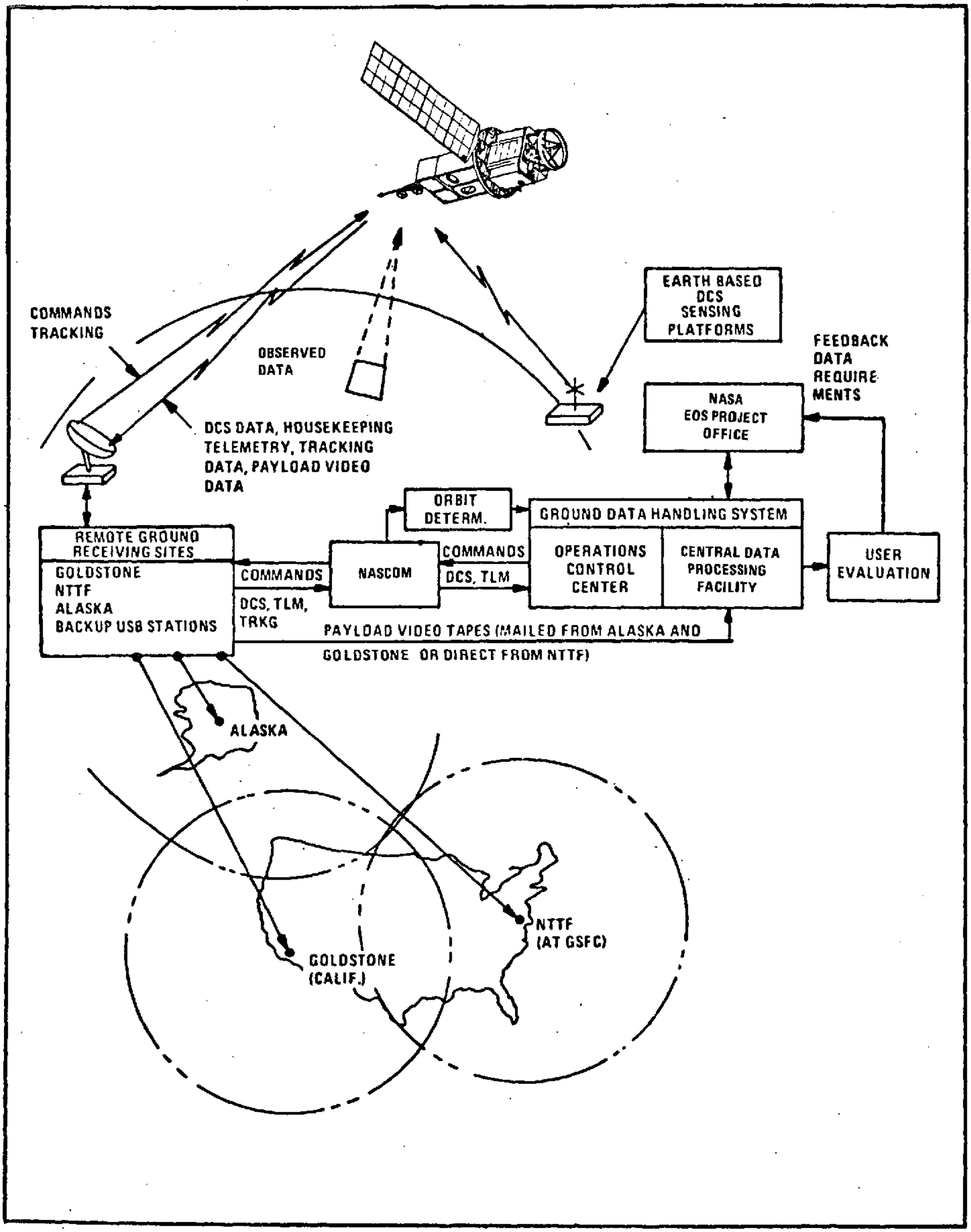


Figure 2-3. Overall EOS System

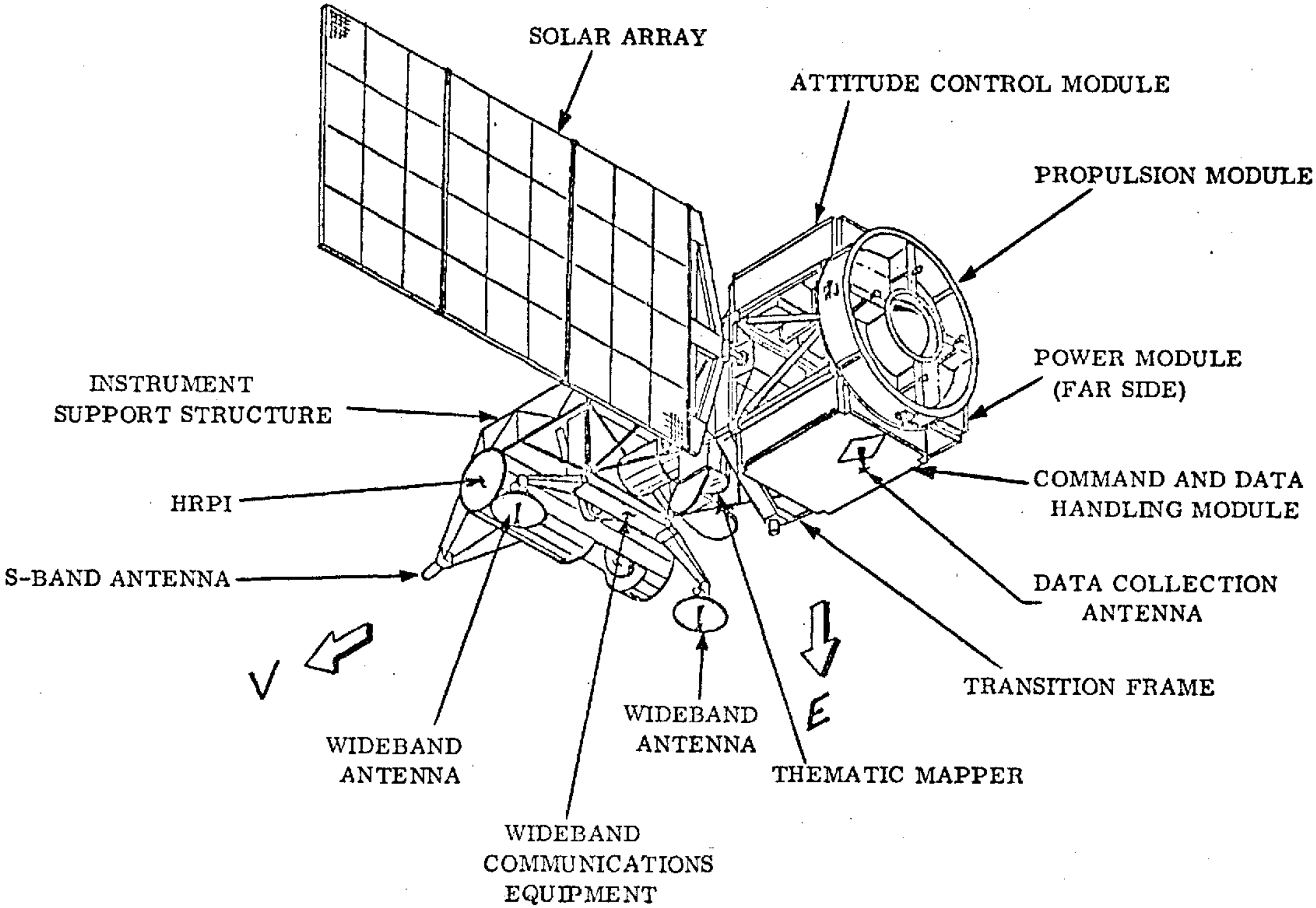


Figure 2-4. Observatory Configuration

control module, the command and data handling module, the power module and solar array, the propulsion module and the support structure are clustered aft of the transition section and form the general purpose spacecraft adaptable to a wide range of missions. All equipment forward of the transition section typically including the instruments, tape recorders, and wideband communications equipment are "mission unique", i.e., peculiar to the mission to be flown.

2.2.3 EOS-A MISSION PECULIAR SECTION

The EOS-A spacecraft consists of the general purpose spacecraft equipment just described plus the EOS-A mission-unique equipment forward of the transition section. As shown in the exploded view of Figure 2-5, this mission unique equipment includes two instruments: a Thematic Mapper and a High Resolution Pointable Imager; high-speed multiplexing equipment; a data compactor to reduce data rate for transmission to local users; the wideband communications equipment including transmitter and pointable antennas; plus the supporting structure. The instruments are:

Thematic Mapper. This seven-band scanner provides nominally 30-meter resolution across a 185 km swath width on the ground. Six spectral bands cover the visible and near infrared (0.5 - 2.35 μm) with a seventh band in the thermal infrared (10.7 - 12.2 μm).

High Resolution Pointable Imager. This is a four-band imaging device providing nominally 10-meter resolution across a 46 km swath width on the ground. The four spectral bands cover the visible and near infrared range from 0.5 to 1.1 μm. The instrument includes the capability for off-nadir pointing and this capability, coupled with the selected orbit, permits access to any point on the globe every three days.

The spacecraft equipment is described in detail in the next Section.

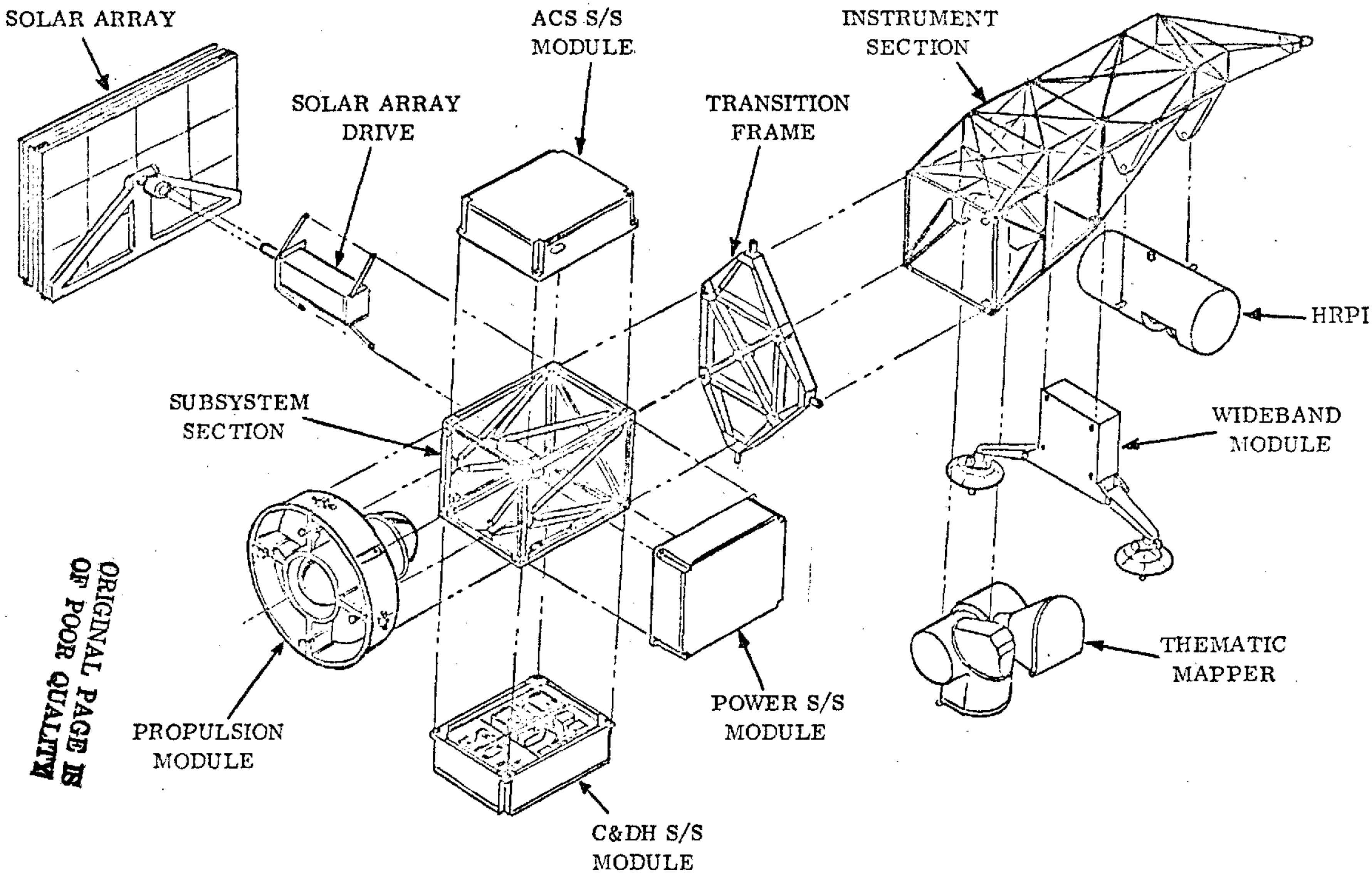


Figure 2-5. EOS Exploded View

SECTION 3

SPACECRAFT SYSTEM AND INTERFACE REQUIREMENTS

This Section defines the spacecraft systems for the EOS-A mission. Each subsection describes a particular subsystem or function, defines the spacecraft side of the payload interface and identifies the information and characteristics required from the payload to achieve a compatible system.

3.1 <u>ATTITUDE CONTROL SUBSYSTEM</u>

3.1.1 DESCRIPTION

The spacecraft uses an inertially referenced control subsystem which orients the spacecraft to the local vertical. Primary attitude reference is provided by a three-axis gyro package (inertial reference unit), with attitude updated periodically by a star sensor. Control is provided by momentum wheels which are continuously unloaded at low torque levels by magnetic coils which torque against the earth's magnetic field. High initial separation rates are eliminated by a propulsion reaction control subsystem, which also serves as a backup momentum wheel unloader. An on-board computer is used to implement many of the ACS functions.

3.1.2 ACS CHARACTERISTICS

The ACS characteristics are listed in Table 3-1.

3.1.3 INTERFACE INFORMATION REQUIRED

Knowledge of certain instrument characteristics is required to verify compatibility with the ACS subsystem. These characteristics are listed in Table 3-2. The minimum value represents the limit of sensitivity of the ACS, i.e., if the parameter value is less than the minimum value it has no affect on the ACS. The maximum value represents the upper limit that can be tolerated by the ACS.

Table 3-1. ACS Characteristics

Parameter	Value	Tolerance
Inertial Reference Knowledge		
Position		
Pitch	0.005^{o}	
Roll	0.005^{o}	
Yaw	0.005^{o}	
Rate (over 30 minutes period)		
Pitch	10^{-6} deg/sec	
Roll	10^{-6} deg/sec	
Yaw	10^{-6} deg/sec	
Jitter (stability over 20 minutes		
Pitch	0.0003^{o}	
Roll	0.0003^{o}	
Yaw	0.0003^{o}	
Control (body axes WRT nadir)		
Position		
Pitch	0.005^{o}	
Roll	0.005^{o}	
Yaw	0.005^{o}	
Rate		
Pitch	10^{-6} deg/sec	
Roll	10^{-6} deg/sec	
Yaw	10^{-6} deg/sec	

Table 3-2. Instrument Parameter Values Required by ACS

Parameter	Limits	
	Maximum	Minimum
Moving Mass Definition		
Momentum	0.2 lb bit sec	---
Period	---	30 sec.
Expended Materials		
Thrust	10^{-4} lb	---
Magnetic Materials		
Residual Dipole	1000 pole-cm	---

3.2 COMMAND AND DATA HANDLING SUBSYSTEM (C&DHS)

3.2.1 DESCRIPTION

A block diagram of the C&DH subsystem is shown in Figure 3-1. The C&DH subsystem provides both ground station (real time or delayed) and on-board computer (OBC) commands to the instrument. It also accepts telemetry data from the instrument for transmission to the ground or use by the OBC. Commands and telemetry data are distributed via data busses which control remote decoders and muxes located within the instrument module. (These remote units will be supplied by the C&DH subsystem). The data busses and remotes also provide the instrument with an interface to the OBC. All command and telemetry servicing will be accomplished under control of the central command decoder and telemetry format generator located in the C&DH module.

A spacecraft clock generates a standard frequency and timecode which is available to the instruments for synchronization and annotation of data.

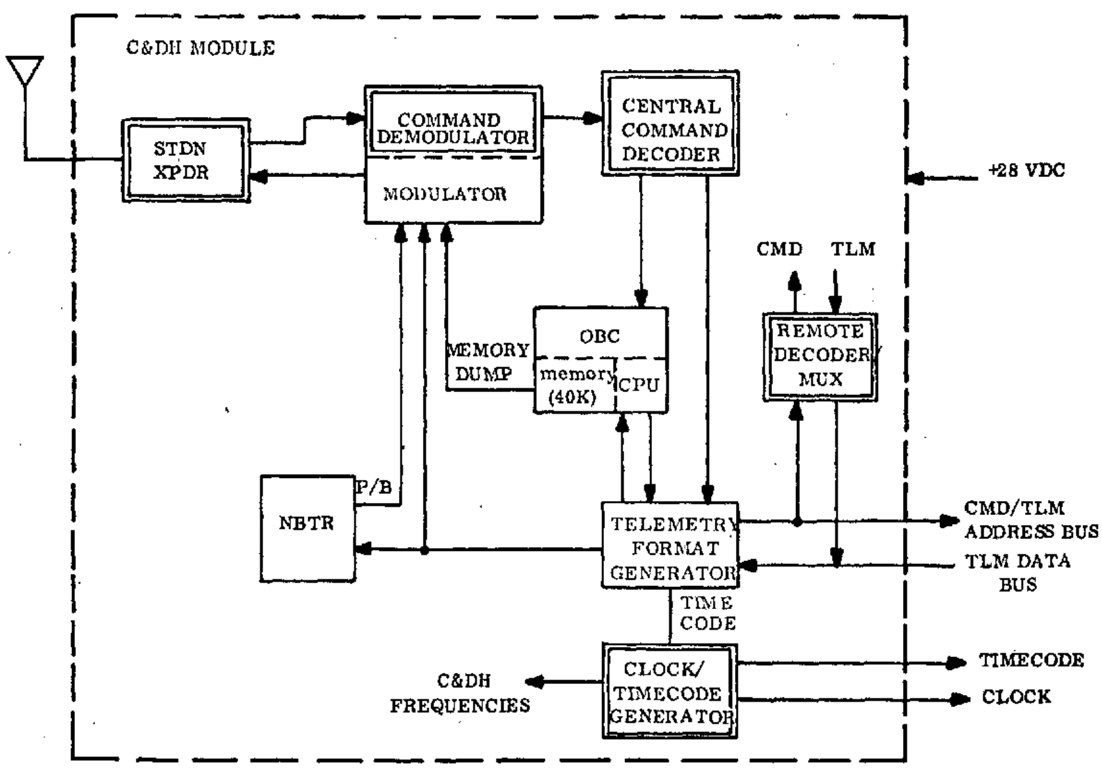


Figure 3-1. C&DH Module

3.2.2 C&DH CHARACTERISTICS

Command Rate: 50/sec (uplink); 31450/sec (OBC)

Command Type:

Pulse Commands -

#/Remote Decoder	64
Level	+5.0 ± TBD V
Duration	4.0 ± TBD MS

Serial Magnitude Commands -

#/Remote Decoder	4
Command Data	16 Bits (NRZ)
Command Clock	16 kHz
Command Envelope	16 Bit Times

Command Storage: Any command; 1 sec resolution

Telemetry Rate: 32,000 words/sec (all to OBC; 500/sec to ground)

#/Remote Mux	64
Types	
Analog	Digitized to 8 Bits
Bi-Level	Groups of 8 one Bit inputs
Serial Digital	16 8-Bit words (maximum/remote)
Input Level	0 to +5 VDC

Clock: 1.6 MHz balanced output; 1 v p-p

Timecode: 32 Bit elapsed time counter; LSB = 1m/sec

3.2.3 C&DH INTERFACE INFORMATION REQUIRED

To achieve compatibility with the Command and Data Handling Subsystem the instrument characteristics defined in Table 3-3 are required.

Table 3-3. Instrument Characteristics Required by C&DH

Commands
Number (per type)
Type (pulse or serial magnitude)
Interface Circuit Schematic
Storage Requirements
Telemetry
Number (per type)
Type
Sample Rate
Usage
Interface circuit schematic
On-Board Computer
Input Definition
Calculation/Decision
Output Definition
Clock Interface Circuit
Timecode Interface Circuit

3.3 INSTRUMENT MODULE SUBSYSTEM

3.3.1 IMS DESCRIPTION

The EOS spacecraft will ultimately be designed to mount instruments in individual modules for resupply using Shuttle as illustrated on Figure 3-2. In this arrangement the instruments are each installed in a module specifically tailored for the instrument and incorporating four corner latch mechanisms for attachment to the spacecraft. Pull-away electrical disconnects will be provided for automatic electrical mating during installation or resupply.

An alternate attachment for early EOS missions is illustrated on Figure 3-3. This method attaches the instrument to a fixed mount structure integral with the spacecraft. The fixed mounting will be required for Delta launched missions where weight and space limitations would preclude use of modules and resupply mechanisms.

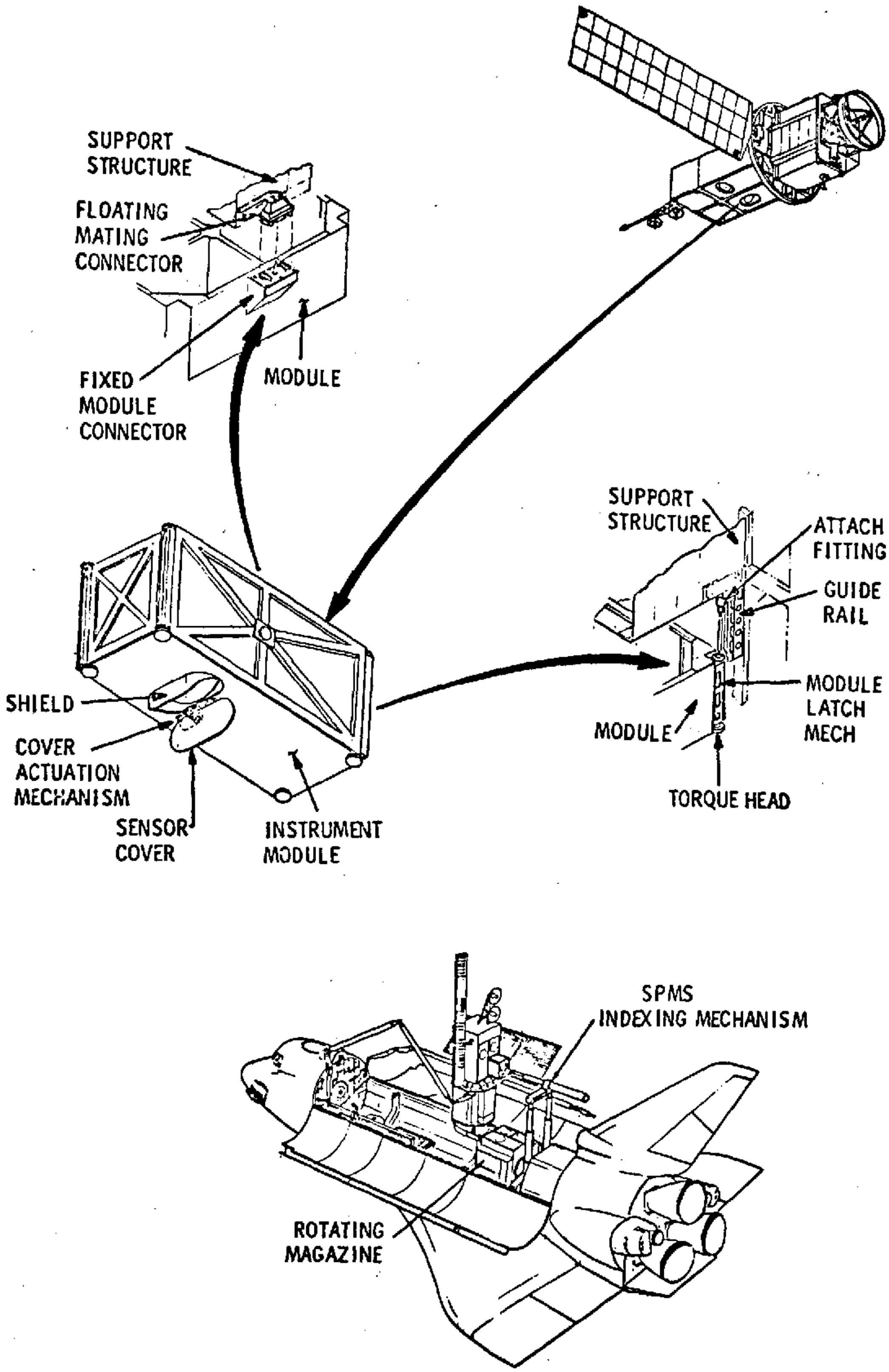


Figure 3-2. EOS Instrument Support
(Resupply Configuration)

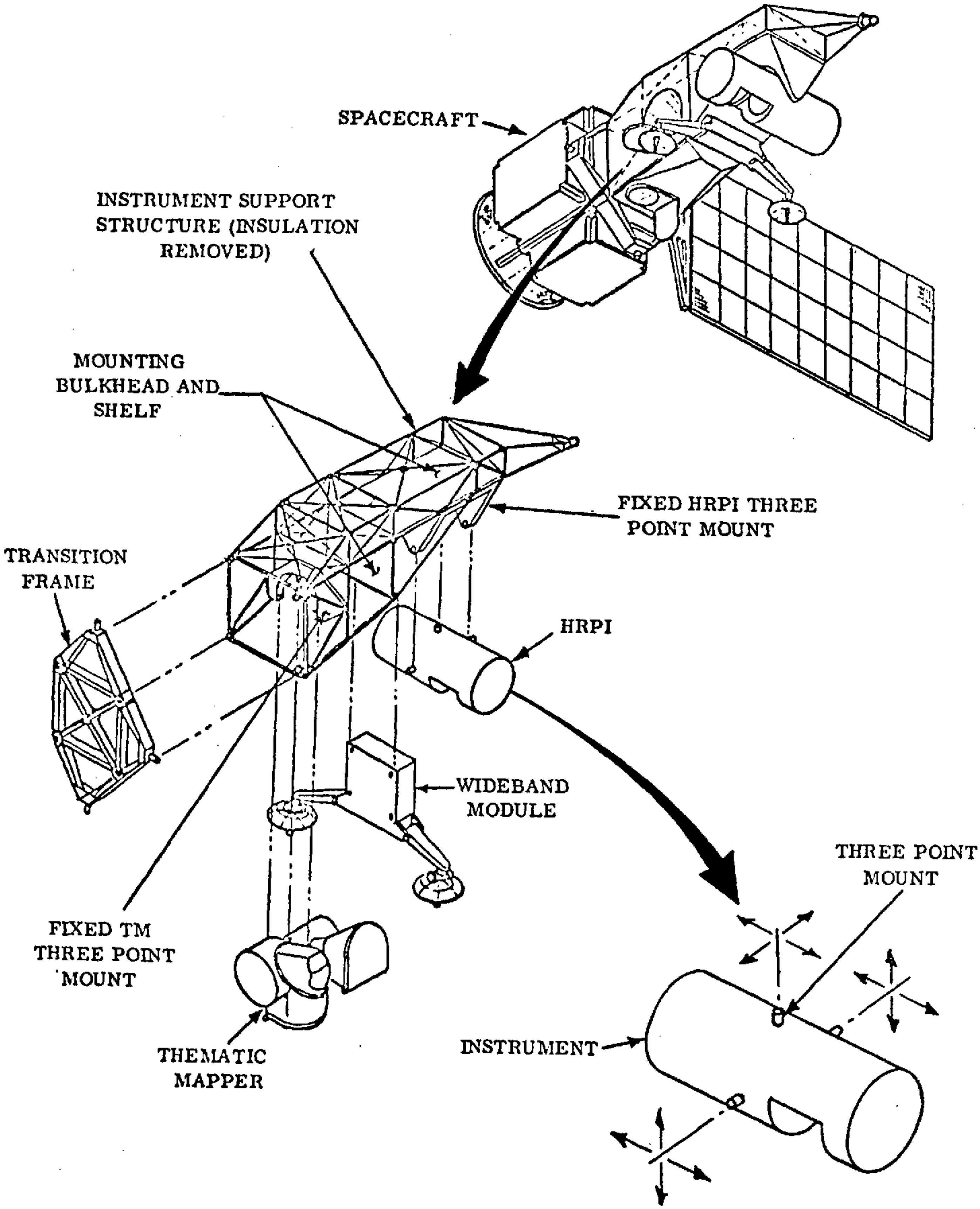


Figure 3-3. Instrument Support (No Resupply Configuration)

In either case the instrument module or support structure will be designed to accommodate a particular instrument mount arrangement and electrical interface connectors will be provided as required.

3.3.2 INSTRUMENT MODULE CHARACTERISTICS

3.3.2.1 Instrument Mounting

Three point trunion mounting systems are recommended for each instrument as shown in Figure 3-4. These three point systems result in determinate planer reactions for the instruments and minimize induced strain into the instrument from support structure deformations. Alternate mounting schemes may be accommodated but will require additional design and analysis for verification.

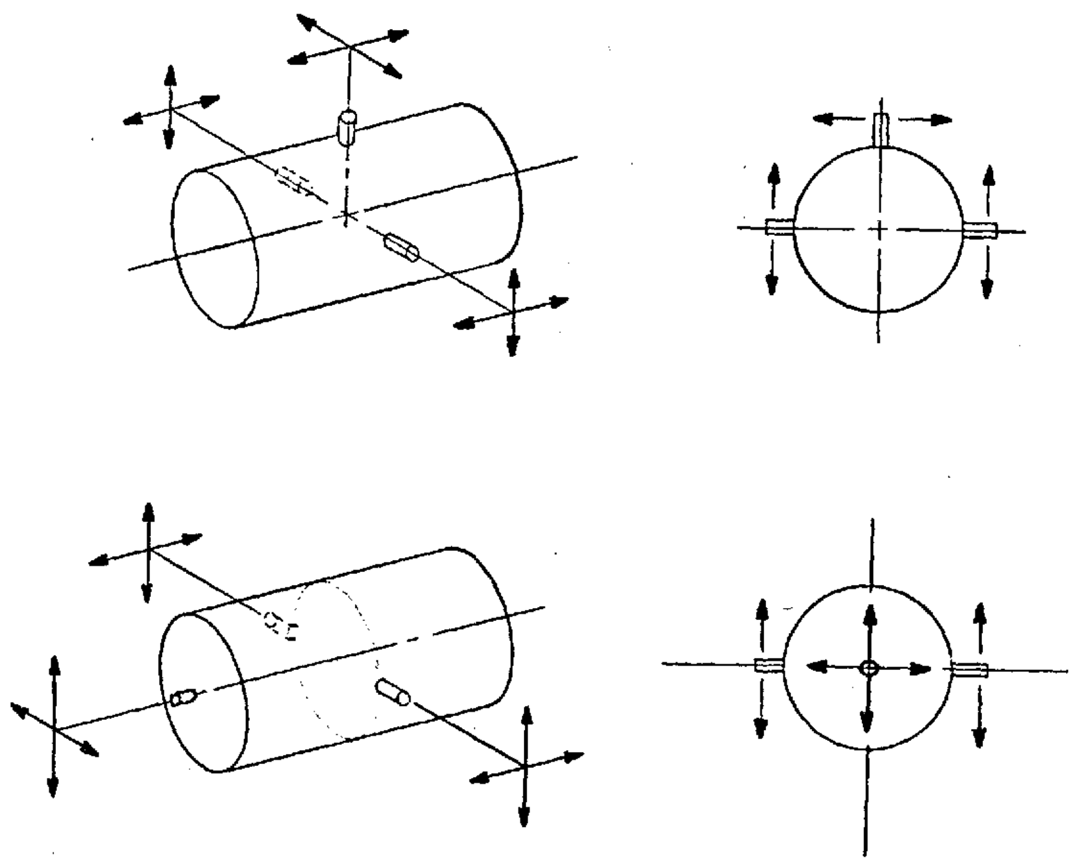


Figure 3-4. Three Point Reaction Systems

3.3.2.2 Thermal Control

Each module or instrument compartment will be designed to provide thermal isolation for the instrument and either the instrument or module will be individually insulated and have independent external heat rejection provisions. The instruments will in general radiate heat from the earth viewing surface and coolers will be oriented outboard on the spacecraft anti-sun side as shown in Figure 3-5.

Instrument temperatures will be maintained by multi-layer insulation and passive heat rejection if possible, however guard heaters may be required for a particular installation.

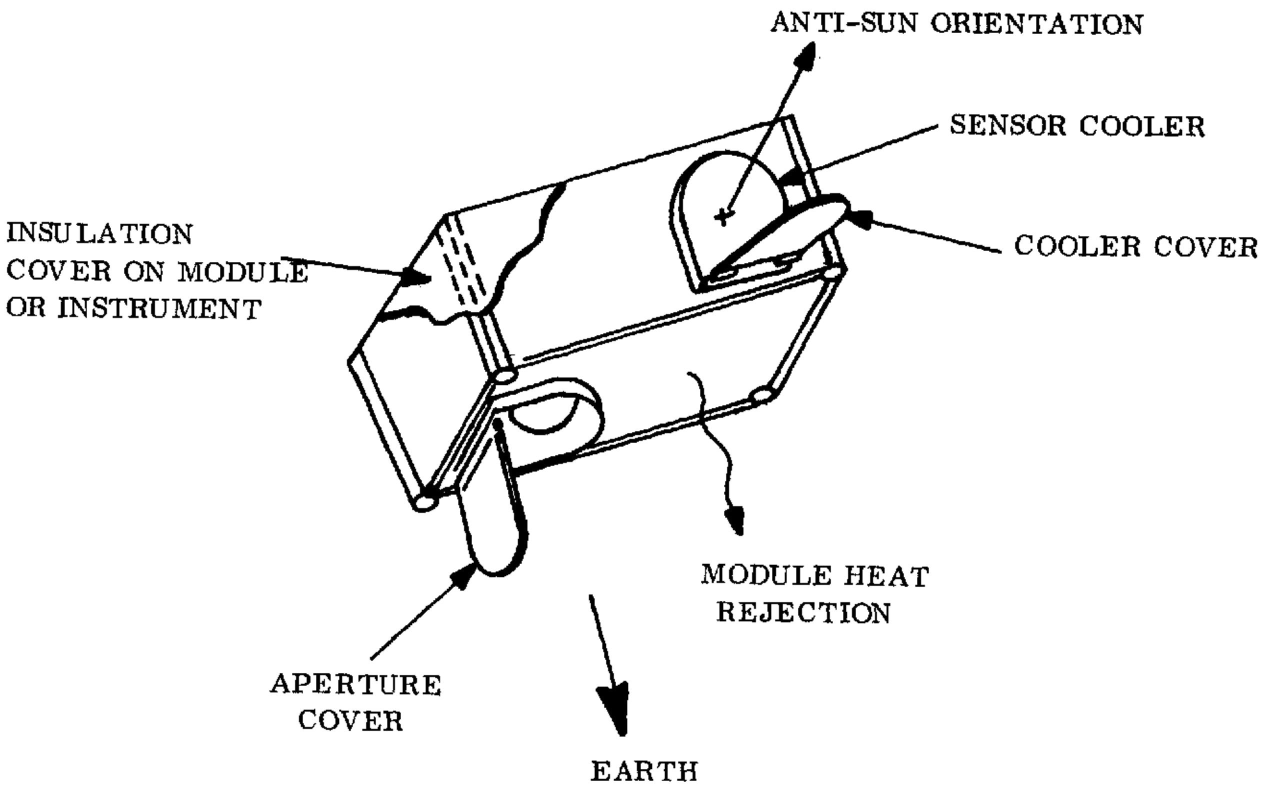


Figure 3-5. Instrument Thermal Control

3.3.2.3 Instrument Modules

A typical instrument module design is shown on Figure 3-6. The module is an aluminum truss structure supporting the instrument at a central three point mounting interface. The lower (earth viewing) surface of the module is open and the sensor cover shown could be integral with the instrument or designed as part of the module. Pull-away electrical disconnects are located on the inboard module surface as required and the module is attached to the structure at the four corners using Shuttle activated latch mechanisms.

Modules of this type will be tailored for each instrument and their design will be dependent on the individual instrument mounting, orientation and field of view requirements.

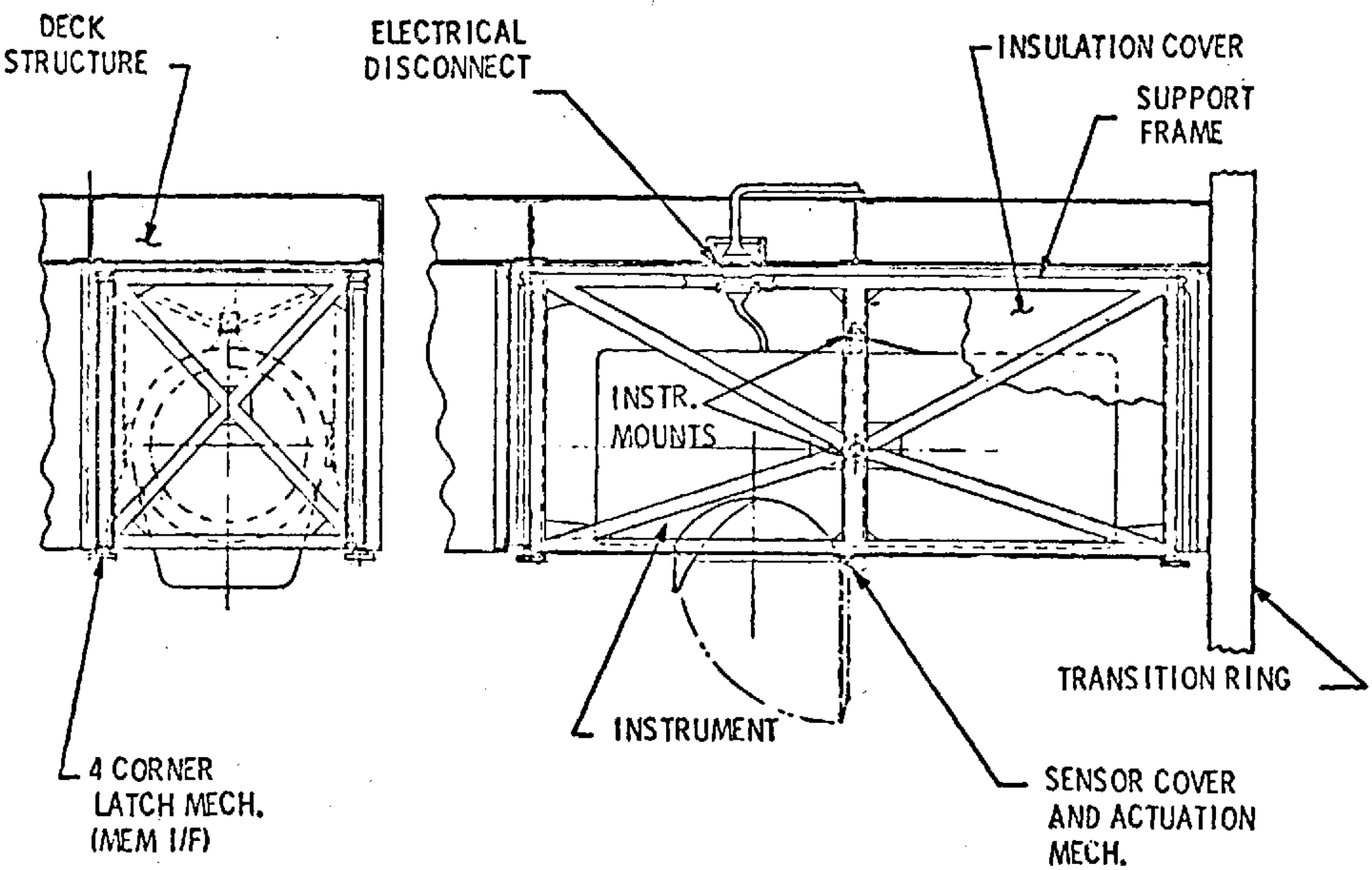


Figure 3-6. Instrument Module

3.3.2.4 Installation Limitations

Allowable Instrument envelopes are determined by the available launch vehicle shroud dimensions for the fixed mounting configurations, and by the module storage and handling limitations for the modular arrangements. The maximum envelope shown on Figure 3-7, is preliminary and should be used as a guide in instrument design. Once a particular instrument complement is selected some latitude in these dimensional limits may be accommodated. Note that this envelope includes all instrument peculiar appendages including coolers and retracted aperture covers.

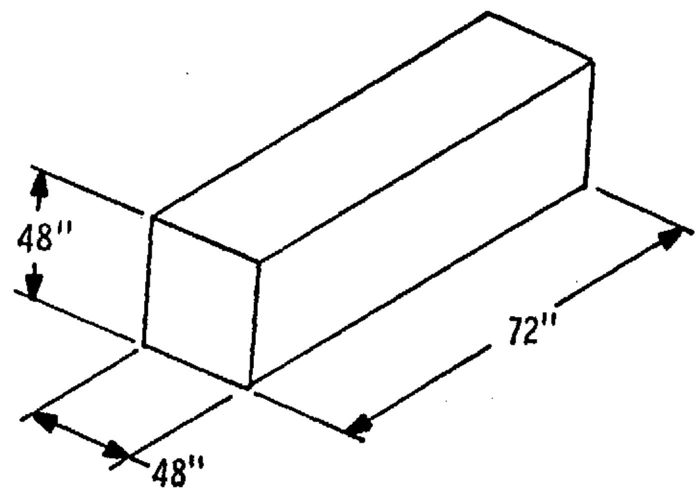


Figure 3-7. Instrument Maximum Envelope

Instrument aperture and cooler fields of view will be defined by the instrument contractor and will be determining factors in final instrument placement and module or support structure design.

Weight, center of gravity, and mass properties of each instrument must be supplied by the instrument contractor.

3.3.2.5 Design and Environmental Requirements and Criteria

Structural criteria will be supplied by the spacecraft contractor for use in design and verification of the instruments. Preliminary spacecraft criteria are presented in Table 3-4 and will be refined dependent on systems analysis, launch vehicle selection and program test philosophy. These values do not include local amplification factors for the instruments.

Table 3-4. Spacecraft Structural Criteria

Launch System		Spacecraft Qualification Test Levels (1.5 x Expected level)								S/C Load Factor	S/C Ultimate Design Load (G's)	
		Acceleration (G's)		Random Vib. (G RMS)	Max. Sine Vib. (G's)		Acoustics db	Shock Resp. (G's Max.)				
		Thrust	Lateral		Thrust	Lateral					Thrust	Lateral
Delta		-18.0	± 3.0	11.3	6.8	2.0	144	1700		1.25	-22.5	3.75
Titan IIID		- 9.0	± 2.6	16.9	3.0	2.0	147	3500		2.0	-18.0	5.2
Shuttle	L/O	-3.45	1.28	7.9 to 24.3	TBD	TBD	143 to 149	TBD		2.0 (1.2 crash)	-6.9	2.56
	B/O	-4.95	.81								-9.9	1.61
	Entry	± .38	4.56								± .76	9.12
	Ldg	±2.25	4.37								±4.5	8.74
	Crash	+9.0	4.5								+10.8	5.4

Thermal control requirements will be determined for each instrument and supplied by the Spacecraft contractor. It is currently planned to hold instrument temperature to 70 ± 5°F nominally.

Alignment requirements are under evaluation and are dependent on the operational system characteristics. Initial studies indicate ground alignment within ±0.10 degrees are tolerable. Orbital variations in alignment due to thermal gradients are more critical and will dictate close control of both temperature levels and gradients in the instrument and supporting structure.

3.4 POWER SUBSYSTEM

3.4.1 DESCRIPTION

The EOS power subsystem is a regulated direct energy transfer system which provides a +28 ± 0.3 VDC output. Regulation is obtained by controlling discharge boost converters, battery charge regulators, and a partial shunt regulator. Separate power lines are provided to each user subsystem. Current sensing and protection are provided within the power module for each power output line.

3.4.2 POWER SUBSYSTEM CHARACTERISTICS

The power subsystem characteristics are delineated in the following table.

Table 3-5. Power Subsystem Characteristics

Parameter	Value
Voltage	
Operating	+ 28 ± 0.3 VDC
Fault	$\leq$ + 45 VDC @ 100 μv/sec $\leq$ -10 VDC @ 250 μv-sec
Output Impedance	$\leq$ 0.1 ohms, DC to 10 kHz
Noise (Output)	$\leq$ 100 mv p-p
Load Transients	$\leq$ + 2 VDC @ 100 μv-sec
Line Drop	$\leq$ 280 mv @ 100 W $\leq$ 500 mv @ > 100 W
Bus Protection	Contained in power module

3.4.3 REQUIRED INTERFACE INFORMATION

Compatibility with the power subsystem requires that the instrument information listed in Table 3-6 be provided.

Table 3-6. Instrument Information Required By Power Module

Parameter
Power Demand (by mode)
Power Input Filter Schematic
Grounding Diagram
Noise (Feedback)
Instrument Stabilization Time
Current Transients (amplitude and rise time)

3.5 SIGNAL CONDITIONING AND CONTROL MODULE (SCCM)

3.5.1 DESCRIPTION

The Signal Conditioning and Control Module is a mission peculiar module which provides circuitry for arming and firing pyrotechnic devices, driving solenoids, controlling mechanisms, or conditioning signals which cannot be conditioned locally. Instrument requirements for non-standard circuitry must be known at an early stage of program development in order to permit incorporation into the SCCM design.

3.5.2 SCCM CHARACTERISTICS

The signal conditioning and control module characteristics are listed in Table 3-7.

Table 3-7. SCCM Characteristics

Parameter	Value/Limit
Pyrotechnic Firing Circuits	Value/Limit
# Available	10
Characteristics	+28 VDC
	10 amps
	100 msec

3.5.3 SCCM INTERFACE INFORMATION REQUIRED

Requests for SCCM circuitry shall include the information shown in Table 3-8.

Table 3-8. Instrument Information Required By SCCM

Function
Pyrotechnics
Number
Function Performed
Solenoid Drivers
Number
Function Performed
Special Circuits
Type
Number
Interface Characteristics
Duty Cycle
Resolution
Timing Accuracy

3.6 WIDEBAND DATA HANDLING SUBSYSTEM (WBDHS)

3.6.1 DESCRIPTION

A block diagram of the WBDHS is given in Figure 3-8. The WBDHS is capable of processing two 120 Mbps input data channels. Both inputs are serial digital with the mux plus A/D functions performed in remote encoders located with the instruments. The 120 Mbps data streams are also selectively compacted in the WBDHS to 20 Mbps for transmission to low cost ground stations. All data are capable of being stored on wideband tape recorders and are transmitted at X-band.

3.6.2 WBDHS CHARACTERISTICS

The characteristics of the Wideband Data Handling Subsystem are given in Table 3-9.

Table 3-9. WBDHS Characteristics

Parameter	Value
Data Compactor Characteristics	TBD
WBVTR	
Record Time	TBD
Input Data Requirements	TBD
PCM/FM Modulator Characteristics	
Input Data Requirements	TBD
RF Frequency	$\sim$8 GHz
RF Output Power	
QPSK	2 W
LCU	30 W
PCM/FM	0.4 W

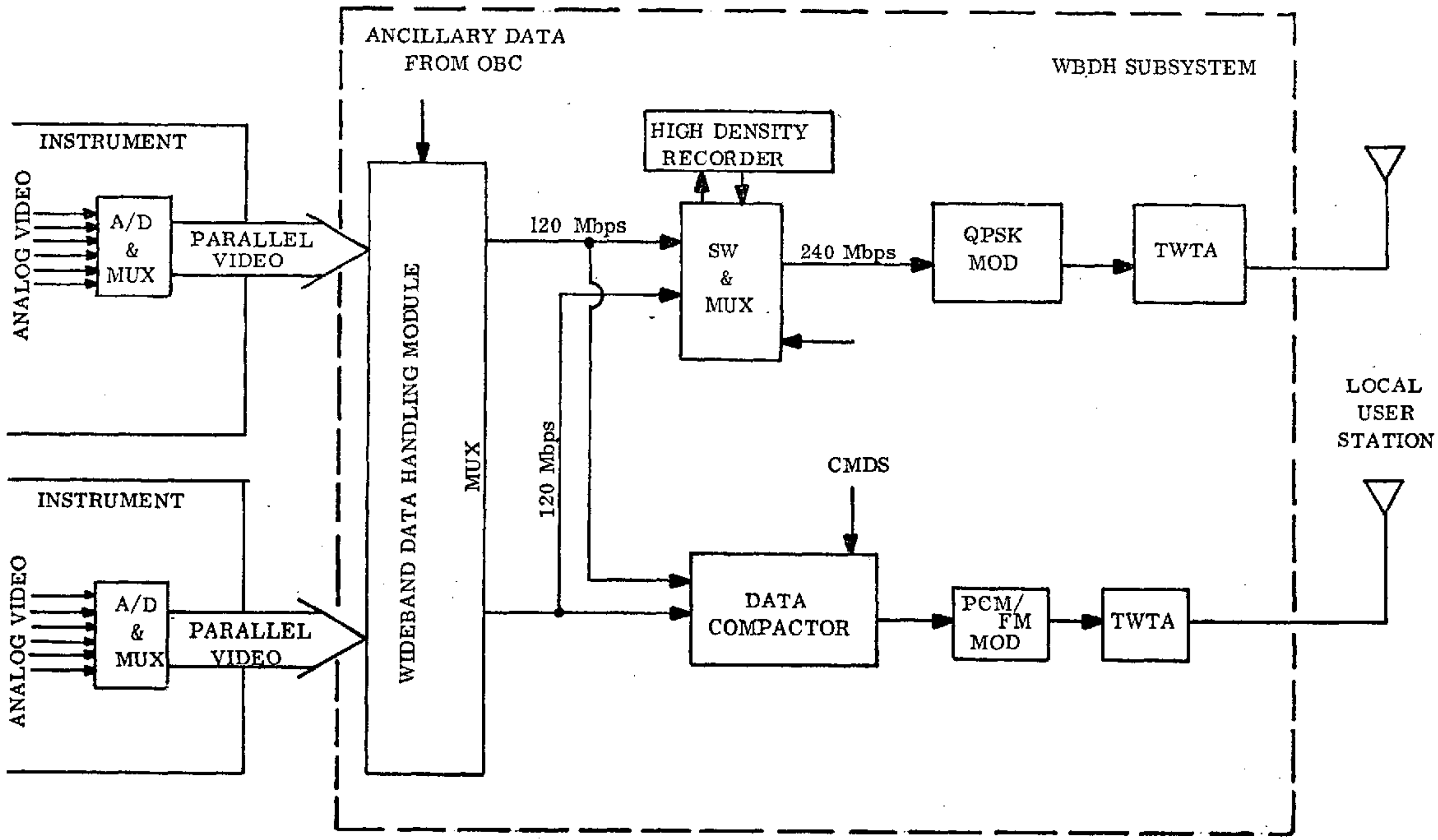


Figure 3-8. WBDH Subsystem Block Diagram

Table 3-10 indicates the characteristics of the Remote Encoder/Multiplexers which are loacted within the instrument modules.

3.6.3 WBDHS REQUIRED INFORMATION

The interface information required by the Wideband Data Handling Subsystem is listed in Table 3-11.

Table 3-10. Remote Encoder/Multiplexer Characteristics

Parameter	Value
Analog Inputs	
Number	100
Range	0.0 to 4.0 V
Input Impedance	TBD
Output	
Quantization	TBD
Sample Rate	
Clock Rate	
Digital Inputs	
Number	
Level	
Impedance	
Size	
Weight	
Power	

Table 3-11. Instrument Information Required By WBDHS

Parameter	Limit
Analog Inputs	
Number	<100
Voltage Range	$<+4$ volts
Signal Frequency	$<$TBD
Source Impedance	$<$TBD
Noise (rms)	$<$TBD
Digital Inputs	
Number	TBD
Voltage Levels	
Pulse Width	
Pulse Repetition Rate	
Source Impedance	
Noise (rms)	
Time annotation requirements	
Special Data Requirements	

SECTION 4.0

INTEGRATION AND TEST PROGRAM

The Integration and Test Program for EOS missions is based upon a definitively instrumented, extensive test program for the first EOS spacecraft coupled with analytical extension of these test results to subsequent mission configurations wherever feasible. The integration and test programs for EOS-A and subsequent EOS missions are shown in Figure 4-1.

4.1 INTEGRATION AND TEST PROGRAM DESCRIPTION

4.1.1 DEVELOPMENTAL TESTING

The developmental test program is configured around four spacecraft models: a Structural Dynamics Model (SDM); a Bench Integration Test (BIT) Model; an Antenna Model (AM); and a Harness Mockup Model (HMU). The structural dynamics model will consist of a full scale primary and secondary structure of the spacecraft including mass models of all major components or assemblies installed in their respective flight configurations. The SDM will be utilized to confirm dynamic analytical models, demonstrate structural integrity of the design, confirm the internal dynamic environments for subsystems and components, confirm the dynamic envelope within the fairing, confirm separation clearances, confirm spacecraft and mechanical AGE compatibility, and to develop dynamic environment test techniques for the proto-flight spacecraft. Test usage of this model includes launch and orbital vibrations, static load or steady-state acceleration, and shock. The SDM will be maintained and modified, as required, throughout the EOS program for the purpose of verifying structural changes on follow-on spacecraft.

The Bench Integration Test (BIT) model will integrate the spacecraft electrical subsystems early in the program to provide an evaluation of system electrical and RF compatibility. It will also serve to checkout electrical test equipment compatibility, test ground station operation, establish test procedures and checkout test software sequences. The model will consist of engineering or flight models for all electrical

	ENGINEERING MODEL	PROTOTYPE MODEL	PROTO/FLIGHT MODEL	FLIGHT MODEL
Component	Developmental Tests Electrical Mechanical Performance	Qualification Tests Electrical Mechanical Performance Environmental		Functional Performance
Subsystem	Developmental Tests Electrical Mechanical Performance	Qualification Tests Electrical Mechanical Performance Environmental		Acceptance Tests Functional Performance Environmental Test
System	Developmental Tests BIT Test SDM Test AM Tests HMM EOS-A and B,C.. as Req.		Acceptance Tests Functional Performance Environmental EOS-A	Functional Performance EOS-B,C...

New Designs: Engineering and Prototype Models

Proven Designs: Flight Models

Figure 4-1. EOS Test Program

simulation of the EOS spacecraft. The interconnecting harness will duplicate wire size, number, shielding and connections of the flight spacecraft system harness. A spacecraft structure will not be utilized. The subsystems will be integrated on a specially shaped bench. The BIT model will be maintained and updated throughout the EOS program and used as a test bed for new experiments.

Antenna model spacecraft are normally provided on new designs. These models are checked out on antenna ranges to assure that the gain and patterns are consistent with the design requirements. These are made of material that provides the RF characteristics of the spacecraft, however, they do not require prime type material in most areas.

A harness mock-up is used as a development tool and is made to prime dimensions. Mock-up harnesses are then assembled in place until the placement of all harness segments is completed. These segments are then removed and three-dimentional boards made up from the mock-up harnesses. All flight harnesses are then fabricated in the proper configuration on these boards. This extremely useful model eliminates the extensive handling required to mount prime hardware in the proper configuration.

For all new design components, performance verification will be obtained by breadboard testing. After the adequacy of basic design has been verified an engineering unit will be built, verifying packaging and fabrication techniques. A functional check will be completed prior to installation on the appropriate engineering subsystem. Following performance verification the engineering subsystem will be integrated with the BIT model for system evaluation.

Wherever possible previously flight-qualified hardware will be selected. For these components there will be no breadboards or engineering units. The first available flight unit will be mounted in its engineering subsystem. This unit will remain in the engineering subsystem on the BIT model until the final flight when it will be mounted in the final flight subsystem.

4.1.2 QUALIFICATION TESTING

Spacecraft hardware intended for use in all EOS missions will be qualified during the EOS-A program to levels based on the most severe environment anticipated. On EOS-A these qualification levels will apply at the component level for all new design and previously unqualified components. Previously qualified hardware will be evaluated on an individual basis to determine whether requalification is necessary. Minor modifications to previously qualified hardware will be reviewed for the extent of design changes and an abbreviated, full or no requalification test will be recommended based upon the impact of design changes on prior qualification.

Qualified components will be assembled into subsystems for performance verification. At the subsystem level no environmental qualification testing will be performed for EOS-A since system level environmental testing will be performed. For subsequent missions, acceptance environmental testing will be completed at the subsystem level with only workmanship vibration at the spacecraft level.

At the system level the spacecraft will be proto flight qualified. Proto flight qualification is defined as testing to qualification levels for flight duration requirements. The proto flight qualification test sequence will double as the acceptance test sequence on EOS-A. Only EOS-A will have a full-scale spacecraft environmental test sequence.

4.1.3 ACCEPTANCE TESTING

At the component level, the environmental qualification test cycle will be the acceptance test cycle for EOS-A. On subsequent missions component acceptance testing will consist of ambient performance tests with no environmental testing. These tests will be performed at the contractor's facility. There will be no acceptance testing at the prime contractor's facility.

Since the qualification hardware will be launched on EOS-A, that mission will have no subsystem acceptance test cycle. On subsequent missions, subsystem acceptance testing will consist of performance verification and a flight level environmental test sequence.

Verification of flight readiness at the subsystem (module) level is necessary if modules are to be interchangeable at any point in the test program with minor effect on the test schedule. Eventual in-orbit module replacement also dictates a heavier emphasis on subsystem testing.

As previously stated there will be no acceptance testing per se for the EOS-A spacecraft. The EOS-A qualification test cycle will satisfy all acceptance test requirements. On subsequent missions the flight general purpose subsystems will be assembled on the spacecraft and functionally tested. When the experiments are delivered and installed, spacecraft electrical and mechanical functional testing along with a workmanship vibration will satisfy launch readiness and acceptance test requirements.

4.2 INTEGRATION AND TEST PROGRAM CHARACTERISTICS

The integration and test program requires that the performance of a module be determinable at any point in the system test sequence. A typical sequence is shown in Figure 4-2.

4.3 INTEGRATION AND TEST PROGRAM - REQUIRED INTERFACE INFORMATION

The interface information required for the Integration and Test Program includes, in addition to the unique test information, the information required by the operations control system and the ground data handling systems as they form an integral part of the test program. The interface information is listed in Table 4-1.

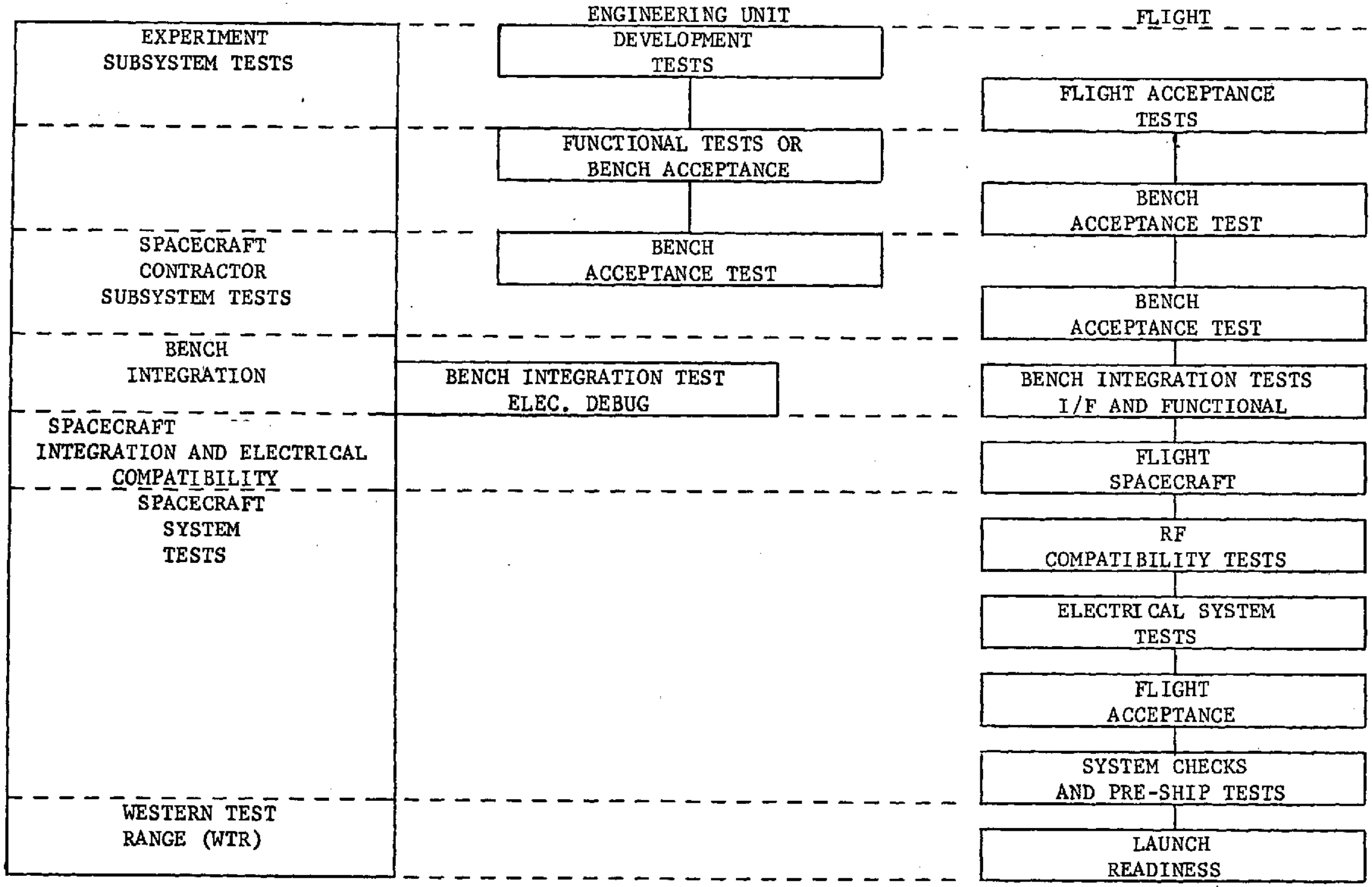


Figure 4-2. Typical Test Sequence

Table 4-1. Instrument Information Required By The Integration and Test Program

I&T Program Requirements

- Function Test Procedure (including limits)
- Required Support Equipment definition
  - Bench Test Equipment
  - Handling Equipment
  - Stimulation Equipment
  - Special Equipment (if any)

Operations Control Requirements

- Instrument Manual
  - Operating Modes
  - Command definition
  - Telemetry definition
  - Operating Restraints
  - Functional Test Results

Ground Data Handling System Requirements

- Data Processing Parameter Manual
  - Data Format
  - Geometric Correction Inputs
  - Radiometric Correction Inputs
  - Ancillary Data Requirements
  - Performance Test Data
    - Calibration
    - MTF
    - SNR
    - Field of View
    - Scan Performance
    - Stability
    - Dynamic Range
    - Internal Alignment
    - Focus

SECTION 5

GROUND DATA HANDLING SYSTEM

The EOS Ground Data Handling System consists of two major segments: the Operations Control Center (OCC) and the Central Data Processing Facility (CDPF). This system is illustrated in Figure 5-1. These two segments will provide for all control of the spacecraft and will process the vast majority of image data.

The EOS-A spacecraft will also transmit a portion of the instrument data directly to users at widely distributed low cost readout stations. These stations also form a portion of the EOS ground system but are essentially independent of the OCC and CDPF.

5.1 OPERATIONS CONTROL CENTER

The Operations Control Center performs the functions of spacecraft command and control, telemetry data acquisition and telemetry data processing and reduction. The functions and characteristics are summarized in Table 5-1.

In order to perform its function of spacecraft command and control, the OCC requires a data base for each spacecraft subsystem. It is anticipated that this information will be in the form of a command and telemetry compendium. The compendium will contain a functional description of the subsystem with particular emphasis upon: (1) the functional effect of each command, including method of verification, resulting instrument mode, and effect upon data obtained; (2) a description of each telemetry function, its derivation, conversion to engineering units, safe operation limits and the information relating to subsystem performance that is monitored by the function; and (3) operational restraints or limitations upon instrument usage such as warm up times, forbidden modes or command sequences and status determinations. This data must also be used prior to launch during integration and test and has been identified in Table 4-1.

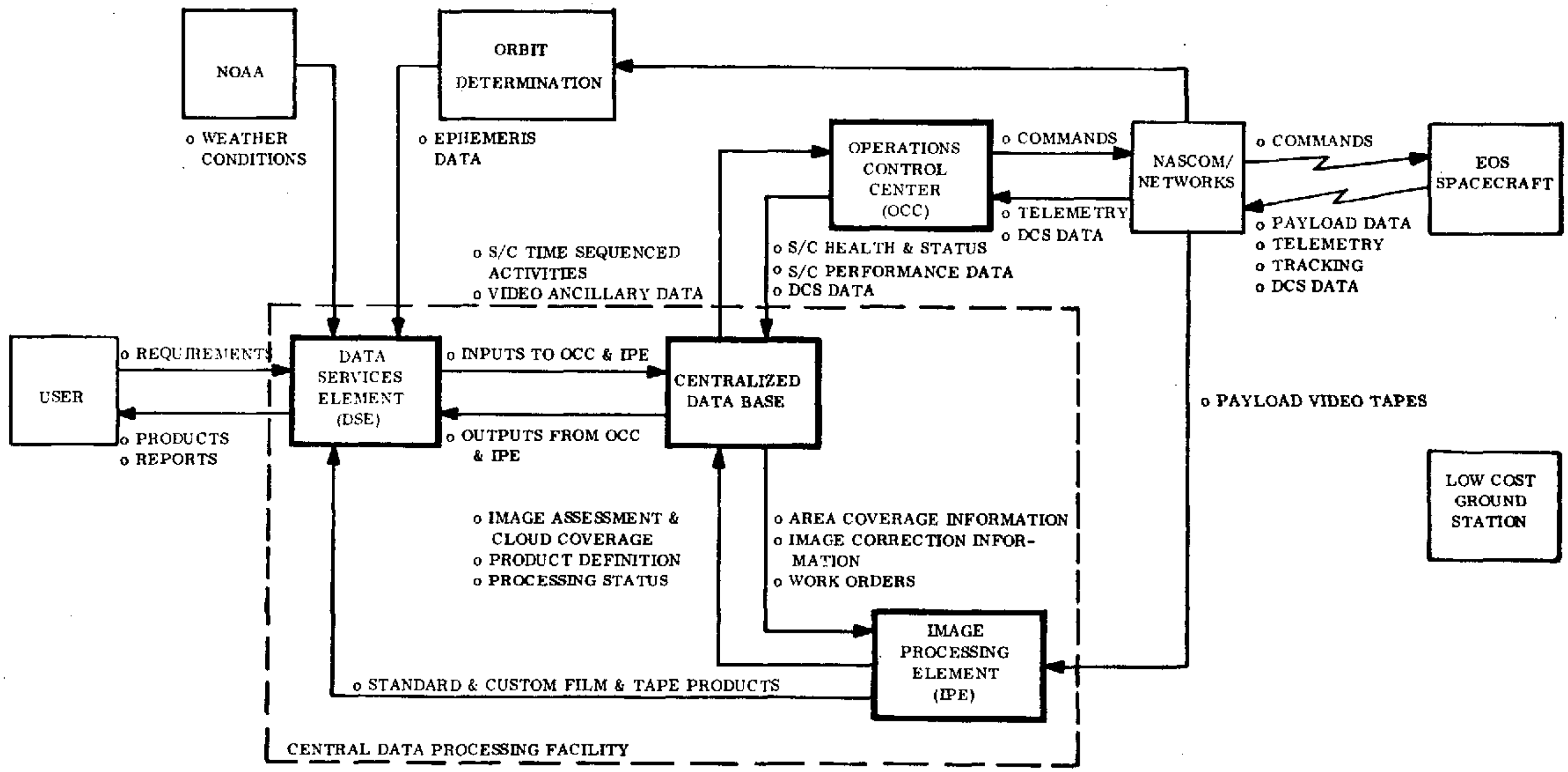


Figure 5-1. EOS Ground Data Handling System

Table 5-1. OCC Characteristics

Functions	Inputs	Outputs
1. Spacecraft command and control 2. Spacecraft telemetry retrieval and processing 3. Determination of spacecraft health and status 4. Generation of displays and reports 5. Command generation 6. Control remote station contact schedule	From Networks 1. Telemetry data via the NASCOM 2. Data Collection System inputs 3. Voice and teletype communications From DSE 1. Schedule of all planned spacecraft activities 2. Ground control point information, calibration data and predicted ephemeris data for inclusion in the video data	To Networks 1. Commands for controlling the S/C 2. Ground control point, ephemeris, calibration, and other auxiliary data to be transmitted to S/C for inclusion in video data. To DSE 1. Spacecraft and ground station configuration and status as an input for the scheduling function. 2. Spacecraft performance data and the data which was actually acquired. 3. DCS data to be used by DSE in generating DCS products.

5.2 CENTRAL DATA PROCESSING SYSTEM

The Central Data Processing System receives raw multispectral data and performs those calibrations, corrections and formatting functions which are necessary to quantatively restore the original fidelity to the data.

The EOS-A ground system is designed to process and correct both Thematic Mapper and HRPI instrument data received in raw form on video tapes and produce output products in the form of High Density Digital Tapes (HDDT's), Computer Compatible Tapes (CCT's), transparencies and prints. All processing and correction will be accomplished in the digital domain to achieve the desired output product accuracy requirements and to satisfy the needs of a user community that is increasingly using digital extractive processing techniques to derive information from the data.

The system design has been configured to perform all functions and meet all requirements utilizing a standard on-line and custom off-line processing approach. The standard on-line preprocessing and image correction functions (consisting of data reformatting, quality assessment, screening, radiometric and geometric correction, initial archival HDDT generation, and film cataloging) are performed on all valid data. The remaining functions (CCT generation, HDDT copying, film production, extractive processing, and browse capability) are performed on a custom off-line basis and are performed only on selected data on request.

The system level error allocations made to the various subsystems define the characteristics of the input data and the system performance requirements determine the output product quality. Together, these determine the type of corrections which must be applied to the data. All data, regardless of the geometric accuracy, will be corrected to the same excellent radiometric quality. All the information necessary to implement this correction is contained in the video data. The geometric accuracy of the correction process, a major cost driver in the total system, is a function of the data utilized in calculating the correction function. The most stringent geometric accuracy requirement is met using ground control points to model uncertainties in knowledge of error sources.

The characteristics of the instruments affect the cost and complexity of only those functions which must be performed to correct the data. A modular design approach has been adopted for the ground system which makes the hardware design relatively insensitive to the parameters of the various instruments. A brief description of the major ground processing functions is given below indicating how they are impacted by the instruments.

Data Reformatting. The format of the input data has a considerable impact on the cost and complexity of the central data processing subsystem. The format is determined primarily by the sensor focal plane configuration, combined with the data sampling and multiplexing strategy employed in the wideband module. The input serial data stream is reformatted from its non-optimum arrangement of pixels to produce one that is band to band registered, spectrally interleaved (all bands) and linearized (all pixels equally spaced along a straight line and in sequence).

Geometric Correction. A major cost driver in the processing of the data on the ground is the stringent geometric mapping accuracy requirements. The instruments are major contributors to the geometric inaccuracies which exist in the data, primarily the high frequency internal distortions. The approach to maximum throughput in the ground processing is to insert into the composite video stream all the information necessary (except best fit ephemeris) to correct the data. For geometric correction, this includes:

- o scan nonlinearity profiles
- o detector offsets
- o line linearity
- o sampling nonlinearities
- o attitude position and rate
- o predicted ephemeris
- o earth rotation effect
- o earth curvature effects
- o boresight alignments
- o time

All data produced has a geometric accuracy falling into one of the following categories:

- o Uncorrected Data - 450 Meter Accuracy
  - Utilizes Predicted Ephemeris
  - Performs X Correction of Each Scan Line (line length, earth rotation, scanning/sampling/array non-linearities, earth curvature and best fit planar projection)
  - All Data Linearized to Straight Lines
- o Uncorrected Data - 170 Meter Accuracy
  - Utilizes Best Fit Ephemeris
  - Performs X Correction on Each Scan Line (same as uncorrected data - 450 meter accuracy)
  - All Data Linearized to Straight Lines

- o Corrected Data - 15 meter accuracy
    - Utilizes Best Fit or Predicted Ephemeris
    - Performs X, Y Correction of all Error Sources
    - Uses Ground Control Points (CCP's) to Model Errors
    - Data Presented in Specified Map Projection
    - Data Gridded with Respect to the Earth

Radiometric Correction. All output data, regardless of its geometric accuracy, will be corrected to the best achievable radiometric fidelity. The EOS-A requirements on radiometric mapping accuracy of output products have a significant cost impact on the central data processing system, particularly for a pushbroom array instrument configuration. The major contributors to the inaccuracies which must be removed from the raw data are the instruments and the A/D converter in the wideband module. As was the case for geometric correction, the approach to maximize throughput is to insert all the ancillary correction information into the data stream from the spacecraft. In addition, a video histogram analysis approach can be utilized if internal sensor calibration devices fail. The ancillary data is:

- o Initial calibration lamp data utilized to remove detector banding and short term instability.
- o Sun calibration data provided to remove long term instabilities,
- o Failed detector annotation required to compensate for necessary data stripes.

All of the instrument parameters which affect the ground processing system must also be available during integration and test prior to launch. These required parameters have already been identified in Table 4-1.

5.3 LOW COST READOUT STATIONS

The EOS-A spacecraft will transmit a portion of the instrument data directly to users at many Low Cost Readout Stations (LCRS). This data will be derived from the Thematic Mapper and the High Resolution Pointable Imager, processed by the spacecraft compactor, and then transmitted. The compactor has several modes in which it can operate, with different modes being used at various times depending on the particular needs of the Local User. The modes include:

- o Reduced number of bands
- o Reduced swath width
- o Reduced resolution

These modes result in a data rate to the Low Cost User of approximately 20 Mbps.

Depending also on the particular user's application of the data, he will implement various levels of radiometric and geometric correction in the Low Cost Readout Station. The corrections required can range from none all the way to corrections which provide output product quality nearly equivalent to that provided in the central data processing facility. The LCRS has been configured to process all data in the digital domain. Figure 5-2 illustrates the data flow through the LCGS.

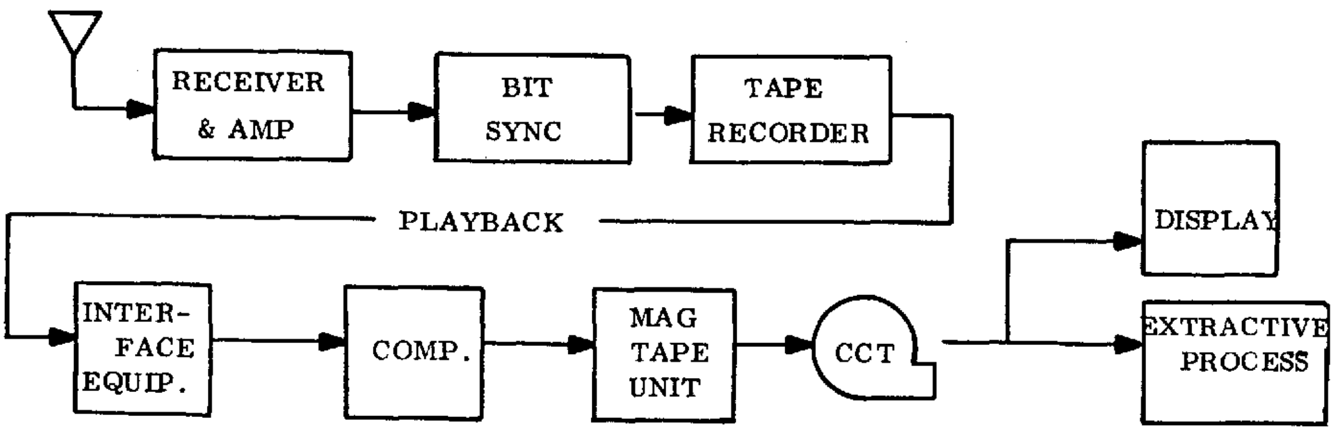


Figure 5-2. Low Cost Readout Stations Block Diagram

In general, the LCGS will receive and process between one and nine 185 x 185 Km scenes over any given 17 day repeat cycle period. Data will be recorded on video tape with multiple playbacks at lower speeds used to buffer the data to rates compatible with the digital computer. Radiometric correction of the data is performed by the computer. The primary output product of the LCRS is computer compatible tape.

In the block diagram, equipment up through the CCT is standard for all LCRS. The station will also be equipped with display and extractive processing capability which is unique to the particular users needs.

Just as in the CDPF, the characteristics of the instruments affect the cost and complexity of the functions which must be performed to correct the data. The geometric correction approach maximizes the amount of processing done on board the spacecraft in order to minimize the costs of performing these functions in a multiplicity of ground stations. Linearity corrections and reformatting are done entirely on board. They are performed in the compactor for LCRS data only. These X-corrections include earth rotation, earth curvature, line length, detector offsets and scan non-linearities. Annotation information such as date, time and image position is also inserted on-board. This is the same type of auxiliary information that is inserted in the wideband data stream for the CDPF.

All radiometric corrections will be done on the ground in the LCRS. The same approach will be utilized as is used for the CDPF; i.e., insert all the ancillary information required in the data stream from the spacecraft which permits the LCRS to correct to the best achievable radiometric fidelity.