

LUNAR ORBITAL SCIENCE STUDY

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LUNAR ORBITAL SCIENCE STUDY

FINAL REPORT
October 1968

Volume I

Contract NAS9-8343

Prepared for
National Aeronautics and Space Administration
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Houston, Texas

Prepared by:

Martin Marietta Corporation
P.O. Box 179
Denver, Colorado 80201

MARTIN MARIETTA CORPORATION
DENVER DIVISION

FOREWORD

This report is submitted in accordance with the requirements of Contract NAS9-8343, Article X, Reports of Work. The report is prepared in two volumes. Volume I contains the basic report. Volume II contains Appendices A and B.

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I INTRODUCTION

In June 1968, Martin Marietta Corporation was awarded a three month study as part of a NASA-MSC Phase B Lunar Orbital Science Study. The study resulted from the 1967 Summer Study of Lunar Science and Exploration at Santa Cruz. The total Phase B study is to further define the scientific use of lunar flights. The Martin Marietta study was directed toward the further definition of the scientific use of the orbital portion of the lunar flights.

A majority of the effort, in accordance with NASA-MSC direction, was applied to baselining experiments cooperatively with experiment scientists and accomplishing a functional flow grouping analysis of large numbers of experiment combinations identified as principal goals at the 1967 Santa Cruz Conference. The experiments were baselined and a new analysis technique devised based on payload, orbit, mission and data management constraints. The study confirms that a lunar orbital science program can be developed for achievement during the 1972-74 period.

II SUMMARY AND RECOMMENDATIONS

This contract was accomplished as a part of the NASA-MSC Phase B Lunar Orbital Science Study resulting from the 1967 Summer Study of Lunar Science and Exploration. The contract has as its objective further definition of the scientific use of lunar orbital flights through analysis, review, updating, and consolidation of experiment requirements, development status and feasibility, grouping, and mission requirements. The Phase B study represents a further step toward the preparation of detailed science mission plans, a process which is necessarily iterative in nature.

The Martin Marietta study placed primary emphasis on achievement of the Santa Cruz Conference objectives through a fundamental scientific analysis of the experiment objectives and their data requirements, and limited iteration of these with proposed experiment sensor capabilities and groupings derived during the study. Experiment mission and subsystem requirements were analyzed, based upon sample experiment groupings. Summarized results of the study are:

1. With the Lunar Science and Exploration Conference as a baseline, scientific objectives were grouped into four regimes: a) Subsurface, b) Surface, c) Atmospheric and d) Cosmic. The subsurface regime was the least defined and required particular attention during the study.

2. An analytical grouping rationale (roadmap) was developed which introduced a more fundamental technical experiment grouping approach than has been used during the early phases of previous experiment programs. The primary result of the roadmap analysis is the identification of correlative experiment groupings required to produce unambiguous data toward the scientific goals. With this roadmap, each experiment was analyzed against a series of seven engineering and scientific constraints. The Phase B analysis shows that all but four of the originally proposed experiments are generally compatible with both scientific objectives and mission constraints. Significant clarification and combination of sensor requirements were achieved, and one new experiment was identified for consideration. It is a swept frequency UHF radiometer, for study of the lunar subsurface regime.

3. Of the 30 experiments analyzed, 6 were in flight hardware configuration, 17 exist in non-flight hardware configuration with a high confidence level that lunar flight can be achieved; 7 require new development. Contact with responsible scientists and industrial sources in each experiment area provides assurance that instruments

can be available for a lunar program in the 1972-74 time period. A preliminary cost estimate for the 30 experiments indicates a total cost span of 60 to 85 million dollars for flight hardware. Further definition work, e.g., specifications and a detailed procurement cycle is recommended to arrive at a closer price definition.

4. Fifty-seven experiment groupings were generated using the roadmap technique. These groupings are categorized into minimum, medium and maximum scientific data accuracy groupings for achievement of the objectives of each regime (Subsurface, Surface, Atmospheric or Ionospheric and Cosmic). Neither selection of specific groupings for particular missions nor programmatic optimization of the number of missions was worked. However, early flights would be expected to use groupings of minimum accuracy since their instruments are generally less complex and more readily available. Additional work in experiment grouping and detailed instrument definition remains for accomplishment after the program constraints are evolved by Phase B.

The large number of groupings (57) indicates the advisability of computerizing the system interface requirements (weight, volume, power, data) for flight grouping since more than one group can be flown by each flight, once the payload allowables are better defined. Computerization allows optimization toward flight schedules and experiment costs. Furthermore, the split of C&SM-borne versus subsatellite-borne experiments (some of the experiment groups can go either way) can be optimized.

5. An optimum mission for most experiments would be polar and of at least 30 days duration to provide adequate coverage of the lunar environment. This points clearly to a requirement for subsatellite usage in the program. The near equatorial orbit constraints of the Apollo Spacecraft permit sensing of only about four percent of the lunar surface during the experiment operating periods; however, substantial achievement of certain experiment objectives can be accomplished. A large amount of data can be collected by use of the Apollo Spacecraft only which for early flights will represent an acceptable data return and will provide a significant contribution to the more comprehensive exploration planned for later flights.

6. There is a significant problem area in the requirement for data reduction and analysis following each flight. Further attention to the entire experiment data management area is advisable. Application of advanced data handling techniques has favorable program cost implications which should be investigated

prior to a Phase C start. An analysis of allowable data compression within the MSFN on the basis of intrinsic experiment data accuracy requirements is also recommended. It will result in an almost real time experiment data availability with a tremendous impact on the visibility of mission and experiment success. If there are new discoveries in lunar science they could be ascertained for announcement immediately after the flight - the present time delay is estimated to be 1½ years.

7. Preliminary timeline analysis shows that most of the crew time is required for basic mission performance. However, the crew requirements for experiment operation are within the design control cognizance of the experiment integrator. EVA to retrieve camera film was found to be the only mandatory requirement for crew participation. Deployment and actuation processes for either experiments and/or subsatellites can benefit from crew control. EVA for film retrieval is probably best performed at a mission time subsequent to the return of the LM when all three astronauts are available.

8. Payload capability varies depending primarily upon the landing site selected for the basic mission. Within the mission constraints of this study, payload available for scientific experiments varies from zero to approximately 1900 pounds.

9. Subsystem studies were made on the basis of selected groupings and show a requirement for various subsystem changes in both the C&SM and the subsatellite candidates. Final definition of these changes is dependent upon experiment group selection and will require additional investigation and iteration. General conclusions can be summarized as follows:

- Data System requirements on the C&SM increase moderately, while on the subsatellites, they range from about a 40% change (Lunar Orbiter) to complete redesign (AIMP).

- Power System requirements on the C&SM also increase moderately, while the subsatellites require almost double their present capacity and significant system design change. Most components are usable, however.

- Guidance and Control appear adequate for most experiments, with the most stringent requirements being $\pm 1^{\circ}$ on attitude control and $\pm 0.05^{\circ}$/sec. on angular rates. Some experiments require special pointing,

such as for a stellar reference or earth pointing antenna. These experiments may impose some modification requirements on the guidance and control system, although these appear to be well within state of the art capabilities.

- Thermal control will be required by some experiment groupings; for example, one stable satellite grouping studied requires an active thermal control system weighing approximately 190 pounds. In this particular case the subsystem would represent a significant portion of the total subsatellite weight, indicating a requirement for further study in the experiment grouping and in the thermal control system itself.

III SCIENTIFIC OBJECTIVES, EXPERIMENT GROUPINGS AND REQUIREMENTS

III.1 Scientific Objectives and Experiment Groupings - Using the Santa Cruz Conference as background, four broad regimes for lunar science exploration were identified, each regime dealing with a different part of the moon and the lunar environment. The four regimes are:

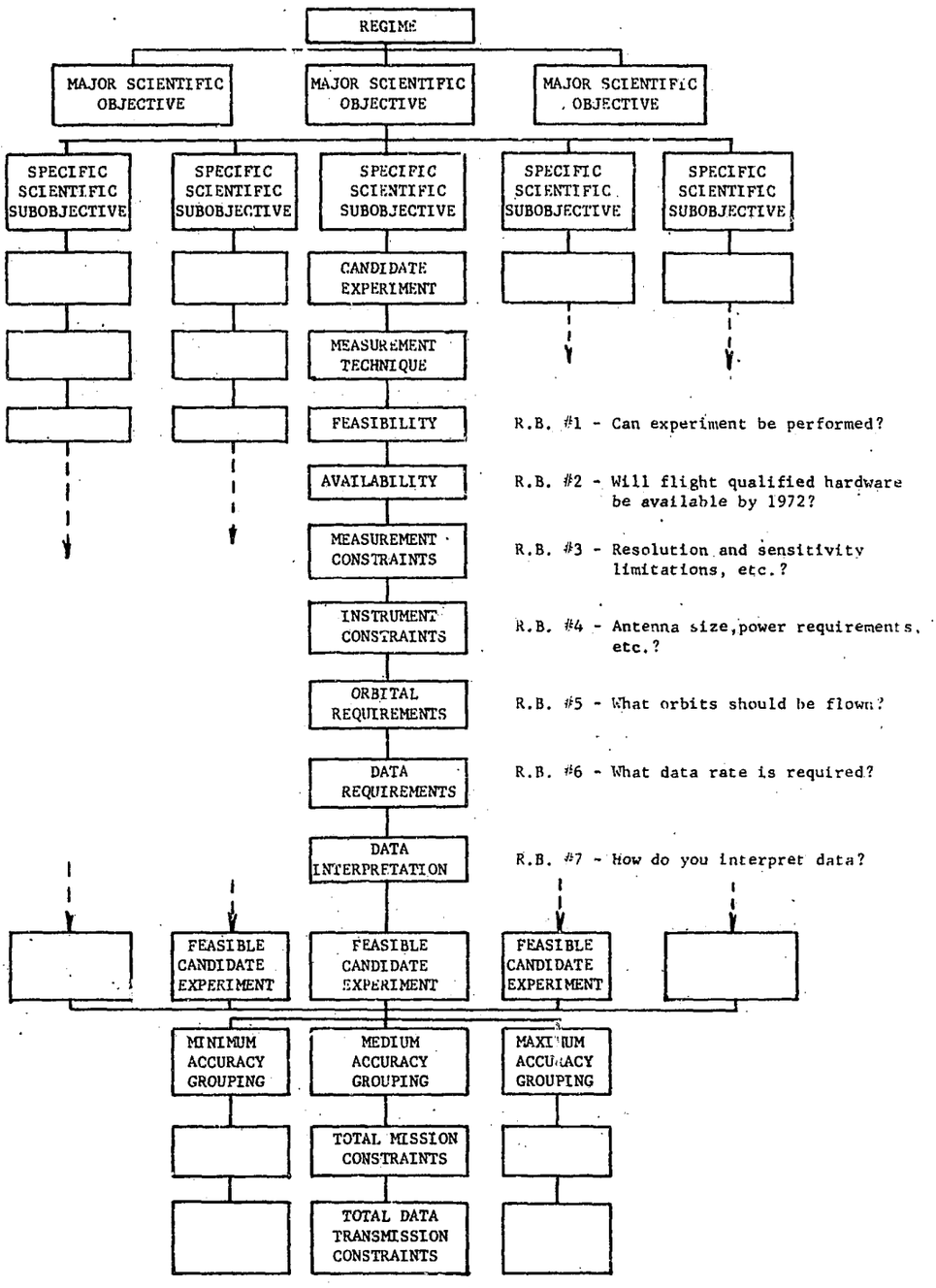
1. Subsurface
2. Surface
3. Atmosphere and ionosphere
4. Cosmic

Although these subdivisions have some overlap, they represent a logical system when considering the grouping of scientific objectives and experiments for lunar orbiting sensors.

Responsible scientists and industry sources were interviewed to establish the objectives and instruments which could meet those objectives. The data on objectives and instruments provided a functional flow or roadmap approach (Figure III.1-1) developed to establish experiment groupings. For each of the four regimes, basic scientific objectives were listed. These were in turn subdivided into specific sub-objectives. Candidate experiments which could meet the objectives were then listed, and fed through a series of seven constraints or roadblocks which defined estimates of 1) feasibility, 2) availability, 3) measurement constraints, 4) instrument constraints, 5) orbital requirements, 6) data requirements and 7) data interpretation. If an experiment failed to satisfy the requirements of a roadblock, it was eliminated from further considerations. The roadblock which was responsible for elimination of an experiment is indicated by a blacked lower right corner. The requirements of surface structure, atmosphere and ionospheric, and of the particles and fields (COSMIC) of the moon are well understood. The Santa Cruz Conference report gives coherent rationale for those regimes. Therefore, as expected, all experiments for these groups pass the roadblocks as shown in Figures III.1-3, III.1-4 and III.1-5 respectively.

The subsurface regime of the moon is the least understood of the baseline exploration considered by the Santa Cruz Conference. Only recently has the possibility of exploring this regime shown up (e.g., the "Raisin Bread" phenomenon has drawn the attention of the scientific community). Therefore, the subsurface regime was emphasized by MMC and the Santa Cruz baseline has been updated in cooperation with the experimenters concerned. Four experiments in the subsurface regime were eliminated by the roadblocks, as shown by Figure III.1-2. The elimination of four experiments led to identification of the swept frequency UHF radiometer.

REGIMES:
1. Subsurface
2. Surface & Near Surface
3. Atmosphere & Ionosphere
4. Cosmic



BASIC BLOCK DIAGRAM FOR LUNAR ORBITAL SCIENCE EXPERIMENT GROUPING "ROADMAP"

Figure III .1-1

REGIME					
MAJOR SCIENTIFIC OBJECTIVE					
SPECIFIC SCIENTIFIC SUBOBJECTIVE	A	To determine variation in mass and shape of moon & local density variations			
CANDIDATE EXPERIMENT	RADAR ALTIMETER	GRAVITY GRADIOMETER	R.F. TRANSPONDER (track from lunar orbit)	R.F. TRANSPONDER (track from earth)	LASER RETROREFLECTORS (track from earth
MEASUREMENT TECHNIQUE	Pulse or FM/CW radar X-band	Gravity gradient 2 linked gravity meters or rotating mass accelerometer,	Doppler and range transponders on lunar surface	Doppler and range transponder on orbiter	Corner - Cube retroreflectors on orbiter
FEASIBILITY	Previously demonstrated in satellites and space vehicles	Gravity meters must be linked together; accelerometer - lab model only	Must erect 3-10 transponders on lunar surface or drop and land units from LO.	Unified S-band equipment on CSM; lightweight units to be developed	Has been used in satellite tracking from earth
AVAILABILITY	Current unit operation to h=15 kmper; available to h=100 km without concept change - 1972	Both require development work for space use	Lightweight units could be developed by 1972	Lightweight transponder available by 1972	Lightweight retroreflectors will be available by 1972
MEASUREMENT CONSTRAINTS	Leading edge range tracker permits ± 25 ft. altitude accuracy	Gravity meter: 10^{-6} gal, accelerometer: 4×10^{-8} gal/cm	Will provide data over complete orbit with sufficient transponders	Provides data on 1/2 of lunar surface only	Provides data on 1/2 of lunar surface only; severe earth atmospheric constraints
INSTRUMENT CONSTRAINTS	MURA modules advantageous; 6 x 6 ant. array required for h = 100 kms, no constraints	No external accelerations permitted; long integration times required	Thermal drift and phase stability	Thermal drift and phase stability	Angular resolution of ground telescopes
ORBITAL REQUIREMENTS	Any orbit depending on mission h < 100 km	85° Inclination h = 50 kms h = 100 km e = .2	Orbit limited by transponder visibility ± 15° Inc., h = 100 kms	30°, 50°, 70°, 85° Inclination h = 600 kms 85° at 50 and 100 kms	30°, 50°, 70°, 85° Inclination h = 600 kms 85° at 50 and 100 kms
DATA REQUIREMENTS	Depends on application, 10 bits/sec. min.	2 bits/sec	10-100 bits/sec	Not applicable no separate data link required	Beacon required to distinguish 6 subsatellites
DATA INTERPRETATION	No range ambiguities in pulsed mode	Need to know accurately orbit and satellite orientations & external accelerations	Need a different lunar potential model & approach to problem	Need a different lunar potential model & approach to problem	Need improved potential model to handle high harmonic content

SUBOBJECTIVE EXPERIMENT GROUP A

EXPERIMENT GROUPING	MINIMUM ACCURACY GROUP Gravity Gradiometer Radar Altimeter Improved altimeter accuracy highly desirable to enhance Gravity Gradiometer accuracy	MEDIUM ACCURACY GROUP RF Transponders Track from lunar orbiter	MAXIMUM ACCURACY GROUP RF Transponders Track from earth, lightweight transponders required to realize multi-orbit mission capability.	MINIMUM ACCU IR Radiometer May require radiometer t surface temp Match resolu orbit height coverage,
TOTAL MISSION CONSTRAINTS	Subsatellite Experiment: map gross structure and anomalies over as large a part of the entire lunar surface as possible	CSM Vehicle: Should allow accurate mapping over limited lunar surface area	Subsatellite Experiment: obtain fine structure over region covering up to 1/2 the lunar surface (far side omitted)	Subsatellite obtain crude brightness t as large a p surface as p
TOTAL DATA TRANSMISSION CONSTRAINTS	Total data rate < 100 bits/sec easily accommodated with current communication link	Total data rate less than 100 bits/sec	No data link requirements	Adjust data s data recorded to be transmit line of sight

To understand processes of internal evolutions and their surface manifestations

B — To measure heat flow from lunar interior

C — To determine subsurface moisture, faults & hazards

	IR RADIOMETER (non-scanning)	K-BAND MICROWAVE RADIOMETER (non-scanning)	MULTI-FREQUENCY MICROWAVE IMAGER
	IR Brightness temperature 5μ - 40μ	Brightness temperature $2 \times 10^{10}\lambda$	Brightness temperature $10^{9}\lambda$; $10^{10}\lambda$; $2 \times 10^{10}\lambda$
	Previously demonstrated in satellites and space vehicles	19.5 gc radiometer; single channel, simple horn ant., "state of the art"	19.5 gc radiometer 7'x7' phased array "state of the art" L-band array ant. "unfurlable"
	All equipment "state of the art" & available now	All equipment "state of the art" & available now	18 mo. - 2yr. development program required; 4, 6, or 8 channel units available 1972
	Spatial resolution to be matched to resolution of microwave sensor	$\Delta T \approx 1°K$ feas. spatial resolution at 19.5 gc = 1°	$\Delta T \approx 1°K$ feas. spatial resolution: 1° at 20 gc & 40 gc 5° at 1.5 gc & 10 gc
	Instrument optics mechanically locked to microwave antenna assembly & attitude stable	High-gain horn ant. or parabola must be stabilized to point down at lunar surface	"L" band array antenna to be "unfurlable" from spacecraft (CSM)
	90° des., ± 15° acc. h = 90 km	90° des., ± 15° acc. h = 80 km	± 15° Inc. equat. CSM spacecraft h = 100 kms
	100 bits/sec, record data on far side of moon on tape and retransmit	100 bits/sec, record data on far side of moon on tape and retransmit	Record on digital tape or color film; alternate: 10 kilobits/sec.
	Multiple wave length calibration required surface temperature readily obtained	Simple calibration on earth requiring truth sites	Needs extensive calibration and truth site information from earth measurements

SUBOBJECTIVE EXPERIMENT GROUP B

MINIMUM ACCURACY GROUP	MEDIUM ACCURACY GROUP	MAXIMUM ACCURACY GROUP
IR Radiometer & K-Band Radiometer May require multi-wavelength IR radiometer to pin down kinetic surface temperature. Match resolution (ground swath) to orbit height to provide contiguous coverage.	IR Radiometer & K-Band Radiometer L-Band Radiometer (matched res.) Same as minimum accuracy group, except add single channel L-band unit.	IR Radiometer & K-Band Radiometer & Swept Frequency U.H.F. Radiometer add single channel U.H.F. unit to achieve maximum depth penetration
Subsatellite Vehicle: obtain crude IR and microwave brightness temperature maps over as large a part of entire lunar surface as possible	Subsatellite Vehicle Same as minimum accuracy group, except better brightness temperature profiles	Subsatellite Vehicle: same as above except maximum depth profiles & superior depth resolution
Adjust data sampling rate to allow data recorded on far side of moon to be transmitted when orbiter has line of sight to earth	Accommodate higher data rate by reducing sampling rate if necessary	Accommodate higher data rate by reducing sampling rate on IR and K-band channels first

SUBOBJECTIVE EXPERIMENT GROUP C

MINIMUM ACCURACY GROUP	MEDIUM ACCURACY GROUP	MAXIMUM A
K&L-Band Radiometers IR Radiometer (nonscanning) measure brightness temperatures at surface and two depths with fixed nonscanning radiometers. One channel per radiometer	Multi-Frequency Microwave Imager K,X, & L-Band M.W. Radiometers (6 channels) IR Mapping Radiometer medium resolution instruments to provide data on subsurface features and faults.	Multi-Fre (2 freq. IR Mappin "Unfocuse video ban and two d features
Subsatellite Vehicle: obtain crude synoptic maps of brightness temperature & emissivity over as large a part of entire lunar surface as possible	CSM or Subsatellite Vehicle: obtain synoptic imaging data for 3 depths to map lunar subsurface	CSM or Su Min: 2 ch Max: 4 ch time shar each freq
Adjust data sampling rate to allow data recorded on far side of moon to be retransmitted when orbiter has line of sight	Screen data on board with astronaut or transmit data back in realtime; record on tape on far side and retransmit	Image dat transmitte recorded side

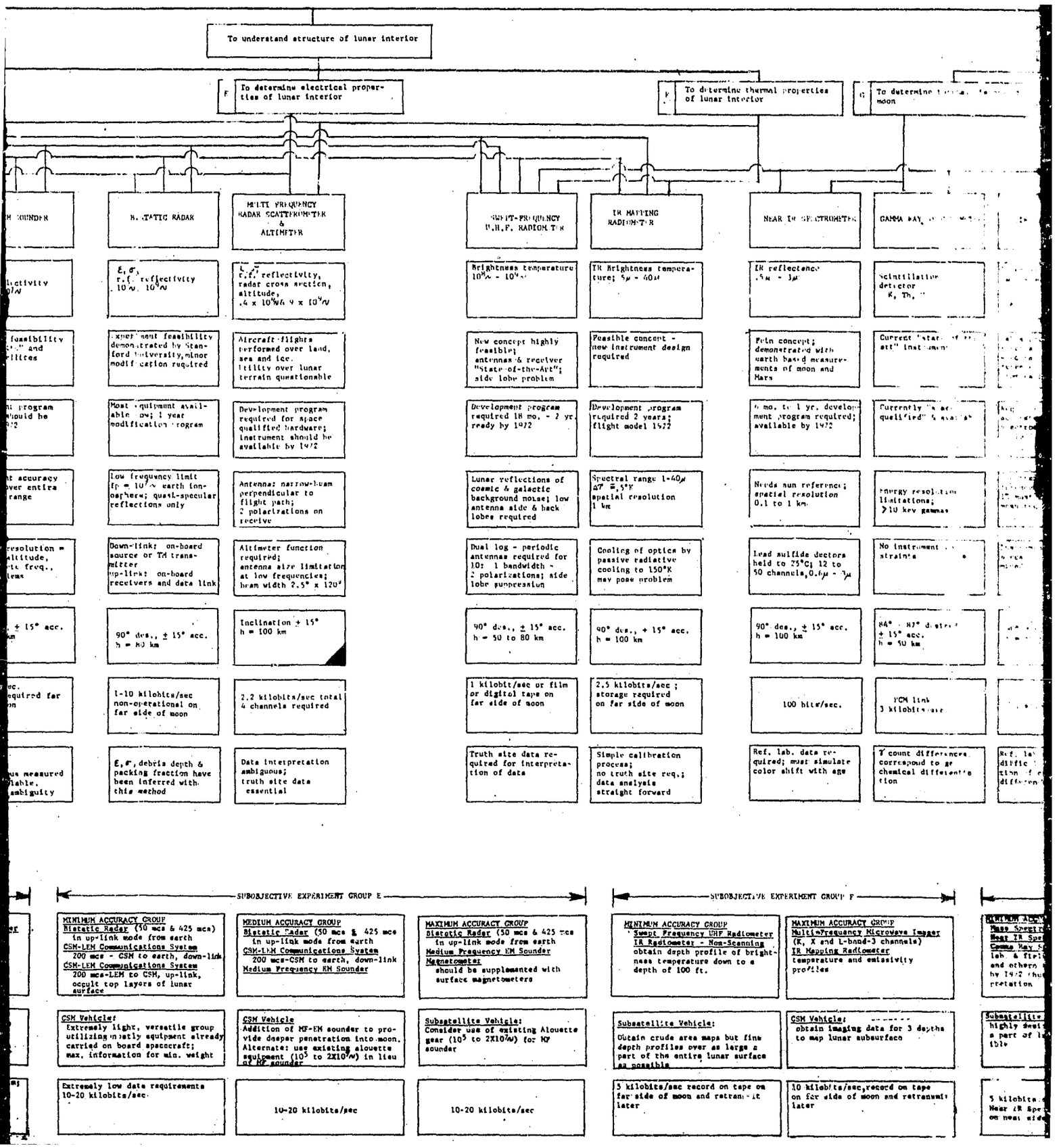
SUBSURFACE

D — To determine structural properties of lunar interior

MULTI-FREQUENCY RADAR IMAGER	MAGNETOMETER	VLF EM SOUNDER	MF EM SOUNDER
ε, σ, r.f. reflectivity 1.2 x 10^9 ~ 8.4 x 10^9 ~	Magnetic field dc - 10	ε, σ, r.f. reflectivity 10^4 to 10^6 ~	ε, σ, r.f. reflectivity 10^6 to 10^7 ~
1.2 gc imager "unfocused mode" flight model built by JPL	Existing sensor previously flown in lunar orbit	Ionosphere "opaque" from 2.8 x 10^6 ~ down; antenna size non-feasible	Equipment feasibility by "Alouette" and "RAE" satellites
18 month development program; available by 1972	All hardware currently available	Techniques demonstrated on earth; equipment available on surface	Development program required, should be done by 1972
80 m resolution unfocused mode "realtime imaging"	Must have magnetically clean & electrically clean spacecraft	Restricted to surface experiment	Measurement accuracy of < 2 db over entire frequency range
Antenna size 2½ x 2 meters in unfocused mode	Sensitivity .25 gamma dynamic range : 64 gamma	Antenna size limitation, RFI problem	Lateral resolution = orbital altitude, 13 discrete freq., RFI problems
90° des., ± 15° acc. CSM spacecraft h = 80 km	90° des., ± 15° acc. h = 50 to 100 km	Restricted to surface experiment	90° des., ± 15° acc. h = 100 km
60 kilobits/sec. per channel	300 bits/sec.	Restricted to surface experiment	100 bits/sec. recorder required far side of moon
Need imaging data on earth and truth site information	Simultaneous lunar surface measurement desirable	Restricted to surface experiment	No previous measured data available. Possible ambiguity

SUBOBJECTIVE EXPERIMENT GROUP D

MAXIMUM ACCURACY GROUP	MINIMUM ACCURACY GROUP	MEDIUM ACCURACY GROUP #1	MEDIUM ACCURACY GROUP #2	MAXIMUM ACCURACY GROUP	MINIMUM ACCURACY GROUP
Multi-Frequency Radar Imager (2 freq., 2 pol., 4 channel) IR Mapping Radiometer "Unfocused" imager-80m res. provides best resolution at surface and two depths to show subsurface features.	Near IR Spectrometer Swept Frequency UHF Radiometer Gamma Ray Spectrometer IR Radiometer (nonscanning) should yield data on location and composition of deposits	Near IR Spectrometer Far IR Spectrometer IR Radiometer (nonscanning) Swept Frequency UHF Radiometer Gamma Ray Spectrometer extend coverage to 40 microns for more detail	Multi-Frequency Microwave Imager (K, X, & L - 3 Channels) Swept Frequency UHF Radiometer IR Mapping Radiometer	Multi-Frequency Microwave Imager (K, X, & L - 3 Channels) IR Radiometer Imager Multi-Frequency Radar Imager (1.2 gc & 8.4 gc - 2 pol.)	Bistatic Radar in up-link mo CSM-LEM Communi 200 mcs - CSM CSM-LEM Communi 200 mcs-LEM occult top l surface
CSM or Subsatellite Vehicle: Min: 2 channels-62.5 Kbits/sec Max: 4 channels-122.5 Kbits/sec time share receive channels at each frequency	Subsatellite Vehicle: highly desirable to map as large a part of lunar surface as possible	CSM Vehicle:	CSM Vehicle: obtain imaging data down to a depth in the vicinity of 100 ft.	CSM Vehicle: operate radar only when passive imagers are turned off	CSM Vehicle: Extremely ligh utilizing inst carried on boa max. informati
Image data processed on board, transmitted back in "real time", recorded & retransmitted on far side	6 kilobits/sec Near IR Spectrometer operates on near side of moon only	5 kilobits/sec Near IR Spectrometer operates on near side of moon only	100 kilobits/sec record on tape on far side of moon	All data recorded on tape or film, astronaut filter helpful	Extremely low d 10-20 kilobits/


To understand structure of lunar interior
F To determine electrical properties of lunar interior
To determine thermal properties of lunar interior
G To determine
moon
M SOUNDER
BISTATIC RADAR
MULTI FREQUENCY RADAR SCATTEROMETER & ALTIMETER
SWEPT-FREQUENCY U.H.F. RADIOMETER
IR MAPPING RADIOMETER
NEAR IR SPECTROMETER
GAMMA RAY
ε, σ, reflectivity
ε, σ, reflectivity, radar cross section, altitude, .4 x 10⁴W/ 9 x 10⁴W
Brightness temperature 10⁴K - 10⁴K
IR brightness temperature; 5μ - 40μ
IR reflectance .5μ - 3μ
Scintillation detector K, Th,
Experiment feasibility demonstrated by Stanford University, minor modification required
Aircraft flights performed over land, sea and ice. Utility over lunar terrain questionable
New concept highly feasible; antennas & receiver "State-of-the-Art"; side lobe problem
Feasible concept - new instrument design required
Feln concept; demonstrated with earth based measurements of moon and Mars
Current "state of the art" instrument
Most equipment available now; 1 year modification program
Development program required for space qualified hardware; instrument should be available by 1972
Development program required 18 mo. - 2 yr. ready by 1972
Development program required 2 years; flight model 1972
6 mo. to 1 yr. development program required; available by 1972
Currently "space qualified" & available
low frequency limit fp = 10⁷ earth ionosphere; quasi-specular reflections only
Antenna: narrow-beam perpendicular to flight path; 2 polarizations on receive
Lunar reflections of cosmic & galactic background noise; low antenna side & back lobes required
Spectral range 1-40μ ΔT ≅ .5°K spatial resolution 1 km
Needs sun reference; spatial resolution 0.1 to 1 km
Energy resolution limitations; >10 key gammas
Down-link: on-board source or TM transmitter
up-link: on-board receivers and data link
Altimeter function required; antenna size limitation at low frequencies; beam width 2.5° x 120°
Dual log - periodic antennas required for 10:1 bandwidth - 2 polarizations; side lobe suppression
Cooling of optics by passive radiative cooling to 150°K may pose problem
Lead sulfide dectors held to 25°C; 12 to 50 channels, 0.6μ - 3μ
No instrument constraints
90° des., ± 15° acc. h = 80 km
Inclination ± 15° h = 100 km
90° des., ± 15° acc. h = 50 to 80 km
90° des., ± 15° acc. h = 100 km
90° des., ± 15° acc. h = 100 km
84° - 87° ± 15° acc. h = 50 km
1-10 kilobits/sec non-operational on far side of moon
2.2 kilobits/sec total 4 channels required
1 kilobit/sec or film or digital tape on far side of moon
2.5 kilobits/sec ; storage required on far side of moon
100 bits/sec.
PCM link 3 kilobits/sec
ε, σ, debris depth & packing fraction have been inferred with this method
Data interpretation ambiguous; truth site data essential
Truth site data required for interpretation of data
Simple calibration process; no truth site req.; data analysis straight forward
Ref. lab. data required; must simulate color shift with age
γ count differences correspond to ge chemical differentia tion
SUBOBJECTIVE EXPERIMENT GROUP E
SUBOBJECTIVE EXPERIMENT GROUP F
MINIMUM ACCURACY GROUP
Bistatic Radar (50 mcs & 425 mcs) in up-link mode from earth
CSM-LEM Communications System 200 mcs - CSM to earth, down-link
CSM-LEM Communications System 200 mcs-LEM to CSM, up-link, occult top layers of lunar surface
MEDIUM ACCURACY GROUP
Bistatic Radar (50 mcs & 425 mcs in up-link mode from earth
CSM-LEM Communications System 200 mcs-CSM to earth, down-link
Medium Frequency EM Sounder
MAXIMUM ACCURACY GROUP
Bistatic Radar (50 mcs & 425 mcs in up-link mode from earth
Medium Frequency EM Sounder
Magnetometer should be supplemented with surface magnetometers
MINIMUM ACCURACY GROUP
Swept Frequency UHF Radiometer
IR Radiometer - Non-Scanning obtain depth profile of brightness temperature down to a depth of 100 ft.
MAXIMUM ACCURACY GROUP
Multi-Frequency Microwave Imager (K, X and L-band-3 channels)
IR Mapping Radiometer temperature and emissivity profiles
CSM Vehicle: Extremely light, versatile group utilizing mostly equipment already carried on board spacecraft; max. information for min. weight
CSM Vehicle Addition of MF-EM sounder to provide deeper penetration into moon. Alternate: use existing Alouette equipment (10⁵ to 2X10⁷W) in lieu of MF sounder
Subsatellite Vehicle: Consider use of existing Alouette gear (10⁵ to 2X10⁷W) for MF sounder
Subsatellite Vehicle: Obtain crude area maps but fine depth profiles over as large a part of the entire lunar surface as possible
CSM Vehicle: obtain imaging data for 3 depths to map lunar subsurface
Extremely low data requirements 10-20 kilobits/sec
10-20 kilobits/sec
10-20 kilobits/sec
5 kilobits/sec record on tape on far side of moon and retransmit later
10 kilobits/sec, record on tape on far side of moon and retransmit later

MARTIN MARIETTA CORPORATION
DENVER DIVISION

To understand major geological processes responsible for character of lunar surface

To determine thermal history of moon

H — To locate and identify mineral deposits & formations

I — To determine emission of gases and locate sources of preferential gas emission

GAMMA RAY SPECTROMETER	IR SPECTROMETER	MASS SPECTROMETER
Scintillation detector K, Th, U	IR emissivity 7μ - 40μ	Atmosphere composition and density 1 to 130 a.m.u. $\geq 10^{-13}$ torr
Current "state of the art" instrument	Lab studies of emission spectra 8-13μ; lunar spectral emission differences less than 1% from earth	Modification of "ALSEP" instrument; CSM produced contaminants limit experiment feasibility
Currently "space qualified" & available	Requires development of accurate multichannel spectrometer, available by 1972	Requires development of electron multiplier technology; flight version by early 1971
Energy resolution limitations; $\geq$10 kev gammas	Spectral emission differences of less than 1% must be accurately measured	Sensitivity marginal; statistical fluctuations must be time averaged
No instrument constraints	Instrument development specs. to be determined	Instrument outgassing constraints; requires shielding from attitude control jets
84° - 87° desired, ± 15° acc. h = 50 km	90° des., ± 15° acc. h = 50 to 100 km	90° des., ± 15° acc. h = 50 to 100 km
PCM link 1 kilobits/sec	1 kilobit/sec.	80 bits/sec.
Y count differences correspond to geochemical differentiation	Ref. lab data required; difficult interpretation of composition differences	Needs a controlled release of tracer gas from surface to calibrate technique

Indicates a roadblock barrier.

SUBOBJECTIVE EXPERIMENT GROUP G, H, & I

	MINIMUM ACCURACY GROUP	MAXIMUM ACCURACY GROUP
	Mass Spectrometer Near IR Spectrometer Gamma Ray Spectrometer lab. & field work by J. Adams and others should become available by 1972 thus facilitating interpretation	Mass Spectrometer Near IR Spectrometer Far IR Spectrometer Gamma Ray Spectrometer extend observations out to 40 microns
Subsatellite Vehicle: highly desirable to map as large a part of lunar surface as possible	Subsatellite Vehicle: highly desirable to map as large a part of lunar surface as possible	CSM or Subsatellite Vehicle:
5 kilobits/sec Near IR Spectrometer operates on near side of moon only	5 kilobits/sec Near IR Spectrometer operates on near side of moon only	5 kilobits/sec Near IR Spectrometer operates on near side of moon only

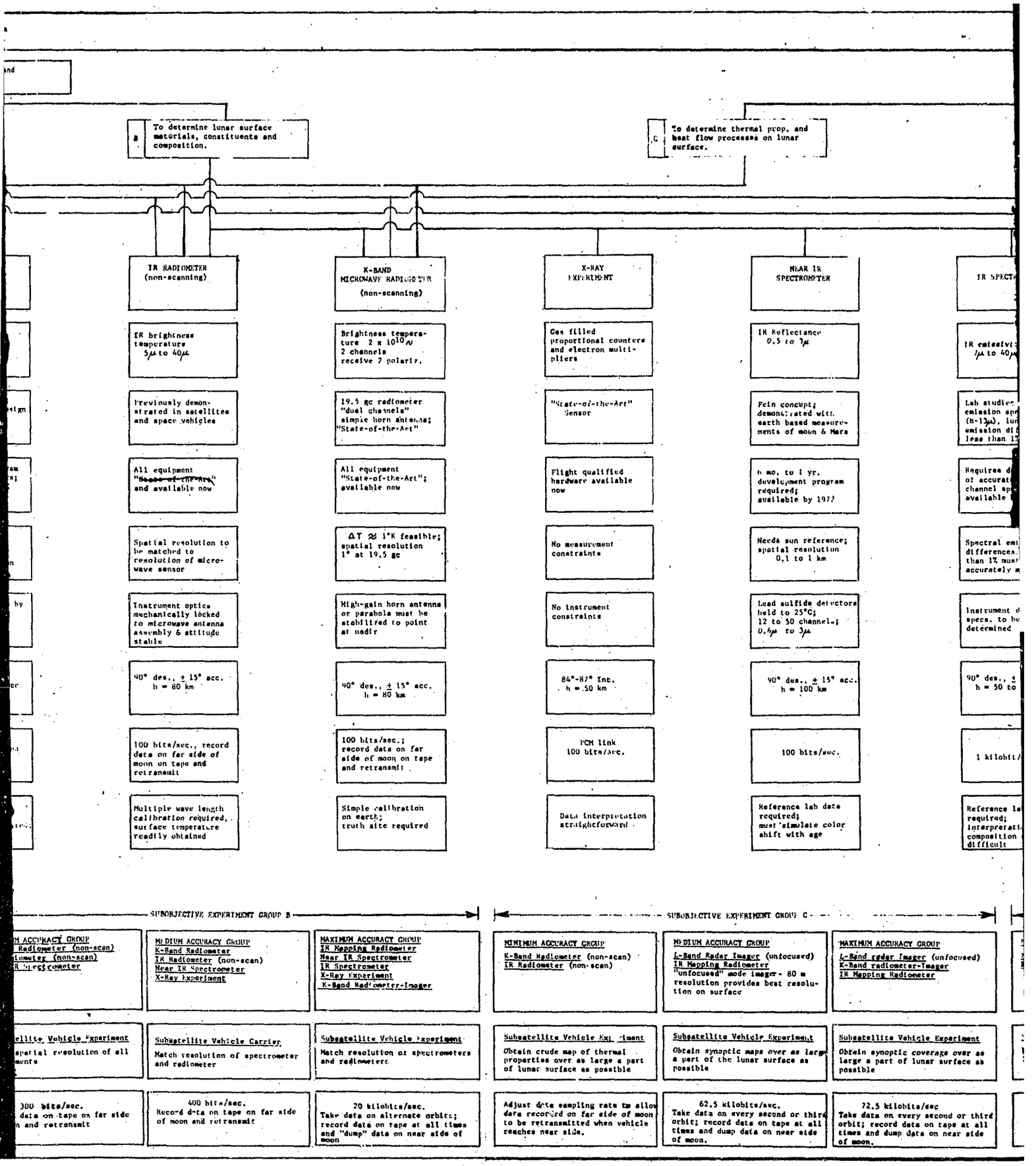
LUNAR ORBITAL SCIENCE EXPERIMENT GROUPINGS FOR SUBSURFACE REGIME

Figure III .1-2

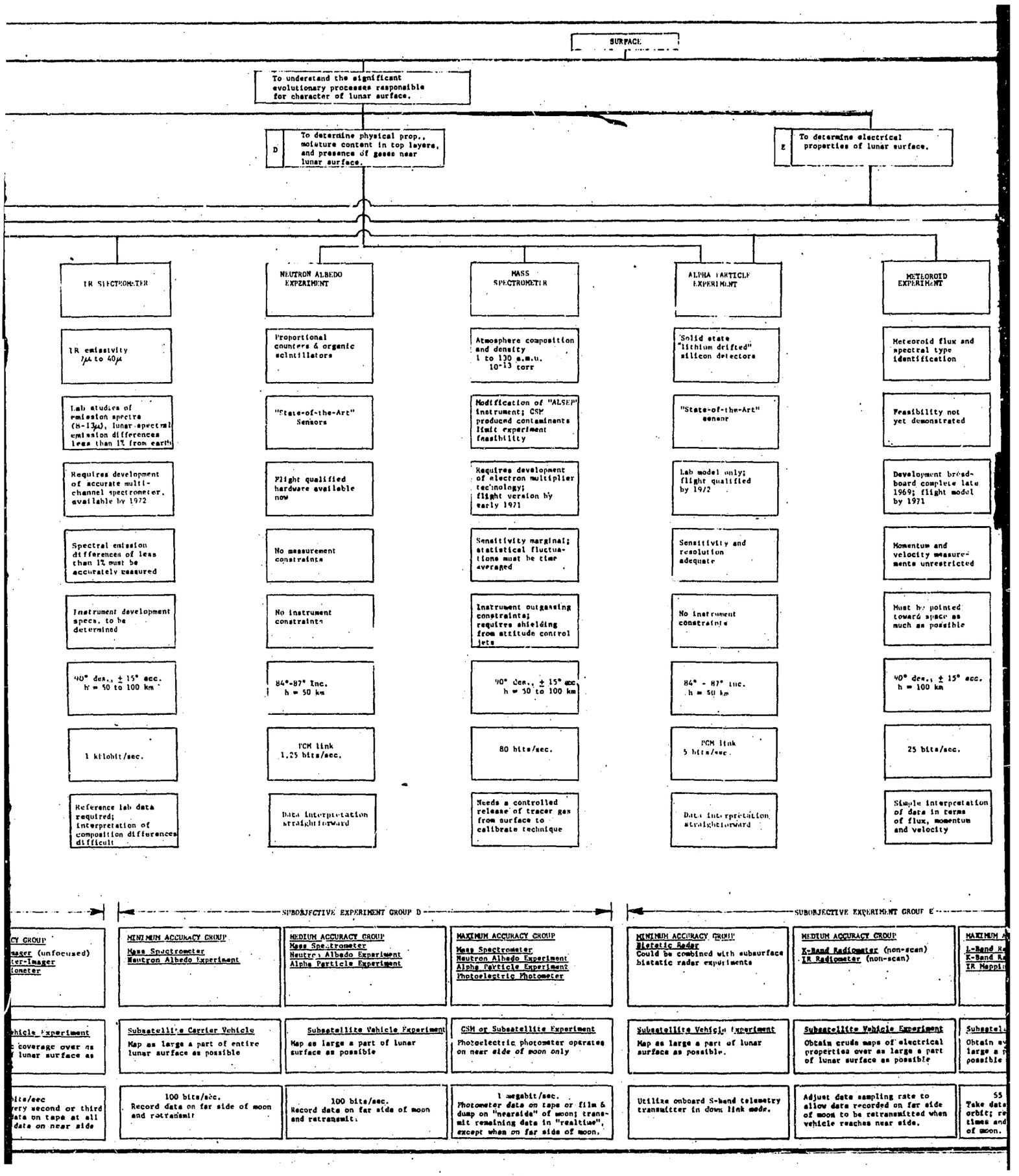
REGIME	To understand lunar surface and near surface characteristics.		
SPECIFIC SCIENTIFIC SUBOBJECTIVE	A — To determine lunar surface structure.		
CANDIDATE EXPERIMENT	K-BAND RADIOMETER-IMAGER	L-BAND RADAR IMAGER	IR MAPPING RADIOMETER
MEASUREMENT TECHNIQUE	Brightness temperature 2×10^{10} °K phase-scanned 2 channels-2 ppl.	ε, σ r.f. reflectivity 1.2×10^{9} °K	IR Brightness temperature 5μ to 40μ
FEASIBILITY	19.5 gc radiometer 7' x 7' phased array "State-of-the-Art"	1.2 gc "unfocused" flight model built by JPL	Feasible concept; new instrument design required
AVAILABILITY	18 mo. develop. program required. Available by 1972	18 mo. development program; available by 1972	Development program required - 2 years; flight model 1972
MEASUREMENT CONSTRAINTS	$\Delta T \approx 1°K$ Spatial resolution = 1° beam width at 20 gc	80 meters "unfocused" "X" meters "focused" "realtime" time imaging possible	Spectral range 1 to 40μ $\Delta T \approx 0.5°K$ spatial resolution 1 km
INSTRUMENT CONSTRAINTS	Rocket exhausts or outgassing from spacecraft	Antenna size: 2m x 2½m "unfocused" 3m x 5m "focused"	Cooling of optics by passive radiative cooling to 150°K may pose problem
ORBITAL REQUIREMENTS	± 15° inc. equat. CSM spacecraft h = 100 km	90° des. ± 15° acc. CSM spacecraft h = 80 km	90° des., ± 15° acc, h = 100 km
DATA REQUIREMENTS	Record on digital tape or film; alternate- 10 kilobit/sec.	60 kilobits/sec, "unfocused mode"	2.5 kilobits/sec; storage required on far side of moon
DATA INTERPRETATION	Requires truth site data for image calibration	Requires truth site data for image calibration	Simple calibration process; No truth site required; data analysis straightforward

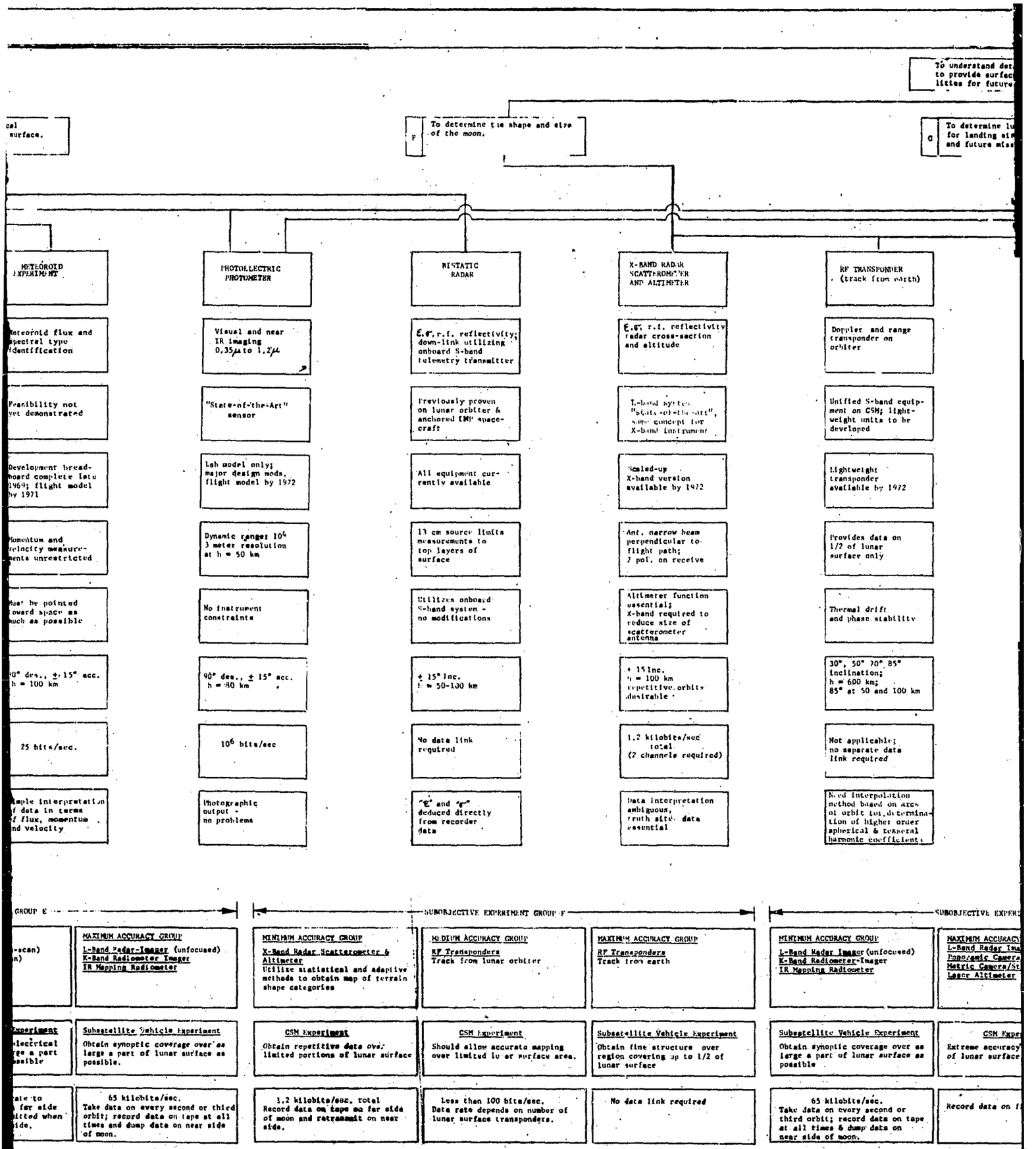
SUBOBJECTIVE EXPERIMENT GROUP A

EXPERIMENT GROUPING	MINIMUM ACCURACY GROUP K-Band Radiometer-Imager L-Band Radar Imager IR Mapping Radiometer	MEDIUM ACCURACY GROUP K-Band Radiometer-Imager L-Band Radar Imager IR Mapping Radiometer Support Camera	MAXIMUM ACCURACY GROUP K-Band Radiometer-Imager L-Band Radar Imager IR Mapping Radiometer Panoramic Camera	MINIMUM ACCURACY GROUP K-Band Radiometer (non-scan) IR Radiometer (non-scan) Near IR Spectrometer
TOTAL MISSION CONSTRAINTS	CSM Experiment Obtain synoptic maps of lunar terrain	CSM Experiment Obtain synoptic maps of lunar terrain	CSM Experiment Obtain synoptic maps of lunar terrain	Subsatellite Vehicle Experimen Match spatial resolution of a instruments
TOTAL DATA TRANSMISSION CONSTRAINTS	Record data on film and return; "realtime dump" of "unfocused" radar image data possible	Record data on film and return; "realtime dump" of "unfocused" radar image data possible	Record data on film and return	300 bits/sec. Record data on tape on far si of moon and retransmit


B
To determine lunar surface materials, constituents and composition.
C
To determine thermal prop. and heat flow processes on lunar surface.
IR RADIOMETER (non-scanning)
K-BAND MICROWAVE RADIOMETER (non-scanning)
X-RAY EXPERIMENT
NEAR IR SPECTROMETER
IR Reflectance 0.5 to 7μ
Fein concept; demonstrated with earth based measurements of moon & Mars
6 mo. to 1 yr. development program required; available by 1972
Needs sun reference; spatial resolution 0.1 to 1 km
Lead sulfide detectors held to 25°C; 12 to 50 channels 0.6μ to 3μ
90° des., ± 15° acc. h = 100 km
100 bits/sec.
Reference lab data required; must simulate color shift with age
SUBOBJECTIVE EXPERIMENT GROUP B
SUBOBJECTIVE EXPERIMENT GROUP C
MINIMUM ACCURACY GROUP
K-Band Radiometer (non-scan)
IR Radiometer (non-scan)
MEDIUM ACCURACY GROUP
L-Band Radar Imager (unfocused)
IR Mapping Radiometer
"unfocused" mode imager - 80 m resolution provides best resolution on surface
MAXIMUM ACCURACY GROUP
L-Band radar Imager (unfocused)
K-Band radiometer-Imager
IR Mapping Radiometer
Subsatellite Vehicle Experiment
Obtain synoptic maps over as large a part of the lunar surface as possible
Obtain synoptic coverage over as large a part of lunar surface as possible
62.5 kilobits/sec.
Take data on every second or third orbit; record data on tape at all times and dump data on near side of moon.
72.5 kilobits/sec
Take data on every second or third orbit; record data on tape at all times and dump data on near side of moon.

III - 4 - B


SURFACE
To understand the significant evolutionary processes responsible for character of lunar surface.
D
To determine physical prop., moisture content in top layers, and presence of gases near lunar surface.
E
To determine electrical properties of lunar surface.
IR SPECTROMETER
NEUTRON ALBEDO EXPERIMENT
MASS SPECTROMETER
ALPHA PARTICLE EXPERIMENT
METEOROID EXPERIMENT
IR emissivity 7μ to 40μ
Proportional counters & organic scintillators
Atmosphere composition and density 1 to 130 a.m.u. 10-13 torr
Solid state "lithium drifted" silicon detectors
Meteoroid flux and spectral type identification
Lab studies of emission spectra (8-13μ), lunar spectral emission differences less than 1% from earth
"State-of-the-Art" Sensors
Modification of "ALSEP" instrument; CSM produced contaminants limit experiment feasibility
"State-of-the-Art" sensor
Feasibility not yet demonstrated
Requires development of accurate multi-channel spectrometer, available by 1972
Flight qualified hardware available now
Requires development of electron multiplier technology; flight version by early 1971
Lab model only; flight qualified by 1972
Development breadboard complete late 1969; flight model by 1971
Spectral emission differences of less than 1% must be accurately measured
No measurement constraints
Sensitivity marginal; statistical fluctuations must be time averaged
Sensitivity and resolution adequate
Momentum and velocity measurements unrestricted
Instrument development specs. to be determined
No instrument constraints
Instrument outgassing constraints; requires shielding from attitude control jets
No instrument constraints
Must be pointed toward space as much as possible
90° dec., ± 15° acc. h = 50 to 100 km
84°-87° Inc. h = 50 km
90° dec., ± 15° acc. h = 50 to 100 km
84° - 87° inc. h = 50 km
90° dec., ± 15° acc. h = 100 km
1 kilobit/sec.
PCM link 1.25 bits/sec.
80 bits/sec.
PCM link 5 bits/sec.
25 bits/sec.
Reference lab data required; interpretation of composition differences difficult
Data interpretation straightforward
Needs a controlled release of tracer gas from surface to calibrate technique
Data interpretation straightforward
Simple interpretation of data in terms of flux, momentum and velocity
SUBOBJECTIVE EXPERIMENT GROUP D
SUBOBJECTIVE EXPERIMENT GROUP E
MINIMUM ACCURACY GROUP
Mass Spectrometer
Neutron Albedo Experiment
MEDIUM ACCURACY GROUP
Mass Spectrometer
Neutron Albedo Experiment
Alpha Particle Experiment
MAXIMUM ACCURACY GROUP
Mass Spectrometer
Neutron Albedo Experiment
Alpha Particle Experiment
Photoelectric Photometer
MINIMUM ACCURACY GROUP
Bistatic Radar
Could be combined with subsurface bistatic radar experiments
MEDIUM ACCURACY GROUP
X-Band Radiometer (non-scan)
IR Radiometer (non-scan)
Subsatellite Carrier Vehicle
Map as large a part of entire lunar surface as possible
Subsatellite Vehicle Experiment
Map as large a part of lunar surface as possible
CSM or Subsatellite Experiment
Photoelectric photometer operates on near side of moon only
Subsatellite Vehicle Experiment
Map as large a part of lunar surface as possible.
Subsatellite Vehicle Experiment
Obtain crude maps of electrical properties over as large a part of lunar surface as possible
100 bits/sec.
Record data on far side of moon and retransmit
100 bits/sec.
Record data on far side of moon and retransmit
1 megabit/sec.
Photometer data on tape or film & dump on "nearside" of moon; transmit remaining data in "realtime", except when on far side of moon.
Utilize onboard S-band telemetry transmitter in down link mode.
Adjust data sampling rate to allow data recorded on far side of moon to be retransmitted when vehicle reaches near side.


To understand det
to provide surfac
ities for future
cal
surface.
F
To determine the shape and size of the moon.
G
To determine lu
for landing sit
and future miss
METEOROID EXPERIMENT
PHOTOELECTRIC PHOTOMETER
BISTATIC RADAR
X-BAND RADAR SCATTEROMETER AND ALTIMETER
RF TRANSPONDER (track from earth)
Meteoroid flux and spectral type identification
Visual and near IR imaging 0.35μ to 1.2μ
ε, σ, r.f. reflectivity; down-link utilizing onboard S-band telemetry transmitter
ε, σ, r.f. reflectivity radar cross-section and altitude
Doppler and range transponder on orbiter
Feasibility not yet demonstrated
"State-of-the-Art" sensor
Previously proven on lunar orbiter & anchored IMP spacecraft
L-band system "state-of-the-art", same concept for X-band instrument
Unified S-band equipment on CSM; lightweight units to be developed
Development breadboard complete late 1969; flight model by 1971
Lab model only; major design mods, flight model by 1972
All equipment currently available
Scaled-up X-band version available by 1972
Lightweight transponder available by 1972
Momentum and velocity measurements unrestricted
Dynamic range: 10⁴ 3 meter resolution at h = 50 km
13 cm source limits measurements to top layers of surface
Ant. narrow beam perpendicular to flight path; 2 pol. on receive
Provides data on 1/2 of lunar surface only
Must be pointed toward space as much as possible
No instrument constraints
Utilizes onboard S-band system - no modifications
Altimeter function essential; X-band required to reduce size of scatterometer antenna
Thermal drift and phase stability
90° dev., ± 15° acc. h = 100 km
90° dev., ± 15° acc. h = 80 km
± 15° Inc. h = 50-100 km
+ 15 Inc. h = 100 km repetitive orbits desirable
30°, 50° 70° 85° inclination; h = 600 km; 85° at 50 and 100 km
25 bits/sec.
10⁶ bits/sec
No data link required
1.2 kilobits/sec total (2 channels required)
Not applicable; no separate data link required
Sample interpretation of data in terms of flux, momentum and velocity
Photographic output - no problems
"ε" and "σ" deduced directly from recorder data
Data interpretation ambiguous, truth site data essential
Need interpolation method based on arcs of orbit for determination of higher order spherical & tesseral harmonic coefficients
SUBOBJECTIVE EXPERIMENT GROUP E
SUBOBJECTIVE EXPERIMENT GROUP F
SUBOBJECTIVE EXPERI
MAXIMUM ACCURACY GROUP
L-Band Radar-Imager (unfocused)
K-Band Radiometer Imager
IR Mapping Radiometer
MINIMUM ACCURACY GROUP
X-Band Radar Scatterometer & Altimeter
Utilize statistical and adaptive methods to obtain map of terrain shape categories
MEDIUM ACCURACY GROUP
RF Transponders
Track from lunar orbiter
MAXIMUM ACCURACY GROUP
RF Transponders
Track from earth
MINIMUM ACCURACY GROUP
L-Band Radar Imager (unfocused)
K-Band Radiometer-Imager
IR Mapping Radiometer
MAXIMUM ACCURACY
L-Band Radar Ima
Panoramic Camera
Metric Camera/St
Laser Altimeter
Subsatellite Vehicle Experiment
Obtain synoptic coverage over as large a part of lunar surface as possible.
CSM Experiment
Obtain repetitive data over limited portions of lunar surface
CSM Experiment
Should allow accurate mapping over limited lunar surface area.
Subsatellite Vehicle Experiment
Obtain fine structure over region covering up to 1/2 of lunar surface
Subsatellite Vehicle Experiment
Obtain synoptic coverage over as large a part of lunar surface as possible
CSM Exper
Extreme accuracy of lunar surface
65 kilobits/sec.
Take data on every second or third orbit; record data on tape at all times & dump data on near side of moon.
1.2 kilobits/sec. total
Record data on tape on far side of moon and retransmit on near side.
Less than 100 bits/sec.
Data rate depends on number of lunar surface transponders.
No data link required
65 kilobits/sec.
Take data on every second or third orbit; record data on tape at all times & dump data on near side of moon.
Record data on f

III-41D

To understand details of lunar surface to provide surface exploration capabilities for future missions

G — To determine lunar surface data for landing site certification and future mission planning.

H — To provide topographic maps of the lunar surface.

	RF TRANSPONDER (track from lunar orbiter)	LASER ALTIMETER	METRIC CAMERA/ STELLAR CAMERAS	PANORAMIC CAMERA
	Doppler and range transponders on lunar surface	Measure altitude to ± 2.5 meters using laser	Stereophotography with concurrent stellar photography for attitude information	Simple high resolution photography
	Must erect 3-10 transponders on lunar surface or drop and land units from LO	Currently under development by TRG, Inc.	Feasible; major modification to existing cameras	feasible, major modification to existing instrument
	Lightweight units could be developed by 1972	Flight qualified unit ready by 1972	2 year development effort; unit ready by 1972	18 mo. development effort; unit ready by 1972
	Will provide data over complete orbit with sufficient transponders	6 pulses/minute limit instrument repetition rate	No measurement constraints	No measurement constraints
	Thermal drift and phase stability	Attitude knowledge required (±0.1°) to measure altitude to ±6 meters	Provide sufficient field-of-view for cameras	Provide sufficient field-of-view for camera
	Orbit limited by transponder visibility ± 15° Inc., h = 100 km	± 15 Inc. h = 80 - 150 km	± 15° Inc. h = 100 km	± 15° Inc. h = 100 km
	10 to 100 bits/sec.	None; Film return	None; Film return	None; film return
	Need interpolation method based on arcs of orbit for determination of higher order spherical & tesseral harmonic coefficients	Data interpretation straightforward	Simple calibration by photography of star fields	Data interpretation straightforward

SUBOBJECTIVE EXPERIMENT GROUP G — SUBOBJECTIVE EXPERIMENT GROUP H

	MAXIMUM ACCURACY GROUP	MEDIUM ACCURACY GROUP	MINIMUM ACCURACY GROUP	MEDIUM ACCURACY GROUP	MAXIMUM ACCURACY GROUP
	L-Band Radar Imager (focused) Panoramic Camera Metric Camera/Stellar Cameras Laser Altimeter	L-Band Radar Imager (Unfocused) Metric Camera/Stellar Cameras Laser Altimeter	Metric Camera/Stellar Cameras Laser Altimeter	Metric Camera/Stellar Panoramic Camera Laser Altimeter	Metric/Stellar Camera Panoramic Camera Laser Altimeter Photoelectric Photometer
	CSM Experiment extreme accuracy over limited part of lunar surface	CSM Experiment Good accuracy over limited part of lunar surface	CSM Experiment Basic mapping mission	CSM Experiment Improved accuracy afforded by inclusion of panoramic camera	CSM Experiment Should yield absolutely highest accuracy available.
	Record data on film and return.	Record data on film and return.	Record all data on film and return.	Record all data on film and return	Record all data on film and return.

III - 41 Eng

MARTIN MARIETTA CORPORATION
DENVER DIVISION

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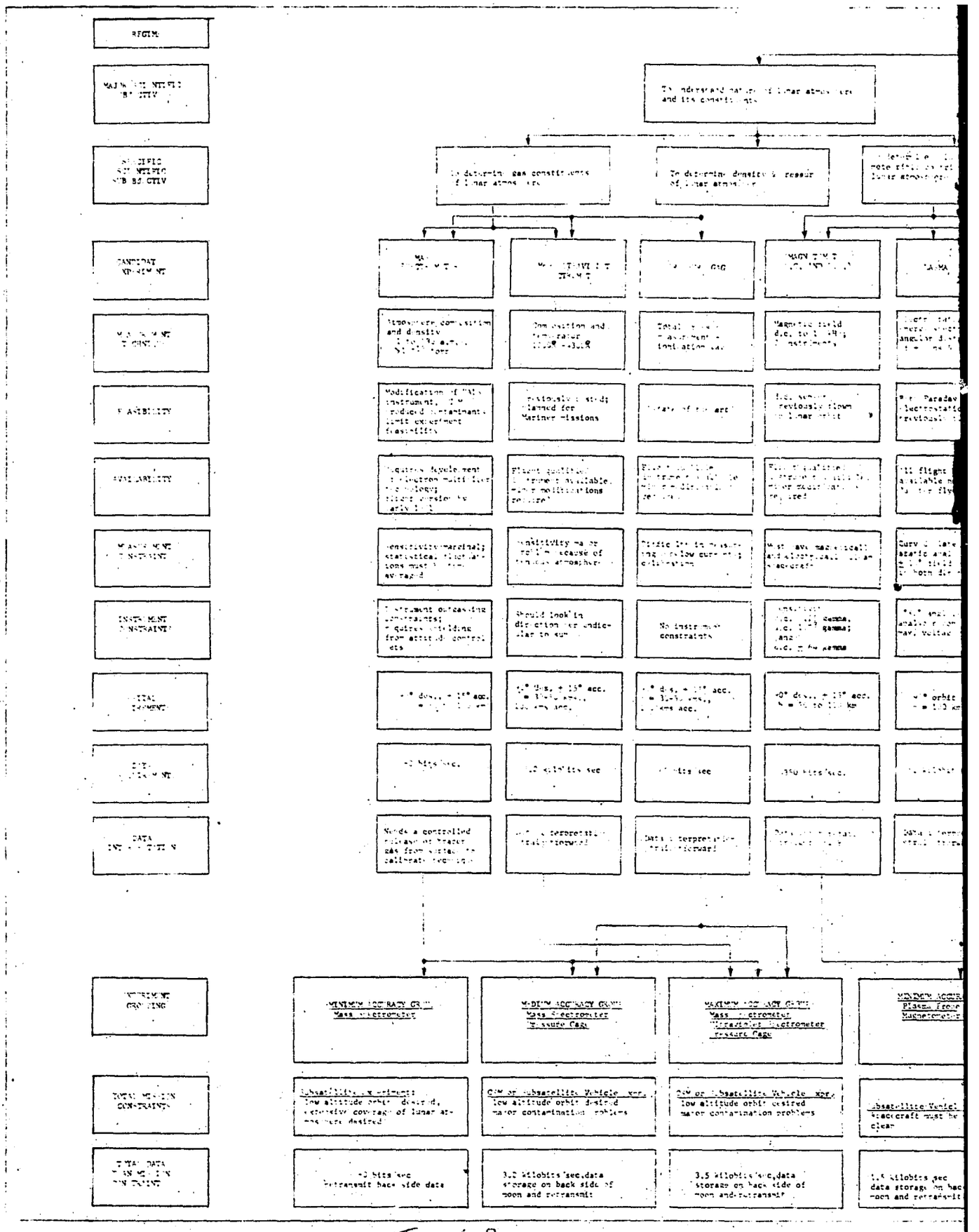
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and

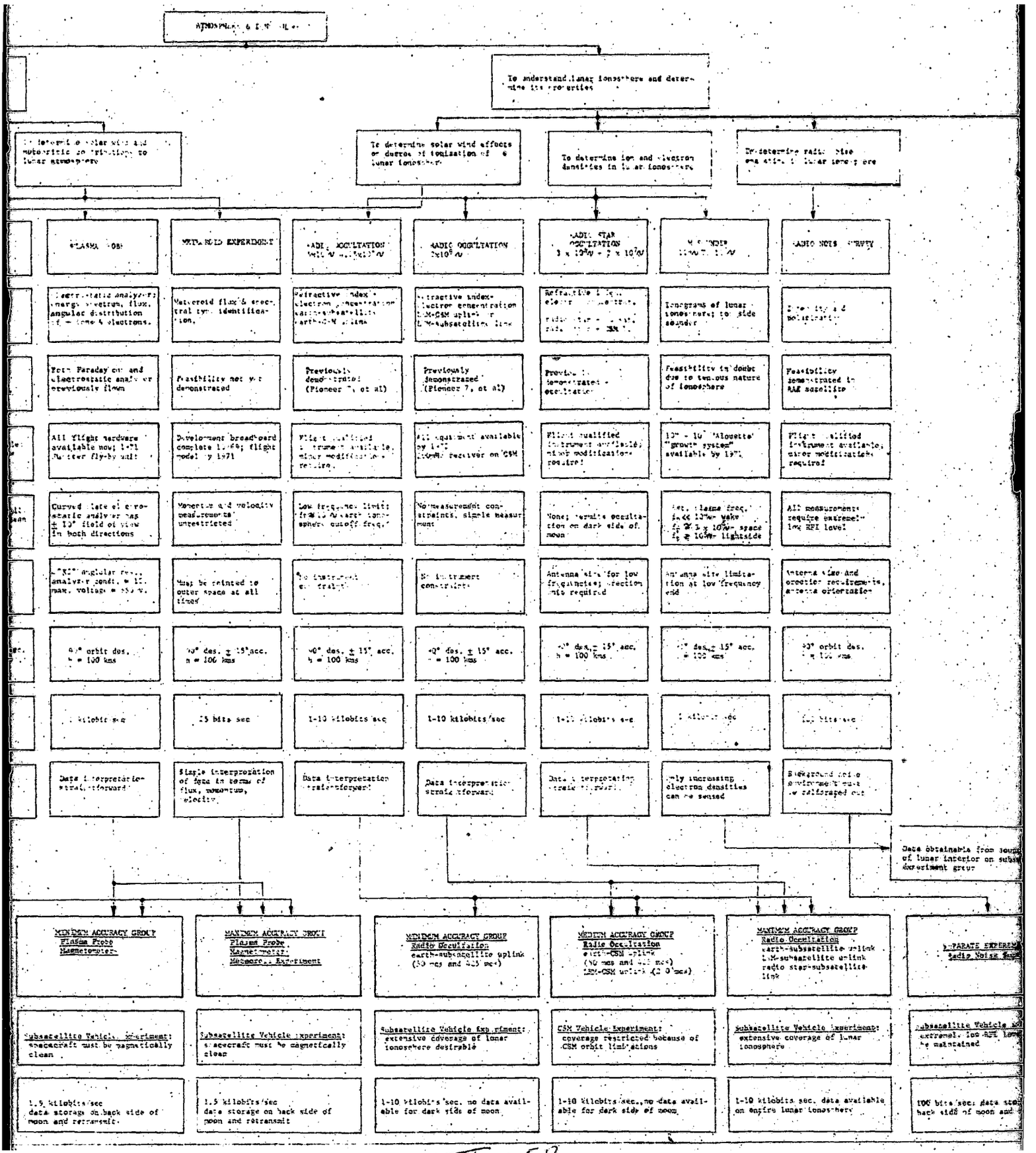
Indicates a roadblock barrier

LUNAR ORBITAL SCIENCE EXPERIMENT GROUPINGS FOR SURFACE REGIME

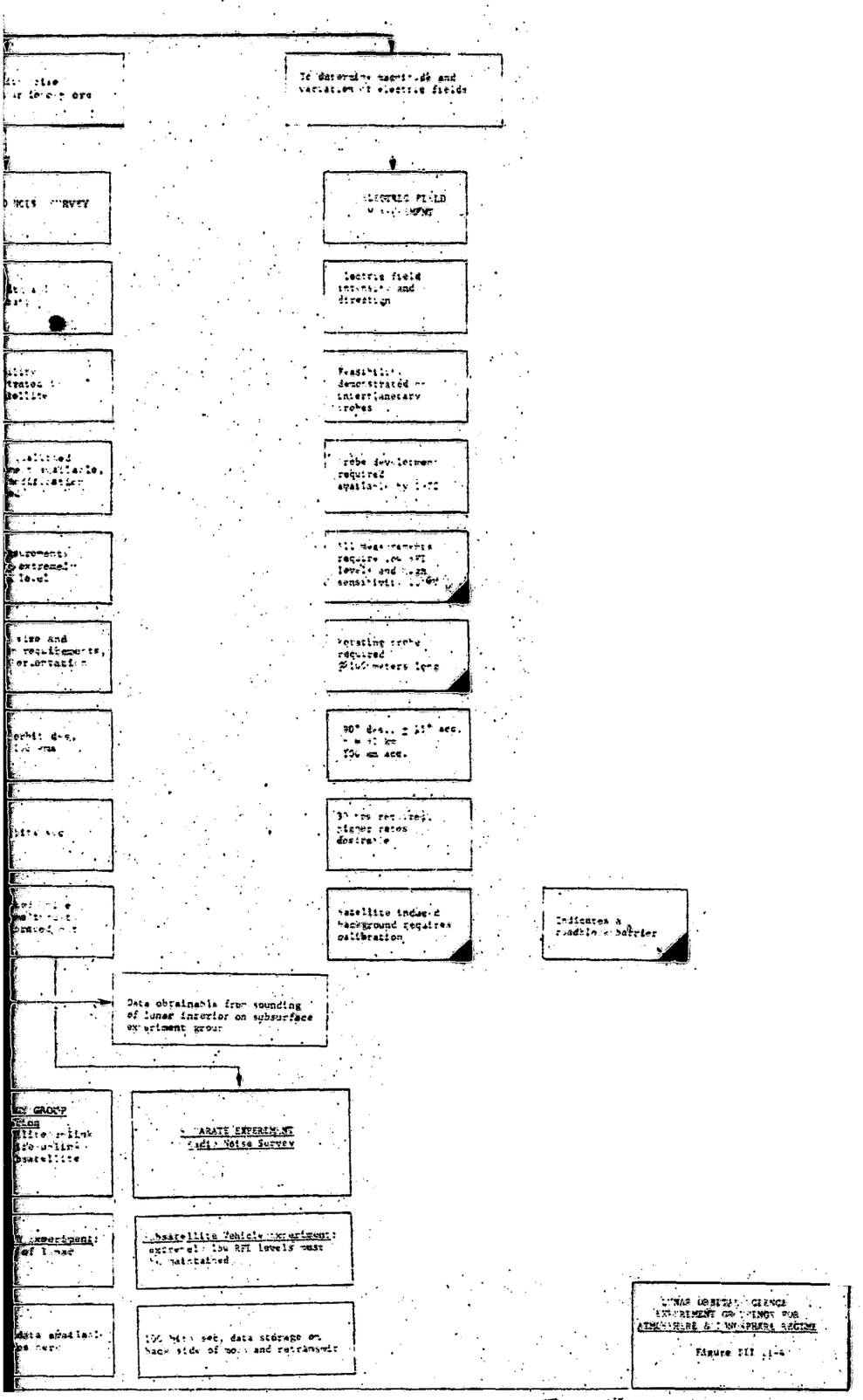
Figure III .1-3


REGIME
To determine gas constituents of lunar atmosphere
To determine density & pressure of lunar atmosphere
Mass spectrometer
Mass spectrometer
Ionization gage
Magnetometer
Thermocouple
Ion gage
Plasma probe
Magnetometer
Mass spectrometer
Ultraviolet spectrometer
Pressure gage
Plasma probe
Magnetometer
Subsatellite w/instruments
Low altitude orbit desired
S/C or subsatellite Vehicle
Low altitude orbit desired
Major contamination problems
S/C or subsatellite Vehicle
Low altitude orbit desired
Major contamination problems
Subsatellite/Vehicle
Spacecraft must be clean
3.2 kilobits/sec, data storage on back side of moon and retransmit
3.5 kilobits/sec, data storage on back side of moon and retransmit
1.6 kilobits/sec data storage on back side of moon and retransmit

III - 5 A

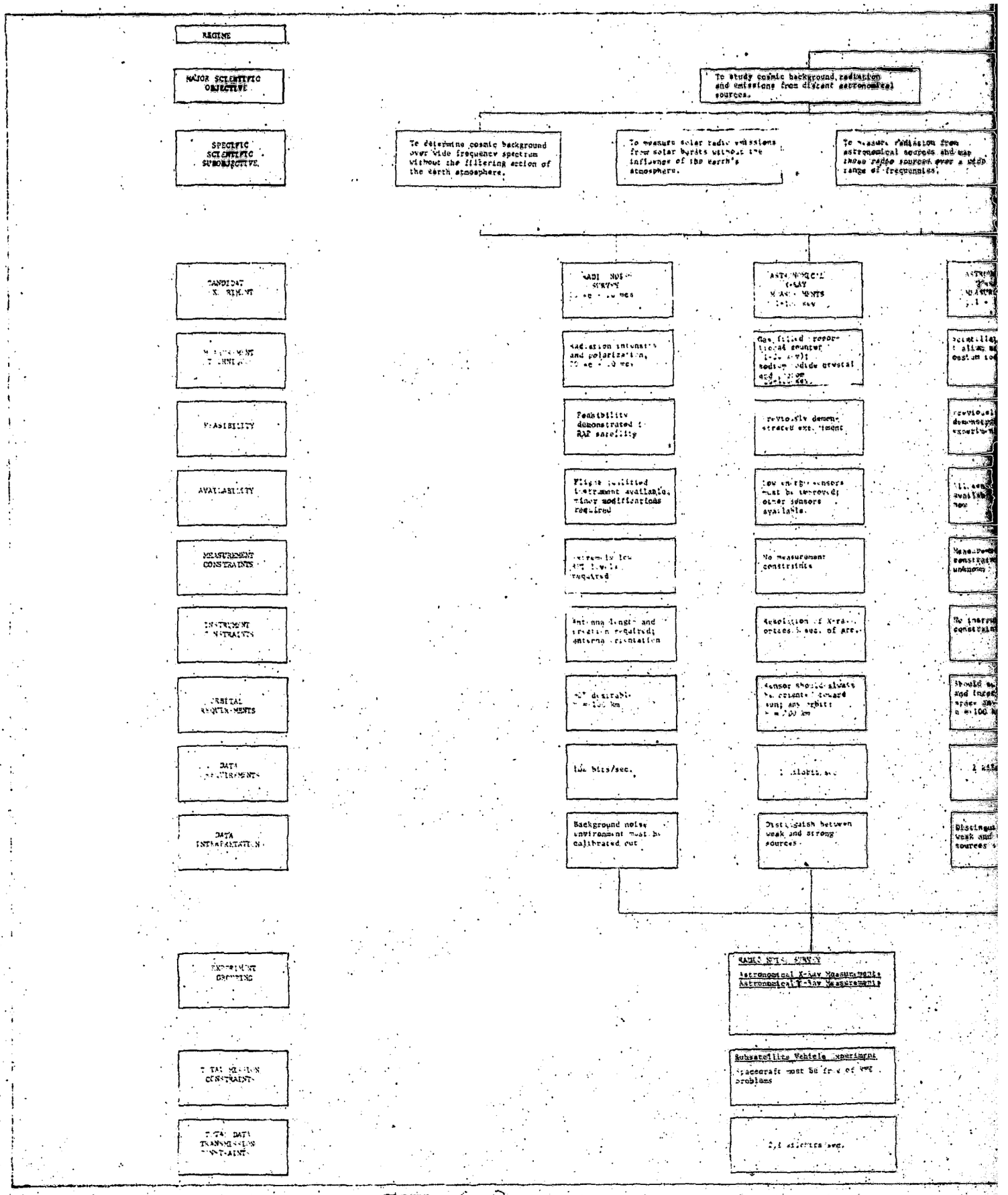

To understand lunar ionosphere and determine its properties
To determine solar wind effects on degree of ionization of lunar ionosphere
To determine ion and electron densities in lunar ionosphere
To determine radio noise environment of lunar ionosphere
PLASMA PROBE
METEOROID EXPERIMENT
RADIO OCCULTATION
RADIO OCCULTATION
RADIO STAR OCCULTATION
RADIO NOISE SURVEY
Data interpretation straightforward
Simple interpretation of data in terms of flux, momentum, velocity
Data interpretation straightforward
Data interpretation straightforward
Data interpretation straightforward
Only increasing electron densities can be sensed
Background noise environment must be extrapolated out
Data obtainable from source of lunar interior on subsatellite experiment group
MEDIUM ACCURACY GROUP
Plasma Probe
Magnetometer
MAXIMUM ACCURACY GROUP
Plasma Probe
Magnetometer
Meteoroid Experiment
MEDIUM ACCURACY GROUP
Radio Occultation
earth-subsatellite uplink
(50 mcs and 425 mcs)
MEDIUM ACCURACY GROUP
Radio Occultation
earth-CSM uplink
(96 mcs and 425 mcs)
CSM-CSM uplink (2.0 kcs)
MAXIMUM ACCURACY GROUP
Radio Occultation
earth-subsatellite uplink
LM-subsatellite uplink
radio star-subsatellite link
SEPARATE EXPERIMENT
Radio Noise Survey
Subsatellite Vehicle Experiment: spacecraft must be magnetically clean
Subsatellite Vehicle Experiment: spacecraft must be magnetically clean
Subsatellite Vehicle Experiment: extensive coverage of lunar ionosphere desirable
CSM Vehicle Experiment: coverage restricted because of CSM orbit limitations
Subsatellite Vehicle Experiment: extensive coverage of lunar ionosphere
Subsatellite Vehicle ... extremely low RFI level must be maintained
1.5 kilobits/sec data storage on back side of moon and retransmit
1.5 kilobits/sec data storage on back side of moon and retransmit
1-10 kilobits/sec, no data available for dark side of moon
1-10 kilobits/sec, no data available for dark side of moon
1-10 kilobits sec, data available on entire lunar ionosphere
100 bits/sec data stored back side of moon and

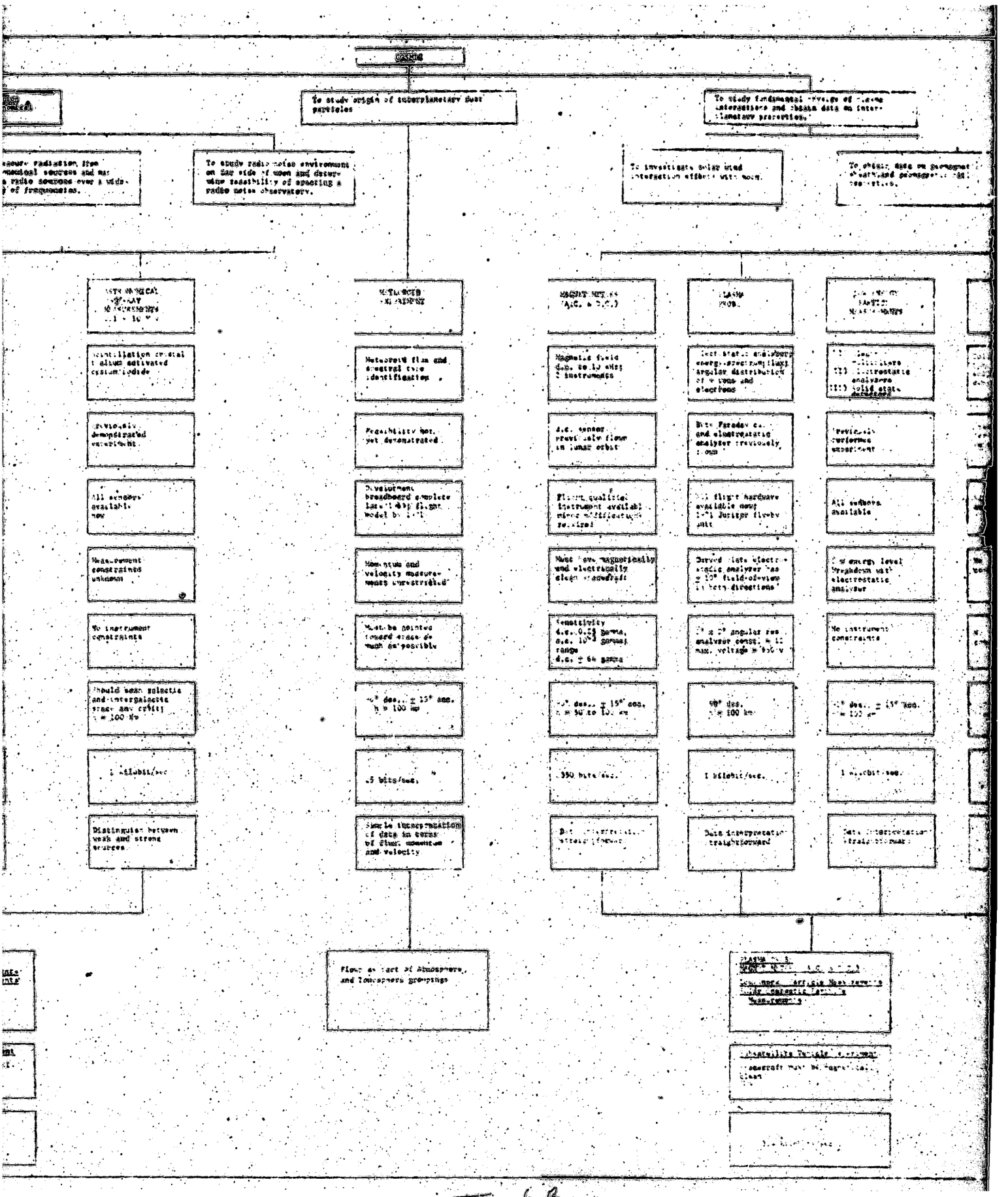
MARTIN MARIETTA CORPORATION
DENVER, COLORADO



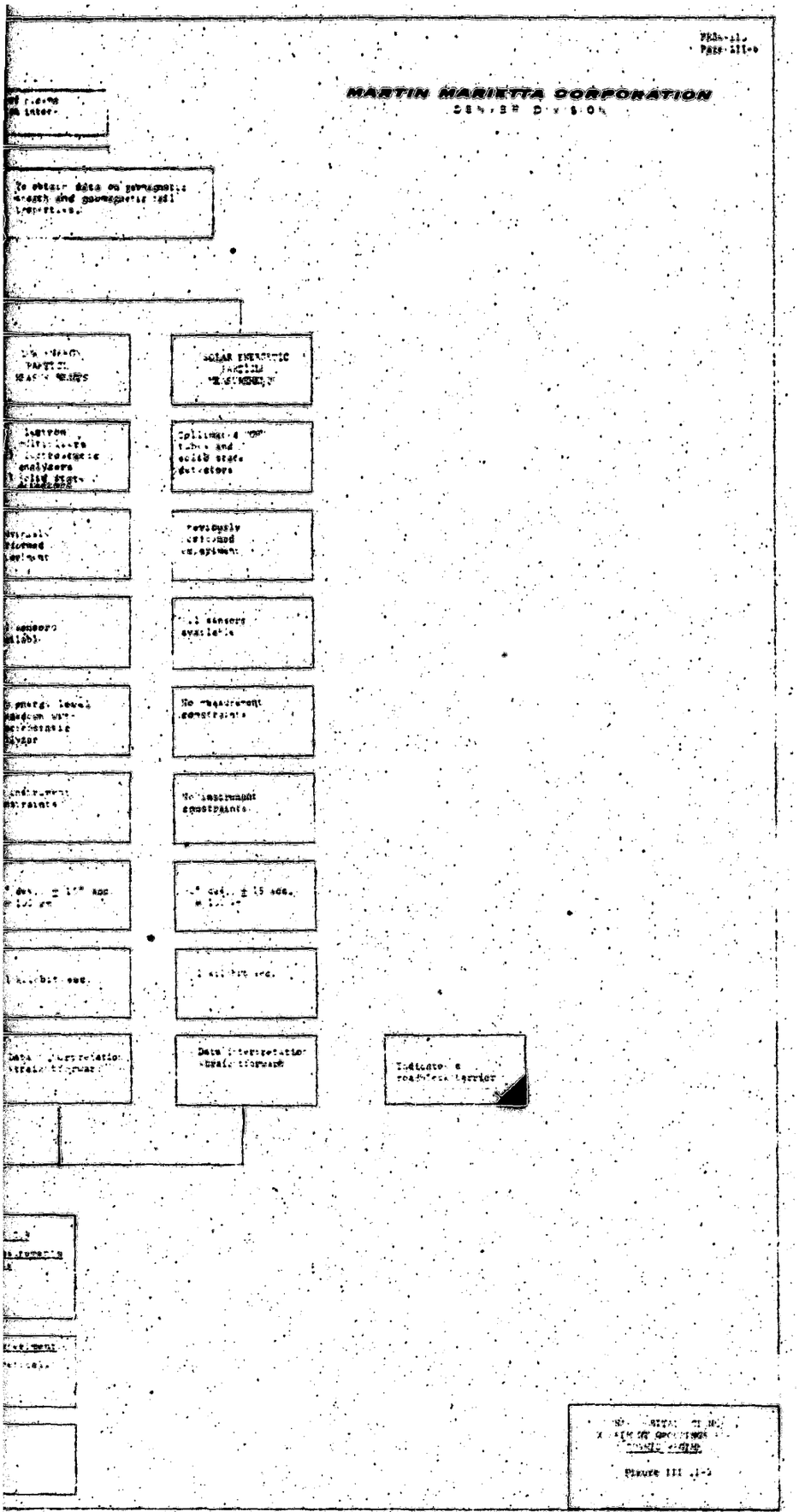
LUNAR ORBITAL SCIENCE EXPERIMENT GROUPINGS FOR ATMOSPHERE & IONOSPHERE SECTION

Figure III 3-4


REGIME
MAJOR SCIENTIFIC OBJECTIVE
To study cosmic background radiation and emissions from distant astronomical sources.
SPECIFIC SCIENTIFIC SUBOBJECTIVE
To determine cosmic background over wide frequency spectrum without the filtering action of the earth atmosphere.
To measure solar radio emissions from solar bursts without the influence of the earth's atmosphere.
To measure radiation from astronomical sources and map these radio sources over a wide range of frequencies.
CANDIDATE INSTRUMENT
MEASUREMENT TECHNIQUE
FEASIBILITY
AVAILABILITY
MEASUREMENT CONSTRAINTS
INSTRUMENT CONSTRAINTS
ORBITAL REQUIREMENTS
DATA REQUIREMENTS
DATA INTERPRETATION
Radiation intensity and polarization, 10 mc - 10 mc
Gas filled proportional counter (1-20 kev) sodium iodide crystal and plastic (20 kev-
Feasibility demonstrated by RAE satellite
Flight qualified instrument available, minor modifications required
Extremely low noise levels required
No measurement constraints
Antenna length and rotation required; antenna orientation
Resolution of X-ra... orces b ecc. of arc.
Sensor should always be oriente toward sun; an orbit r = 7,00 km
100 bits/sec.
Background noise environment must be calibrated out
Distinguish between weak and strong sources
INSTRUMENT GROUPING
Astronomical X-ray Measurements
Astronomical X-ray Measurements
TOTAL MISSION CONSTRAINTS
Subsatellite Vehicle Experiment
Spacecraft must be free of RFI problems
TOTAL DATA TRANSMISSION REQUIREMENT
2.1 minutes/day.


To study origin of interplanetary dust particles
To study radio noise environment on far side of moon and determine feasibility of operating a radio noise observatory.
To investigate solar wind interaction effects with moon.
To study fundamental aspects of plasma interactions and obtain data on interplanetary properties.
To obtain data on geomagnet
Meteoroid flux and spectral type identification
Feasibility not yet demonstrated
Development breadboard complete Last flight model
Momentum and velocity measurements unrestricted
Must be pointed toward areas of much composition
Sensitivity
Should scan galactic and intergalactic areas any orbit
Distinguish between weak and strong sources
Simple interpretation of data in terms of flux, momentum and velocity
Flow characteristics of Atmosphere, and Ionosphere groupings
III - 6B

MARTIN MARIETTA CORPORATION
ORLANDO DIVISION



shown in FIGURE III.1-2. Now this regime emerges with a full complement of groups necessary to solve the subsurface exploration problem.

The result of the roadmap development was the identification of a large number (57) of experiment groupings. These groupings reduced the extremely large number of possible combinations of 30 experiments to more manageable proportions. A further reduction of combinations was achieved by defining minimum, medium, and maximum accuracy groups. These groups reflect estimates of experiment sensitivity, and additionally provided a measure of system complexity. Minimum accuracy groupings allow early flights to obtain some of the data useful in meeting the lunar science objectives. Maximum accuracy groupings provide more and better data, but generally at somewhat higher cost, weight, and data transmission rates.

The number of missions required to meet the lunar science objectives was not estimated. However, the groupings presented as the roadmap output allow future mission estimates to be made, based on the ground rules and vehicles available at that time. The type of satellite on which each experiment candidate should be flown in shown on Table III.1 -1.

Table III.1-1
Satellite Requirements for Lunar Orbital Experiments

Command and Service Module	C&SM or Stable Satellite
Metric and Stellar Camera Panoramic Camera Laser Altimeter Radar Imager (if image on film)	IR Mapping Radiometer IR Spectrometers Microwave Radiometers Radar Imager Radar Altimeter RF Reflectance Swept Frequency UHF Radiometer Bistatic Radar Electromagnetic Measurements Photoelectric Photometer RF Transponder** UV Spectrometer** Meteo[illegible]s Solar [illegible]d Foil Laser Retroreflector Gravity Gradiometer Electric Field* Radio Noise Survey
Stable Satellite	
Geochemistry Experiments* Mass Spectrometer*	
Spinning Satellite	
AC Magnetometer DC Magnetometer Plasma Probe Low Energy Particles Solar Energetic Particles	

*Can use on C&SM; however, possible background noise, contamination problem. Prefer more extensive coverage provided by higher inclination, longer mission.
**Can also use on spinning satellite.

Four missions were identified by NASA as candidates for the lunar science program. Four representative groupings, shown on Table III.1-2, were developed. Subsequent sections of this report dealing with subsystem support requirements are based on these groupings. It should be noted that the groupings, derived jointly by NASA and MMC early in the study, hold up quite well when compared to the roadmap.

Table III.1-2
Potential Experiment Groupings

Command and Service Module	Stable Satellite
Metric and Stellar Camera Panoramic Camera Photoelectric Photometer Laser Altimeter Solar Wind Foil	Geochemistry Experiments Mass Spectrometer RF Transponder Meteoroids
Experiment Weight 712 lbs Inclination 0-15° Coverage 1-4%	Experiment Weight 526 lbs Inclination 90° Coverage 100%
Stable Satellite	**Spinning Satellite**
IR Mapping Radiometer IR Spectrometer Microwave Radiometer RF Reflectance Bistatic Radar Radar Altimeter RF Transponder Electromagnetic Measurements Radio Noise Survey	AC Magnetometer DC Magnetometer Plasma Probe Electric Field Low Energy Particles Solar Energetic Particles RF Transponder
Experiment Weight 644.5 lbs Inclination 90° Coverage 100%	Experiment Weight 73.5 lbs Inclination 0-90° Coverage 100%

There will be further groupings in terms of payload capability which can be achieved by combining the scientific discipline groups identified on the roadmap. The final selection of the groupings will be based on constraints such as weight, lunar orbit inclination, nodal points and duration desired.

III.2 Experiment Requirements - Experiment requirements were obtained by a survey of responsible sciantists and potential industry vendors. The requirements obtained were then compiled and documented

in Experiment Description Documents (EDD) enclosed as Appendix A of this report. The EDDs were prepared to the format of the NASA/ MSC Phase A report, Appendix A, and include experiment objectives and measurement concepts, with the requirements of specific hardware which most nearly satisfies the objectives.

Many states of development are represented by the experiments. Of the thirty experiments surveyed, six are of flight hardware configuration, requiring only adaptation to the potential spacecraft. Another seventeen experiments are state-of-the-art. These have not necessarily been previously flight qualified, but suitable instruments are available which can be modified for lunar flight. The remaining seven experiments will require sensor technology improvement and development. The state of sensor development can be deduced from Table III.2-1

Experiment requirements are summarized in Table III.2-2. These requirements are also presented in somewhat greater detail in the EDDs of Appendix A. The majority of the experiments have many common requirements; however, some differences can be seen.

Power requirements for most experiments are compatible for use with 28 volt dc systems. The K-band microwave radiometer is an exception since it is an application of a Nimbus unit which uses -24.5 volt power. Another exception is the bistatic radar which requires 50 volt, 2400 Hz square wave primary power. The Mariner 1967 Venus probe unit is reflected, again an application of existing hardware.

Temperature limits are reasonably broad, with the exception of film cassettes, some of the geochemistry units, and the IR spectrometer. This latter unit presents the tightest temperature control ($23^{o}C \pm 5^{o}$) limit.

Data storage and transmission requirements range from zero to a video link requirement of 12 megabits/second for the photoelectric photometer. All instruments requiring greater than eight bit resolution have individual analog to digital converters, and present a digital data stream to the spacecraft data system.

Capability to hold the spacecraft within $\pm 1^{o}$ of local vertical should be adequate for all sensors requiring this pointing mode. Control of attitude rates on the order of 0.05 degrees per second appears to be the most stringent requirement. Several of the sensors require pointing modes other than local vertical. The solar X-ray and cosmic ray portions of the geochemistry experiment and

Table III.2-1
Experiment Development Status, Other Programs

Experiment	*	Development Program
Metric and Stellar Camera	2	Fairchild study for MSC
Panoramic Camera	2	None for space application
Laser Altimeter	3	Control Data Corp. development contract
Radar Altimeter	2	None orbital, SIB unit flown to near orbit
Geochemistry Experiments**	2	AS&E study for GSFC
Mass Spectrometer	2	ALSEP (similar)
IR Mapping Radiometer	3	MSC study
IR Spectrometer	2	None
Microwave Radiometer	2	Nimbus E (Space General)
RF Transponder	1	C & SM, Lunar Orbiter
Bistatic Radar	2	Mariner 1967
Photoelectric Photometer	2	None
RF Reflectance	2	MSC aircraft units; no orbital development
Electromagnetic Measurements	3	None
Noise Survey	3	RAE Explorer in earth orbit
Radar Imager	2	Venus probe (JPL - Mariner)
Meteoroids	3	MSC study
Solar Wind Foil	2	ALSEP
Magnetometer (D.C.)	1	Already developed for many space missions (AIMP, Pioneer)
Magnetometer (A.C.)	1	
Plasma Probe	1	
Low Energy Particles	1	
Solar Energetic Particles	1	
Electric Field	3	None
UV Spectrometer	2	Mars probe - C.U. Mariner
Laser Retroreflector	2	ALSEP
Gravity Gradiometer	3	None

*Category 1 - Adaptable Flight Hardware Available
2 - State-of-the-art
3 - New Development
**Combination of four experiments

Experiment	Physical Electrical: Weight	Ret Weight	Dimensions	Vltg (DC)	Avg Pwr	Temp Limit (°C)	Data Requirements: Class	Format	Bit Rate	Attitude Cont: Type	Max Error	Controls	Remarks
Electromagnetic Meas. 75' Boom ant (4) 25' short ant (4) [illegible] Receiver	70 7 130 20 227	0 0 0 0	4" x 6" x 12" (each) 12" x 3" x 6" (each) 2" x 4" x 8" (each) 6" x 6" x 12"	- - 28±4 28±4	- - 2 50	+[illegible] to +50°	1	Digital	1[illegible] BPS	Loc Vert	±1°	1. Power On-Off 2. Antenna Deploy	* Stowed Dimensions
Mass Spectrometer Mass Spectrometer	15	0	11.5" x 7.5" x 12"	28±4	12	-20 to +70°	1 10	Digital Analog	80 BPS 0.1 BPS (Total)	Loc Vert	Not Crit.	1. Power On-Off 2. Deploy Seal Cover	
Magnetometer (AC) Sensor (3 reqd) Electronics	1 4 5	0 0 -	1" x 1" x 1.2" (each) 6" x 6" x 6"	[illegible]	-	[illegible] to +70° -50 to -50°	3 5	Digital Analog	300 BPS each 1 BPS each	Spin Stabi-lized *	[illegible]°	None Required Desired: 1. Sample Rate Select 2. On-off (each sensor) 3. Gain control (each of 3 BPS)	* Axis of spinning satellite normal to plane of ecliptic. Attitude knowledge ±5° Position knowledge ± 1 ER
Magnetometer (DC) Sensor Electronics	1 2 3	0 0	4" x 6" x 3.5" 6" x 6" x 3"	28±4	1.5	-50 to -50° -5 to +40°	1	Digital	24 BPS	Spin Stabi-lized *	±5°	None Required Desired: 1. Sample Rate Select 2. Calib. Sequence 3. On-off (each sensor) 4. Gain control	* Axis of spinning satellite normal to plane of ecliptic Attitude knowledge ±5° Position knowledge 1ER
Plasma Probe	6.5	5	7" x 7-3/4" x 6-3/4"	28±4	3.5	-15 to +50°	1 1	Digital Analog	16.7 BPS. 3 BPS	Spin Stabi-lized *	±5°	None Required Desired: 1. Power On-Off 2. Calibrate Mode 3. Data Record (4 modes)	Requires knowledge of sun angle to ±[illegible]° attitude knowledge ±5°, position knowledge of 1 RM. * Axis of spinning satellite normal to plane of ecliptic.
Electric Field Antennas (4 req'd) Electronics	[illegible] 5	0	7" x 11" x 7200" (ea) (7" x 22" x 4" as stored) [illegible]" x [illegible]" x 7"	28±4 28±4	- 3	-10 to +55° -10 to +55°			30 BPS	Spin Stabi-lized *	±5° ±0.5°/sec	1. Power On-Off 2. Antenna Deploy	* Axis of spinning satellite normal to plane of ecliptic
Meteoroids Impact/Flash Inst.	22	0	4" x 22" dia	28±4	5	-55 to +80°		Digital	25 BPS (400 bits twice/day)	Any	Not Crit.	1. Power On-Off 2. Expt. Deploy (If necessary)	Preferred orientation is away from lunar surface, however any orientation is sat. Attitude knowledge required post flight.
Laser Retroreflector Retroreflector Array	12	0	15" x 15" x 4"	None	None	-50 to -50° est.	None	-	-	Inert. Earth Orient.	±5°	None Required	Requires earth based telescope within ±12° of retroreflector ϕ axis.
Noise Survey Receiver (2) Antenna	2 40 lb *	0 0	7" x 4" x 2" (ea) 120' max tip-to-tip	28±4 -	5 -	-10 to +65° Thermal Cont. Coatings	1	Digital	40 BPS	Inert.	±15° ±0.5°/s	1. Power On-Off 2. Receiver Mode Control 3. Antenna Deploy and	* Includes deployment mechanism
Gravity Gradiometer Gravity Meter (2)	110	0	1-1/2" dia x 40"	28±4	5	-10 to +50° est.	2	Digital	10.2 BPS	Loc Vert	-	1. Power On-Off	Attitude knowledge to ±5°
Solar Energetic Particle Electronic Detector	10	0	6" x 5" x 10"	28±4	5	-10 to +50°	1 2	Digital Analog	100 BPS 7 BPS	Any	*	1. Power On-Off 2. Deploy Protec. Covers	* Attitude & position histories required. Attitude knowledge ±5°
Low Energy Part Detectors	10	0	4" x 7" x 12"	28±4	[illegible]	[illegible] to [illegible]	1 2	Digital Analog	1000 BPS 7 BPS	Any	*	1. Power On-Off 2. Deploy Protec. Covers	* Attitude & position histories required. Attitude knowledge ±5°

Table III-4. C&SM LUNAR ORBITAL SCIENCE - EXPERIMENT REQUIREMENTS SUMMARY

Experiment	Physical Electrical						Data Requirements			Attitude Control		Controls	Remarks
	Weight	Ret Weight	Dimensions	Vltg (DC)	Avg Pwr	Temp Limit (°C)	Chan	Format	Bit Rate	Type	Max Dev		
Metric and Stellar Camera													
Metric	171	21	13.9" x 27.3" x 14.3" Return Dimens. 7" x 8" dia.	28±4	60	+5 to +30° (est)	5 (est)	Analog	1 BPS (total)	Loc Vert	±2° ±0.05° per sec	1. Power On - Standby - Off	Boresight Metric and Stellar to 5 arc sec
Stellar (2)	72	12	15.2" x 19.2" x 7.6" (each) Return Dimens. 3.8" x 3.3" dia. (each)	28±4	20	+5 to +30°	5 (est)	Analog	1 BPS (total)				Roll S/C once (minimum), pref. 3 times to view space to calibrate metric/stellar alignment
	243	33			80								
Panoramic Camera	315	65	20" x 5" x 20" Return Dimens. 5" x 20" dia.	28±4	500	+5 to +30° (est)	5 (est)	Analog	1 BPS (total)	Loc Vert	±2° ±0.05° per sec	1. Power On-Off 2. Mode Select (2 modes)	Primary operate mode in conjunction with Metric/Stellar system
Radar Altimeter													
Antenna	15.0	0	15" x 18" x 3"			-20 to +70°	1	Digital	72 BPS	Loc Vert	+5° ±0.05° per sec	1. Power On - Standby - Off 2. Antenna deploy (if required)	2 ft maximum between antenna and electronics
Electronics	26.5	0	8" x 8" x 11"	28±4	70		20	Analog 0 - 5V	8 BPS (total)				
	41.5	0			70								
RF Reflectance													
Electronics	20	0	8" x 8" x 8"	28±4	70	-20 to +70°	1	Digital	1440 BPS 160 BPS (total)	Loc* Vert	±1° P ±2° Y,RR ±0.05° per sec	1. Power On - Standby - Off 2. Dynamic Range Switch	Can combine altimeter and scatterometer if both expts fly together
Antenna	5	0	36" x 3" x 2"				20	Analog 0 - 5V					
	25	0											
Laser Altimeter	100		20" x 18.5" x 15"	28±4	100	-55 to +71°	1	Digital	2 BPS (max)	Loc Vert	±2° ±0.05° per sec	1. Power On - Standby - Off	Boresight with metric camera to 10 arc seconds
Geochemistry													
Gamma-Ray	85	0		28±4	18	20 to 30				↑ Loc Vert ↓ Solar Solar None	±2° ±2° ±2° ±2° ±15° ±15° ±0.05° per sec	1. Power On-Off 2. Boom Deploy 3. Calibrate Mode	Roll S/C to view space with gamma-ray sensor 2 times, 15 minutes each preferably on dark side
Neutron Albedo	15	0		28±4	3	15 to 35							
Alpha Particle	20	0	Dimensions	28±4	2	15 to 35							
X-Ray Sensors	175	0	Unknown	28±4	28	0 to 50	1	Digital	280 BPS				
Solar X-Ray	15	0		28±4	2	0 to 50							
Cosmic Ray	15	0		28±4	2	0 to 50							
Electronics	150	0		28±4	20	0 to 50							
	475	0			75								
Rendezvous													
Electronics	12	0	12" x 10" x 3"	28±4	17	-12 to +43		Internal		Internal		1. RF On-Off 2. Ranging On-Off	Use existing on-board equipment for CSM rendezvous
Antenna	2	0		-	-								
	14	0											
Solar Wind Foil	15	0.25	14" x 10" x 10" Return Dimens. 36" x 1.5" dia.	28±4	0	Extender: -10 to +65° Foil: -65 to +100	3	Analog 0 - 5V	3 BPS	Solar	±15° ±0.5°/sec	1. Boom Extend/Retract 2. Foil Extend/Retract	Foil Area X Experiment Time = 90 ft² - hrs minimum

Table III.e-2 (Cont'd) C&SM LUNAR ORBITAL SCIENCE - EXPERIMENT REQUIREMENTS SUMMARY

Experiment	Physical Electrical						Data Requirements			Attitude Control		Controls	Remarks
	Weight	Ret Weight	Dimensions	Vltg (DC)	Avg Pwr	Temp Limit (°C)	Chan	Format	Bit Rate	Type	Max Dev		
IR Mapping Radiometer													
Radiometer	75	0	15.8" x 15.8" x 9.8"			-20 to	1	Digital	2500 BPS	Loc	±2°	1. Power On-Off	*Radiator on radiometer must view space (boom mounted probably)
Electronics	75	0	TBD	28±4	150	+40°				Vert	±0.2° per sec	2. *Boom Extend/ Retract	
	150	0											
Microwave Radiometer													
Antenna (L&K Band)	70	0	12' x 12' x 2.5" / 3' x 3' x 2.5"	-24.5		-10 to +60	2	Digital	200 BPS 1 SPS	Loc	±5°	1. Power On - Standby - Off	For K-band only; weight, power, and data rate are 1/2 the amounts shown. Attitude knowledge of ±2° required.
Electronics (L&K Band)	20	0	8" x 6" x 13" / 8" x 6" x 13"	+0.5	40		5	Analog 0 - 5V	per chan x 8 bits = 40 BPS		±0.1° per sec	2. Mode Select (2 Modes)	
	90	0											
IR Spectrometer													
Spectrometer	15	0	16 1/2" x 14 1/2" x 14"	28±4	15	22±10°	1	Digital	4x10^5 bits/ orbit (min) 1x10^7 bits/ orbit (max)	Loc Vert	±40° 0.50° per sec	1. Power On-Off	Roll S/C to acquire sun in sensor F.O.V. for calibration at least once; prefer once/day.
Electronics	15	0	8" x 8" x 12"	28±4									
	30												
Photoelectric Photometer	20	0	24" x 14" dia	28±4	4	-120 to +120°	1	Video	1.0 MHz	Loc	±1°	1. Power On-Off	Attitude knowledge to ± .° required for post flight correlation.
							1	Analog 0 - 5V	1 BPS (total)	Vert		2. Filter Select for Multispectral	
Radar Imager													
Antennas (2)	30	0	118" x 196" x 3" / 19.6" x 31.5" x 2"	28 +6	0 120	-20 to +75° -20 to +75°	2	Digital Analog	20 BPS sec.		±1° V&P ±2° R	1. Power On - Standby - Off	2 ft. maximum between antenna and trans/rec
Data Processor (4)	80	0	12" x 8" x 6" (ea)	-4	140	-20 to +75°	20	0 - 5V	20 BPS (total)	Loc	±0.05°	2. Antenna deploy (If required)	
Trans/Rec (2)	160	0	36" x 20" x 12"(ea)							Vert	per sec	3. Focus/Unfocused Mode Switch	
	270	0											
Film Recorders (2) (Optional)	80	5	9" x 9" x 18" (ea) Return Dimens. 9" x 5" x 3"		80 340	+5 to +30°							
U.V. Spectrometer	35	0	25" x 11" x 6"	28±4	24	-20 to +40°	2	Digital	1600 BPS each	Loc Vert	±3° ±.02°/ minute	1. Power On-Off 2. Deploy aperture cover or sun cover if used	Spectrometer must view above lunar horizon
Bistatic Radar													
Receiver	5	0	6" x 7" x 8.3"	30V, 2400 Hz Square Wave	2	0 to +50°	1	Digital	10^3 to 10^4 BPS Use 2 MHz	Loc Vert	±10°	1. Power On-Off 2. Antenna Deploy (If required)	
Antenna (VHF)	10	0	118" x 118" x 3"										
Antenna (UHF)	5	0	13 1/2" x 13 1/2" x 3"										
	20	0											

the solar wind foil require solar orientation. The laser retro-reflector requires an earth orientation. The bistatic radar is generally compatible with the local vertical orientation mode. However, the antennas for this instrument receive the reflected wave from the lunar surface resulting from an earth transmitted signal. Best data will be obtained by orienting the antennas perpendicular to the return reflected from the lunar surface, an orientation which changes with orbital position.

Interface with the C&SM display and control panel should be minimal. Of the experiments in the C&SM grouping, power on/off switching is the only requirement. If the geochemistry experiments are considered as part of the C&SM grouping, then an additional boom deployment function may be required. Subsatellite groupings of experiments can be operated by use of the subsatellite sequence system.

Environmental control requirements identified to this point in the program show a concern for electromagnetic interference, principally caused by simultaneous operation of experiments and supporting subsystems. Several requirements, peculiar to specific experiments, are seen. The neutron albedo portion of the geochemistry experiment must be shielded from potential neutron sources, such as the SM propellant tanks. (Note: The propellants have not been positively identified as sources of background noise. However, an evaluation should be made.) The laser altimeter will probably require some spacecraft provided coolant. The instrument is not designed at this point to dissipate all of the heat generated within the unit. Operation of the IR mapping radiometer on the dark side of the moon will require a cold reference source. Use of a radiator for this purpose is presently specified, which may present a difficult integration problem.

The experiments listed in Table III.2-3 will require some form of protection from spacecraft contamination. This is particularly true for those experiments operated on the C&SM. Contamination mechanisms and sources are not fully understood. However, those experiments operating at wavelengths approaching the depth of potential particle layers should be protected. Apertures and surfaces of all optical systems should be shielded while not in use.

Table III.2-3
Experiment Contamination Summary

Experiment	Requirement
Metric and Stellar Cameras	Shield lens; inhibit Reaction Control System during exposure
Panoramic Camera	Same as Metric
Laser Altimeter	Shield aperture of transmitter and receiver telescopes
Mass Spectrometer	Shield entrance aperture from attitude control jets and known outgassing
IR Mapping Radiometer	Shield entrance aperture
IR Spectrometer	Protect telescope and mirrors
Photoelectric Photometer	Shield optics
Solar Wind Foil	Protect from acidic atmospheres; do not expose to rare gases
UV Spectrometer	Protect entrance aperture from attitude control jets and known outgassing
Laser Retroreflector	Shield surfaces from jet plume, outgassing

The orbit requirements of most experiments are compatible with an 80km (approximate), circular, low inclination orbit. While many of the experiments could obtain more data with a high inclination, long duration mission, the objectives of most experiments are substantially met using the Apollo orbit. Some exceptions can be seen in Table III.2-4.

The crew requirements for operation of experiments in the C&SM grouping are summarized in Table III.2-5. As seen in the table, with the exception of EVA for data retrieval, the requirements on the astronauts are minimal. Crew requirements for all other experiments are shown in the experiment descriptions of Appendix A, should it be desired to include them in the C&SM grouping. Otherwise, these experiments are actuated automatically or on ground command. Potential astronaut functions are therefore limited to deployment of experiments and/or subsatellites and switching on/off. Other functions such as experiment scientific data monitoring are considered to be options only at this time.

Table III.2-4
Experiment Orbit Requirements Summary

Experiment	Altitude	Eccentricity	Inclination	Remarks
RF Transponder, Laser Retroreflector	600, 100, 50Km	0 to 0.3, altitude dependent	30, 50, 70, 85°	Desires multiple missions to achieve objectives
AC and DC Magnetometers, Plasma Probe	100, varying to 1000Km	circular changing to elliptical	0°	Initial circular, 100Km; boost in steps to 1000Km apogee
Gravity Gradiometer	65Km or less	zero	polar preferred, 15° acceptable	

Table III.2-5
Crew Requirements Summary

Experiment	Function
Metric and Stellar Cameras	Switch standby/on/off. Retrieve film at experiment conclusion.
Panoramic Camera	Same as Metric Camera
Laser Altimeter	Switch standby/on/off
RF Transponder	None required
Photoelectric Photometer	Switch standby on/off
Solar Wind Foil	Actuate experiment boom deployment, fire cover squibs. Retrieve foil at experiment conclusion.

Deployment requirements are summarized in Table III.2-6 Below. Of the experiments in the C&SM grouping, only the solar wind foil requires deployment.

Table III.2-6
Experiment Deployment Requirements Summary

Experiment	Requirement
IR Mapping Radiometer	Must extend experiment so that radiator can view space
Microwave Radiometer	Must erect L-band antenna (12' x 12' dimensions)
Electromagnetic Measurements	Deploy two mutually perpendicular 1500 ft. dipole antennas
Noise Survey	Deploy and extend "V" or dipole antennas (~60 to 120' length)
Radar Imager	Deploy L-band antenna (3 x 5 meter dimensions)
Solar Wind Foil	Deploy such that foil surface is not shielded from the sun
AC and DC Magnetometers	Deploy at least 10 feet from spacecraft to minimize interference Must rotate experiments if spacecraft is non-rotating
Electric Field	Deploy antenna mechanism, then extend antenna elements; antenna must spin
Gravity Gradiometer	Deploy gravity meters 15 feet from spacecraft

Operating timelines for all experiments are shown in Figure III.2-1. The timelines reflect operating periods which allow complete coverage of the lunar surface beneath the spacecraft or complete accomplishment of the experiment (as far as mission duration and inclination will permit). As seen from the figure, operation of the metric and panoramic cameras is not required every orbit. In fact, due to slow rotation of the moon and the wide fields of view (74°), the film capacity is not approached for complete coverage of the area under a 15° inclined orbit of three days duration. Operation of the laser altimeter is shown as continuous, although its prime function is to obtain altitude for camera correlation. The accurate altimetry data obtained can be used with other experiments. The radar imager and the microwave imager cannot operate concurrently. However, some mutual surface coverage can be achieved by operating the two

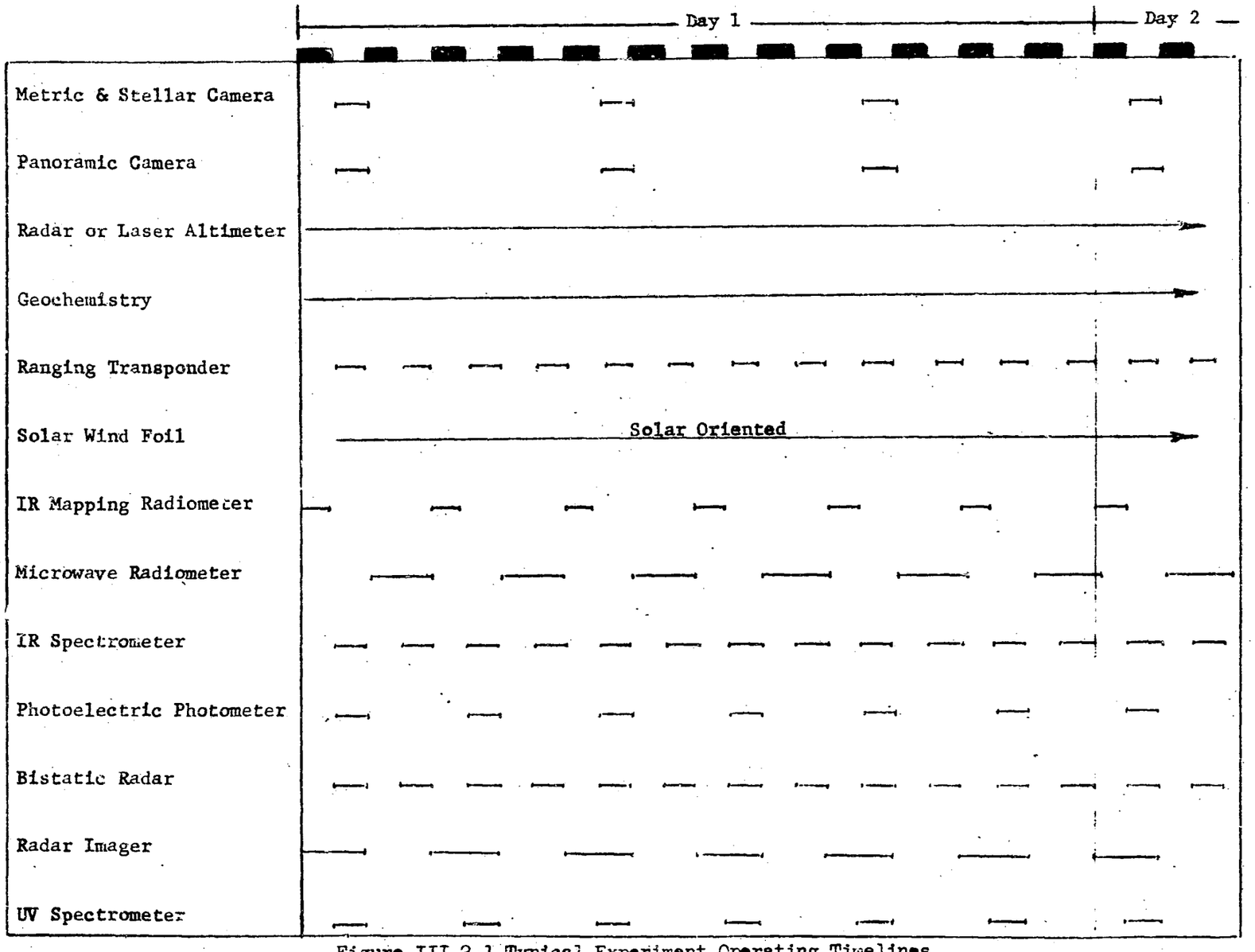


Figure III.2-1 Typical Experiment Operating Timelines

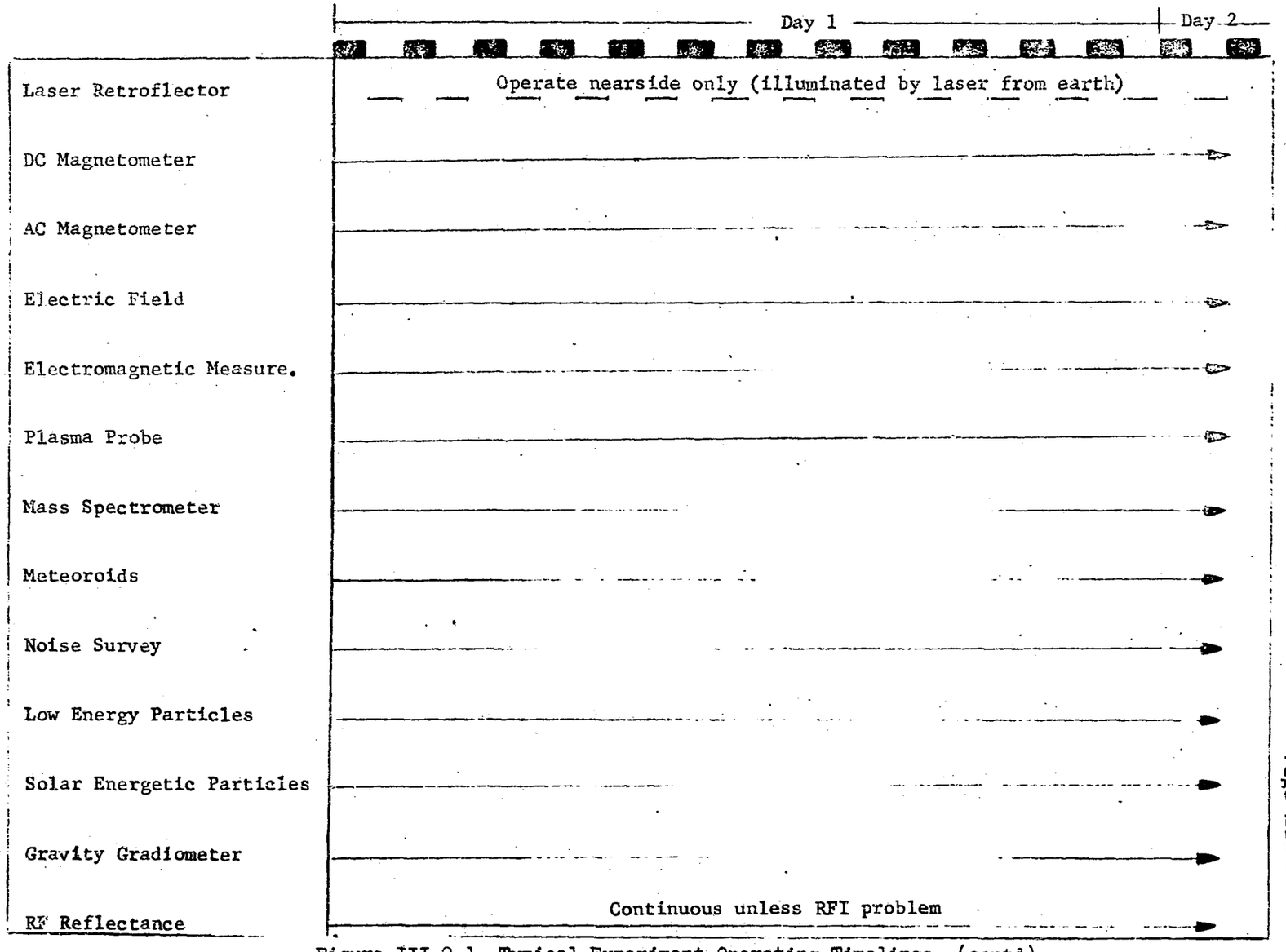


Figure III.2-1 Typical Experiment Operating Timelines (contd)

instruments alternately on every other orbit since their fields of view will overlap. As seen from the scientific roadmap (Section III.1) an IR radiometer and the microwave radiometer should operate concurrently.

Ground servicing, checkout and launch requirements are not well defined at this point. Some general observations can be made, however. An optical alignment facility must be provided to permit alignment of experiments to the spacecraft and to each other. All experiments requiring precise alignment will be delivered with an optical surface which defines the centerline of the field of view. Laboratory and checkout space which is temperature and humidity controlled should be provided. However, no requirement has been identified for clean rooms. There are a number of experiments which require deployment due to size and/or operation location on the spacecraft. An area will be required in which to checkout the deployment devices. Pyrotechnic simulators must be provided for vehicle checkout to simulate deployment.

A combined systems test of experiments (or simulators) and the spacecraft should be planned, not only to confirm system operating adequacy, but to test interference among simultaneously operated experiments. Experiment data should be recorded in these tests to be sure that no experiment affects the data collected by any other. Contamination covers should be actuated during the systems test and during prelaunch activities.

Peculiar checkout and servicing requirements are shown in Table III.2-7

Table III.2-7
Peculiar Ground Checkout and Servicing Requirements

Experiment	Requirement
Metric and Stellar, Panoramic Cameras	Load film within 3 days of launch. Maintain film temperatures. Check intervalometer, camera operation.
Laser Altimeter	Provide ground cooling if experiment operated more than a few cycles during checkout.
IR Mapping Radiometer	Provide cryogenics for detector cooling (or cryogenic heat sink for experiment radiator)
AC and DC Magnetometers	"Magnetically clean" area required for calibration, checkout. Possibly use manufacturer's facility during experiment acceptance for this purpose. Additional study required as to necessity of degaussing satellite.

Peculiar subsystem and support requirements are summarized in Table III.2-8. Reference should also be made to the other tables of Section III, since some peculiar requirements are shown there and are not repeated below.

Table III.2-8
Peculiar Subsystem and Support Requirements

Laser Altimeter	Spacecraft coolant at 15 $\pm$ 7° C required.
Mass Spectrometer	Controlled gas emission from lunar surface desired (not required).
RF Transponder, Laser Retroreflector	Spacecraft orbit and attitude should be unperturbed by onboard systems.
AC and DC Magnetometers, Plasma Probe, Electric Field	Antennas must be spinning during operation--spinning subsatellite candidates. Plasma probe requires accurate sensor for knowledge of spacecraft-sun line.
Metric and Stellar, Panoramic Cameras	Intervalometer required, either using spacecraft sequence system or included in experiment.
Microwave Radiometer	Requires 100 pps timing signal from spacecraft.
Optional Operation:	
Radar Imager	If operated in high resolution focused mode, film recorder (or ability to handle 10^8 BPS) required. EVA will then be required to retrieve film.
Microwave Radiometer	If included in C&SM grouping, provision of on-board processing and display will permit astronaut to make observations and adjust antenna scan, FOV, and gain to obtain data on areas of particular interest.

Experiments not operated manually will require control by the spacecraft sequence system. Furthermore, based on past experience, interface with spacecraft timing will be required for experiments in addition to the microwave radiometer listed above.

IV EXPERIMENT COST AND AVAILABILITY

A summary of cost and schedule data is presented in Figure IV-1.

These data are based upon the detail cost and schedule estimate sheets, for each experiment, found in Appendix B of this report.

These data are presented by experiment and portray a cost range from an estimated minimum to a maximum. Since the quantity of hardware varies among experiments, the cost should be related to the hardware required per the detailed estimate. Estimates have been influenced by the confidence level of the organization(s) contacted to translate the experiment definition into cost information. Many experiments are well defined and hardware available, while other experiments, not as well defined, require a degree of conjecture; therefore, the cost "range" allows for this adjustment.

The schedule spans reflect quarters necessary from the order to delivery of the flight article(s) for each experiment. For reasons stated above a schedule minimum and maximum time is presented.

Various NASA agencies, university personnel, and manufacturers were contacted in gathering these data. In some cases it was determined that similar items are (1) in use on an existing satellite, (2) about to be launched on a satellite within the next six months, (3) in the development stage for a funded program, or (4) undergoing a design and feasibility study. One example of existing hardware is the Laser Altimeter built by Control Data Corporation and scheduled to be placed in an Earth Orbit during the spring of 1969. At the other extreme, the Gravity Gradiometer Experiment is still undergoing a feasibility study to determine the technique to be used.

Experiment	Cost Range $ in Millions	Quarters											
		1	2	3	4	1	2	3	4	-1	2	3	4
		△ Order Placed							Min		Max		
Metric and Stellar Camera	5.5 - 7.7												
Panoramic Camera	9.1 - 12.7												
Laser Altimeter	0.261 - 0.313												
Radar Altimeter	0.825 - 1.285												
Geochemistry Experiments	8.8 - 12.3												
Mass Spectrometer	1.2 - 1.5												
IR Mapping Radiometer	1.0 - 1.4												
IR Spectrometer	5.0 - 7.0												
Microwave Radiometer	1.85 - 5.2												
RF Transponder	0.27 - 0.33												
Bistatic Radar	0.9 - 1.3												
Photoelectric Photometer	2.0 - 2.9												
RF Reflectance	0.5 - 0.6												
Electromagnetic Measurements	2.6 - 3.1												
Radio Noise Survey	0.2 - 0.3												
Radar Imager	6.3 - 8.9												
Meteoroids	1.2 - 1.6												
Solar Wind Foil	0.5 - 0.7												
DC Magnetometer	0.4 - 0.5												
AC Magnetometer	0.35 - 0.46												
Plasma Probe	2.0 - 2.4												
Electric Field	2.5 - 3.1												
Low Energy Particles	1.2 - 1.7												
Solar Energetic Particles	1.0 - 1.5												
UV Spectrometer	1.6 - 1.9												
Laser Retroreflector	0.4 - 0.6												
Gravity Gradiometer	2.5 - 3.6												
Total	59.956 - 84.338												

Figure IV-1 Experiment Cost and Schedule Estimate Summary

V MISSION CONSTRAINTS

V.1 Lunar Coverage - C&SM and Subsatellite - The basic lunar landing mission is not compatible with total lunar surface survey by C&SM mounted sensors. The surface coverage from a lunar landing orbit with up to 2000 pounds of payload capability and a maximum time of 3 days is a very small fraction of the total. Therefore, subsatellite systems should be considered to enhance the coverage through high inclinations and long stay times or the extended viewing of certain lunar sites should be elevated in value above the total surface coverage objectives stated at the start of this program.

The coverage of the lunar surface is discussed in terms of general area coverage, specific orbit accessibility and the impact on experiment utilization and specific target coverage. Subsatellite requirements and capabilities are discussed when they vary from the basic C&SM.

The portion of the lunar surface that is available for viewing is presented in Figure V.1-1 as a function of orbit inclination and lunar stay time. A polar orbit with a stay time of about 14 days is required for complete surface coverage. The coverage areas for the 10.1 and 14.3 day total time missions (1 day and 3 day stay time, respectively) are noted along with the 14 day stay time data. An orbit altitude of 60 n m is assumed.

Select the 15 degree inclination that is required for the Abulfeda landing and establish the lighting such that it is favorable to the landing mission. The lunar surface accessibility for the 10.1 day (1 day stay) and 14.3 day (3 day stay) missions are shown in Figure V.1-2. The effect of lighting on experiment utilization is shown. Experiment utilization is shown in Figure V.1-3. No restrictions have been assumed on the available time due to crew duties or spacecraft pointing capabilities and the data represent a maximum. Very low percentages of the lunar surface are available to the C&SM on a nominal landing mission.

To improve lunar surface coverage a subsatellite could be deployed at high inclinations. The size of the subsatellite is a function of the propulsion system requirements. The subsatellite propulsive requirements to achieve a high inclination from the initial C&SM orbit inclination can be defined from the turn requirements. Figure V.1-4 presents the minimum turn requirements to achieve a given orbit inclination from any C&SM orbit. An orbit altitude of 60 n m is assumed for the subsatellite data. The turn angle requirement can be converted to a velocity increment requirement as shown in

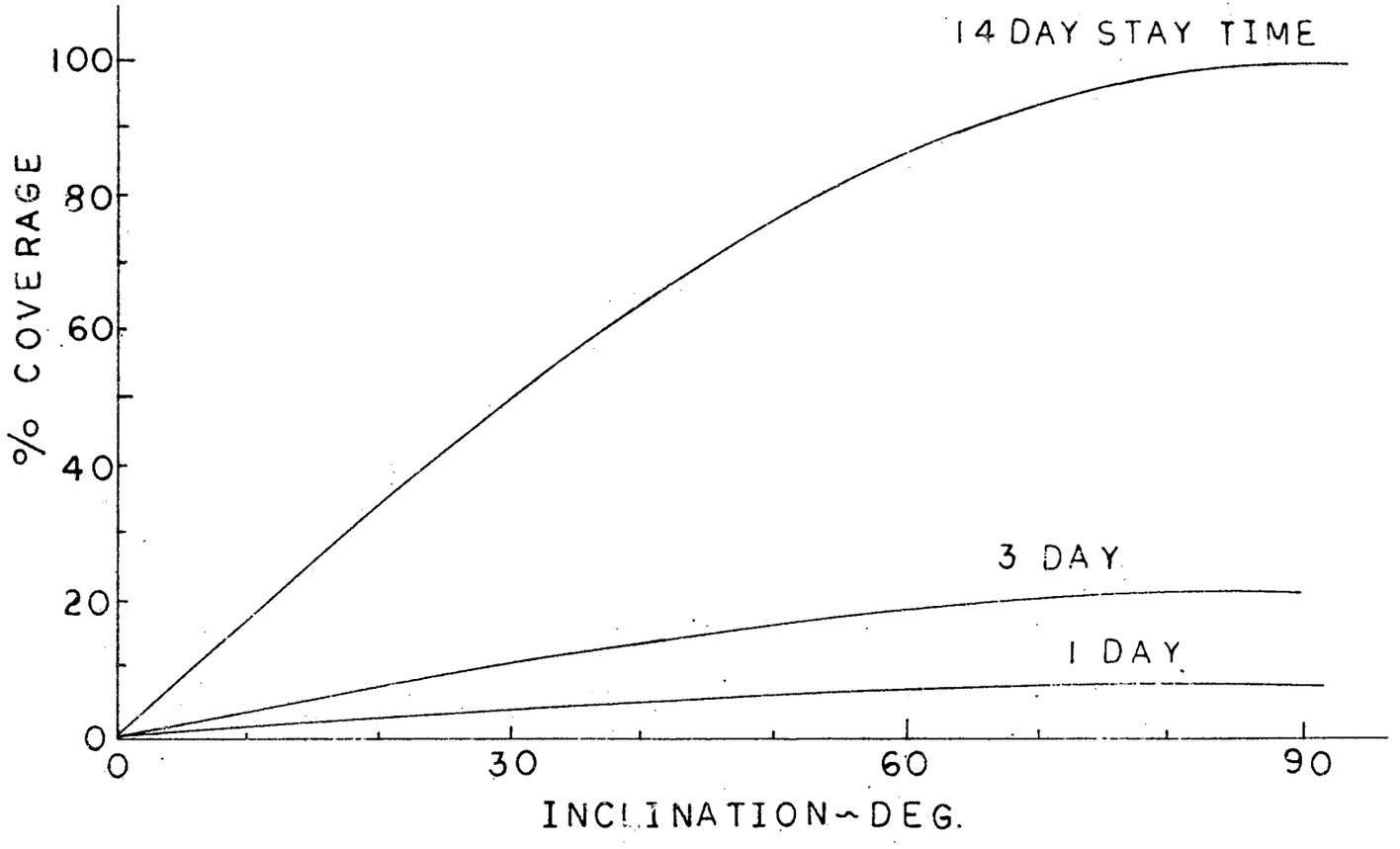


Figure V.1-1 Lunar Surface Coverage

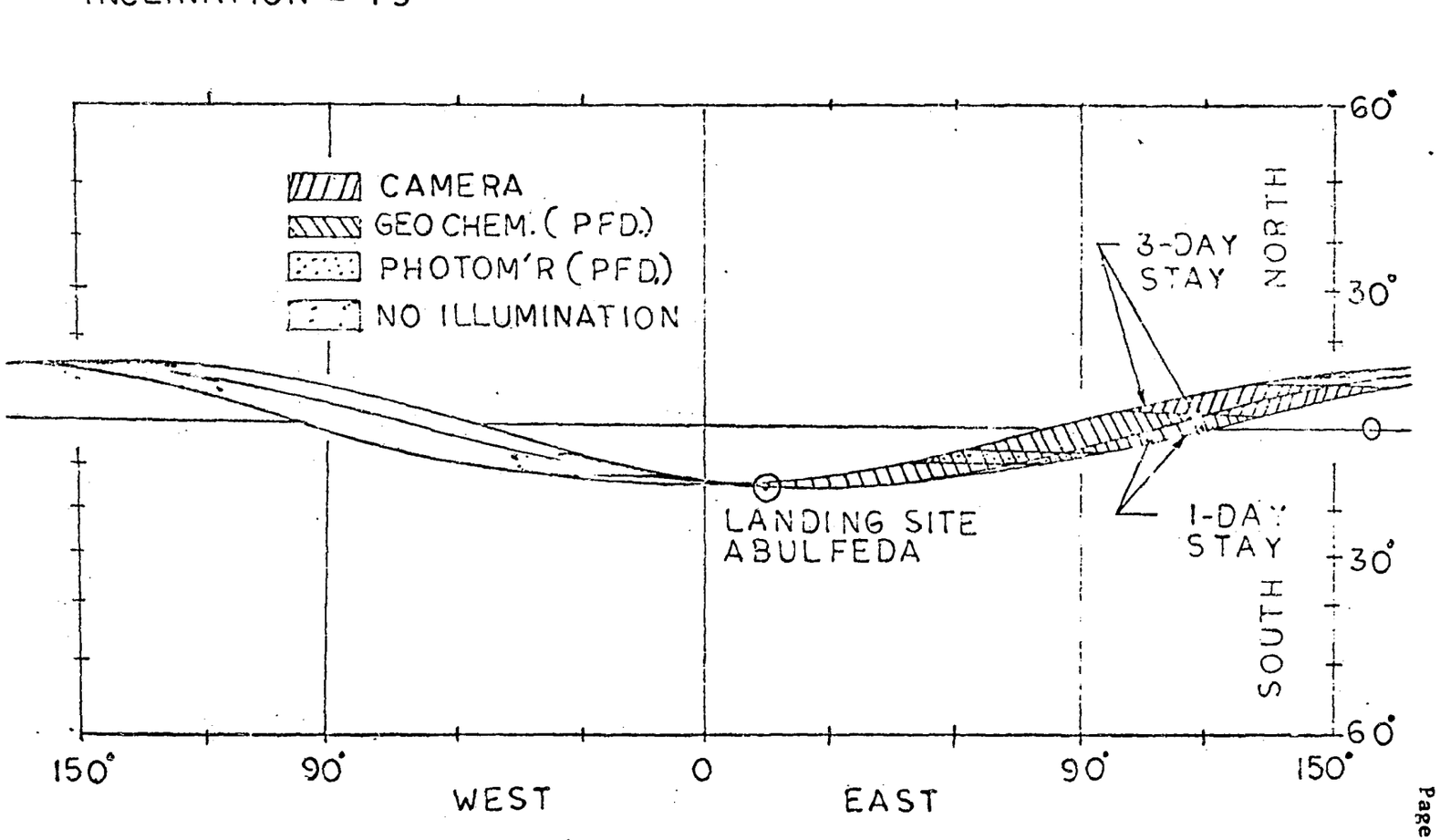
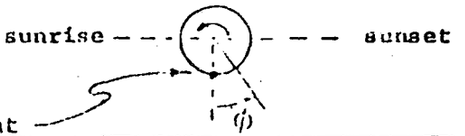


Figure V.1-2 Areas Accessible for Orbital Experiments

Inclination: 15°

Landing Site: Abulfeda



Experimental Equipment	Illumination (ϕ) Acceptable	Illumination (ϕ) Preferred	% Coverage 1.08 days Accept	1.08 days Pref	3 days Accept	3 days Pref
Metric Camera	+45° to +75° -45° to -75°		0.2		0.6	
Panoramic Camera	same		0.2		0.6	
Radar/Laser Altimeter	any ϕ		2.5		7.0	
Geochemistry	any ϕ	±45° light side	2.5	0.4	7.0	1.2
Ranging Xponder	N/A					
Solar Wind Foil	N/A					
IR Mapping Radiometer	±90° dark side		1.2		3.5	
Microwave Radiometer	any ϕ		2.5		7.0	
IR Spectrometer	±90° light side		1.2		3.5	
Photoelectric Photometer	±90° light side	±10° light side	1.2	0.1	3.5	0.2
Bistatic Radar	any ϕ		2.5		7.0	
U.V. Imager	any ϕ		2.5		7.0	
Radar Imager	any ϕ		2.5		7.0	

Figure V.1-3 Experiment Utilization

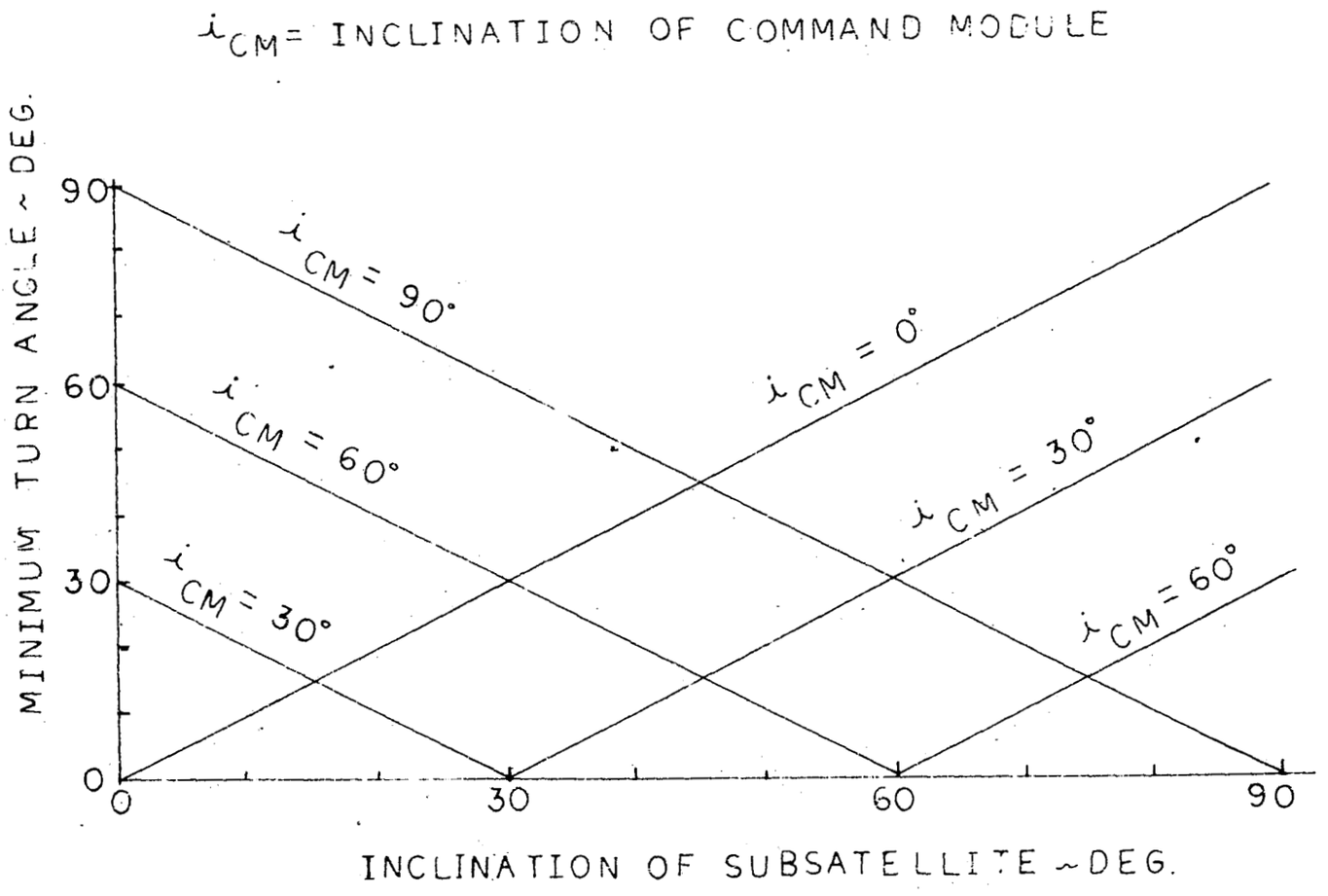


Figure V.1-4 Subsatellite Turn Analysis

Figure V.1-5. These data, in turn, can be presented in terms of the mass ratio required to develop a given velocity increment. The mass ratio is expressed as a precent of the subsatellite weight that must be expended in propellant to achieve a given turn in Figure V.1-6. Two specific impulse values are noted for comparison. The subsatellite mass shown does not include the ejection mechanism to deploy it from the C&SM.

The C&SM coverage picture for general surface characteristics has been noted. Subsatellites will improve the coverage characteristics of the system. Now, consider the coverage of specific targets or landing sites for multiple viewing and study. Figure V.1-7 shows the general coverage problem for a circular orbit at 60 n m. A maximum sensor viewing half angle, τ of 37 degrees is noted. Coverage of any chosen landing site is given in Figure V.1-8 as a function of the latitude of the landing site. A sensor viewing half angle of 37 degrees is assumed. The mission plan for this coverage is such that the inclination of the orbiting C&SM is only slightly (less than 2 degrees) higher than the latitude of the landing site. Also, the illumination angle at the time of lunar module landing ($\Phi = 70^{\circ}$-83°) is also approximately satisfactory for camera viewing ($\phi = 45^{\circ}$-75°). The illumination angle becomes smaller at the rate of 12.2 degrees/day due to surface rotation. Thus the landing site receives satisfactory illumination not only at lunar module landing but at all times thereafter for the stay times considered.

Coverage of the lunar landing site starts when the C&SM passes approximately over the landing site and continues until the landing site passes out of the field of view of the sensor as the orbit moves westward. The C&SM orbit is at an inclination slightly greater than the latitude of the landing site. The number of orbits passing within sensor coverage of the landing site (based on 74 degrees field of view) ranges from approximately 49 orbits for a site at 10 degrees north or south latitude to some 39 orbits at 45 degrees north or south latitude. Since, however, the 3 day surface stay time of the lunar module limits the number of orbits to some 36 orbits, it is evident that the landing site is within view of the C&SM-mounted camera part of each orbit for entire stay time. Figure V.1-8 also relates the number of orbits to the amount of viewing time of the camera. For the 3 day surface stay time (with 36 orbits), the amount of viewing time varies with site latitude and inclination of the orbit. For all landing sites the total viewing time is about 1 hour for a 3 day stay time.

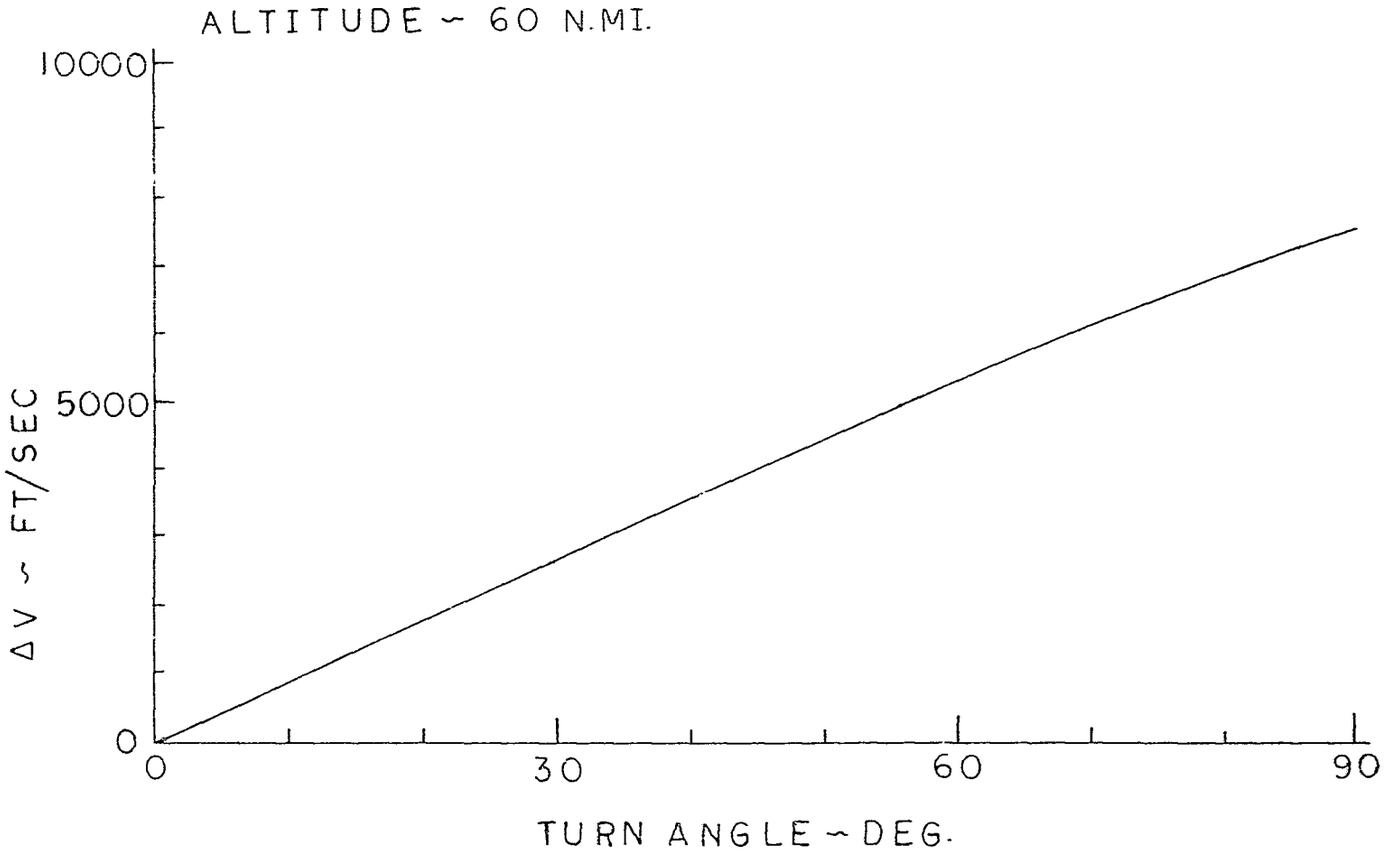


Figure V.1-5 Subsatellite Analysis, Δ V

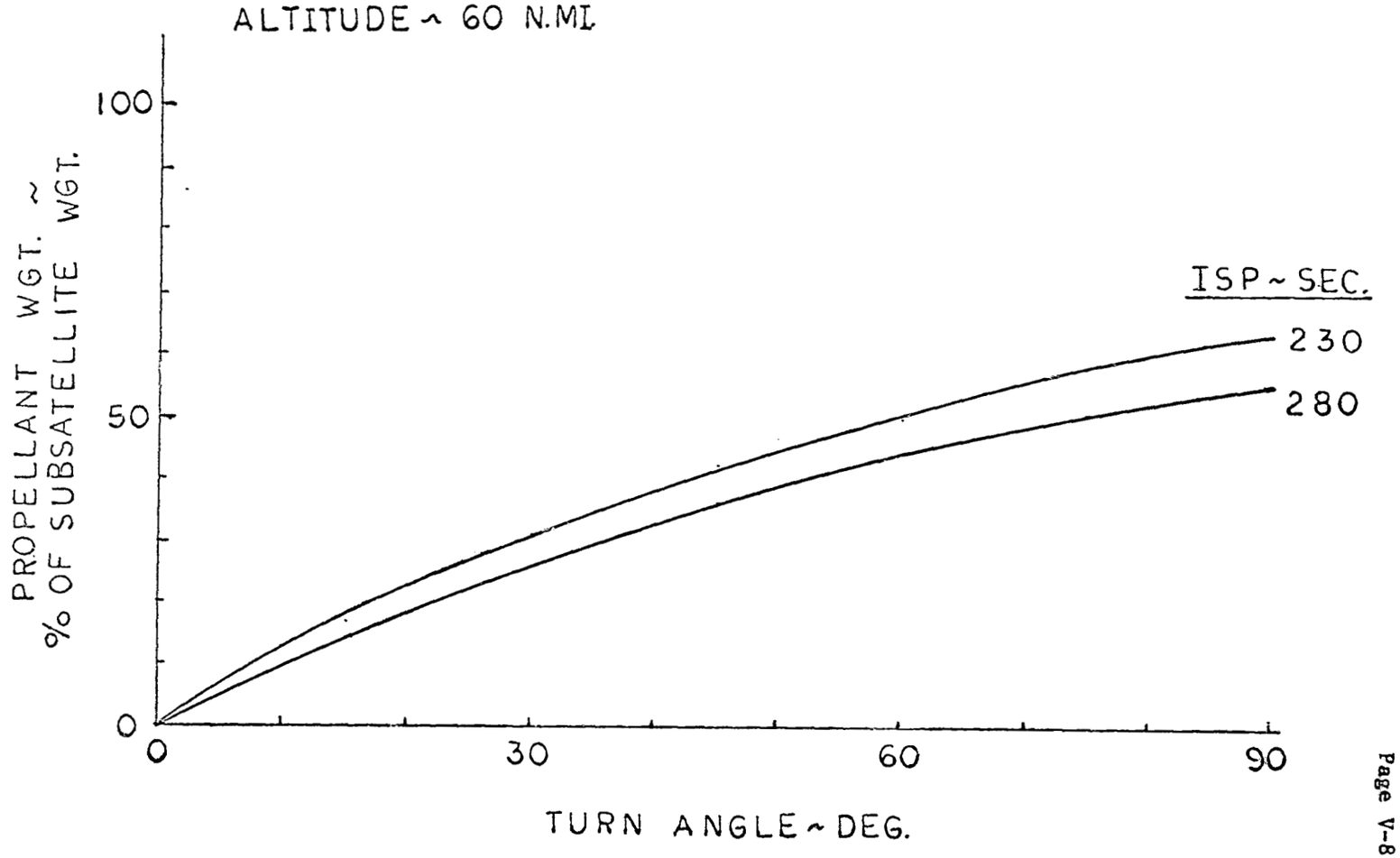
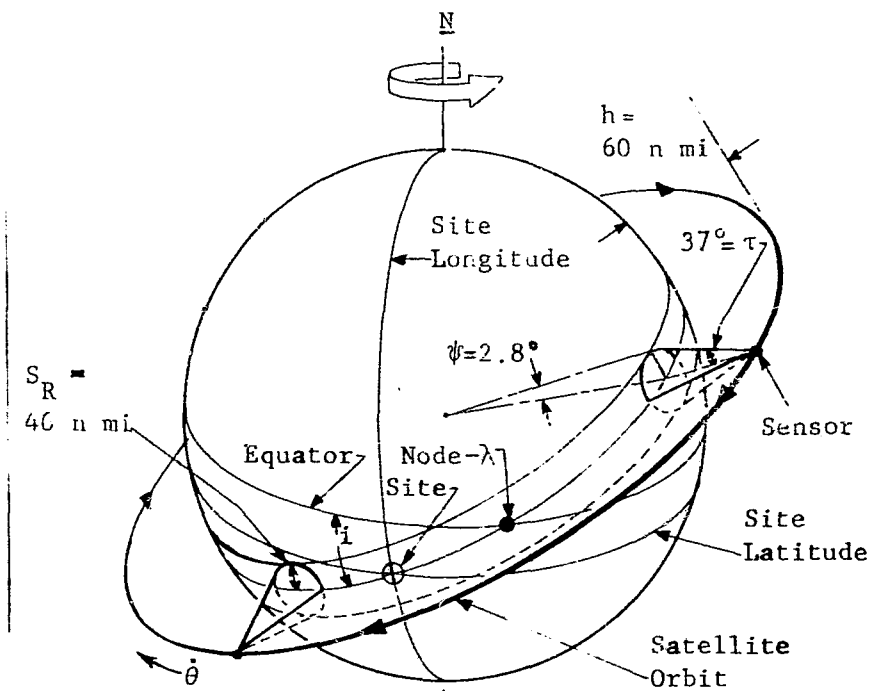


Figure V.1-6 Subsatellite Analysis, Propellant Required



C Circumference of moon ≅ 5900 n mi

T Time of satellite orbit ≅ 2 hr

$\dot{\theta}$ Velocity of satellite ≅ 180°/hr

i Inclination of orbit

τ ½ angle of sensor field of view ≅ 37°

ψ Equivalent arc on moon surface of sensor coverage ≅ 2.8°, ~46 n mi

λ Apparent motion of node to west = 1.1°/orbit, ~18.0 n mi/orbit

Figure V.1-7 Sensor Coverage of Lunar Site

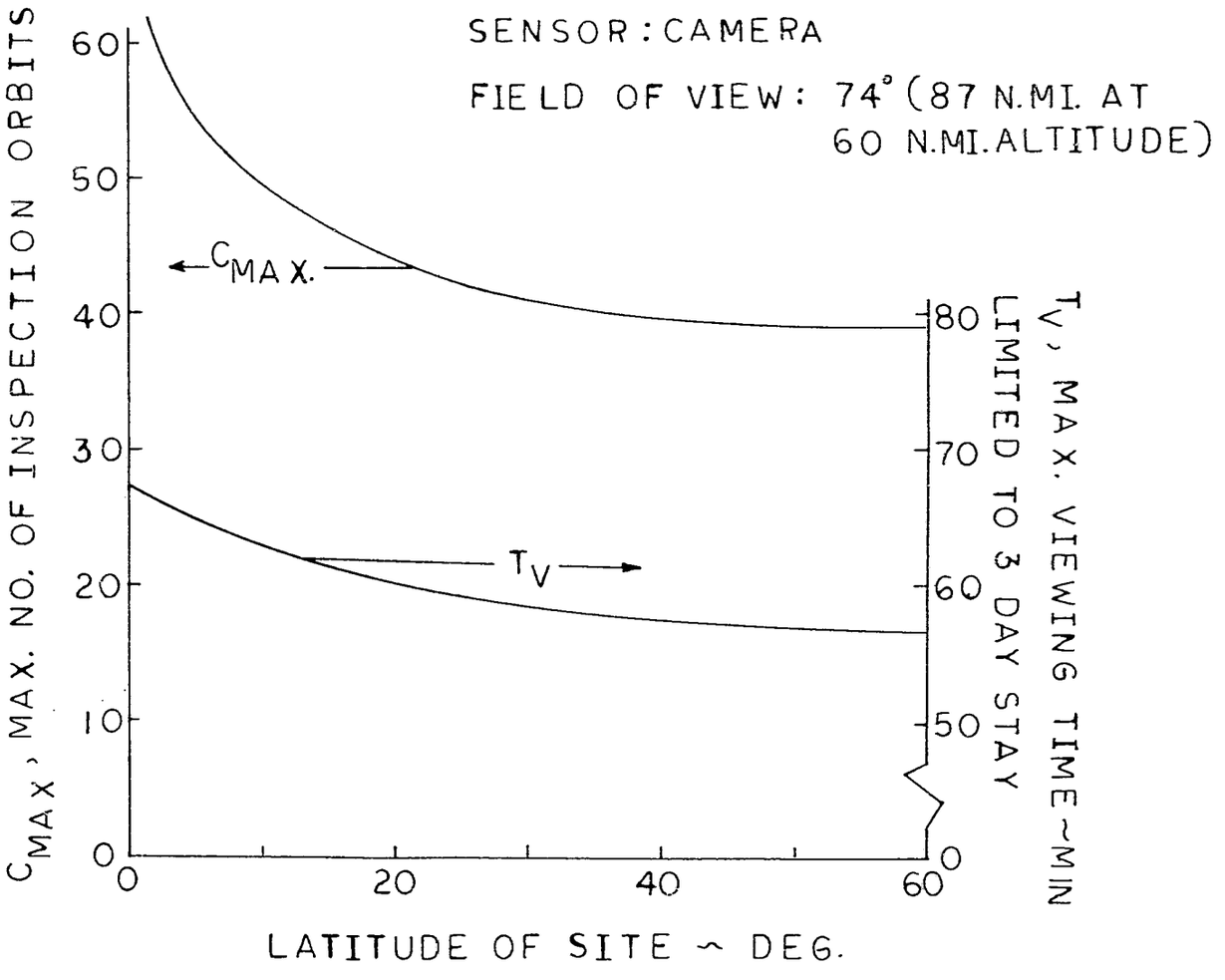


Figure V.1-8 C&SM Camera Coverage of Landing Site

The previous coverage numbers assumed that the initial pass is directly over the landing site or target. In this case only one half of the ground viewing circle is available for the coverage sweep with time since the target starts in the middle of the circle and will not be viewed once it has moved beyond the circle or returned to the center where nominal LM pickup is achieved. Now consider the case where the target is started anywhere within the coverage circle or as a maximum on the edge of the circle. This condition tends to maximize the accessibility of the target. The site longitude and latitude will fall within view of a given orbit inclination and nodal position if the distance between the site and the orbit (angle φ) is less than the radius of the viewing circle (Ψ). The smallest angle, φ, between any orbit and any site is computed as

$$\sin \varphi = \sin \nu \cos i - \cos \nu \sin i \sin \Delta \Omega$$

where ν is the site latitude, i is the orbit inclination, and $\Delta \Omega$ is the difference between the orbit node, λ_o and the site longitude, λ ($\Delta \Omega = |\lambda_o - \lambda|$). If $|\varphi| < \Psi$, the arc of the lunar surface cut off by γ includes the target site. Ψ is computed as

$$\psi = \cos^{-1}\left[\frac{R_M + h}{R_M} \sin \gamma\right] - \gamma$$

where R_M is the radius of the moon and h is the altitude of the orbiting vehicle.

Figure V.1-9 gives the maximum number of orbits with sensor coverage as a function of inclination and site latitude. For maximum coverage, the inclination should be within several degrees of the site latitude. Figure V.1-9 is based on a sensor field of view of 74° and an orbiting vehicle altitude of 60 n m.

When the initial orbit is eastward from the site by exactly Ψ degrees ($\varphi = \Psi$), the coverage is a maximum and Figure V.1-9 can be used. However, when this is not the case, some modifications need to be made. For instance, when the initial orbit is over the site, $\varphi = 0$ and the coverage C is approximately one-half of C_{MAX}. In general,

$$C = \frac{C_{MAX}}{2}\left[1 + \frac{\varphi}{\Psi}\right]$$

where C is the coverage in orbits, and φ is negative if the initial orbit is westward from the site.

The viewing time for the orbiting vehicle can be obtained from Figure V.1-10 where the parameter C_{MAX} is taken from Figure V.1-9 and φ is computed as above. If the orbiting vehicle is a sub-

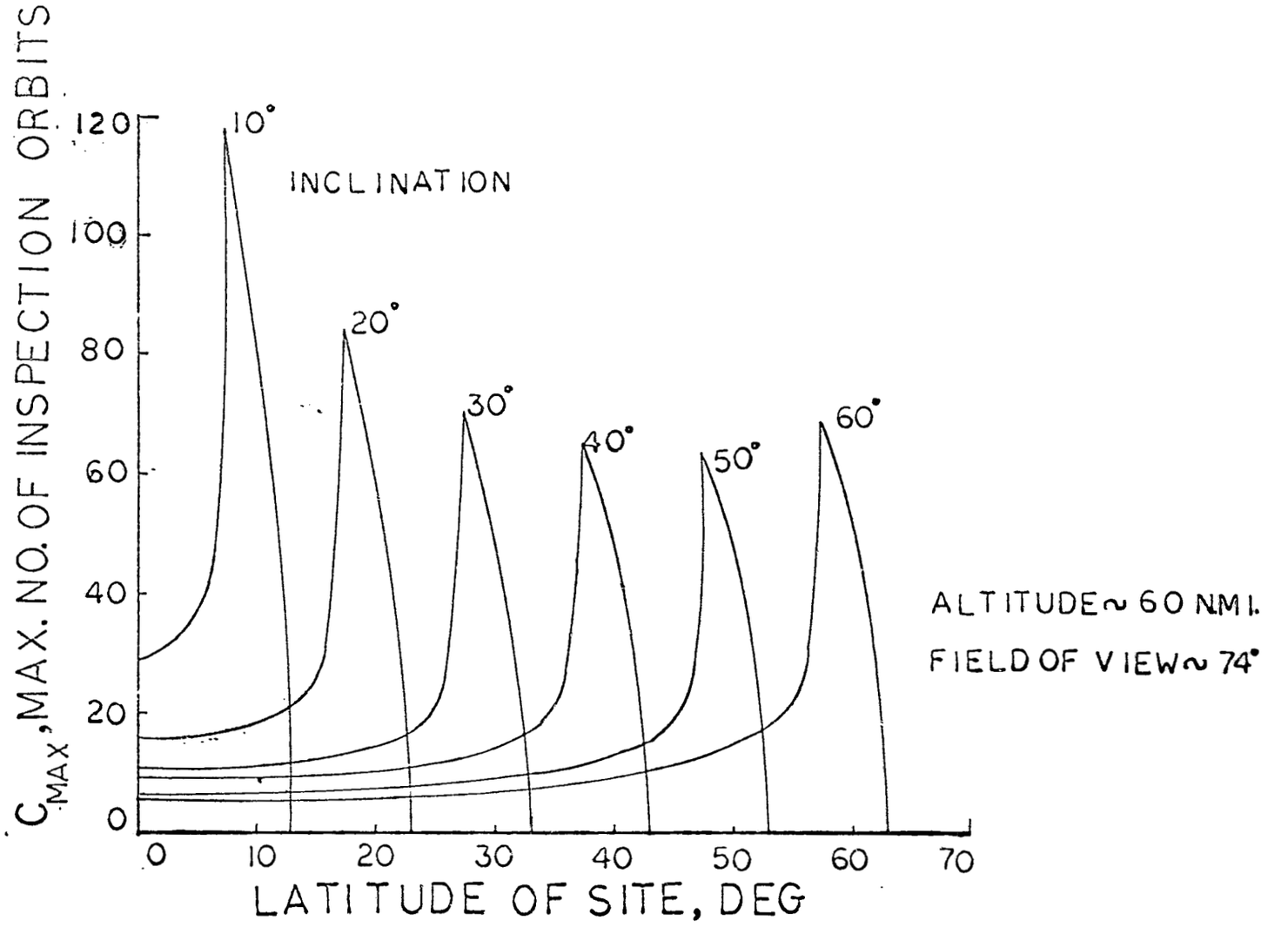


Figure V.1-9 Extended Sensor Coverage

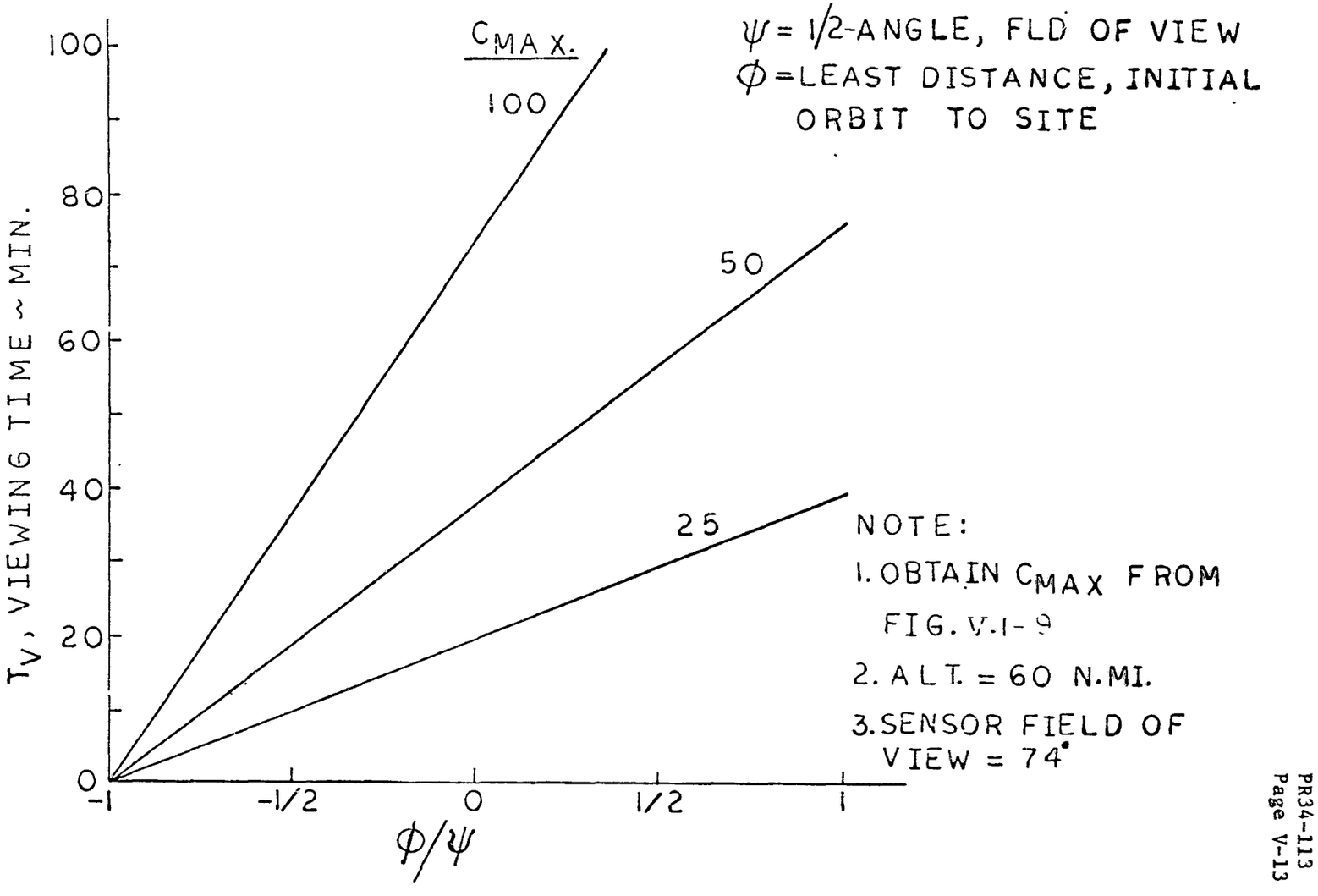


Figure V.1-10 Extended Sensor Viewing Time

satellite, there is no stay time limitation on viewing time as there would be for the C&SM. Also, after approximately 14 days, the initial conditions are repeated but from a southerly direction if the original orbits had been from a northerly direction, and conversely.

V.2 Payload/Delta V Considerations - The payload capability is dependent upon the selection of a specific landing site and mission date. The maximum capability is 2800 pounds. This capability is reduced to 2300 pounds for a landing at Abulfeda (inclination about 15 degrees). The tradeoffs between payload and the mission variables are discussed in the following paragraphs.

The payload is a function of the velocity increments required for lunar orbit insertion (LOI), technique of LM pickup and trans-earth insertion (TEI). These velocity increments, in turn, are a function of the translunar trip time, and lunar orbit nodal position and inclination required for the mission. For a mission where the principal objective is a lunar landing the nodal position and inclination are established by the landing site location and technique of LM pickup. For any case the velocity increment is minimized by utilizing a three impulse technique that employs a highly elliptic initial orbit for plane changes before achieving the final circular orbit.

Consider the landing site location and its effect on the mission. Twenty landing sites have been identified and are shown in Figure V.2-1. These sites are:

1.	Littrow	11.	Schröter's Valley
2.	Hyginus	12.	Aristarchus
3.	Tycho	13.	Dionysius
4.	Gassendi	14.	Sea of Alexander
5.	Censorinus	15.	Tobias Mayer
6.	Abulfeda	16.	Copernicus C
7.	Rima Bode	17.	Marius Hills
8.	Mösting C	18.	Hadley Rille
9.	Fra Mauro	19.	Alphonsus
10.	Hipparchus	20.	Harbinger Mountains

The first four sites have been designated by NASA-MSC as those of primary interest. Data for Abulfeda are included as a comparison baseline to take advantage of the extensive site analysis already completed for this site. For specific trip times, a nodal position can be defined that will yield a minimum LOI velocity increment. Similarly the maximum LOI velocity increment is associated with a specific nodal position. The positions of the descending nodes for these limiting cases and a three impulse insertion technique are also shown in Figure V.2-1. The surface traces shown are representative of 60 n m orbits at an inclination of 30 deg. Data are shown in Figure V.2-1 for total translunar trip times of 84 hrs, 96 hrs and 108 hrs. This total time includes the time in the arrival ellipse that is a part of the three impulse technique and corresponds to actual trip times of 66 hrs, 72 hrs and

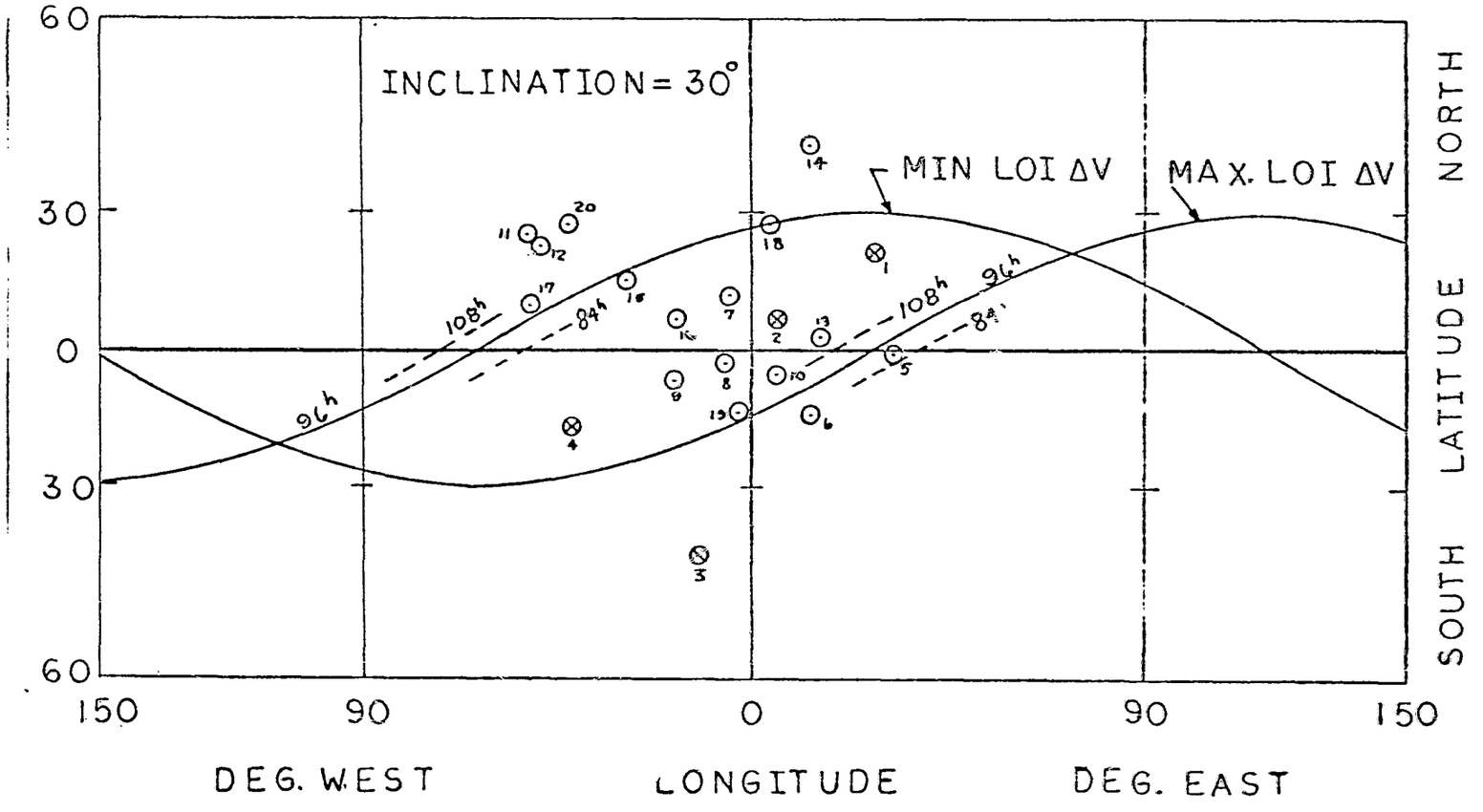


Figure V.2-1 Site Analysis

78 hrs respectively. The nodal positions will not change with inclination and can be either ascending or descending nodes. The corresponding ascending node picture can be achieved by a rotation about the lunar equator. These data do not reflect a requirement for free return trajectories since the study was not constrained in this way.

The magnitude of the LOI velocity increment (three impulse technique) is presented in Figure V.2-2 as a function of inclination. The data are presented for nodal positions yielding minimum and maximum conditions and it is apparent that the nodal position is a significant factor in the energy requirements at high inclinations. Trip time is also a factor with increasing trip times resulting in reduced energy requirements.

The transearth insertion (TEI) velocity increments will be similar to the LOI increments. For stay times of about three days the maximum TEI increment will occur with the maximum LOI increment and the minimums will occur together. The orbiting weight equivalent to a change in the velocity increment can be computed on the basis of about seven pounds per fps for LOI and three pounds per fps for TEI. Specific site analysis, then, will define the velocity requirements and the attendant payload capability in lunar orbit. Two approaches that are different in the LM pickup technique have been investigated in determining the payload capability.

The first method considered consists of picking an initial inclination that passes through the lunar site of landing and utilizing a plane change for LM pickup that increases with stay time. The mission profile would show a wait of 18 hrs after LOI for checkout and landing preparations. At the end of the wait the site would pass under the orbit plane and the LM would descend and land. A stay time of three days is utilized and the lunar rotation has moved the landing site out of the orbit plane. A final orbit inclination is determined that passes through the landing site and intersects the orbit plane at a minimum turn angle. This is the plane change required for LM pickup. Also, the initial and final orbit inclinations are determined (graphically in most cases) as well as the initial and final orbit nodal positions.

The plane change maneuver is presented in Figure V.2-3 as a function of stay time and CM orbit inclination for a lunar orbit altitude of 60 n.m. Two specific stay times are noted to provide a comparison between the requirements for the 10.1 day and the 14.3 day mission. The landing site is assumed to be Abulfeda.

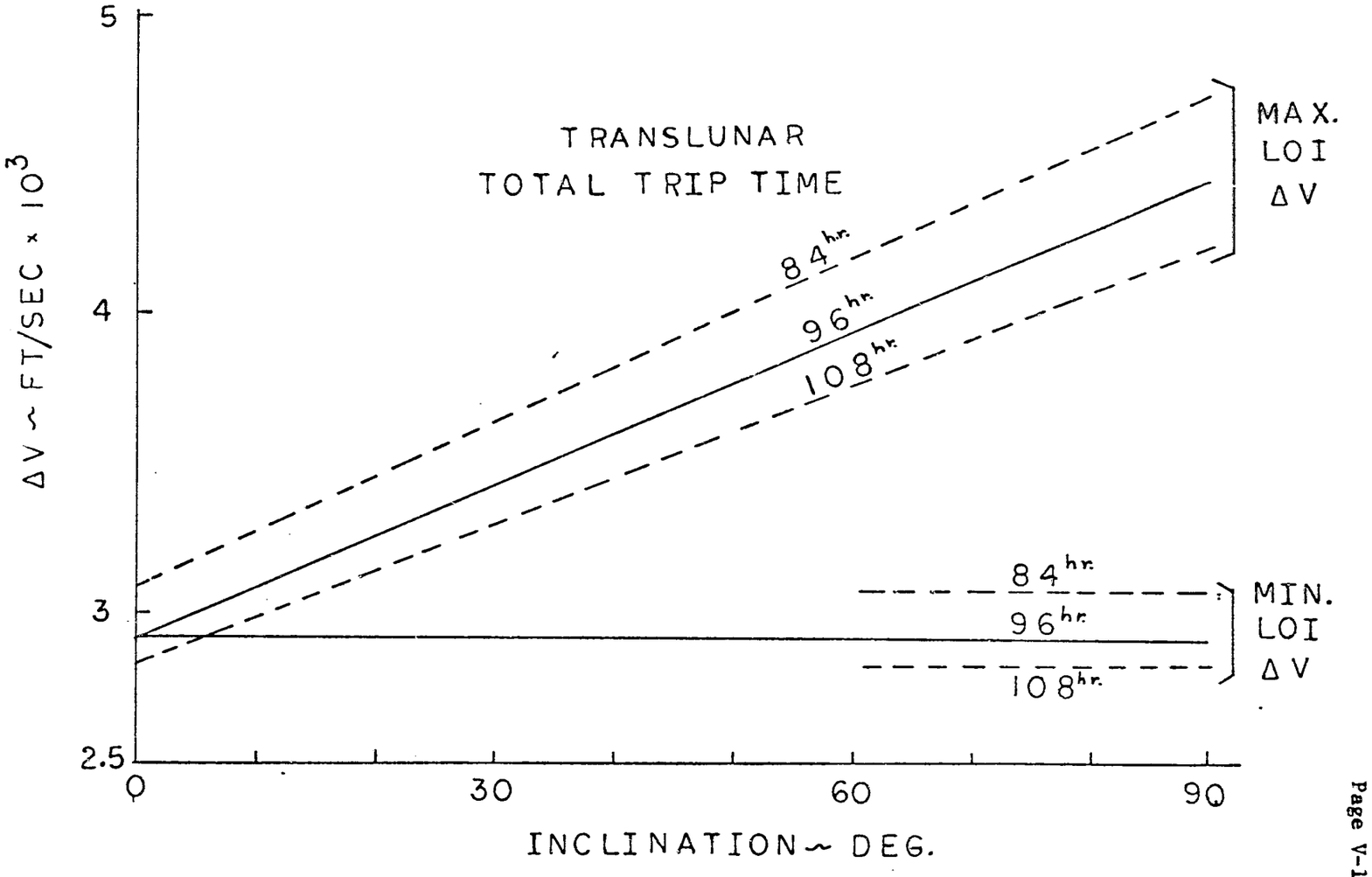


Figure V.2-2 Site Analysis ~ ΔV

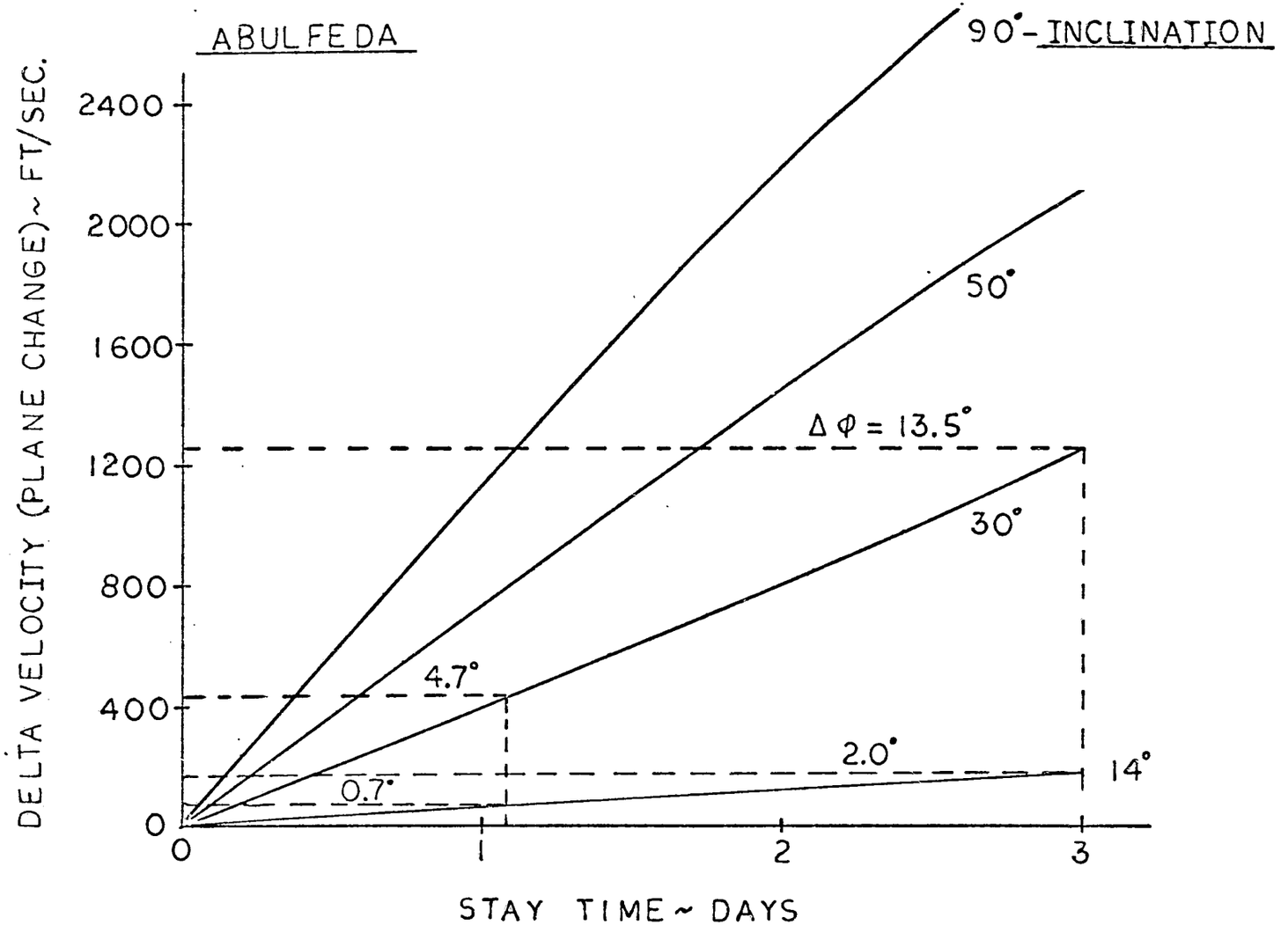
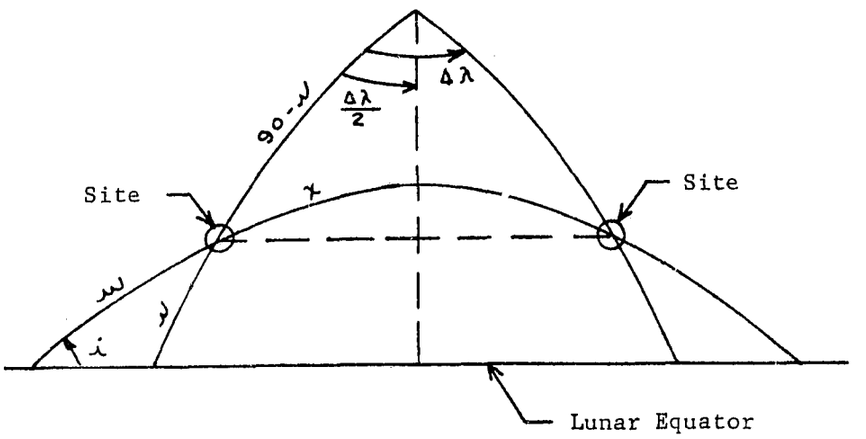


Figure V.2-3 Plane Change Requirements

The LOI and TEI velocity increments are evaluated from Reference 1 which assumes that the three impulse technique is used. Provision is made for the pickup plane change and adjustment is made for midcourse corrections and a LM rescue mode of operation. The final velocity requirements set the propellant needed and the resulting payload available for scientific experiments. For a landing at Abulfeda this method yields a payload of about 1700 pounds.

This approach allows for any orbit abort within its nominal performance. The plane change increases to a maximum at the nominal departure time; therefore, early LM pickup requires less energy. However, an abort results in a shorter mission time and nodal positions displaced from the nominal case. Depending on the choice of landing sites the new departure node could result in either higher or lower TEI velocity increments. The capability of any orbit abort, then, will require a payload penalty for some landing sites or possibly the provision for waiting in lunar orbit for the nominal departure time.

The second method is faster computationally than the first and is generally less restrictive in terms of payload. In this case, an orbit is determined that passes over the landing site both initially and at the nominal LM pickup time. No pickup plane change is required for the nominal flight plan. However, a small plane change is required for any orbit abort. The initial and final orbit inclinations are the same and slightly greater than the latitude of the landing site. This condition is noted in the following sketch.



The initial and final nodal positions can be determined from simple expressions that are dependent only on the longitude of the landing site and the lunar surface stay times. The equations for the inclination required are:

$$\text{Sin } i = \frac{\text{Sin } \nu}{\text{Sin } w} \qquad \text{(inclination)}$$

where $w = (90\text{-}x)$ and

$$\text{Sin } x = \text{Cos } \nu \ \text{Sin } \frac{\Delta\lambda}{2}$$

The initial node can be determined from:

$$\Omega_o = \text{Long}_{site} - 90 + \frac{\Delta\lambda}{2} \qquad \text{(uncorrected for wait time after LOI)}$$

The final node:

$$\Omega_f = \text{Long}_{site} - 90 - \frac{\Delta\lambda}{2} \qquad \text{(uncorrected for wait before TEI)}$$

Knowing the inclination and initial and final nodal positions, the velocity increments for LOI and TEI can be determined from the parametric data in Reference 1. The payload capability for a specific site can be defined.

This approach is more consistent with the data and assumptions of the existing Abulfeda site analysis and is the technique used for the investigations of this report. Any orbital abort requires a small plane change which is assumed to be achieved by utilizing the LM rescue mode propellant in the C&SM. The abort will shift the departure nodes and will affect the TEI velocity increment. Depending on the site chosen, the TEI velocity increment will increase or decrease and may require an additional payload penalty. This condition can be alleviated by waiting in the lunar orbit for the nominal departure time.

Based on the second method, specific payload penalty curves have been determined and are presented in Figure V.2-4. A stay time of three days is assumed and nominal LOI and TEI velocity increments are utilized. The node positions allow for the initial wait between LOI and landing and the final wait between LM pick-up and TEI. Any site along the equator is accessible without penalty. As the site latitude is increased north or south, the cost in terms of payload penalty increases. A basic capability of 2800 pounds of scientific payload is assumed for equatorial orbits.

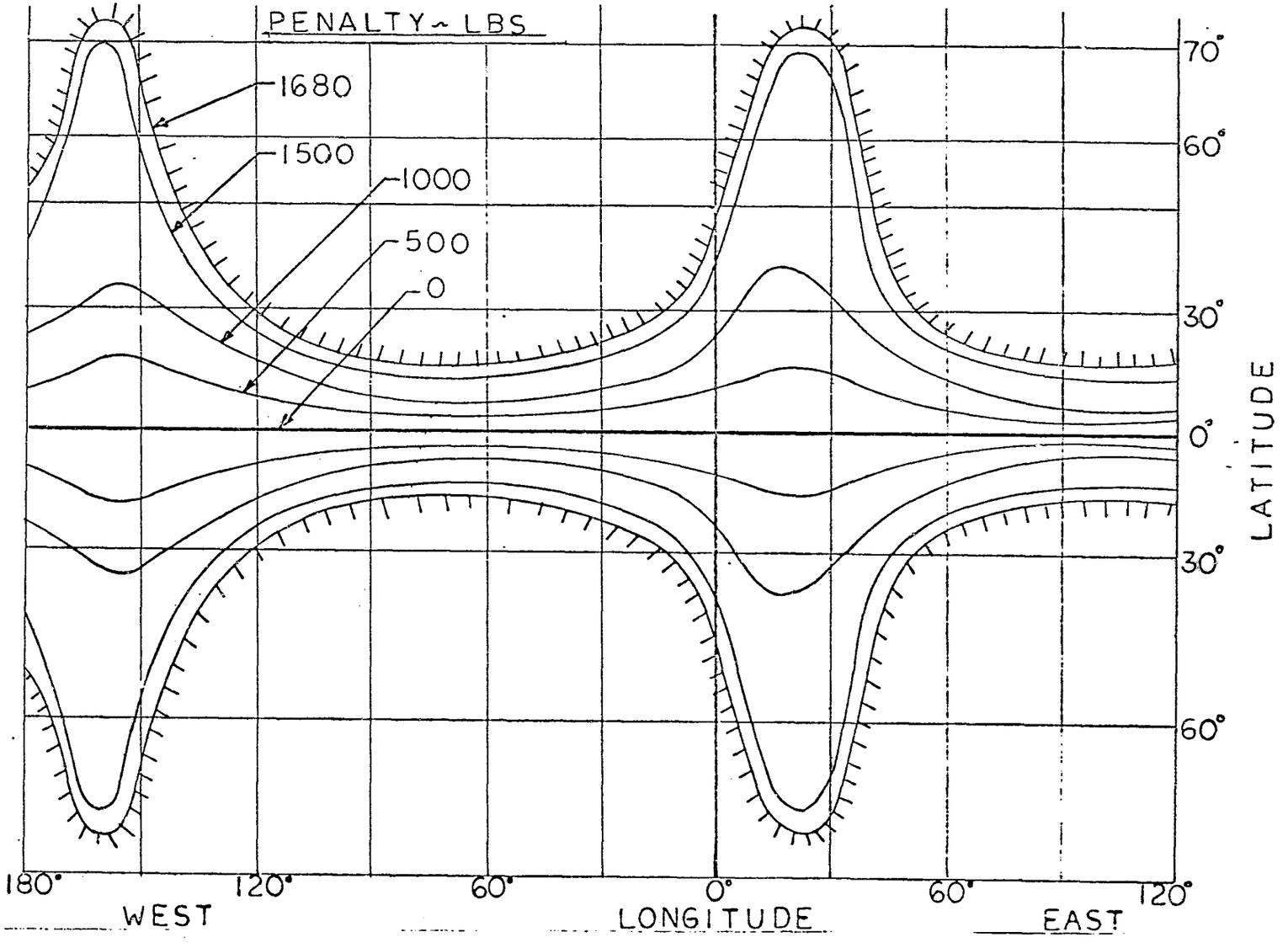


Figure V.2-4 Site Analysis ~ Payload

The payload penalty is subtracted from this value. At Abulfeda a payload penalty of 500 pounds is noted resulting in a payload capability of 2300 pounds. The maximum payload penalty of 1680 pounds reflects the maximum propulsive capability of the C&SM based on the weight as presented in Table V.2-1. Areas outside of the 1680 pound envelope are not accessible under these constraints. The data are cross-plotted in Figure V.2-5 to indicate the effect of inclination on payload penalty for the most favorable landing site location and least favorable landing site longitudinal location. It should be noted that the LM pickup technique chosen establishes the inclination for a specific site at an angle slightly larger than the value of latitude at the site. Lunar orbit inclination is not freely variable in this case. Should other inclinations be desired the landing site must be changed or the LM pickup technique must be changed resulting in new LOI and TEI values.

Specific payload data are presented in Table V.2-1 for an Abulfeda landing. The data assume a heavy LM weight to account for the extended mission time (three day surface stay). This assumption requires the C&SM to bring the LM to an 8 n m orbit for descent and landing and necessitates a C&SM transfer from the initial 60 n m circular orbit to the 8 n m circular orbit and back to the 60 n m circular orbit. Two sets of numbers are shown to compare MMC and NAR data. The only significant differences are in the LOI and TEI velocity increments which can be attributed to differences in the accuracy of reading parametric data (MMC) and specific case computations (NAR). NASA-MSC has directed the use of the NAR payload data.

Reference:

1. "Lunar Orbital Survey Missions," Final Report. Volumes I, II, and III, Lockheed Missiles and Space Company, 16 January 1967.

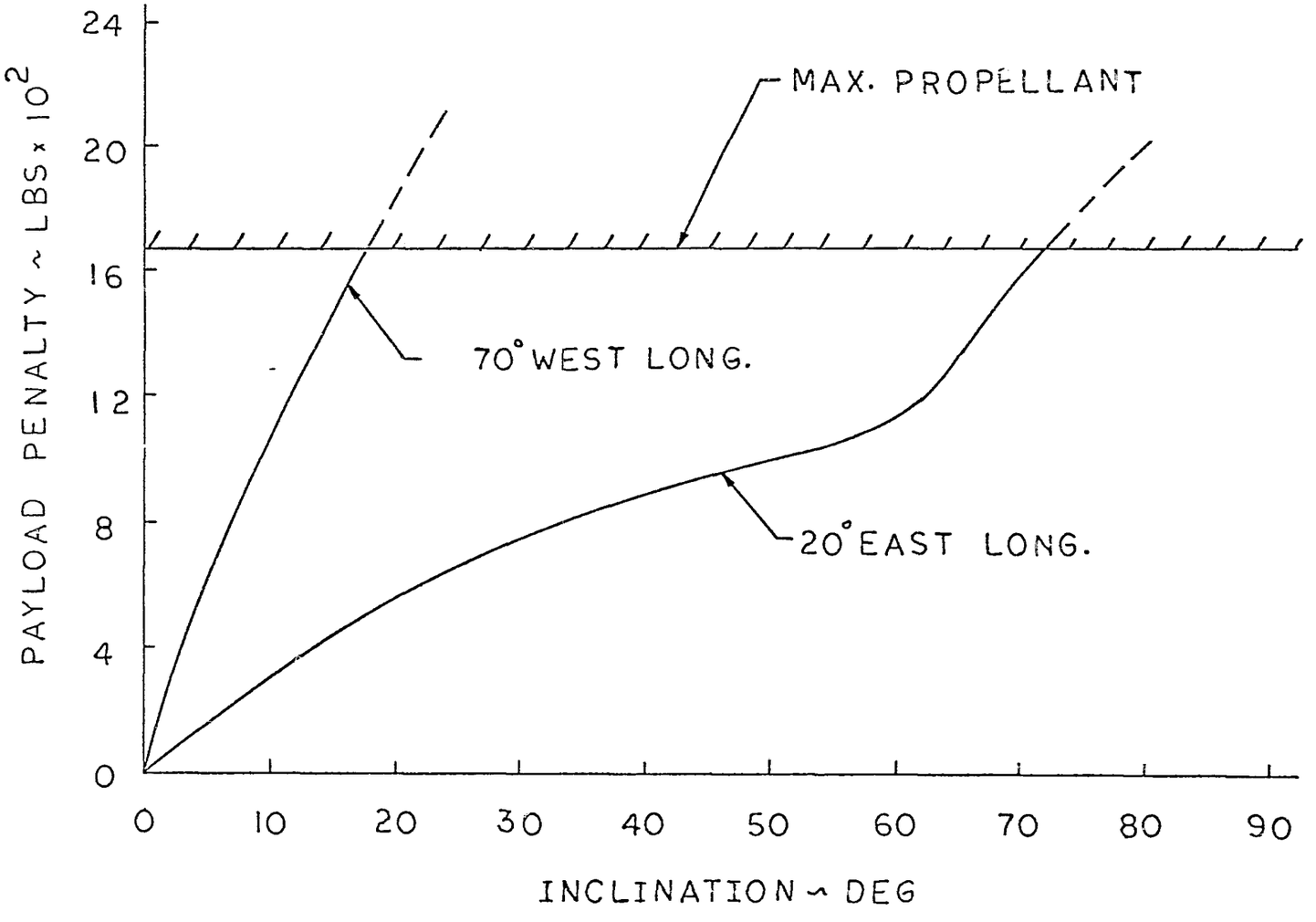


Figure V.2-3 Inclination Effects

Table V.2-1 Payload Comparison Abulfeda

	MMC	NAR
i ~ (DEG)	14.85	15.00
$\Delta V_{(LOI)}$ ~ FPS	2903	2871
$\Delta V_{(TEI)}$ ~ FPS	3625	3456
$W_{(LM)}$ ~ LB	33805	33805
$W_{(CSM)}$ ~ LB	25000	25000
$W_{(INJ)}$ ~ LB	99165	99165
ISP ~ SEC	313	313
$\Delta V_{(HVY-LM)}$ ~ FPS	290	290
$W_{(PAYLOAD)}$ ~ LB	1900	2300

V.? CREW SYSTEMS

The impact of one-man CM operations in orbit during phase 11.0, Lunar Surface stay, was investigated for both the 10.1 day and the 14.3 day missions. Very limited crew time, only 2-3 hours would be available during the 10.1 day mission for science experiments. The 14.3 day mission provides a nominal 72 hours in lunar orbit and permits periodic cycles for S/C systems management and IM support tasks to provide 12-15 hours crew time for identified LOS experiment tasks. Candidate LOS experiments require crew participation that involves primarily experiment switching tasks and EVA data retrieval at the end of the mission. Manual control of all experiment on-off cycling is not recommended, and automated experiment control to assist the crew is necessary. The recommended S/C attitude is "streamlined" (x-axis parallel to velocity vector) with the C&SM -Zaxis at local vertical. This choice permits crew visibility of the ground track, but requires canted orientation and mounting of Sector I LOS experiments. The most feasible time for EVA data retrieval is during trans-earth coast. The EMU/PLSS, already provided for the Lunar Surface mission, should be reused to minimize overall mission weight and stowage requirements. Since primary mission contingencies may cause cancellation of EVA, alternate data retrieval techniques should be investigated.

V.3-1 Crew IVA - The Lunar Orbital Science (LOS) mission superimposes selected experiment management tasks on the reference Apollo Mission G (lunar landing) crew operations. Throughout this study the primary objective of a successful lunar landing mission was considered for new landing sites and experiments. The LOS crew operations must complement the lunar landing mission while satisfying the science experiments requirements. Crew IVA is restricted for the science experiments to one-man CM operations during lunar orbit while the IM and two crewmen are on the lunar surface.

During one-man CM operations in lunar orbit, the CM pilot must manage the C&SM, support the IM, and support the LOS experiments while continuing the established crew work/rest cycle at Lunar Orbit Insertion (LOI). Critical mission events just before IM-undocking and after the lunar orbital rendezvous period preclude major adjustments to the required S/C tasks and crew work/rest cycle to accommodate LOS operations.

Crew experiment tasks are dependent upon the scientific experiment grouping considered. For the LOS mission, the experiment groupings fall into two categories: C&SM mounted, and sub-satellite mounted. After a sub-satellite is placed in lunar orbit (post-LOI), the CM pilot would have no active control over it. Any required sub-

satellite functions would be controlled by MSFN/DSIF commands. Therefore, emphasis in the study was placed on C&SM mounted experiments. In order to insure the greatest experiment grouping flexibility, crew/experiment operations and interfaces were investigated for all candidate LOS experiments. The crew/experiment tasks are summarized in Table V.3-1 for C&SM -mounted experiments. Table V.3-2 shows these interfaces for sub-satellite-mounted experiments should they be placed in the C&SM.

Figure V.3-1, Gross Mission Timeline for Lunar Orbital Period, shows those mission and experiment events that impact the CM pilot. The bar graph indicates those critical mission periods when the CM pilot would be fully occupied. Obviously, independent switch functions for On/Off cycling of experiments are excessive for manual control and monitoring by him, even during less critical mission periods. Those experiments requiring this regular cycling should be provided with on-board timers pre-set by the CM pilot or by ground commands to relieve him of these repetitive operations. Crew participation in such activities should be limited to primarily backup or selective control and periodic updating of automatic sequencing functions. The regularity of experiment on-off cycling lends itself to various solutions from relatively simple intervalometer type controls monitored by the CM pilot to computer-sequenced operations requiring little crew participation. MSFN/DSIF command control of experiment on-off functions may provide a simpler alternative to completely automating the on-board experiment. Experiment sequential timing functions could be integrated into the individual experiments or made part of selected experiment group sequencing functions. Integrated experiment group control would be preferable if the CM pilot is required to update, set or otherwise perform simultaneous control functions. Controls and displays should be portable utilizing inflight disconnects that could be abandoned in the LM after completion of the Lunar experimentation program. Experiment panels may need to be carried in from the LM and mounted temporarily at the center couch area near the main C&D panel to permit simultaneous operations by the CM pilot of S/C and experiment operations while maintaining out-the-window visibility of the Lunar surface.

All EVA for the LOS mission is based on evaluation of minimum necessary crew interface with the LOS experiments because of the crew time commitments to the primary lunar landing mission. Increased crew utilization in some areas is desirable including more effective camera film usage. The metric cameras carry considerably more film than can be usefully exposed over the ground track as now planned (each 4th orbit). Crew options to photograph targets of opportunity on in-between orbits would permit increased usage of the available 1200 frames from 50% to 75%-80%.

TABLE V.3-1. CREW/CM/SM/EXPERIMENT TASKS

EXPERIMENT	TARGET ORIENTATION	ALIGN. OR CAL. REQ'D	CREW CONTROLS DISPLAYS	CREW MON. REQ'TS	IMU REALIGN	DATA RETRIEVAL REQ'TS	ATTITUDE AND POINTING REQ'TS
Metric and Stellar Camera	Lunar Surface (Local Vert.)	Yes, check alignment by photo-graphing star fields	Pwr On/Off	NONE	Yes	Retrieve film cassettes	Attitude hold - 9 hrs. (each 4th orbit - 1hr. in day period) ± 2°, ± 0.05° sec
Panaramic Camera	Lunar Surface (Local Vert.)	No	Pwr On/Off	NONE	No	Retrieve film cassettes	Attitude hold - 2 hrs (1 hr-da 1; 1 hr-da.2; 1 hr-da.3) ±2°; ± 0.05°/sec.
Photo-electric Photometer	Lunar Surface (Local Vert.)	No	Pwr On/Off	NONE	Yes	NONE	Attitude hold-18 hrs (every 2nd orbit-day period)
Laser Altimeter	Lunar Surface (Local Vert.)	No	Pwr On/Off	NONE	Yes	NONE	Attitude hold - continuous (72 hrs - nominal ±2°; ±0.0.5°/sec)
RF Trans-ponder	Earth	No	Pwr On/Off	NONE	No	NONE	NONE
Solar Wind Foil	Sun	No	Boom Ext/Ret Foil Ext/Ret	NONE	No	Retrieve foil	Solar ± 15° for 30 hrs.
Radar Imager	Lunar Surface (Local Vert.)	No	ON/OFF Antenna Deploy	NONE	No	Retrieve film	± 1° Y&P; ± 2°R; ± 0.05° sec.

TABLE V.3-2 - CREW/EXPERIMENT TASKS

EXPERIMENT	ORIENTATION	ALIGN. OR CAL REQ'D	CREW CONTROLS/ DISPLAYS	CREW MON. REQ'TS.	DATA RETRIEVAL REQ'TS.	ATTITUDE & POINTING REQ'TS
Meteoroids	Space (Not Critical)	No	Pwr ON/OFF	None	None	(Sub-satellite)
Geochemistry	Lunar Surface	No	Pwr ON/OFF Boom deploy	None	None	(Sub-satellite)
IR Mapping Radiometer	Lunar Surface	No	Deploy Exper.	None	None	(Sub-satellite)
Mass Spectrometer	Lunar Surface	No	Pwr ON/OFF Seal Cover Deploy	None	None	(Sub-satellite)
IR Spectrometer	Lunar Surface	No	PWR ON/OFF	None	None	(Sub-satellite)
MW Radiometer	Lunar Surface	No	Pwr ON/OFF	None	None	(Sub-satellite)
RF Reflectance	Lunar Surface	No	None	None	None	(Sub-satellite)
BI-Static Radar	Lunar Surface	No	None	None	None	(Sub-satellite)
Radar Altimeter	Lunar Surface	No	Pwr ON/OFF Antenna deploy	None	None	(Sub-satellite)
Electromagnetic Meas.	Lunar Surface	No	Antenna deploy	None	None	(Sub-satellite)
Noise Survey	N/A	No	Pwr ON/OFF BW Change Ant. Orient & errection	None	None	(Sub-satellite)
Magnetometer (DC)	Any	No	None	None	None	(Sub-satellite)

TABLE V. 3-2 - CREW/EXPERIMENT TASKS - (Continued)

EXPERIMENT	TARGET ORIENTATION	ALIGN. OR CAL REQ'D	CREW CONTROLS/ DISPLAYS	CREW MON. REQ'TS.	DATA RETRIEVAL REQ'TS.	ATTITUDE & POINTING REQ'TS.
Magnetometer (AC)	Any	No	None	None	None	(Sub-satellite)
Plasma Probe	Any	Yes	Pwr ON/OFF Data Record Deploy	None	None	(Sub-satellite)
Electric Field	Lunar Surface	No	Pwr ON/OFF Antenna deploy	None	None	(Sub-satellite)
Low Energy Particles	N/A	No	Pwr ON/OFF Deploy Covers	None	None	(Sub-satellite)
Solar Energetic Particles	N/A	No	Pwr ON/OFF Deploy Covers	None	None	(Sub-satellite)

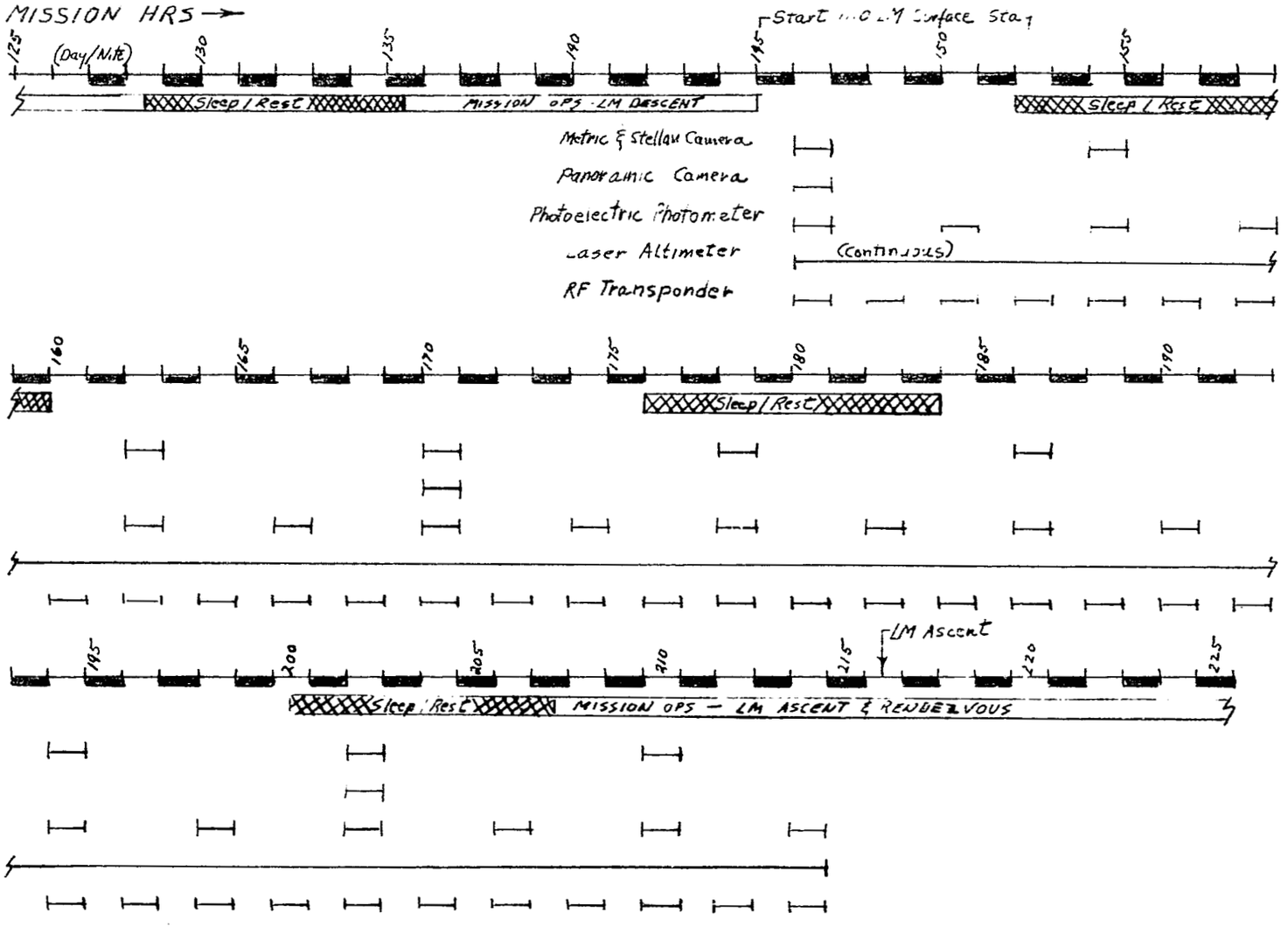


Figure V.3-1 Crew/Experiment Timeline for Lunar Orbital Period

Experiment data may be enhanced by increasing the crew participation in the Microwave Radiometer experiment. Color maps of the Lunar surface viewed on a color Cathode Ray Tube in real time by the CM Pilot could reveal interesting phenomena that would be selectively investigated in greater detail on succeeding orbits. Through repetitive observations, the CM Pilot could distinguish small changes in color (indicating temperature changes) and correlate these with information from optical or radar sensors. An on-board color CRT monitor would be required in the CM as part of the carry-on control and display.

V.3-2 Crew Visibility & Spacecraft Orientation - The recommended S/C orientation is "streamlined" (Local vertical with -Z axis toward nadir). This attitude is chosen from the several alternatives available to provide crew visibility of the ground track and Lunar Landing site through the docking windows while minimizing RCS propellant usage for attitude control. Rolling the SM Z-axis away from the local vertical (e.g. to align Sector I center line with the ground track to simplify experiment mounting) introduces cross-coupling into RCS thruster firings causing more complicated crew maneuvers and less efficient propellant usage. This S/C orientation and visibility footprint is shown in Figure V.3-2. As indicated the SM Sector I centerline is aligned 38 degrees off the ground track, and experiments located in Sector I must be mounted to accommodate this offset position in order to point the -Z axis toward local vertical. Crew visibility of the nadir is not possible unless the C&SM is pitched downward. While this pitched attitude is not recommended for orbital flight, it remains as a crew option anytime viewing the nadir is necessary. In the "streamlined" attitude the near edge of the visibility footprint is 60 NM downrange. Pitching maneuvers to bring this visibility footprint nearer to the nadir would interrupt LOS experiment Lunar Surface (local vertical) pointing and should be performed only as required.

The nose down S/C attitude is not considered acceptable because LOS experiment mounting in SM Sector I would be seriously complicated. The in-between attitude, pitched 45 degrees toward the Lunar Surface, provides the optimum crew visibility, but also causes cross-coupling of the RCS thrusters and spacecraft interference with LOS experiment field of view in Sector I.

In Figure V.3-2, the visibility footprint is shown for the right couch docking window. In one-man lunar orbit, a crewman in either the left or right couch could control the S/C with little change in crew visibility. If the FDAI (Flight Director Attitude Indicator) is required by the crewman, the left seat would be used.

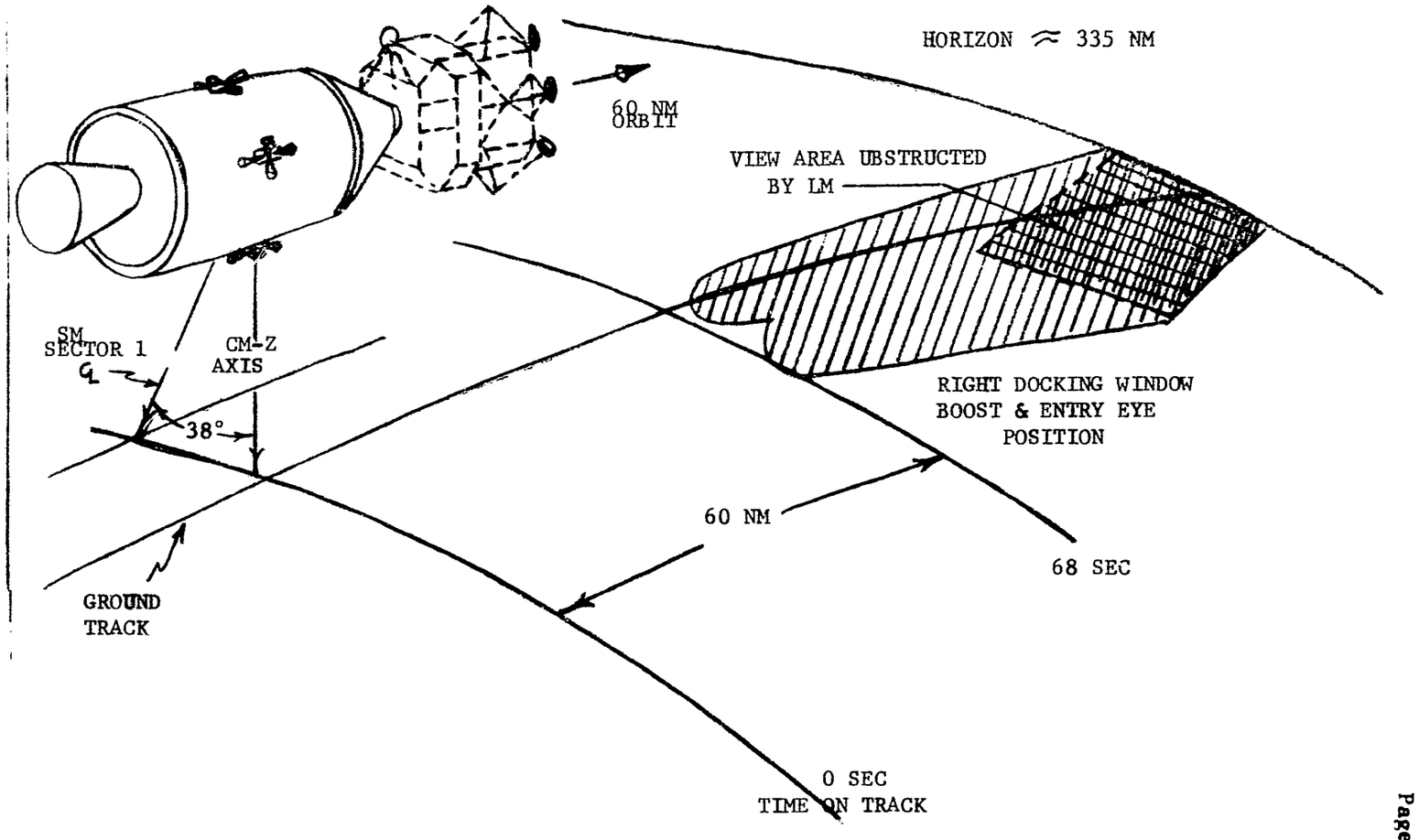


Figure V.3-2 Crew Visibility

In making this analysis, the Weber couch modification in the CM was assumed to have minimum impact on the eye positions formerly specified for boost and entry, and docking.

To maintain the local vertical, "streamlined", orientation within the $\pm$ 1° attitude constraints required by the LOS experiments, the spacecraft must be manually controlled using the ORDEAL system with attitude reference from the Stabilization Control System (SCS). Automatic stabilization utilizing IMU reference is not feasible due to the power-down requirement for the IMU in orbital flight. High drift rates for the SCS attitude reference (approximately 2°/hr) require that the Gyro Display Coupler (GDC) be re-aligned each orbit. This is most simply performed using the Crew Optical Alignment flight (COAS) and the lunar horizon. Manual S/C control is not acceptable for long periods, as required by certain LOS experiments. Addition of an automatic pitch programming capability is required for local vertical control.

V.3.3 EVA Data Retrieval - As presently conceived, LOS experiment payloads mounted in the C&SM require EVA data retrieval prior to earth return. EVA requires all three crewmen to be aboard the CM, limiting EVA opportunities to lunar orbit after the surface mission while the CM and LM are joined or to the trans-earth coast period after the LM has been jettisoned. Crew safety dictates that no EVA be performed from the CM during lunar orbit except in emergency situations. Uncertainty of equipment functioning after depressurization/repressurization and crew fatigue factors after lunar surface stay warrant no other choice. If EVA were performed from the LM prior to LM-jettison, an additional day in lunar orbit and a much more complicated external surface traverse to SM Sector I would be required for data retrieval. In view of these considerations, EVA for LOS data retrieval during lunar orbit is not recommended. No plan to retain the LM during TEI is considered due to limited SPS propellant for the earth return maneuver.

The EMU/PLSS already carried for the lunar surface mission is recommended for the LOS data retrieval EVA to minimize overall mission weight and volume stowage requirements. Even though long, high pressure oxygen and water umbilicals offer certain advantages, the required CM modifications to accommodate the umbilicals and water cooling, and the additional boost stowage of the umbilicals and Pressure Control Units (PCU) cause the concept to be discarded. PLSS refilled from the LM ECS are required for both EVA crewman so that the backup crewman may egress the CM, if necessary, to assist the primary EVA crewman.

Figure V.3-3 shows the general configuration of EVA required. The CM would remain depressurized throughout the EVA. The EVA crewman would proceed over the SM surface to Sector I, backed up by a 2nd standby crewman in the CM crew hatch. The 3rd crewman would monitor the CM systems.

Access to Sector I from the CM would be provided by a deployed handrail mounted on the SM with a short telescoping section to reach to the CM main crew hatch. Fixed handholds at the access openings and fixed handholds on transfer packages are recommended rather than use of portable handholds. Since the film cassettes and solar wind foil requiring EVA retrieval are in the 5 to 65 lb. mass range, they should not require more complicated transfer aids.

In the study of EVA concepts for the Apollo Experiment Pallet (AEP), Martin Marietta evaluated, through 6 degree of freedom simulation, various concepts for retrieving data cassettes from SM Sector I. Results of these studies showed the feasibility of EVA maneuvering from the CM main crew hatch to Sector I and return via a deployed handrail and handholds. Evaluation of the transfer of similar size data cassettes resulted in selection of these deployed handrail and handholds. More complex transfer aids are considered unnecessary for the EVA operations required. Since LOS experiment data retrieval is very similar to that evaluated in previous AEP simulation, results are applicable to the planned LOS EVA operations. Subsequent to work in AEP, the results of Gemini EVA's on flights 9 through 12 provided substantial neutral bouyancy and actual space flight data on the utility and design of handrails, work site tethers and restraint devices.

In the mainline Apollo and AAP programs, continued neutral buoyancy and KC-135 simulation has refined the criteria for design and effective use of handrails, restraints and tethers during surface translation and data retrieval.

Detailed analysis of the specific handrail, tether, and other EVA equipment devices would follow the preliminary vehicle/experiment configuration. Figure V.3-4 shows a typical timeline for EVA during trans-earth coast.

Although EVA data retrieval during trans-earth coast is feasible, the value of the data to be retrieved seems marginal in terms of possible jeopardy to the completion of an already successful lunar landing mission. It is strongly recommended that alternative methods not requiring EVA retrieval be considered. Film chutes may be designed to return exposed film to a cassette attached to the CM or even

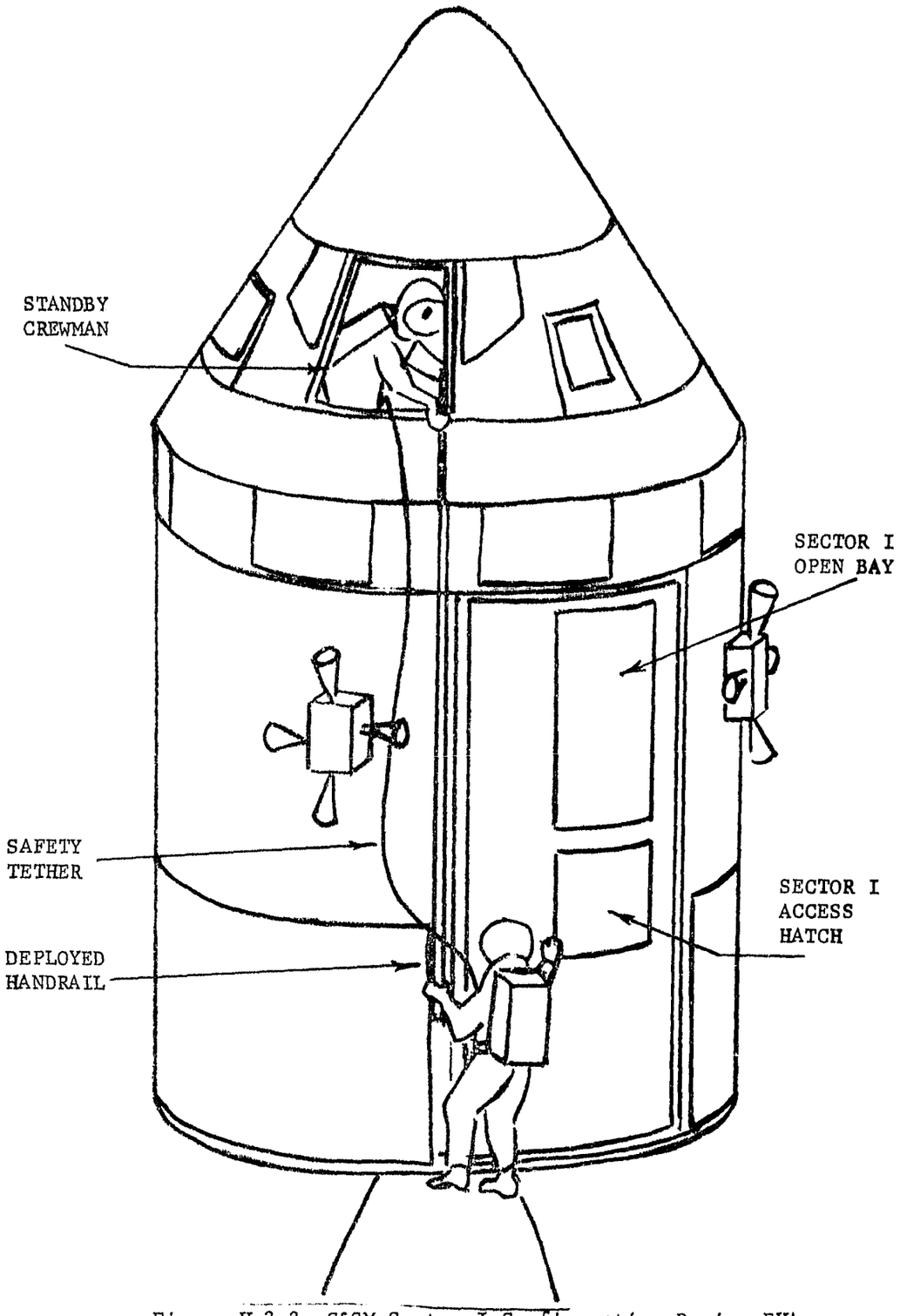
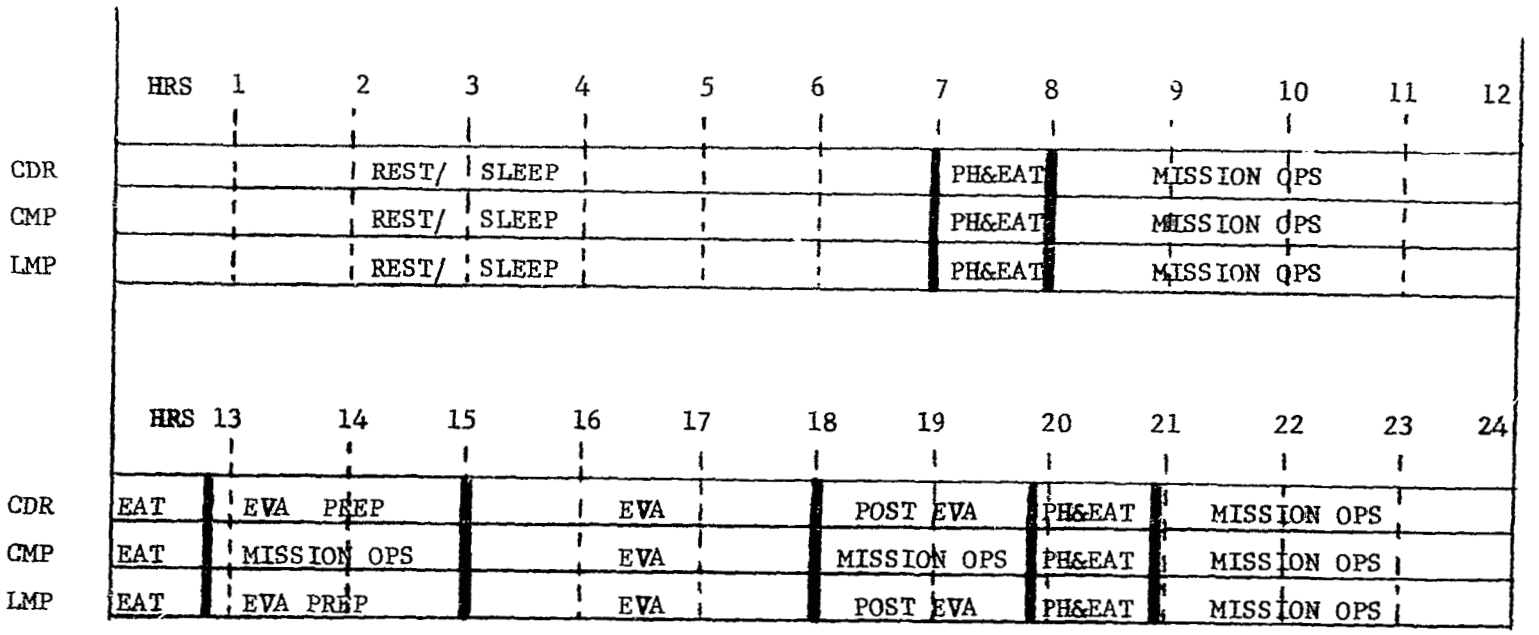


Figure V.3-3 C&SM-Sector I Configuration During EVA



(PH - PERSONAL HYGIENE)

MISSION POSTURE: C&SM IN TRANS-EARTH COAST. THIS T/L STARTS WITH THE FIRST SLEEP PERIOD AFTER THE FIRST MID-COURSE CORRECTION AT SPHERE OF INFLUENCE BOUNDARY

Figure V.3-4 Typical Timeline EVA

into the CM itself. Alternately, the weight penalty should be investigated for a separate reentry film pod that could be injected into earth entry from the SM prior to SM jettison upon earth return. This concept would take advantage of the techniques for film return presently in use from earth orbiting satellites where film pods are snagged by aircraft after reentry. The solar wind foil could be manually deployed and recovered on the lunar surface during LM surface operations obviating any requirement for EVA retrieval.

V.3.4 CM Boost/Reentry Stowage Requirements - Experiment and LOS mission peculiar crew equipment required to be stowed in the CM is shown in Table V.3-3 CM Stowage Requirements. These stowage requirements are based on the recommended concept for EVA data retrieval.

Table V.3-3
CM Stowage Requirement

Equipment	No.	Est. Wt.	Size (in.)	Volume	Boost	Entry
1. Portable Life Support System (PLSS)	2	Charged-86.1 Unchgd -66.0	26 x 17.8 x 10.5	4,850 in^3		X
2. Safety tether	2	10.0	1.6 x 0.75 x 30 ft	450 in^3	X	X
3. Metric camera film cassette	1	21.0	7 x 8 dia	56 in^3		X
4. Stellar camera film cassette	2	6.0	4 x 7 dia	28 in^3		X
5. Panoramic camera	1	65.0	6 x 20 dia	120 in^3		X
6. Solar wind foil	1	0.25	36 x 1.5 dia	60 in^3		X
7. Film canister (radar imager)*	1	5.0	9 x 5 x 3	135 in^3		X

*-Experiment may not be on CM/SM

VI SUBSYSTEM DESIGN REQUIREMENTS

VI.1 DATA AND COMMUNICATION SYSTEMS

The data system capability of four possible carriers (C&SM, Lunar Orbiter, AIMP, and Pioneer) was compared to the experiment requirements derived from responsible scientists and industry sources. The analysis below is based on a representative set of groupings of experiments presented to North American Rockwell in August, 1968. For these groupings, the general type and level of modification required to each of the four carriers was determined.

The integration of the experiments into a carrier in almost every case resulted in a modification, (ranging from a 40% redesign of the Lunar Orbiter system to a complete redesign of the AIMP) requiring addition or replacement of some portion of the data system. A summary of these modifications is presented in Table VI.1-1, based on the experiments of Table VI.1-2. This table further verifies that the changes required are dependent upon the mission experiment grouping. The major components affected are the data storage system, the modulation system, the PCM multiplexing system and the wiring and distribution of commands. The prime areas of concern are the multiplexing of digital data and the storage of high bit rates data (up to 60 kbps) for up to two hours. Further definition of hardware required to accomplish these tasks should be studied in future programs.

VI.1-1 Stable Satellite Analysis - Lunar Orbiter - The basic Lunar Orbiter satellite contains a telemetry (T/M) system capable of transmitting 50 bits per second PCM and 250 kilo hertz video data plus a 38 KHZ pilot tone over an S-band carrier frequency. The satellite also has a receiver which is used for receipt of commands and coherent ranging when used in conjunction with the T/M transmitter. The 50 bps data are used for monitoring analog and bilevel measurement of satellite performance, command verification, flight programmer data and micrometeoroid detectors. Removal of the micrometeoroid detectors, the photographic system, and redundant measurements will provide capacity for approximately 300 bits of data at very low sample rates. Removal of the photographic system also provided the RF link for wideband inputs. The Lunar Orbiter system is incapable of handling serial digital bits, and with its low bit rate, the capability of handling experiment data without modification to the data system is limited.

To establish the level and scope of modifications required to the Lunar Orbiter several experiment groupings were reviewed and their impact on the system ascertained. Figure VI.1-1 is a system which results from the four sample groupings of Table VI.1-2. The experiments considered and their data rates are enumerated on the figure. This system will accommodate all the experiment data. The basic

TABLE VI.1-1

SUMMARY OF DATA SYSTEM MODIFICATIONS REQUIRED TO IMPLEMENT EXPERIMENT GROUPINGS PRESENTED

	STABLE SATELLITE		SPINNING SATELLITE			C&SM			
						GROUP I		GROUP II	
MAJOR COMPONENTS	LUNAR ORBITER PROPOSED MOD	WEIGHT POUNDS	PIONEER PROPOSED MOD	AIMP PROPOSED MOD	WEIGHT POUNDS	PROPOSED MOD	WEIGHT POUNDS	PROPOSED MOD	WEIGHT INCREASE
Antennas & Ant Ckts	No Chg.	14.95	New	New	5.02	No Chg.	------	No Chg.	------
Traveling Wave Tube Amp	No Chg.	5.5	New	New	1.89	No Chg.	------	No Chg.	------
Transmitter	No Chg.	▷1	New	New	1.32	No Chg.	------	No Chg.	------
Modulator	New	4.0*	New	New	4.0*	No Chg.	------	New in SM	4.0*
Command Receiver	No Chg.	▷1	Minor Mod	New	6.13	No Chg.	------	No Chg.	------
Command Decoder	No Chg.	6.42	Minor Mod	New	5.644	No Chg.	------	No Chg.	------
Command Decoder Driver	No Chg.	▷2	Minor Mod	New	▷2	No Chg.	------	No Chg.	------
Transponder Circuits	No Chg.	12.18	Major Mod	New	.5*	No Chg.	------	No Chg.	------
Data Storage System	New	20.0*	New	New	20*	New in SM	20*	2-New	40*
PCM Encoder/Digital Multiplexer	Pioneer or Equiv.	18.46	New	New	10.0*	New in SM	10.00*	New in SM	10.00*
Signal Conditioning	New	1.5*	New	New	.5*	New in SM	1.5*	New in SM	1.5*
Timing (Time of Day)	New	▷1	New	New	1.5*	No Chg.	------	No Chg.	------
Wiring & Controls Ckts	Major Mod	20*	Major Mod	New	20*	New in SM	20*	New in SM	20*
Command Distr. System	Major Mod	3*	Major Mod	New	3*	New in SM	3*	New in SM	3*

*Estimated ▷1 Included in Transponder ▷2 Included in Command Decoder

NOTE: See Table VI.1-2 for experiments included above.

TABLE VI.1-2
SAMPLE EXPERIMENT GROUPING - INPUT TO NORTH AMERICAN ROCKWELL
AUGUST 1968

Command and Service Module		Subsatellite	
Group I	Group II	Stable	Spinning
Metric & Stellar Camera	IR Mapping Radiometer	Geochemistry Expts.	DC Magnetometer
Panoramic Camera	IR Spectrometer	Mass Spectrometer	AC Magnetometer
Photoelectric Photometer	Microwave Radiometer	Radar/Laser Altimeter	Plasma Probe
Geochemistry Experiments	Metric & Stellar Camera	RF Transponder/Laser Retroreflector	Electric Field
Laser Altimeter	RF Reflectance	Electromagnetic Measurements	Low Energy Particles
RF Transponder	Bi-Static Radar	Radio Noise Survey	Solar Energetic Particles
Solar Wind Foil	Radar Altimeter	Gravity Gradiometer	Meteoroids
	RF Transponder	Meteoroids	RF Transponder/Laser Retroreflector
	Radar Imager		
	UV Spectrometer		

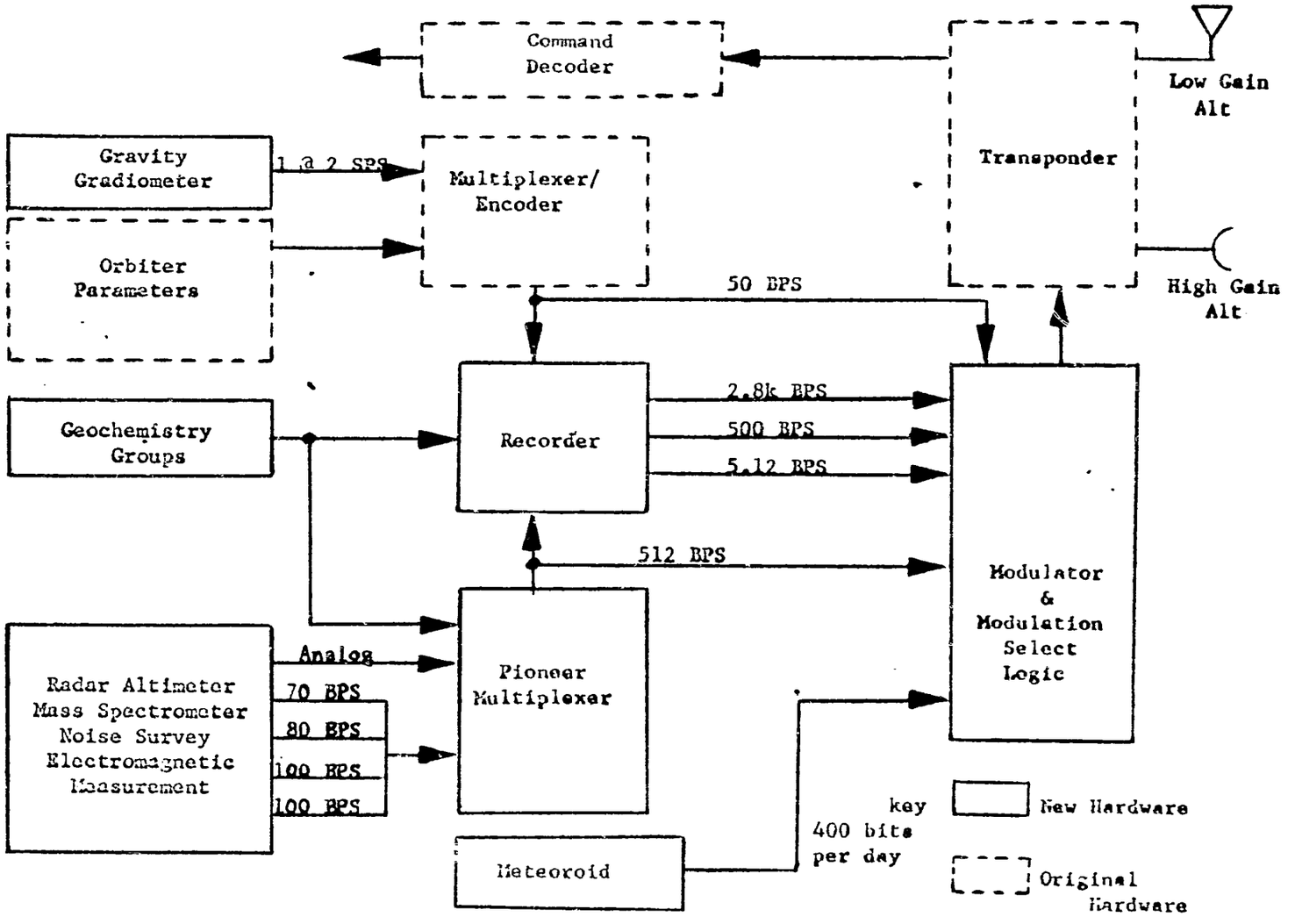


Figure VI.1-1 Stable Satellite - Lunar Orbiter - Data System

data, command, ranging and transponding systems are left intact. The analog data from the experiments would be encoded to six bit resolution and sampled at one sample per sixteen or sixty-four frames. The encoder would provide synchronization and clock signals to the experiments to provide the correct cycling of data into the PCM main frame. The data storage system would be capable of recording data for two hours with a playback speed of 10:1. Three data tracks would be required and the recording speed would be 1-7/8 ips or less since the data rates are very low.

The modulator and modulation select logic system would contain the circuitry required to place the incoming data on separate subcarriers, and provide mixing and signal conditioning of the proper signal for input to the transmitter. The configuration changes required to implement this grouping are noted in Table VI.1-1.

VI.1-2 Spinning Satellite Analysis - Pioneer - The basic Pioneer satellite data system has the capability of transmitting real time and stored data at bit rates of 512, 256, 64, 16 and 8 bits per second. It has four PCM main frame formats for analog and digital data plus two subframes for housekeeping data which are multiplexed into the selected main frames. The system output is bi-phase modulated onto a 2048 hertz (HZ) subcarrier which then modulates the S-band carrier of the transmitter. The satellite also contains a command and transponder system for controlling operation of onboard systems and doppler tracking of the vehicle (Ranging is not available). Each main frame contains 26 words that are available for experiment data. Each word is seven bits long (six data plus one parity bit). Using the maximum bit rate of 512 bps provides 355 bps for experiment use. The Pioneer data system as described does not have the capacity to transmit realtime and stored data simultaneously, nor does it contain a ranging system. Adaptation of this sytem to a particular experiment grouping presents the same type of problems as noted for the Lunar Orbiter. Figure VI.1-2 presents a data system configured to manage the data from the experiments presented in Table VI.1-2. The total bit rate to be handled (3,150 bps) is much greater then the 355 bps available; therefore the system requires the addition of new hardware. Table VI.1-1 summarizes the changes required to satisfy the experiments.

The basic Pioneer System transmitted 2048 Hz signals over its S-band link. Therefore the bandwidth of the transmitter, TWT's and antennas, if optimized to this signal, will not contain sufficient bandwidth to handle the playback data. Modification

or replacement of these components would be required. The tape storage system would require 3-3/4 ips recording speed, with a 10:1 playback ratio. The PCM encoder would have data capacity as noted in Figure VI.1-2 and the modulation system would perform the same function noted for the Lunar Orbiter modulator. A major modification of the transponder circuits would be required to provide capability for ranging, to support tracking and experiment functions. The modifications required on the Pioneer are to a greater level than those of the Lunar Orbiter but the basic approach to handling the data is the same.

VI.1-3 Spinning Satellite Analysis - Anchored Interplanetary Monitoring Platform (AIMP) - The basic AIMP satellite telemeter utilizes pulse frequency modulation and was designed to provide telemetered data at distances up to 600,000 KM. This system utilizes VHF frequencies (136 & 142 MHZ) for ranging, commands, and transmission of telemetered data. The system is capable of encoding analog data and multiplexing digital data. Both data are then pulsed frequency modulated onto the carrier utilizing frequency bursts which have an equivalent binary value of four bits.

The T/M output is composed of frequency bursts that are equivalent to a sequence of 1320 digital data bits, 974 analog data bits and 128 snychronization bits which require 81.92 seconds to transmit, resulting in an average bit rate of 29.6 bits per second. The data experiment requirements for the AIMP and the Pioneer are the same (see Table VI.1-2). The AIMP system is totally inadequate to handle the data; therefore, a new system is required. Since the experiment requirements are the same as for the Pioneer, the system shown in Figure VI.1-2 can represent the AIMP system with all new hardware.

VI.1-4 Command and Service Module (C&SM) Analysis - The C&SM data and ranging system has as its prime function the support of the Apollo Mission. Excess data capacity and time sharing are therefore the only areas that may be considered for support of the experiments.

The data requirements for the sample experiment groupings for the C&SM (Table VI.1-2) are indicated in the data and system shown on Figures VI.1-3 and VI.1-4 respectively. The major obstacles to accommodation of all the experiment data directly into the Apollo format for either of the two C&SM experiment groups of Table VI.1-2 are:

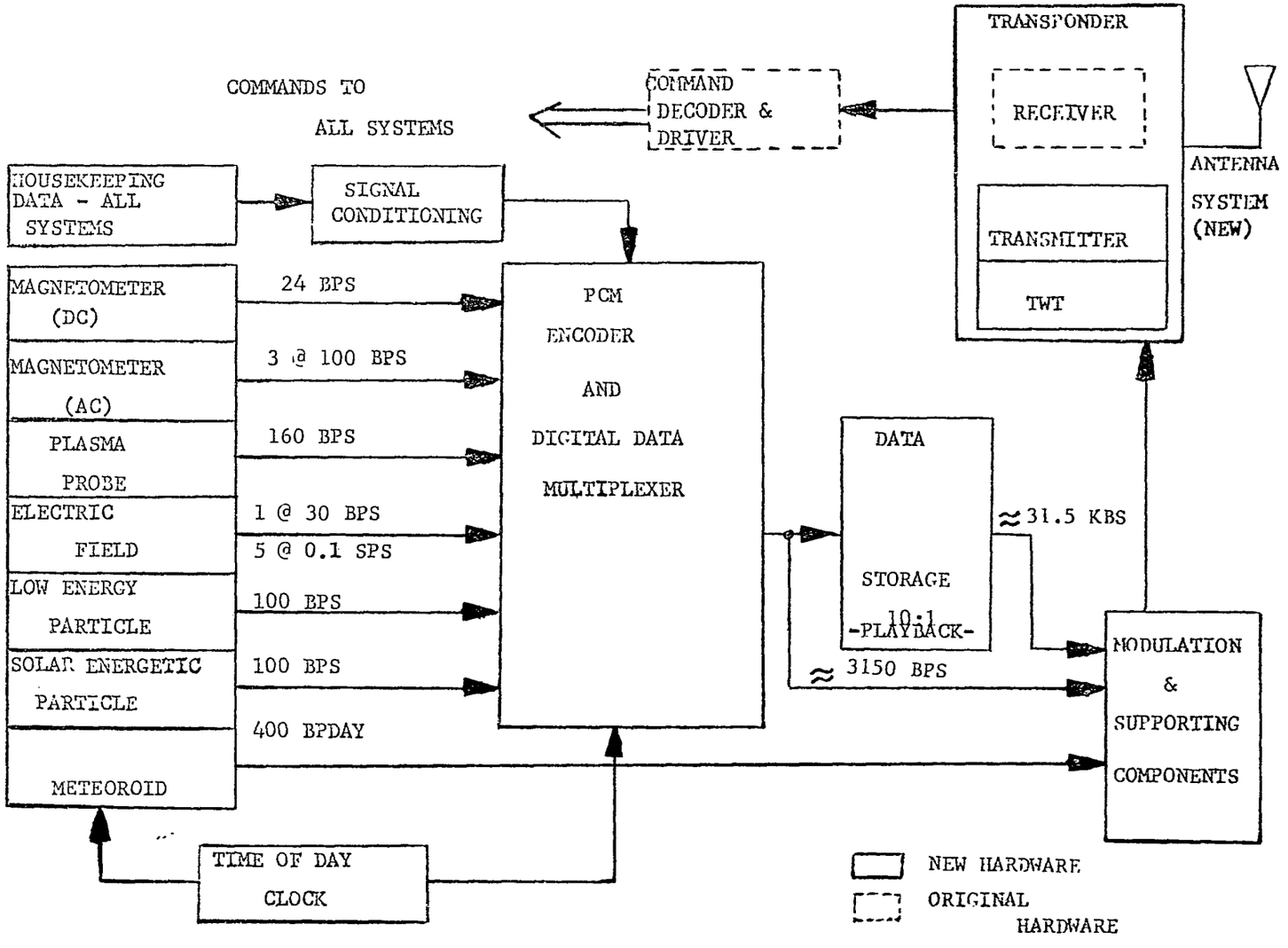
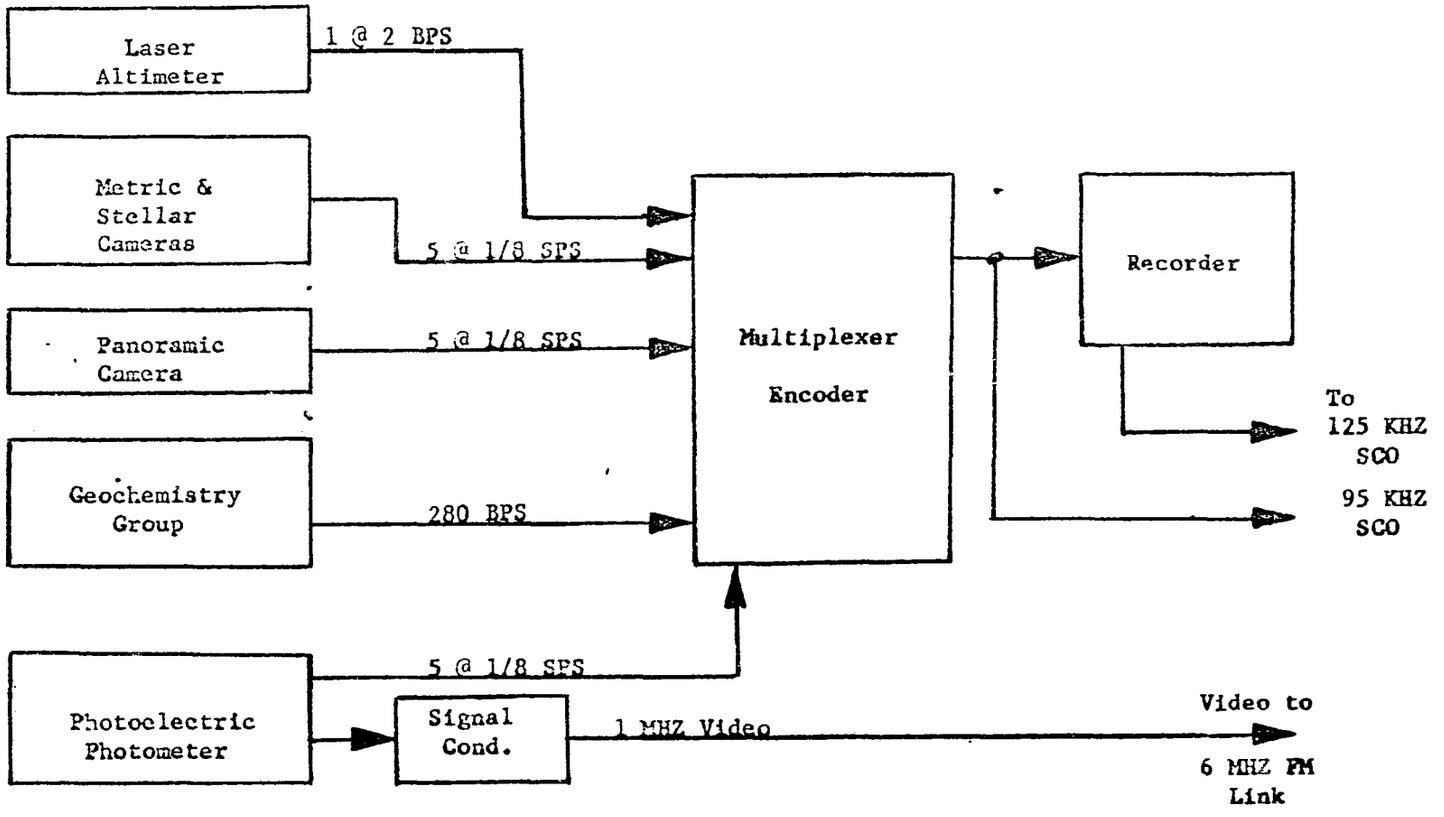


Figure VI.1-2 Pioneer - Spinning Satellite Data System



*Group I Experiments per Table VI.1-2

Figure VI.1-3 C&SM - Group I Experiments* - Data System

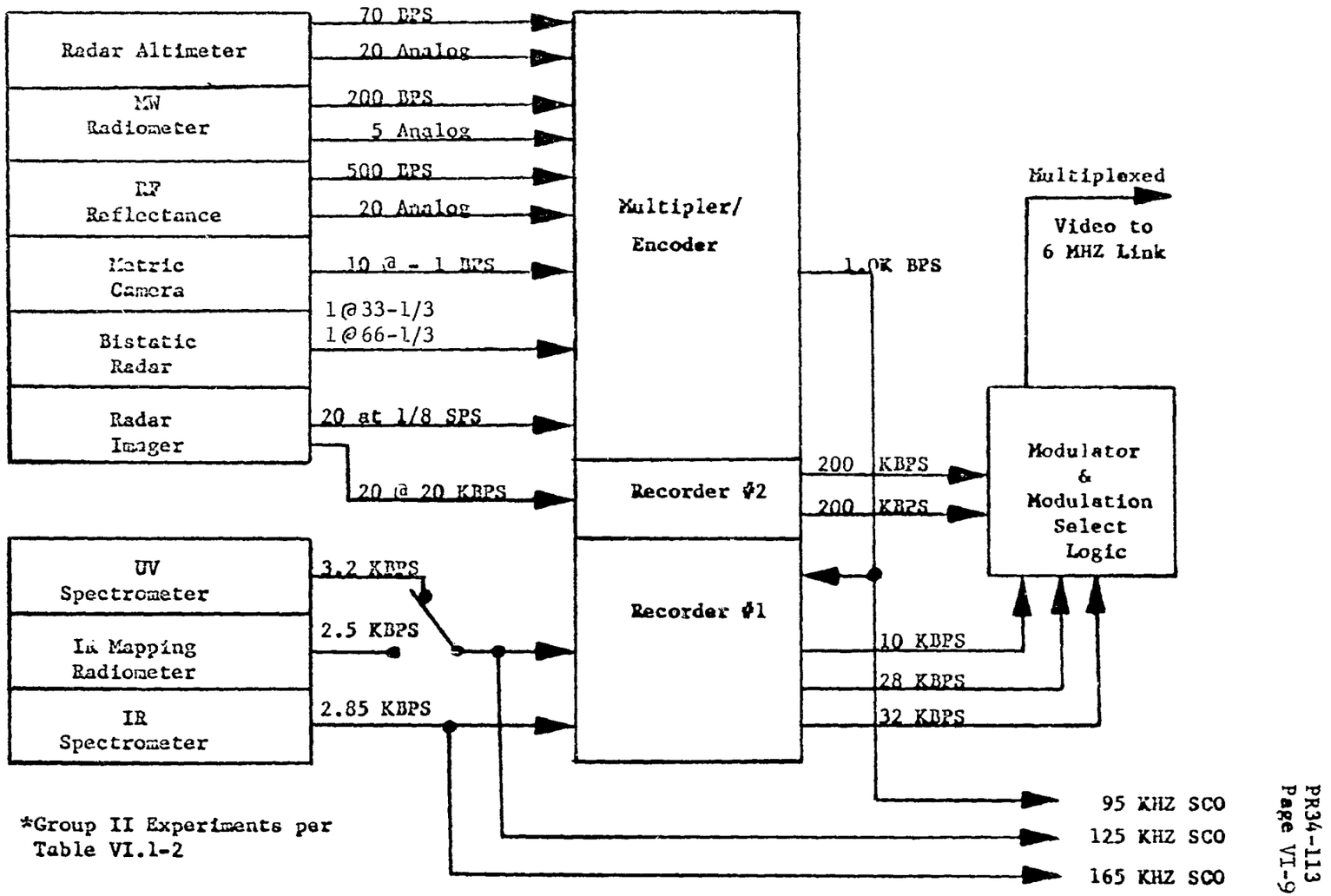


Figure VI.1-4 C&SM - Group II Experiments* - Data System

(a) Limited digital data capacity,

(b) Insufficient data storage (digital or FM), insufficient storage time (duration) and insufficient playback to record ratio (1:1) of the recorder (in the high bit rate mode);

(c) Time sharing of the encoding system and the FM links which could impair the Apollo data acquisition or the experiment data acquisition or both, due to the high data rates required by the experiments.

The adaptation of the groupings to the C&SM system could be accomplished in several ways. Figures VI.1-3 and VI.1-4 present simplified methods of accommodating the data. Figure VI.1-3 represents a system that would utilize the biomedical subcarriers and the 6 MHz wideband FM carrier identified by North American Rockwell for transmission of realtime and playback data. Updata link, manual commands, and ranging would be accomplished by the existing C&SM onboard systems. The tape recorder used for this grouping would be operated at 1-7/8 ips and playback at a 10:1 ratio for a two hour recording time. Therefore, the use of the 125 KHz subcarrier would be limited to about 12 minutes per orbit. This data system has a minimum impact upon the C&SM system.

The experiment data system shown on Figure VI.1-4 is more complex and places greater requirements upon the basic C&SM system. The grouping contains experiments with high data rates and requirements for continuous operation; therefore, realtime and playback capability is required to support them. The system shown would require a PCM Encoder/multiplexer, two recorders and a modulator located in the SM. The three subcarriers of the C&SM (85 KHz, 125 KHz, and 165 KHz) and the six MHz wideband FM channel would also be required to support this mission. If the six MHz channel is not available, a separate S-band transmitter would be required. Total transmission time of the wideband FM would be limited to approximately twelve minutes per orbit.

VI.1-5 Ground Checkout and Launch Requirements

(a) Subsatellite Data Systems - The location of the subsatellite during prelaunch checkout will be in the Sector I of the SM. In this position, all the deployable antennas will be stowed and open loop verification will not be possible. Therefore, all checkout after satellite installation will be accomplished closed-loop or by using test outputs that will be designed into the system. The interface after installation will be manual-disconnect-umbilicals with terminations inside

Sector I therefore not accessible after cover installation. Critical parameters that require monitoring after Sector I is sealed and during the interval of time prior to deployment could utilize spare channels in the C&SM data system. Therefore, they could be monitored as part of the normal C&SM data flow throughout prelaunch and pre-deployment phases of flight.

The data system checkout requirements under the constraints noted previously would be limited to verifying the correct operation of prime components. Parameters such as output power and voltage levels and the verification of subcarrier and carrier frequencies, the bit rates, command and command responses and other prime parameters required will be monitored to establish the integrity of the system. Due to the limited number of measurements and the non-standard bit rates of each of the experiments, complex automatic checkout equipment will probably not be used to verify systems operation. The ground checkout equipment will utilize standard ground station equipment with the possible exception of the PCM decommutator. The decommutator will require the flexibility of handling non-standard bit rates.

(b) C&SM Data System - The experiments studied thus far for flight in the C&SM utilize the RF loops and command system provided by the C&SM. Therefore, the checkout of any portion of the data system located in Sector I can be performed using the normal C&SM checkout methods. Self calibrate or simulators may be required to support checkout but verification of the parameters defining experiment integrity will be required in any case. Since the recommended systems do utilize the six MHz wideband FM channel, automatic checkout equipment can be by-passed. This allows use of ground station facilities tailored to meet the Sector I data reduction requirements with reduced complexity in equipment and checkout methods.

VI.1-6 Peculiar Subsystem Support Requirements - It is anticipated that any data subsystem used in the C&SM or subsatellites will be designed to meet the expected environments without special support. Protection will be required for extreme environments such as micrometeorites, high radiation sources, temperatures below $0^{\circ}F$ and above $160^{\circ}F$.

VI.2 POWER SYSTEM

The power system requirements necessary to support the selected experiment groupings on each of four carriers, the C&SM, the Lunar Orbiter (small stable satellite), the Pioneer (small spinning satellite) and the Anchored Interplanetary Monitoring Platform (small spinning satellite) were analyzed. The latter three carriers are subsatellites deployed from the C&SM Sector I. This section presents the existing vehicle power configuration, reflects the power required to support the sample groupings, and suggests the revisions to the existing vehicle power systems to provide the required power.

Based on the experiment requirements and groupings of Section III, the required power for experiments exceeded the existin capacity of all four carriers. The capacity of the three su.- satellites with power output approximately doubled is summarized as follows:

Lunar Orbiter	Solar array - double size - total output	766 watts
	Battery - add 2 for a total of 3	252 watts
Pioneer	Solar array - double size - total output	170 watts
	Battery - add Lunar Orbiter type	84 watts
AIMP	Solar Array - double size - total output	130 watts
	Battery - add Lunar Orbiter Type	84 watts

However, even the increases shown above are inadequate to meet the grouped experiment requirements of Table III.1-2. The power system revisions necessary to support the experiment groupings in the C&SM Sector I were analyzed for a three-day mission. Since the C&SM does not have a solar array, all power required must be provided by the addition of batteries to the system.

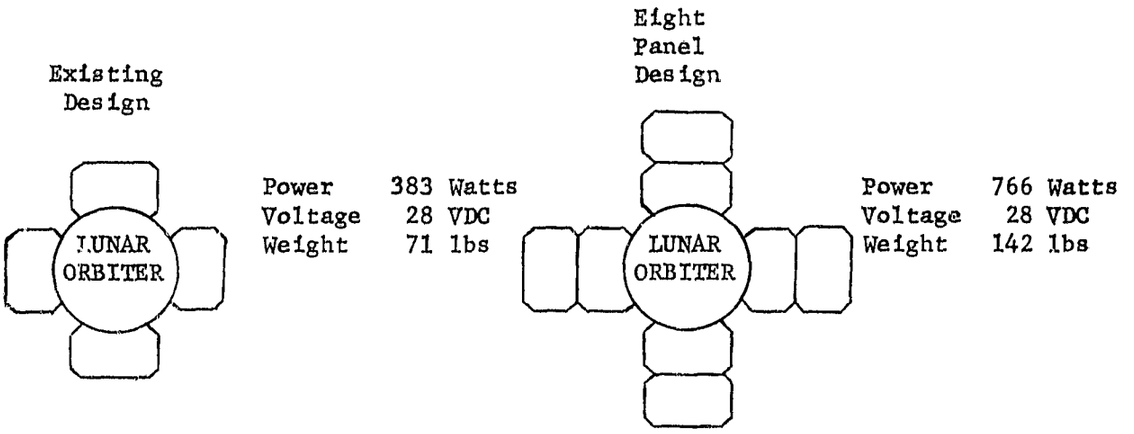
The revisions to the power systems suggested herein reflect the maximum feasible increases based on using multiples of the existing solar panels and batteries. However, further studies may reveal that it would be possible to provide the required power by complete redesign of the solar array and use of a battery with an increased output capacity. It should also be noted that such

perturbations as changing the orbit from circular to elliptical could improve the day to night ratio and reduce the battery power required, thus reducing the battery charging power required of the solar array. Experiment operating timeline revisions can also be used to reduce the total power required as opposed to the continuous operation now required of most experiments.

The experiment requirement that the experiment axis remain aligned to the lunar local vertical for the small stable satellite experiment groups, together with the power system requirement that the solar array be aligned to the sun, necessitates a gimbal system for either the solar array, the experiments or both. However, since the power output varies as the cosine of the illumination angle a sixty degree change will still provide one half the total power. The proposed battery dissipation during experiment operation is only 25% and the battery should provide full power for 6000 cycles or 500 days.

VI.2-1 Subsatellite Power - Each of the three subsatellite vehicles, the Lunar Orbiter, the Pioneer and the AIMP, has a solar array/battery power system. Only the Lunar Orbiter, however, has a system capable of recharging the battery to provide power for vehicle operation during sun occult for a circular lunar orbit. The use of these vehicles as subsatellites for future Apollo missions will require a system to provide power during both orbital day and orbital night operation. The power subsystem requirements are based on a sixty nautical mile, circular lunar orbit which provides a near 1:1 day to night ratio. Since all power dissipated by the battery during orbital night must be replaced by the solar array via the charge controller during orbital day and since the present timelines call for continuous operation of most experiments, the day and night experiment power requirements are almost identical; therefore, the solar array output capability must be approximately double the battery output. Based on a thirty day prime mission and a two hour orbital period the battery would require 360 cycles.

VI.2-2 Small Stable Satellite Power - The present Lunar Orbiter power system is a solar array/battery system. The solar array is a four flat panel n-p solar cell power source composed of 10816 2x2 cm cells connected in a series-parallel arrangement to provide a voltage of 26.6 to 31.0 VDC with a maximum power of 383 watts. The total output power of the solar array may be increased by increasing the number of solar panels composing the solar array system. The maximum number of panels feasible considering the structural and weight constraints imposed on the vehicle is probably about eight. The existing system and a system with double the power are shown below:



The Lunar Orbiter battery is a 30-pound 12-ampere hour nickel-cadmium power source to provide vehicle power for sun occult operation. The major design consideration for establishing the number of batteries required however, is the fact that the orbital 1:1 day/night ratio the current dissipated during the sun occult operation must be replaced by the solar array via the charge controller during the illuminated portion of the flight. Therefore, the maximum charge rate of three amps at 28 VDC limits the battery output to 84 watts. Based on the weight and space constraints for the Lunar Orbiter, the maximum number of batteries to be added would be two for a total of three batteries. The three battery system will provide a total continuous power during sun occult of 252 watts at 28 VDC. The weight of the three battery system is 90 pounds.

The total vehicle power to support the proposed missions includes subsystem power as well as the power to operate the experiments. The subsystem power requirements are given by Table VI.2-1. The required experiment power and total vehicle power for the small stable satellite experiment groupings follow in Table VI.2-2.

The total maximum power available from the eight panel solar array and the three battery system is not sufficient to support the Group A and C experiments as noted in Table VI.2-2. Therefore, a redesign of the solar array/battery system would be required to further increase the output capacity to support such a mission. The Group B experiment totals, however, reveal that the eight panel, three battery system would be capable of supporting the mission as the nighttime discrepancy is sufficiently small and should be easily solvable.

TABLE VI.2-1
STABLE SATELLITE SUBSYSTEM POWER REQUIREMENTS

	ORBITAL DAYTIME WATTS	ORBITAL NIGHTTIME WATTS
Communications Subsystem - Total	(142.5)	(39.5)
Timing (Time of Day)		
Transponder System	17.0	4.0
Command Receiver		
Transmitter		
TWT	70.0	0
Data Storage System	35.0	15.0
Signal Conditioning	0.5	0.5
PCM Encoder/Multiplexer (2)	14.0	14.0
Modulator	2.0	2.0
Command Decoder	4.0	4.0
Command Decoder Driver		
Power Subsystem - Total	(16.0)	(9.0)
Battery Charge	(Varies with total power - See groupings)	
Shunt Regulator	2.0	2.0
Charge Controller	14.0	7.0
Attitude Control Subsystem - Total	(125.0)	(125.0)
Thermal Control Subsystem - Total	(43.0)	(43.0)
Total Subsystem Power (without active thermal control)	283.5	173.5
(with active thermal control)	326.5	216.5

Note: Thermal control is based on the stable satellite experiment Group C on Table VI.2-2.

TABLE VI.2-2
STABLE SATELLITE POWER SUMMARY

GROUP A		GROUP B		GROUP C	
Experiment	Power Watts	Experiment	Power Watts	Experiment	Power Watts
Geochemistry	75	Geochemistry	75	IR Mapping Radiometer*	150
Radar Altimeter	70	Mass Spectrometer	12	IR Spectrometer	15
Gravity Gradiometer	5	Meteoroids	5	MW Radiometer	40
Electromagnetic Meas.	50			RF Reflectance	70
Mass Spectrometer	12			Bi-Static Radar**	2
Meteoroids	5			Radar Altimeter	70
Noise Survey	5			Electromagnet Meas.	50
				Noise Survey	5
Total	222 watts		92 watts		402 watts
Daytime Operation:					
Experiment Total	222.0 watts		92.0 watts		***252.0 watts
Total Subsystem Power	283.5 watts		283.5 watts		326.5 watts
Battery Chrg. Power	500.0 watts		375.0 watts		800.0 watts
Total Daytime Power	1005.5 watts		750.5 watts		1378.5 watts
Max Power Available from 8 panel solar array	766.0 watts		766.0 watts		766.0 watts
Power margin from 8 Panel Solar Array	- 239.5 watts		+ 15.5 watts		- 612.5 watts
Nighttime Operation:					
Experiment Total	222.0 watts		92.0 watts		402.0 watts
Total Subsystem Power	173.5 watts		173.5 watts		216.5 watts
Total Nighttime Power	395.5 watts		265.5 watts		618.5 watts
Power Available from 3 batteries	252.0 watts		252.0 watts		252.0 watts
Power Margin from 3 batteries	- 143.5 watts		- 13.5 watts		- 366.5 watts

* Operates every other orbit, nighttime
** Operates every orbit nearside
*** Daytime power is total power minus the IR mapping radiometer

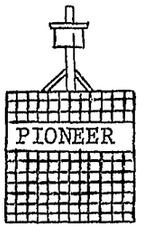
The three groups discussed here provide a total power range sufficiently broad to allow any grouping from the Section III roadmaps to be analyzed. Assuming the subsystem power is constant, the total experiment power for any such grouping may be compared with the total experiment power of the sample groupings given in Table VI.2-2 and the feasibility determined.

VI.2-3 Small Spinning Satellite Power - Two existing satellites, the Pioneer and the Anchored Interplanetary Monitoring Platform (AIMP), were considered for use as spinning subsatellites. Both have solar array/battery power system and the main difference is in structural design configuration.

The Pioneer solar array system consists of approximately 10,000 n-p type solar cells 2x2 cm spread over the cylindrical spacecraft surface. The cells are connected in a series-parallel combination to provide 24 to 33 VDC with a maximum power of 85 watts. From a power point of view, spinning satellites are normally used when the total power required from the solar array is 100 watts or less. When the power required is in excess of 100 the small stable satellite with the sun oriented solar array is selected.

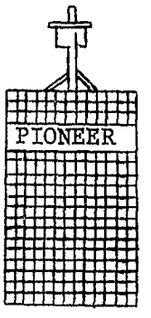
For the purpose of this study, doubling the size of the solar array was assumed to be feasible, thus doubling the power output capacity. The existing system and the double capacity configuration are shown below.

Existing Design



Power 85 Watts
Voltage 28 VDC
Weight 17 lbs

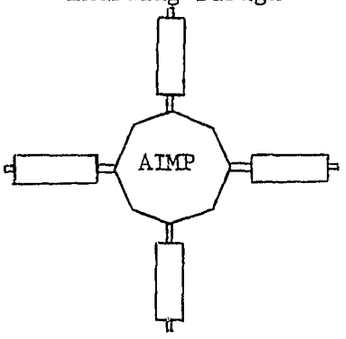
Double Output Design



Power 170 Watts
Voltage 28 VDC
Weight 34 lbs

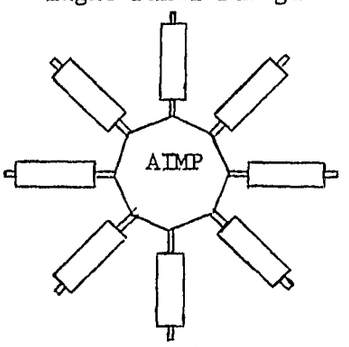
The AIMP solar array is composed of four flat panels with 7680 2 x 2 cm solar cells. The cells are connected in a series - parallel combination to provide 19.8 VDC with a maximum power of 66 watts. The maximum number of solar panels feasible is probably about eight. The eight panel configuration will double the solar array output capacity. The number of cells in series will be increased to provide the 28 VDC required by the experiments. The existing system and eight panel configuration to double the power output are shown below.

Existing Design



Power	66 watts
Voltage	19.8 VDC
Weight	21 lbs

Eight Panel Design



Power	130 watts
Voltage	28 VDC
Weight	42 lbs

Neither the Pioneer two ampere-hour silver-zinc battery nor the AIMP eleven ampere-hour silver cadmium battery have the capacity to provide the continuous power required for sun occult operation. To support the proposed groupings one Lunar Orbiter type thirty pound, twelve ampere hour, nickel cadmium battery replaces the existing battery for either vehicle. The maximum charge rate of 3 amps at 28 VDC limits the total dissipation to 84 watts per orbit.

The total vehicle power to support the proposed missions includes subsystem power as well as the power required to operate the experiments. Table VI.2-3 gives the subsystem power requirements.

The required total vehicle/experiment power for the small spinning satellite is given in Table VI.2-4.

The total maximum power available from the solar arrays as shown in Table VI.2-4 is not sufficient to support either Group A or Group B. Therefore, a redesign of the solar array system would be required to further increase the total output capacity. The Lunar Orbiter type battery added to either vehicle would provide sufficient power for sun occult operation as shown in Table VI.2-4.

TABLE VI.2-3
SPINNING SATELLITE SUBSYSTEM POWER REQUIREMENTS

	ORBITAL DAYTIME WATTS	ORBITAL NIGHTTIME WATTS
Communications	(75.75)	(25.25)
TWT	30.0	0
Transmitter	0.5	0
Modulator	0.5	0.5
Command Receiver	0.5	0.5
Command Decoder	0.5	0.5
Command Decoder Driver	0.5	0.5
Data Storage System	35.0	15.0
PCM Encoder/Digital Multiplexer	7.0	7.0
Signal Conditioning	0.5	0.5
Time (Time of Day)	0.25	0.25
Power	(8.0)	(4.5)
Shunt Regulator	1.0	1.0
Charge Controller	7.0	3.5
Battery Charge (Varies with total power - see groupings)		
Attitude Control	(5.0)	(5.0)
Thermal Control	(5.0)	(5.0)
Total Subsystem Power	93.75	39.75

TABLE VI.2-4
SPINNING SATELLITE POWER SUMMARY

GROUP A EXPERIMENTS		GROUP B EXPERIMENTS	
EXPERIMENT	POWER WATTS	EXPERIMENT	POWER WATTS
Magnetometer - DC	1.5	Magnetometer - DC	1.5
Magnetometer - AC	2.0	Magnetometer - AC	2.0
Plasma Probe	3.5	Plasma Probe	3.5
Electric Field	3.0	Electric Field	3.0
Low Energy Particles	5.0	Low Energy Particles	5.0
Solar Energetic Particle	5.0	Solar Energetic Particles	5.0
Meteoroids	5.0		
Experiment Total	25.0 watts		20.0 watts
Daytime Operation:			
Experiment totals	25.00 watts		20.00 watts
Subsystem Totals	93.75 watts		93.75 watts
Battery Charging	100.00 watts		100.00 watts
Total Daytime Power	218.75 watts		213.75 watts
Max. Power Avail from Double Capacity Pioneer	170.00 watts		170.00 watts
Max. Power Avail from 8 Panel AIMP	130.00 watts		130.00 watts
Pioneer Power Margin	- 48.75 watts		- 43.75 watts
AIMP Power Margin	- 88.75 watts		- 88.75 watts
Nighttime Operation:			
Experiment totals	25.0 watts		20.00 watts
Subsystem totals	39.75 watts		39.75 watts
Total Nighttime Power	64.75 watts		59.75 watts
Power Avail from Lunar Orbitor Type Battery on Both AIMP & Pioneer	84.00 watts		84.00 watts
Power Margin Pioneer & AIMP	+19.25 watts		+24.25 watts

The feasibility of using the small spinning satellite to support other experiment groupings resulting from the roadblocks of Section III can be analyzed by comparing the total experiment power required for such a grouping with the experiment power of Group A and B of Table VI.2-4, assuming that the subsystem power is constant.

4. C&SM Power - Experiments hardmounted in Sector I of the Service Module (SM) are supplied 10 KWH power from the C&SM power system. All additional power required will come from batteries added for that purpose. The total power requirements are based on a 60 nautical mile, circular lunar orbit and reflect a three day mission.

Table VI.2-5 reflects the power required by subsystems added to the C&SM to support the proposed experiments. They interface with the appropriate C&SM systems to complete the desired functions where applicable.

Two groupings of experiments, Group A and Group B, are identified by Table VI.2-6 and VI.2-7. The total power for each experiment is computed on the basis of total operational time for the designated missions. Ten KWH power is reserved for experiments from the basic C&SM power system. The batteries added provide for the additional power required.

VI.2-5 Power System Ground Checkout and Launch Requirements

(a) Subsatellite Power Systems - Ground checkout of the subsatellite power system will be completed prior to the installation of the subsatellite in Sector I of the C&SM. Complete checkout will include battery load tests, solar panel output verification and continuity checks of the electrical power and distribution system. No further testing will be required after the subsatellite is installed in Sector I. The subsatellite power system is not operational until the subsatellite is deployed from the C&SM.

(b) C&SM Experiment Power System - The power system required to support the experiments hardmounted in Sector I of the C&SM will be checked out using the C&SM checkout umbilical and will be part of the normal C&SM checkout sequence.

TABLE VI.2-5
C&SM SUBSYSTEM POWER REQUIREMENTS

	WATTS	
Communications	(42.0)	Experiment Group A (See Table VI.2-6)
PCM Encoder/Multiplexer	7.0	
Data Storage System	35.0	
Communications	(75.0)	Experiment Group B (See Table VI.2-7)
PCM Encoder/Multiplexer	7.0	
Data Storage System	67.0	
Modulator	1.0	
Power	(5.0)	
Altitude Control	(Provided by C&SM)	
Thermal Control	(5.0)	Experiment Group A
Thermal Control	(43.0)	Experiment Group B
Total Subsystem Power	52	Experiment Group A
Total Subsystem Power	123	Experiment Group B

TABLE VI.2-6
C&SM POWER SUMMARY - GROUP A EXPERIMENTS

EXPERIMENTS	POWER WATTS	OPERATION	TOTAL POWER
Metric & Stellar Camera	80.0	Every 4th Orbit Daylight	.7
Panoramic Camera	500.0	1 Hr 1st Day 1.2 Hr 2nd day 1 Hr 3rd Day	1.1
Photoelectric Photometer	4.0	Every Other Orbit Daylight	.1
Geochemistry Experiment	75.0	Continuous	5.4
Laser Altimeter	100.0	Continuous	7.2
Solar Wind Foil	0	Every Orbit Daylight (30 Hrs required)	0
Total	759.0	Total	14.5 KWH

TOTAL EXPERIMENT SUPPORT POWER REQUIREMENTS

Experiment Total	759 Watts	14.5 KWH
Communications Subsystem	42 Watts	3.0 KWH
Power Subsystem	5 Watts	.4 KWH
Attitude Control Subsystem	C&SM PROVIDED	-
Thermal Control Subsystem	5 Watts	.4 KWH
Total Experiment Loads	811 Watts	18.3 KWH

VEHICLE POWER SYSTEM REVISION

Power Available from C & SM	10 KWH	
Add 1 Silver-Zinc Battery 3 Day Mission	12 KWH	130 LBS.

TABLE VI.2-7
C&SM POWER SUMMARY - GROUP B EXPERIMENTS

EXPERIMENTS	POWER WATTS	OPERATION	TOTAL POWER
IR Mapping Radiometer	150.0	Everyother Orbit Darkside	5.4
IR Spectrometer	15.0	Continuous	1.1
MW Radiometer	40.0	Continuous	2.9
Metric and Stellar Camera	80.0	Every 4th Orbit Daylight	0.7
RF Reflectance	10.0	Continuous	0.7
Bi-Static Radar	2.0	Every Orbit Near Side	0.1
Radar Altimeter	70.0	Continuous	5.0
Radar Imager	340.0	Every 4th Orbit Day & Night	6.1
UV Imager	24.0	Every other Orbit, Day	0.4
Total	731.0	Total	22.4 KWH

TOTAL EXPERIMENT SUPPORT POWER REQUIREMENTS

Experiments	731 Watts	22.4 KWH
Communications Subsystem	75 Watts	5.4 KWH
Power Subsystem	5 Watts	.4 KWH
Attitude Control Subsystem	C&SM Provided	-
Thermal Control Subsystem	43 Watts	3.1 KWH
	854 Watts	31.3 KWH

VEHICLE POWER SYSTEM REVISION

Power Available from C&SM	10 KWH	
Add 2 Silver-Zinc Batteries 3 Day Mission	24 KWH	260 LBS

VI.3 GUIDANCE & CONTROL

This section describes the attitude control subsystem support for lunar orbital science experiments. The experiments require evaluation of three separate control concepts:

Continuous orientation of SM Sector I mounted experiments along the lunar local vertical.

Continuous long term orientation of experiments on a subsatellite along the lunar local vertical.

Spin stabilization of a subsatellite.

Subsystem study effort was limited to an investigation of the feasibility of meeting experiment requirements and it was concluded that provision of the necessary attitude control to meet these requirements is feasible. No unreasonable problems were identified, although not all hardware is available "off the shelf" at the present time. In all cases, requirements may be met by alternate methods. Principal identified alternatives are described, but mode studies have not been performed and no selection is implied.

VI.3-1 Command and Service Module - Three systems for achieving local vertical orientation and attitude hold of Sector I mounted experiments are described below:

a. PGNCS and RCS;

b. Orbital Stabilization System (OSS) and RCS with Sector I mounted horizon sensors ;

c. Independent auxiliary system in Sector I.

A detailed accuracy study has not been performed, however, previous work has shown that a pointing accuracy on the order of 1 deg. is reasonable. The PGNCS could be expected to do somewhat better than this dependent on the frequency of Inertial Measurement Unit alignments. The PGNCS would provide local vertical pointing through appropriate computer programming to calculate the direction of local vertical from an inertial coordinate system plus further transformations to determine the orientation of Sector I relative to the local vertical. On this basis the vehicle could be maneuvered to the proper attitude and maintained there in the automatic command module computer

attitude hold mode. This system would require no additional hardware and is obviously lowest in component weight. Cost would depend on the extent of computer reprogramming required and availability of memory. Crew time would be required for periodic realignment of the IMU. Power requirements are relatively high and the additional operating time of C&SM components would somewhat reduce the probability of mission success.

The OSS and RCS system would obtain a local vertical attitude reference from horizon sensors mounted in Sector I and aligned with the experiment sensitive axis. Signal processing, including compensation, pulse modulation and thruster command logic, is provided by the MSC developed OSS. The OSS outputs would interface with the RCS for control accelerations. However, in this case the difference between the coordinate system which would be defined by the sensors and the coordinate system established by the RCS must be considered. The relationship between the sensor reference (R) system and the body (B) system established by the thrusters is indicated in Figure VI.3-1 where superscripts denote the coordinate frame and $X^R = X^B$.

Since the angle B (on Figure VI.3-1) is about 32 degrees the cross coupling is severe and, in addition, the normal horizon sensor output polarity is reversed. To correct this, the system must include resolver circuitry. Also, since a horizon sensor provides pitch and roll attitude information only, a gyrocompass or equivalent technique is necessary for 3-axis attitude control. The resulting system block diagram is shown in Figure VI.3-1. This system overcomes some of the disadvantages associated with the PGNCS but with a weight and possibly a relative cost penalty. A horizon sensor for lunar application is not presently available but a suitable instrument has been under development by the Barnes Engineering Co. If a separate power supply is furnished for the components in Sector I, it would also be necessary to provide isolation between the resolver output and the OSS.

Both these systems would utilize the C&SM RCS for control and therefore it is necessary to evaluate RCS propellant usage rates. In the applicable lunar orbit, the principal dynamic disturbance torque would result from gravity gradient effects. However, the following discussion demonstrates that this effect may be neglected for this case and that RCS propellant may be calculated on the basis of an undisturbed limit cycle.

Taking $\emptyset$, θ and ψ as roll, pitch and yaw respectively and moments of inertia as customarily defined for the C&SM, the gravity gradient torques will appear essentially in pitch if it is assumed that angular displacements are small and that $Iyy \sim Izz$.

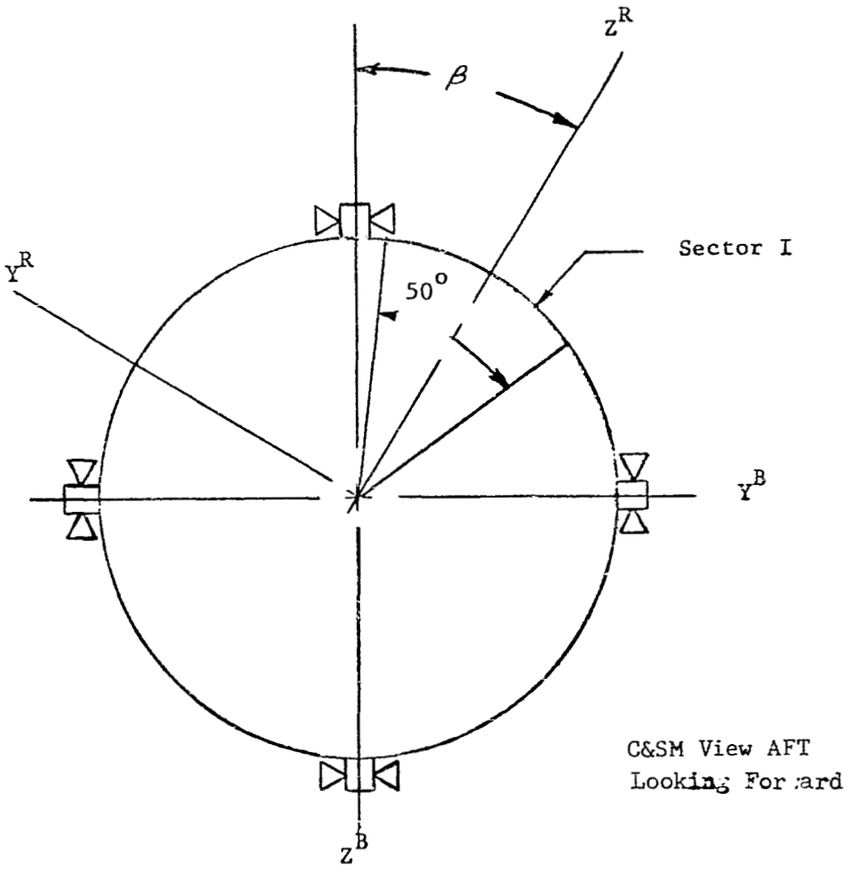


Figure VI.3-1a Orbital Stabilization Coordinate Systems

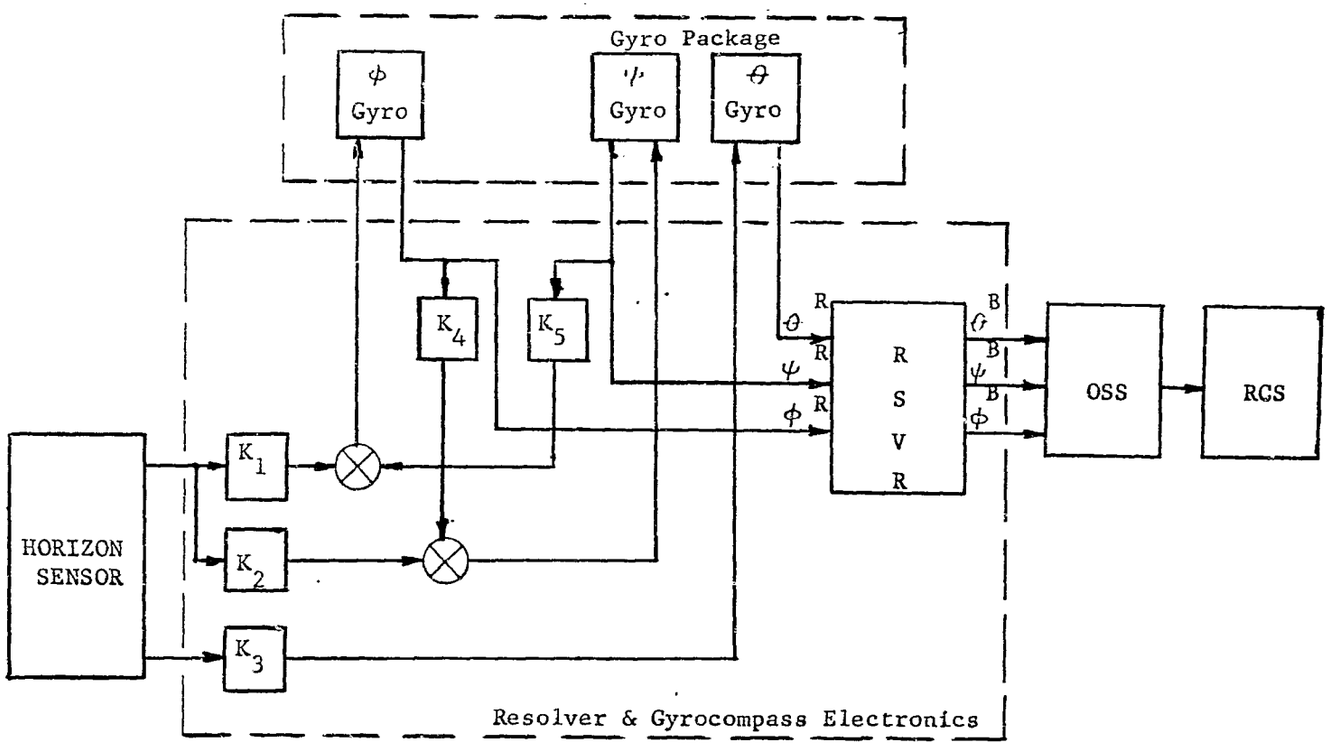


Figure VI.3-1b Orbital Stabilization System and RCS Mechanization

$$\ddot{\theta} = \alpha^2 \theta$$

Where $\alpha = \left[3w_o^2 \left(\frac{I_{zz} - I_{xx}}{I_{yy}} \right) \right]^{1/2}$

The solution to the equation is:

$$\theta(t) = \theta \cosh \alpha t + \frac{\dot{\theta}_o}{\alpha} \sinh \alpha t$$

where θ_o and $\dot{\theta}_o$ are the attitude deadband and the attitude rate produced by a thruster firing respectively.

It is possible to plot propellant usage, W_p, as a function of $\frac{\alpha \theta}{\dot{\theta}}$ and this results in Figure VI.3-2 where a one jet limit cycle has been assumed with a minimum impulse propellant usage, W_{MP_1} of 5.48×10^{-3} lb/pulse. Values obtained directly from the figure would be valid only for a symmetric case with $\dot{\theta}_o$ the same after each impulse although it appears possible to obtain W_p for non-symmetric cases by summing the ΔW_p for segments of a limit cycle. For the study, however, it is adequate to note that the astronaut could re-initialize and eliminate particularly bad cases and inspect the figure for the applicable value of which is approximately 1.8×10^{-3} sec $^{-1}$. Inserting numerical data for θ_o of 0.5 deg. and a $\dot{\theta}_o$ given by:

$$\dot{\theta}_o = -1/2 \frac{M \Delta t}{I_{yy}} = 5.47 \times 10^{-5} \text{ rad/sec}$$

results in:

$$\frac{\alpha \theta_o}{\dot{\theta}_o} = -0.28 \quad \text{and} \quad W_p \theta \cong 0.08 \text{ lb/hr}$$

By contract, the propellant for an undisturbed limit cycle is given by:

$$W_{p\theta} \cong \frac{3600 \, rn^2 \, (I_{min})^2}{4 \, I_{yy} \theta_d \, I_{sp}}$$

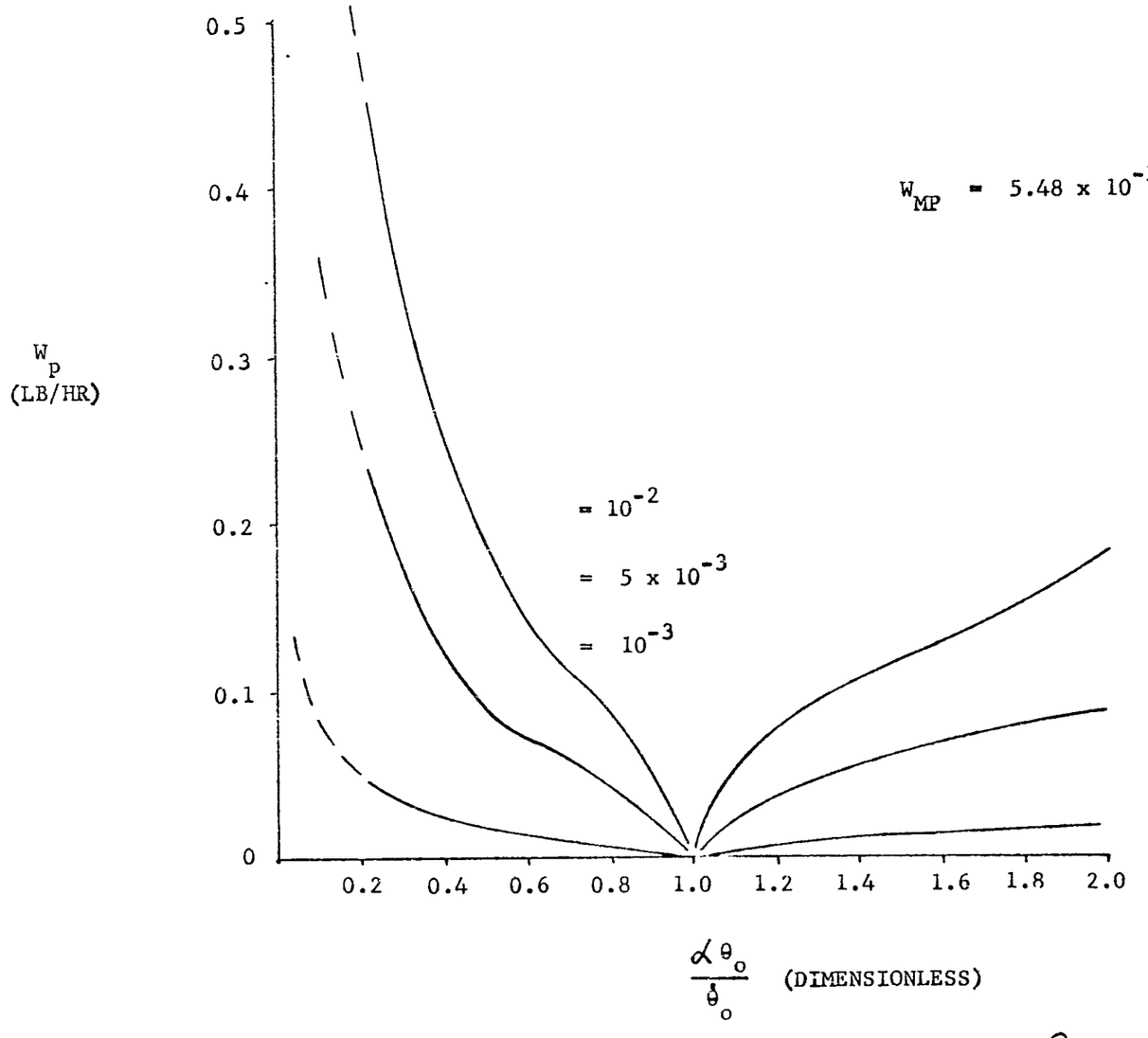


Figure VI.3-2 Propellant Usage as a Function of $\alpha \frac{\theta}{\dot{\theta}}$

where:

r = 6.4 ft., moment arm
n = no. of thrusters fired
I_{min} = 0.93 lb-sec., minimum impulse bit
I_{yy} = 5.43×10^{4} ft-lb-sec^2, pitch mount of inertia
θ_d = 8.73×10^{-3} rad., pitch deadband
I_{sp} = 170 sec., pulse specific impulse

from which:

$w_{P\theta}$ = 0.062 lb/hr.

The total undisturbed symmetric limit cycle propellant is:

$$W_{PT} = \frac{3600n^2 (I_{min})^2}{4\,\theta_d\, I_{SP}} \sum_{i=1}^{3} \frac{r_i}{I_{ii}} \qquad W_{PT} = 0.298 \text{ lb/hr.}$$

A North American Rockwell (NR) report, IL190-320-S & C-68-70, 8 Aug 1968, includes a factor of 1.1 of minimum impulse variations based on simulation results and gives a propellant consumption figure of 0.388 lb/hr. The value computed above closely agrees with the NR value and is considered a good estimate. It is also seen that most of the propellant is consumed by the limit cycle about the roll axis and that the effects of gravity gradient disturbances in the pitch axis are inconsequential for nominal parameter values. However, changes such as a reduction of the deadband to 0.1 deg. could substantially increase pitch axis propellant usage as the nominal operating point would be on the nonlinear portion of the curve.

The third alternative, an independent auxiliary system mounted in Sector I, would also utilize a horizon sensor/gyrocompass system as the attitude reference. Electronics similar in function to the OSS would be necessary to process the sensor signals and provide output commands to the cold gas valves.

A simple block diagram is shown in Figure VI.3-3 where gyrocompassing is assumed but resolver circuitry is unnecessary. The cold gas valves could be oriented to avoid any direct impingement into the field of view of experiment sensors and would not produce combustion product contaminants. It is also possible

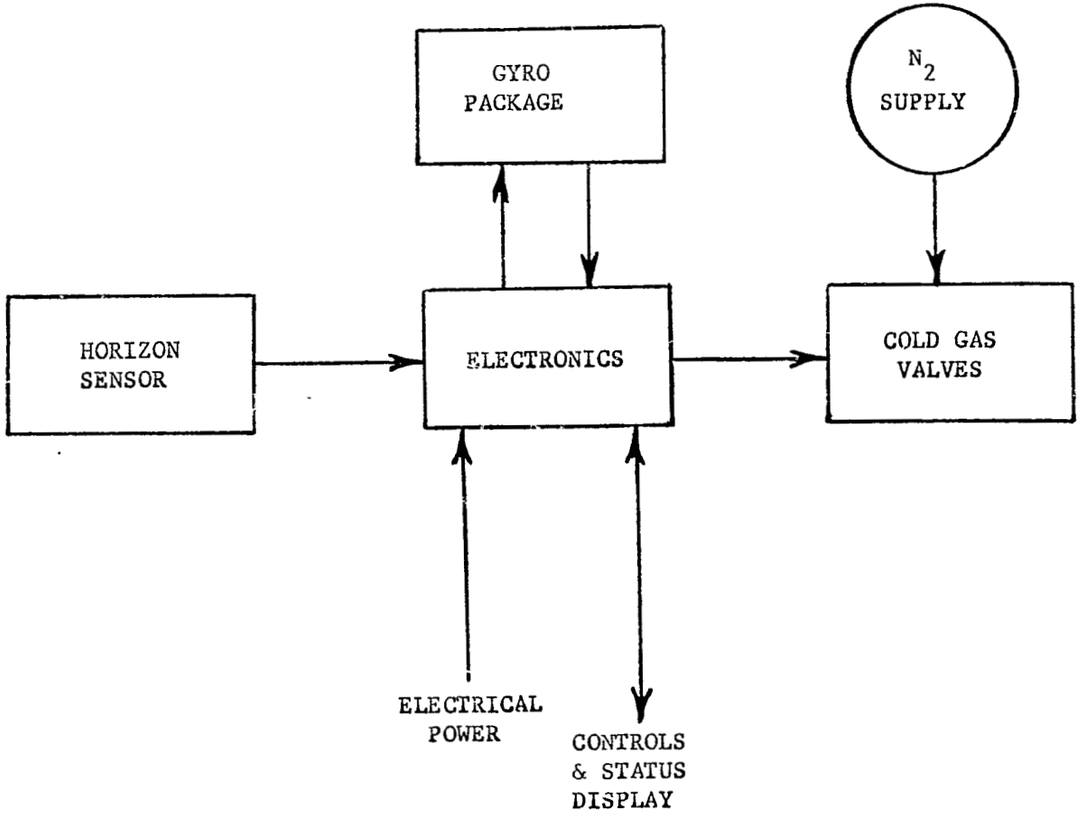


Figure VI.3-3 Auxiliary Command and Service Module System For Lunar Orbital Experiment Orientation

to substantially reduce the minimum impulse with a cold gas system. With a value of W_{MP} = 0.1 lb-sec., extremely low limit cycle rates result and substitution in the preceding equation for W_{PT} gives:

$$W_{PT} = 0.014 \text{ lb/hr.}$$

However, this value and the associated rates are so low that crew motion disturbances could be expected to be the determining factor in propellant usage rates. The low limit cycle rates would make it possible to reduce the attitude deadbands for more accurate pointing without incurring significant propellant weight penalties. An estimate of subsystem add-on weight and power for both the OSS/RCS and auxiliary system is given in Table VI.3-1. The additional component weight of 25 lb for the auxiliary system compared to the OSS/RCS would be partially offset by lower attitude control propellant requirements. However, the total weight penalty of the auxiliary system would exceed that of the OSS/RCS up to the point where sufficient propellant is included in both system weights for 75-100 hrs of operation.

VI.3-2 Subsatellite Mounted Experiments

Spin Stabilized Sub-Satellite

The attitude control study of a spin stabilized satellite reduces to the questions of spin axis stability and spin axis orientation accuracy. Two satellite configurations, Pioneer and AIMP, have been briefly investigated and some pertinent characteristics are presented in Table VI.3-2.

It is well established that stability of the spin axis in the absence of external torques requires that the spin axis coincide with the principle axis of maximum inertia. This is a matter of obtaining the proper mass distribution during design and fabrication and not a question of feasibility. An area requiring further study is the effect of the external torque resulting from gravity gradient dynamics. A quantitative evaluation of this is beyond the scope of the present study but the results cited in a NASA report, NASA CR-433, Effect of Gravity Gradient Torque on the Motion of the Spin Axis of an Asymmetric Vehicle, April 1966, indicate that perturbations could be very small. Further, the magnitude of the torque is a function of the vehicle mass distribution and is, therefore, subject to control.

Table VI.3-1
Command and Service Module Attitude Control Add-on Alternates

	OSS/RCS		Auxiliary Sys	
	Weight	Power	Weight	Power
Horizon Sensor System	22 lb	20 W	22 lb	20 W
3-Axis Gyro System	15	40	15	40
Interface Electronics	10	10	13	15
OSS	5	5		
Gas, Valves, Plumbing, Tank			27	
Total	52 lb	75 W	77 lb	75 W

Table VI.3-2
Subsatellite Characteristics Attitude Control

	Lunar Orbiter	AIMP	Pioneer
Spin Axis Orientation Accuracy	NA	1.6 Deg[A B]	0.3 Deg [A]
Spin Rate	NA	25 $\pm$ 5 RPM	60 RPM
Impulse Capability	8.6×10^4 lb-sec	2.1×10^4 lb-sec	0
Spacecraft and P/L Weight	960 lb	181 lb	140 lb
Attitude Control Propellant	4 lb	0	0.9 lb

A - Estimated
B - Dependent on spin-up and separation mechanisms

The Pioneer system includes a command link, electronics and a single gas jet. This system makes it possible to adjust the orientation of the spin vector about either or both of the two axis orthogonal to the spin axis by pulsing the gas jet at the proper fixed point in the spin cycle. Control is furnished by earth commands and the necessary attitude reference is the received signal strength from the vehicle high gain antenna. The relatively narrow antenna pattern leads to an estimate of 0.3 deg. for the achievable orientation accuracy. This technique has been successfully used on past Pioneer flights although ambiguity in the apparent attitude may require some trial and error pulsing of the gas jet to determine attitude from the resulting response. The Pioneer configuration does not include a propulsion system and would not be capable of performing orbital changes.

The AIMP configuration does include a propulsion system capable of performing a maneuver along the fixed thrust vector axis. However, the AIMP does not include a system for adjusting the spin vector orientation. Thus, the spin characteristics, including precession and nutation of the spin vector, would be determined by the separation and spin-up mechanisms which establish the initial conditions. A paper presented by Lunar Orbit Mission Analysis for the Improved Delta Launch Vehicle and AIMP-D Spacecraft, AIMA 4th Aerospace Sciences Meeting, June 1966, cites three sigma elevation and azimuth flight path angle errors of 1.1 deg. each during engine burn of the spin stabilized stage. Spin-up is achieved on a spin table prior to separation from the preceding stage.

Stable Subsatellite

The Lunar Orbiter Configuration was investigated as an available satellite with an active, non-spinning, attitude stabilization system. This vehicle was developed to photograph discrete, selected portions of the lunar surface. The attitude control system is of course compatible with this particular set of requirements. A simplified block diagram is given in Figure VI.3-4. In operation, the system is inertially stabilized with solar arrays oriented toward the sun. The inertial hold orientation is established by the vehicle-sun axis detected by the sun sensor system and the vehicle-canopus axis detected by the star tracker. The programmer initiates required functions through time reference sequential stored commands or real time earth uplink commands. Maneuvers are performed by summing rate mode gyro output with a fixed "sleeving" voltage to command the

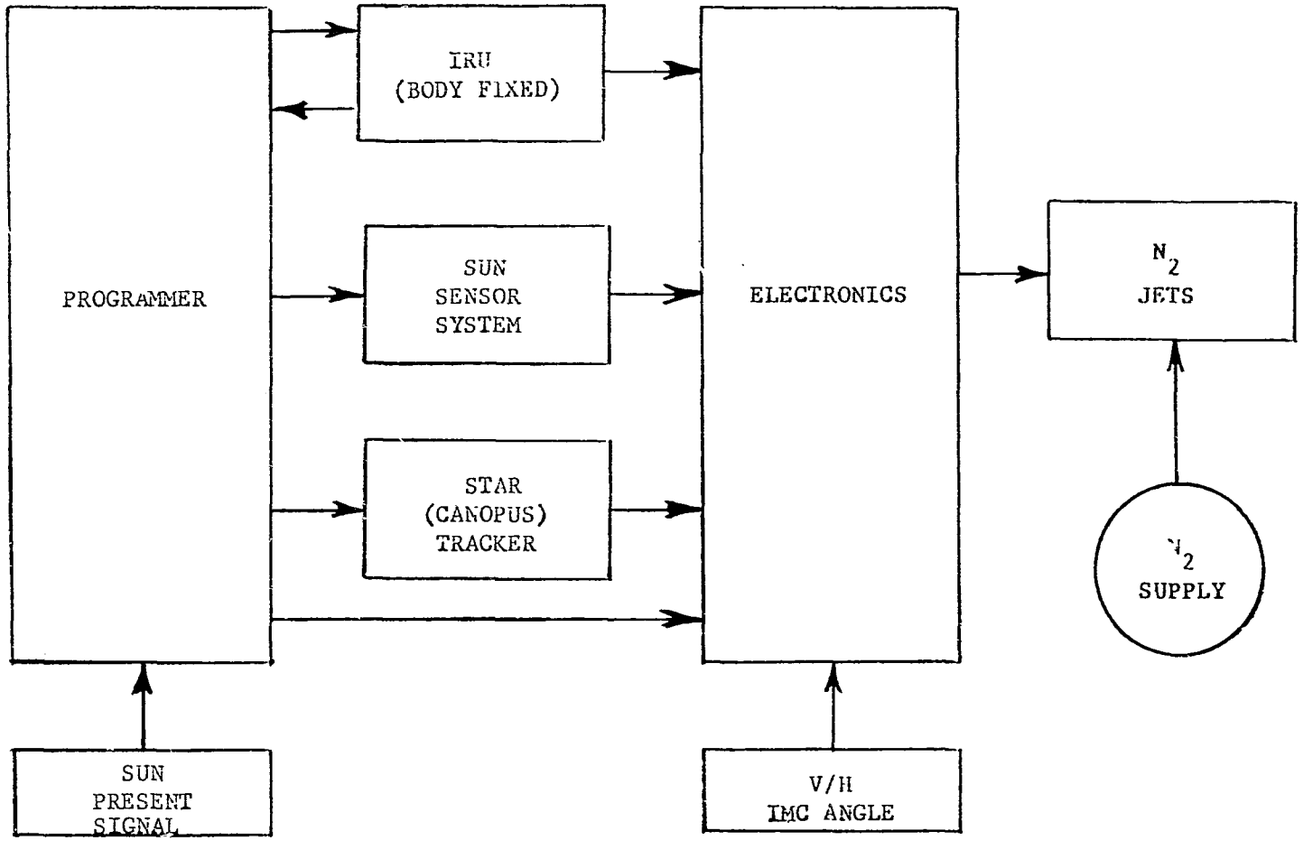


Figure VI.3-4 Lunar Orbiter Block Diagram

gas jets. Maneuver attitude excursion is determined by integration of the rate signal. This maneuver procedure must be followed to orient the cameras to the nadir for a photographic sequence and then reorient to the basic inertial attitude for reacquisition of the sensors.

The various orbital relationships are depicted schematically in Figure VI.3-5 where angular relationships are greatly simplified for clarity. The plane of the earth's equator is, of course, inclined at about 23 degrees to the plane of the ecliptic and the moon's orbital plane is inclined 5 degrees to earth's equatorial plane. The star Canopus is in the region of 50 degrees south latitude. It is adequate for system illustration to consider that the sun (s), earth (E) and moon (M) all lie in the plane of the paper of Figure VI.3-5. Canopus then is roughly along the line to the reader's eye. The low inclination orbit of the Lunar Orbiter mission then also lies roughly in the plane of the paper and is denoted by the circle around M in the position labeled "Launched." Although the sun is occulted by the moon on each orbit, Canopus is continuously visible from the orbit plane and the plane of the orbit does not deviate radically in its relation to the sun during the annual rotation of earth about the sun. In the mission presently considered, however, the lunar orbit plane would be inclined at 90 degrees. This is shown in Figure VI.3-5 by the line through M in the "Launch" position representing an edge view of the orbit plane--assumed for convenience to be about the terminator at this time. Under this condition, Canopus is occulted by the moon on each orbit. Further, in the next position, "3 months" later, the orbit plane has rotated through 90 degrees relative to the sun. Continuing on to "6 months," etc. it is seen that the orbit plane rotates 360 degrees relative to the sun in a year, corresponding to the earth's one year orbital period. As a result, the inertial reference frame established by the lunar orbital sensors is not suitable for a long duration mission in a polar lunar orbit. Further, the sensors establish an inertial frame rather than the moon centered coordinate frame required for continuous experiment operation. Even if each sensor were somehow provided with 360 degrees of angular freedom to maintain the line of sight to the reference source, the computations required to determine the direction of lunar local vertical from the inertial frame would require some general purpose digital computer capability. The programmer in the present system does not have this capability. These considerations consider that it would be necessary to replace the existing Lunar Orbiter attitude reference system. The best alternative would again utilize a horizon sensor system to provide a closed loop lunar surface attitude reference. The resulting system would be essentially as previously shown in Figure VI.3-3, although further

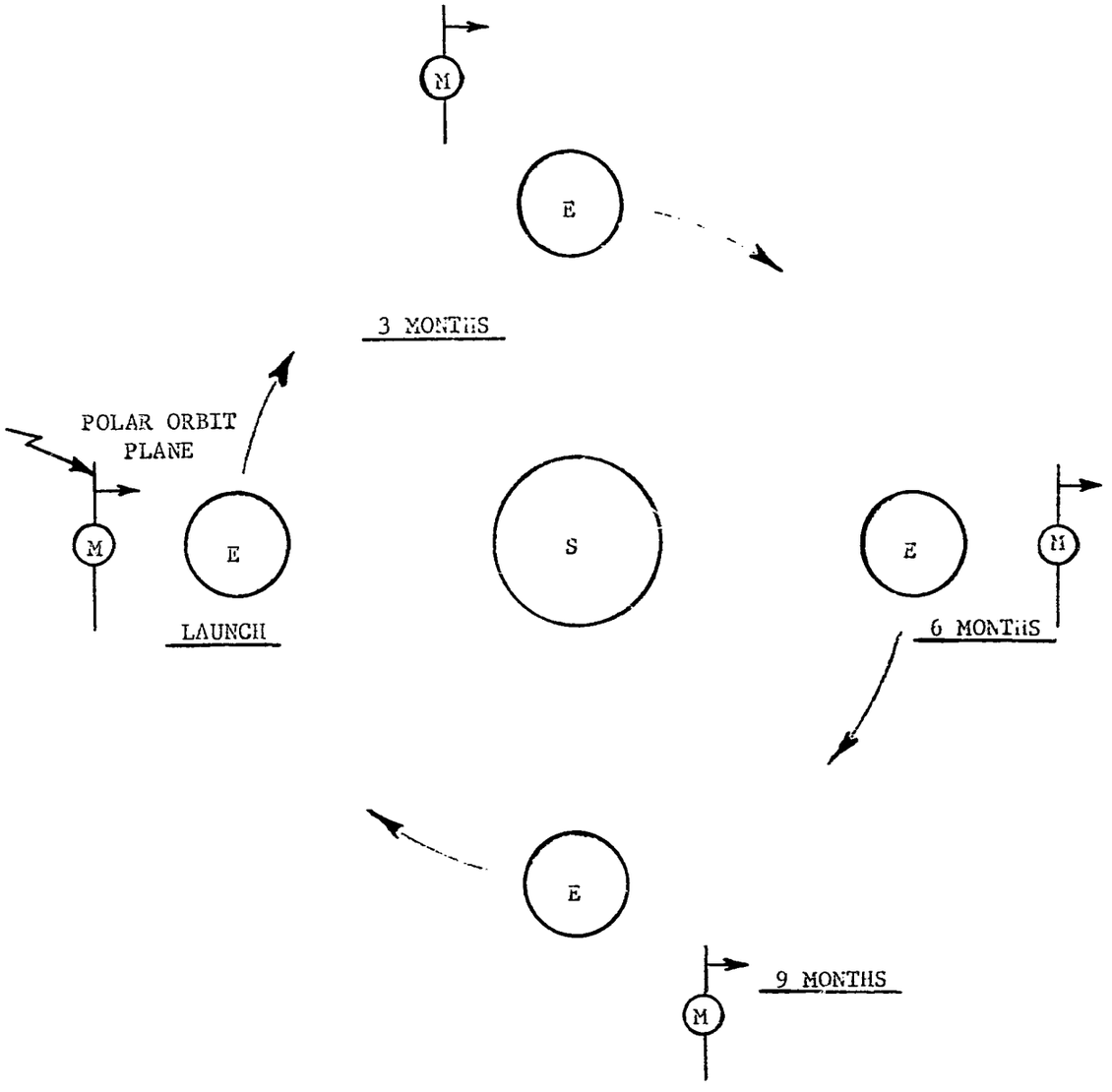


Figure VI.3-5 Lunar Orbital Relationships

studies could be expected to lead to the addition of momentum storage device with the cold gas system used for desaturation only.

Figure VI.3-5 also shows a problem with respect to the present, fixed, solar arrays. To provide the anticipated electrical energy requirements for a 30-day mission in a polar orbit with modes in the region of the earth-moon line it would be necessary to provide a single degree of freedom gimbal system for the solar arrays. The axis of rotation would be normal to the vehicle orbit plane. For long duration missions it would be necessary to add a second degree of freedom with the axis of rotation parallel to the orbit plane. Due to the slow rate of rotation required about this second axis it could be mechanized with discrete changes in response to earth commands.

VI.4 THERMAL CONTROL SYSTEM - Thermal control systems for three subsatellite candidates and the C&SM were reviewed and compared to experiment requirements for several sample groupings generated during the study. The impact on the thermal control systems of the four carriers is as follows:

1. The first and third C&SM groupings shown in Table VI.4-2 can be passively controlled, with the exception of the laser altimeter which requires its own cooling system (or interface with the carrier). A second C&SM grouping also shown in Table VI.4-2 represents a far greater heat load and will require an active coolant system.

2. Three experiment groupings were analyzed for a stable satellite (Lunar Orbiter). Two of the groupings imposed no stringent thermal control requirements on the satellite systems. However, the third grouping required an active coolant system in addition to the passive Lunar Orbiter system.

3. Two spinning satellite experiment groupings were analyzed and compared to the thermal control systems of the Pioneer and AIMP satellites. The existing passive thermal control systems appear adequate for these groupings.

The analysis results show a great dependence of the thermal control systems of the carrier vehicles on experiment groupings selected.

VI.4-1 Existing Thermal Control Configurations - The existing thermal control systems on each of the three subsatellite candidates are presented in Table VI.4-1 and briefly discussed in the following paragraphs.

Anchored Interplanetary Monitoring Platform (AIMP) - The AIMP satellite utilized a passive thermal control subsystem (TCS) which consisted of a fiberglas heat shield located between the spacecraft, and the retromotor, multi-layer insulation blankets, and thermal control coatings on the internal components and external surfaces. The multi-layer insulation blanket consists of between 20 and 30 layers of three mil mylar aluminized on one side as the outer shield and 1/4 mil mylar aluminized on both sides as the inner shield. The small packages (less than 1 ft^2) within the satellite were protected with 20 to 30 layers of crinkled (NRC-2) with aluminum vapor deposited on one side only. Larger components (more than a few ft^2) utilized 20 to 30 layer mylar shields with either dacron or nylon separators between the

Table VI.4-1
Thermal Control Requirements for Satellite Missions

Parameters	Satellites		
	AIMP	Lunar Orbiter	Pioneer
System	Passive	Passive & Heaters	Passive & Active
Heat Loads, BTU/Hr			
Orbit	170	1300 to 1400	290
Thermal Barrier	Heat shield between spacecraft/retromotor (fiberglas) Multilayer insulation blankets	Silicone oxide on aluminized mylar 3 layers dacron 3 layers aluminized mylar	Multi-layer insulation blankets
Coatings			
Internal	Components painted black		
External	White paint α = .363, ϵ = .85 Cat-A-Lac black paint α = .95, ϵ = .87 Buffed aluminum α = .15, ϵ = .04	Silicone base paint with zinc oxide Pigment ϵ = .9, α = .22	
Temperatures, ^{o}F			
Prelaunch			77 $\pm$ 5
Orbit			
Internal	41 to 122^{o}F	35 to 85	30 to 90^{o}F
External		30 to 425	

either dacron or nylon separators between the aluminized mylar shields. The internal components within the satellite were coated with Cat-A-Lac black paint, while the external surfaces (top and side) were primarily coated with Dow Corning's white paint. The optical properties of the coatings are respectively α = .95, ϵ = .87, and α = .363 and ϵ = .85.

Post flight data evaluation revealed that the increase in temperature of certain components was due primarily to absorptivity changes, which increased by a factor of two. A major contributor to the absorptivity change was contamination of the white painted surfaces by the exhaust products from the retromotor.

NASA-Goddard is presently eliminating any use of white painted surfaces and will utilize teflon coated with either vapor deposits of aluminum or silver. Future flights are also considering Kapton instead of mylar, and the application of heat pipes.

The average heating load to be dissipated on these satellites was 170 btu/hr.

Lunar Orbiter - The Lunar Orbiter satellite utilized a passive thermal control system and mounted electric heaters on the critical components. A heat shield barrier and multi-layer insulation blankets, consisting of 3 layers of aluminized mylar with layers of dacron separators were used. The outer mylar shield was coated with silicone oxide. The equipment mount deck surfaces were also coated with white paint, containing zinc oxide pigment and a silicone base. The optical properties were absorptivity, α = .22, and emissivity, ϵ = .9.

Post flight data evaluation showed that the internal temperatures frequently exceeded the limits of 108°F. The high temperatures, as for those of the AIMP, were attributed to degradation of the external coating absorptivity property.

To stabilize the temperatures during orbit, the attitude control subsystem was utilized to change the vehicle sun line attitude long enough to reduce temperatures.

The internal heating loads ranged between 1300 and 1400 btu/hr.

Pioneer - The Pioneer spinning satellites utilized primarily a passive thermal control subsystem, which consisted of multilayer insulation blankets. Some of the vehicles had provisions for an active system in the platform area, which consisted of louvers individually actuated by bimetallic springs, which sensed the temperature of the equipment platform in the region of the louvers.

VI.4-2 C&SM Thermal Control System Analysis

Three experiment groupings were investigated. The first two were given to North American Rockwell early in the study; the third grouping was generated later. All three groupings should be considered as representative samples of potential C&SM experiments and are listed in Table VI.4-2 for convenience. Allowable temperature limits for all experiments are given in Table III.4, Experiment Requirements Summary.

Three typical duty cycles were generated based on the groupings shown in Table VI.4-2. These are shown in Figures VI.4-1, VI.4-2, and VI.4-3 for groups I, II, and III respectively. The average heating load for the first C&SM experiment grouping (Table VI.4-2) is 715 btu/hr, with a peak of 2700 btu/hr for a duration of 1 hour. The peak load is due primarily to the operation of the Panoramic camera. However, the camera has sufficient thermal capacity to absorb the load and maintain its components within their allowable temperature limits. The laser altimeter experiment appears critical for continuous operation, and a cooling system is recommended for this experiment only.

For the second experiment grouping, the average internal heating load to be dissipated is 1260 btu/hr, and peak loadings between 2330 and 2430 btu/hr will be generated. The average 1260 btu/hr could be dissipated by the present Apollo Block II C&SM coolant loop and environmental control subsystem (ECS) radiator configuration. The radiator has a maximum capability of approximately 8425 btu/hr with an effective total radiator area of 56.0 ft^2. The mission orbital requirements for the C&SM are approximately 6500 btu/hr, leaving a margin of 1925 btu/hr. However, to avoid extensive requalification of the C&SM environmental control system due to experiment interface, use of a separate experiment coolant loop is recommended. A schematic of a TCS capable of dissipation of the average heating load is shown in Figure VI.4-4. The coolant loop would utilize the fluid E-1 at a coolant flow rate of 400 lbs/hr. As shown, the system would include a by-pass arrangement, temperature control valve and two of the present Apollo Block II LM pumps. The radiator size required would be 21.0 ft^2. Cold plates are probably required for six experiment components. These are: (1) IR Mapping Radiometer, (2) MW Radiometer, (3) Metric Camera, (4) RF Reflectance, (5) Radar Altimeter, and (6) Radar Imager. The total weight of the system would be 169 pounds.

TABLE VI.4-2 C&SM EXPERIMENT GROUPINGS

GROUP I	GROUP II	GROUP III
Metric Camera	IR Mapping Radiometer	Metric & Stellar Camera
Stellar Camera	IR Spectrometer	Panoramic Camera
Panoramic Camera	Electronics	Photoelectric Photometer
Photoelectric Photometer	MW Radiometer	Laser Altimeter
Laser Altimeter	Metric Camera	RF Transponder
RF Transponder	Stellar Camera	Solar Wind Foil
Solar Wind Foil	RF Reflectance	
Extender	Bistatic Radar	
Foil	Radar Altimeter	
Geochemistry Expts.	RF Transponder	
Gamma-Ray	Radar Imager	
Neutron Albedo	Antennas	
Gamma Particle	Data Processor	
X-Ray Sensors	Trans/Rec.	
Solar X-Ray	Film Record (Optional)	
Cosmic Ray	U.V. Imager	
Electronics	Imager	
	Film Recorder	

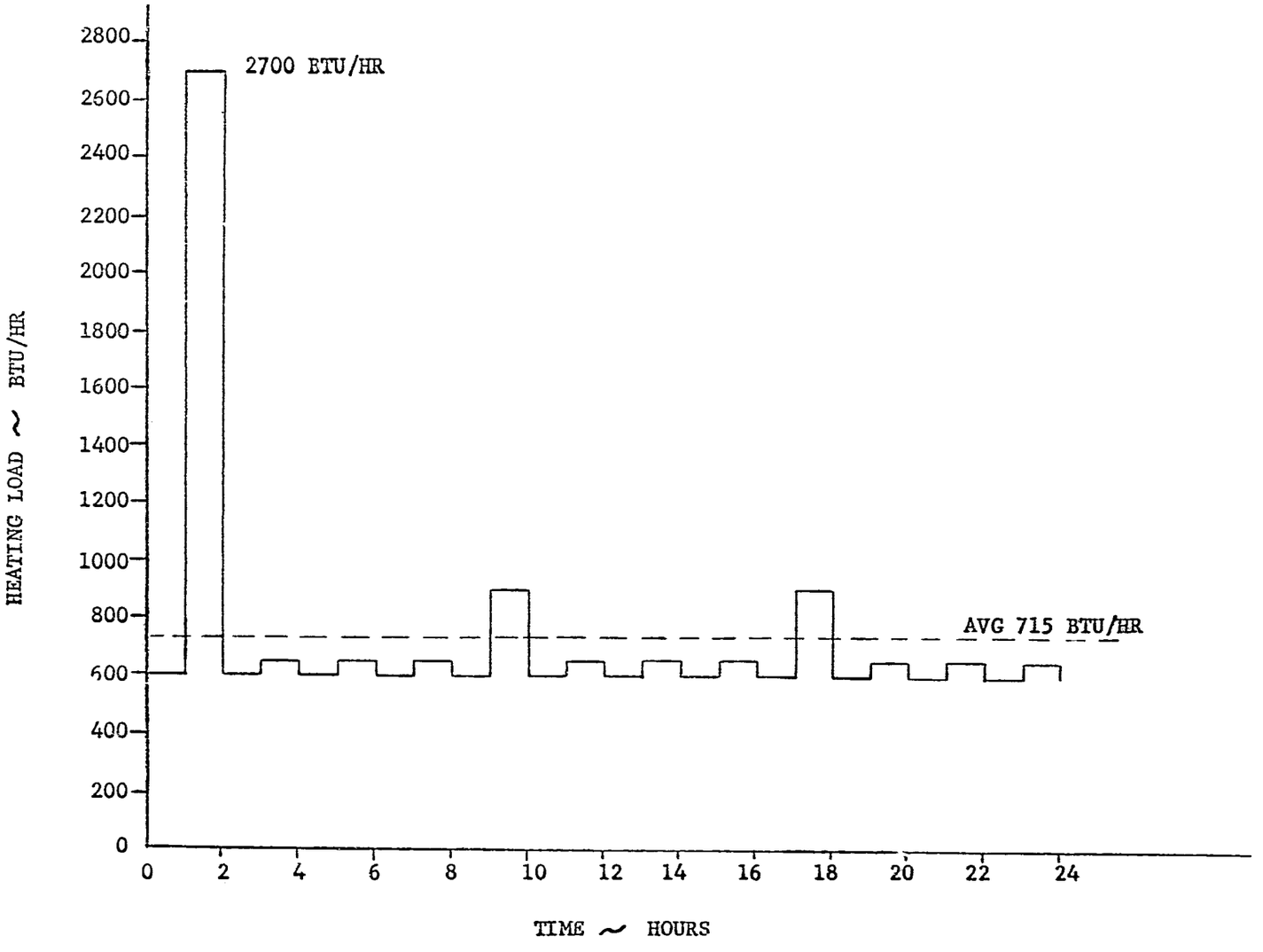


Figure VI.4-1 Heating Loads - C&SM Group I Experiments

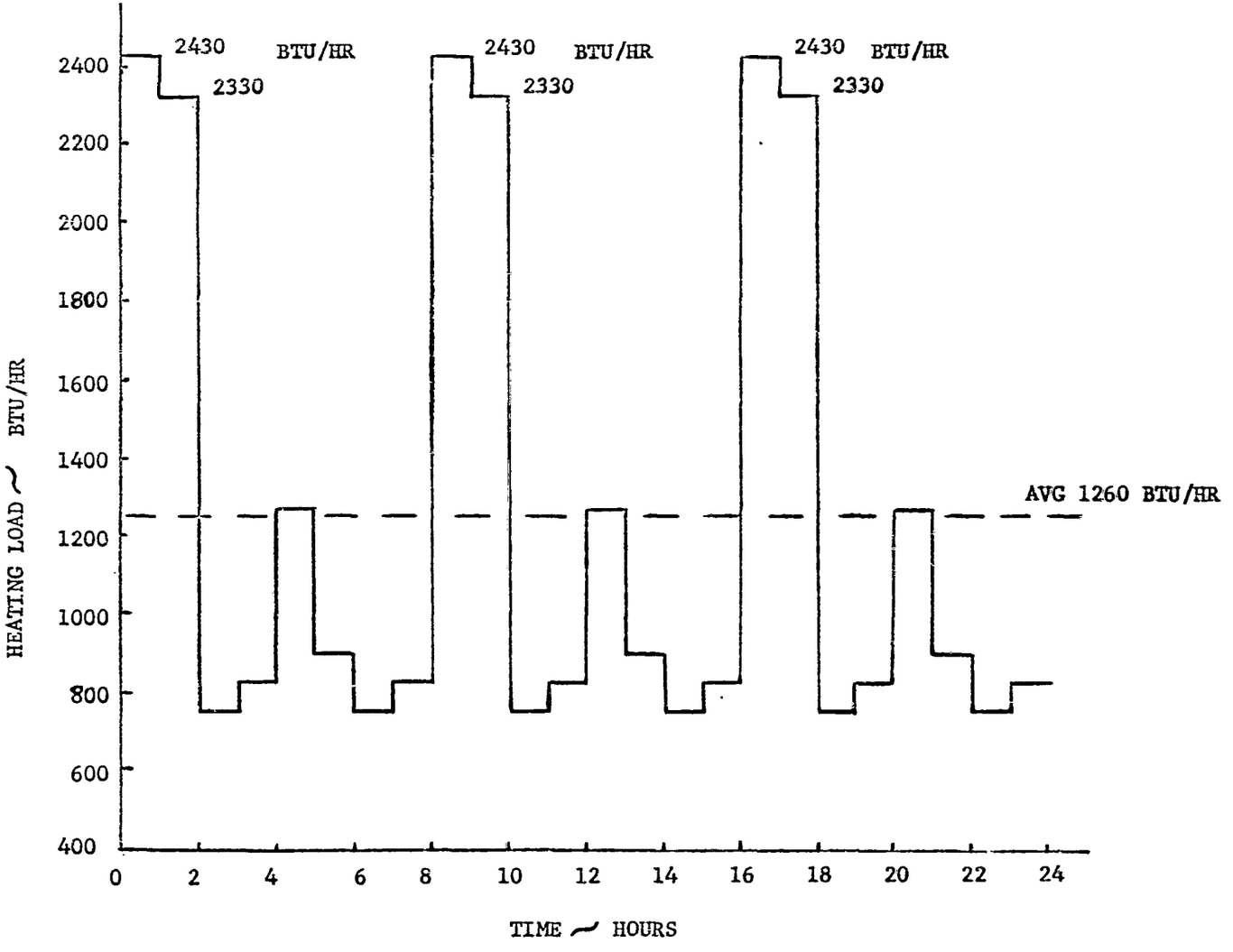


Figure VI.4-2 Heating Loads - C&SM Group II Experiments

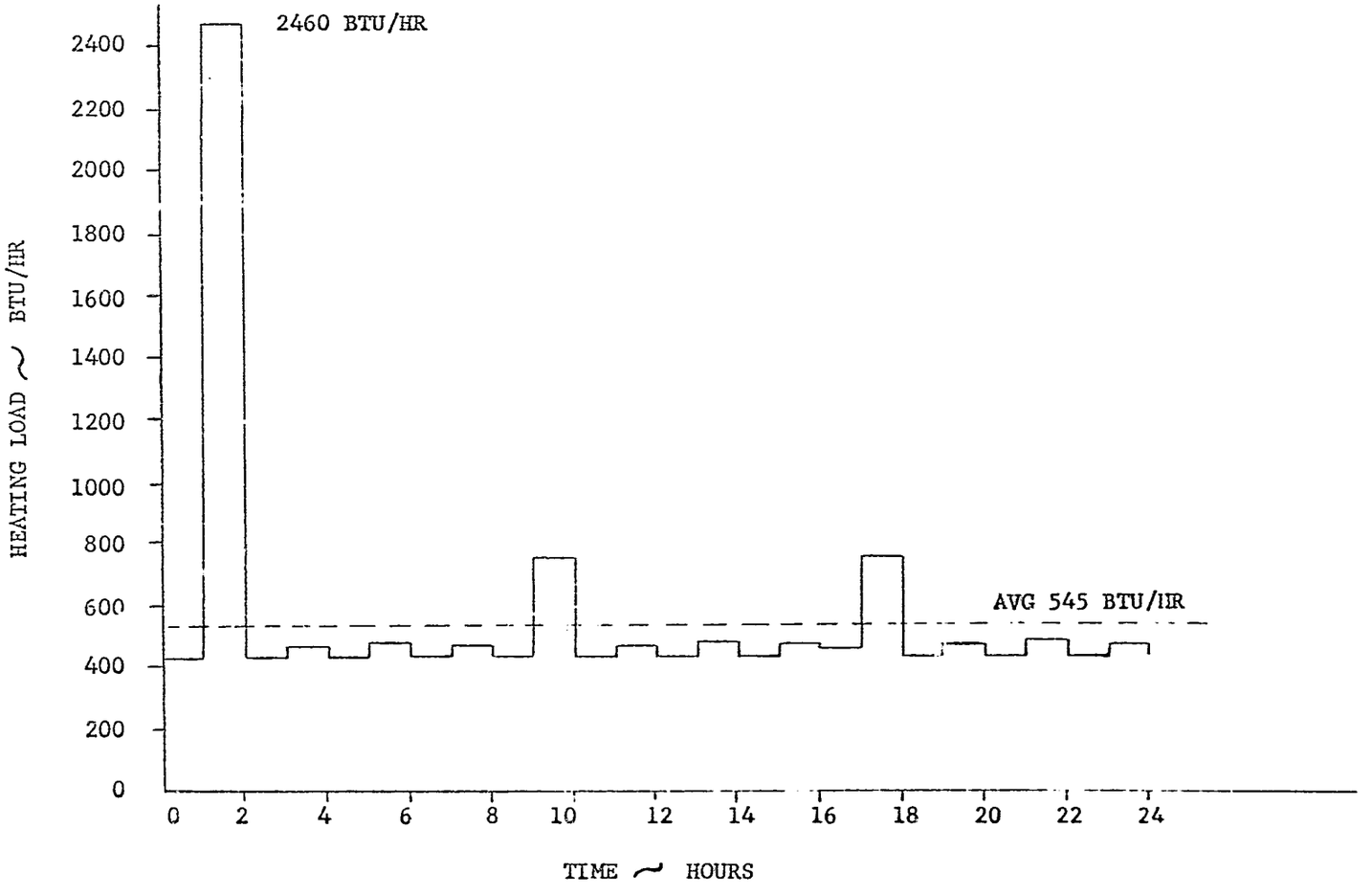


Figure VI.4-3 Heating Loads - C&SM Group III Experiments

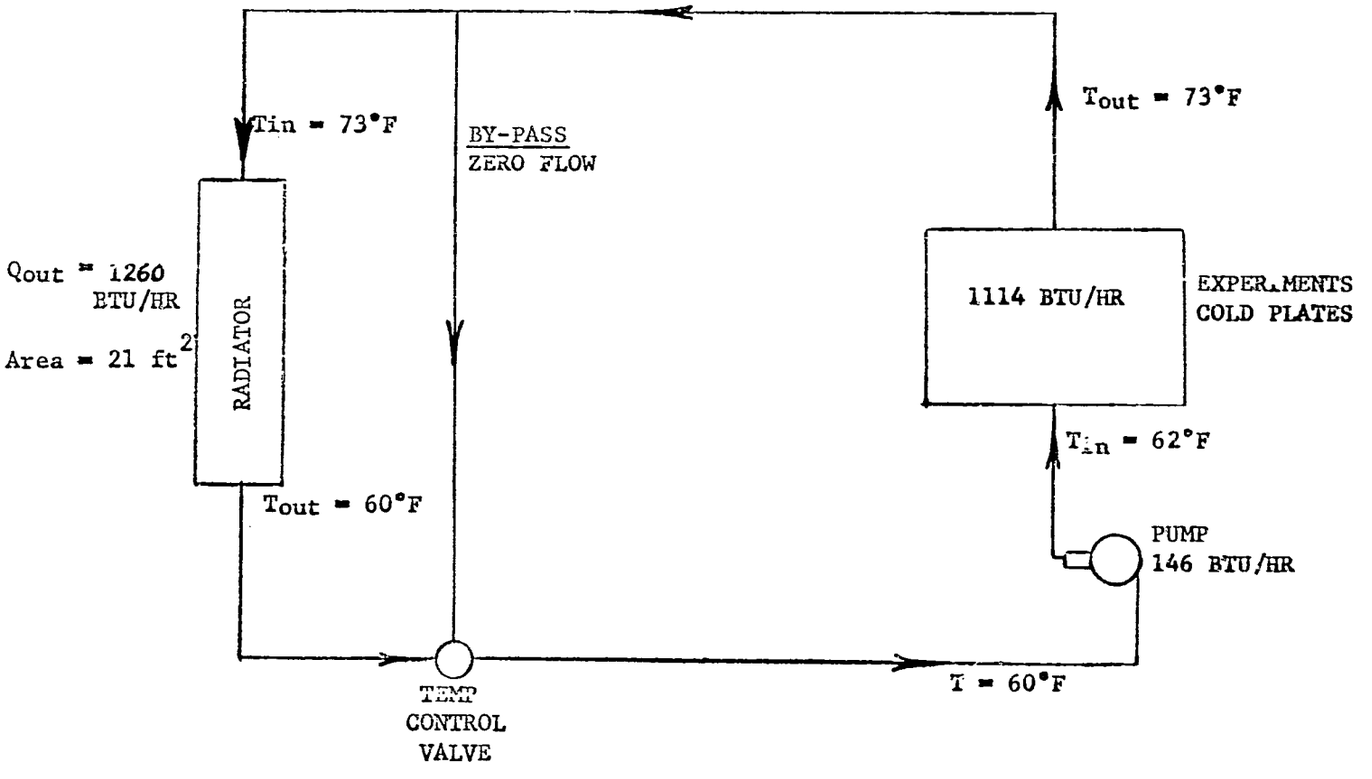


Figure VI.4-4 Maximum Heating Load Condition - C&SM Group II Experiment Coolant System

The IR mapping radiometer is seen as a thermal control problem, since the experiment requirements specify that its optical mirror be maintained at 150°K by radiation techniques. This places a requirement that the experiment radiator views deep space during experiment operation. The orbital environment otherwise will not enable this temperature level to be maintained. The average heating load for the third C&SM experiment group is 545 btu/hr, with a peak load of 2460 btu/hr when all experiments are in operation. Operation of the experiments of this grouping should not require interface with the C&SM thermal control system with the possible exception of the laser altimeter. Its continuous operation will cause a component heating problem. As in Group I, a separate cooling system should be considered for this experiment. All experiment and subsystem components should receive coatings similar to those of the Lunar Orbiter satellite components, with good thermal conduction paths to the mounting structure.

VI.4.3 Stable Satellite (Lunar Orbiter) Thermal Control Analysis - Three candidate experiment groups were investigated for use on a stable subsatellite. The groupings are presented in Table VI.4-3.

Table VI.4-3
Stable Satellite Experiment Groupings

Group I	Group II	Group III
Geochemistry	Geochemistry	IR Mapping Radiometer
Mass Spectrometer	Mass Spectrometer	IR Spectrometer
Radar/Laser Altimeter	RF Transponder	MW Radiometer
RF Transponder/ Laser	Meteoroid	RF Reflectance
Retroreflector		Bi-Static Radar
EM Measurements		Radar Altimeter
Noise Survey		RF Transponder
Gravity Gradient		EM Measurements
Meteoroid		Noise Survey

Heating loads for the first two stable satellite groupings are as follows:

Heating Loads (btu/hr)

	Group I		Group II	
	Min	Avg	Min	Avg
Experiments	0	760	0	314
Subsystems	324	324	198	198
TOTALS	324	1084	198	512

These heat loads are compatible with the Lunar Orbiter passive thermal control system. Therefore, with the exception of local insulation, no changes are required.

The third stable satellite experiment group presents a greater thermal control problem. Figure VI.4-5 shows the heating loads for this case. An active coolant loop circuit is required to dissipate these heating loads. A schematic of a typical system is shown in Figure VI.4-6. The coolant fluid recommended is E-1 at a flow rate of 600 lbs/hr. The coolant flow is divided equally to the subsystem and experiment components. The radiator size required to dissipate the average 1840 btu/hr to space is 30.0 ft^2. The size of this radiator poses a significant integration problem. Cold plates are probably required on five experiment packages. These are:

1. IR mapping radiometer
2. MV radiometer
3. RF reflectance
4. Radar altimeter
5. Electromagnetic measurements

Also, some subsystem components will use cold plates. The total weight of the TCS is estimated as 192 lbs., including 20 lbs. of multi-layer insulation blanket.

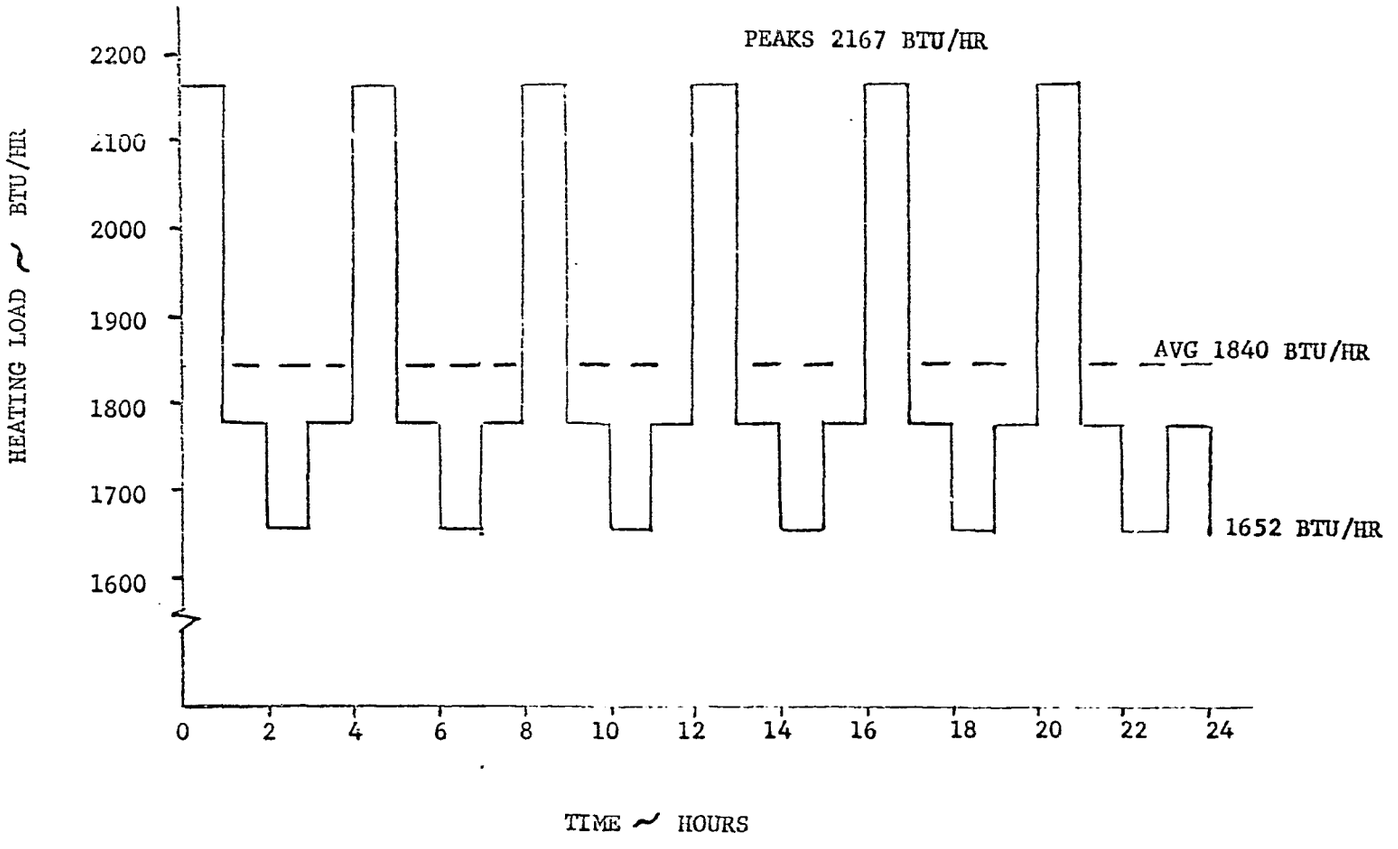


Figure VI.4-5 Heating Loads - Stable Satellite Group III Experiments

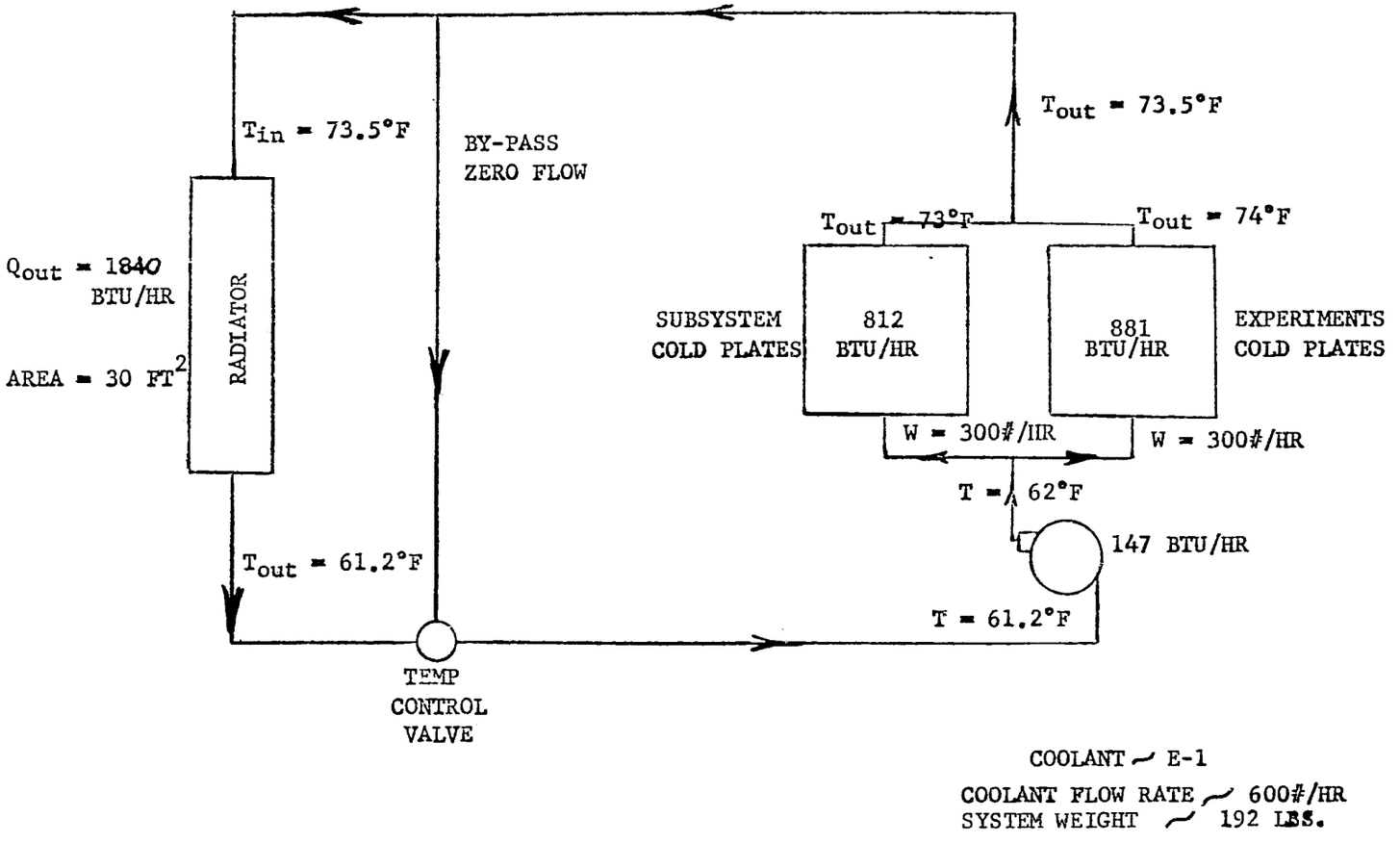


Figure VI.4-6 Maximum Heating Load Condition - Stable Satellite Group III Experimer c Coolant System

VI.4-4 Spinning Satellite (AIMP) Thermal Control Analysis - Two candidate experiment groups were investigated for the spinning satellite missions. These are listed in Table VI.4-4.

Table VI.4-4
Spinning Satellite Experiment Groupings

Group I	Group II
Magnetometer (D.C.)	Magnetometer (D.C.)
Magnetometer (A.C.)	Magnetometer (A.C.)
Plasma Probe	Plasma Probe
Electric Field	Electric Field
Low Energy Particles	Low Energy Particles
Solar Energetic Particles	Solar Energetic Particles
Meteoroid	RF Transponder
RF Transponder/Laser	
Retroreflector	

The average heating loads for Groups I and II are respectively 135 and 279 btu/hr. These heating loads are within the capabilities of the passive thermal control techniques applied on the AIMP satellites.