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EXPERIENCE APPLICABLE TO THE VIKING LANDER FROM A STUDY OF RELATED SPACE FLIGHT PROJECTS

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AUTHORS' FORWARD

This work was performed in support of the Viking Lander during the time when the Viking Mission was rescheduled from a 1973 Launch to the 1975 Launch Opportunity.

The changes to the Viking Lander and the Viking Flight Mission have been kept current to the time of publication, and thus the publication date establishes the baseline configuration for the comparisons contained in this report. Modifications to both the Lander and Flight Mission will continue, however they are not expected to have any major effect upon the application of data presented within the report.

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March 25, 1971

EXPERIENCE APPLICABLE TO THE

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FROM A STUDY OF

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DEFINITION OF TERMS

Within the context of this study the general terms listed below are used as defined:

COMPONENT: A combination of parts, devices and structure, usually self-contained, which performs a distinctive function in the operation of the overall equipment. A "Black box." In the course of a Project, qualification testing, replacements within a spacecraft and provisioning for flight spares are performed at the component level.

DEVICE: Electromechanical or mechanical items which perform a specific function and are intermediate in complexity between piece parts and components. Examples are: valves, small motors, gyros, solenoids.

FAILURE/ANOMALY: The inability of an article to function as intended or within its specified tolerance per the applicable documents. For this study these are the formally documented incidents which have occurred during ground test or flight operation.

FLIGHT ACCEPTANCE: A series of operating tests and environmental exposures performed on a flight item to determine compliance with acceptance requirements.

PART: One piece or two or more pieces joined together which are not normally subject to disassembly without destruction of design use. For this study parts are considered as elements of electrical circuits.

QUALIFICATION: A test or series of tests conducted to determine whether an item of hardware meets qualification requirements. For this study, qualification infers completing a series of operational tests performed in conjunction with environmental exposures.

SUBSYSTEM: A subsystem consists of one or more interconnected components and performs a major operating function within a spacecraft.

ABBREVIATIONS

AGE: Aerospace Ground Equipment. These are the electrical and mechanical test equipment employed in ground operation of a spacecraft.

DSN: The deep space communication network.

ETR: The Eastern Test Range, Cape Kennedy, Florida.

RTG: Radioactive Thermoelectric Generators. A unit which employs the heat of radioactive decay to generate electricity by thermoelectric phenomenon.

WTR: Western Test Range, California.

1.0 SUMMARY

This report documents a study undertaken to identify actual experience contained in recent space programs having features related to the Viking Lander. The features considered were mission science operation, long life, system complexity, atmosphere entry and RTG operation. The study does not include rocket engine terminal descent, inertial guidance, or extended on-planet operations. The data is intended to provide a baseline to support evaluation and planning for the Viking Lander Project.

Four space programs were selected for the study: Lunar Orbiter, Mariner 69, Biosatellite, and Nimbus B. Data from the four programs have been collected and ordered in detail sufficient to define the flight and ground failure/ anomalies, hardware and mission configurations, and test programs. The data has been processed into tables, charts, graphs and analyses in depth sufficient to establish the cause-effect correlations between space program activities and potential flight problems. The specific direction of the study has been toward early identification of potential problems for the Viking Lander, however the data presented in this report could prove useful for space projects in general.

2.0 INTRODUCTION

This report describes a study undertaken at the direction of the Viking Project Office of the NASA Langley Research Center. The study consisted of compiling available data from recent space projects, identifying the cause and effect of problems, and then making an assessment for the potential impact of these problems upon a new space project. For this study the data was taken from the Lunar Orbiter, Mariner '69, Biosatellite, and Nimbus B programs and applied to the Viking Lander Project.

All four of the completed projects achieved major successes, yet each suffered flight anomalies which resulted in some data loss and required in-flight work-arounds. The repetitive nature of a number of flight problems and the occurrence of unexpected problems are indications of less than complete understanding of the design requirements and operational environments. The flight problems also suggest that improvements are in order for the controls employed throughout design, manufacture, test and flight operations.

2.1 Expected Application of the Report

This report has been prepared to identify the actual experience of past space programs having features related to the Viking Lander in order to provide a baseline for evaluating program activities and design features (mission and hardware) in the Viking Lander program. While the attention of the study activities has been directed to application for the Viking Lander, this report should prove useful to support space programs in general.

This report will support planning and trade-off decisions by providing material to judge the cause-effect correlation between program activities and flight problems in the following areas:

(a) Design - Previous design problems, failure modes, persistent problems, limitations of advanced hardware, effects of late modifications, and etc. have been identified and reviewed to support future design and assurance activities.

(b) Test Program - The past test programs have been identified and reviewed relative to flight results to provide data for evaluating future test programs.

(c) Assurance Activities - Failure modes and classes of risk have been identified and evaluated in terms which can be correlated with the effectiveness of the applicable program controls.

2.2 Selection of Space Programs for Study

Four space programs were selected for this study on the basis of their ability to contribute space program experience pertinent to the Viking program. The criteria governing this selection consisted of:

- Configuration - Resemblance to features of Viking Lander mission and hardware configuration.

- Management - Programs originating with several NASA centers and performing contractors, representing different approaches to a successful space program.

- Complexity and Life - Heavy units with complex interacting subsystems and long flight duration.

- Maturity - Projects including both single mission and continuing projects, representing different opportunities for growth.

The comparison of the Viking Lander and the selected programs are shown summarized in Tables 2-1 and 2-2. The following subsections discuss the features of these programs.

2.2.1 VIKING LANDER - NASA-LRC Management, MMC Contractor

The Viking mission includes requirements for long life, multi-vehicle mission operation, complex science powered from a nuclear source, entry, and deceleration into an atmosphere, a "soft" landing on Mars and a 90-day period of science and engineering measurements. Many of Viking's general mission requirements are common within the four space flight programs selected for analysis in this study. Their characteristics are summarized in Tables 2-1 and 2-2. The following features of the Viking Lander contributed to their selection:

Mission Phases - Space cruise, entry, landing, and surface operations.

Hardware Elements - Science, RTG, guidance and control, programmer-computer, communications.

Complexity - Weight, number of microcircuits, parts count, devices, complex interacting subsystems.

Life - Fourteen months dormancy in space, followed by planetary entry, landing, and 90 days surface operations.

2.2.2 LUNAR ORBITER - NASA-LRC Management, Boeing Contractor

This program represented an example of a short term project response (2 years from contract start to first launch) for hardware design, fabrication, test, and flight. The Lunar Orbiter spacecraft was designed to operate for a 30-day

Lunar photographic primary mission followed by reduced level operation for up to one year. It accomplished this objective and culminated in 5 flights (1966-1967), as planned. The spacecraft employed many mechanical and electro-mechanical devices. Large quantities of microcircuits were contained within the Programmer-Computer which was commandable and reprogrammable with both real time and stored commands. Significant relative to Viking, Lunar Orbiter flights were conducted with simultaneous operation of more than one spacecraft.

2.2.3 MARINER 69 - JPL Management and Contractor

Mariner 69 provided two successful flights (Mariner 6 and 7) from one launch window in 1969. These flights of the Mariner series offer the unique instance of an ongoing program incorporating major design changes to the flight spacecraft systems. The inclusion of the digital Central Computer and Sequencer into the 69 configuration introduced the use of more than 2000 microcircuits into the spacecraft subsystems. Moreover, the Mariner Spacecraft relates to Viking because of successfully accomplishing a Martian mission and because of the wealth of technical and operation background at JPL.

2.2.4 BIOSATELLITE - NASA-Ames Management, General Electric/RESD Contractor

The Biosatellite program consisted of two phases with a degree of overlap. The initial "3-day" (Biosatellite I and II) configuration culminated in two flights of nominally three days (1966-67). In this phase the spacecraft heat shields, attitude control, command, telemetry, deorbit and recovery capability were developed. The "Primate" configuration (Biosatellite III) was designed for 30-day flight (1969) and added fuel cell power, life support, and active thermal control. The Biosatellites were automated orbiting biological laboratories with all the resulting systems and operational complexity. Some of the precision hardware capabilities realized in flight included such diverse items as acceleration rate control of less than 10^{-5} earth g's; controlled gamma ray irradiation of live biologicals in flight; psychomotor games played by a primate in flight; and real time telemetry of brain waves (electro-encephalograms). Of the four spacecraft studied, only the Biosatellites made controlled atmospheric reentries from orbit.

2.2.5 NIMBUS B - NASA-Goddard Management, General Electric/SD Contractor

Nimbus is another example of a continuous program with numerous evolutionary changes incorporated throughout its history; Nimbus flights are used for R&D testing and operation of weather satellite (meteorological) equipment and other related experiments. In particular, the B spacecraft payload consisted of a camera, interferometer spectrometer, infrared spectrometer, UV solar energy monitor, and an experiment for locating remote terrestrial sensors. The Nimbus systems required complicated programming to operate multiple experiments and record and play back their data. Nimbus B carried two SNAP 19 RTG power sources as experiments to evaluate their performance and reliability. The SNAP 19 RTG power source is essentially the same type Viking will employ. Nimbus B also carries two sets of tape recorders, one set similar to that proposed for Viking. Nimbus equipment has operated continuously for 18 months in space and approximately equals that proposed for the Viking Lander.

2.3 Study Approach

The study included collection, processing, and evaluation of project data as described in the following two paragraphs.

2.3.1 Data Collection

The following types of data were gathered for each of the four programs studied:

(1) All Failure/Anomalies in Flight - Identifying failure description, cause, effect, date of occurrence, and report code number. This material is tabulated in full in Section 3 herein, and in addition the data has been processed to establish patterns in Sections 4 and 5.

(2) All Failure/Anomalies in Ground Testing - Identifying failure description, cause, effect, date of occurrence, and report code number. This material also was processed into useful patterns in Sections 4 and 5. The individual failures are too voluminous for inclusion in this report.

(3) Hardware and Mission Configurations - Identifying systems, components, parts counts, and mission profiles. This material is reduced to block diagrams and tables in Section 6. Preliminary Viking Lander configuration data has been included for comparison.

(4) Test Programs - Identifying hardware quantities and flow through the test program, environment types and levels in detail. This information is interpreted by hardware utilization flow block diagrams and detailed comparative charts of the test environment types and levels in Section 7. Preliminary Viking Lander test program planning has been included for comparison.

The collection of the data for the four selected programs ranged from review of published summary reports on Lunar Orbiter to sifting of original failure reports in Nimbus B log books.

2.3.2 Data Processing and Evaluation

The failure/anomaly data from ground test and flights were reduced to incident, cause, effect and previous history. These reduced data were combined for each program into groups of interest to this study. These groups included failure classes, which are related to hardware risks; flight and test failure modes, which are related to responsible agency; chronology of ground test incidents; and various categories of failure mechanisms. Unpublished data and consultations with project technical authorities assisted in the interpretation of the failure mechanism data.

Other information was organized to provide mission and equipment details, use and flow of hardware, nature of the test programs, equivalent parts count, and functional block diagrams.

The data described above were compared with comparable material for the Viking Lander for reference. Basic comparisons were then made and graphically portrayed to present the relative distributions of failure classes for each program as well as distributions of failure modes and failure mechanisms for both flight and ground test.

As a result of comparing the reduced data among all the programs considered in this study, a number of conclusions and recommendations are presented.

2.4 Descriptions of Report Contents

Each section of this report contains material which can be used relatively independently of the others. To facilitate the location of specific information, the contents of each succeeding section is described as follows:

Section 3, Failures in Space - Tables itemize all flight failure/anomaly incidents which occurred in the projects. Cause and effect descriptions and prior history are provided for each incident along with the designation of failure class and mode. The text defines significant flight incidents and flight failure classes. The flight data is then summarized in terms of these defined concepts.

Section 4, Failure Modes - The contents include definitions of failure modes, tabulation of the failure modes for each program for both flight and ground test, and graphical representations of ground test failure mode distributions and chronological patterns. The data are interpreted and summarized in the text.

Section 5, Application of Experience to Viking - This section presents tabulations of incidents with hardware closely related to the Viking Lander, describes interface problems, and identifies persistent problem areas.

Section 6, Description of Spacecraft - This section provides details of the missions, equipments, functions, and parts counts for each program.

Section 7, Test Programs - This section contains diagrams of hardware utilization (including test articles), detailed tables of the system and component qualification and acceptance test programs, and discussions of the comparative test programs.

Section 8, Conclusions and Recommendations - This section presents the conclusions drawn from this study and resulting recommendations.

TABLE 2-1 SUMMARY OF SELECTION AND COMPARISON CONSIDERATIONS

COMPARISON ITEM		VIKING DEFINITION		RELATED PROGRAMS STUDIED			
		VIKING LANDER	VIKING ORBITER	LUNAR ORBITER	MARINER '69	BIOSATELLITE 30 Day	NIMBUS B
Responsible NASA Center		LRC	LRC	LRC	JPL	Ames	GSFC
Performing Contractor		MMC	JPL	Boeing	JPL	GE (RESD)	GE (Space Div.)
Mission:	Type	Mars Landing	Mars Orbit	Lunar Orbit	*Mars Fly-By	Earth Orbit Low Angle with atmosphere *entry.	Earth Orbit Polar
	Science Carried	Entry; Atmosphere Reconstrt'n Surface: Photo Biology Weather Seismology	Planet Map Photo, High Alt. Atmosphere Measurements	Lunar *Photo Map Solar Rad'n Micrometr'd	*Photo map. Atmosphere; High Alt. Measurements	*Biology of Primate; Psychology of Space Flight	Earth Meteorology *Photo, *RTG, *Gas Bearing Rate Gyro
*Items of special interest to Viking Lander							
Weights:	Spacecraft	2350#	1800#	578#	821#	1380#	1249#
	On Board Expendables	Fuel and Gasses	Fuel and Gasses	Fuel and N_2	Fuel, H_2 and N_2	Liquid H_2, O_2 and gasses	Freon Gas
	Weight Expendables	300#	1800#	275#	29#	150#	20#
	Total Weight at Separation from Booster	2650#	3000#	853#	850#	1530#	1269#
Nominal Flight Duration		17 Mo.	17 Mo.	1 Mo. Prim. 12 Mo. Limited	9 Mo.	30 Days	1 Year
Project Type		New	Continuing Mariner	New	Continuing Mariner	New	Continuing Nimbus

TABLE 2-2 SUMMARIES OF FLIGHT SPACECRAFT DATA

PROJECT	PROJECT DESIGNATION		FLIGHT DES'GN'	LAUNCH DATE	DAYS IN FLIGHT	COMMENT
Lunar Orbiter	S/N-4	A	I*	8-10-66	80	32 Days Photography; Flight terminated to avoid interference with next flight.
	S/N-5	B	II*	11-6-66	339	30 Days Photography; Terminate to avoid Lunar Eclipse
	S/N-6	C	III*	2-5-67	246	27 Days Photography; Terminate to avoid Lunar Eclipse
	S/N-7	D	IV*	5-4-67	74	28 Days Photography; Failed and lost
	S/N-3	E	V*	8-1-67	183	25 Days Photography; Failed, Terminated
Mariner '69	M69-3	F	VI*	2-25-69	160+	Days to Mars encounter; Mission ended officially on 11-1-69 however tracking continued until 3-1-70
	M69-2	G	VII*	3-27-69	127+	
Biosatellite	RV301	A	I*	12-14-66	63	Failed to deorbit, entry by orbit decay
	RV302	B	II*	9-7-67	2	Called down early to avoid storm
	RV501	D	III*	6-27-69	8-45	Primate returned early. Spacecraft section continued to operate.
Nimbus		A	I	8-24-64	28	Bearing failure, tumbled, lost
		C	II	5-26-66	966	Attitude control gas finally expended
		B1*	Lost*	5-18-68	-	Booster Failed
		B2*	III*	5-14-69	568	Continuing as of 11-1-70
		D	IV	4-8-70	207	Continuing as of 11-1-70

*Indicate Data Included in this Study

3.0 FAILURE AND ANOMALY EXPERIENCE FROM SPACE FLIGHT

This section presents the failure and anomaly data from space flight and establishes relationships between the flight data and ground test results. The relationships established for these data become one of the base points for assessing potential flight problems in new programs. The material appearing in this section provide descriptions and discussions as follows:

- The basic flight data is described and presented in terms of flight failure anomaly incidents.

- The flight data is screened and classified in terms of significant flight incidents (defined in 3.2) and risk classifications (defined in 3.3.1) and failure modes (defined in 4.0)

- Relationships between flight data and ground test results are established by observations drawn from the distribution of significant flight incidents within the classification of risk.

The material in this section is presented in the following figures and tables:

3.1 Data From Space Flight

Tables 3-2 through 3-5, inclusive, present descriptions of all the failure/anomaly incidents reported in the course of flight operations for each of the projects within the study. These data are presented as separate tables for each of the projects. Wherever flight incidents had a previous history from ground testing or other flights, the related experience is described. The tables presenting the individual programs have been organized as follows:

Lunar Orbiter, Table 3-2 - The project consisted of 5 flights and reported 63 incidents. The data are presented in 16 groups of either similar incidents or a series of incidents involving a single item of spacecraft equipment (e.g., Star Trackers, Power Losses). For reference the individual incidents are identified by their documented failure/anomaly report number such as LO-II-A7 (Roman numeral identifies flight spacecraft).

Mariner '69, Table 3-3 - The project had 2 simultaneous flights and reported 43 incidents. The data are presented in 18 groups arranged in flight chronology (e.g., Mariner 6 Launch, Mariner 7 Launch, Star Sensor problems). Individual incidents are identified by the JPL problem/failure report number (PFR) with a suffix digit to show which spacecraft (6 or 7).

Biosatellite, Table 3-4 - The project had 3 flights and reported 47 incidents. The data are presented in 14 groups which identify the function or equipment involved (e.g., failure to de-orbit, reversed G switches, IR sensors). The individual incidents are identified by the flight spacecraft and the failure transmittal numbers.

Nimbus B, Table 3-5 - This project has one continuing flight and reported 46 incidents. These data are presented in 12 groups in order of the first appearance of a problem or a defect in an item of spacecraft equipment (e.g., electrical part failures, tape recorder problems). The individual incidents are identified by the item number assigned in the Nimbus monthly report.

Common Problems, Table 3-6 - Table 3-6 lists problems which occurred on more than one project and shows cross references to the failures and anomalies presented in the tables of flight incidents. A total of 46 incidents covering 13 problems have occurred; these included both operating anomalies and defects in spacecraft equipment.

3.2 Significant Flight Incidents

The failure and anomaly incidents reported from flight describe problems which have taken many forms. For example, components and sub-assemblies have slowly degraded in the course of a long flight. Complex problems have appeared which perturbed more than one subsystem within the spacecraft and involved several interfaces. Also, the flight work-arounds for one anomaly have generated side effects which later resulted in new problems. Therefore, flight data were screened to identify the actual incidents which perturbed the flights. Such incidents are termed significant, and these significant flight incidents are used as the basis for the comparisons presented in this study. The following criteria were employed to identify significant flight incidents:

1. Spacecraft Origin - The problem or anomaly must have had a potentially damaging affect on the flight spacecraft.

2. Same Incidents in Separate Spacecraft - These incidents are counted separately. This accounting includes the cases where operating difficulties persisted in spite of corrective modifications incorporated into succeeding spacecraft. An example is the Lunar Orbiter Star Sensor.

3. Continuing or Cyclic Incidents - These incidents are considered as one event. An example appears as the spurious switchings experienced by the Nimbus spacecraft during over-flight of specific areas on earth.

4. Multiple Failure Mechanisms within a Particular Component or Subsystem - These incidents are counted separately. Typical examples are tape recorders which can simultaneously degrade due to both flutter and head contamination effects.

5. Secondary Problems Arising from a Flight Work-Around - These incidents are not included unless there was a further work-around. An example is the case of Lunar Orbiter camera thermal door. The secondary effects of film struck by stray light, fogged windows, and distorted video signals are not included as significant flight incidents.

6. Transitory or Undamaging Incidents of a Predictable Nature - These incidents are not included. An example is the Lunar Orbiter Launch into the earth shadow.

7. Ground Equipment Problems - These incidents are not included. Lunar Orbiter data reported problems within the DSN and the Photo Ground Reconstruction Equipment during the course of flight operations.

Significant flight incidents appear as separate entries throughout the tables of flight data. Multiple occurrences of the same event are listed one time with the repetition shown by number. Flight incidents which do not meet the criteria are appropriately indicated as "no failure", "not in spacecraft", "predictable", "continuing", or "secondary".

3.3 Relationship of Flight Failures to Ground Testing

3.3.1 Flight Risk Classifications

Flight incidents have been classified as risks in terms of ground test problems which carried over into flight. For the purposes of this study the risks assigned to flight incidents have been classified as follows:

I. Standard Risk - The risk associated with well established parts, materials, and processes.

II. Accepted Risk - Problems with a previous history from ground testing. In each case the problem was not completely removed, and the potential failure was considered an acceptable risk as to chance and consequence of occurring.

III. Undetected by Ground Test - The ground tests and inspections employed were intended to uncover these defects; however, by some means they escaped and later appeared in flight.

IV. New Effect - The first appearance of the problem was during flight.

The risk classifications appear in the tables of flight incidents (Tables 3-2 through 3-5) as Roman Numerals in the column identified as "Class". In addition, these classifications appear in the cross references for common flight problems (Table 3-6).

3.3.2 Risk Classifications Within the Significant Flight Incidents - Ground test results were related to flight failures in terms of sources and effects, respectively. Table 3-1 and Figure 3-1 present distributions of risk classifications for the significant incidents of each project, as precentages and as numbers of incidents, respectively. Comparing the distributions within the projects and across the projects permit the following observations:

I. Standard Risks - Well established parts, materials, and processes have not made the major contribution to the number of reported flight failure incidents in any of the projects. For Biosatellite and Nimbus B the contributions from standard risks were increased as the result of project decisions:

- Biosatellite Thermistors: The temperature monitor thermistors were redundantly supported by other sensors within the 3-Day spacecraft. Consequently thermistors failures were not repaired during prelaunch operations and such failures account for 3 of the 5 standard risk incidents reported for Biosatellite. (A further discussion appears in Section 4.1)

- The Nimbus B spacecraft employed a number of commercial grade parts and electrical part failures account for all but one of the incidents reported for this class.

II. Accepted Risks - Problems with a history of failures in ground testing originated the largest number of flight incidents in all the projects except Nimbus B. The problems reported in flight came from two sources. One source was advanced hardware which had been configured for the particular spacecraft application. These equipments included such items as tape recorders, rate gyros, transponders, and pyro valves. The second source was from system operating problems. In the course of ground testing these problems had been resolved to a condition considered tolerable for the particular mission (an example is an EMI sensitivity shown by test to have a 6 db operating margin). In the particular case of Nimbus B, the flight incidents all originated in advanced hardware; the system operating problems had been resolved with earlier spacecraft.

III. Undetected by Ground Test - Significantly, all the projects in the study had problems which escaped the intended tests or inspections and caused failure incidents in flight. The circumstances which allowed these problems to escape testing and the mechanisms which caused these problems are the subjects for the later portions of this study. In relating the incidents to ground test results (or absence of results in this case) the originating sources can be summarized as follows.

- Defects traceable to manufacturing operations appeared in all projects.
- Additional sensitivities to transient pulses appeared in all but Nimbus B.
- Reversed polarities with direction sensitive equipment appeared on Mariner and Biosatellite.
- A complex electromechanical unit in Biosatellite was flown without previous environmental qualification.
- Evidence of undetected thermal vacuum and vibration sensitivities appeared in Lunar Orbiter.

IV. New Effects - New effects appeared in all the projects studied. The sources reported by flight incidents are summarized as follows:

- Untried hardware problems such as operation of star sensors.
- Loss of contact or unexpected damage to the spacecraft (Lunar Orbiter IV, Mariner VII).
- Encounter of new or unexpected environments. Incidents from this source have continuously perturbed the operation of Nimbus B.

3.3.3 Risk Classification within Compromising Flight Incidents

Throughout the previous discussions, flight incidents have been considered without regard to the impact upon the particular flight mission. The projects in this study all enjoyed successful space flights and by a large majority anomalous incidents fell within the capabilities of the individual spacecraft. Therefore, a further correlation of flight failures and risk sources has been performed. This correlation limited the flight incidents to those which were considered compromising to the flight mission.

A flight failure was considered compromising if the incident had any of the following results:

- The incident caused the loss of any mission science data.
- The incident caused any loss to the control of the spacecraft.
- The incident resulted in termination of RF contact.
- The incident resulted in premature termination of flight operations.

The numbers of incidents which compromised flights and the numbers of spacecraft suffering early termination (or loss) are summarized in Table 3-1. The distribution of these data into risk classifications appear below: (These incidents are indicated by "x" in the Tables 3-2 through 3-5)

Project	Compromising Incidents	Risk Class I	II	III	IV
LUNAR ORBITER	9	3	2*	3	1*
MARINER	7	0	4	2	1
BIOSATELLITE	13	1*	6	5	1
NIMBUS	7	1	2	0	4
TOTALS	36	5	14	10	7

*Includes an incident which resulted in premature flight termination.

Observations drawn from these data are as follows:

I. Standard Risk - Standard risk items have provided sources which compromised flights. The Biosatellite failure to deorbit was the outstanding example. For Lunar Orbiter 2 of the incidents were part failures in the Photo Subsystem, and 1 was a capacitor in the traveling wave tube amplifier (evidence exists that an improper capacitor had been installed). For Nimbus B a capacitor failure in the Infra Red Spectrometer degraded one channel after about 700 hours of flight.

II. Accepted Risk - Problems with a previous history in ground testing provided 40 percent of the compromising flight incidents. Of the 14 incidents reported 1 was related to a system operating problem of low voltage stability (LO-V-A4); the balance were caused by advanced hardware as follows:

- Unproven Hardware, Biosatellite Psychomotor game - pellet feeder, 5 incidents.
- Airborn Tape Recorders, 5 incidents involving Mariner, Biosatellite and Nimbus B.
- TV Cameras Mariner
- Plugged Cryostat, Mariner
- Camera Thermal Door, Lunar Orbiter

III. Undetected in Ground Test - The second largest contributer to mission data losses were effects which escaped detection in the tests and inspections applied. Reviewing these incidents revealed the following sources:

- EMI caused the Mariner incidents and one incident with Lunar Orbiter.
- Biosatellite was compromised twice by reversed polarities (one mechanical, one electrical) and the unproven hardware item described previously.
- The Lunar Orbiter Photo Systems showed evidence of a vibration induced change to a delicate adjustment during one Flight (IV) and for the last flight, a shortened segment of film/developer Bimat was installed in the supply magazine.

IV. New Effects - The abilities to accomodate unexpected or new effects are measures of the margins built into the spacecraft and designed into the total system. The margins include:

- Extra capability to withstand environments
- Extra operating capability (power, controls, expendables)
- Alternate operating modes

New effects comprised 33 of the 139 significant flight incidents. The causes of new effects have ranged from untried hardware to new environments. In the course of 10 flights new effects have compromised data only 7 times and 2 of these events were loss of contact with the spacecraft. (Lunar Orbiter IV, completely - Mariner 7, temporarily). The other five compromising events were induced by unexpected environmental factors. These environmental effects are of particular interest for comparisons to the Viking Lander and are the subject of further discussion elsewhere in this study (Section 5). The ability to accomodate new effects as shown by the reported flight data confirms the original selection premise that these projects were well conceived and well executed; therefore, they provide a suitable base for making further comparisons.

TABLE 3-1 SUMMARY OF FLIGHT FAILURE AND ANOMALIES FOR THE PROGRAMS STUDIED

PROGRAMS	LUNAR ORBITER	MARINER '69	BIOSATELLITE	NIMBUS B
Number Flights	5	2	3	1
Number of Incidents Reported	63	43	47	46
Number of Significant Incidents	51	33	29	26
Number of Flight Compromising Incidents	9	6	13	7
Number of Flights Terminated by On-Board Anomaly	2	0	1	Flight Continuing
Risk Assessment Data, Percent of Significant Flight Incidents				
I. Percent due to well established parts materials and processes.	16	3	18	27
II. Percent due to assumed risks	41	55	48	31
III. Percent which went undetected by ground test.	25	12	24	4
IV. Percent new effects	18	30	10	38
Number of effects in common with other programs (Ref Table 3-6)	9	9	10	7

LISTING OF FLIGHT FAILURES AND ANOMALY INCIDENTS

The recorded incidents from spacecraft flight are presented in the following tables:

Table 3-2 — Lunar Orbiter - 5 Flights - 63 Incidents
Presented as 16 groups. Individual incidents identified by anomaly report number, i.e. LO-I-A7

Table 3-3 — Mariner '69 - 2 Simultaneous Flights - 43 Incidents
Presented as 18 groups in flight chronology. Individual incidents are identified by PFR number and a suffix digit (6 or 7)

Table 3-4 — Biosatellite: 3 Flights - 47 Incidents
Presented as 14 groups. Individual incidents identified by flight spacecraft and the failure transmittal number.

Table 3-5 — Nimbus - 1 Flight - 46 Incidents
Presented as 12 groups presented in order of first appearance for a particular problem or effect. Individual incidents identified by number assigned in the Nimbus monthly report.

*Indicates common problems between programs. See Table 3-6)

Modes: A failure mode designator has been applied for correlation with Section 4. The modes are: D, Design Origin; M, Manufacturing Defect; P, Electrical Part; G, Mechanical Device; X, Miscellaneous or Unknown. Flight incidents often show evidence of contributions from more than one failure mode. In such cases, a probability for the relative contribution by each mode was estimated and is expressed as a decimal.

Risk Classification: Risk classifications are identified by Roman numerals as follows:

I. Standard risk. Well established parts materials and processes
II. Accepted Risk. Problems known from ground testing
III. Undetected in Tests. Flight incidents which were not uncovered or detected by the intended ground test.
IV. New Effects. First appearance occurs during flight.

Compromising Incidents indicated by x preceding the description.

TABLE 3-2 SUMMARY OF FAILURES AND ANOMALIES FROM LUNAR ORBITER FLIGHT EXPERIENCE

MISSION OR FLIGHT LIMITING EFFECTS (ITEMS 1 THRU 5 INCLUSIVE)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
LO-I-1 LO-II-A7 LO-III-A2 LO-IV-A3	* 1) Canopus Star Tracker used for cruise reference was sensitive to stray light; attitude reference not stable. Workaround required; Spacecraft controlled from earth.	None: Ground tests did not adequately simulate flight conditions.	4	IV	D
	2) Power loss or degradation:				
LO-V-A4	x • Evidence of power loss for more than one hour. Programmers had changed state, attitude control gas was lost. Flight was terminated.	Low voltages at the spacecraft bus had caused instabilities in the program during previous ground tests.	1	II	D
LO-III-A14	• Shunt regulator shows sporadic leaks up to 0.5 amperes	Leakage in flight on LO-II, LO-V, no evidence from ground test.	1	I	.5P .5M
LO-I-6	• Shunt regulator transistor failed 1.2 amp leak.	Similar to a failure during ground test caused by a solder ball in the transistor.	1	II	.5P .5M
LO-II-8	* 3) Attitude control compromised by Gyro Gimbal hangup. Loss of motion reference. (Roll)	Anomalous gyro operation had been reported 16 times during ground test.	1	II	G
	* 4) Spurious commands				
LO-II-A5	• Spacecraft responded to commands intended for another unit. Executed uncontrolled unwanted actions.	Susceptability to false commands discovered in ground compatible tests.	1	II	D
	5) Loss of contact with spacecraft				
LO-IV-A8	x • Spacecraft ceased all RF activity.	None: spacecraft lost.	1	IV	X
LO-II-5	x • TWTA exhibited anomalous behavior then ceased to transmit (would no longer turn on at command). A filter capacitor in the high voltage assembly was assumed to have failed. Failure resulted in loss of 17 photo frames.	None: later experienced in ground tests for LO-IV mission simulation. Capacitor failure as in ground test then assumed for flight.	1	I	P

TABLE 3-2 SUMMARY OF FAILURES AND ANOMALIES FROM LUNAR ORBITER FLIGHT EXPERIENCE (Contd.)

EFFECTS COMPROMISING DATA ACQUISITION AND TRANSMISSION (ITEMS 6 THRU 12 INCLUSIVE)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
	* 6) Telemetry signal strength variations.				
LO-I-5 LO-IV-A1	• Small fluctuation in the output power due to temperature and vibration sensitivity of system.	None: Problem not severe for L.O.	2	III	.5D .5P
LO-I-7	• Step function of -7.3 dbm in received RF signals at DSN stations and -4 dbm in SC received RF signals. Original signal strength recovered later. Filter was suspected.	No history in subsystem or system test. Filter suspected based on a failure in component testing of transponder.	1	II	.5P .5M
LO-III-3	• TWTA showed high turn on current plus changes in RF drive. Conservation action taken to minimize potential for over current or loss. Readout time reduced.	Ground tests had shown the sensitivities for TWTA units. These were the basis for precautionary action.	1	II	.5P .5M
LO-III-A5	• RF power in TWTA changes 8-12 mw due to temperature sensitivity of a directional coupler within the RF transponder.	Same as for LO-I-7 (above)	1	II	.5P .5M
LO-III-A4	• RF power varies cyclically with transponder temp., degraded filter and RF coupler.	Same as for LO-I-7 (above)	1	II	.5P .5M
LO-III-A7	• RF Signal strength fluctuated during tests and countdowns due to multipath effects.	Known condition at ETR	No Fail.		
	* 7) EMI Effects				
LO-III-A3	• TWTA Helix voltage measurement indicates erratic by TM sensor: EM noise in sensor line is suspected cause.	None	1	III	D
LO-III-A12	• Transient pulse generated in photo system changes TM data bit and generates spurious "off" command. (Read out time limited to 28 seconds, this incident related to LO-III-2)	None	1	IV	D

TABLE 3-2 SUMMARY OF FAILURES AND ANOMALIES FROM LUNAR ORBITER FLIGHT EXPERIENCE (Contd.)

EFFECTS COMPROMISING DATA ACQUISITION AND TRANSMISSION (ITEMS 6 THRU 12 INCLUSIVE)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
	7) Cont'd				
LO-V-A1	● EM spike generated by command receiver causes loss of lock. Condition critical when controlling multiple vehicles.	Susceptability known from ETR tests prior to first unit flight.	1	II	D
LO-I-8	x ● Picture smeared, caused by an EM spike generated within system which was interpreted as a shutter operating command. 156 photo frames smeared.	Spike was generated in the VH sensor. Evidence shows that a grounding loop in AGE masked the effect in tests performed. Tests of photosystem with VH operating not performed with spacecraft.	1	III	D
	8) On-Board Electrical Problems:				
LO-III-2	x ● Read out shut off immediately after commanded "on"; no further film data from this mission. Data limited to 138 frames. Suspected DC-DC converter, diode or transistor.	None:	1	I	0.5P 0.5M
LO-III-4	● High voltage section of photo system failed to operate. Suspected short in either a high voltage lead, transformer, or tube anode.	Ground test had shown a series of arcing failures at low pressure	1	II	.5D .5P
LO-IV-2 LO-IV-5	x ● Readout stopped without command after each film exposure. Readout of each frame required an individual action from the ground. Brush in film position monitor circuit suspected.	None (192 of 212 photo frames were read out) (Evidence of vibration sensitivity)	2	I	.5G .5M
LO-IV-3	● Spurious indication of "A/C thruster "on", suspect multivibrator failed.	None	1	I	.5P .5M
LO-V-1	● Video signal interrupted during scan transmission. Ground film shows 3 to 6 white lines on each of 7 frames. Suspected breakdown in high voltage connector or capacitor.	Appeared related to high voltage breakdown failures at low pressure as in component tests.	1	II	.5D .5P
LO-II-1	● Spurious telemetry counts of shutter actuation without actual operation of shutter;part failure.	None	1	I	.5P .5M

TABLE 3-2 SUMMARY OF FAILURES AND ANOMALIES FROM LUNAR ORBITER FLIGHT EXPERIENCE (Contd.)

EFFECTS COMPROMISING DATA ACQUISITION AND TRANSMISSION (ITEMS 6 THRU 12 INCLUSIVE)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
	9) Mechanical Device Problems				
LO-II-3	• Film readout sequence failed to stop on command; video transmission persisted. Cause attributed to cam adjustment or wear. Shut off index was missed, stop positions contacts did not actuate.	Known vibration sensitivity.	1	III	G
LO-II-7 LO-III-1	• Film failed to advance past scanner, one framelet was repeated 18 times (LO-II-7) another 14 times (LO-III-1) plus other cases. Cause was attributed to over length mounting screw in film take up holder. The screw deflected the film guide and caused binding.	None. Attributed to tolerance buildup	2	III	M
LO-IV-1	x • Camera thermal door failed to open on command. Later the door was opened successfully but never allowed to close again. Cause was attributed to jam or binding in unlocking mechanism. Resulted in film data loss from secondary effects.	Door opening failures had been experienced on 2 occassions during thermal vacuum operation of system ground tests.	1	II	.5D .5M
LO-IV-4	x • Wide angle frame unexposed, evidence of double shutter trip on telephoto camera. Cause attributed to tolerance build up plus wear coupled with delicate adjustment required in set up of shutter control mechanism.	Testing of the photo subsystem had shown vibration sensitivity. (Loss of one photograph)	1	III	.5D .5M

TABLE 3-2 SUMMARY OF FAILURES AND ANOMALIES FROM LUNAR ORBITER FLIGHT EXPERIENCE (Contd.)

EFFECTS COMPROMISING DATA ACQUISITION AND TRANSMISSION (ITEMS 6 THRU 12 INCLUSIVE)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
	10) Problems Particular to the Photo Subsystem				
LO-II-6 LO-III-A8	• Exposures from wide angle and telephoto cameras not the same. Balancing filters in line not correct density.	Measurement of this type not considered in acceptance criteria.	2	IV	D
LO-IV-A6	• Line scan tube tilted beyond tolerance in vehicle. Alignment instructions unclear for field preparation.	Final precision of alignment is dependent upon skill of the individual performing.	1	II	D
LO-II-2 LO-III-A10	• Film processing defects due to Bimat fabrication, and on board operation. Resulted in some degradation to film.	Incident occurred 1 time during subsystem flight acceptance in thermal vacuum. Not able to duplicate or repeat effect.	2	II	M
LO-V-A3	x • Short Bimat in magazine, film data loss	None, (Inspection oversight)	1	III	M
LO-V-2	• Film broke during rewind	None	1	IV	M
LO-III-A9	• Loose washer found in photo subsystem during ETR preparation.	Loose film segment found in subsequent unit.	No Failure		
	*11) Thermal Effects				
LO-I-2	• Thermal paint degraded at rate exceeding predictions. Vehicle operated off-sun to maintain temperature control.	Inadequate ground evaluation technique applied.	1	III	D
LO-II-4	• 30KHz oscillator drifted 110 Hz due to temperature change associated with paint degradation. Ground equipment biased to accomodate shift.	None	1	II	D
LO-II-A2	• Photo system casing loses pressure after 300 days. Suspected a thermal cycling crack or meteoroid penetration.	This casing had leaked in ETR preparation; Pin hole discovered and patched.	1	II	M
LO-II-A6	• False indication of automatic gain control measurement increase. Change was due to aging of resistors in TM channel circuit.	Same problem later observed in preparation of LO-III.	1	III	.5P .5D

TABLE 3-2 SUMMARY OF FAILURES AND ANOMALIES FROM LUNAR ORBITER FLIGHT EXPERIENCE (Contd.)

EFFECTS COMPROMISING DATA ACQUISITION AND TRANSMISSION (ITEMS 6 THRU 12 INCLUSIVE)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
	* 11) Thermal Effects (Cont'd)				
LO-II-A1	• Photo system Lo temperature due to launch in earth shadow.	Predictable condition	No fail.		
LO-III-A1	• Battery temperatures show difference due to off-sun (Canopus) work around.	Predictable condition	No fail.		
	* 12) Sensor Problems				
LO-I-10	• Loss of accelerometer; on board sensor failed to indicate during launch. Failure was in the spacecraft circuitry. No checkout or continuity measurements were performed after mate to booster.	None	1	III	.5M .5P
LO-I-3	• Pressure transducer in N_2 supply system was erratic at low voltage with internal power. Flight proceeded without transducer fix or change.	None	1	II	.5G .5M
LO-III-A15	• Solar flare occurred, radiation level saturated the sensor.	None. This is a space environment to be considered.	No Failure		

GROUND BASED EQUIPMENT EFFECTS (ITEMS 13 & 14)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
	13) AGE Problems				
LO-III-A6	• Ground power failed to reapply during a simulated count down. Failure due to missing load resistor and blocking diode in transistor circuit. Vehicle experienced transients (Change in AGE not incorporated).	Change out in power supply to launch complex equipment between Flights 2 and 3.	1	III	M
LO-I-9	• Camera GRE operation showed foggy pictures due to halos from a Kinescope tube in AGE. Tube masked to minimize effect.	Tube design problem. Could not be eliminated only reduced.	Not in Spacecraft		

TABLE 3-2 SUMMARY OF FAILURES AND ANOMALIES FROM LUNAR ORBITER FLIGHT EXPERIENCE (CONCLUDED)

GROUND BASED EQUIPMENT EFFECTS (ITEMS 13 & 14)

Lunar Orbiter Designation	Description of the Failure/Anomaly Incident	Previous History, Comments	Number	Class	Mode
	14) DSN Operations Problems				
LO-II-A4	• Each DSN station reported a different automatic gain control measurement from the on board transponder. Calibration techniques for DSN stations caused apparent effect.	Problem not in S/C	No Failure		
LO-V-A2	• Video data showed high noise content at the same time a DSS station was operating a test transponder at the same frequency.	Problem not in S/C	No Failure		
MISCELLANEOUS EFFECTS (ITEMS 15 & 16)					
	*15) Valves and Regulators				
LO-II-A3 LO-III-A13	• Pyro "shut-off" valve failed to seal completely; gas leaked past seat.	Valve not adequate design for the application.	2	II	D
LO-III-A11	• Fuel fill valve leaked past poppet seat due to insufficient spring force and poor alignment with the seat.	Observed on previous unit during flight preparation.	1	II	M
LO-I-4	• N_2 pressure regulator leaks. Gas transferred to unwanted area.	None	1	I	G
	16) SecondaryProblems Arising from Other Incidents				
LO-IV-A4 LO-IV-A2	• Film light struck thru open camera door during orbit cruise. Transmission of light struck film results in abnormal video signals	None; Limitations to work around for LO-IV-1 (Camera thermal door)	Secondary		
LO-IV-A5	• Video signal lost for 24 seconds during start-stop readout from LO-IV-2, part failure in position brush indicator.	None	Secondary		
LO-IV-A7	• Fogged window due to camera door open, window temperature drops below dew point during orbit cruise.	None; Result of work around for LO-IV-1	Secondary		

TABLE 3-3 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHT OF MARINER '69 VEHICLES M-VI, VII

PRELAUNCH THRU SEPARATION FROM CENTAUR (ITEMS 1 and 2)

Mariner Designation PFR No. S/C	Description of Failure, Cause, Effects (Date)	Previous History - Comments	Number	Class	Mode
	* 1) Mariner 6 Launch				
204695-6	• RF output from vehicle drops 0.8 db in 2 steps (2-24)	Potentially a multipath effect.	1	II	X
204696-6	• Receiver frequency shifts when operated out-of-lock in ranging mode. In the course of the countdown the automatic gain control showed a 2 db drop when ranging mode is commanded on. (2-24)	Identified as a EMI problem and involves a feedback loop in a 15 volt power supply.	1	II	D
203894-6	• Anomalous high frequency accelerations recorded during boost (2-24)		1	IV	X
	* 2) Mariner 7 Launch				
204690-7	• RF power fluctuates during boost phase indicated by telemetry (3-27)		1	II	X
204689-7	• CC and S executed memory dump sub routine without ground command to readout mode. Coincident with Centaur separation. (3-27)	Close and bounce of the separation relay generates transient pulses. These pulses travel within the spacecraft by noise coupling within harness bundles. The CC and S interprets the transient pulse as the command for memory readout.	1	II	D
INTERPLANETARY CRUISE - (ITEMS 3 THRU 8 INCLUSIVE)					
203895-6	* 3) Gas vented from scan latch mechanism perturbs velocity ~ 9 mm/sec unexpected result (2-28)	Unbalanced force on S/C due to gas jet. Not considered in original flight plan.	No Fail.	(IV)	

TABLE 3-3 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHT OF MARINER '69 VEHICLES M-VI, VII (Contd.)

INTERPLANETARY CRUISE - (ITEMS 3 THRU 8 INCLUSIVE)

Mariner Designation PFR No. S/C	Description of Failure, Cause, Effects (Date)	Previous History - Comments	Number	Class	Mode
	* 4) Canopus Star Sensor Problems				
204694-6	• Intensity increase in Star Sensor: Dust particles suspected (2-28)	Dust particles dislodged from exterior of S/C by maneuvers or shock. Suspected source of dust is dirt from inside of shroud.	1	IV	.5D .5M
204692-6	• Star sensor lost lock at scan platform unlatch (3-6)		1	IV	D
204680-6	• Could not update Canopus cone angle (4-20)	New effect. Anomalous operation not experienced in ground test.	1	IV	D
204672-6	• Tracker will not hold lock on a dim star. Suspect noisy tube. (4-30)		1	I	P
204670-6	• Could not update cone angle (5-26)	Same effect as 204680, above.	Contd.		
204686-7	• Intensity increased in Star Sensor Suspect Dust particles (4-8)	Same effect as for M-VI, (See 204694 above)	1	IV	.5D .5M
204673-7	• Bright particles in Star Sensor field of view (5-8)	"	Contd.		
204642-7	• Command sequence causes Canopus loss (7-26)	New effect, M-71 design changed.	1	IV	D
	* 5) RF Lock Problems				
204691-6	• RF system appeared to go into an internal self lock mode when DSIF transmitter was off (3-21)	RF system exhibited false lock and supurious locks in ground test on at least 2 occasions. Also appeared related to lift off condition of frequency shift.	1	II	D
204675-6	• Flight command system dropped lock then reacquired. (4-6)		1	II	D
204678-7	• Ranging loop lost lock, regained (4-20)		1	II	D
204677-7	• Command receiver lost, lock then regained (4-25)		1	II	D
204679-7	• Automatic gain control measurement via TLM. shows fluctuating voltage during RF lock events (4-25)		No Fail.		

TABLE 3-3 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHT OF MARINER '69 VEHICLES M-VI, VII (Contd.)

INTERPLANETARY CRUISE - (ITEMS 3 THRU 8 INCLUSIVE)

Mariner Designation PFR No. S/C	Description of Failure, Cause, Effects (Date)	Previous History - Comments	Number	Class	Mode
204687-6	* 6) The telemetry input signal was reversed from one solar panel temperature transducer (4-7)		1	III	M
	* 7) RF Power Fluctuations				
204671-6	• Auxiliary oscillator #2 had a short term instability (4-8)		1	II	.5D .5M
204681-6	• Power decrease noted in oscillator drive (4-18)		Contd.		
204674-6	• RF drive to TWT varied beyond set limits (5-10)		Contd.		
204667-6	• Auxiliary oscillator for S band drifted excessively over a period of hours (6-18)		1	II	.5D .5P
204668-7	• Auxiliary oscillator drifts excessively over a period of several hours (6-18)	Same as for M-VI. See 204667 Appeared related to RF lock problems	1	II	.5D .5P
204665-7	• Change in RF power TLM channel reading when rangine mode switched on using low gain antenna (6-20)		No Fail.		
204662-7	8) CCS unable to switch to hi gain antenna, suspect defective IC (7-7)	Ground operation of IC units showed history of problems during fabrication and test.	1	II	.5D .5M

ENCOUNTER (ITEMS 9 THRU 14 INCLUSIVE)

Mariner Designation PFR No. S/C	Description of Failure, Cause, Effects (Date)	Previous History - Comments	Number	Class	Mode
204661-6	9) CCS shows unexpected event in counter #4 during pre-encounter check test, wiring error. (7-25)		1	III	M
	10) Infrared Radiometer Problem at Encounter				
201349-6 201348-7	x • Data from IRR shows changes with position of scan platform; EMI noise effects (7-28 and 8-1)	Problem exhibited in ground test only when AGE cables in place, and therefore attributed to test configuration.	2	III	D

TABLE 3-3 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHT OF MARINER '69 VEHICLES M-VI, VII (Contd.)

ENCOUNTER (ITEMS 9 THRU 14 INCLUSIVE)

Mariner Designation PFR No. S/C	Description of Failure, Cause, Effects (Date)	Previous History - Comments	Number	Class	Mode
204636-6	11) TV far encounter pictures lack constrast x (7-29)	TV had shown incorrect constrast in tests on PTM	1	II	D
204650-6	12) Infrared spectrometer channel 1 Cryostat x fails to cool, data lost for longer wave length (4-14 microns) (7-30)	History of IRS cryostat plugging during ground test due to contaminants (~ 10 incidents)	1	II	.5D .5M
204897-6	13) Unexpected doppler residuals during encounter (7-31)		1	IV	X
	14) Mariner VII Pre Encounter Anomaly (7-30)				
204660-7	x ● An unexpected abrupt signal loss from the spacecraft occurred with side effects described in 20 PFR documents. The spacecraft was reacquired and the encounter sequence successfully performed.	This event represented a significant flight failure of nearly catastrophic nature.	1	IV	X

POST ENCOUNTER AND EXTENDED MISSION (ITEMS 15 THRU 18 INCLUSIVE)

Mariner Designation PFR No. S/C	Description of Failure, Cause, Effects (Date)	Previous History - Comments	Number	Class	Mode
	*15) Tape Recorder Problems				
204657-7	● Analog tape recorder erase problem. Command sequence changed (8-3)	Power subsystem transients have changed states in controllers; ground test.	1	II	D
204658-6	x ● Track 1 of analog tape recorder, magnitude reduced by half (8-8)	Changes in output of analog channel were observed in ground test.	1	II	.5D .5P
204659-6	x ● Track 1 of digital tape recorder not readable (8-8)	Oscillations observed in digital recorder preamplifier during ground tests, data would be lost.	1	II	.5D .5P

TABLE 3-3 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHT OF MARINER '69 VEHICLES M-VI, VII (CONCLUDED)

POST ENCOUNTER AND EXTENDED MISSION (ITEMS 15 THRU 18 INCLUSIVE)

Mariner Designation PFR No. S/C	Description of Failure, Cause, Effects (Date)	Previous History - Comments	Number	Class	Mode
204633-6	* 16) The memory dump routine for the CCS was not executed on command (10-30)	Non programmed changes in memory have occurred in ground tests. Taken as Flight risk.	1	II	D
204632-7	17) In an attempt to update the Canopus sensor position an anomalous power switching occurred; the spacecraft rolled, telemetry channels were perturbed, spurious command data appeared in the CCS and the scan platform apparently moved. Side effects described in 4 PFR documents (11-3)	This represents a second significant flight anomaly for this unit; suspect relationship to previous incident of 204660.	1	IV	X
	18) Mariner 7 Continuing Effects				
204624-7	• The event counter deviates by one count from predicted after command transmission (12-16)	Suspect relationship to previous events 204660, 204632.	1	IV	X
204623-7	• Ranging mode turns off without command (12-31)	"	Contd.		
204622-7	• Canopus cone angle in unexpected position after command (1-8)	"	Contd.		
204621-7	• Routine check of clock on CCS generates spurious ranging off command (1-15)	"	Contd.		

TABLE 3-4 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHTS OF BIOSATELLITE VEHICLES

EFFECTS CONTRIBUTING DIRECTLY TO THE LOSS OF BIOSATELLITE I (ITEMS 1 TO 3 INCLUSIVE)

Failure Transmittal Numbers	Type of Failure/Problem	History, Significance, Comments, Etc.	Number	Class	Mode
Flight I (3 Day) 1 Instance	1) x Failure to deorbit vehicle: Attributed to an open connection in the wiring to the Retro-rocket Initiator.	The Retrorocket and associated wiring were single point failure areas. At time of the first flight there had been no prior incidents of failures to initiate this type of rocket.	1	1	.1D .9M
Flight I (3 Day) 1 Instance Flight II (3 Day) 1 Instance	*2) x Deorbit "G" Sensing Switch for releasing parachute installed backwards.	Most probable reason for failure to retrieve Flight I. The capsule re-entered after the orbit deteriorated. The mounting was corrected for Flight II just before launch. These incidents were the generators for a very strict policy on direction sensitive equipment (gyros, accelerometers, etc.) covering drawings, mountings, verification by test and reviews of installations.	2	III	D
Flight I (3 Day) 1 Instance Flight II (3 Day) 1 Instance	*3) x Erratic operation of the IR Sensors due to overpowering thermal influence of the sun and apparent earth discontinuities caused by high altitude clouds.	These conditions could not be simulated by tests. The condition was improved by suitable optical filtering and logic & gain changes in the sensor circuitry.	2	IV	D
EFFECTS WHICH COMPROMISED FLIGHT OPERATIONS (ITEMS 4 THROUGH 7 INCLUSIVE)					
Flight II (3 Day) 1 Instance	*4) x Command Link Problems • The outputs of the two redundant Command Receivers were 180° out of phase and virtually cancelled the input to the decoder.	This effect circumvented system redundancy. An uncoordinated change was introduced. The true effect was masked by test equipment. Flight control was difficult. (See Tables 5-1 and 5.5.8)	1	III	D

TABLE 3-4 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHTS OF BIOSATELLITE VEHICLES (Contd.)

EFFECTS WHICH COMPROMISED FLIGHT OPERATIONS (ITEMS 4 THROUGH 7 INCLUSIVE)

Failure Transmittal Numbers	Type of Failure/Problem	History, Significance, Comments, Etc.	Number	Class	Mode
	*4) (Continued)				
Flight I (3 Day) BIOS-6-52-10 BIOS-6-52-11	• Command access difficult. Decoder address words are changed in the command receiver. Spurious bits enter from telemetry transmission because word frequency is an harmonic of the telemetry bit rate.	Tests of separate links showed proper operation. Combined RF (transmit-receive) testing was difficult, problems were attributed to test equipment and facility sources.	1	III	D
	*5) EMI Effects				
Flight I (3 Day) BIOS-6-52-3 BIOS-6-52-2	• Attitude control switches modes due to the action of self generated transients and noise sensitive logic.	Similar problems encountered in ground test did not receive sufficient corrective action since the test equipment was considered the principal source of EMI noise. It was subsequently found necessary to use redundant phase locking and to further desensitize the circuits by extensive redesign. The mission risk was acknowledged.	2	II	D
Flight III (30 Day) 2 Instances - 1 Case	• Spurious firing of attitude control sets due to self generated transients.		1	II	D
Flight II (3 Day) 3 Instances- 1 Case	• Transmitted commands not executed or untransmitted commands executed.	This problem was generally attributed to system noise and spikes on circuit lines. The effect was reduced by addition of a redundant Decoder enabling circuit.	1	II	.75D .25P
	*6) Attitude Control Equipment Problems				
Flight III (30 Day) 1 Instance	• Excessive gyro offset; corrected by switching to back up gyro	Gyro offset in ground test was caused by contamination of the gyro fluid which introduced excessive friction.	1	II	.2G .8M
Flight III (30 Day) 1 Instance	• Boom which holds Magnetometer required for yaw deorbit deploy-failed to extend within 5° of deployed position due to high bearing friction.	Similar problems were encountered in ground test. The changes introduced for correction did not prove adequate for operation in the vacuum environment of orbit.	1	II	D

TABLE 3-4 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHTS OF BIOSATELLITE VEHICLES (Contd.)

Failure Transmittal Numbers	Type of Failure/Problem	History, Significance, Comments, Etc.	Number	Class	Mode
EFFECTS WHICH COMPROMISED FLIGHT OPERATIONS (ITEMS 4 THROUGH 7 INCLUSIVE)					
	*7) In Flight Leakage Problems				
Flight III (30 Day) 1 Instance	• Decrease of 1/2 psi in regulated nitrogen pressure after attitude control operates.	Test problems of the Regulator were caused by contamination, test damage to diaphragm, and material transfer between titanium seat and tungsten carbide ball valve due to relative motion in vibration.	1	II	.5G .5D
Flight II (3 Day) 1 Instance	• Leakage of A/C gas after separation of capsule	The leak occurred after separation of the capsule most likely at the quick disconnect fitting.	1	1	G
EFFECTS WHICH COMPROMISED DATA (ITEMS 8, 9)					
Flight III (30 Day) 9 Cases	8) x Inconsistencies between count of food dispensed and quantity authorized by psychomotor game or command.	More than one failure in the psychomotor game (Electromechanical item). Suspected causes include: noise susceptibility of digital counting registers; irregular loss of commanded signal between Decoder and Controller; high resistances resulting from long cables of too-small wire size and large number of connectors and terminals. Mission risk was acknowledged.	9	II III	.6D .3P .1G
Flight I (3 Day) 3 Cases	*9) Failures of thermistors, presumably random. 2 prior to launch and 1 during flight.	Thermistor failures during acceptance test were not deemed a sufficient problem to warrant action.	3	I	P

TABLE 3-4 SUMMARY OF FAILURES AND ANOMALIES FROM FLIGHTS OF BIOSATELLITE VEHICLES (CONCLUDED)

MISCELLANEOUS EFFECTS (ITEMS 10 THRU 13 INCLUSIVE)

Failure Transmittal Numbers	Type of Failure/Problem	History, Significance, Comments, Etc.	Number	Class	Mode
Flight I (3 Day) BIOS-6-52-8	*10) Low relative humidity due to condensation on cooling tubes.	Temporary out of spec. conditions at beginning of mission with rapid regulation into specified range.	No Fail.	(IV)	
Flight III (30 Day) 1 Instance	*11) Capsule temperature noted as slightly below spec. limit. Re-calibration after recovery shows operation was within range.	Interface definition for primate did not cover all potential heat loads. Water vapor output of primate was in excess of any previous experience, system over cooled.	No Fail.	(IV)	
Flight II (3 Day) 7 Known Incidents	*12) Out of spec. angular rates due to out gassing of S/C with A/C not operating.	This was a secondary problem since the A/C ould not be turned on early in the mission due to the command problem noted in item 4, however the extent of out-gassing is shown.	No Fail.	(IV)	
Flight I (3 Day) 1 Instance	13) Lo deorbit spin rate. Spin and despin nozzles partially masked by structure, full impulse of jet not available for spin up, and de-spin. Correct by mixing N_2 with Freon, more dense efflux.	Biosatellite used a well established design for the spin-despin system, however no effective ground testing for this system could be performed,	1	IV	D
Flight II 3 Day Incident	14)* x Tape Recorder (Recovered post test evaluation of data shows section of tape with low signal levels, data degraded.	Ground testing of the tape recorder had shown 3 cases of tape degradation	1	II	D

TABLE 3-5 SUMMARIES OF FAILURES AND ANOMALIES FOR THE FLIGHT OF NIMBUS B

Nimbus Monthly Report Identification		Type, Description of Incidents	Previous History, Comments	Number	Class	Mode
		1) A total of six parts have failed.				
1		• Pressure transducer in the RTG at launch	Nimbus used commercial parts in portions of the spacecraft	1	I	D
13		• Thermistor in electronics bay (100 orbits)		1	I	P
20		• Zener diode changed reference (400 orbits)		1	I	P
22, 25	x	• Evidence of capacitor failure (519, 825)	Capacitors eventually degraded output of one channel in the IR spectrometer.	2	I	P
40		• Transistor. Located in infrared science (Orb. 1900)		1	I	P
	*	2) Tape Recorder Problems: Nimbus carries 4 assemblies 2 redundant pairs. One pair stores PCM data, (1 KBPS), the second pair records and plays back at 66 KBPS	A total of 24 tape recorder problem of failure incidents occurred during ground preparation.			
2		• PCM Recorder #2 stops during playback. Suspect - jam, motor, or clutch (Orbit 10)	The PCM units were late in the program change-outs due to non performance of the original design.	1	II	G
30, 36, 37		• PCM Recorder #1 degrades to non operable orbits 1203 to 1886.		1	II	D
39	x	• High speed recorder #1 data flutter problems begin orbit 1100		1	II	D
44, 45	x	• Both high speed recorders sustained degraded playback due to tape head contamination.		1	II	G
		3) Transmitter-receiver isolation margin.				
3		Command receiver saturates when simultaneous transmit-receive operation unintentionally attempted. Receiver has sensitivity greater than specification, responds to low energy sidebands from telemetry transmitter.	Nimbus spacecraft were not designed for simultaneous transmit-receive; operational error.	No Fail.		
	*	4) Radiation Effects. Early in the Nimbus III flight a solar flare incident occurred, radiation levels were high for a short period.	These incidents indicate insufficient margin within the design to accommodate the solar radiation environment.			

TABLE 3-5 SUMMARIES OF FAILURES AND ANOMALIES FOR THE FLIGHT OF NIMBUS B (Contd.)

Nimbus Monthly Report Identification		Type, Description of Incidents	Previous History, Comments	Number	Class	Mode
		4) (cont'd)				
4	x	• Ultra violet sensors degraded; 3 of 5, in the instrument, 2 began to fail after 3 days, the 3rd after 8 days.		1	IV	.8D .2P
6		• Neon reference in IR science degraded.		1	IV	.8D .2P
5,8		• Sensor housings over temperature		No Fail.		
7		• Neon reference shows noise bursts passing thru South Atlantic radiation anomaly.		1	IV	D
	*	9) Day-night transition attitude error effects: Attitude perturbations occur in the course of transition due to switching of mode control, etc.	No record from system test. Causes attributed to pneumatic and inertia errors. These problems were transitory and did not impair system operation.			
9,24		• Sun saturates IR Sensors for Science and horizon control (orbits 30, 790, 1208)		1	IV	D
34, 47		• Gyro switching unexpectly. 1345, 1404, 1455, 3573		1	IV	D
	*	10) Spurious switching effects	No previous history.			
10, 11, 15, 19	x	• Paddle wheel motor regulator switches to back up unit upsetting clock and telemetry. 4 times	These events all occur near the equator at locations over Borneo and Brazil. Apparent unknown RF environment imposes spurious commands.	1	IV	D
26, 32	x	• Clock resets to zero. 15 times.		1	IV	D
12, 14		• FM demodulator shifts to redundant mode approximately one time each 18 orbits		1	IV	D

TABLE 3-5 SUMMARIES OF FAILURES AND ANOMALIES FOR THE FLIGHT OF NIMBUS B (CONCLUDED)

Nimbus Monthly Report Identification	Type, Description of Incidents	Previous History, Comments	Number	Class	Mode
	*16) S Band Transmitter, High Speed Data Link (66 KBPS)	Ground test showed 6 incidents with the S band transmitter and 9 with the S band antenna.			
16, 21	• Power level makes sudden drop of 2 db. orbit 337		1	IV	X
	• Evidence that a capacitor lead has opened in the first stage cavity.		1	III	.7D .3M
	17) Solar array drive effects:	Ground test recorded 5 incidents with the solar array drive and noisy telemetry from potentiometer.			
17, 18, 23, 27, 35, 38, 41, 46	• Drive amplifier saturates and operates erratically. Suspect dirty potentiometer in feed back circuit or dirty slip rings. (Progressive from Orbit 317)		1	II	.8P .2D
43	• Telemetry from the Solar Array Drive position indicator became noisy. Suspect worn potentiometer (Begin Orbit 2300)	Inductive devices appear more suitable for this type of application.	1	II	P
28	28) Synchronization pulse increased in amplitude until a phasing problem occurred. Accommodated by ground station adjustment (Orbit 1138)		1	II	D
29	29) Telemetry from the attitude control section thermal shutter became erratic at about Orbit 1000		1	I	.5P .5G
33	* 33) Infrared inteferometer spectrometer fails upsets clock, turns off one battery. x Item shows noise but not signal. Suspect Sensor. Orbit 1332.	No previous history	1	IV	.8P .2M
42	*42) Noise level in Gyro Telemetry doubled by Orbit $\approx$ 2200	This effect occurred in an earlier flight just prior to gyro failure.	1	II	.8G .2M

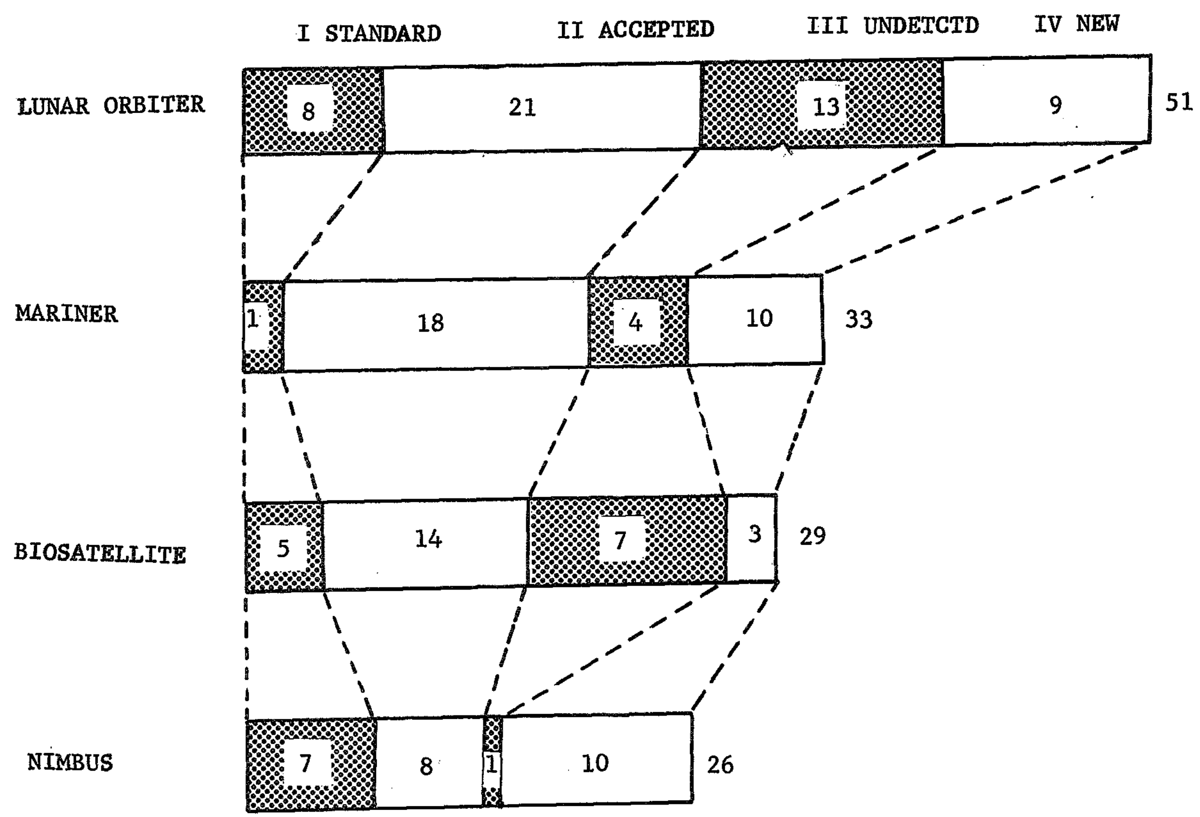
TABLE 3-6 SUMMARY OF FLIGHT PROBLEMS EXPERIENCED ON MORE THAN ONE PROGRAM

Effect	Appearance	(Table Ref.)	Class
1. Canopus star sensor stray or spurious light effects. Operational problems encountered.	Lunar Orbiter	3-2 Item 1	IV
	Mariner	3-3 Item 4	I, IV
2. Rate gyro anomaly, motion reference degraded or lost, (Evidence of internal damage to unit)	Lunar Orbiter	3-2 Item 3	II
	Biosatellite	3-4 Item 6	II
	Nimbus	3-5 Items 9, 42	IV, II
3. Spurious commands, either unwanted signals are introduced into the command link, or an on-board effect negates the execution of a command.	Lunar Orbiter	3-2 Item 4	II
	Mariner	3-3 Items 2, 16	II
	Biosatellite	3-4 Items 4, 5	III, II
	Nimbus	3-6 Item 10	IV
4. RF fluctuation or perturbations, transmitted power and frequencies show drift change and/or step changes.	Lunar Orbiter	3-2 Item 6	II, III
	Mariner	3-3 Items 1, 2, 7	II
	Nimbus	3-5 Item 16	IV
5. EMI perturbations to spacecraft operation, appearing as spurious or degraded data from telemetry or unwanted actions caused by internal sources or disturbances.	Lunar Orbiter	3-2 Item 7	II, III, IV
	Mariner	3-3 Items 2, 10	II, III
	Biosatellite	3-4 Item 5	II
6. Thermal Effects: temperature control of the spacecraft impaired or difficult. Design of the system did not have sufficient operating range.	Lunar Orbiter	3-2 Item 11	II
	Biosatellite	3-4 Items 10, 11	IV
	Nimbus	3-5 Item 4	IV
7. Sensor anomalies. Problems include inadequate dynamic range; inadequate for the operating environment; incomplete ground test; data loss	Lunar Orbiter	3-2 Item 12	III
	Mariner	3-3 Item 6	I
	Biosatellite	3-4 Item 9	I
	Nimbus	3-5 Items 4, 33	IV
8. Leaking valves and regulators	Lunar Orbiter	3-2 Item 15	I, II
	Biosatellite	3-4 Item 7	II

TABLE 3-6 SUMMARY OF FLIGHT PROBLEMS EXPERIENCED ON MORE THAN ONE PROGRAM (Continued)

Effect	Appearance	(Table Ref.)	Class
9. RF links compromised in operation by loss of lock.	Lunar Orbiter	3-2 Item 7	II
	Mariner	3-3 Item 5	II
10. Polarity reversal compromises data or operations	Mariner	3-3 Item 6	III
	Biosatellite	3-4 Items 2, 4	III
11. Extraneous out gassing or release of gasses perturb the spacecraft in altitude or velocity.	Mariner	3-3 Item 3	IV
	Biosatellite	3-4 Item 12	IV
12. Tape recorders. Data degraded or lost; recorder units degraded or failed to operate.	Mariner	3-3 Item 15	II
	Biosatellite	Post flight analysis of re-covered units. 3-4 Item 14	II
	Nimbus	3-5 Item 2	II
13. IR Horizon Sensors perturb attitude control system under particular circumstances	Biosatellite	3-4 Item 3	IV
	Nimbus	3-5 Item 9	IV

FIGURE 3-1 SUMMARY OF DATA, FLIGHT FAILURE CLASSIFICATION



4.0 FAILURE MODES

The previous section presented details of failure mechanisms which caused flight failures. To organize the data into useful tabulations, the following modes of failure were assigned as appropriate. The parenthesized letter for each mode is the corresponding code used elsewhere in this report for brevity.

- Design Problems (D) - An incident attributed to a shortcoming in the design of an item such that a change to the design was required to eliminate or minimize the problem.

- Electrical Part Failure (P) - A hardware failure correctable by replacement of an electrical part with another of the same or equivalent type and requiring no circuit design change.

- Device Failure (G) - A hardware failure correctable by replacement of an electromechanical or mechanical item with another of the same or equivalent type.

- Manufacturing (M) - A hardware failure caused by improper producing operations or processes.

- Test (T) - An incident or irregularity incurred during a test, originating from sources under the control of test organizations (and not directly attributable to the hardware under test).

- Miscellaneous or Unknown (X) - An event or anomaly not attributable to one of the preceding modes or inconclusively described or analyzed.

These failure modes classify the incidents as to the agency responsible for the deficiency, and thus support decision making to prevent recurrence of the failures. For example, Design Engineering and Reliability clearly must resolve the design deficiencies and must also evaluate the part and device failures to determine whether the reliability of these items is sufficiently high to support program goals. Similarly, manufacturing failures require corrective and preventive action by manufacturing and quality operations. Test failures require corrective action by the test and/or quality organization in test operations, facilities, or the preparation of test procedures. The miscellaneous or unknown category merits special attention by all parties to resolve uncertainties.

These failure modes are identified for flight as well as ground test problems. The addition of mission operational difficulties due to the communication interface is considered a design problem since it requires engineering action.

The failure mode distributions by program are presented in Figures 4-1 and 4-2, and evaluated in Sections 4-1 and 4-2 herein. This data has led to the following general observations:

1. Manufacturing and test failure modes generated half or more of the failure incidents.

2. Design deficiencies were a major effect; the uncovering of these deficiencies continued across all phases of test operation and persisted into flight.

3. The most critical or mission limiting flight problems were design deficiencies known from ground test.

4. Part and device failures were not a major quantitative cause of ground test failures, however they became more significant in flight.

5. Continuing type programs did not enjoy fewer flight anomalies.

6. Relative to environments, the majority of failures were uncovered in functional testing under ambient conditions.

7. In flight, 10 of the 11 vehicles studied completed their missions; however, all experienced serious malfunction or degradation to varying degrees.

4.1 Failure Modes in Space Flight

In Section 3 the flight failures and anomalies of the four programs are described, together with the cause of each. Tables 3-2 through 3-5 additionally identify the failure mode for each incident. In most cases these were identified by the project records or by contact with project personnel. In a few cases a single failure mode selection could not be determined with certainty, and prorated probabilities were assigned to each possible mode, based on judgment. The combinations were so determined that the total of all the mode probabilities for a given incident was unity. The application of this technique minimizes individual errors which tend to be averaged out when a number of incidents are involved. In all cases subjective interpretation was kept to a minimum.

Table 4-1 and Figure 4-1 present a summary of the flight failure mode distributions by percent for each program studied. The numbers of failures and of flights are also included. The Test (T) failure mode does not appear in this table since it is not applicable to flight failures.

The distributions in Table 4-1 and Figure 4-1 clearly indicate that Design (D) failure modes quantitatively dominated flight incidents. Mariner experience was based on reported Problem/Failure reports without the more definitive data available for the other programs studied, so that a relatively high percentage (21%) of the flight incidents were classified Unknown (X). For that program it is also notable that no flight failures could be attributed to Devices (G).

The Biosatellite flight failure mode distribution presented in Table 4-1 and Figure 4-1 was split in two, based on the two phases of the program in which the 3- and 30-day flight spacecraft were developed and flown. Part failures of the former were thermistors which were overstressed in the sterilizing process due to their mounting arrangement in specimen bottles. Many were used due to the nature of the science experiments. The relatively high ratio of Design (D) failures in both program phases was due largely to interface problems caused by novel mission and payload requirements such as deorbit from random attitude and biological maintenance of a primate, respectively.

Finally, note the high percentage (32%) of Nimbus part failures (P). This is believed to result from the use of commercial parts on the program.

Consideration of Table 4-1 and Figure 4-1 and the details from which they were developed, together with the qualifications noted immediately above reveal the following:

a. Design failures persisted into flight operations to a major extent.

b. Many of the design deficiencies uncovered in flight had been apparent in ground test; for example:

- EM phenomena, Biosatellite, Mariner
- Tape Recorders, Nimbus
- False Commands, Lunar Orbiter, Biosatellite
- Camera Doors, Lunar Orbiter
- RF Locks, Mariner
- Psychomotor Tester, Biosatellite

c. Maturity expressed as a continuing series of flights does not necessarily reduce the incidence of flight anomalies (compare: Nimbus with 26 for 1 unit and Mariner with 33 with 2 units to Lunar Orbiter with 51 in 5 flights and Biosatellite with 29 in 3 flights).

4.2 Failure Modes in Ground Testing

In addition to the determination of flight failure modes discussed in the preceding section, this study included the analysis of ground test failures of the same four programs. Addition of the latter provided the following information:

- Indication of the extent to which problems encountered in ground test lingered to affect flight (a significant factor, as noted in Section 3).

- Determination of the effectiveness of the testing program in screening out latent defects.

- Appraisal of the adequacy of design fixes in solving problems arising in ground test.

- Additional background data relative to specific equipments.

- A larger data base than available from flight to permit a broader and more accurate perspective of program problem areas.

Processed details of ground test failure data were available for the Lunar Orbiter and Biosatellite Programs. Nimbus data was obtained from the vehicle logbooks augmented by contact with the responsible contractor subsystem engineers. The Mariner material consisted of preliminary incident descriptions which were adequate in most cases for indicating failure causes. The failure descriptions from all these programs were studied with respect to equipment design and complexity (see Section 6) and to the nature of the test program (see Section 7) and other project constraints in order to assign failure modes to the ground test incidents.

4.2.1 Relative Occurrence of Failure Modes

Table 4-2 presents a summary of the ground test failure modes by percent for each program studied. The total number of applicable failures for each is also listed in the bottom row. The format of this table is the same as Table 4-1, with addition of the Test (T) failure mode which is here applicable, and which in fact is the most repetitive. Note that the failure modes for all programs follow a generally similar pattern of distribution.

The Nimbus ground test failure mode distribution departs appreciably from the pattern of the other programs because this data includes system level assembly and testing only. For this reason the Nimbus ground test data was omitted from Table 4-2.

The individual program failure mode distributions are plotted in bar chart form in Figure 4-2 except for Nimbus which was omitted for reasons explained in the preceding paragraph. The Lunar Orbiter and 3-Day Biosatellite patterns presented in Figure 4-2 are very similar and typical of programs involving adaptations of previous experience as well as development of new concepts in about equal degree. Interestingly these two programs pursued closely related chronological paths.

The relatively large percentage of the Part (P) failure mode for Mariner 69 is believed to be due to the extensive use of integrated circuits.

The relatively high percentage for the 30-Day Biosatellite Test (T) mode is largely due to the difficulties in testing biological interfacing hardware. The fact that many of the process problems were solved in the previous 3-Day program phase accounts for the comparatively low level of the Manufacturing (M) mode.

4.2.2 Failure Modes vs. Time

Figures 4-3 through 4-7, inclusive, show the incidence of failures by mode plotted against calendar time during the Project (dates were taken from failure report documents). The shape of these curves are generally similar, all showing a trend towards leveling off as launch dates approach. In the case of Biosatellite 3-Day (Figure 4-5), the curve shows the least leveling prior to flight, and this was the unit which failed to complete a mission. The 3-Day Biosatellite covers the delivery of 2 flight units. The second unit was flown (with some modifications) in September of 1967. Nimbus B (Figure 4-7) shows the effect of the 2 vehicles, with the leveling effect for the second unit somewhat less than for the first; this effect has been attributed to the use of reworked hardware. The common shape of these curves suggest that a normalizing process can be applied and a characteristic shape described for a program. Coupled with predictions for totals of failure incidents, such a characteristic can allow a real time assessment of project status as data is received from test operations.

4.2.3 Failure Modes vs. Environment

The obvious pattern disclosed by this study has been the predominance of failures uncovered by functional testing under ambient conditions as indicated in Figure 4-8, wherein the failure distributions by environment are shown. This follows a deductive logic, in that many design deficiencies and manufacturing mistakes (mis-wiring, poor bonds, etc.) do not need environmental inducements to inhibit operation. Within the study, these failures largely appeared during the functional operational phases prior to environmental exposures. In addition to the ambient conditions, vibration and thermal vacuum environments show about equal effect for uncovering failures and both appear to be effective screening concepts.

TABLE 4 - 1 SUMMARY OF FLIGHT FAILURE MODE DISTRIBUTIONS

FAILURE MODE		LUNAR ORBITER	MARINER '69	BIOSATELLITE 3-DAY	BIOS. 30-DAY	NIMBUS B
DESIGN	D	40%	56%	68%	40%	48%
MANUFACTURING	M	32%	14%	6%	6%	3%
PARTS	P	17%	9%	20%	42%	32%
DEVICES	G	9%	0	6%	12%	13%
NOT EXPLAINED (UNKNOWN)	X	2%	21%	0	0	4%
TOTAL %		100%	100%	100%	100%	100%
TOTAL NUMBER		51	33	16	13	26
NUMBER OF FLIGHTS		5	2	2	1	1

TABLE 4 - 2 SUMMARY OF GROUND TEST FAILURE MODE DISTRIBUTIONS

FAILURE MODE		LUNAR ORBITER	MARINER '69	BIOSATELLITE 3-DAY	BIOS. 30-DAY
DESIGN	D	26%	25%	35%	34%
MANUFACTURING	M	30%	20%	24%	15%
PARTS	P	4%	12%	1%	1%
DEVICES	G	1%	6%	1%	1%
TEST	T	35%	26%	32%	45%
MISCELLANEOUS	X	4	11%	7%	4%
Total %		100%	100%	100%	100%
Total Quantity		1514	1059	483	1861

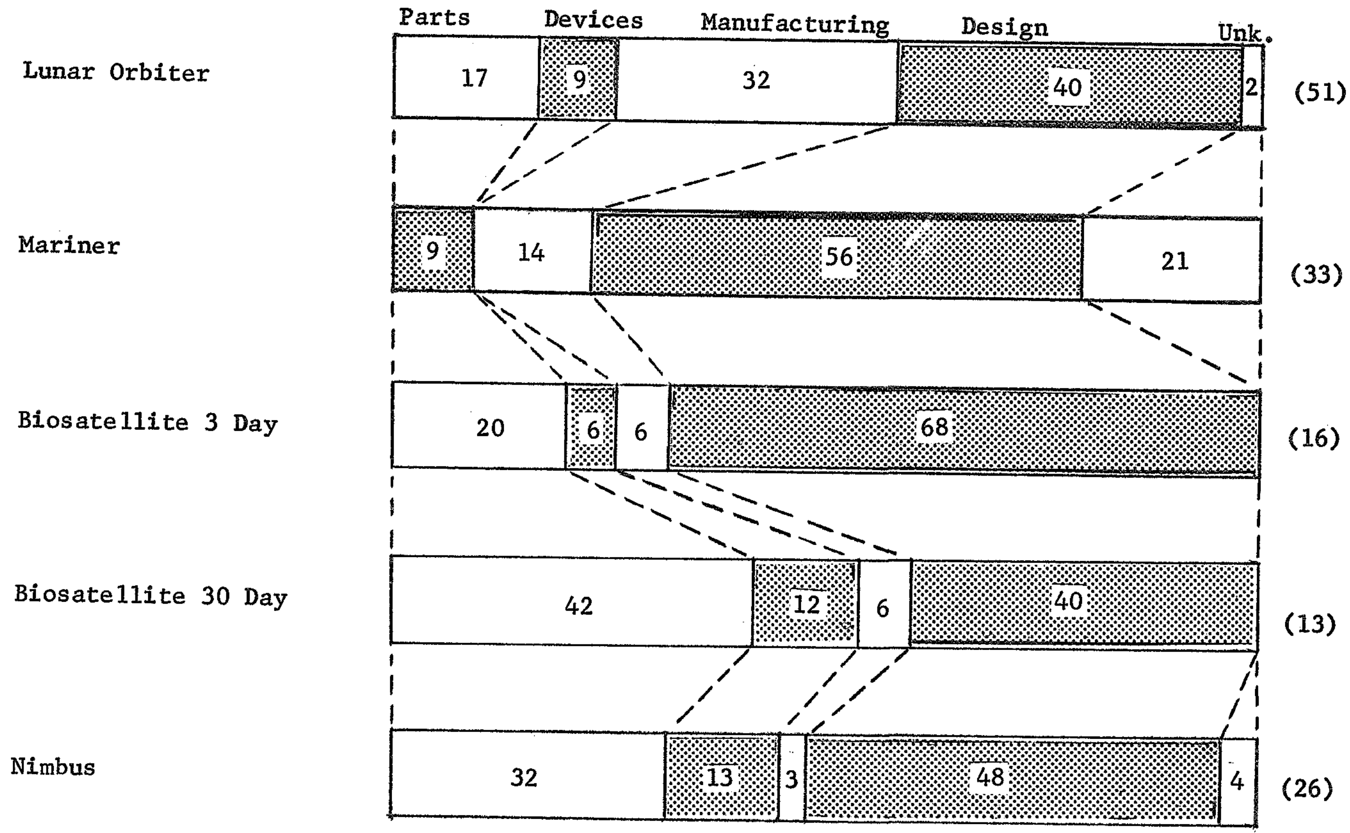
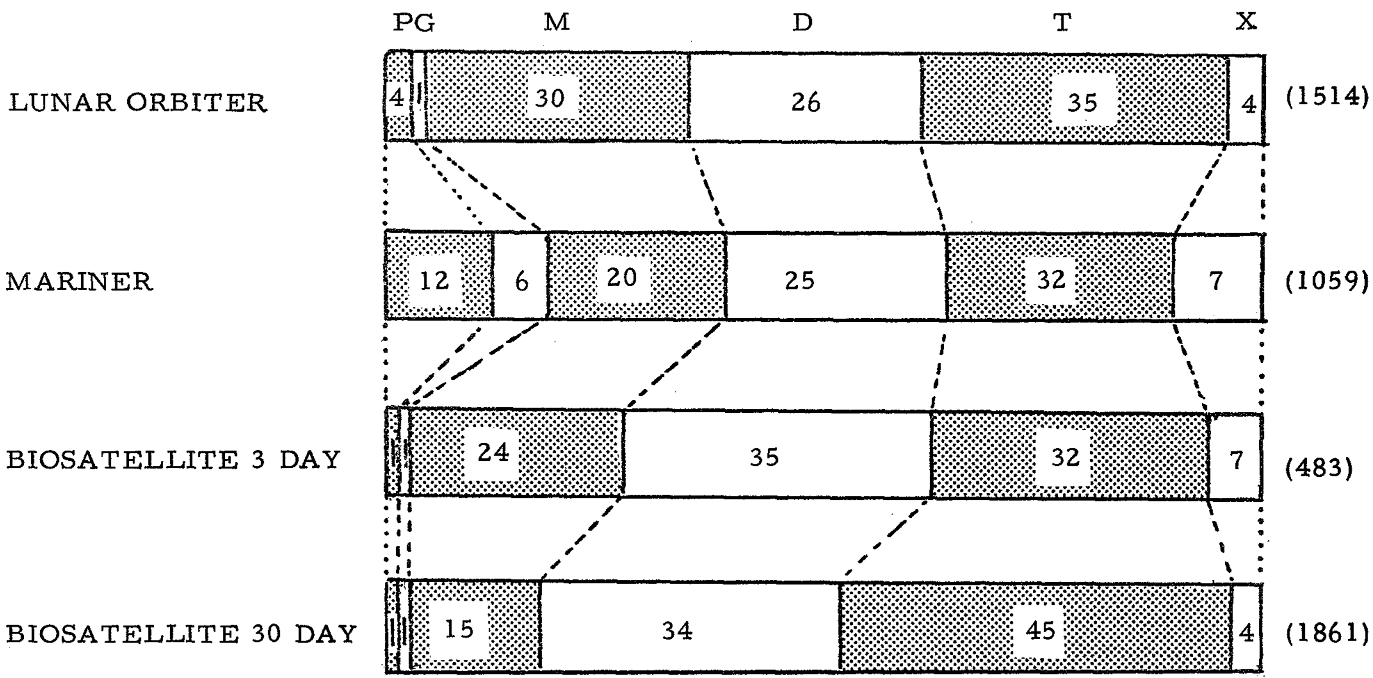


FIGURE 4-1 SUMMARY OF FLIGHT PROBLEM/FAILURE MODE DISTRIBUTIONS, PERCENT

FIGURE 4-2

DISTRIBUTION OF GROUND TEST PROBLEM/FAILURE MODES, PERCENT



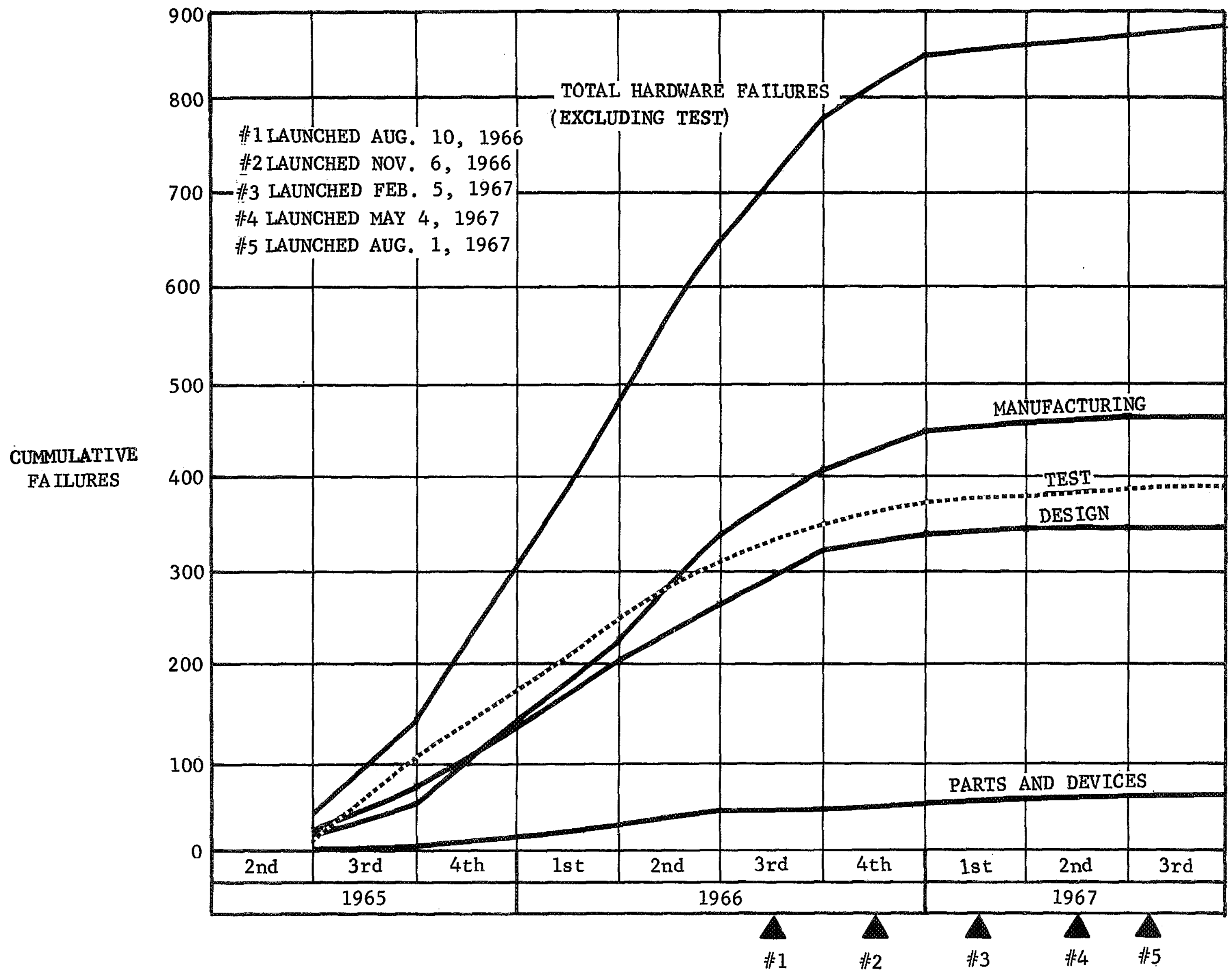


FIGURE 4-3 FAILURE TIME HISTORY, LUNAR ORBITER.

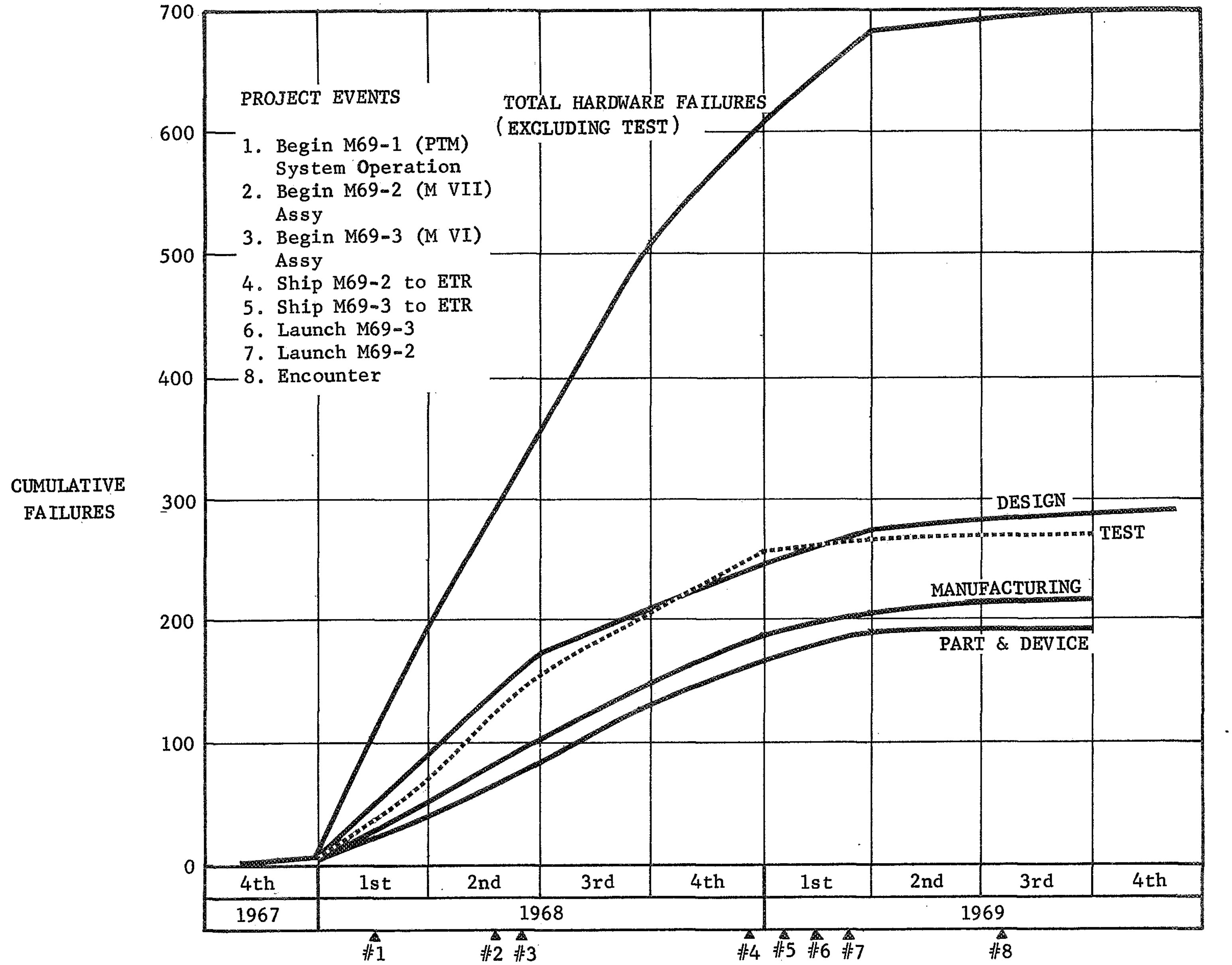


FIGURE 4-4 FAILURE TIME HISTORY, MARINER '69.

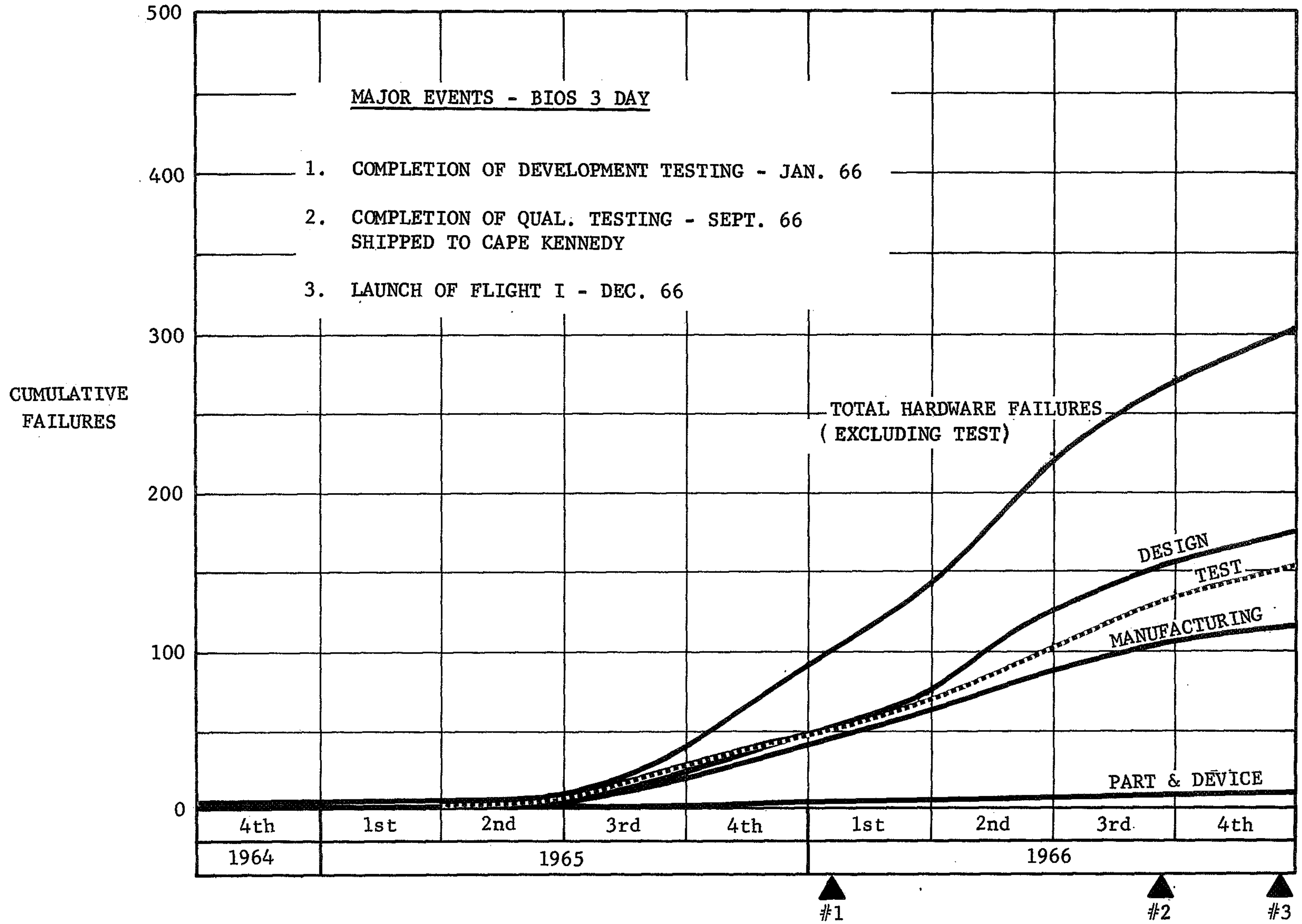


FIGURE 4-5 FAILURE TIME HISTORY, BIOSATELLITE 3-DAY.

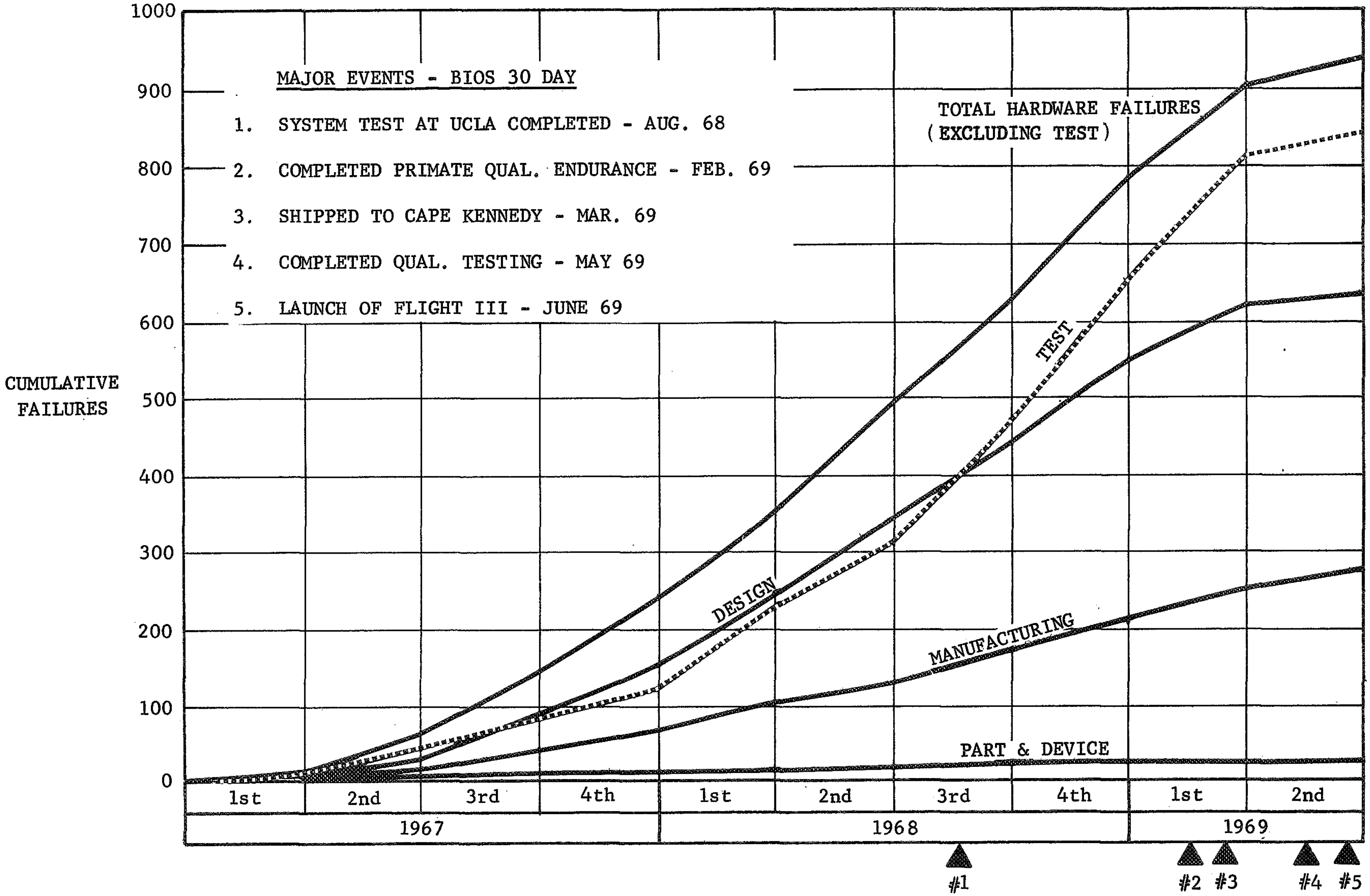


FIGURE 4-6 FAILURE TIME HISTORY, BIOSATELLITE 30-DAY.

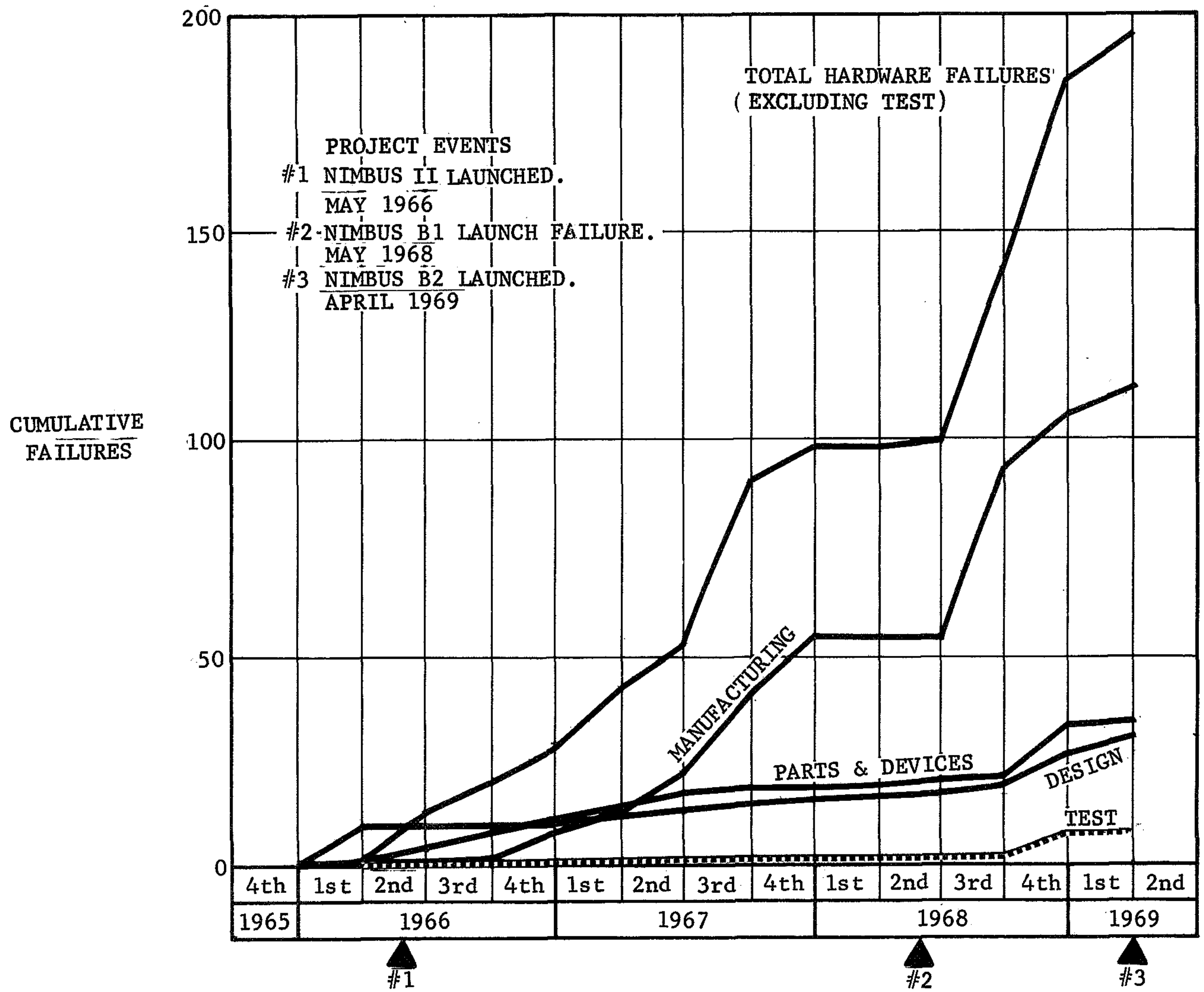
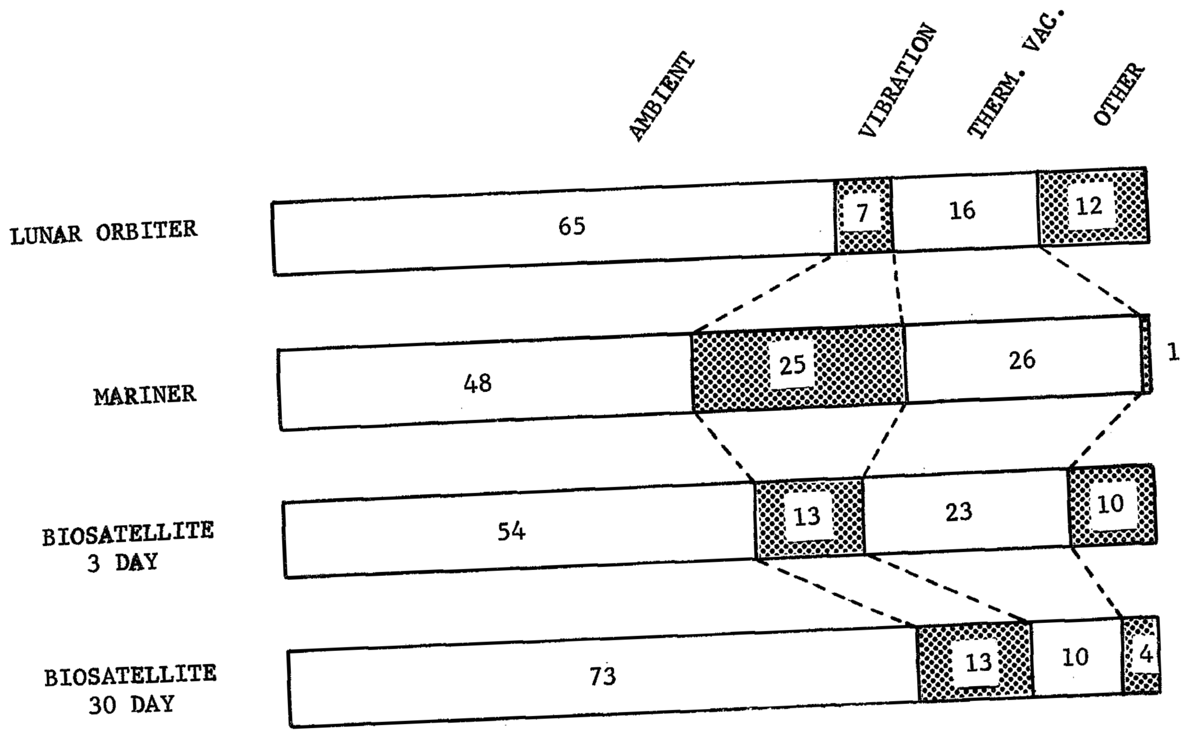


FIGURE 4-7 FAILURE TIME HISTORY, NIMBUS B.

FIGURE 4-8 DISTRIBUTIONS OF GROUND TEST FAILURE BY ENVIRONMENTS



NOTE: MARINER DISTRIBUTION DOES NOT INCLUDE SPACECRAFT TESTING

5.0 EXPERIENCE APPLICABLE TO THE VIKING LANDER

This section describes potential flight problems for the Viking Lander based upon the data from the four space projects in this study. The problems are discussed in terms of failure mechanisms, causes, and effects. These data are presented in 5 tables which are listed below and described briefly in the following paragraphs.

Table 5-1	Interface Problems
5-2	Experience with Advanced Hardware Proposed for the Viking Lander
5-3	Persistent Problems
5-4	Unique Incidents
5-5	Extended Descriptions

5.1 Interface Problems

A total of 9 interface problems are presented in Table 5-1. These problems all had causes which originated in one component or subsystem and had effects elsewhere in the spacecraft. In some cases the observed incidents were the end result of a series of contributing causes or defects. The appearance of these problems within the Viking Lander would impair the ability to satisfy all the mission objectives. Within the table the problems are presented as follows:

- The problem is described and supported by an illustrative example.

- The impacts to the projects are shown by number of incidents or the flight which was compromised.

- The generating causes are briefly identified and followed by a description of the flight work-around or similar action.

- The application to the Viking Lander is shown by identifying the area of principal concern.

- References are provided to the extended descriptions in Table 5-5.

5.2 Experience with Advanced Hardware Proposed for the Viking Lander

The experience from ground test and flight for 9 types of spacecraft hardware are presented in Table 5-2. All of these will be employed in the Viking Lander. For each type of hardware the table describes:

- The suppliers, the flight spacecraft, and the functional use within the spacecraft.

- Ground test problems, these are listed showing causes, effects, and degree of repetition.

- Flight experiences - these are shown as numbers of failures and descriptions of the failure incidents.

The items described in the table were the results of intensive development efforts within each of the projects. The degree of effort expended was no less than that being planned and applied to the hardware under development for the Viking Lander. For the items listed, technology development has continued and the problems described may have been resolved. However, in tailoring these types of equipment to fit the exacting requirements imposed by the Viking Lander, it is important to appreciate what has occurred in past programs in order to avoid repetition of previous problems.

5.3 Persistent Problems

Some types of problems have appeared in common and persistently to each of the projects in the study. Table 5-3 describes the 7 types of persistent problems encountered. Evidence indicates that where these problems were not resolved in ground tests they persisted into flight and compromised mission objectives. Attention to these potential problems during design and test can minimize their impacts upon both ground testing and carry over into flight. The persistent problem data are presented as follows:

- The problem is described in general and defined in terms of the principal forms or occurrences.

- Persistence is shown by number of incidents.

- The causes and the resulting impact upon the projects are described. The descriptions are shown in relation to the principal concern or most probable appearance in the Viking Lander.

5.4 Unique Incidents

Table 5-4 presents 19 one-of-a-kind incidents which have a potential for occurrence in the Viking Lander. These problems generally came as surprises; and in a number of cases they arose as a result of incorporating changes which were intended to eliminate other problems. The table has been arranged into 5 groupings of incidents under the headings of power, noise, communications, devices, and parts-materials. For each incident, the data are presented as follows:

- The incident is described showing the causes, the effects, and the project of origin.
- The similarity or potential for occurrence to the Viking Lander is identified.
- Supporting comments offer descriptions of the means by which the Viking Lander can avoid repetition.

5.5 Extended Descriptions

Table 5-5 describes 10 areas in which the data from the previous tables are expanded by extended descriptions; references to Table 5-5 appear throughout the first 3 tables of this section. These extended descriptions are coupled with check-list information recommended to the Viking Lander Project.

TABLE 5-1 INTERFACE EFFECTS, PROBLEMS COMPROMISING TO FLIGHT OPERATIONS

FAILURE OR PROBLEM DESCRIPTION	OCCURRENCE	CAUSE-EFFECT, ACTION TAKEN	VIKING COMMENT FURTHER DESCRIPTION
1. Spacecraft experienced spurious commands. Examples from flight. • Mariner VII executed a memory dump at separation (unwanted action) • Mariner VI did not execute a memory readout at command during post encounter (no action)	Biosatellite 11 Incidents During Flights. Mariner 4 Incidents During Flight Lunar Orbiter 2 Incidents During Flight	Transient pulses generated within the S/C were accepted as "bits" by the command circuitry. These bits altered active words or changed stored memory states and resulted in changed commands (unwanted action) or scrambled commands (action lost). The presence of transients pulses was known from ground testing. Flight work arounds have consisted of repetitive commands and reprogramming of memory storage.	Viking would experience impaired system control. Also entry and landing depends upon the stability of stored digital data. This effect is related to EMI. Further discussion appears as Table 5-5.1.
2. Spacecraft responded to commands addressed to another unit. Incomplete system isolation during multi-vehicle operation.	Lunar Orbiter 4 Incidents During Flight	During multi-vehicle operations, the Command RF made secondary locks with the other spacecraft. The low signal level at the secondary lock imposed over an extended period of time generated sufficient pulse responses to change stored commands and thus caused erratic actions. The sensitivity had been discovered in ground test and the system buffered to thresholds which were acceptable risks. For flight operation the work around consisted of increasing the power at the DSIF transmitter. The secondary lock then recognized the vehicle address and actively rejected transmission.	System Isolation recognized as a critical need. A secondary RF lock allowed spurious signals to enter memory storage and change stored commands. Further discussion appears as Table 5-5.2.
3. Polarity reversal effects, appear during flight Examples: • Spacecraft responded to commands from only one tracking station • G Switch discovered installed backward.	Biosatellite 3 Day Flights	The command receivers and decoders were redundant and cross linked. The transformers in the output of the receivers were unintentionally set to apply the signals 180° out of phase resulting in almost a complete cancellation. Low receiver output signals were known from ground test (See item 4 below) but not identified for cause. In flight operations the command station with access was operating with an off-frequency transmitter. The drawing for the G-Switch installation called out the wrong direction. The error was caught at the last minute by an engineer personally familiar with the switch operation.	The reversed transformers represent a unique case of redundancy introducing a mode for failure. Further discussion appears as Table 5-5.8
4. Flight problems masked by ground equipment or test effects Examples: • Low output from command receivers (from item 3) • Infrared radiometer data is degraded over portion of scan sweep.	Biosatellite II Mariner VI, VII at Encounter Lunar Orbiter 1 (Suspected)	In ground test, hardwire operation by-passed the offending transformers. For RF link operation in the factory, low signal output was a common problem across all vehicle programs. During countdown a poor RF link was cured by using the backup command test set (high signal levels) Mariners showed noisy data over portions of IRR scan during ground testing only when operated with AGE Cables in place Lunar Orbiter had evidence of AGE introducing a ground loop which suppressed a compromising EMI transient. There existed no capability to work around these effects	Viking must operate with a complex AGE system employing long connecting cabling, a large number of test interconnect cables, and a large number of breakout boxes. A discussion of test related effects appear as Table 5-5.10

TABLE 5-1 CONCLUDED. INTERFACE EFFECTS

FAILURE OR PROBLEM DESCRIPTION	OCCURRENCE	CAUSE-EFFECT, ACTION TAKEN	VIKING COMMENT FURTHER DESCRIPTION
5. RF links for command and telemetry are incompletely isolated. Examples: . Mariner makes an internal self lock during cruise . Lunar Orbiter loses lock from frequency shift . Biosatellite and Nimbus Harmonic of telemetry bit rate matches command address.	Mariners 4 Incidents During Flight Lunar Orbiter 1 Incident (LO-V-A1) Biosatellite I 2 Incidents Nimbus 1 Incident During Flight	Mariner: Spurious RF signals entered the receiver lock circuitry thru a feedback loop in a power supply. False locks occurred particularly in ranging mode operations. Lunar Orbiter: During multi spacecraft operation the RF lock circuitry was operated at an off design point. A "ranging mode-off" command from ground caused a frequency shift which broke the RF locks. Biosatellite, Nimbus: The command address frequency was the 3rd harmonic of the telemetry bit rate. The effect circumvented the RF isolation and spurious bits occurred in the decoder address words. Work Arounds: Mariner - Limit ranging mode operation; Lunar Orbiter - Frequency search for reacquisition; Biosatellite, Nimbus - Command and telemetry not operated at the same time.	These problems and the resulting work-around are not compatible with the long delay time RF trans-missions inherent in the Viking Mission. Further discussion appears in Table 5-5.3.
6. High Voltage Breakdown at reduced pressure. • Photo System - LO-III • Video Interrupt - LO-V	Lunar Orbiter 2 Incidents in Flight	Flight interrogation of the photo system high voltage measurement showed a sudden drop which later self corrected. The video interrupts were caused by loss of 1000 volts impressed upon a screen grid. Corona and arcing related phenomenon were suspected for these incidents. Arcing can be self healing, and Lunar Orbiter had experienced arcing at low pressures during ground test of the photo system. There was no capability to work around these effects.	Viking science experiments and communications items must operate with high voltages in reduced pressures for an extended period of time. Further discussion appears as Table 5-5.4
7. Spacecraft switches to backup modes due to localized RF energy entering the command system. The local RF fields have intensities and frequency content sufficient to simulate an uncoded "tone" type command.	Nimbus Continuous Incidents in Flight	Nimbus B achieves a full earth weather coverage by means of a polar orbit. Anomalous switchings to the backup power regulator system have occurred during sweeps over Borneo and the South Atlantic near Brazil. Considerable RF energy exists over these areas and for no apparent reason. (The Asian War Zone is suspect for the Borneo incidents, but the South Atlantic Anomaly has no explanation). Work around has con-sisted of a reset by ground command.	Unique events. Identifies an environmental condition present in near Earth operation. Further discussion appears in Table 5-5.4
8. Infrared Horizon Sensor unable to distinguish Earth-Space Interface. Flight Attitude Precision Lost.	Biosatellite Flight I	The design of the IR Sensor failed to account for the presence of cold cloud surfaces over portions of the Earth. These local cold spots perturbed the warm earth-cold space interface as seen by the sensor and resulted in hunting and attitude confusion. The horizon sensors provided 2 of the 3 references for de-orbit positioning. The error introduced by this effect had to be accepted in the entry path angle.	System design requirements were incompletely defined. Further discussion appears in Table 5-5.4
9. Ultra Violet Radiation Sensor degrades in 8 days. Lunar Orbiter thermal paint degrades.	Nimbus Science Equipment Lunar Orbiter Flight Incident LO-I-2	The sensor units of the Nimbus "Monitor of Ultra Violet Solar Energy" (MUSE) experiment was intended to measure over 5 broad spectral bands in the UV range. UV energy began to degrade the sensors within 3 days. 3 of the 5 channels were lost after 8 days. The Lunar Orbiter thermal paint was intended to degrade slowly but maintain emissivities and absorptivities within limits for a period of one year. Limitations on test techniques masked a more rapid degradation than anticipated. Work around consisted of operating the vehicle at other than the most desired attitude.	Unique Incidents wherein a known environment was not accomodated in the equipment design. Further discussion appears in Table 5-5.4

TABLE 5-2 EXPERIENCE WITH ADVANCED HARDWARE PROPOSED FOR THE VIKING LANDER

EQUIPMENT	PROGRAM	USAGE	DEVELOPMENT & TEST EXPERIENCE	MISSION EXPERIENCE
1. TAPE RECORDERS				
Lockheed Continuous Loop Type	NIMBUS 1, 2, 3	Continuous recording and frequent (≥ once/orbit) playback of PCM telemetry data.	Three instances included broken shaft, leakage, head contamination.	Two instances involved a tape break and a stalled motor, failing both redundant units on Nimbus 3.
			Five years application on OGO.	
RCA Reel Type	NIMBUS 1, 2, 3	Continuous recording alternated between two recorders.	Development problems included head contamination, sticky tape, tape variability, friction of limit switch, broken drive belts, flutter, and leakage.	One instance of head contamination and flutter.
Cook Reel Type	BIOSATELLITE II	Periodic recording during orbit and descent with no playback during mission.	Three instances of tape degradation, and one instance each of tape loosening in vibration, electrical grounding problem, ball bearing friction, brake shoe material problem, and motor wearout	Tape was wrinkled by transport guides and rollers. Transport mechanism was redesigned for 30 day flight.
Lockheed Continuous Loop Type	MARINER 69	Analog & digital recorders served as buffers between high data rate recording and low data rate playback.	Three instances of drive tape leaving reels; twelve instances of head contamination by oxide and/or graphite; 2 instances of degraded high frequency response; and one instance each of noisy digital output, analog output level change, degraded analog high frequency response, and over spec. pressure.	Analog recorder erased some inputs and output was reduced by half. One track of digital tape recorder was not readable.
See Table 5-5.5				
2. RTG - SNAP 19 See Table 5-5.7	NIMBUS 3	Experimental test of SNAP 19 performance.	Thermoelectric sublimation due to argon leakage, deterioration of high temperature thermoelectric bond, and contamination from insulation.	One instance of output degradation of 21-1/2% after 10,000 hours due to the three causes noted in development experience in unknown proportions.
3. Memory Cores	NIMBUS 3	Memory for IRLS experiment.	No problems reported.	Interruptions of data less than 1% due to input loss from unknown cause.
	LUNAR ORBITER	Memory for Attitude Control Computer.	One instance of a cracked memory core due to mis-handling.	No problems identified.

TABLE 5-2 CONTD. EXPERIENCE WITH ADVANCED HARDWARE

EQUIPMENT	PROGRAM	USAGE	DEVELOPMENT & TEST EXPERIENCE	MISSION EXPERIENCE
4. RATE GYRO				
Sperry Gas Bearing	NIMBUS 3	Experimental test of gas bearing gyro performance and reliability.	Shaft contamination, high gimbal torque due to relaxed ligaments, motor failure due to warped thrust pads.	One instance of Excessive drift.
Kearfott Alpha Ball Bearing	NIMBUS All Flights MARINER	Rate Control Sensor.	One instance of non-repeatable torques on Mariner. No failures reported on Nimbus.	Two instances on Nimbus of bearing friction, one resulting in loss of the gyro. No failures reported on Mariner.
Nortronics Ball Bearing	BIOSATELLITE All Flights	Rate control sensor. Single phase for monitoring to conserve power and two phase for control accuracy.	One instance of broken flex leads, 2 instances of nul shift due to inadequate temperature compensation, 3 instances of contaminated flotation fluid, and one instance of fluid leakage.	No failures recorded
Sperry Ball Bearing	LUNAR ORBITER All Flights	S/C Attitude or Rate Control.	Two instances of fluid contamination due to a bubble and to soldering flux causing gimbal hangup, 2 instances of broken pivots, and 12 instances of mass unbalance cured by coil form redesign.	One instance of gimbal hangup from unknown cause.
See Table 5-5.6				
5. Lamp Bulbs - Incandescent	BIOSATELLITE III	Two sets of parallel redundant bulbs used alternately to establish day or night capsule lighting for the primate experiment and to permit photography.	The light bulbs originally specified did not withstand the vibration and required extensive redesign.	No problems reported.
6. S BAND TRANSMITTER	NIMBUS 3	Transmission of High Data Rate Stored Telemetry.	Three instances of failures of unreported causes.	One instance of Transmitter failure due to open capacitor connection to first stage cavity due to lack of lead attachment to supplement solder.

TABLE 5-2 CONTD. EXPERIENCE WITH ADVANCED HARDWARE

EQUIPMENT	PROGRAM	USAGE	DEVELOPMENT & TEST EXPERIENCE	MISSION EXPERIENCE
7. PYRO VALVES				
Lockheed N. C. Explosive Valve	Biosatellite All Flights	Redundantly pyro initiated to release cold gas for spin and despin.	About 1000 previously used in space vehicles. Yet on this program two instances occurred of external leakage due to rupture of the valve body by plunger tip particles which was cured by addition of a protective cover.	No failures reported.
N. O. Nitrogen Shut-off valve	Lunar Orbiter All Flights	Provides pneumatic isolation between Attitude Control and Velocity Control Subsystems to conserve nitrogen gas for the extended mission.	One instance of failure to close in confidence test.	Two instances of leakage through valve after closing.
N. C. Propellant Valve	Mariner 69	Maintains tank seal until stored hydrazine is required to operate engine.	One instance of low flow rate due to contamination after firing cured by changed manufacturing procedures.	No problems reported.
N. O. Scan Platform Unlatch Valve	Mariner 69	Unlatches Scan Platform.	One instance of post fire leakage through a gasket cured by redesign.	No failures recorded.
8. POTENTIOMETERS	MARINER 69	Scan position telemetry, TV resolution adjustment, and thrust vector control.	Six instances of high friction, three instances of assembly error, one instance of an open connection, one instance of vibration induced setting change, two instances of wiper-detent interference, and two instances of improper lubrication.	No known failures
	LUNAR ORBITER All Flights	Thrust vector control linearity	One instance of improper brush pressure.	No known failures
	NIMBUS 3	Thermal control shutter position indicators	Twenty six instances of excessive backlash.	No known failures
	BIOSATELLITE All Flights	Pressure transducers for O_2 and H_2 measurement (1 to 15 psia) Voltage adjustments	Continuous operation over a narrow pressure range and jitter caused local wear. Pressure data became erratic. One instance each of an open poteniometer, unstaked setting, and sheared bushing.	No known failures

TABLE 5-2 CONCLUDED, EXPERIENCE WITH ADVANCED HARDWARE

EQUIPMENT	PROGRAM	USAGE	DEVELOPMENT & TEST EXPERIENCE	MISSION EXPERIENCE
9. Microelectronic Parts MOS	NIMBUS 3	Used in redundant command memory.	"Swiss cheese" effect caused by breakdown in metallized surface; redundant cells both failed due to common electrical contact problem external to part.	Frequent clock abnormalies presumed as not directly attributable to MOS because they appear to be externally induced.
Texas Inst. IC's	NIMBUS 3	Part / Qty. / Type SN510/1 34 Mono. R-S Flip-Flop SN512/3/4 46 Mono. NAND/NOR Gate SN515 4 Mono. Exclusive or Gate	One instance of an IC failure.	No failures recorded.
IC's	MARINER 69	2678 IC's, mostly digital employed, throughout 14 subsystems	Process Control related problems continued throughout project: . Open circuits from metalization cracks . Intermetallic compounds between Au-Al, (Purple Plague) generated by case sealing temperatures . Contamination entrapped in lead wire (Cl) cause precipation of Al compounds at joints. Static discharges in handling damaged one series of IC, and forced replacement with less sensitive configuration.	Suspected contributor to flight problems.
IC's	BIOSATELLITE All Flights	108 IC's of 5 types were used in the Attitude Control Programmer logic and 100 in the Telemetry and Command Subsystem.	No failures reported	No known flight failures.
IC's	LUNAR ORBITER	620 IC's used in the Attitude Control Computer, 142 in the Command Decoder, and 9 in the Inertial Reference Unit.	Many development problems caused by lead migration from the glass seal and by cracks in the metalization. The latter accountef.or 4 test failures.	No failures reported.

TABLE 5-3 PERSISTENT PROBLEM AREAS

PROBLEM DESCRIPTION	OCCURRENCE	EFFECT ON PROGRAMS, ACTION,	VIKING CONCERN
1. Electromagnetic Interference Type Problems a. Pulse transients generated by switching and power transfer. b. Noise generated within S/C circuitry distorts telemetry. c. AGE pickup introduces noise and transients into circuitry. ▼ Centaur Telemetry radiated interference.	Program — Incidents L. O. — 12 Mariner — 46 Bios. — >25 Nimbus — None Reported L. O. — 7 Mariner — 36 Bios. — >10 Nimbus — Not Reported L. O. — Not Reported Mariner — ~60 Bios. — >20 Nimbus — Not Reported ▼ Mariner — 6	EMI problems were the major reason for circuit changes after the functional design had been established. These effects extended diagnostic testing and required retest after rework. Specifically for spacecraft testing, filters, suppressors or modifying correction totaled: ~ 10 identified for Lunar Orbiter ~ 14 described for Mariner 14 defined and incorporated to assure 6db operating margins in Biosatellite 30 day flight unit. In addition, all the programs had identified areas of known EMI susceptibility which were accepted as risk. The EMI problems for Nimbus were resolved in early spacecraft (Nimbus A and C); Nimbus B had no changes within the basic spacecraft subsystems. The experiments, as potential sources of EMI, were all buffered from the basic spacecraft. ▼ The Centaur RF affected the high impedance sensors and instrumentation circuitry within the Mariner Spacecraft. These leads had only modest shielding with single end groundings; RF coupling occurred at lower power densities than specified (the IRR showed sensitivity at 30db below specified levels).	EMI sensitivity of a similar nature potentially present for the Viking Lander in areas of • Stored commands necessary for entry and ground operation • High speed digital feed back calculations during entry • High speed switching of power for attitude control, deorbit, and descent landing • Digital data recording for PCM transmission. Within the Viking Lander Sensors and data lines necessary for launch and boost phase operations must be free of RF interference. Further discussion. See Table 5-5.1
2. Ringing and Oscillation Effects a. RF falselock or selflock effects b. Regenerative oscillation thru feedback loops such as power supplies. c. Beat frequency effects generated between related or interconnected items. d. Regenerative oscillations thru loops involving AGE. e. Regenerative oscillations thru loops involving mechanical equipment	Totals for all effects Lunar Orbiter 2 Flight 6 Ground Mariner 4 Flight 16 Ground S/C 22 Ground AGE Biosatellite 2 Flight 10 Ground Nimbus 1 Flight None Reported in Ground Test Mechanical: Lunar Orbiter Camera Door Biosatellite Coolant Pump	Regenerative oscillations within a spacecraft complex generally stop effective testing (or operations) until eliminated by some corrective action. However, the identification of an oscillation problem can be complicated by masking effects such as a threshold for signal levels (Mariner) or burned out parts (Biosatellite). In such cases the underlying cause escapes immediate detection and requires a deliberate search. Oscillation and ringing instabilities have required design modifications or corrective actions as follows: Lunar Orbiter — 4 Spacecraft circuit modifications. 2 flight work arounds Mariner — 6 Spacecraft circuit modifications, 4 flight work arounds (AGE not reported) Biosatellite 30 Day — 8 Spacecraft circuit modifications, with AGE, 2 Flight work arounds Nimbus — 1 Flight work around.	Viking areas with particular concern for ringing effects a. RF link isolation. S band Simultaneous receive-transmit, (Beat frequency susceptibility) b. Feed back attitude control, deorbit and entry. (Body Dynamics contributing) c. Terminal Descent, feedback control. (Body Dynamics contributing) d. AGE installation, Long cable operation Table 5-5.4 describes some specific incidents of ringing and oscillation problems. Further comment. See Table 5-5.3

TABLE 5-3 CONTINUED PERSISTENT PROBLEM AREAS

PROBLEM DESCRIPTION	OCCURRENCE	EFFECT ON PROGRAMS, ACTION,	VIKING CONCERN
3. High Voltage Breakdown. Corona and Arcing Effects.		Arcing and Corona effects generally appeared during the first tests at low pressures. Corrective action required redesign or rework. The effects persisted into flight equipment due to design changes or lack of development testing with an identical configuration.	Viking inventory of potentially critical items stands as . Transmitters, S Band, UHF . TWTA, S Band . Diplexers ScienceEquipment . Mass spectrograph Entry . Retarding Potential Analyser . Cameras - Imagery . Biology . Gas Chromatograph Mass Spectrograph
a. High voltage arcing at reduced pressure.	Lunar Orbiter; Photo System. 20 incidents during ground test (2 incidents in flight)	The voltage breakdowns in Lunar Orbiter photo system was attributed to process control in high voltage potting. The multiplicity of incidents plus the carry over into flight cast suspicion upon the adequacy of the design approach.	
b. Intermediate voltage/current breakdown effects.	Biosatellite; 5 incidents of Corona on diplexers, 1 incident on Antenna Nimbus; 1 incident on antenna, early unit	The corona sensitivity for Biosatellite diplexers appeared in early acceptance testing. Corrective action consisted of encapsulation with pressurization, the diplexers never experienced low pressure operation.	
▼ Induced pressure effects trigger breakdown.	▼ Mariner; Gas leaks from pyro valve.	▼ The Mariner incident involved gas leaked from the pyro valves which unlatched the scan platform. The gases leaked into the instrumentation bay of the scan platform, pressures rose the critical levels for voltage breakdown in the science equipment. The seals in the valve chambers were redesigned to contain the evolved gasses.	Further comment. See Table 5-5.4
4. Dynamic Sensitivity Effects			
a. Inadequate mounting attachment or damping of a sub unit allows shifting, break up or fall off during test.	L. O. 11 Mariner 6 Bios. > 20 Nimbus Not Reported	These incidents all required drawing, process and assembly changes for corrective action. They represented degrees of incompleteness in detail designs and areas where development testing was bypassed (i.e. significant changes between prototype and qualification units). The basic structures, circuits, or approaches usually went unmodified. For the most part these difficulties were isolated and corrected during qualification, however for Lunar Orbiter, shifting of the photo-multiplier tube within the Star Tracker persisted well into the flight acceptance testing.	Viking dynamic requirements are relatively modest, however, the uncertainty of the entry and landing condition will not allow relaxation of design.
b. Unexpected sensitive resonance condition fails sub unit in overstress or fatigue.	L. O. 2 Mariner 8 Bios. > 10 Nimbus 1	These effects are the type of failures for which dynamic tests are performed. They arise from the compromises associated with detail design and generally represent areas of difficult or inexact analyses. These effects were generally resolved in tests with the qualification unit.	
▼ Crystal sensitivity to vibration unexpected for configuration.	Mariner Power Conversion Unit	▼ The design for the frequency control crystal in the power conversion assembly had accounted for the vibration environment. However, during one axis of exicitation (parallel to boost) the output wave broke up continually during random and at 600 and 800 Hz during sine sweeps. The crystal was replaced with a ruggedized unit and the mounting modified to allow more damping.	Crystal type control units will be employed, which must operate throughout entry and landing environments.

TABLE 5-3 CONTINUED PERSISTENT PROBLEM AREAS

PROBLEM DESCRIPTION	OCCURRENCE	EFFECT ON PROBLEMS, ACTION,	VIKING CONCERN
5. Problem areas and defects attributed to manufacturing operation.		The incidents cited in the 7 problem areas described below are cases which escaped the Quality Controls interlaced throughout manufacturing and appeared during either acceptance or subsequent testing. The necessary repairs, reworks, or changes constituted extra expenses, delays, and the potential for additional damage to spacecraft hardware.	Further comment and comparison of Manufacturing considerations appear in Table 5-5.9.
a. Errors in drawings and procedures propagated into equipment.	L.O. 26 Mariner 16 Bios. 51	This type of incident voids the Quality Assurance provisions. Hardware can pass all inspections with a built in defect. Corrective actions are summarized as follows: Change to drawing or procedure only (Inspectors or assemblymen recognized the error, hardware produced correctly). 12 Hardware used with modest rework. 40 Hardware required extensive repair 24 Hardware lost 18	. Vendor Control.
b. Soldered joints . Cold solder joints . Cracked joints . Excess solder interference . Splatters and splashes	L.O. 15 Mariner 5 Bios. 31	Inspections kept the majority of poor solder joints out of finished assemblies. Those that appeared were either a missed inspection or had intermittent contact. In many cases test indications of intermittent contacts were called bad joints and the item returned for rework of the suspect area. Rework incidents 24 Multiple reworks or change outs to modules 17 Modules lost 10	. Microcircuit Assembly.
▼ Solder ball effect in Zener diodes and transistors	L.O. 2 incidents on ground, 1 suspect in flight LO-I-6	▼ Zener diodes used in the transponder and transistors used in shunt regulator could form internal solder balls in the course of lead weldings. The effect degraded the part; failure could result at later time. Detection required X-ray examination, which this could be marginally successful. The flight failure involved a transistor in the shunt regulator which had been X-rayed, accepted, and operated for 835 hours in ground test and 320 hours in flight.	. Part Selection
c. Leaks and leakage effects other than valves . Faying surfaces . damaged or poor seals . Damaged or poor pressure lines	L.O. 21 Mariner 16 Bios. 28	Leakage in all forms constituted a continuous ground test annoyance. However, once the system integrity became established leakage problems tended not to reappear. Incidents of leakage during flights were singular. Rework of mating surfaces with reassembly 35 Replacement of sealing unit and rework 23 Loss of significant assembly 8	Flight transit time of 340 days exceeds any previous leak integrity requirement.

TABLE 5-3 CONTINUED PERSISTENT PROBLEM AREAS

PROBLEM DESCRIPTION	OCCURRENCE	EFFECT ON PROGRAMS, ACTION,	VIKING CONCERN
5. Manufacturing Effects Cont'd d. Contamination Effects . Particles inside assembly casings . Particles imbedded in surfaces of control elements . Particles in fluids	L. O. 30 Mariner 8 Bios. 18	Contamination was often the base cause for other failures, leaky valves had seats damaged by imbedded particles. Shorts appeared in integrated circuits and relays failed to close due to particles left within component assemblies. In the latter case the particles have ranged from gold flakes within a Mariner micro circuit assembly to odd bits of wire and loose washers which fell out of a Lunar Orbiter photo sub system assembly. Rework incidents as clean ups 31 Repair or replaced damaged parts 17 Assemblys lost 8	Cleanliness requirements for Viking exceed all other programs. Contamination effects may be reduced in hardware assembly and operations.
▼ Mariner IRS cryostat plugs due to contaminants in cooling gas supply.	Ground test 10 incidents; Flight: loss of one channel on Mariner VI	▼ The ground testing of the cryostat for the Mariner '69 infrared spectrometer showed a great sensitivity to small quantities of particle contamination in the gas supply. Process controls were improved in the course of spacecraft preparation however the correction was not completely adequate; The IR channel covering the wave length 4 to 14 microns did not work for Mariner VI.	Potential problem source to GGMS, and biology experiments.
e. Electrical Shorts . Internal to components . Spacecraft Harness . Shorts to Ground . Shorts between voltage sources	L. O. 16 Mariner 8 Bios. 19	These effects represented insulation loss or spurious contacts made after assembly checkout. Bent connector pins, tightly placed components, lead flexings, marginal assemblies all contribute to the generation of shorts. Modest rework to correct (terminals, connectors) 23 Repair or replace items in modules (shorted parts) 25 Items destroyed or lost (burn outs) 5	Viking potential exceeds other programs due to heat sterilization environment . Plastic flow of insulation . conductive film depositions . Dense loading of lander body
f. Miswiring . Missing wires . Extra wires . Wires in the wrong place	L. O. 9 Mariner 15 Bios. 12	Miswiring represented a breakdown in Quality Control procedures in the manufacturing areas in that inspections of wire routings and connections were usually 100%. Considering the potential for misapplying power, the effects have been modest. Rework, wire changes only 19 Repair damaged parts 14 Replace lost modules, harness segments 3	
g. Alignments . Internal alignment for operation . Boresight type alignment of science sensors . Electrical alignments of functioning systems	L. O. 9 Mariner 3 Bios. 9	These problems are diffenentiated from adjustment in that they require 2 or more assemblies to work in conjunction. The incidents cover such items as the antenna position sensors on Lunar Orbiter (Electrical Mechanical), Separation-release assemblies on Biosatellite (Mechanical) and Science Sensors for Mariner (Electro-Optical) Straight forward rework iterative shim and adjust 11 Interdependent, relies on the skill of the technician 7 Electrical items, alignments for frequencies, phases, power matching 4	Viking alignment requirements include: Cameras Radars Accelerometers Telemetry S Band Thrusters Structure Inertial Equipment

TABLE 5-3 CONTINUED PERSISTENT PROBLEM AREAS

PROBLEM DESCRIPTION	OCCURRENCE	EFFECT ON PROGRAMS, ACTION,	VIKING CONCERN
6. Reversed Polarity Effects . Circuits with voltage or current directional sensitivity receive input in opposite direction. ▼ Sensor outputs reversed into other circuits.	Total all programs exceeds 10 incidents in which a circuit component was destroyed after delivery to test. ▽ • Mariner VII Solar Panel Temp. Sensor Flight Incident • Accelerometers in Ground Test, more than 10	These effects appeared as both manufacturing errors and operator errors in the course of test. In either case they represented a breakdown in the Quality Assurance procedures. The incidents cover such events as capacitors destroyed, diodes reversed, sensors elements lost. Biosatellite destroyed a significant number (more than 20) of tantalum wet slug capacitors by wrong installation. For the most part these incidents occurred during inspection operations before final unit assembly. ▼ The Mariner temperature sensor on the solar panels shows the degree to which small effects can escape the inspection processes.	The Viking Lander employs polarity sensitive part and sensors throughout all subsystems. Further Comment and Biosatellite incident, Table 5-5.8
7. Test Related Effects a. Specifications and procedural problems . Specifications incomplete or incorrect (limits, tolerances, capabilities) . Procedures incomplete, incorrect or unclear. b. AGE configuration not compatible with flight equipment . Improper current, voltages . Improper sequencing or timing (Does not include cabling, see item "e")	LO 91 Mariner 25 Bios. 118 L.O. 24 Mariner 16 Bios. 61	The effects tabulated below all required significant corrective actions such as document changes; flight equipment retest/repair/replacement or support equipment modification. Incidents of "accept as is" or "continue to test" have not been included. Since specification and test procedures are the basic sources for Quality Assurance procedures, these incidents represent areas of weakness and potentials for damage. . Paper changes, documents revised to correspond to test results 155 . Retest required after document change, hardware undamaged 55 . Flight equipment repaired, retested, document changed 20 . Flight equipment lost, replaced 4 The incompatible AGE configurations listed included modifications as part of the corrections; and in addition, a significant action with the flight hardware was involved (i.e. remove, reinspect/repair/replace, then retest) . Flight hardware removed for inspection and extra diagnostic retest 61 . Flight equipment required repair 30 . Flight items replaced 10	Further comment and comparison of test considerations appear in Table 5-5.10. Lunar Orbiter Experience includes an incident of damage to AGE as blown transistors during a launch countdown test. An AGE modification was not incorporated, power switching allowed over current. (LO-III-A6)

TABLE 5-3 CONTINUED PERSISTENT PROBLEM AREAS

PROBLEM DESCRIPTION	OCCURRENCE	EFFECT ON PROGRAMS, ACTION,	VIKING CONCERN
7. Test Effects Cont'd. c. Operator errors . Incomplete or incorrect set up . Procedure not followed resulting in incorrect or missed controls	L. O. 22 Mariner 6 Bios. 35	For these incidents both AGE and procedures were correct for the test. The error committed was of a magnitude to potentially damage flight hardware. . Flight items removed, reinspected, retested 35 . Flight items required repair 26 . Flight items replaced 6	
d. Facility problems; Facility or Fixturing cause potentially damaging test condition . Capability limited . Control anomaly . Instrumentation malfunction . Unexpected action, i. e. shaker dump	Mariner 8 Bios. 16 L. O. Not Reported	In addition to facility changes, equipment modifications or procedure changes these incidents required specific attention to flight hardware . Flight items removed, reinspected, retested 13 . Flight items requires repair 7 . Flight items replaced 4	
e. Cabling effects; Mistest due to defects in cables within the AGE complex	Mariner 7 Bios. 13 L. O. Not Reported	These cases necessitated significant action to flight hardware in addition to rework of the cable involved. . Flight items removed, reinspected, retested 13 . Flight items repaired, retested 4 . Flight items replaced 2	
▼ Repairs Changes to AGE system complex cabling	▼ Bios. 34	▼ Biosatellite data from Spacecraft Acceptance and Qualification testing identified the extent of repairs to interconnecting cables. The incidents covered shorts in connectors, broken leads, damaged shields, grounds, miswiring and fatigue failures. Of this sample 6 incidents defined the replacement of a cable.	

TABLE 5-4 UNIQUE FAILURE AND ANOMALY EXPERIENCES RELATABLE TO VIKING LANDER

PROGRAM/INCIDENT	DESCRIPTION - CAUSE, EFFECT, CURE	VIKING CONCERN	COMMENTS/RECOMMENDATIONS
A. POWER LINE NOISE AND ABERRATIONS			
Biosatellite Regenerative oscillation of power inverter circuitry.	The configuration of the power supply monitoring and control network produced a regenerative oscillation due to positive feed-back and required relocation of the voltage sensing point.	Viking will also have a complicated power supply and distribution problem due to its variety of equipment. Changes in voltage needs can also be expected due to unanticipated hardware problems.	To the greatest extent possible voltage regulating circuits should be located adjacent to their loads and interactions within the power distribution should be eliminated.
Biosatellite Noise Filters pumped noise energy back into power lines.	Noise sensitive components such as the Attitude Control Programmer were protected from noise by filters on the power lines. However, these filters discharged the noise energy by pumping it back into the power lines where it caused interference elsewhere.	Cable runs susceptible to pickup will be long, uncertain power demands by the science may produce sharp spikes, excessive communication turn-around will preclude contingent reaction.	Noise filtering should be treated as a system problem and EMI tests should include monitoring throughout the system.
Mariner Lack of transient immunity.	Part of the CCS logic generates a false output in response to power switching as a result of which a delay circuit is bypassed and an error can be introduced into the memory. This was prevented by isolating the logic.	The Viking CCS will have similar circuits and spurious signals are very likely to occur.	Particular provisions should be taken in the CCS to suitably isolate digital timing circuits.
Biosatellite Lack of circuit voltage design margin and voltage reduction from IR drop in cabling.	Some components such as the Storage Programmer lacked voltage design margin such that they malfunctioned due to IR drop through the cabling.	Long cable runs are likely on Viking due to its large size. A similar problem is quite possible.	Assure realistic voltage specifications at system, subsystem, and component levels which reflect the successive reductions due to normal load and distribution losses. Select wire sizes to minimize harness IR drops.

TABLE 5-4 UNIQUE FAILURE AND ANOMALY EXPERIENCES RELATABLE TO VIKING LANDER

PROGRAM/INCIDENT	DESCRIPTION - CAUSE, EFFECT, CURE	VIKING CONCERN	COMMENTS/RECOMMENDATIONS
B. POWER INTERRUPTIONS			
Nimbus Electrical Short	So much grease was applied to an O-ring used for assembly in the IR Scanner that it caused an electrical short.	O-rings are widely used in intricate mechanisms such as the science experiments.	When used on O-rings, grease should be used sparingly.
Biosatellite Loss of power due to open circuit of a power monitoring resistor.	Loss of all power was caused by the open circuit of an hermetically sealed power resistor used in a power monitoring circuit. Although seemingly ruggedly constructed, the resistive element (about 1/16 inch diameter) fractured at the terminals due to lack of flexibility to accomodate the prescribed environmental range.	No single failure mode capable of opening the power supply line can be tolerated on Viking.	Redundancy should be provided for any circuit element such as a diode or resistor in such critical locations as the power source.
Pyrotechnic Damage	A pyrotechnically separated IFD struck a wiring cable with such force that the insulation was damaged causing a short circuit which interrupted power, stopping a programmer and preventing a recovery sequence.	Viking has many pyrotechnic devices and events.	Consideration should be given to accomodating and dissipating the energy produced by pyrotechnics.

TABLE 5-4 UNIQUE FAILURE AND ANOMALY EXPERIENCES RELATABLE TO VIKING LANDER

PROGRAM/INCIDENT	DESCRIPTION - CAUSE, EFFECT, CURE	VIKING CONCERN	COMMENTS/RECOMMENDATIONS
C. COMMUNICATIONS			
Biosatellite Single Command transmitted - multiple commands generated by the decoder. Decoding circuit interaction.	The AVCO Command Decoder had been successfully used on a number of NASA satellites but was found excessively sensitive to line noise. This condition was aggravated on Biosatellite because the receiver and decoder were separately packaged rather than contained in one enclosure which minimized their interfaces. The fix required resectioning of circuitry and re-routing of wiring to provide circuit isolation plus the addition of a current limiter to provide back up disabling of more than one output per address.	This is one more example of the dangers attendant in attempting to adapt equipment, particular micro wave, designed for different system conditions and applications	The Viking CSS in particular will require isolation of various functions and outputs to prevent interactions.
Mariner RF False Lock Anomaly Disturbs ranging and command activities.	A feedback path existed in the RF subsystem receiver, resulting in a potential false lock effect, a change in effective receiver bandwidth, changes in command modulation level, and an anomalous output. The problem was caused by spurious signals on the ±15 volt supply line to the Voltage Controlled Oscillator (power supply triggered). The spurious signal modulated the VCO producing side bands which after amplification are interpreted as an indication of lock-on, which then generated receiver AGC voltages, etc. The supply circuit to the VCO was filtered for low frequencies (such as 2.4 kHz) and was apparently adequately by-passed for RF. The operational difficulty was inhibited by transmitting ranging of commands.	Viking will also employ ranging equipment which is vital due to the nature of the mission. Ringing and Oscillation effect which persisted into flight.	Choose operating conditions such that transient parasitics cannot occur.
Biosatellite Irregular outputs from Storage Programmer - lack of circuit biasing.	The Storage Programmer consisted of majority logic shift registers which produced spurious outputs. It was found that the circuits malfunctioned in response to voltage supply fluctuations such as those caused by power demands elsewhere. The problem was found to be lack of provision in substrates for adequate biasing of the circuits and required redesign of the substrates.	Viking will employ many timing circuits some of which will lack redundancy. They form the heart of the system and so their stability is imperative.	Worst case tests of all timing circuits should be conducted to assure their stability.
Interruption of Transmitter Output due to frequency shift caused by case distortion.	The Transmitter output frequency shifted as a result of transient dimensional changes caused by rapidly decreasing boost phase ambient pressure. This phenomenon was undetected in test because only initial and final conditions were monitored.	Viking Lander equipment will be subject to dynamic effects during descent and science operation.	Significant parameters of all equipments under test should be continuously monitored to provide assurance that no transient effects are overlooked.

TABLE 5-4 UNIQUE FAILURE AND ANOMALY EXPERIENCES RELATABLE TO VIKING LANDER

PROGRAM/INCIDENT	DESCRIPTION - CAUSE, EFFECT, CURE	VIKING CONCERN	COMMENTS/RECOMMENDATION
D. ELECTROMECHANICAL DEVICES Lunar Orbiter Loss of Accelerometer accuracy due to extended storage.	The accuracy of the Linear Accelerometer located in the IRU shifted out of tolerance. It had been stored so long in the same position that the critical axis bias adjustment had changed.	The twofold problem can be expected on Viking of extended storage of hardware and many mechanical elements subject to such failure mechanisms as structural deformation, cold welding, and plastics (as nylon) flow.	Mechanical devices should be exercised and storage positions should be changed periodically.
Lunar Orbiter Loss of Accelerometer accuracy due to beating of two frequencies in the IRU.	The Linear Accelerometer output was out of tolerance because of pulsing problems in the IRU. The power for driving the gyro spin motor was 400 Hz. This beat with a pulsed power at 9600 Hz, used in the IRU for heaters, to disturb the accelerometer supply line. The gyro power was detuned to eliminate this problem.	There may be various frequencies in Viking systems which could beat together in this manner. The attitude control and science systems are likely.	AC use in Viking should be limited to one frequency. If more are needed they should be isolated.
Lunar Orbiter Camera Thermal Door Resonance	The Camera Thermal Door failed to open because of the excessive load resulting from its resonant vibration which coincided with the applied motor pulses. The condition was corrected by addition of a mass damper which changed the resonant frequency of the door.	The science experiments include many mechanical devices which may contain resonances to produce this condition.	Perform worst case dynamic studies and tests for all mechanical devices. Review and check electrical frequencies for correspondences to or harmonics of mechanical resonant frequencies.

TABLE 5-4 UNIQUE FAILURE AND ANOMALY EXPERIENCES RELATABLE TO VIKING LANDER

PROGRAM/INCIDENT	DESCRIPTION - CAUSE, EFFECT, CURE	VIKING CONCERN	COMMENTS/RECOMMENDATION
E. MATERIALS AND PACKAGING			
Biosatellite Fuorolube Contamination	Fluorolube, a halogen based lubricant used to prevent galling of hydraulic and pneumatic fittings, reacted with aluminum under conditions of high temperature and pressure to explode in extreme cases and to contaminate the fluid with a white powder under more moderate conditions.	Plumbing of the hydrazine storage vessels are exposed to highly corrosive material for a very long period.	All exposed materials require careful selection and testing
Lunar Orbiter High gain antenna jambs during Thermal Vacuum test, Mercury Corrosion.	The commutator and brushes for the antenna position indicator froze to the antenna shaft. Analysis showed that mercury vapor from a chamber manometer had chemically attacked the surface.	Extensive vacuum testing particularly in vendor facilities.	Review Instrumentation planned for supporting equipment.
Biosatellite 4 Identical Connectors Side by Side, cross connected, 2 occasions.	The interface connectors in the recovery capsule were 4 identical 61 pin Bendix units. The limited use and spare part considerations prevented independent keying. They were individually identified, yet on 2 occasions AGE interconnect cases were mixed - once with some damage.	Side by Side identical connectors are an invitation to trouble.	The same potential situation appears for Viking. Limited numbers in use, Long interconnect cables.
Biosatellite Identical Fluid Couplings Side by Side	In a similar manner, coolant fluid was loaded into an oxygen line, disassembly and cleaning required. This later incident nearly caused a flight delay.	Same comment.	Same comment.
Biosatellite Contaminated circuit board encapsulation.	There were a number of instances of electrical leakage of circuit boards due to lack of cleaning for encapsulation and poor preparation of potting modules.	Contamination can be a problem due to the sheer quantity of potted modules.	New techniques of non destructive testing should be explored to provide assurance that encapsulated modules are not contaminated.

TABLE 5-5 - EXTENDED DESCRIPTIONS

The extended descriptions are presented as a series of 10 sub-tables titled as follows:

5.5.1	EMI Effects
5.5.2	Lunar Orbiter System Isolation
5.5.3	Ringing and Oscillation
5.5.4	Corona and Arcing
5.5.5	Tape Recorders
5.5.6	Rate Gyros
5.5.7	Radioactive Thermoelectric Generators (RTG)
5.5.8	Polarity Effects (Biosatellite Command Receivers)
5.5.9	Manufacturing Effects
5.5.10	Test Effects

TABLE 5-5.1 EXTENDED DESCRIPTIONS, EMI EFFECTS

DISCUSSION OF EFFECTS	VIKING PROJECT IDENTIFIED NEED
SOURCES, CAUSES: The transients result when power transfers through hard or insufficiently suppressed switches (relays, steppers, toggles, reeds, step diodes, etc.). The principal sources identified within spacecraft have been gyro power and phasing; telemetry turn on/off; heaters; deadfacing (separation); and attitude control (solenoids). Hard switching within AGE presents externally imposed transients. Transient rates for currents range from 10^4 amps/sec (relays) up thru 3×10^5 amps/sec (solid state, microsecond responses). The induced pulses trigger state changes in flip-flops and alter digital or discrete command/control signals. Noise represents responses to the operating environment complicated by conditions such as leads coupling through incomplete shielding. Noise effects increase when power regulation is performed as a central function. Nimbus records a minimum of effects, the spacecraft supplies only 28 volt dc regulated power with all secondary requirements performed within the science instruments or related subsystems. Lunar Orbiter, Biosatellite and Mariner show increasing EM noise effects with the more complex power systems. The 2.4 KHz square wave alternating current in Mariner was a significant noise source particularly for the higher harmonics. CONTRIBUTING EFFECTS: The transient pulses from switchings rarely matched the intended control signals, but represent cases of responses to off-design conditions; Mariner experience shows 2 graphic examples: a. A circuit designed to operate with input pulses of 6 volts for 75 to 125 milliseconds duration responded to pulses of 2 volts for 4 to 5 milliseconds duration. The system response was interpreted as spurios commands. b. A similar circuit sensitivity responded to double pulses such as contact bounce within a relay. The first pulse initiated the action; the second, following immediately, became the inhibitor/reset signal. System response was interpreted as lost commands. These conditions have been aggravated by cases where circuits were designed for more than one application (Mariner) or existing circuit designs have been directly incorporated into the Spacecraft system (Biosatellite). Low signal, high impedance circuitry are prey to EM noise problems; this type of circuit is a necessity for sensors and sensitive science instrumentation. Noise obscures or distorts the response from the desired measurement. Noisy telemetry sensors are recorded on all programs. The extreme cases appear in lost science (Mariner) and noise coupling of commands (Lunar Orbiter, See 5-5.2 below). Noise conditions are controllable. Witness Biosatellite 30 Day where telemetry processed the microvolt level signals associated with electro-encephalograms (brain waves). Finally the experiences from both Biosatellite and Mariner indicate that AGE and cabling are rarely transient surpressed or noise shielded to the same degree as spacecraft equipment with the resulting compromises in operating effectively. TEST EFFECTIVITY: EMI testing has been successful in identifying sources and needs for corrective action; yet System level testing can be painfully slow. Biosatellite 30 Day reports 60 days to complete a 6db margin demonstration which consisted of measurements of EM ambients followed by amplified noise and transient injections to establish the capability. With respect to the 6db margin again experience with Biosatellite suggests that for complex systems 10 to 12 db are required to provide uncompromised performance.	Viking will need specific criteria for: 1. Voltage and current transients generated within component and transmitted to lines, i.e., levels, rise times, shapes, maximum rates of change. 2. Input rejections to components (levels, times, shape) 3. Harness routing to avoid coupling. (particular attention to separation of power and sensor leads) 4. Harness and cables, shielding, grounding and effectiveness at frequency. 5. Application or limits applied to AGE

TABLE 5-5.2 EXTENDED DESCRIPTIONS, LUNAR ORBITER SYSTEM ISOLATION

DISCUSSION	VIKING PROJECT IDENTIFIED NEED
The following description has been drawn from the analysis of Lunar Orbiter flight anomaly identified as LO-II-A5, False Commands. Flight History Lunar Orbiter II was operating in the extended mission phase (post photography) at the time of these incidents. Between the tracking periods on February 17 and February 19, 1967, the camera thermal door (CTD) opened and the photo subsystem electronics experienced several apparent logic changes which were noted in telemetry. A flight programmer memory scramble was noted on February 19, presumably as a result of signals intended for Lunar Orbiter III. The memory was restored on February 20. Further apparent logic changes in the photo system electronics were noted on March 8, however the photo system operated normally, thus the changes in telemetry were spurious rather than actual. The flight programmer memory scrambles occurred when Lunar Orbiter II was not in full view during command activity with Lunar Orbiter III; consequently the power level with respect to Lunar Orbiter II was below the minimum required during command activity. The corrective action consisted of raising the DSIF transmitter power to 10 kw. In effect the signal level at LO-II was increased to the point where the LO-III address was recognized; the isolation logic then prevented any further spurious commands. Previous Knowledge and Analysis Prior to the first flight, during DSIF compatibility testing with Spacecraft "C", it was discovered that a condition could exist where the command decoder could sporadically respond to random noise output from the transponder. If this condition was allowed to continue, false commands were issued to the programmer. The number of false commands issued was a function of the noise level at the command decoder input and the length of time the condition existed. To alleviate this condition a squelch circuit was added to the transponder to eliminate the spurious command condition when the transponder was out of lock and a modification was made to the command decoder to decrease its sensitivity to noise when the transponder was in lock. These false commands (signal level from -128 dbm to -150 dbm) remained for a finite time (2 to 4 seconds) after the up link carrier was turned off and before the transponder squelch circuit was activated. This allowed the command decoder to be vulnerable but with the desensitized command decoder it was felt that this period was not long enough for generation of false commands. This proved to be true since no false commands occurred during normal tracking of any of the five spacecraft. The problem occurred when two or more spacecraft were operated on the same ratio frequency. When tracking the prime spacecraft, it was possible to lock up the other spacecraft inadvertently. If this occurred at a signal level between -128 dbm and -159 dbm and was allowed to continue for long periods of time, false commands could be generated and would result in a scrambled programmer. This is what happened in this particular case.	The System Isolation for Viking cannot admit to the problems described for Lunar Orbiter. Viking isolation schemes must show proofs against: . Signal threshold effects . Internal noise effect . Inter-vehicle effects (Relay transmission couplings)

TABLE 5-5.3 EXTENDED DESCRIPTIONS, RINGING AND OSCILLATION

COMMENTS, OBSERVATIONS.	VIKING PROJECT IDENTIFIED NEED
Ringing and oscillation problems as a persistent area tend to appear as spacecraft or system level complications. As such design modification are more difficult to incorporate, corrective actions are "fixes" with acceptance of risk. The effects persist into flight and carry the need for work arounds. The problems which have been identified as relatable to Viking (See Table 5-4) can be considered as somewhat generic; although each incident appeared as essentially a sneak effect. Designers were well aware of feedback stabilities, beat frequencies and regulator stabilities with their analytical treatments (Nyquist, Bode, et al). The struggle with ringing and oscillations during Biosatellite spacecraft testing led one senior system engineer to comment, "If it can ring, it will". The areas of identifiable concern to Viking in which ringing sensitivity may be anticipated are summarized as: a. Spurious RF: Paticular attention must be paid to harmonics, RF feedback thru loops with incomplete shielding and RF leaks at couplings (Biosatellite, Mariner) b. Beat frequencies: A vulnerable area appears to be the 400 cycle A. C. associated with the rate gyros (L. O., Bios.) c. Regulators in Power Supplies: Wide dynamic range requirements for input to the regulation seem to aggravate the sensitivity (Mariner). d. Simulation loops particularly with AGE cabling (Biosatellite). e. Mechanical feedback couplings into an electrical loop (Biosatellite, Lunar Orbiter). This last condition was observed to only a minor extent in the program studied. However, for the Viking Lander pulsing attitude control jets and high force throttling during terminal descent is expected to receive stability margin evaluation and testing.	In addition to the stability studies current planned, Viking Lander could employ a means to: . Review interfaces and interface signals specifically for potential ringing and oscillation inducing conditions. . Evaluate the effects of unexpected signals which will appear generated within the system as development proceeds (i.e. resonance levels different from prediction, voltage ripples or spikes) . Suspect and look for ringing effects behind seemingly random part failures or similar effects.

TABLE 5-5.4 EXTENDED DESCRIPTIONS, CORONA AND ARCING

COMMENT	
Corona and arcing effects seem to be detectable and correctable from testing of components or subassemblies. Corrective action shown effective by test eliminates further incidents (Biosatellite, Nimbus). Where effects persist to later test or flight, the design approach must be questioned. The incidents from Lunar Orbiter in the Photo System were attributed to a process control problem. The comment can then be raised, in such a critical area, a process of such difficulty to control suggests the need for an alternate solution. VIKING CONCERN In Lunar Orbiter, Biosatellite and early Nimbus the means employed to eliminate corona and arcing sensitivity cannot be considered acceptable to Viking. Encapsulation is not compatible with weight, sterilization heat transfer or the 2 year life requirement; Reduced power is feasible only for near earth; marginal processes are not compatible with heat sterilizations. Viking will be forced to design-out any detected sensitivity to low pressure operating conditions. ENVIRONMENTAL CONCERN The incidence of corona and arcing type problems which do not allow precise analyses together with the environmental anomaly incidents described as items 7, 8, 9 of Table 5-1. present an overall case for environmental margin testing for the Viking Lander. Flights have been compromised by effects overlooked (Bios. IR Sensor); designs which missed or could not account for conditions (Nimbus, Lunar Orbiter); unanticipated environments (Nimbus); and inability to completely cope with a defined condition (Lunar Orbiter arcing effects). Considering the uncertainties of the Mars-atmosphere, known, extensive, operatingmargins appear to offer the best insurance for successful data return.	Environmental evaluation and verification for voltage breakdown effects with extended margins. Particular emphasis on: . Early Testing . Correct (Flight) Geometry . Range of Atmospheric capability Items considered potentially most critical are science . Retarding potential analyzer . Entry Mass Spectrograph . Cameras . Biology . Gas Chromatograph., Mass Spectographs

TABLE 5-5.5 EXTENDED DESCRIPTIONS, TAPE RECORDERS

DETAILED EXPERIENCE	VIKING PROJECT IDENTIFIED NEED
Nimbus 3 Lockheed Continuous Loop Recorder Development problems were a broken drive shaft in vibration test leakage of the pressurized containers, and reduced output level due to contamination of the heads with graphite tape lubricant. Flight failures were caused by tape breakage at the tape splice (probably) and by excessive torque of the tape pack, motor bearing, and/or clutch as indicated by intermittent operation and high motor current through diagnostic telemetry.	Reel to reel type recorders appear preferable to continuous loop type such as this one.
Nimbus 3 RCA Reel Type Tape Recorder Background was 5 years noncontinuous operation on OGO. Development problems were: a. Tape sticking to the head cured by head selection and contour and by controlled humidity of 30% in an atmosphere of 78% N_2, 12% O_2 and 10% He pressurized to 17 psia. The tape becomes too smooth at lower humidity and too sticky at higher humidity. b. Progressively reduced output caused by oxide buildup on the playback head. Cured by addition of a strip of abrasive chrome oxide "green tape" to the end of the magnetic tape so that the head is periodically cleaned. Enough time has not ensued to determine the effectiveness of this fix. c. There was so much variation in the quality of purchased tape that RCA made their own with adequate processing control. This included a maximum ambient temperature of 35°C. d. Erratic reproduction caused by tape pressure from the end-of-travel (EOT) switches. This was cured by using an optical EOT switch in conjunction with a transparent section of tape and backed up by a mechanical switch beyond the "green tape". e. Broken drive belts cured by rigorous inspection. The standard construction of these belts for recorders is a flat doughnut shaped ring punched from 0.0005 inch mylar and turned to fit on pulleys. f. Flutter caused by resonance of the mechanical drive including 5 drive belts, a planetary drive, two motors, and ball bearings. Reduced but not cured by mechanical de-tuning and by replacing, readjusting, and cleaning elements. g. Pressure leak at connector O-ring seal due to lack of key permitting distortion of ring when connector lock nut was tightened. Eliminated with proper parts. Flight failures were gradual deterioration of outputs from same three channels on both recorders suppositionally caused by oxide buildup on part of heads in relatively light contact with the tape due to its canting by the mechanical drive configuration. This should be cured by use of the "green tape".	Use of a hermetic seal and internal atmosphere as shown with the same humidity is recommended. The most successful solution to the head contamination problem is the use of an abrasive tape section with command provision to select it. On Nimbus this was accomplished by locating the cleaning tape between the primary and backup end-of-tape switches and unrolled by transmitting a second playback command.
Biosatellite Recoverable Tape Recorder Development Problems were: a. Tape stored over 100 F subsequently lost flexibility and bent in incremental steps. b. Tape came off the reels during vibration. c. The Recorder was grounded through a flexible strap which was inductive at high frequencies upsetting the frequency response and operation due to external interference. Cured by grounding through solid bars.	Careful attention must be given to grounding details.

TABLE 5-5.6 EXTENDED DESCRIPTIONS, RATE GYROS

DETAILED EXPERIENCE	VIKING PROJECT IDENTIFIED NEED
Nimbus 3 Sperry Gas Bearing Gyro used in Rate Measuring Package (RMP) Experiment The gas bearing clearance of 3 to 7 microinches becomes contaminated with outgassing polymers which tend to bind it sufficiently to prevent the motor from starting after being stopped for a period. The gimbal suspending ligaments tend to relax with time producing increased, and variable torque which determines the drift rate. The RMP gyro drift error was excessively high--20 degrees per hour--at the beginning of the mission over a year ago, and has slowly been decreasing since. As a result the experiment was not repeated on Nimbus D launched in April 1970 and the ball bearing gyro will continue to be used for primary rate control on future Nimbus satellites. Some motors failed due to warped end thrust pads which limited the rotor freedom.	The attitude control requirements for the Viking Lander cannot accept drift rates of 20° per hour. Development of gas bearing rate gyros has continued, better designs are presently available.
Biosatellite Nortronics Ball Bearing Gyro Although a standard design for some time, this gyro sustained test failures due to broken flex leads during vibration, nul shift due to inadequate temperature compensation, and contamination of the flotation fluid. The latter was cured by more stringent manufacturing cleanliness requirements.	Very stringent procedures are advisable in preventing contamination of flotation fluid of Viking gyros. Unfortunately contaminants in the fluid are randomly suspended and adversely affect gyro operation if they drift into the bearings where clerances are very small.

TABLE 5-5.7 EXTENDED DESCRIPTIONS, RADIOACTIVE THERMOELECTRIC GENERATORS (RTG)

DETAILED EXPERIENCE	VIKING PROJECT IDENTIFIED NEED
Nimbus 3 SNAP 19 RTG — Apollo SNAP 27 RTG The output of the SNAP-19 flown on Nimbus 3 has degraded 21.5% instead of the expected 10% in about 10,000 mission hours. There have been three main problems in its development which contribut to the degradation. The first was caused by contamination of the thermopile by impurities in the insulation. The second was deteriorization of the thermoelectric hot bond since both hot and cold ends are bonded without provision for relative linear motion. The third problem was increased thermoelectric sublimation due to relatively low pressurization level; in particular the Nimbus unit has leaked so that there is no internal pressure.	The following successful practices have been followed on SNAP-27 to overcome the noted problems of SNAP-19: 1. Specify stringent processing control to remove contamination for the insulation. 2. Use a pressure connection with axial variability at the thermoelectric hot ends. 3. Use argon pressurization of 25-27 psia. 4. Use welded joints and fillings to minimize leakage.

TABLE 5-5.8 EXTENDED DESCRIPTIONS, POLARITY EFFECTS (BIOSATELLITE COMMAND RECEIVERS)

DISCUSSION	VIKING PROJECT IDENTIFIED NEED
Comment. Polarity reversals either from manufacturing or operator error have generally proven costly as blown capacitors, burned sensors or no data. The incident with the Biosatellite II command receiver is a startling example of how a reversed polarity crept into a system, circumvented redundancy, and skirted the edge of catastrophe. Biosatellite II Command Receiver Incident. The command receivers employed in all Biosatellite spacecraft were vendor supplied, "off the shelf" items in more or less series production. The original design was not for Biosatellite, however they had been adapted and operated successfully during the flight of Biosatellite I. The vehicle installation employed 2 redundant receivers and command decoders. For complete redundancy, a cross-over network was incorporated between the output of the receivers and the inputs to the decoders such that either receiver could feed either decoder; 4 effective paths existed for receiving commands. During the time between the flights of Biosatellite I and II the vendor made a block change in the receiver; the units were assured as interchangeable. In the changes was a reversal of the leads to the output transformer, the signal was the same, except the phase was shifted 180°. In pairs or singly the receivers acted identically to the earlier units. In the course of Biosatellite Acceptance testing, vehicle operation had been established with telemetry and command monitored by both hardwire and RF radiation. Within the factory RF links required special care due to limits on radiated power levels and the nature of the building itself. At a point very near the end of factory acceptance, changeout of a command receiver was necessary. During the balance of factory testing the RF link proved difficult, however nothing uncommon. In field preparation the RF links responded, with difficulty. However, RF fluctuations and path problems at ETR had been experienced on many occasions (Ref. Mariner, Lunar Orbiter total 3 incidents). During the countdown, command access became marginal; and the AGE was suspected. The backup command control unit was then employed and access restored. In flight command access was effectively nil except for the tracking station at Carnarvon, Australia. This station then provided control until the successful recover of Biosatellite II. Post Test Evaluation Revealed the Problem: . The reversed output transformer on the later unit, used in pairs with an earlier model resulted in virtual signal cancellation. (Had one unit failed completely, the problem would have disappeared). • Access thru the system was obtained by either very high signal levels (AGE) or by operating the command transmitter slightly off the specified frequency. (Australia) . In preparation for the flight, all the other tracking stations had fine tuned to the assigned frequency COMMENT RELATIVE TO VIKING LANDER. The actions leading to this failure can be summarized: . A change was made which was not completely evaluated for the application. Receivers worked singly and in matched pairs. Mixed pairs were overlooked; and mixed pairs circumvented the system redundancy. . The symptoms from test were overlooked. A commitment for flight had been made; annoyances were not subject to time consuming analyses. . No one was really looking for an incident of this type.	Polarity and Phasing effects can be anticipated as problem generators. Viking will need a mechanism for: 1. Seeking out areas of potential polarity difficulty. 2. Identifying the kinds of effects which would be generated. 3. Verifying the effectiveness of corrective actions. 4. Checking for completion in finished hardware.

TABLE 5-5.9 EXTENDED DESCRIPTIONS, MANUFACTURING EFFECTS

<table>
<tr><th>DISCUSSION</th><th>VIKING PROJECT IDENTIFIED NEED</th></tr>
<tr><td>SAMPLE SIZE: The tabulation of persistent effects comprises ~ 40% of the failure/anomaly incidents attributed to manufacturing causes. The tabulation was intended to show program impact therefore defects uncovered and reported before delivery and incidents involving minor items such as fastener changes, adjustments, and interpretations have been excluded. Nimbus data has been excluded in this area because of an inherent bias. Only data from manufacturing operations performed at GE-Valley Forge were included; data from the fabrication and test of science equipment worked at vendor plants were not available for review.

COMMENT:

The most significant effect falls in the area of greatest control, drawings and manufacturing procedures. While not identified in the failure/anomaly reports the impression from review is that drawing and procedure problems are related to change and change controls during the fabrication of flight hardware. Initial releases may be the most accurate, however the incidence of the Biosatellite reversed "g" switch stands as a warning that mistakes can escape the best of detections.
In the other areas, a persistent failure must be suspected for having design problems in the background. The configuring of seals such that "O" rings are destroyed in assembly; shorting in densly packaged electronics; fatigue of wires due to short harness; all are amenable to refined design practice.

SPECIFIC COMMENTS

a. Soldered Joints: Each program has experienced solder problems and for different reasons. Biosatellite which used conventional parts found that modules were less prone to bad joints if an automatic (wave) process were employed. For Mariner the fabrication of the flight command assemblies had to employ hand soldering to attain good joints. Wave soldering was remelting the pretrimmed coatings on heads and forming a gold-solder amalgam which subsequently cracked.</td>
<td>Comparison of Manufacturing considerations between Viking Lander and Mariner '69 (Based on parts count and use of microcircuits)
<table>
<tr><th></th><th>VIKING*</th><th>MARINER</th></tr>
<tr><td>Electrical Parts Count each Spacecraft</td><td>33729</td><td>21595</td></tr>
<tr><td>Microcircuits each Spacecraft</td><td>9470</td><td>2678</td></tr>
<tr><td>Number of electrically operating assys. produced.</td><td>~ 530</td><td>~ 190</td></tr>
<tr><td>Estimated Number of Electrical Leads (Joints)</td><td>1.5×10^6</td><td>3.2×10^5</td></tr>
</table>
*Parts totals and estimate of leads drawn from Tables 6-2 and 6-4
Estimate of numbers for assemblies drawn from Table 6-4 and Figures 7-2, 7-5</td></tr>
<tr><td>b. Solder ball effect Lunar Orbiter. The analysis of the flight incident of Aug. 24, 1966 (LO-I-6) is summarized: At sunset the shunt regulator current was shown to be 1.2 amperes high. Analysis of telemetry data led to the conclusion that a power transistor 2N2016 had shorted from the collector to the emitter. Eight parallel 2N2016 transistors, in series with resistors, were used in each shunt regulator. The discrepant transistors were constructed using resistance welds of a lead plated nickel lead to connect base and emitter posts. Solder balls were formed when improper weld schedules caused weld splatters. At the time of failure, the power transistor assembly had been subjected to 1155 hours of operation, including 835 hours in ground test and 320 hours in flight. On all subsequent vehicles a modified joining technique was employed using these transistors.</td><td></td></tr>
<tr><td>c. Contamination stands as an annoyance throughout all phases of a project. Failure incidents show that lack of controls during any facet of operations can have a major impact to subsequent activities. The contamination in the Biosatellite circuit boards increased costs and constrained the schedules for acceptance testing of flight equipment. The plugged cryostat on Mariner cost science data. Also for Mariner dirt shaken from the inside of the flight shroud compromised the careful clean room controls applied during spacecraft assembly and provided the source for the bright particles which confused the canopus star sensors during in-flight maneuvers.</td><td>Contaminants deposited from these shrouds are an item for consideration relative to maintaining Lander quarantine.</td></tr>
<tr><td>d. Cable Shorts: Pinched wire effect, Extrusion of Insulation; Pinched wire problems appeared to a minor degree ~15 incidents total for the study. However, in Biosatellite shorting or harness degradation occurred due to insulation extruded (cold flow) by tight or undersized harness clamps, Correcting for the effect required repair of harness segments and close control of clamp sizes. . For Viking Lander, post sterilization inspection of harness will be impossible; considering the heat softening of wire insulation material, clamp size discipline must be maintained to control plastic creep effects within harness assemblies.</td><td></td></tr>
</table>

TABLE 5-5.9 EXTENDED DESCRIPTIONS, MANUFACTURING EFFECTS (CONCLUDED)

COMMENT	VIKING PROJECT IDENTIFIED NEED
e. Alignments: Mariner experience is the most outspoken. Science instrumentation often had little or no capability for bore sight type alignments. This condition is a design oversight. For Lunar Orbiter, incident LO-IV-A6 identifies an alignment problem which need not have existed.	
From the analysis performed for this incident: . . . . . . During evaluation of the Goldstone test film, it was determined from the data read out that the line scan tube was tilted approximately 0.78 degrees. Since the tilt of the line scan tube is a component adjustment, line tilt was evident in all Mission IV video data and could not be corrected during the mission. (Acceptable tilt is ±0.17 degrees) The analysis disclosed that the ground reconstruction system manual being used by Eastern Test Range personnel erroneously called for the measured film data to be divided by the magnification factor of the ground reconstruction equipment. This error was subsequently corrected in all documents but inadvertently not incorporated in this specific copy. Consequently until the apparent line tilt was noticed in this flight readout, all units were considered to have negligible tilt. A tilt of 0.17 degrees of arc is considered to be acceptable. Since the data evaluated at ETR were divided by 7.2 (magnification), the results appeared to be acceptable. An investigation into the method of accomplishing the alignment of the line scan tube to the optical mechanical scanner was made. The alignment procedure makes the best fit of the line in the readout gate. Two parallel mirrors are aligned to provide X and Y motion (scale and line tilt). Since these cannot be treated individually, the final alignment depends on the skill of the person making the adjustment.	. Procedure Error. . Marginal Technique.

TABLE 5-5.10 EXTENDED DESCRIPTIONS, TEST EFFECTS

DISCUSSION

SAMPLE SIZE: The tabulation of persistent effects is intended to show a trend and comprises ~23 percent of the total incidents attributed test causes. To show project impact the tabulation excludes items which resulted in no change (i.e. "use as is" or correct in immediate retest). The trivial cases of these events were dropped from the study totals wherever review of documentation would allow adequate interpretations. The relative impact is shown in the Mariner Subsystem Environmental Test Summary where the following count was defined.

Specification Procedure Errors	21 incidents, 5 changes, 1 rework
Operator Error	33 incidents, 5 retests, 1 damaged unit

Of the total incidents described above only half were retained for the study in that they involved a significant overtest.

SPECIFICATIONS AND PROCEDURES

The preponderance of this type of difficulty is not unexpected however unfortunate. The review of data indicates 2 areas of potential danger to Viking.

- Design by Test: Many incidents of specification or procedure changes gave evidence that an estimated value, limit or approach would be employed and the actual experience later substituted. The approach may be tolerable in establishing the telemetry value for a quiescent sensor but is inherently dangerous and very costly in qualification and acceptance testing (Biosatellite).
- Size of Procedure Documents: Long duration tests with complex procedures tend to experience more changes (anticipatable). Mariner experience suggests breaking up of long tests into separable items such that shift change carry overs and small shadings to procedures do not impact the total document.

AGE, CABLING, OPERATOR ERROR

These items are interrelated in that program changes seem to be a major source for these difficulties. Biosatellite 30 Day vehicle had to employ a block type configuration control for changes due to difficulties late into the flight acceptance phase of the program. AGE and procedures became numerous and complex. On the other hand experiences with Lunar Orbiter seems to show that a limited AGE complex can introduce problems. The paper tape printout for telemetry with virtually no provisions for simultaneous continuous monitor slowed the detection of operating anomalies. The effect was to increase the number of recorded anomaly incidents and thereby slow the progress of testing.

FACILITIES:

Experience seems to show some evidence of "make do" with existing equipment. Mariner and Biosatellite both report incidents of inadequate shakers for initial tests, also temperature control problems in thermal vacuum exposures. Viking must be on guard against this type of difficulty particularly in the area of vendor capability.

RELATIVE IMPACT OF TEST PROBLEMS

The incidence of test related anomalies appears to allow the following summarization.

- Biosatellite 30 day had the largest impact. The spacecraft was difficult to operate because of a series of interdependent fluid flow systems. The system was new, changes persisted until late into Flight Acceptance.
- Lunar Orbiter; median impact. Spacecraft had a relatively straight forward operation. The system was new with: effectively 3 vehicles of Flight configuration before the first launch.
- Mariner; Minimum Impact. Major changes introduced into an established concept. Principal difficulties were associated with the areas of change. Majority of problems resolved with the qualification units.

VIKING PROJECT IDENTIFIED NEED

Comparison of Test Considerations Between the Viking Lander and Mariner '69

	ESTIMATE FOR VIKING LANDER	MARINER
Total No. of Assemblies Subject to Envir. Tests	620	208
Max. No. of Separate Qual. Env. for any Item	15	10
No. of Individual Assy. Env. Tests	2010	1020

6.0 COMPARATIVE DESCRIPTION OF SPACECRAFT

The material presented in this section provides a summary description of the hardware and mission configurations of the programs in this study. This material is provided, since differences in hardware and mission can be expected to effect the type and quantity of failure/anomalies. This section contains the following tables and figures:

6.1 Comparison of Missions

Table 6-1 presents the Viking Mission Phases alongside corresponding flight activities of the other spacecraft. For each mission phase, the principal events and constraints are listed together with an estimate of the duty cycles and elapsed time envelopes for the mission duration. The Viking Lander can be seen to have the most complex mission, with total operating and flight times matched only by Nimbus B.

6.2 Comparison of Subsystems

Table 6-2 presents the comparison of subsystems and components proposed for the Viking Lander with similar items for the other programs. To the extent possible the arrangement is by subsystems or equipment groupings which essentially accomplish the same functions within their respective spacecraft. Component assembly items in the programs studied which have known similarities to proposed Viking Lander equipment are identified. The following comments bring attention to items of special interest in Table 6-2.

Except for Viking and Biosatellite, thermal control of these spacecraft was limited to passive or semi-passive radiative elements. Biosatellite contained a sophisticated heat transfer subsystem consisting of a circulating fluid, heat exchangers, and differential displacement valves which absorbed heat from some elements, surrendered it to others, and radiated the excess externally with both a radiator and a water boiler. All spacecraft studied used thermostatically controlled electrical heaters for short duration needs, as in conditioning dormant components such as gyros, IR scanners, etc.

The Biosatellite communication equipment consisted of VHF beacon, receivers, and transmitters with a data rate of 22.4 K bps. All data was transmitted in real time except the outputs of three shift registers and a recovered tape recorder. The other spacecraft all employ an S-Band for direct communications links. Viking and Mariner share the additional problem of transmission delay (about 40 minutes round trip for Viking).

Lunar Orbiter, Mariner, and Nimbus employ solar cells as the basic power source. Biosatellite used a fuel cell which converted oxygen and hydrogen into power with water as a by-product; the oxygen and hydrogen were stored cyrogenically. The Viking Lander will employ a SNAP 19 RTG, of the same type now flying as an experiment on Nimbus B.

All the four spacecraft utilized gyro sensing rate control, maintaining a fixed attitude through either infrared sensors or star trackers. Active control employed jetted cold gas which for Nimbus supplemented inertia wheel fine control. By comparison the preliminary approach for Viking Lander attitude control will employ a pulsed hot gas reaction system supported by rate sensing from gyros.

6.3 Comparison of Part Counts

Table 6-3 presents an electronic parts count by subsystem and part type of each spacecraft. The Viking Lander counts are based on the best estimates available at the date of publication. It is realized that since integrated circuits and MSI are counted as parts, total parts count is not an accurate measure of equipment complexity; however, this information is generally provided in technical descriptions.

In order to provide circuit complexity comparisons an "equivalent discrete parts" count is listed in Table 6-4. In computing the equivalent total parts, the total number of discrete parts within an integrated circuit or MSI is counted, rather than counting the IC or MSI as one part.

For hardware complexity comparison, the numbers of part connections for each spacecraft are listed in Table 6-4. For example, a resistor has 2 connections, a transistor has 3, and a typical integrated circuit has 14.

6.4 General Comparisons of Other Spacecraft with Viking Lander

The Viking Lander is expected to weigh about twice as much as Nimbus or Biosatellite and about three times as much as Mariner or Lunar Orbiter (see Table 6-4). Like Mariner, it will experience long delay in signal time of flight, and so must have the self-contained capability to carry out its functions (which are considerably more varied and complex); like Biosatellite, it will enter (with a much longer entry into an unfamiliar environment and with the requirement for precise alignment); like Lunar Orbiter, it will deploy complex mechanisms; like Nimbus, it will perform a number of dissimilar experiments (but of greater intricacy and involving laboratory type devices). Uniquely Viking has three unprecedented novel constraints: the sterilization requirement, the extensive and intricate science program, and the scope of the planetary entry program.

Based on the several electrical parts counts shown in Table 6-3, the Viking Lander design complexity is projected as 3 to 4 times that of Mariner 69, which was sizeable in itself. Since the latter benefitted considerably from prior program experience, development of the Viking Lander is clearly an unprecedented task for an unmanned space unit. In addition to the electronic complexity is the wide variety of electromechanical elements such as the RTG, thermal switch, soil sampler, terminal descent engine, and the science experiments. Furthermore, the Viking Lander must accomplish what has never been done, performance of a major exploratory mission after extended dormancy and planetary entry.

TABLE 6-1 COMPARISON OF MISSIONS

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE 30 DAY	NIMBUS B
GENERAL: 2 Flights to Mars Oribts and Landings. Spacecraft: Each unit has an Orbiter and a sterilized, soft lander section Mission Data . Atmospherics; measurements from orbit, during entry and on the surface. . Photography; from orbit and from the surface . Biology and Seismology, from the surface. Mission Time, 17 month design minimum requirement. Constraint: Both launches performed during opportunity window of Aug.-Spet. 1975.	GENERAL: 5 Flights to Lunar Oribt. Spacecraft: Self contained orbiting wet film photo reconnaissance unit. Mission Data: . Photo reconnaissance, wide angle and closeup viewing of entire Lunar surface with emphasis on Apollo landing sites . Extended operation to measure Solar radiation fluxes, micrometeroid population and determine mass asymetry of the moon. Mission Times varied from 74 to 339 days. Constraint: 5 launches completed within a 12 month period, 1966, 67.	GENERAL: 2 Flights to Mars using heliocentric fly-by. . Spacecraft: Design drawn from previous ('64) successful sys. Increased data handling capability, extended range, science, improved operating flexibility. . Mission Data: • Photographic: Far encounter and closest approach. • Radiation, Infrared and ultra violet for atmospheric constituents, concentrations, and planet environment temperatures. . Mission Times: 130 to 170 days with capability for extended operation up to 1 year. . Constraint: Both launches accomplished within the opportunity window of Feb-Mar. 1969	GENERAL: One flight of extended duration in near earth orbit ending with atmospheric entry and recovery. . Spacecraft: Automated orbiting biological laboratory containing a live primate. . Mission Data Biological and psychological data from flight and during atmospheric entry. Final data from post flight examination of primate passenger. . Mission Times. 8 days for primate capsule, 45 days for spacecraft bus. . Constraint: Post flight recovery of primate alive.	GENERAL. One flight of extremely long duration, Near earth orbit, global coverage for meteorology measurements. Spacecraft: Fourth unit in a series of nearly identical items, modifications were increased data handling/storage, up rated science for meteorology and addition of 3 new experiments, 2 are Viking relatable (Gas Bearing rate Gyro and SNAP19 RTG) Mission Data, •Meteorology: Measurements to to support global weather predictions using ultra violet visible and infrared radiations. •Ground Tracking. Relay monitor from remote RF emitters. •Space Operating Environment Snap 19-RTG, and gas bearing rate gyro. Mission Time: Continuous, now exceeds 17 months from launch. Constraint: First Nimbus B lost due to booster failure, Unit in flight fabricated from spares and reworked assemblies.
LAUNCH ACTIVITY Range. AFETR, Complex 39 Booster. Titan III-Centaur Preparation Time for one flight assembly totals ~45 days. Special Preparations . Terminal Sterilization of the encapsulated lander . Sterile insertion of fuel and gasses to the Lander . Fueling and gas charging of the Orbiter . Mating of Orbiter and Lander sections Powered Flight: Titan III into Earth orbit (partial); Centaur into Heliocentric Type II Mars Transit. Constraints: Spacecraft units must be prepared to provide a ready spare for each launch.	LAUNCH ACTIVITY Range. AFETR. Booster. Atlas-Agena Preparation Time for one flight assembly ~40 days Special Preparations . Mating of photographic section to spacecraft . Fueling and charging of gasses Powered Flight: Direct Ascent into Lunar Transit Constraints: Special Conditions, Timing of program was such that the photo system never operated with the spacecraft until launch preparation. All testing at Boeing was performed with photo simulators.	LAUNCH ACTIVITY Range. AFETR. Complex 36 Booster: Atlas-Centaur Preparation time for one one flight assembly ~40 days Special Preparations . Mount solar panels . Fuel and charge with gasses Powered Flight. Direct ascent into Mars transit Constraints . Spacecraft launched one month apart. . Each launch has a ready spare spacecraft.	LAUNCH ACTIVITY Range. AFETR., Complex 17A Booster: Long Tank, Thrust augmented delta. Preparation time for Spacecraft ~120 days Special Preparations . Load cryogenic hydrogen and oxygen . Energize fuel cell . Install live primate Powered Flight. Ascent into earth orbit, 220-240 miles, 33.7° inclination Constraint: Primate kept alive.	LAUNCH ACTIVITY Range. VAFB Booster: Thrust augmented Thor - Agena D. Preparation Time for Spacecraft ~90 days Special Preparations . Install RTG units . Mate to companion spacecraft, Secor 13 Powered Flight: Ascent to Polar, near Earth, Orbit. Page 1 of 6

TABLE 6-1 COMPARISON OF MISSIONS (CONTINUED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE 30 DAY	NIMBUS B
<u>CRUISE OPERATION</u> <u>Duration:</u> 305 to 340 days <u>Principal Events</u> . Solar panel and antenna deployment . Attitude reference acquisition (sun, canopus) . Midcourse corrections at 5-20 days post launch and 5-20 days pre-encounter <u>Spacecraft Functioning</u> . 100% for orbiter power, attitude control and lander RTG . 10% Orbiter RF tracking . 5% Housekeeping telemetry . 1% Command . 30Sec max, propulsion for midcourse. <u>Constraints:</u> . Control of 2 units after second launch . Transmission delay time for RF continuously increases and reaches ~ 15 minutes for a oneway transit.	<u>CRUISE OPERATION</u> <u>Duration</u> ~3 days <u>Principal Events</u> . Solar panel and antenna deployment . Attitude Reference acquisition (Sun, Canopus) . Midcourse corrections as required (2 nominal) <u>Spacecraft Functioning</u> . *100% for power, and attitude control* . 25% tracking and housekeeping telemetry . 2% command . 10 Sec. Max. for Propulsion <u>Constraints</u> Multi vehicle operation began with Launch #3, however Lunar transits were performed one at a time.	<u>CRUISE OPERATION</u> <u>Duration:</u> 160 and 127 days <u>Principal Events</u> . Solar Panel and antenna deployment . Attitude reference acquisition 2 steps. Sun-Vega, Sun-Canopus. . Midcourse corrections, one each. M-VI after 4 days; M-VII after 12 days . Scan platform (Science) unlatch, performed after 8 days for M-VI, and 42 days for M-VII <u>Spacecraft Functioning</u> . 100% Power and Cruise attitude control . 5% tracking and housekeeping telemetry . 2% Command . *6-8 seconds propulsion for* midcourse <u>Constraints</u> . Control of 2 units after second launch . Transmission delay time increase to ~6 minutes for one way transit.	<u>ORBITAL DWELL.</u> <u>Duration:</u> 8 days for primate capsule, 45 days for spacecraft. <u>Principal Events</u> . Primate Science data on a continuous basis . Electrocardiograms . Electroencephalograms . Psychomotor games . Urinalysis . Vital Functions <u>Spacecraft Functioning</u> . 100% Power, attitude rate sensing, life support, data acquisition . 15% telemetry tracking and command during station passes averaging 2 per orbit, 7 min. each. <u>Constraints</u> . Attitude rate control to 10^{-5} *earth* g. . Live recovery of primate requires near real time evaluations of vital functions (i.e. between station passes).	<u>ORBITAL DWELL.</u> <u>Duration:</u> 18 months and continuing. <u>Principal Events</u> . Solar Panel deployment . Attitude stabilization and control to local vertical . Continuous global coverage of atmospheric measurements from science sensors . Continuous *monitor of* rate package and RTG performance . Commanded receive store and retransmit thru IRLS experiment. <u>Spacecraft Operation</u> . 100% power, attitude control atmospheric science data handling and storage . 10% RF links for command transmission, data playback *as station passes.* 7 min. each averaging 2 per orbit. . ~1% IRLS operation <u>Constraints:</u> Simultaneous orientations of Science - camera axis aligned to local vertical, and solar panels facing the Sun.
<u>MARS ORBIT INSERTION</u> <u>Duration:</u> ~5 days <u>Principal Events</u> . Attitude position for burn . Orbit insertion velocity burn ~20 minutes . Orbit determination and trim for reconnaissance <u>Spacecraft operation</u> . *100% Orbiter* power, and attitude control, Lander RTG . ~25% Ranging mode telemetry and tracking . 2% command . ~20 minutes, Orbiter propulsion <u>Constraints</u> . Time required for command transmit and verify is ~ 30 min . Spacecraft perform orbit insertions one at a time.	<u>LUNAR ORBIT INSERTION</u> <u>Duration</u> ~5 days nominal <u>Principal Events</u> . Attitude position for Lunar Orbit Insertion . Orbit insertion velocity burn (7.5 to 10 minutes) . Orbit determination . Orbit trim to final for photographic operations <u>Spacecraft Functioning</u> . 100% power and attitude control . ~15% tracking telemetry . ~3% command . *10.5 minutes max. propulsion* operation <u>Constraints:</u> . More than one vehicle in Lunar Orbit after flight 3 . Spacecraft placed in orbit one at a time.			

TABLE 6-1 COMPARISON OF MISSIONS (CONTINUED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE 30 DAY	NIMBUS B
MARS ORBIT DWELL, RECONNAISSANCE Duration: 5 to 20 days Principal Events . Reconnaissance of pre-selected landing sites by photography IR spectroscopy and water detection . Orbit trim to synchronous for the selected landing point Spacecraft Operation . 100% orbiter power and attitude control, Lander RTG . ~50% orbiter science . ~20% telemetry of science data . ~5% command Constraints . Reconnaissance mode for first unit completed and landing executed before second unit encounter. . 3 RF sources during second unit operation. . RF delay time ~ 30 minutes.	LUNAR ORBIT PHOTOGRAPHY Duration: 15 to 30 days Principal Events . Photo exposures (211) each containing wide angle and telephoto sections . Orbit trims for photo coverage . On-board processing of film . Read out of film via analog video RF link (varied from 70 to 100% of exposures performed) Spacecraft Operation . 100% power, attitude control, photo subsystem . 50% nominal, communications as command, telemetry tracking and video readout (operate when earth is in view and not in direct line with sun) Constraints: . Multiple units in orbit after 3rd launch, up to 3 maximum . Only one spacecraft in photo operation at a time. . Video transmission as analog FM possible only because of relatively short transmission distance (Lunar, not inter-planetary)	ENCOUNTER Duration ~3 days Principal Events: . Energize Science and perform spacecraft readiness check . Attitude orient for encounter . Photograph at early approach i.e., 7 x 10^5 miles . Operate all science during period of close approach . Store data . Transmit data real time and playback of stored information. Spacecraft Operation . 100% all operating sub-assemblies Constraints: . Fly by close approachs are 6 days apart. M-VI on July 31, M-VII on Aug 5, 1969. . RF transit time was ~6 minutes one way at the time of encounter.		

TABLE 6-1 COMPARISON OF MISSIONS (CONTINUED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE 30 DAY	NIMBUS B
SEPARATION ACTIVITY Duration: ~ 3 days Principal Events: . Precise orbit determination to program lander for entry. . Energize and checkout lander. . Batteries brought to full capacity. . Open Bioshield . Eject lander Spacecraft Operation . 100% Orbiter attitude control and Power (maximum draw from orbiter to lander during checkout and battery charge) . ~50% Orbiter telemetry for checkout of Lander and tracking for orbit determination . ~5% command operations Constraints: . RF transit times are ~ 15 minutes one way . Control of 3 RF sources after first lander ejection, 4 RF sources at second lander eject . Only one unit in separation mode at a time . Operation of lander system after ~1 yr. of storage.	EXTENDED MISSION LUNAR ORBIT Duration 45 to 309 days Principal Events . Monitor of radiation levels . Monitor for meteoric material impacts . Tracking of Orbits . Orbit changes as desired . Spacecraft environmental response Spacecraft Operation . 100% power and attitude control . ~10% Telemetry operation for tracking and data from S. C. . ~2% Command Operation • Propulsion as required Constraints: A maximum of 3 spacecraft in simultaneous operation.		SEPARATION AND DEORBIT Duration: ~30 minutes Principal Events . Attitude position for deorbit . Separate Entry Capsule . Spin up to 60 rpm . Deorbit burn 10 sec. . De spin to 10 rpm . Separate deorbit system . Primate data recorded Spacecraft Operation . 100% Power, Life Support and full attitude control stabilization. Capsule RF . 100% Deorbit program and separation system . 50% spacecraft RF for command, tracking monitor of events. Constraints: . Entry conditions must not injure primate	

TABLE 6-1 COMPARISON OF MISSIONS (CONTINUED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE 30 DAY	NIMBUS B
MARS ENTRY AND LANDING Duration: 4 to 8 hours Principal Events: . Orbiter operates as relay RF station . Lander attitude control and deorbit burn $\sim$20 min. . Atmosphere Entry, Heatshield . Parachute, Deceleration . Terminal Descent and Landing . Science data during all atmospheric penetration Spacecraft Operation . 100% Orbiter in relay mode, Power, attitude control Data storage, relay receiver, RF transmission . 100% lander power, attitude control, guidance, control. . 3% to 100% Relay RF, real time entry data. . 30 minutes entry science . 12 minutes heatshield . 2 minutes parachutes . 40 sec. terminal descent engines Constraints: . Landing sequence is automatic . During first landing control 3 RF sources, during second landing control 4 RF sources.	TERMINATE: Duration: $\sim$1 hour Principal Events . Attitude position for burn . Velocity change to cause Lunar Impact $\sim$30 sec. . Spacecraft crashed on Lunar Surface. #1. Crashed to prevent interference with flight 2, (80 days) #2, 3, crashed to allow single vehicle operation during Lunar eclipse, Oct. 67, 339, 264 days #4. Lost contact, 74 days crash by orbit precession #5. Crashed after sudden loss of attitude control gas. 183 days Spacecraft Operation . 100% Power, attitude control . 1% command sequence 10 min. telemetry readout . $\sim$30 sec. propulsion Constraints Destruction of #2 and #3 done in conjunction with #5 in extended mode operation.		EARTH, ENTRY AND RECOVERY Duration $\sim$30 minutes Principal Events . Atmospheric entry with heat shield . Recovery and Landing by 3 stage parachute . Retrieval beacon energized . Primate biological data throughout entry, recorded Capsule Operation . 100% power, life support and data handling. . $\sim$10 minutes Parachute . $\sim$10 minutes Beacon Constraints: Control of entry corridor for preferred retrieval via air snatch.	

TABLE 6-I COMPARISON OF MISSIONS (CONCLUDED)

VIKING LANDER	VIKING ORBITER	MARINER'69	BIOSATELLITE 30 DAY	NIMBUS B
<u>LANDER OPERATION</u> <u>Duration:</u> 90 days minimum <u>Principal Events</u>: . Photography of site . Biological life determination . Identification of materials, organics. . Atmospheric measurements . Seismic measurements . Orbiter acts as relay for photographic data <u>Lander Operation</u> . 100% Power, and control sequencer, data conditioning and record. . ~50% Biology . ~10% Photograph, Gas Chromotograph, Mass Spectrograph, Relay and Direct Telemetry . 2-5% Meteorology, Seismology and command. <u>Constraints</u>: . After second lander 4 RF sources must be accomodated . Orbiters have missions beyond lander support . Surface mission extends thru superior conjunction with the Sun. Maximum for RF delay times and Solar interferences with RF links.	<u>ORBITER MISSION</u> <u>Duration</u>: 90 days minimum <u>Principal Events</u> . Relay link for photographic data for first 5 days after landing and again 20 days post landing. . Mapping and reconnaissance capability for items of defined interest. <u>Orbiter Operation</u> . 100% Power, attitude control . ~50% Orbiter science and data storage, handling. . ~10% Relay link operation . ~10% Earth Link operation . ~2% command . ~4 min. propulsion in increments <u>Constraints</u>: . Either orbiter must serve either lander after second landing. . Control 4RF sources after second landing . Mission time extends through superior conjunction with the Sun, RF link has maximum distance and maximum solar interferences.		<u>RETRIEVAL</u> <u>Duration:</u> 3 hours <u>Principal Events</u>: . Air snatch attempt missed due to cloud cover. . Water impact, helicopter retrieval alternate employed. . Capsule opened and primate removed (alive). <u>Capsule Operation</u> 100% Power, life support, data handling and record, TLM, recovery beacon. <u>Constraints</u>: . Power system has limit of 12 hours for water impact dwell . Data from entry and in-flight photography dependent upon retrieval. <u>POST RETRIEVAL</u> <u>Duration</u>: 37 days for S/C - Indefinite for primate <u>Principal Events</u>: . Spacecraft operates in extended mode for experimental purposes until fuel cell exhausts supply gasses . Primate receives examinations in vivo and post mortem. . Data reduction of on-board flight records <u>Spacecraft Operations</u> . 100% Power, rate control attitude. . 10% Telemetry . 1% command. <u>Constraints</u>: Tracking of spacecraft required until orbit decay.	

TABLE 6-2 COMPARISON OF COMPONENTS AND SUBSYSTEMS

VIKING SUBSYSTEM AND COMPONENTS	SUBSYSTEMS WHICH PERFORM THE SAME OR SIMILAR FUNCTIONS AS IN VIKING			
	LUNAR ORBITER	MARINER '69	BIOSATELLITE (30 DAY)	NIMBUS B
1. POWER: RTG plus NiCd Batteries, D. C. with control and regulation. 1. RTG; SNAP 19, 2 units 2. Battery Ni-Cd, 2 units 3. Power Distribution Assembly 4. Motor Driven Switch 5. Power Conditioning Assemblies (Charge Controller D. C.-D. C. Converters) 6. Load bank 7. Shunt Regulator 8. Bioshield Power Assembly (Float Chargers and Regulators) 9. Staging Connector	1. POWER: Solar array plus NiCd Batteries, D. C. with control and regulation. 1. Solar panels; 4 units 2. Storage Battery, 2 units, *2 3. Battery Charge Controller, *4 4. Battery Voltage Shunt Regulator, *7 5. Booster Regulator * Identifies similarity to Viking Component as identified by Item Number. i.e., Ni-Cd Battery, Viking Item 2	1. POWER: Solar array plus AgZn Battery, D. C. plus A. C. with conversion, and regulation. A. C. to, 2.4 KHz, Square Wave 400 Hz Single Phase 400 Hz Three Phase 1. Solar Panels; 4 units 2. Battery 3. Power Conversion Assy. *3, 5 4. Power Regulator Assy. *3, 5, 8	1. ELECTRICAL POWER AND DISTRIBUTION: Fuel Cell Plus Subordinate functioning Batteries. D. C. with control and Regulation 1. Fuel cell: H_2-O_2 Cryogenic 2. Oribital Battery Ag-Zn 3. Entry Battery Ag-Zn 4. Recovery Battery Ag-Zn 5. Deorbit Battery (thermal) 6. Capsule Heater Battery Ag-Zn 7. Inverter Power Supply, *5 8. Inverter Power Supply R/V, *5 9. Inverter Power Supply S/C, *5 10. Disconnect Assy; Cable, *9 11. Separation Switch. 12. Timer Programmer 13. Fuel Cell Controller 14. Disconnect Assy; Cable, *9 15. Separation Switch.	1. POWER AND RTG EXPERIMENT: Solar Array plus Ag-Zn batteries supplemented by RTG. D. C. with control and regulation 1. RTG; SNAP 19, 2 units (Expt.) 2. Solar panels, 2 units 3. Batteries, 8 units 4. Power control module *5 (regulator chg. controller) 5. Shunt regulator, *7 6. Panel Drive Motor Assy, *4 7. Power Conditioner, *5, 3 (DC/DC Conv. Switching) 8. Dummy Load, *7
2. GUIDANCE AND CONTROL: Position and velocity references feed into a computer; outputs are signals to a valve drive amplifier. Subsystem provides orientation and stabilization for entry; terminal descent maneuver; and all sequencing from pre-separation thru landing. . Thrusters are part of propulsion 1. Velocity Reference Unit 2. Attitude Reference Unit (includes rate gyros) 3. Flight Computer (Guidance and Control Modes) 4. Terminal Descent and Landing Radar 5. Radar Altimeter 6. Engine cut of Sensor 7. Valve Drive Amplifier 8. Aeroshell Altimeter Antenna 9. Lander TDLR Antenna	2. ATTITUDE CONTROL: Inertial and rate reference plus computation. Controls: attitude positioning, stabilization, specific orientation hold, and provides sequencing for flight path control. . Thrusters are part of velocity and reaction control 1. Sun Sensor and Remote Eyes 2. Canopus Star Tracker 3. Inertial Reference Unit, *1 4. IRU Gyro, *2 5. Accelerometer 6. Control Assembly, *3 Flight Electronics 7. Programmer 8. Crystal Controlled Oscillator 9. Magnetic Core Memory, *3 10. Closed Loop Electronics, *3 11. Switching Assembly, *7	2. ATTITUDE CONTROL: Sun and Star Sensors plus Rate Sensors. Provides Position and Stabilization by means of cold gas (N_2) thrusters. Control calculations performed by Central Computers - Sequencer 1. Control Electronics, *2 2. Gyro Control Assy, *2 3. Sun Sensor Primary 4. Sun Sensor Secondary 5. Sun Gate Detector 6. Canopus Sensor 7. Thrust Vector Control 8. A/C Gas Assembly 9. A/C Jets	2. ATTITUDE CONTROL AND DEORBIT: Rate and attitude sensors plus control unit. Rate stabilization to 10^{-5} Earth G, plus position for deorbit separation. Cold gas reaction thrusters part of system. 1. A/C Programmer, *2 2. Rate Gyro, 3 axis, *2 3. Magnetometer 4. Magnetometer Programmer 5. IR Horizon Sensor 6. Pneumatic AC Controller 7. Pressure Regulator 8. Valves, Solenoid, 8 units 9. Jet Controller 10. Filter Pneumatic 11. Deorbit Programmer, *2	2. ATTITUDE CONTROL AND RATE MEASURING PACKAGE EXPERIMENT: Continuous control for axis alignment to local vertical and solar array exposure to Sun. Inertia wheels plus cold gas (freon) thrusters 1. Flywheel Assembly, 2 axis 2. Flywheel Electronics, *1, 2 3. Rate Gyro, 1 Axis, *2 4. Pneumatic Assembly 5. Sun Sensors 6. Solenoid Valves 7. Horizon Sensor 8. Jet Nozzles 9. Rate measuring package, *2 containing air bearing rate gyro and electronics

TABLE 6-2 COMPARISON OF COMPONENTS AND SUBSYSTEMS (CONTINUED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE	NIMBUS B
3. COMMAND CONTROL: Performs Timing and Control functions Digital Logic Circuitry 1. Flight Computer (Command and Control Modes) 2. Soil Sampler Control Unit 4. COMMUNICATIONS Contains RF Equipments in UHF and S Bands. Relay and Direct Links for Commands and Transmission 1. UHF Radio Assy (Transmitter, Beacon, Receiver) 2. S Band Radio Assy (Command Receivers and Dectors, Ranging Module Modulator Exciters TWTA (2) 3. Antenna, UHF 4. Antenna S Band Lo Gain 5. Antenna S Band Hi Gain 5. TELEMETRY SUBSYSTEM: Data Gathering, Storage and Encoding for PCM Transmission. 1. Cruise-Entry Data Aquisition Unit 2. Lander Data Acquisition and Processing Unit 3. Data Storage Memory 4. Tape Recorder 5. Sterilization Multiplexer 6. Transducers 7. Bridge Completion Units	3. COMMUNICATIONS: Contains Command Receivers and Decoders, Data Encoders and Transmitters in S Band. Handles PCM and FM Analog Data. No Separate Storage. 1. TWT Amplifier and Power Converter *4-2 2. Travelling Wave Tube *4-2 3. Hi-Gain Antenna 4. RF Rotary Joint Assembly 5. Antenna Position Controller 6. Command Decoder *4-2 7. Modulation Selector 8. Transponder *4-2 9. PCM Multiplexer Encoder*5-12 10. Instrumentation Signal Conditioner *5-1,2 11. Low-Gain Antenna *4-4 12. Temperature Transducer Installations *Indicates Corresponding Item for Viking by Subsystem and Component, i.e. Traveling Wave Tube, *4-2, is found in Lander Communication Subsystem (4) S Band Radio Assy (2)	3. RADIO FREQUENCY: Contains Antennas, Command Receivers, and Data Transmitters in S Band PCM. 1. RF Assembly *4-2 (Command plus Receivers, S Band Exciter and TWTA) 2. Antenna Hi-Gain *4-5 3. Antenna Lo-Gain *4-4 4. FLIGHT COMMAND: Provides Command, Detection and Decoding 1. Flight Command Assembly *4-2, 3-1 5. CENTRAL COMPUTER SEQUENCER: Provides Internal Timing, and Command Storage Using Digital Logic Capacity 128 Words 1. C. C. S. Assembly *5-3 6. DATA STORAGE: Tape Recorders for Analog and Digital Storage with Digital Output 1. Data Storage Assembly *5-4 7. DATA AUTOMATION: Provides Control of Data for Telemetry in Real Time to Record and Playback Modes 1. Data Automation Assembly *5-1,2 8. FLIGHT TELEMETRY (PCM) Provides Bit Rate Selection, Encoding, Signal Conditioning, and Modulation. 1. Flight Telemetry Assembly *5-1,2 2. Sensors for Eng. Data	3. TELEMETRY TRACKING AND COMMAND: Provides Command Receivers, Decoders, Storage, Tracking Beacon, and UHF Telemetry. PCM and Continuous Wave Data Transmission 1. Command Receiver *4-1 2. Command Decoder, Digital *3-1 3. One Word Storage Prog. *3-1 4. Beacon *4-1 5. PCM Multicoder *3-1 6. Entry Transmitter *4-1 7. Converter, Controller *5-1 8. Commutor, PAM, 30x10 *5-2 9. Antenna Coupler 10. Antenna Power Divider 11. Tape Recorder Bio Med. *5-4 12. Diplexer 13. Transmitter, Orbital *4-1 14. Subcarrier Oscillator 15. Sensors 16. Antenna, TM, and Command 17. Antenna, Entry 18. Current Limiter 19. Premod Filter In addition Command Control Functions were performed by the following items listed in the Biosatellite Power Subsystem. a. Power Controller, RV *3-1,2 b. Power Controller S/C *3-12 c. Timer Programmer, *3-1 d. Fuel Cell Controller, *3-1	3. COMMAND Provides Receivers, Decoders, Time References, and Control. 1. Command Receiver *4-2 2. Interface *3-1 3. Clocks (2 Units) *3-1 4. PCM TELEMETRY Provides Signal Conditioning, Encoders, Storage and Transmission of Data From Engineering Sensors and Experiments 1. Multicoder *5-2 2. Transmitter *4-1 3. T/M Electronics *5-2 4. Tape Recorders (2 Units) *5-4 5. REAL TIME TRANSMISSION SYSTEM: A High Speed Data Transmission Capability in Real Time for Handling Cameras and IR, Radiometer Data. 1. Radiometer Data Conditioner *5-2 2. Transmitter *5-2 6. HIGH DATA RATE STORAGE SYSTEM: A Storage and Transmission Capability for 66KBPS PCM with Redundant Channels 1. Recorders Tape (2 Units) *5-4 2. Multiplexers (2 Units) *5-2 3. Electronics Assy (2 Units) *5-2 4. Repeaters (2 Units) *5-2 5. Transmitters S Band (2) *5-2

TABLE 6-2 COMPARISON OF COMPONENTS AND SUBSYSTEMS (CONTINUED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE	NIMBUS B
6. SCIENCE INSTRUMENT On-Board Experiments for Entry and Surface Measurements. A. ENTRY 1. Accelerometers Triaxial 2. Pressures; Stagnation and Ambient 3. Temperatures: Stagnation and Ambient 4. Mass Spectrometer 5. Retarding Potential Analyser B. SURFACE 6. Imagery, Cameras 7. Biology; Pyrolytic Release Labeled Release Light Scattering. Gas Exchange 8. Gas Chromotograph - Mass Spectrometer Organic Analysis 9. Meteorology (2 Units) Wind Velocity, Direction, Temperature. Water Content (Humidity) 10. Seismometer	4. EXPERIMENTS Carried as Follows: 1. Photo subsystem *6-6 self-contained assy. of 34 items with 2 cameras, wet film development, readout to telemetry. 2. Micrometeroroid Detector, (Listed as part of the Communications Subsystem) 3. Radiation Dosimeter *6-7 Scintilation Counter Assy. (Listed as part of the Communications Subsystem)	SCIENCE EXPERIMENTS are carried as 5 Subsystems. 9. SCAN CONTROL SUBSYSTEM Moveable Platform with Sensors and Control Electronics Provides mounting for instruments. 1. Scan Control Electronics See 3-2 2. Narrow Angle Mars Gate 3. Far Encounter Planet Sensor 10. ULTRA VIOLET SPECTROMETER Measures UV Spectrum 1050 to 4350A, Solar and Planet 1. UV Spectrometer Assy. *6-9 11. TELEVISION Contains 2 Cameras Wide Angle and High Resolution Plus Electronics 1. TV Electronics *6-6 2. Wide Angle Camera 3. High Resolution Camera 12. INFRA RED SPECTROMETER A 2 Channel Unit providing Overlapping Coverage for Wave Lengths from 1.9 micron to 14.3 microns 1. IR Spectrometer and Cryostat *6-9 2. IRS Gas Storage 3. IRS Gas Valves 13. INFRA RED RADIOMETER A 2 Channel unit with Wave Length from 8 to 25 u. 1. IR Radiometer Assy. *6-9	SCIENCE EXPERIMENTS Related to a Live Primate. 4. PRIMATE PAYLOAD: Primate Support and Control 1. Pellet Food Dispenser *6-7 2. Water Dispenser *6-7 3. Feces Collection and Storage *6-7 4. Light Assy. *6-7 5. Camera 16 MM 6. Camera Controller *6-6 7. Life Support Controller *3-2 8. Trace Gas Contamination Control Assembly 9. Reservoir Disinfectant 10. Couch Primate 5. PRIMATE EXPERIMENTS 1. Primate (Live, Instrumented) 2. Psychomotor Game Assy. 3. Sensors - EEG Vital Function 4. Heparin Equipment *6-7, 8 6. URINE TRANSPORT Urine analysis & monitor 1. Storage Tanks Assembly 2. Valves, 3 configurations 3. Collection and Emergency Storage Sub Assembly 4. Urine Analysis Expt. *6-7 7. LIFE SUPPORT Cabin Atmosphere Control 1. Gas Management Assy. *6-7 2. Valves Pressure Relief (2 configurations) 3. Filter 4. Amplifier O_2 Sensor Partial Pressure	A total of 9 experiments, 7 are carried as sub-systems and 2 are integrated into other sub-systems (RTG, and Rate Measuring Package) 7. IMAGE DISSECTOR CAMERA 2 Chanel television system 1. Camera and Electronics Assembly *6-6 8. SATELLITE INFRARED SPECTROMETER (SIRS) 1. Control Module 2. Sensor Section 3. Sensor Electronics 4. Data Storage Unit 8 channel 5. Power Module 9. HIGH RESOLUTION IR RADIOMETER (HRIR) 1. Radiometer Sensor 2. Radiometer Electronics 10. MEDIUM RESOLUTION IR RADIOMETER (MRIR) 1. Sensor 2. Electronics 3. TM Module Sig. Cond. 11. INFRARED INTERFEROMETER SPECTROMETER (IRIS) 1. Sensor 2. Electronics 12. MONITOR ULTRA VIOLET SOLAR ENERGY (MUSE) 1. Sensor 2. Electronics 13. INTERROGATION, RECORDING AND LOCATION SYSTEM (IRLS) Receives Transmissions from remote sensors on Earth, stores the data, then re-transmit to a tracking Station 1. Antenna 2. Receiver 3. Memory (see 5-3, 4) 4. Data Handling Units 5. Transmitter (See 4-1) 6. Control Module

TABLE 6-2 COMPARISON OF COMPONENTS AND SUBSYSTEMS (CONTINUED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE	NUMBUS B
7. PROPULSION: Monopropellant N_2H_4 Provides Thrust for Attitude Control; Velocity Change for De-Orbit; and Terminal Descent Propulsion. 1. Deorbit Thrusters (8 Units) 2. Tank Assy. Deorbit (2 Units) 3. Filter Deorbit 4. Rocket Engine Assembly Terminal Descent (3 Units) 5. Tank, Terminal Propellant 6. Filter, Terminal System 7. Hand Valve	5. VELOCITY AND REACTION CONTROL: BiPropellant Hypergolic 100# Thrust. Provides Impulse for Midcourse and Orbit Trim Maneuvers. 1. Rocket Engine, Velocity Control *7-4 2. Tank-Fuel *7-5 3. Tank-Oxidizer *7-5 4. Actuator-Thrust Vector Control 5. Thruster Assembly *7-1 6. Tank-Nitrogen *7-2 7. Velocity Control Pressure Regulator *7-2,5 8. Reaction Control Pressure Regulator *7-2,5 9. Nitrogen Pressure Sensor 10. Test and Fill Valve Assy. 11. Pressurization Squib Valve 12. Nitrogen Filter 13. Filters *7-6	14. PROPULSION. Mono propellant N_2H_4 50# Thrust, Provides thrust for mid-course maneuvers. 1. Rocket Engine *7-4 2. Propellant Orifice 3. Filters, 2 configurations *7-36 4. Valves (11 configurations) *7-7 5. Pressure Regulator *7-2,5 6. Tanks (2 confs.) *7-2	8. DEORBIT ROCKET A solid motor which provides ΔV for deorbit and entry (see Attitude Control) 1. Retro Rocket Assy. 9. TANKAGE Provides on-board storage for Cryogenic gasses (O_2H_2), Life Support Gasses (Air, O_2) and Fluids (water & coolants) 1. Cryotankage Assy. *7-2,5 2. Accumulator Water 3. Tank Water Storage 4. Tank Storage Metabolic Liquid 5. Tank, N_2 *7-2 6. Valves (7 configurations) 7. Regulators O_2, N_2 *7-2 8. Filters, Water 9. Sensors	None.
8. DECELERATOR Single Stage Mortared Parachute System 1. Mortar Assembly 2. Pressure Cartridge 3. Main Parachute Assembly	None	None.	10. RECOVERY SUBSYSTEM 3 Stage Parachute plus Retrieval Aids 1. Parachute Assy, 3 stage *8-3 2. Recovery Programmer (see 3-1) 3. G Switch 4. Beacon (*4-1) 5. Cutters (Bagline, Reefing) (see 12)	None.
9. THERMAL CONTROL: Active System . RTG on Pad Cooling, Water Circulation. . On Planet, Lander Body Temperature Control. 1. Thermostats 2. Water Circulation Coil 3. Thermal Switch 4. Phase Change Material 5. Insulation 6. Heaters	6. THERMAL CONTROL: Passive System Insulation Thermal Paint	15. TEMPERATURE CONTROL Active System, senses radiant energy. Control by insulation, heaters, and louvers over the electronics bays. 1. Temperature Control Flux Monitor Electronics 2. TCFM, Sensor 3. Louver Assy. 4. Insulation Blankets *9-5	11. THERMAL CONTROL Provides Temperature Control for cabin during launch, orbit and entry; employs a liquid circulation primary with heat rejection by evaporation and radiation. 1. Pump Dual *9-2 2. Heat Exchange 2 position *9-3 3. Thermal Control Assy. *9-3 4. Accumulator Coolant 5. Heaters *9-6 6. Boiler Evaporative Assy 9-4 7. Switches (2 types) 8. Valves (4 types)	14. TEMPERATURE CONTROL Active System. Senses Temperatures at Component Locations. Heaters or Cool thru Louvers. 1. Control Module 2. Heaters 3. Louver Assy. 4. Insulation

TABLE 6-2 COMPARISON OF COMPONENTS AND SUBSYSTEMS (CONCLUDED)

VIKING	LUNAR ORBITER	MARINER '69	BIOSATELLITE	NIMBUS B
10. MECHANISMS: Release and Actuator Units 1. S Band Antenna Deployment Mech. 2. Soil Sample Devices and Distribution Unit 3. Soil Sample Collector Head 4. Camera Boom Deployment Mech. 5. Sampler Boom Assembly 6. Leg Assembly Deployment Mech. 7. Bioshield Valves 8. Bioshield Filter	7. STRUCTURE & MECHANISMS Plus Specific Items from other Subsystems 1. Actuators (Antenna and Solar Panel) *10-4 2. Camera Thermal Door 3. Actuator Thrust Vector Control (from Velocity and Reaction Control) *9-6	16. MECHANICAL DEVICES: 1. Solar Panel Boost Dampers 2. Solar Panel Cruist Control Dampers 3. Lo Gain Antenna Support Dampers 4. Pyro Arming Switch 5. Separation Initiated Timer 6. Scan Platform Valves 7. Scan Actuator (From 9) *10-6	12. SEPARATION AND DEORBIT 1. Feed line disconnect 2. Baro Switch 3. Separation Switch 4. Fluid Disconnect Assy. 5. Disconnect Assy Elect. Cable	No Separate System 1. Solar Panel Unfold Actuator 2. Separation Release Assy.
11. PYROTECHNIC CONTROL: Firing Circuitry for Lander. 1. Lander Pyro Control Assy. 2. Bioshield Pyro Cont. Assy. 12. PYROTECHNIC DEVICES: 1. Valves N.O. 2. Valves N.C. 3. Tubecutters 4. Seprn. Nuts 5. Linear Charge 6. Mortar Charge 7. Pressure Cartridge Conf. 8. Detonators 9. Initiators 10. Pin Pullers	No Separate Subsystem, Parts of Structure and Mechanisms and Reaction Control. 1. Pin Pullers *12-10 2. Cartridges *12-8.9 3. Shut Off Valves (VRC) *12-1 4. Fuel-Oxidizer Valves (VRC) *12-1	17. PYROTECHNICS 1. Pyro Control Electronics *11-12 2. Propulsion Squibs *12,1,2 3. Pin Puller and Squibs 4. Release Device and Squib Assy. *12,8,9	No Separate Sub-System Device and Control Throughout Others Recovery 1. Ejector Assy. *12-8,9 2. Pyro Units in Cutters (Squibs, Charge) *12-8,9 Separation 1. IFD Squib *12-8,9 2. Squib Separation Actuator *12-8-9 Attitude Control 1. Actuator, Magnetometer Boom *12-8,9 De-Orbit 1. Explosive Bolt *12-4,8,9 2. Explosive Act. Valve *12-1,2	No Separate System 1. Squib Solar Panel Release Actuator *12-8,9 2. Explosive Bolt, Separation *12-4,8,9

TABLE 6-3 SUBSYSTEM PARTS COUNTS

Notes indicated ( )* appear on continuing page.

SUBSYSTEM/PROGRAM	TOTAL	Capacitors	Crystals	Diodes & Rectifiers	Fuses	Inductors	Microcrts.	Resistors	Thermistors	Xformers	Transistors	Motors	Pots	Relays	Switches	Connectors	Filters	Other (2)*
I. POWER																		
VIKING LANDER	3315	479		978		46	893	569		36	115			165		(15)*	34	
LUNAR ORBITER (1)*	496	57		76	8	8		239	4	4	65			2		23		10
MARINER (1, 19)*	1994	300	2	411	47	8	137	803	2	46	180		8	21	17			12
BIOSATELLITE	781	110		173				312		7	69	1		59		15	20	15
NIMBUS	1230	32		301		3		632			156			1		105		
II. GUIDANCE AND CONTROL																		
VIKING LANDER (4, 5, 6, 7)*	10178	1010	1	545		60	1365	2466	10	22	614			4			3	4078
LUNAR ORBITER (2)*	7208	775	1	897		21	629	3685	3	62	973			2		153	7	0
MARINER (2)*	1615	274		356		2	31	738		24	138	4		25				23
BIOSATELLITE	2293	320		472		49	108	1004			281	4		5		24	20	6
NIMBUS	5515	610		1404		73		2587			713			21		78		29
III. COMMAND																		
VIKING LANDER (5, 8)*	13251	2085		2210			2635	4175			1858							288
LUNAR ORBITER (3)*	2788	892	4	203		105	142	942	1	118	208	1		4		112	55	1
MARINER (8)*	4930	309		1361	6	75	478	1793		95	694			119				
BIOSATELLITE	2445	460		564		69	6	974			226			37		12	12	85
NIMBUS	3943	407		1689		38		1360			381			2		37		29
IV. TELEMETRY																		
VIKING LANDER (5)*	3540	370		250			1632	882			42							364
LUNAR ORBITER (3)*	4995	408		2063	1	21		1850	1	64	549				20	6	9	3
MARINER (10)*	7945	904	5	900	10	105	1508	3396		80	1016	2		11				8
BIOSATELLITE	2037	239	1	564		49	94	819			229	1		4		12	20	5
NIMBUS (11)*	5959	513		730		654		1570			2412			26		16		38
V. SCIENCE - PHOTO																		
VIKING LANDER	2000						2000											
LUNAR ORBITER (12)*	4324	496		624	16	6		2474	21	5	566	12		6		56	17	25
MARINER (13)*	2780	399		423		16	464	1086		14	324	8	32	10	1			3
BIOSATELLITE (14)*	157	7		47		2		67			28	1				1	4	
NIMBUS (12)*	4006	592		648		17		1973			675			34		50		17
VI. SCIENCE - OTHER																		
VIKING LANDER (5, 16)*	1445	70		60			945	270			80	3			5			12
LUNAR ORBITER (18)*																		
MARINER	2331	408		391	6	34	60	1082	8	6	323	3		2				8
BIOSATELLITE	947	32		88			300	279			200	7		21		5	15	
NIMBUS (17)*	2394	490		270		29	84	1028			278			16		145		54
VII. TOTALS																		
VIKING LANDER	33729	4014	1	4043		106	9470	8362	10	58	2709	3	50	169	5	(15)*	37	4742
LUNAR ORBITER	19811	2628	5	3863	25	161	771	9190	30	253	2361	13		14	20	350	88	39
MARINER	21595	2594	7	3842	69	240	2678	8898	10	265	2675	17	40	188	18	(15)*		54
BIOSATELLITE	8660	1168	1	1908		169	508	3455		7	1033	14		126		69	91	111
NIMBUS	23047	2644		5042		814	84	9150			4615			100		431		167

TABLE 6-3 SUBSYSTEM PARTS COUNTS (CONCLUDED)

NOTES ( )*

1. Does not include solar cells (L.O. 10,656, Mariner 17432)

2. Does not include memory cores (L.O. 3242 ceramic, Mariner 4373 ferritic)

3. Command includes Communications "A" Subsystem less the TWTA.
   Telemetry includes Communications "B" Subsystem less the TWTA.

4. Radar Altimeter & ECOS from Letter PM/3563/WRS

5. Based on RA-3701068 Part Count

6. Includes TDLR Part Count from Ryan Prop. #29170-152

7. ARU Part Count from Ham. Std. Prop.
   VRU Part Count from Bell Aero. Prop.

8. Includes Command and CC&S

9. Deleted

10. Includes Data Storage and Automation

11. PCM Subsystem

12. Photo S/S Only.

13. Includes TV and Scan Control

14. Recovery Camera

15. No Part Count Available

16. Entry Science Mass Spectrometer and Landed Science Soil Acq. Mech. and GC/MS Only.

17. High Resolution and Medium Resolution Infrared Radiometer Experiments

18. No significant other experiments

19. Includes Pyro, Temp. Control and Devices

TABLE 6-4 SUMMARY COMPARISON OF FIVE SPACECRAFT

	VIKING LANDER	LUNAR ORBITER	MARINER	BIOSATELLITE	NIMBUS
Weight Dry	2350#	578#	821#	1380#	1249#
Weight Total	2650#	853#	850#	1530#	1269#
No. Subsystems	12	7	17	12	14
No. Science Experiments	12	4	4	6	9
Total Discrete Parts (1)	33,729	19,811	21,595	8,660	23,047
Number of Microcircuits	9,470	771	2,678	508	84
Equivalent Discrete Parts (2)	262,907	38,315	85,867	20,852	25,063
Number of Part Connections	168,480	54,898	79,169	25,799	61,112
Number of Months From Program Start to First Flight	72	29	37	33	N/A
Flight Duration of Primary Mission - Months	17	1 - Primary 12 - Limited	5	1	12
Total Flight Duration	-	74-339 Days	9 Months	8 Days (Experiment) 45 Days (Engrg. S/S)	18 Mos.
Configurations Qualified at the Component Level	Est. 88	28	52	155	53
Component Assemblies Built	653	420	360	946	No Data
Individual Test Exposures	2010	1396	1020	2312	No Data

(1) Includes microcircuits (2) Includes a count of 25 for each microcircuit

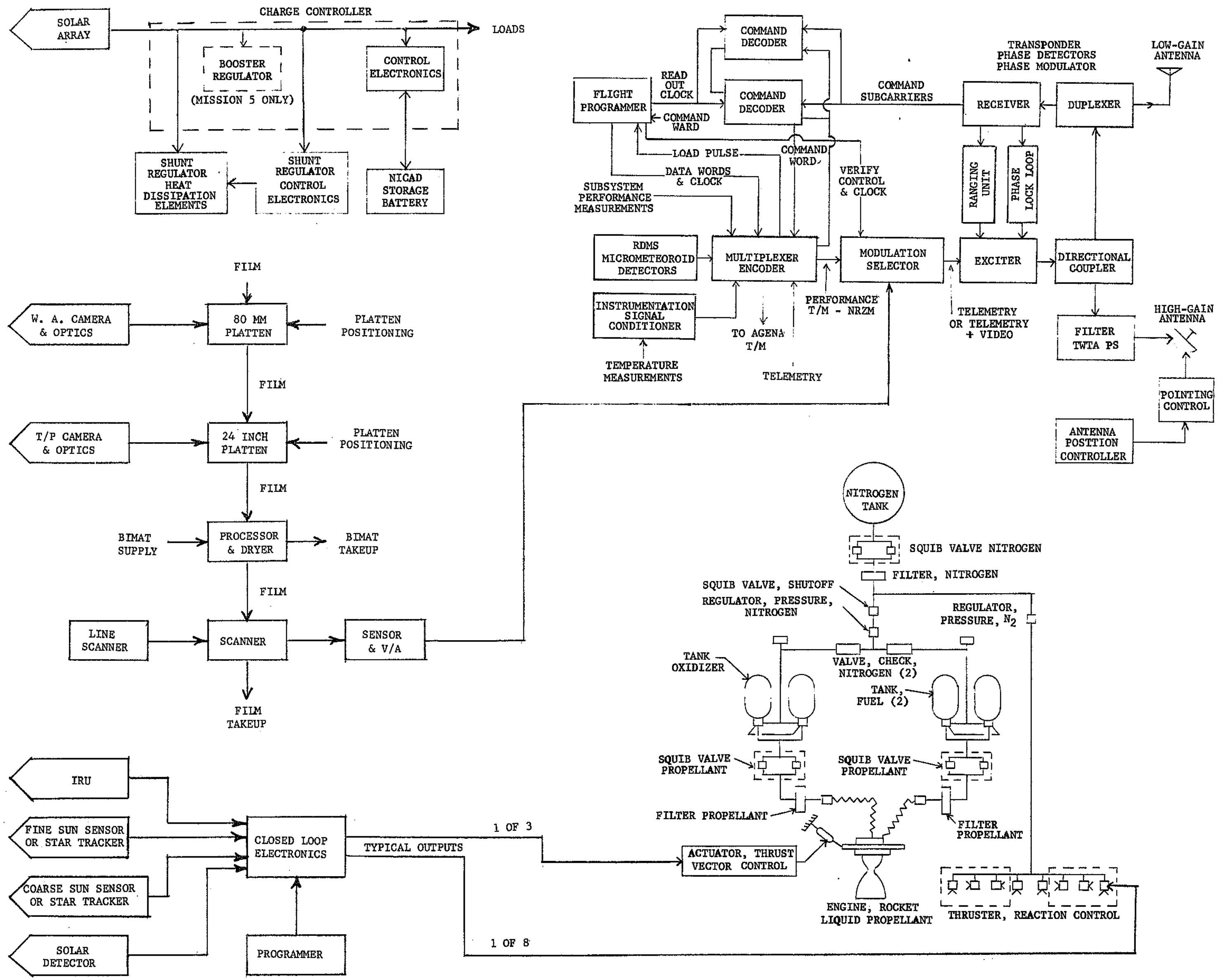


Figure 6-1. Block Diagram, Lunar Orbiter

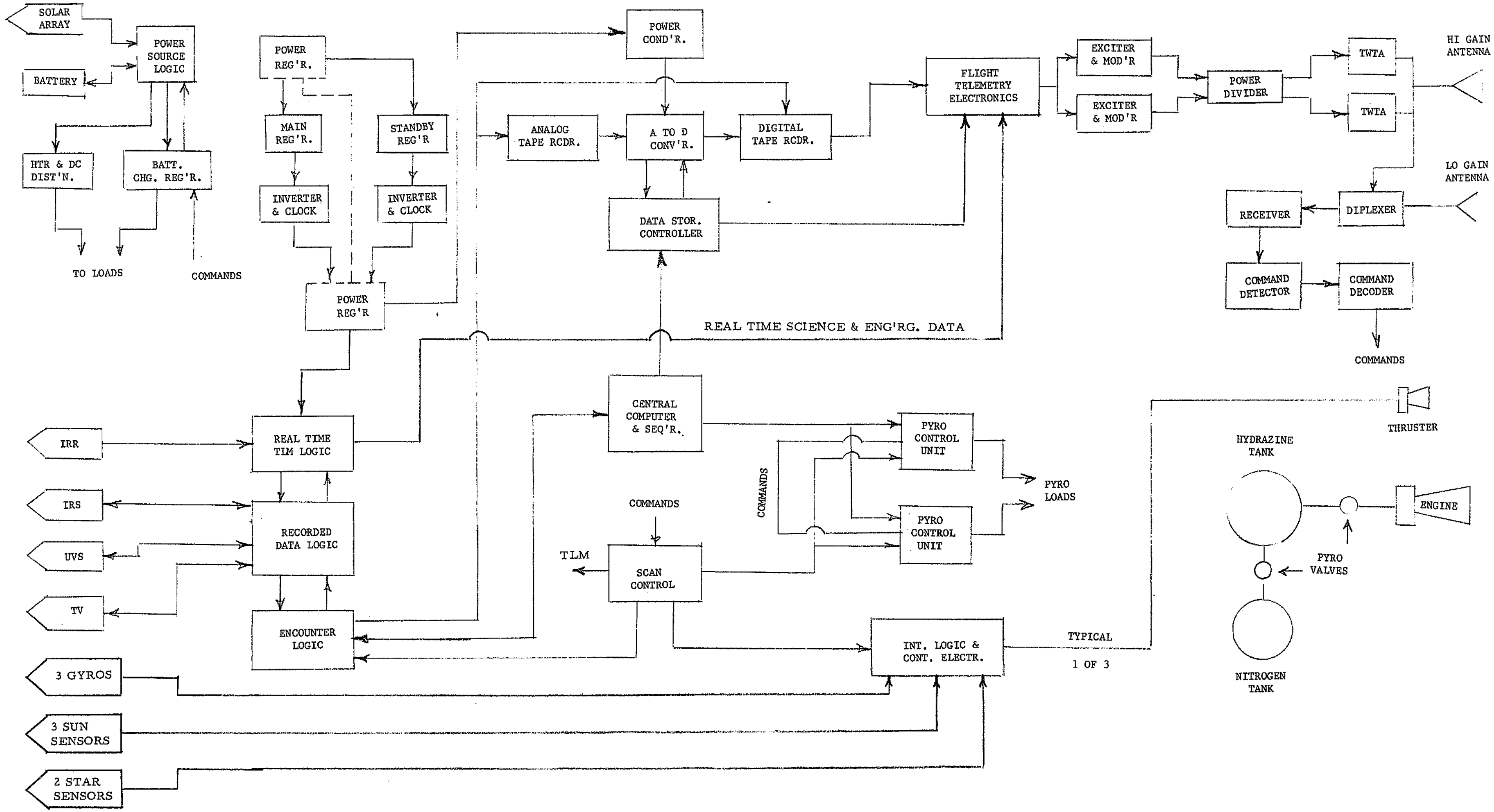


Figure 6 - 2. Block Diagram, Mariner

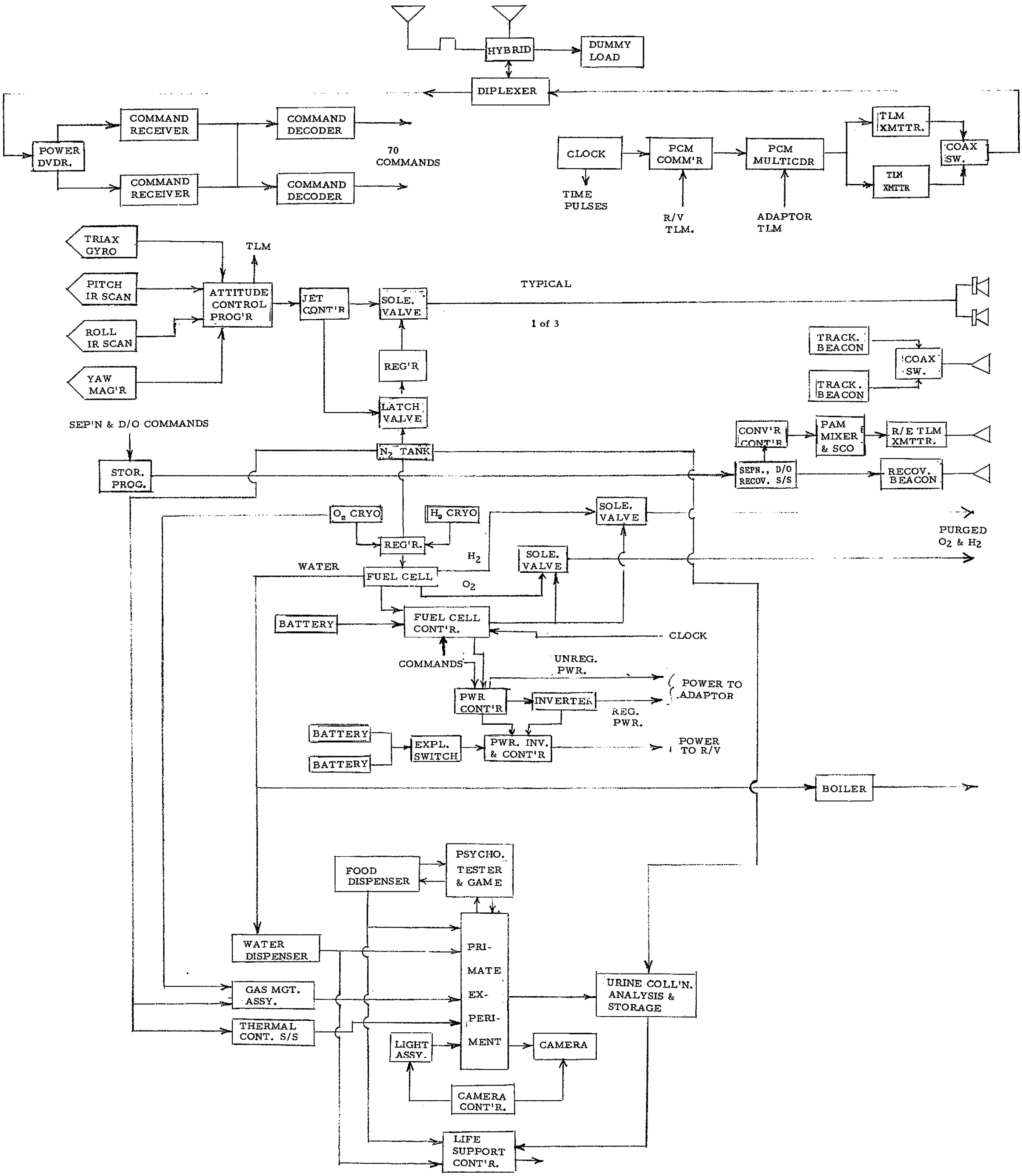


Figure 6 - 3. Block Diagram, Biosatellite

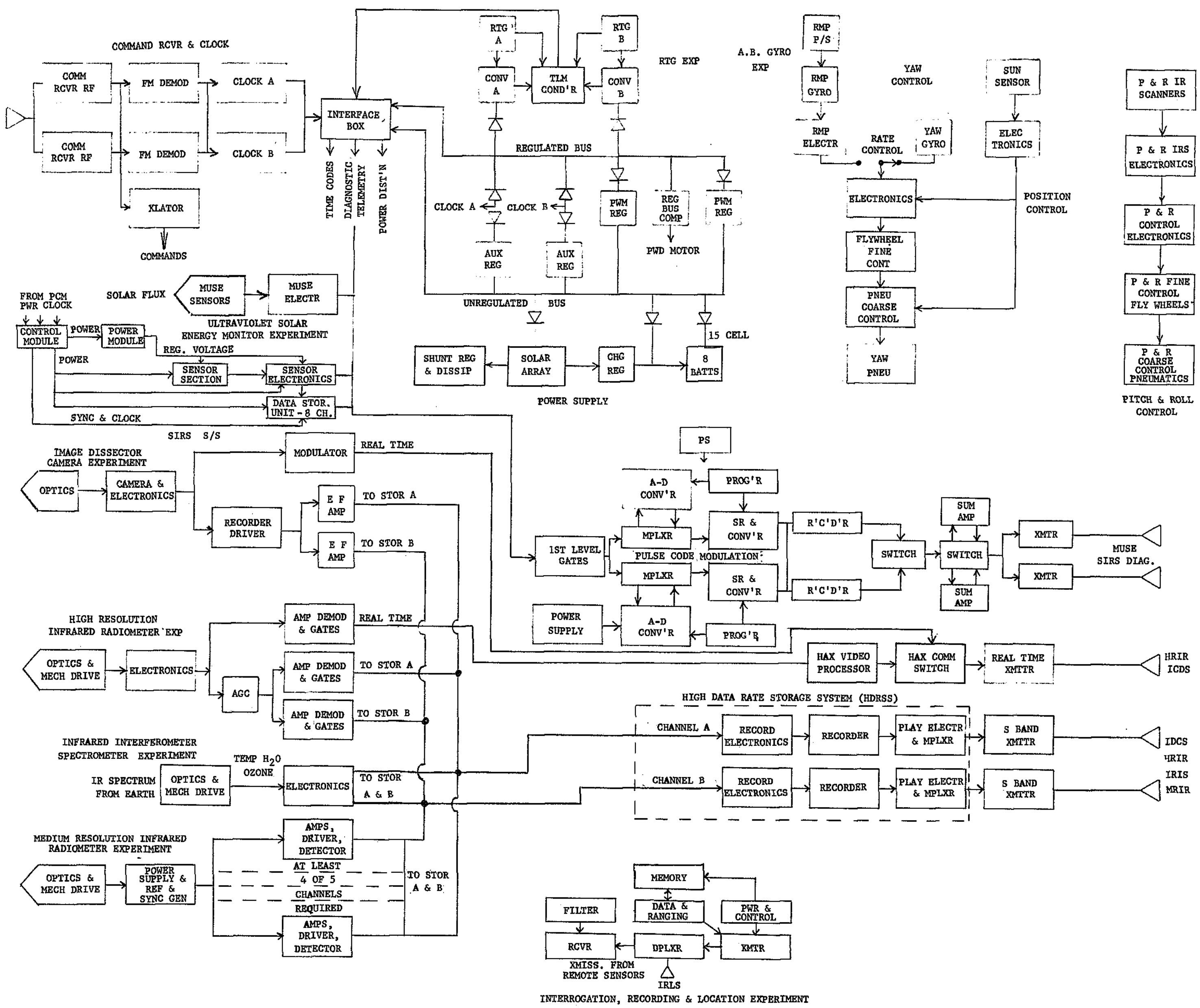


Figure 6 - 4. Block Diagram, Nimbus

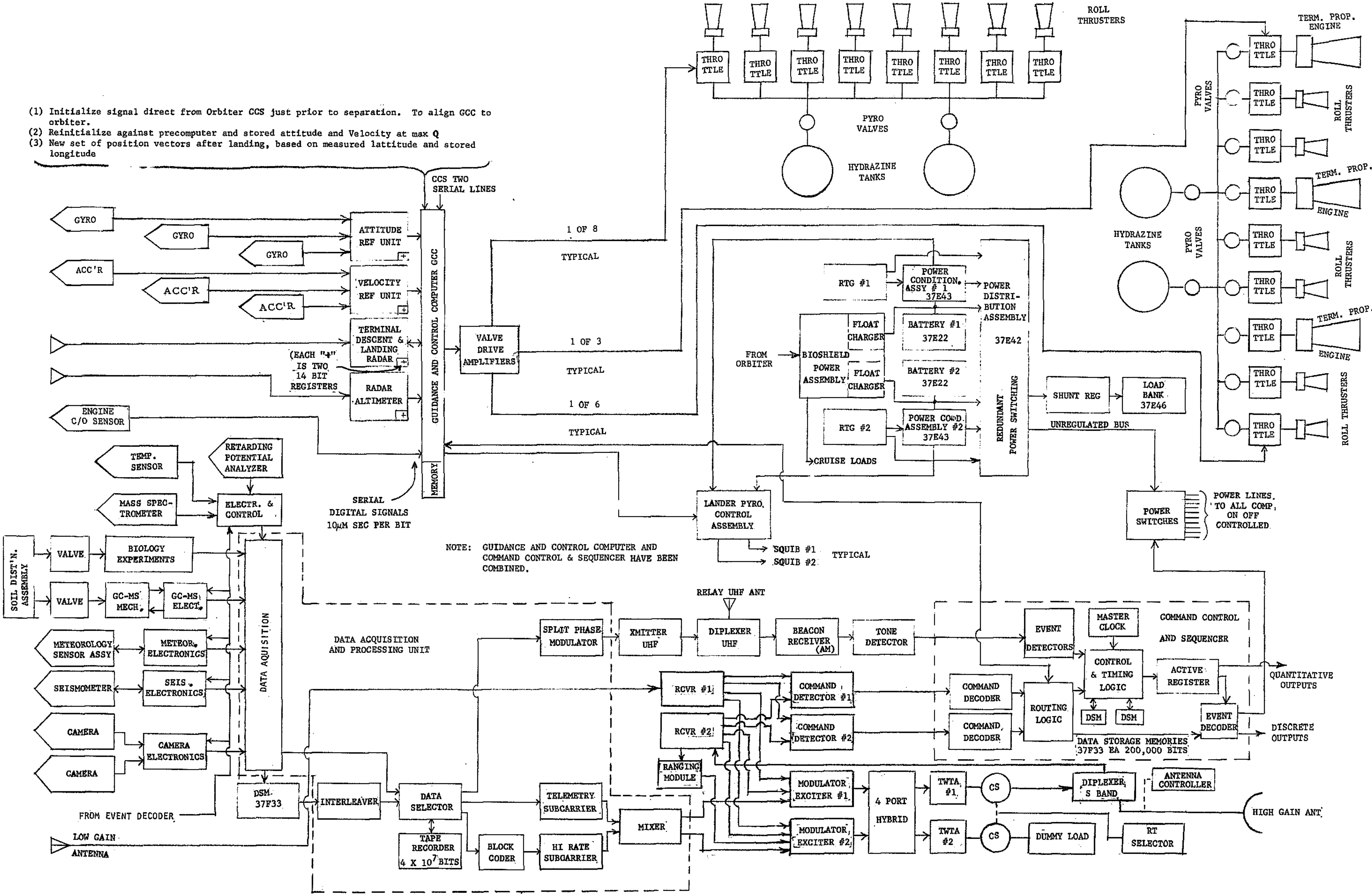


Figure 6-5 Block Diagram, Viking Lander

7.0 COMPARATIVE DESCRIPTION OF TEST PROGRAMS

The material presented in this section provides detailed tabular summary description of the ground test programs in the four space projects in this study. In addition, the data presented has been evaluated against the resulting space flight performance for each of the space programs.

This material is provided since the types and levels of testing can be expected to effect the type and quantity of flight failure anomalies. This section contains the following tables and figures:

For each of the programs in the study, Section 7.1 provides a review of the test opportunities, environmental exposures, and test problems in conjunction with the flight problems shown in Section 3. Section 7.2 defines criteria for use in establishing or evaluating future test programs. These criteria are extracted from the reviews presented in Section 7.1 and test related data appearing in sections 4 and 5.

7.1 Evaluation of the Test Programs in the Study

An evaluation of each test program is presented in paragraphs 7.1.1 through 7.1.4 below, with the discussions organized as follows:

A. Opportunity for Test - Comments describe completeness in provisions for uncovering or detecting potential flight problems. The basic data are the block diagrams, which show the project utilization of hardware sets, types of testing and sequences employed.

B. Environments - Describes the deviations from the baseline test program environments appearing in Tables 7-1 and 7-2. The test levels and durations contained in the tables include the exposures applied to both components and spacecraft for conditions of qualification, flight acceptance, and special demonstrations.

C. Test Problems - Identification of difficulties encountered in performance of the test program. These problems tended to compromise the test programs outlined in the block diagrams.

D. Undetected by Ground Test - Identifies flight failures (from the tables of section 3) for which ground testing had been provided, but the applied tests did not uncover the problem. These incidents are described in context with the test program and the applied environments.

7.1.1 LUNAR ORBITER (Figure 7-1 and Tables)

A. Opportunity for Test - The project employed a thorough ground test program with extensive component testing leading up to ground test of a complete spacecraft. The ground testing planned to precede the first flight was completed, and in addition, at the time of first flight the component acceptance testing for the remaining spacecraft equipment was also completed. The early completion of component and most spacecraft testing resulted in early recognition of operating problems.

B. Environments - The environment exposures followed those defined in the tables except for specific items as follows:

Photo Subsystem: The photo subsystem was treated as a special case. Most components contained within the subsystem assembly were not individually subjected to qualification or flight acceptance; instead, the subsystem was tested as a whole. In addition, the sine vibration tests for the photo subsystem were conducted at spacecraft rather than component environmental levels (see Table 7-2).

Component Vibration: A vibration test of a spacecraft assembly was performed prior to component design completion. Data from this test was used as a basis to reduce vibration levels as follows: Component Sine Vibration: Input levels were revised downward for 10 components which included the transponder, inertial reference unit and the traveling wave tube amplifier. Component Random Vibration: Spectrum adjustments and level reductions as much as 35 percent were allowed to 12 components in the communications and propulsion subsystems.

C. Test Problems - The test problems stemmed from the slow delivery of the photo subystem assemblies. Spacecraft tests at the factory were not able to employ flight photo subsystems and the use of simulators did not necessarily provide appropriate system interface exposures. The flight photo subsystems received flight acceptance testing in the subcontractor plant and were installed in the spacecraft at the launch site for subsequent functional tests as part of the pre-launch preparation.

D. Undetected by Ground Test - The records from flight show 13 incidents which were undetected by ground tests provided for their control. The incidents which either resulted in loss of data or tedious flight operations work-arounds were:

1. Thermal Paint Degradation (LO-I-2) - The thermal emissivity characteristics deteriorated in space, and required operation of the spacecraft pitched "off sun". The test of the thermal paint did not detect the deterioration because measurements of emissivity were made before and after, but not during the solar vacuum environment.

2. Photo Subsystem EMI Problem (LO-I-8) - An unexpected transient pulse effect smeared all of the high resolution image motion compensated telephoto pictures during the first flight. The high resolution photos were of primary interest. During photo subsystem tests and testing at the ETR, no integrated operations were performed which would have detected this failure mode. This problem was corrected by a redesign and verified by test before the second flight.

3. Mechanisms Degradation - Several mechanisms degraded in flight. The Readout Scanner cam did not return to "spot stop" position sporadically throughout the mission (LO-II-3); backup commands overcame the problem with minor loss of readout time. The Readout Scanner Encoder terminated readout sporadically with no commands given (LO-IV-2 and LO-IV-5); work-arounds controlled the problem until the failure became complete, which resulted in a 10 percent loss of photos. A camera shutter sustained double trip which lost one photo (LO-IV-4). The deterioration of these mechanisms was potentially due to either wear or launch vibration; design changes provided increased resistance to both effects.

7.1.2 MARINER '69 (Figure 7-2 and Table)

A. Opportunity for Test - The Mariner '69 test program employed the same approach as that successfully used for Mariner '64 to establish the quantities of hardware, and flow of test operations. For Mariner '69, early operation of an assembled spacecraft was achieved by installing prototype subsystems and components into the proof test model (Qualification Unit). This action generated some extra problem failure reports,however, operating problems were thus identified and resolved early.

B. Environments - In the environmental qualifications of subsystems,applicable data from Mariner '64 was used to reduce testing. The environmental exposures described in Table 7-1 were subject to the following interpretations and modifications:

1. Sine Vibration: Local resonance conditions were superimposed upon the sweeps; input force levels reached as much as 2 times nominal over frequency bands of 100 Hz.

2. Acoustic Testing: Testing of subassemblies was limited to exterior items only (antennas, star trackers).

3. Transportation Vibration and Bench Handling Shock: Applied to pyrotechnic devices only.

4. Pyro Shock: Applied to science and close proximity equipment (6 items)

5. Humidity: New equipment only (18 items).

6. Explosive Atmosphere: Applied to items operating with high voltages (6).

7. Electron-Proton Radiation: Waived in all cases.

C. Test Problems - The test problems encountered by Mariner '69 came from 3 separate sources. First, in design, the Mariner '69 and Mariner '71 missions were both considered in the setting of interfaces and final use of equipment. The double mission use necessitated designing for wider dynamic ranges in regulators and wider acceptance of signals through interfaces. The spacecraft was more sensitive to some types of disturbances such as EMI pulse transients. The double mission used also restricted quantity of spacecraft hardware available for test purposes only. The spares from the '69 project were allocated to the '71 project. Late deliveries relative to the inflexible launch date compromised the life testing of equipment. Life tests of 7000 hours duration were planned for 25 items. Only 15 were begun and 4 completed. The uncompleted tests had exposure times ranging upwards from 500 hours. The principal problem experienced by the entire project and the one which caused the most difficulty to testing was associated with microcircuits. The introduction of microcircuits to the degree and sophistication employed, apparently taxed the capabilities of the suppliers to deliver in both the quantities and quality required. The change-out of microcircuit modules continued throughout the test program and extended into the final phases of launch preparation.

D. <u>Undetected in Ground Test</u> - The Mariner data from flight shows four incidents which ground testing failed to identify. Of these, 2 were manufacturing defects, and 2 were EMI effects. The problems are described as follows:

1. Reversed Polarity (Manufacturing): The reversed thermal sensor was not detected during space simulation thermal vacuum tests.

2. Wiring Error (Manufacturing): This problem was not discovered during ambient or environmental spacecraft tests. The available information does not say, but it is possible the wiring error was in a module changed late during flight preparation.

3. EMI Noise over Portions of the IRR Scan: This incident is described in Table 5-1. The Mariner records show 3 problem/failure reports dealing with this problem. In ground testing the variances in IRR data during the scan platform movement were observed only when AGE cabling was attached. However, the problem appeared in flight. The investigation of the problem has continued in anticipation of more sensitive equipment to be used in the '71 flight.

7.1.3 BIOSATELLITE (Figure 7-3 and Table)

A. Opportunities for Test - The original concept for the Biosatellite project called for 3 separate flight missions. The first, of 3 days duration, carried general biology experiments; the second, of 30 days duration, carried a live primate; and the last, of 21 days duration, was to have carried live rats and special plants (the 21 day spacecraft was never built). In all 3 spacecraft configurations common equipment was to be used to the maximum extent. Space proven hardware in current series production was also to be employed to the maximum extent. These latter items included recovery and de-orbit components from Discoverer, fuel cells from Gemini and a command subsystem which had been flown in 4 previous applications.

Each of the Biosatellite flight missions was preceded by a thorough test program. Subsystem operations were verified as breadboards employing prototype components. These components were subsequently assembled to produce an early spacecraft. Throughout the program, components common between missions were shared in the assembly of the early spacecraft. The primate (30 day) mission had extra complexity due to the life support interface. The testing for the special interfaces with the primate required an additional spacecraft assembly (spacecraft 601). These data from early tests provided a base for performing the acceptance and qualification of the particular spacecraft fabricated for the missions. Each of the 2 configurations built included a qualification unit and 2 flight spacecraft.

B. Environments - The environmental testing for all spacecraft components applied the conditions associated with the 30 day flight. Within the 30 day mission the environments described in Table 7-1 included 2 limited applications:

1. Sterilization: This environment was only applied to items in the urine transport subsystem and portions of the life support equipment.

2. Fungus: This environment was only applied to equipment located in the recovery capsule (equipment in contact with the primate).

C. Test Problems - The test program for Biosatellite suffered a series of compromises. In design, the multiple use of components and the use of existing flight equipment generated problems with interfaces. A large segment of the failure incidents arose from interface problems and some of the problems persisted into flight. The failure to recover the first flight initiated a redesign and requalification effort. As a consequence of the failure, thermal vacuum was added as an environment in flight acceptance testing of components. All existing flight equipment was recycled to include test of the additional exposure.

The difficulties introduced by the primate mission resulted in continuing changes in the flight spacecraft relative to the qualification and development units. Problems associated with the primate necessitated a major redesign of the psychomotor game and the urine transport equipment just before the flight spacecraft was due for shipment to the launch site. Testing of the redesigned equipment was limited only to flight acceptances as components and indirectly through the flight acceptance test of the spacecraft. The proofing environments at qualification levels were not applied.

D. Undetected in Ground Test - The records from the flights of Biosatellite spacecraft show 7 incidents which were not discovered by the intended ground test. These problems are described as follows:

1. Reversed Polarities: These problems total 3 incidents involving "g" switches and the outputs of command receivers. These are discussed in detail in Table 5-1 and 5.5.8.

2. Psychomotor Game - Food Dispenser: The late redesign of the game unit precluded exhaustive ground testing. Flight records indicate 3 incidents which most probably would have appeared in further ground testing.

3. Transmitter-Receiver Overlap: This incident occurred during the first flight and a discussion of the details indicates a type of problem which could appear with future testing or test equipment. The designs of the Biosatellite telemetry transmitters and command receivers provided for ample margins of isolation, although the receiver showed sensitivities 6 to 7 db below the specification threshold. However, the 7000 Hz frequency associated with the decoder address word turned out to be a harmonic of the 1792 bits per second telemetry rate. The presence of the harmonic was sufficient to allow telemetry bits to enter the receiver and become intermixed with the digital address for the command decoders. The spurious bits changed the addresses, the decoders did not accept the commands. During flight command access was difficult.

Ground tests with either commands or telemetry separately operated without difficulty. Combined operation with air link RF transmissions showed occasional problems with command access. On many previous projects using similar equipment in the same facility,air link transmission had always shown problems generally traceable to either the facility or the ground test equipment. Consequently the command link problems during test were attributed to sources other than the spacecraft.

7.1.4 NIMBUS B (Figure 7-4 and Table)

A. Opportunities for Test - The testing of the Nimbus B spacecraft was treated as additional assemblies in a series of like spacecraft. Changes in equipment were relatively minor and had a minimum effect upon interfaces.

B. Environments - The experience from the first 2 flights were factored into the exposure levels and test sequences. In comparison to earlier Nimbus spacecraft,the environmental exposures were reduced in number of environments but increased in the levels and duration of exposures.

1. Vibration: Levels were approximately 2 times those applied to the early spacecraft. In addition,the sinusoidal sweeps simulated local resonances in a manner similar to Mariner '69. The local increases reached 2.5 times nominal over frequency bands of 100 Hz.

2. Thermal Vacuum: A solar panel bearing failure terminated the first Nimbus flight and could be traced to a thermal vacuum induced cause. Subsequent testing was therefore increased in both duration and temperature limits for thermal cycling. Flight acceptance of Nimbus equipment totaled more than 500 hours exposure to temperature cycling in hard vacuum.

C. Test Problems - The testing of the Nimbus B project was compromised from two sources. Commercial grade electrical parts were used in portions of the spacecraft; parts failures in flight comprise the largest single group of significant failure/anomaly incidents. The first Nimbus B spacecraft was lost from a booster failure. The spacecraft flying as Nimbus III was the second unit of the Nimbus B project. This spacecraft was fabricated by using a number of reworked qualification items and the spares from the first unit.

D. Undetected in Ground Test - The Nimbus flight records show only one incident which escaped ground test. A capacitor in the S-band transmitter developed an intermittent contact after approximately 500 hours in flight. The capacitor was structurally unrestrained and supported only by the electrical leads and solder joints. The condition was a design and manufacturing oversight which escaped the pre-flight acceptance vibration and thermal vacuum exposures intended to uncover such deficiencies.

7.2 Summary of Test Related Experiences

Facets of ground testing have been described throughout this report. Section 4 presents data showing failures relative to the applied environments (see Figure 4-8 and paragraph 4.2.3). Section 5 shows the persistent problems related to testing, and Section 7 herein presents an evaluation of the test programs for each project. From these data, criteria emerge which can be employed in both the establishment and evaluation of new test programs. These criteria appear as Table 7-3 and are grouped as follows:

1. Early identification of operating problems - These test program features enhanced their projects by either uncovering operating problems at an early time or reducing the scope of later testing.

2. Establishment of Operating Margins - These features provided operating margins against the potential contingencies associated with unknown or unexpected flight conditions.

3. Test Problems - Conditions present or occurring within a project which either diminished the value of early testing or contributed to flight anomalies.

4. Environments - These environments had exposures applying to all the programs. These data are further expanded in Table 7-4 which summarizes the test exposure levels applied for flight acceptance.

7.3 Comparison of Failures Occurring in Project Time

Figure 7-6 shows a comparison of hardware failures as they occurred throughout the ground test and flight phases for 3 of the projects in the study. The time scale has been normalized into percent of the ground test phase preceding the first flight. For each project, the time begins with the acceptance testing of the qualification components. The ground test phase includes qualifications, flight acceptances, and any special demonstrations which were performed concurrently. Failures are shown as percent of the total occurring in both ground test and flight. These include all incidents due to sources in design, manufacturing, parts and devices.

The relative rates at which problems were uncovered by ground testing is similar for each of the 3 projects, although the test programs had different schedules and quantities of equipment. Lunar Orbiter showed a difference from the others at the time of first flight since 3 additional spacecraft were still in ground test. On the other hand, ground testing for Mariner and Biosatellite culminated with a single launch opportunity.

The idealized curve represents a goal for test planning and project scheduling. The early identification of problems allows more ample time to complete any necessary reworks and retests. The final phases of testing can then proceed more smoothly, and with less potential for failures to carry over into flight. Within each of the projects, test program features have been identified which could help to bring future programs closer to the ideal. These features are part of the criteria presented in Table 7-3.

TABLE 7-1 COMPONENT ENVIRONMENTAL TESTS: SUMMARY OF EXPOSURES FOR QUALIFICATION AND FLIGHT ACCEPTANCE.

PROGRAM	DYNAMIC ENVIRONMENTS					GROUND ENVIRONMENTS, THERMAL		BENCH OPERATION, LIFE
	VIBRATION	ACOUSTIC	SHOCK	ACCELERATION	STATIC LOAD	THERMAL CYCLING	TEMPERATURE HUMIDITY	
VIKING LANDER	Sine: 5-250 Hz, 3 axes, 0.4 in. da to 5.3grms Random: 20-2000Hz 3 axes, 10 grms ovr'll. . Res Surv: Manual Swp. to 2000Hz, 2.12grms . Sine(Trnsnt Sim): Swp. 2 oct/min, 5-250-5Hz . 7.5 min/axis Shaped spectrum: Roll up at 3db/octave to 0.075 g^2/Hz at 250-1000Hz Roll of 6 db/octave to 2000 Hz FAT. Sine: As above to 3.52 grms Random: As above to 6.0grms ovr'll . Sweep 4 oct/min up and down . 1 min/axis, Shaped spectrum, eased to 0.025 g^2/Hz 250-1000Hz	144 db Overall, 7.5 minutes Spectrum covers 20-10,000 Hz Reference Points: Level db / Frequency Hz Lo / Hi 134 / 300 peak 130 / 120 / 1030 120 / 40 / 8500 118 / / 10,000 112 / 20 /	PYRO 3000 g peak at 2000 Hz "near" 1200 g peak at 2000 Hz "far" 3 axes, apply 3 times in each direction, 18 shocks total. Shock described as a relative response spectrum with 10% at 113 Hz, 50% at 10,000 Hz LANDING SHOCK 30g for 30 mil'sec, with 5 mil'sec rise and 5 mil'sec decay 3 axes, apply 3 times in each direction, 18 shocks total	19g, 3 axes. 5 min. both directions. 30 min. Total. (Non Entry Items 12g)	Load equals sustained acceleration or landing shock. Direction according to mounting (Applies to tanks and structural items only)	HEAT COMPATIBILITY 275°F for 380 hours total. Atmosphere: dry N_2 Apply as 3 cycles of 60 hrs. each Followed by 5 cycles of 40 hrs. each. FAT: 256°F for 60 hrs, 1 cycle.	-35F to 160F and 95% RH, 346 hrs. Total. Apply as 14 cycles 24 hrs each 2 hrs dwell -35 F ambient RH 2 hrs to 160F, and 95% RH 14 hrs. dwell 160F and 95% RH. 6 hrs to -35F	Operate equal to a Mission Operating hours approx. equal Operation cycles approx. equal Operate at Ambient Conditions The operating times and cycles for functional tests associated with environmental exposures are included in the life requirement.
LUNAR ORBITER	Sine: 5-2000 Hz, 3 axes, to 10.6 grms max. Random: 20-2000 Hz, 3 axes, 20.4 grms ovr'll . Sweep 1 oct/min, Thrust Axis, 0.5"da. to 10.6 grms, Lateral axes, 0.5"da to 7.07 grms . 200 seconds total per axis in steps. 80 sec. at 1.2 grms 40 sec at 7.4, 15, and 20.4 grms ea. Spectrum 0.005 g^2/Hz at 20-277 Hz and 0.45 g^2/Hz at 700-1200 Hz, Roll 12 db/oct 277-700 Hz and 200-2000 Hz. FAT: Sine: as above to 7.07 and 4.73 grms. Random. 17.2 grms 100 sec/axis stepped 40 sec. 2 grms, 20 sec at 6, 12, 17.2 grms. Spectrum shaped as above from 0.3 g^2/Hz.	System Test Only	System Test Only	11.7g Thrust Axis with 4.7g Lateral Axes. 5 min. both directions. 30 Min. Total.	Tests on Structural Model	None	None	RELIABILITY DEMONSTRATION 10^{-5} Torr 60 days with Thermal Cycling. Temperatures, cycle times, and operating periods simulate 2 missions of 30 days each
MARINER '69	Sine: 5-2000 Hz, 3 axes to 13.5 grms. max. Random: 5-2000 Hz, 3 axes 18.1 grms. ovr'll . Transportation sine. 1 Hr. each axis 2-34 Hz at 0.92 grms; 35-48 Hz at 2.12 grms, 48-500 Hz at 3.52 grms. . Sine Sweeps; 1 oct/min. 5-30 Hz 0.75 grms 30-250 Hz at 7.5 grms; 250-600 at 4.5 grms 600-2000 Hz at 13.5 grms. (Bus. Assy. items) . Random; Shaped Spectrum, peaked toward low frequency (30-250 Hz) 60 sec. each axis FAT. Sine sweeps 3 oct/min. 3 axes, levels eased 33% as above. Random: Shaped Spectrum 10.8 grms, 20 sec/axis	148 db overall, 60 seconds Shaped spectrum 50 to 10000 Hz control 1/3 octave bands. Center Frequency Hz / db 80 / 130.5 100-200 / 136 250-400 / 141 500 / 138 1000 / 128 2000 / 119 4000 / 109 8000 / 100	PACKAGE DROP: In shipping container 6 drops on corners and edges 42 inch fall for weights to 20 lb. 36 inch fall for weights to 50 lb. SHOCK (PYRO): 200 g 0.7 mil'sc. terminal peak saw tooth. 3 axes, 5 times each axis 15 shocks total	9g 3 axes, 5 minutes both directions, 30 minutes total time.	Tests on Structural Model	None	Expose to RH of 75% or higher The temperatures timing and numbers of cycles are defined individually for each item	EXTENDED OPERATION Time equal to a flight mission (9 months nominal). Environmental conditions are those of known criticality For temperature and vacuum.
BIOSATELLITE	Sine: 10-2000 Hz, 3 axes, to 8.47 grms max. Random 20-2000 Hz, 3 axes, 21.2 grms ovr'll . Sweep 2 oct/min; 10-14 Hz at 0.5"da; 14-40 Hz at 3.56 grms; 40-62 Hz at 0.06"da; 62-2000 Hz at 8.47 grms. . 4 min/axis, shaped spectrum; Roll up 6 db/octave to 0.30 g^2/Hz at 50-1000 Hz. Roll off 12 db/octave to 2000 Hz FAT: Sine: 11-2000 Hz, 3 axes to 5.65 grms. Sweep 4 oct/min. 11-40 Hz at 2.12 grms; 40-65 Hz at 0.035"da. 60-2000 Hz at 5.65 grms.	System Test Only	35g, 11 mil'sec half sine pulse 3 axes, apply 3 times in each direction. 18 Shocks total (Air Snatch, or Water Impact)	18.8g 3 axes 5 min, both directions. 30 min. total. (Non Entry Items 12.5g)	Limited Tests on Structural Assemblies	STERILIZATION Autoclave at 260°F for 20 Min. FAT: same TEMPERATURE CYCLES 160 to -25F, 45 hrs. total, 5 cycles . 4 hrs. dwell at 160°F . 4 hrs. dwell at -25°F . 0.5 hr. Transit between extremes FAT: 125F for 1 hour	Ambient to 160°F, and 95% RH 120 hrs. total Apply as 5 cycles 24 hrs. each 2 hrs. Amb. to 160F and 95% RH 6 hrs Dwell at 160F 95% RH 16 hr to Ambient at 95-100% RH (Ref: Mil-Std 810B, Method 507 Proc. 1, 5 cycles)	FUNCTIONAL CHECK OUT . Perform before and after each environmental exposure . Time of operation applies to life requirement LIFE The minimum operation must equal ground preparation plus flight. Estimate an operating total, then test. For an estimate of: / test: 0 - 500 hrs / 2 times est. 500-100 hrs / 1000 hrs. 1000 + hrs. / equal to est.
NIMBUS B (B1 and B2)	Sine: 5-2000 Hz, 3 axes to 7.07 grms nom. Random: 20-2000 Hz, 3 axes, 20 grms ovr'll . Sweep at 1 oct/min nominal. At frequencies below 30-40 Hz; 0.25"da to 3.52 grms. For items mounted in structural resonance areas input levels are increased over the response frequency band (to 10.5 - 19 grms) . 4 minutes per axis flat spectrum 0.2 g^2/Hz FAT: Sine: 5-2000 Hz 3 axes to 3.52 grms. Sweep 4 oct/min. Levels for resonance bands range from 5.3 to 10.5 grms. Randon: 11.7 grms, (0.07 g^2/Hz) 2 min/axis	None	None	30g for 5 minutes: Direction corresponds to vector from powered flight thrust	None	None	87F and 95% RH for 24 Hrs. Then reduce to 77F and stabilize.	None

TABLE 7-1 Continued COMPONENT ENVIRONMENTAL TESTS: SUMMARY OF EXPOSURES FOR QUALIFICATION AND FLIGHT ACCEPTANCE.

PROGRAM	FLIGHT AND MISSION ENVIRONMENTS, PRESSURE, THERMAL					OTHER		
	ASCENT SIMULATION (QPD)	CRUISE THERMAL VACUUM	ENTRY THERMAL	MARS SURFACE	CORONA, ARCING	SEAL, LEAKAGE	INTERNAL PRESSURE	EMC
VIKING LANDER	760 Torr to 117.2 Torr in 45 Sec. Pressure reduction must exceed 25 Torr/sec for a period of 10 Seconds (Ascent profile curve)	10^{-5} Torr $\sim$312 hrs. with 10 cyclings between temperature extremes based on component locations. High Range 125-160F; Low Range -50-0F. . Alternate dwells after stabilization of 1 hr. each for 5 high, 4 low. . 24 hr. dwell low . 280 hr. dwell high FAT: 10^{-5} Torr $\sim$72 hrs. with 10 cyclings . Alternate dwells 1 hr each, 5 Hi, 4 Lo . 8 hrs dwell Lo . 48 hr dwell Hi Temperature limits eased 25F	3 Torr $\sim$24 hrs total at 3 cycles each ending in a heat pulse and immersion in CO_2 at 15 Torr and -180F . 8 hr dwell at 3 Torr, nominal Temp. . Radiant heat pulse to 315F in 320 Sec. . CO_2 to 15 Torr in 2 minutes; Final temperature of gas is -180F FAT: 3 Torr $\sim$ 8 hrs with one cycle as above.	CO_2 at 15 Torr for 120 hrs total apply as 10 cycles with 6 hr dwells alternating high and low temperature Items internal to the lander cycle between -25F and 125F. External items range from -190F to 145F FAT: CO_2 at 15 Torr 60 hrs. total 5 cycles with 6 hr dwells. Temperature limits eased 25F.	Monitor for Discharges and Breakdowns. During test operations which occur over the critical pressure ranges, identify the most sensitive pressure condition and show sustained operation. Regimes for Investigation: Cruise, 10^{-5} Torr Entry 3 to 15 Torr Surface 15 Torr	Show Seal Integrity by an Appropriate technique. . Liquid Immersion: bubbles . Vacuum-Helium, in leak . Pressure-Helium, out leak . Pressure-Helium, to Vacuum, out leak . Radiflow, in leak FAT: As above.	PROOF AND BURST TESTS The ratios to nominal working levels are as follows: Item — Proof — Burst Accumulators, etc. 2.0 4.0 Actuators (Pneu.) 1.6 2.5 Lines, Fitting, etc. 2.0 max. - Main Propellant 1.25* - (No. Person.Hzrd.) 1.15* - *These are the safety factors for crack propagation using fracture mechanics design. FAT: Proof Pressures	Test for compliance with defined limits and measur thresholds for susceptibil Test methods as describe Mil-Std 461 and 462 cover Conducted Emittance Conducted Susceptance Radiated Emittance Radiated Susceptance
LUNAR ORBITER	760 Torr to 1 Torr in 90 sec. Follow an ascent profile curve with intermediate points: 620 Torr at 20 sec. 129 Torr at 50 sec. Max. rate of change, 20 Torr/Sec.	10^{-5} Torr. for $\sim$430 hrs. in 10 cycles 5 cycles: Dwell 40 hrs at 10F end with a 3.7 hr excursion to 110F and return. 5 cycles: Dwell 40 hrs at 110F end with a 3.7 hr excursion to 10F and return. FAT: 10^{-5} Torr $\sim$15.5 hrs in 2 cycles . Dwell 5 hrs at 35F end with a 2.7 hr excursion to 85F and return. . Dwell 5 hrs at 85F end with a 2.7 hr excursion to 35F and return.	NA	NA	None	LEAK Defined individually for each applicable item. FAT: As above.	PROOF PRESSURE 1.5 times nominal working levels BURST Rupture must occur at pressures in excess of 2.5 times working levels. FAT: Proof Pressure	Test for compliance with limits. Tests performed on indivi items or related groups. Measurements identify ou and establish minimum th for: Conducted Emittance Conducted Susceptance Radiated Emittance Radiated Susceptance
MARINER '69	Pump down at an average rate of 15 Torr per second. perform in conjunction with cruise thermal vacuum and FAT thermal vacuum.	10^{-5} Torr. for 312 hours total. Controlled temperatures 40°F for 24 hours followed by 167°F for 288 hours. Heat transfer by conduction (internal) or radiation (external items). Thermal shock (external items) 167°F to -50°F maximum rate attainable using radiant heat. FAT. 10^{-5} Torr. for 68 hours total 32°F for 8 hours followed by 131°F for 60 hours.	NA	NA	None	LEAK: Initial: Soap bubbles Final: Helium Mass Spectrometer Attitude control nozzle valves have a limit of 3 standard cc/hr. FAT: As above.	PROOF PRESSURE: Ambient: 85% of Bulk Yield Stress Cryo: 1.4 times operating pressure BURST: Minimum of 2.2 times operating pressure level Items cycles 3 times to 85% of bulk yield stress must burst within 4% of single cycle burst pressure. FAT: Proof pressure.	Test for compliance with limits (specification) Tests are performed on s items; measurements cov Conducted Emittance Conducted Susceptance Radiated Emittance Radiated Susceptance
BIO-SATELLITE	None	10^{-5} Torr for 72 hours or 30 days with temperature cycles to simulate orbits . 72 hrs for components having proven stable materials. . Orbit cycles have dwells of 30 min. and minimum change rates of 20F/hr. Temperature Ranges as follows: Body -30,+25F to 130, 160F Cabin 40F to 105F Extr'r -100F to 200F FAT: 10^{-5} Torr for 36 hours max. with temperature cycling as above.	None	Earth Entry with Air snatch or Water Impact No Surface Environment Simulation Tests	Operate Over a Pressure Cycle. 720 Torr to 1.5×10^{-1} Torr and return to 720 Torr . Monitor for corona or breakdown effects during the pressure change The pressure cycle is applied as the pump down and release from the Orbit Thermal Vacuum Test.	LEAK Hermetically sealed items only. Pressurize with helium or radioactive Kr. Monitor for out-leak in vacuum . He Mass Spectrograph . Radioactive counters (Ref. Mil-Std 202, Method 112 Procedure III) FAT: As above	PROOF PRESSURE 1.5 times nominal working level. BURST: Rupture must occur at pressures in excess of 4 times working levels FAT: Proof Pressure.	Test for compliance with d limits (specification) Tests are performed on se items and assembled sub-s Test Methods drawn from Mil-I-26600
NIMBUS B	None	10^{-5} Torr for 286 hrs with 7 cyclings The dwells and temperature sequence is as follows: (1) 24 hrs at 131F (5) 48 hrs at 32F (2) 24 hrs at 23F (6) 48 hrs at 122F (3) 48 hrs at 32F (7) to ambient (4) 48 hrs at 122F Transit times are 6 hrs to/from ambient 8 hrs for extremes and 2 hrs 23-32F. FAT: Same schedule as above with temperatures eased 9F.	NA	NA	None	LEAK Hermetically sealed units only . Fill with Helium . Place in Vacuum 10^{-5} Torr . Monitor for Helium out Leak FAT: As above.	PROOF PRESSURE 1.5 times nominal working level BURST Rupture must occur at pressures in excess of 4 times working levels FAT: Proof Pressure	System Test Only

L TESTS: SUMMARY OF EXPOSURES FOR QUALIFICATION AND FLIGHT ACCEPTANCE.

FLIGHT AND MISSION ENVIRONMENTS, PRESSURE, THERMAL				OTHER			
CRUISE THERMAL VACUUM	ENTRY THERMAL	MARS SURFACE	CORONA, ARCING	SEAL, LEAKAGE	INTERNAL PRESSURE	EMC	SPECIAL
5 Torr ~312 hrs. with 10 cyclings ween temperature extremes based component locations. High Range -160F; Low Range -50-0F. ternate dwells after stabilization 1 hr. each for 5 high, 4 low. hr. dwell low 0 hr. dwell high T: 10^{-5} Torr ~72 hrs. with 10 cyclings ternate dwells 1 hr each, 5 Hi, 4 Lo rs dwell Lo hr dwell Hi nperature limits eased 25F	3 Torr ~24 hrs total at 3 cycles each ending in a heat pulse and immersion in CO_2 at 15 Torr and -180F . 8 hr dwell at 3 Torr, nominal Temp. . Radiant heat pulse to 315F in 320 Sec. . CO_2 to 15 Torr in 2 minutes; Final temperature of gas is -180F FAT: 3 Torr ~ 8 hrs with one cycle as above.	CO_2 at 15 Torr for 120 hrs total apply as 10 cycles with 6 hr dwells alternating high and low temperature Items internal to the lander cycle between -25F and 125F. External items range from -190F to 145F FAT: CO_2 at 15 Torr 60 hrs. total 5 cycles with 6 hr dwells. Temperature limits eased 25F.	Monitor for Discharges and Breakdowns. During test operations which occur over the critical pressure ranges, identify the most sensitive pressure condition and show sustained operation. Regimes for Investigation: Cruise, 10^{-5} Torr Entry 3 to 15 Torr Surface 15 Torr	Show Seal Integrity by an Appropriate technique. . Liquid Immersion: bubbles . Vacuum-Helium, in leak . Pressure-Helium, out leak . Pressure-Helium, to Vacuum, out leak . Radiflow, in leak FAT: As above.	PROOF AND BURST TESTS The ratios to nominal working levels are as follows: Item — Proof — Burst Accumulators, etc. — 2.0 — 4.0 Actuators (Pneu.) — 1.6 — 2.5 Lines, Fitting, etc. — 2.0 max. — - Main Propellant (No. Person.Hzrd.) — 1.25* — - — 1.15* — - *These are the safety factors for crack propagation using fracture mechanics design. FAT: Proof Pressures	Test for compliance with defined limits and measure thresholds for susceptibility. Test methods as described in Mil-Std 461 and 462 covering: Conducted Emittance Conducted Susceptance Radiated Emittance Radiated Susceptance	MARS SAND AND DUST 6 hrs exposure to fine particles Velocity: 2500 + 500 ft min with 0.1-0.5 gram of solids in each cubic foot of air. Ambient Pressure: 30% RH. Solids are S_1O_2, 0.006" dia. or less. PROPELLANT COMPATIBILITY Hydrazine at 90F for 168 hrs. cover normally wetted surfaces with 100% Hydrazine.
5 Torr. for ~430 hrs. in 10 cycles ycles: Dwell 40 hrs at 10F end with .7 hr excursion to 110F and return. ycles: Dwell 40 hrs at 110F end with .7 hr excursion to 10F and return. T: 10^{-5} Torr ~15.5 hrs in 2 cycles vell 5 hrs at 35F end with a 2.7 hr ursion to 85F and return. vell 5 hrs at 85F end with a 2.7 hr ursion to 35F and return.	NA	NA	None	LEAK Defined individually for each applicable item. FAT: As above.	PROOF PRESSURE 1.5 times nominal working levels BURST Rupture must occur at pressures in excess of 2.5 times working levels. FAT: Proof Pressure	Test for compliance with defined limits. Tests performed on individual items or related groups. Measurements identify outputs and establish minimum thresholds for: Conducted Emittance Conducted Susceptance Radiated Emittance Radiated Susceptance	None
5 Torr. for 312 hours total. ntrolled temperatures F for 24 hours followed by °F for 288 hours. Heat transfer conduction (internal) or radiation ternal items). ermal shock (external items) °F to -50°F maximum rate ainable using radiant heat. T. 10^{-5} Torr. for 68 hours total F for 8 hours followed by °F for 60 hours.	NA	NA	None	LEAK: Initial: Soap bubbles Final: Helium Mass Spectrometer Attitude control nozzle valves have a limit of 3 standard cc/hr. FAT: As above.	PROOF PRESSURE: Ambient: 85% of Bulk Yield Stress Cryo: 1.4 times operating pressure BURST: Minimum of 2.2 times operating pressure level Items cycles 3 times to 85% of bulk yield stress must burst within 4% of single cycle burst pressure. FAT: Proof pressure.	Test for compliance with defined limits (specification) Tests are performed on selected items; measurements cover: Conducted Emittance Conducted Susceptance Radiated Emittance Radiated Susceptance	EXPLOSIVE ATMOSPHERE: Aviation gasoline - air mixture at 68 to 122F. Item must operate without causing ignition. Total of 9 cases with rich median and lean mixes each evaluated at 760 Torr, 523 Torr, and 141 Torr. ELECTRON RADIATION PROTON RADIATION Expose to fluxes and energy spectrums as predicted for each sensitive item.
5 Torr for 72 hours or 30 days with perature cycles to simulate orbits hrs for components having proven ble materials. bit cycles have dwells of 30 min. d minimum change rates of 20F/hr. mperature Ranges as follows: dy -30,+25F to 130, 160F bin 40F to 105F tr'r -100F to 200F : 10^{-5} Torr for 36 hours max. temperature cycling as above.	None	Earth Entry with Air snatch or Water Impact No Surface Environment Simulation Tests	Operate Over a Pressure Cycle. 720 Torr to 1.5 x 10^{-1} Torr and return to 720 Torr . Monitor for corona or breakdown effects during the pressure change The pressure cycle is applied as the pump down and release from the Orbit Thermal Vacuum Test.	LEAK Hermetically sealed items only. Pressurize with helium or radioactive Kr. Monitor for out-leak in vacuum . He Mass Spectrograph . Radioactive counters (Ref. Mil-Std 202, Method 112 Procedure III) FAT: As above	PROOF PRESSURE 1.5 times nominal working level. BURST: Rupture must occur at pressures in excess of 4 times working levels FAT: Proof Pressure.	Test for compliance with defined limits (specification) Tests are performed on selected items and assembled sub-systems Test Methods drawn from Mil-I-26600	FUNGUS 28 day exposure to 86F at 95% RH Component is sprayed with live spores together with a spore charged nutrient.
Torr for 286 hrs with 7 cyclings dwells and temperature sequence s follows: 4 hrs at 131F (5) 48 hrs at 32F 4 hrs at 23F (6) 48 hrs at 122F 8 hrs at 32F (7) to ambient 8 hrs at 122F nsit times are 6 hrs to/from ambient s for extremes and 2 hrs 23-32F. : Same schedule as above with peratures eased 9F.	NA	NA	None	LEAK Hermetically sealed units only . Fill with Helium . Place in Vacuum 10^{-5} Torr . Monitor for Helium out Leak FAT: As above.	PROOF PRESSURE 1.5 times nominal working level BURST Rupture must occur at pressures in excess of 4 times working levels FAT: Proof Pressure	System Test Only	None

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TABLE 7-2 SYSTEM ENVIRONMENTAL TESTS, SUMMARY OF EXPOSURES FOR QUALIFICATION AND FLIGHT ACCEPTANCE

PROGRAM	DYNAMIC ENVIRONMENTS				GROUND ENVIRONMENTS, THERMAL		OPERATION, LIFE
	VIBRATION	ACOUSTICS	SHOCK	STATIC LOAD	THERMAL CYCLING	TEMPERATURE HUMIDITY	
VIKING LANDER	Sine: 5-200 Hz 3 axes 1.06 grms Nominal . 0.29 in. da 5-10Hz, 1.06 grms to 125Hz then 0.0018 in. da to 200 Hz, (2.69 grms) . Sweep 2 oct/min up and down FAT. As above, to 0.707 grms Nom. . 0.193 da to 10Hz, 0.0012 da. to 200Hz . Sweep 4 oct min up and down The above levels are defined for the Viking Spacecraft assembly. These levels also apply to the Viking Lander tested separately.	3 configurations as follows: Ovr'll SPL db / Time sec. Launch: (S/C Assy.) Q 149 / 300; FAT 143 / 60 Entry: (Lander w/ A. Shell, B. Covr.) Q 144 / 150; FAT 138 / 30 Term. Des: Lander: Q 123 / 100; FAT 121 / 20	Pyro Shock: Fire all items in Mission Sequence (~60 devices) Landing Shock: 3 drops at 24 ft/sec. terminal velocity (one drop each leg) Lander at 14° to horizontal surface at 10° to horizontal Impact angle 24°	Static load to design ultimate, no failure. Static test model. Specific Areas: . Base cover, pressure . Aeroshell, pressure, temp. . Bioshield, pressure, temp. . Lander Legs, . Trusses-Adapters; launch loads	275°F for 200 hrs. total Atmosphere dry N_2 Apply as 5 cycles of 40 hrs. each. FAT: (Dry N_2 Atmosphere) Decontamination; 256F for 30 hours one cycle Terminal Sterilization: 256F for 12 hours one cycle Recycle limit, 3 times for flight equipment.	NONE	ALTERNATE MODE OPERATION . Verify Performance and Operation at extremes of voltage and signal levels. Establish margins for operation. . Exercise and investigate redundant mode switching and alternate path operations. Establish levels, capabilities for lesser mode operations
LUNAR ORBITER	Sine: 7-400 Hz, 3 axes, to 2.6 grms. Max. Random: 20-2000 Hz, 3 axes, 20.4 grms ovr'll Sweep 1 oct/min. Shaped input. Thrust: 0.85 grms to 2.11 grams with 1.41 over 60-200 Hz Lat. 0.35 to 2.26 grms with 0.64 at 90 Hz Inputs are notched at first system resonances Random. Exposure same as components Torsion: 20-150 Hz at 4 oct/min 8.6 rad/sec^2 to 60 Hz, 17.2 rad/sec^2 to 150 Hz. FAT: Sine: 7-400, 3 axis, 4 oct/min as above, levels eased 33% Random: 20-2000 Hz, 3 axes, 17.2 grms overall, Exposure same as component	140 db overall, 2 minutes Spectrum shaped as follows with octave band control Center Freq. Hz / Band level db 37.5 / 118 75 / 127 150 / 133 300 / 136 600 / 135 1200 / 133 2400 / 131 4800 / 129	Pyro device operation: a. Explosive bolts, V-band release for spacecraft separation. b. Explosive Releases, for solar panels and antennas.	Static load simulation of powered flight and critical reaction conditions. Tests to yield on Structural Model.	NONE	NONE	MISSION SIMULATION 10^{-5} Torr for 576 Hours as in Cruise Thermal Vacuum Exposure. Cruise Mode: 90 hrs. sinks at 20 -65F and 10F. Orbit simulation ~140 cycles, sink temperatures as in FAT . Test completed before first launch.
MARINER '69	Sine: 5-2000 Hz, 3 axes, to 13.5 grms. Max. Random: 20-2000 Hz, 3 axes, 7.0 grms. ovr'll. Sweep 1 oct/min. Thrust axis 0.75 grms. 5-120 Hz roll to 13.5 g, 500-2000 Hz. Laterals 0.5 grms, 5-50 Hz, 0.75 grms, 50-250 Hz. Random. 60 sec, axis shaped spectrum 0.0225 g^2/Hz, 100-255 Hz, 0.063 g^2/Hz 350-750 Hz, Roll; up 6db/oct; off, 12db/oct. Torsion: Sweep 1 min/oct; 2.25 rad/sec^2 at 5 Hz step to 7.5 at 20 Hz and 15.0 at 50-150Hz FAT: Sine: As above, eased 30%, 3 oct/min Random: Thrust 20 sec, eased to 4.2 grms. Lateral, 20 sec each eased to 2.8 grms.	148 db overall for 60 seconds. Spectrum same as for sub-assemblies 50 to 10,000 Hz Peak Range 100-200 Hz 136 db 252-400 Hz 141 db Roll off to 10,000 Hz 100 db	Pyro Firings of Installed Devices: 1. V Band, Booster Separation 2. Pin Pullers, Solar Paddles 3. Valves, Propulsion System 4. Release Mech., Scan Platform 5. Valve, IRS enable. FAT. V band, Booster Separation	Structural Model Employed for Static Determinations in Specific Critical Areas.	NONE	NONE	No extra requirement
BIOSATELLITE	Sine: 10-2000 Hz, 3 axes to 5.3 grms. Max. Random: 20-2000 Hz, 3 axes to 12.3 grms ovr'll . Sweep: 2 oct/min. Thrust, 1.59 grms to 150 Hz ramp to 5.3 grms at 500 Hz. Lat'l. 1.06 grms to 250 Hz ramp to 5.3 grms at 400 Hz. Inputs notched for system resonances. . 4 min. axis shaped spectrum. 0.0225 g^2/Hz 20-150 Hz; Roll up 4 db/oct to 0.09 g^2/Hz 425-1200 Hz; Roll off 2 db/oct. FAT. Sine 10-2000 Hz, 3 axes, 4 oct/min. Input as above, eased 33%, 3.5 grms. max. Random. 2 min/axis 8.2 grms. overall spectrum as above eased 33%	145 db overall. for 30 sec. spectrum as defined in Mil Std. 810, Method 515, Fig. 1 i.e. Peak, 150-1200 Hz at 141 db Roll offs to 125 db at 50 Hz and 500 Hz	Pyro Separation Device: 5 Firings at 20 Torr and ambient temperature. Double pendulum suspension within an altitude chamber . Spacecraft . Interface plus a mass simulator for the delta booster	Static evaluation at critical areas on structural model equipment. . Booster adapters, proof load . Deorbit Thrust cone, proof load. . Tank and Fuel cell support members, yield.	NONE	50-100F at 95% RH 72 hours total, cycled; 3 cycles of 24 hrs. each 2 hrs. raise from 50F to 100F at 95% RH 6 hrs. Dwell 100°F, 95% RH 16 hrs. Reduce 100F to 50F RH above 95%	MISSION SIMULATION 30 DAY Continuous Ambient Operation for 25 consecutive days. . All systems operate to flight schedules . Live Primate
NIMBUS B (B1 and B2)	FAT. Sine 5-2000 Hz, 3 axes to 3.52 grms. Random. 20-2000 Hz, 3 axes, 11.7 grms ovr'll . Sweep 2 oct/min, Thrust 0.5" da to 3.52 grms. Laterial 0.5" da to 0.707 grms. . 2 min/axis. Flat spectrum random at 0.07 g^2/Hz	NONE	Pyro Separation device: FAT: V band separation bolts, 4 firings. One firing performed after Vibration exposure.	NONE	NONE	NONE	No extra requirement

TABLE 7-2 Continued SYSTEM ENVIRONMENTAL TESTS, SUMMARY OF EXPOSURES FOR QUALIFICATION AND FLIGHT ACCEPTANCE

PROGRAM	FLIGHT AND MISSION ENVIRONMENTS, PRESSURE, THERMAL				OTHER			
	ASCENT SIMULATION (QPD)	CRUISE THERMAL VACUUM	ENTRY THERMAL	MARS SURFACE	SEAL, LEAKAGE	INTERNAL PRESSURE	EMC	SPECIAL
VIKING LANDER	NONE	10^{-5} Torr. 384 hrs. total. Solar simulation for near planet, equilibrium with -320°F Cryowall. . Near Earth. 120 hrs. dwell Solar Flux. 486 BTU/hr Ft^2 . Near Mars. 72 hrs dwell Solar Flux. 145 BTU/hr Ft^2 Flux incidence angle as in cruise. Apply above schedule 2 times FAT. 10^{-5} Torr. 192 hrs. total as above, apply schedule one time.	10^{-5} Torr 192 hours total. Solar simulation for near Mars. Equilibrium with -320 °F Cryowall. Operate for 96 hr. dwell. Solar Flux 185 BTU/hr. Apply to base cover for entry configuration. FAT. 10^{-5} Torr 96 hrs as above.	Simulate 3 Surface conditions Hot. 32 Torr, 120 hrs. CO_2 Atmosphr. Cold. 1.5 Torr, 120 hrs N_2+A Atmosphr. Nom. 7.6 Torr 288 hrs. CO_2+N_2+A. Apply as 24 hr day temp. simulation. Dawn Noon Sunset Midnight Not 375R 575R 500R 390R Cold 310R 475R 450R 330R Nom. ~340R ~525R ~475R ~360R (Temperatures of Ground Simulator) FAT. 120 hrs. nominal conditions as above.	FAT . Bioshield Separation Joint: Bio integrity. . Tankage and lines: System Integrity.	Bioshield: Internal pressure to design ultimate, No Failures. FAT. Proof Pressure. Test assemblies to levels established for components (Fracture Mechanics)	Demonstrate 6 db margin Test techniques, critical areas drawn from results of development tests.	CAPTIVE FIRINGS Exercise all reaction co[illegible] Attitude, De-orbit, Term[illegible] . Open Loop Firings; Con[illegible] cold flow laboratory . Closed loop firings; Cor[illegible] Hybrid simulation labor[illegible] DESCENT PERFORMAN[illegible] Make simulated Mars lar[illegible] on-board guidance and er[illegible] LLRF provides gravity s[illegible]
LUNAR ORBITER	760 Torr to 94 Torr in 75 seconds following an ascent pressure profile schedule. i.e. 630 Torr at 20 sec. 470 Torr at 30 sec. 310 Torr at 40 sec. 207 Torr at 50 sec. Maximum rate of change is 16 Torr/sec.	10^{-5} Torr for 117 hours total. Temp. from -290°F cryowall and local heat sinks. Cruise: 54 hrs. sinks at 64F, 35F. Hot Orbit: 31.5 hrs. as 9 cycles Cold Orbit: 31.5 hrs. as 9 cycles FAT: 58.5 hrs. including 9 cycles Heat Sink Temperatures Orbit / 1 Hr. / 0.5 Hrs. / 2 Hrs. Hot / -290°F / 90, 140F / 65, 35F Cold / -290°F / 65, 170F / 40, -5F FAT / -290°F / 85, 145F / 60, 15F	NA	NA	System Seal and Leak Integrity. . Velocity Reaction Control . Attitude Control . Photo Pressure Shell	Proof Pressure all units to 1.5 times working levels. . Velocity Reaction Control . Attitude Control	Demonstrate 6 db Margin by: . Comparison of System EM signatures with known component susceptabilities. . Illumination by RF. Relative to 10^{-6} Volt/Meter narrow band reference. Show 6 db margin above 140 db incidence . Operate with illumination of 153 db	NONE
MARINER '69	760 Torr to 152 Torr in 65 seconds or less. Maximum rate of change is 108-124 Torr/sec attained in less than 10 sec.	10^{-5} Torr 350 hours with shroud temperature -290°F or less, solar simulation within 3° of roll axis. Steady State 250 Hours Solar Simulation 65 watt/ft^2 to 143 watt/ft^2. Temperatures in accord with subsystem testing. FAT: 10^{-5} Torr 200 hours minimum as above, 150-200 hours steady. 50 to 100 hours solar, 65 to 130 watt/ft^2	NA	NA	System Integrity Helium leak detector. All spacecraft.	System Integrity. Proof pressure. All tanks and lines. All spacecraft.	System Environment Simulation . RF: Atlas Centaur, AFETR radiations. from 2 min. before lift to 4 min post Seprn. . Transients: Umbilical and IFD actuation; . S Band: Self generated noise; . Signature: Measure spacecraft. RF over range 30 Hz to 10 GHz in simulated near encounter mode. FAT. As above for RF and S Band	OPERATING MARGIN EVA[illegible] Extended operation at extre[illegible] signal levels, etc. to establ[illegible] margins FAT: Inspection; Calibratio[illegible] At the completion of enviro[illegible] the spacecraft is disassemb[illegible] reassembled, final calibrat[illegible] ally operated.
BIO SATELLITE	NONE	10^{-5} Torr for 72 hours total. Orbit temperature phases simulated. Control by selective heating within a 16 zone thermal shroud; Scheduled: 8 hrs. ambient to Phase I temp. distribution. 24 hrs. dwell, Temp. Range -144 to 85F 8 hrs. transition to Phase II distr. 26 hrs. dwell, temperatures -16 to 103F 6 hrs. return to ambient FAT. Same Test.	NONE	NONE	System Integrity Leakage: by Appropriate Means (See Pressurize)	System Integrity Proof Pressure to 1.5 times working level, monitor leak rate . Attitude Control Gas . Cabin Air . De-Orbit Attitude Control Gas . Fluid Lines, Coolants . Cryo Fuels. Liquid Hydrogen, Oxygen	Demonstrate 6 db Margin: . Measure EM signatures conducted and radiated. . Show compliance with limits imposed. . Inject 1 volt RMS on paternal bus at frequencies defined, maintain stable operation. . Illuminate at frequencies defined with stable operation.	MAGNETIC FIELD EFFEC[illegible] (Magnetometer Attitude Co[illegible] Measure and Show: . Spacecraft has static mag[illegible] less than 2.5 milligauss . Spacecraft generated tran[illegible] than 25 milligauss and 0.1[illegible] . Exposure to a 5 gauss fiel[illegible] induce residuals.
NIMBUS B	NONE	FAT, 10^{-5} Torr, 286 hrs, 7 cyclings the dwells and temperature sequence are as follows 1. 24 hrs. at 122F 5. 48 hrs. at 416 2. 24 hrs. at 32F 6. 48 hrs. at 113F 3. 48 hrs. at 41F 7. To ambient 4. 48 hrs. at 113F Transit times are 6 hrs. to/from ambient. 8 hrs for extremes (i.e. 122 to 32, 41 to 113, etc.) and 2 hrs. for 32 to 41F. Note. Test is same as component FAT.	NA	NA	System Integrity . All lines, Tankage (Attitude Control)	System Integrity Proof Pressure to 1.5 Working Level . Tanks . Lines	Demonstrate 6 db Margin: . Measure ambient EM signature on main bus. . Inject noise 6 db above measured level, show operation. . Illuminate with range RF levels established for 6 db margin. Show operation.	NONE

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2 Continued SYSTEM ENVIRONMENTAL TESTS, SUMMARY OF EXPOSURES FOR QUALIFICATION AND FLIGHT ACCEPTANCE

FLIGHT AND MISSION ENVIRONMENTS, PRESSURE, THERMAL				OTHER			
ASCENT SIMULATION (QPD)	CRUISE THERMAL VACUUM	ENTRY THERMAL	MARS SURFACE	SEAL, LEAKAGE	INTERNAL PRESSURE	EMC	SPECIAL
NONE	10^{-5} Torr. 384 hrs. total. Solar simulation for near planet, equilibrium with -320°F Cryowall. . Near Earth. 120 hrs. dwell Solar Flux. 486 BTU/hr Ft^2 . Near Mars. 72 hrs dwell Solar Flux. 145 BTU/hr Ft^2 Flux incidence angle as in cruise. Apply above schedule 2 times FAT. 10^{-5} Torr. 192 hrs. total as above, apply schedule one time.	10^{-5} Torr 192 hours total. Solar simulation for near Mars. Equilibrium with -320°F Cryowall. Operate for 96 hr. dwell. Solar Flux 185 BTU/hr. Apply to base cover for entry configuration. FAT. 10^{-5} Torr 96 hrs as above.	Simulate 3 Surface conditions Hot. 32 Torr, 120 hrs. CO_2 Atmosphr. Cold. 1.5 Torr, 120 hrs N_2+A Atmosphr. Nom. 7.6 Torr 288 hrs. CO_2+N_2+A. Apply as 24 hr day temp. simulation. Dawn Noon Sunset Midnight Not 375R 575R 500R 390R Cold 310R 475R 450R 330R Nom. ~340R ~525R ~475R ~360R (Temperatures of Ground Simulator) FAT. 120 hrs. nominal conditions as above.	FAT . Bioshield Separation Joint: Bio integrity. . Tankage and lines: System Integrity.	Bioshield: Internal pressure to design ultimate, No Failures. FAT. Proof Pressure. Test assemblies to levels established for components (Fracture Mechanics)	Demonstrate 6 db margin Test techniques, critical areas drawn from results of development tests.	CAPTIVE FIRINGS Exercise all reaction controls for: Attitude, De-orbit, Terminal Descent. . Open Loop Firings; Control from cold flow laboratory . Closed loop firings; Control from Hybrid simulation laboratory. DESCENT PERFORMANCE TESTS Make simulated Mars landing using on-board guidance and engines LLRF provides gravity simulation
760 Torr to 94 Torr in 75 seconds following an ascent pressure profile schedule. i.e. 630 Torr at 20 sec. 470 Torr at 30 sec. 310 Torr at 40 sec. 207 Torr at 50 sec. Maximum rate of change is 16 Torr/sec.	10^{-5} Torr for 117 hours total. Temp. from -290°F cryowall and local heat sinks. Cruise: 54 hrs. sinks at 64F, 35F. Hot Orbit: 31.5 hrs. as 9 cycles Cold Orbit: 31.5 hrs. as 9 cycles FAT: 58.5 hrs. including 9 cycles Heat Sink Temperatures Orbit 1 Hr. 0.5 Hrs. 2 Hrs. Hot -290°F 90,140F 65,35F Cold -290°F 65,170F 40,-5F FAT -290°F 85,145F 60,15F	NA	NA	System Seal and Leak Integrity. . Velocity Reaction Control . Attitude Control . Photo Pressure Shell	Proof Pressure all units to 1.5 times working levels. . Velocity Reaction Control . Attitude Control	Demonstrate 6 db Margin by: . Comparison of System EM signatures with known component susceptabilities. . Illumination by RF. Relative to 10^{-6} Volt/Meter narrow band reference. Show 6 db margin above 140 db incidence . Operate with illumination of 153 db	NONE
760 Torr to 152 Torr in 55 seconds or less. Maximum rate of change is 108-124 Torr/sec attained in less than 10 sec.	10^{-5} Torr 350 hours with shroud temperature -290°F or less, solar simulation within 3° of roll axis. Steady State 250 Hours Solar Simulation 65 watt/ft^2 to 143 watt/ft^2. Temperatures in accord with subsystem testing. FAT: 10^{-5} Torr 200 hours minimum as above, 150-200 hours steady. 50 to 100 hours solar, 65 to 130 watt/ft^2	NA	NA	System Integrity Helium leak detector. All spacecraft.	System Integrity. Proof pressure. All tanks and lines. All spacecraft.	*System Environment Simulation* . RF:Atlas Centaur, AFETR radiations. from 2 min. before lift to 4 min post Seprn. . Transients: Umbilical and IFD actuation; . S Band: Self generated noise; . Signature: Measure spacecraft. RF over range 30 Hz to 10 GHz in simulated near encounter mode. FAT. As above for RF and S Band	OPERATING MARGIN EVALUATION Extended operation at extremes of voltages, signal levels, etc. to establish functional margins FAT: Inspection; Calibration, Retest At the completion of environmental exposures the spacecraft is disassembled for inspection reassembled, final calibrated and functionally operated.
NONE	10^{-5} Torr for 72 hours total. Orbit temperature phases simulated. Control by selective heating within a 16 zone thermal shroud; Scheduled: 8 hrs. ambient to Phase I temp. distribution. 24 hrs. dwell, Temp. Range -144 to 85F 8 hrs. transition to Phase II distr. 26 hrs. dwell, temperatures -16 to 103F 6 hrs. return to ambient FAT. Same Test.	NONE	NONE	System Integrity Leakage;by Appropriate Means (See Pressurize)	System Integrity Proof Pressure to 1.5 times working level, monitor leak rate . Attitude Control Gas . Cabin Air . De-Orbit Attitude Control Gas . Fluid Lines, Coolants . Cryo Fuels, Liquid Hydrogen, Oxygen	Demonstrate 6 db Margin: . Measure EM signatures conducted and radiated. . Show compliance with limits imposed. . Inject 1 volt RMS on paternal bus at frequencies defined, maintain stable operation. . Illuminate at frequencies defined with stable operation.	MAGNETIC FIELD EFFECTS (Magnetometer Attitude Control) Measure and Show: . Spacecraft has static magnetic field less than 2.5 milligauss . Spacecraft generated transients are less than 25 milligauss and 0.1 sec. . Exposure to a 5 gauss field does not induce residuals.
NONE	FAT, 10^{-5} Torr, 286 hrs, 7 cyclings the dwells and temperature sequence are as follows 1. 24 hrs. at 122F 5. 48 hrs. at 416 2. 24 hrs. at 32F 6. 48 hrs. at 113F 3. 48 hrs. at 41F 7. To ambient 4. 48 hrs. at 113F Transit times are 6 hrs. to/from ambient. 8 hrs for extremes (i.e. 122 to 32, 41 to 113, etc.) and 2 hrs. for 32 to 41F. Note. Test is same as component FAT.	NA	NA	System Integrity . All lines, Tankage (Attitude Control)	System Integrity Proof Pressure to 1.5 Working Level . Tanks . Lines	Demonstrate 6 db Margin: . Measure ambient EM signature on main bus. . Inject noise 6 db above measured level, show operation. . Illuminate with range RF levels established for 6 db margin. Show operation.	NONE

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TABLE 7-3 - SUMMARY OF TEST PROGRAM EVALUATION CRITERIA

CRITERIA	EXAMPLE, EFFECT ON PROJECT, COMMENT
1. Early Identification of Operating Problems	
a. Early testing of flight type components includes both operation and environmental exposures	Example - Lunar Orbiter reliability demonstration testing which consisted of extended operation in thermal vacuum. The data from Section 4 (Figures 4-1 thru 4-7) shows that ambient operation, followed by vibration and thermal vacuum, are the most effective environments for uncovering problems.
b. Subsystem breadboards using flight type components.	Accomplished on Biosatellite and Lunar Orbiter system design verification test. These tests provided early identification of subsystem interface conditions and problems.
c. Early operation of a spacecraft	Examples are Mariner PTM with prototype subsystems and Lunar Orbiter Spacecraft C. Mariner provided insight to system sensitivity toward EMI effects; Spacecraft C identified problem areas with DSN and multi-vehicle operations.
d. Early measurement of spacecraft dynamic responses	Lunar Orbiter completed a vibration survey test before component designs were completed. Data from those test verified design requirements for components and allowed application of vibration test levels based upon local structural responses.
e. Early definition of science interfaces	Accomplished on Nimbus B - Project proceeded without a prototype or qualification spacecraft.
2. Establishment of Operating Margins	
a. Extended operations of components under environmental conditions	Extended operation tests were performed on Lunar Orbiter (RDT), Mariner, and Biosatellite. Data enhanced operating capabilities.
b. Extended mode and alternate mode operation of the spacecraft	Test included mission simulations for both Lunar Orbiter and Biosatellite plus alternate mode operation for Mariner. Value of these tests shown in a flight example from Mariner. Mariner VII was reacquired and brought through a successful encounter after a major anomaly which tumbled the spacecraft and disabled a battery.

TABLE 7-3 - SUMMARY OF TEST PROGRAM EVALUATION CRITERIA (Continued)

CRITERIA	EXAMPLE, EFFECT ON PROJECT, COMMENT
c. EMI Margin Demonstrations	Tests included in all spacecraft. Provided operating margin against difficult conditions for both mariner and Biosatellite. EM problems were not eliminated, however, adequate work-around had been prepared.
3. Test Problems	
a. Late delivery of science equipment	For Lunar Orbiter late delivery of the photo systems did not allow complete interface testing. Data loss in flight could be traced to incomplete interface testing. Life testing of Mariner subsystems was not completed.
b. Changing science interfaces and requirements	The changes in the primate interface requirements contributed to flight incidents with thermal control and the psychomotor game. Biosatellite.
c. AGE not consistent with spacecraft in shielding and transient suppression.	AGE effects mask potential flight problems shown in Mariner and Biosatellite. AGE is a source of EMI noise and transients operation difficulties (see Table 5-5-1).
d. Part and manufacturing controls not maintained in subcontracted equipment	Lunar Orbiter photo system shows incidents of part or manufacturing control deficiency. Difficulty in control of manufacturing film and film processor items resulted in degradation and loss of flight data.
e. Requirements for complex part items (integrated circuits) reach the limits of supplier capabilities	Mariner integrated circuits - minimum of 85 changes recorded in ground testing with changes continuing into launch site preparations
4. Environmental Exposures	
a. Vibration. Sine and Random exitations have been applied to both spacecraft and components during both acceptance and qualification.	Data from Section 4 (See Figure 4-8) shows vibration testing an effective means for uncovering problems. The data from the study shows no evidence which justifies eliminating either mode of exitation in any of the testing.

TABLE 7-3 - SUMMARY OF TEST PROGRAM EVALUATION CRITERIA (Conclusion)

CRITERIA	EXAMPLE, EFFECT ON PROJECT, COMMENT
b. Thermal Vacuum. Tests include thermal cycling and solar simulation. Time of exposure in balance with duration of the mission.	Data from Section 4 shows thermal vacuum an effective means for uncovering problems. Data from flight shows minor incidents of undetected sensitivities.
c. Corona and Arcing. Tests for corona and arcing sensitivity are incorporated into other exposures such as pumpdown, thermal vacuum.	Testing has been effective. Data from Section 5 (Tables 5-3 and 5.5-4) have shown that ground tests will uncover sensitivities toward voltage breakdown. Persistence into flight can be avoided.
d. Acoustics. Testing has been applied as qualification only.	Data from Section 4 shows only a small contribution from acoustic testing toward uncovering problems. Acoustic exposure may be overshadowed by vibration tests.
e. Shocks. Pyrotechnic shocks have been applied for both qualification and acceptance.	Data from Section 4 shows only a small contribution due to shock tests. Flight records show evidence that pyro shocks induce responses which can upset star sensors (dust particles, or movements of light reflection).

TABLE 7-4 - SUMMARY OF FLIGHT ACCEPTANCE TEST EXPOSURES

FLIGHT ACCEPTANCE ENVIRONMENT	LEVELS APPLIED FOR EACH PROJECT			
	LUNAR ORBITER	MARINER '69	BIOSATELLITE	NIMBUS
Vibration 15-2000 Hz.				
Component Sine	7 grms	9 grms	5.6 grms	3.5 grms
Component Random	17.2 grms	10.8 grms	None	11.7 grms
Spacecraft Sine	1.74 grms(max.) to 400 Hz only	9.0 grms(max)	3.5 grms (max.)	3.5 grms
Spacecraft Random	17.2 grms	4.2 grms	8.2 grms	11.7 grms
Thermal Vacuum (10^{-5} Torr)				
Component	15 Hrs. 2 cycles	68 Hrs. Solar Simul'n.	36 Hrs. 18 cycles	268 Hrs. 7 cycles
Spacecraft	58 Hrs. 9 cycles	200 Hrs. Solar Simul'n	72 Hrs. 2 cycles	268 Hrs. 7 cycles
Other	None	Pyro Shock Separation	None	Pyro Shock Separation

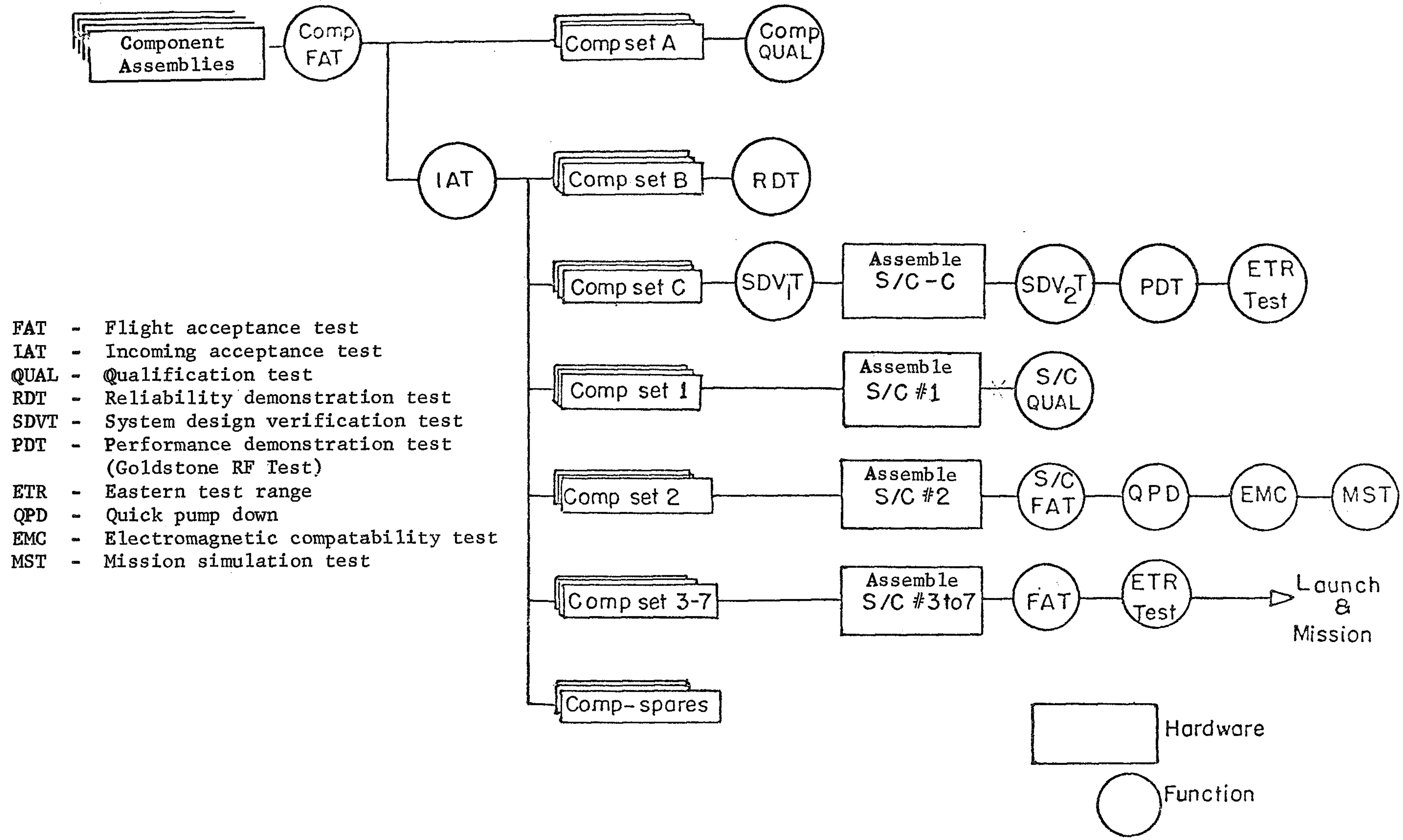
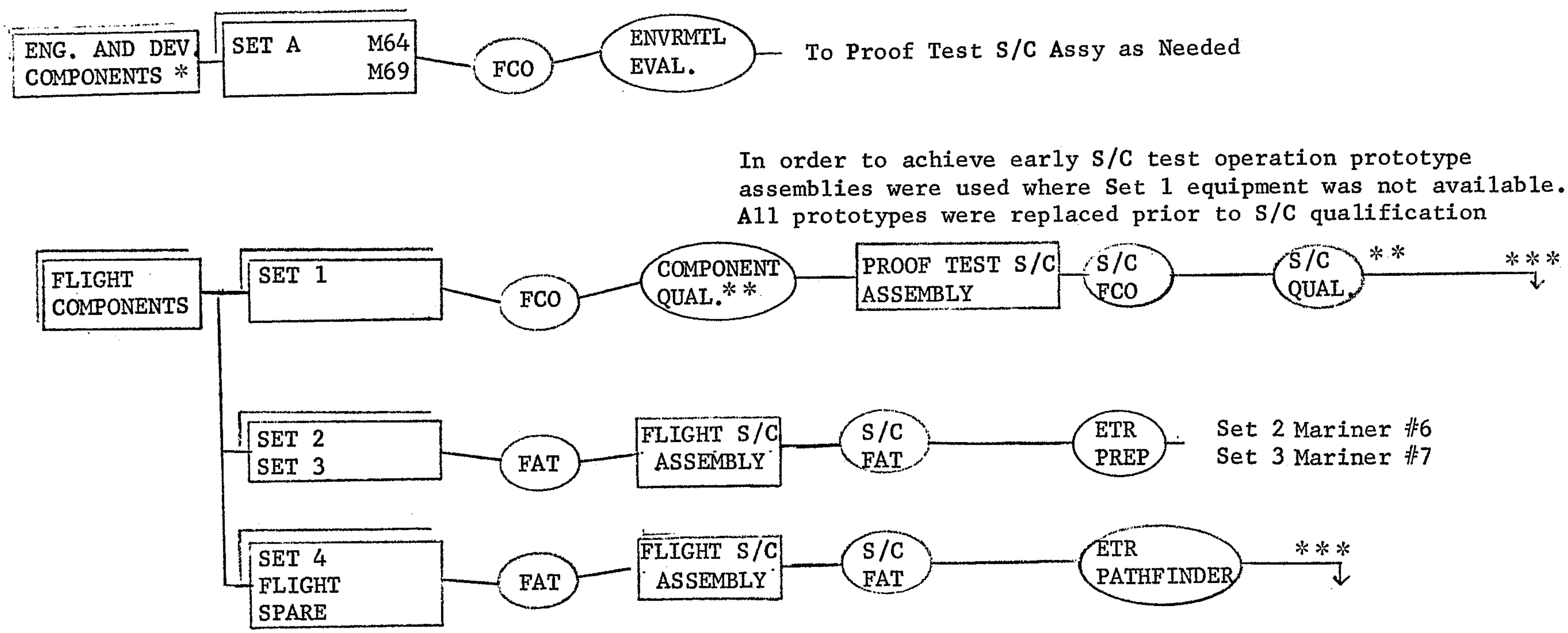


FIGURE 7-1 BLOCK DIAGRAM OF HARDWARE UTILIZATION, LUNAR ORBITER

FIGURE 7-2 BLOCK DIAGRAM OF HARDWARE UTILIZATION, MARINER 69



FCO Functional Checkout
FAT Flight Acceptance Test
S/C Spacecraft

* JPL TERM IS SUBSYSTEM ASSEMBLIES
** JPL TERM IS TYPE APPROVAL
*** THESE ITEMS WILL BE REWORKED INTO THE MARINER '71 CONFIGURATION

HARDWARE (box)
FUNCTION (oval)

FIGURE 7-3 BLOCK DIAGRAM OF HARDWARE UTILIZATION, BIOSATELLITE

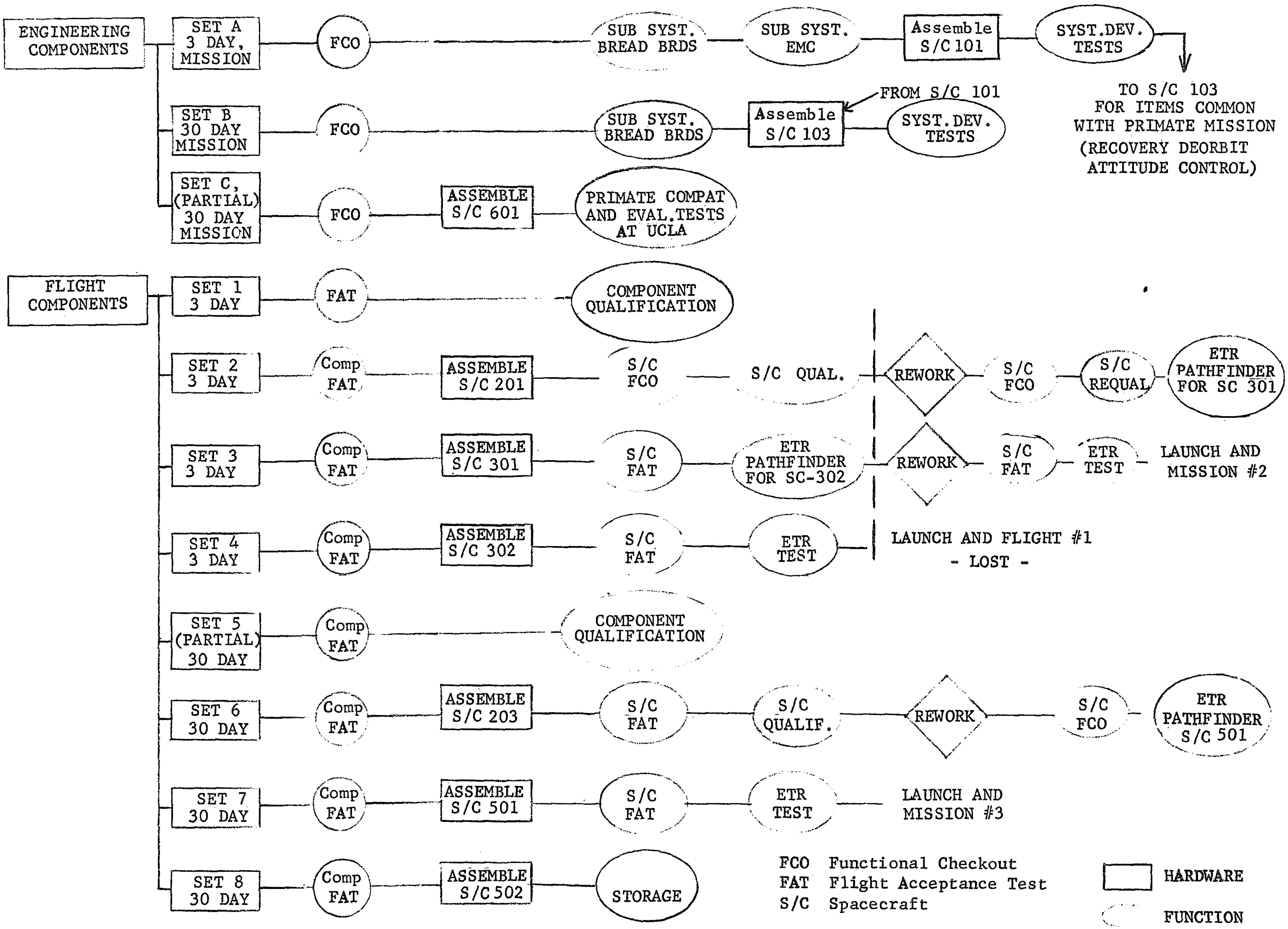
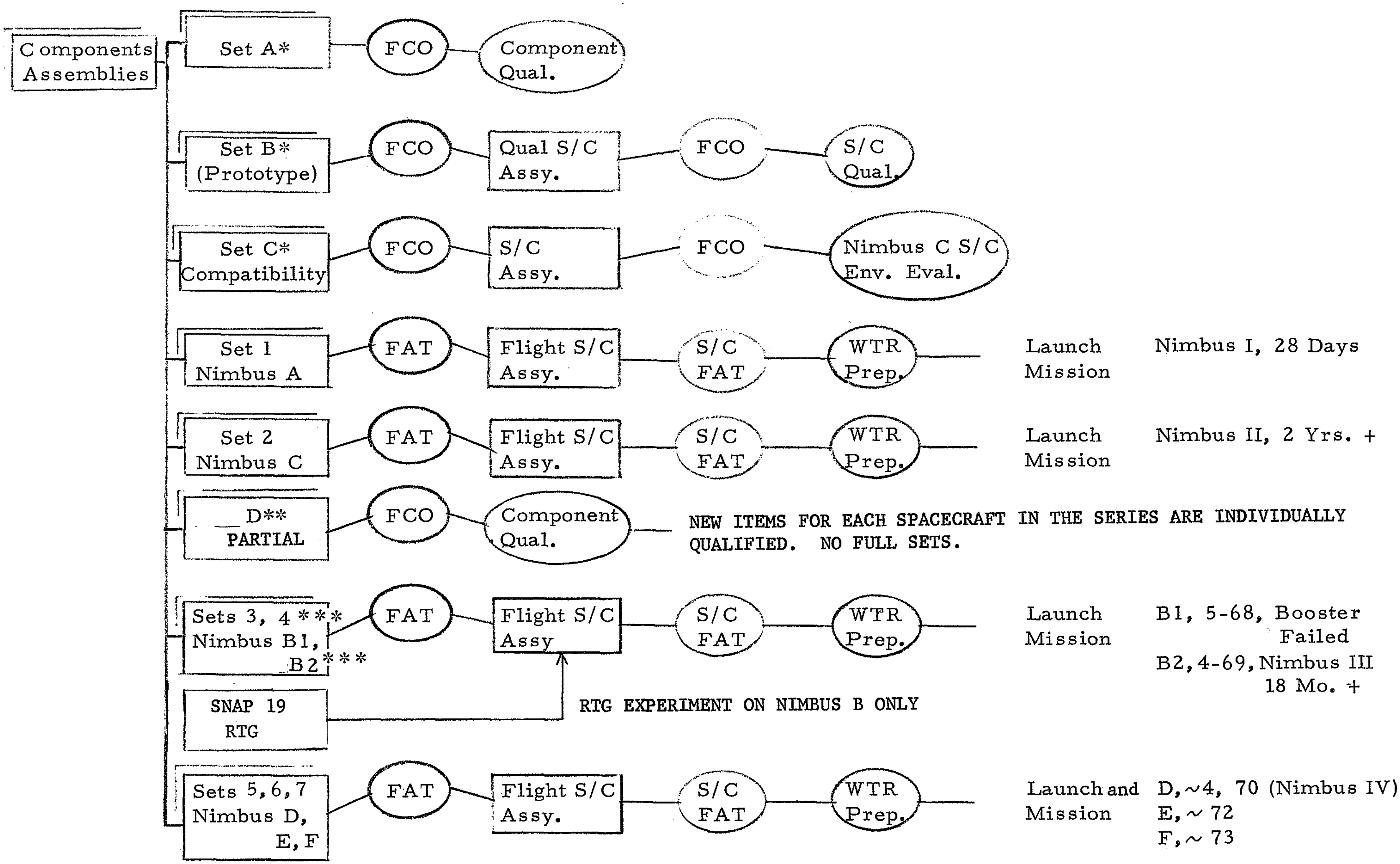


FIGURE 7-4 BLOCK DIAGRAM OF HARDWARE UTILIZATION, NIMBUS



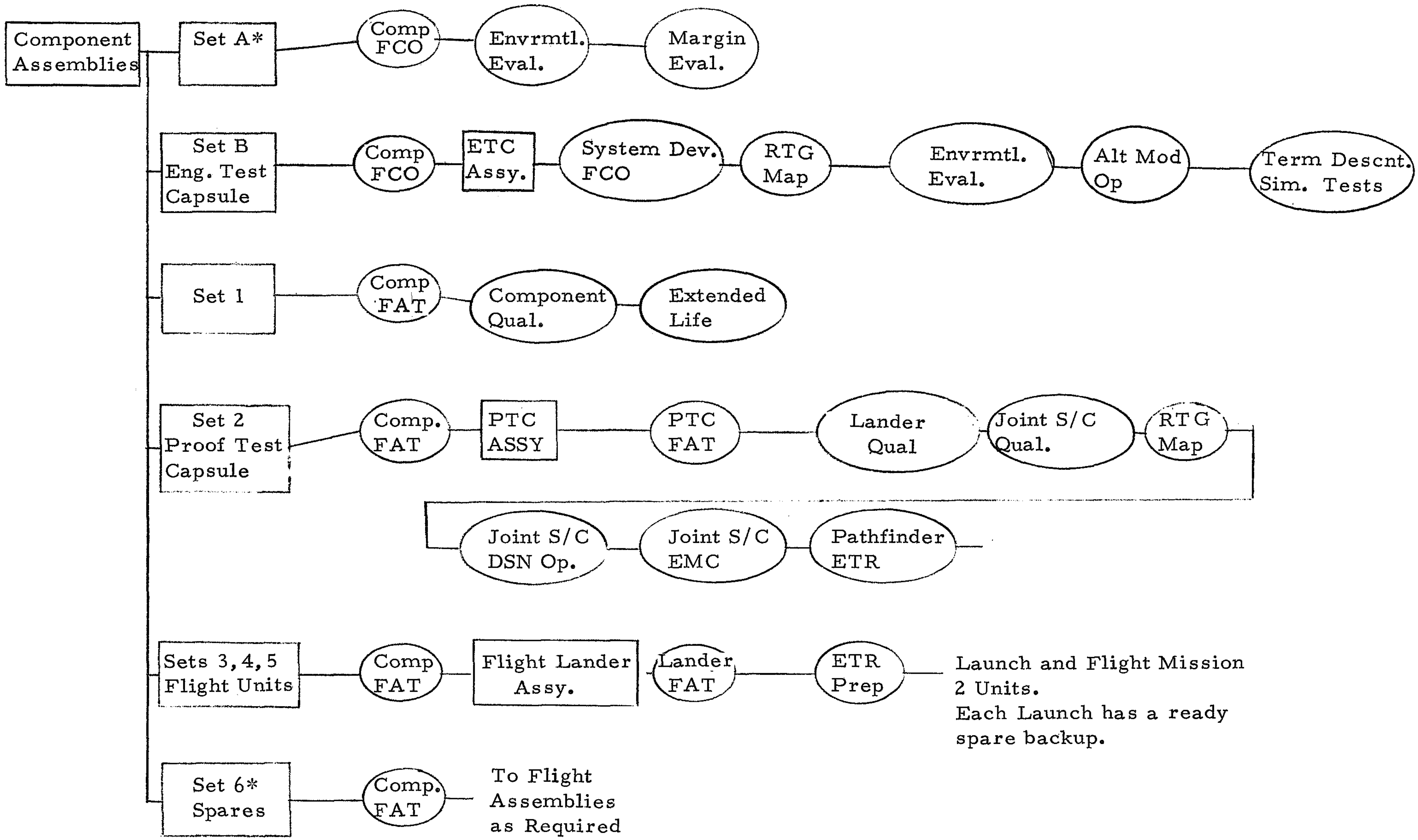
* Component inventory changes with each Spacecraft

**Set D applies to Nimbus B, Set E to Nimbus D, etc.

***Set 4 was made up from spares and reworked qualification units.

FCO Functional Checkout
FAT Flight Acceptance Test
S/C Spacecraft

FIGURE 7-5 BLOCK DIAGRAM OF HARDWARE UTILIZATION, VIKING LANDER



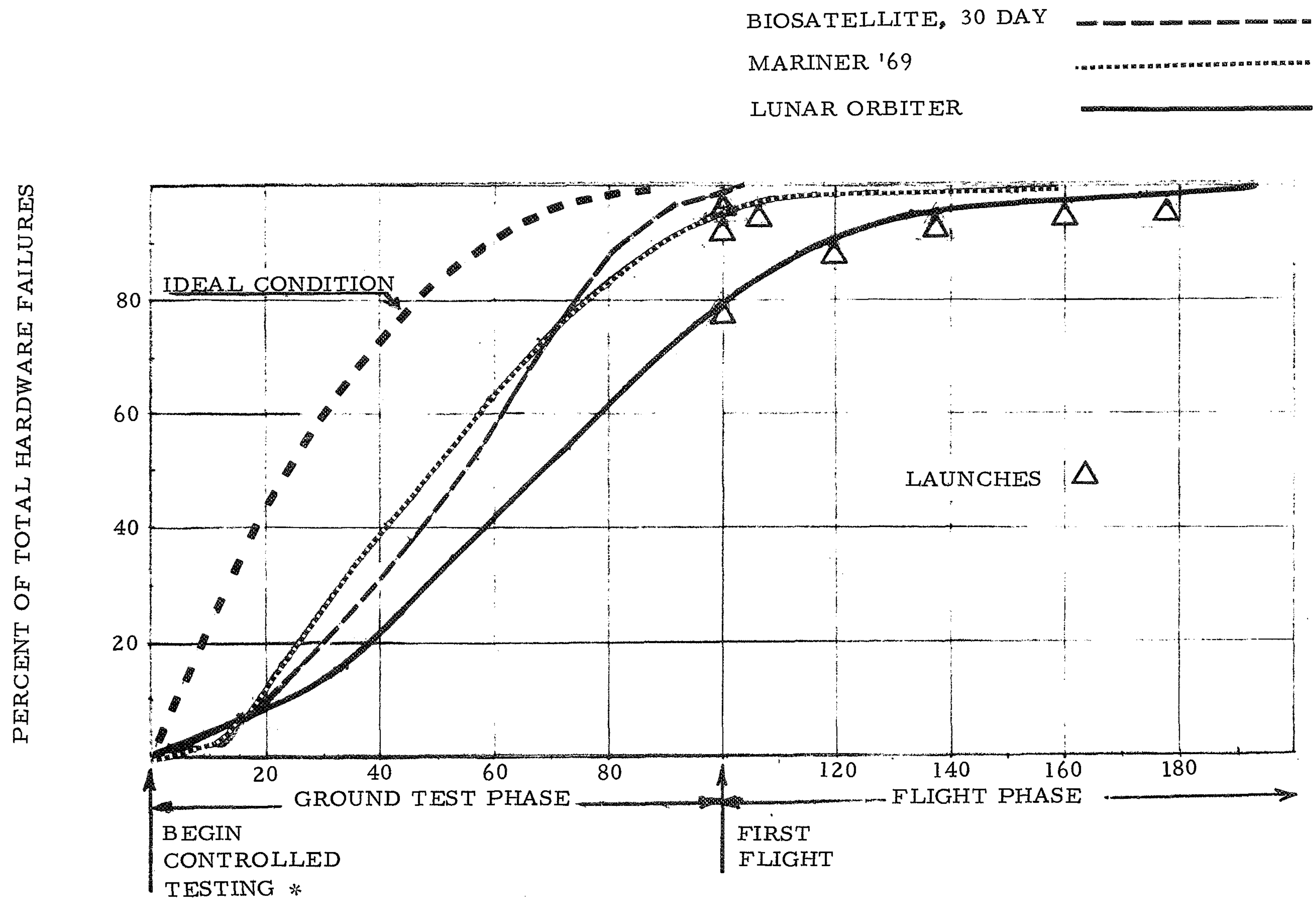
*Note: These sets may contain more items than just the inventory for one Lander Capsule Assembly.

FCO Functional Checkout
FAT Flight Acceptance Test
Comp. Component
S/C Spacecraft (Orbiter plus Lander)
DSN Deep Space Network

HARDWARE

FUNCTION

FIGURE 7-6 · COMPARISON OF HARDWARE FAILURES IN RELATIVE TIME TO FIRST LAUNCH



* Acceptance Tests of
Qualification Components

8.0 CONCLUSIONS AND RECOMMENDATIONS

This report provides data and correlations from the experience of the four space programs studied. These programs were selected because of features related to the Viking Lander, and the report is offered to support decision making for that program. However, the material herein should prove useful for space programs in general.

The problem descriptions in this report can be used to develop check list items for design, manufacture, quality, test, and operations; however, it is not believed possible to develop a list of rules which would assure program success. The potential value of the material in this report is to support the trade-offs among reliability, schedule, cost, weight, and function which are necessary at each step in a space program. It is with this understanding that general conclusions and recommendations are presented in Table 8-1. Particular attention is drawn to the following observations from flight experience data:

- 53% of flight failures were caused by design weakness.

- 38% of flight failures had occurred previously during ground testing. In each case the potential failure was considered to be an acceptable risk as to chance and consequence of occurrence. This represents an undesirable pattern of not recognizing potential flight failures.

- 24% of flight failures were new effects which had not occurred previously. These failures showed the need for design margins in the spacecraft and more realistic simulation in test.

- 18% of flight failures escaped their ground test. The failure cause was known and ground tests had been provided for their detection. It could be concluded that these tests were not adequate to demonstrate the required functional/environmental capability.

- 10% of flight failures were attributed to environments encountered during the mission. It can be concluded that the ground test environmental exposures were effective, however not sufficiently complete.

(Note: These items are not mutually exclusive, and therefore can add to more than 100%.)

TABLE 8-1 STUDY CONCLUSIONS AND RECOMMENDATIONS

SIGNIFICANT OBSERVATIONS AND CONCLUSIONS	RECOMMENDATIONS
A. FLIGHT FAILURES	
1. Well established parts, materials, and processes - standard risks -	
a. Parts	a. Parts
Problems can be expected with single source parts, particularly multifunctional items such as integrated circuits. These require extensive development and are prone to processing quirks which can be veiled in proprietary technology. Examples are in the microcircuits experiences with Lunar Orbiter and Mariner.	For spacecraft projects which do not have flexible schedules, single-source parts should be avoided. In addition Procurement and Specification controls must be closely managed.
b. Potentiometers	b. Potentiometers
Wire wound potentiometers are highly susceptible to localized wear and contamination which results in erratic output. This condition also applies to potentiometric elements used in transducers, position feedback elements, pickoffs, circuit adjustments, etc.	Inductive pickoffs avoid the problems inherent with potentiometers. However wherever potentiometric items are employed, the application must be carefully evaluated for the impact of these known problem areas.
c. Solder Joints	c. Solder Joints
Poor solder joints have been a persistent source of failures. They were the more likely candidate cause of the Nimbus S-Band transmitter failure and the failure of Biosatellite to deorbit.	It should be required and verified that all electrical connections be mechanically secured before soldering. For connectors, where this cannot be done, stresses on the joints should be relieved before encapsulation. Cable bends should be formed before lacing. Design review should extend to cover these items and include verification of the mock ups used for harness routing.

SIGNIFICANT OBSERVATIONS AND CONCLUSIONS	RECOMMENDATIONS
2. Flight Failures with a Previous History in Ground Test The most frequent flight failures and the cause of nearly half of flight data loss were incidents which had a previous history in ground test operations. These were considered as acceptable risks at the time of launch. The sources of these incidents are either advanced hardware configured for the particular spacecraft or system operating problems. Examples - Rate Gyros, Tape Recorders - EMI Sensitivities	2. Problem areas uncovered by ground tests need to be resolved more conclusively than has been done in past space programs. The material presented in Section 5 identifies potential problems which can reasonably be expected to impact a major space program and which merit special attention from the outset of a new project. A particular emphasis is needed in the evaluation of test "anomalies" and "recurrent glitches" which may signify fundamental operational weakness. A thorough review of re-design, re-work and re-test for "re-qualification" and flight acceptance is also necessary.
3. Flight Failures without Precedent Flight failures from new effects included untried spacecraft equipment such as star trackers; new environments encountered such as unexpected RF fields; and unprecedented loss of contact or control of a spacecraft. Each mission has encountered new effects in the course of flight.	3. An approach toward circumventing such incidents is through failure mode and effects analyses performed in depth early in the design phase by multi-disciplinary teams. The team should analyze for unexpected flight events such as thermal variations, RF interferences, etc. These analyses can provide the basis for establishing the desired margins and potential work arounds for the spacecraft and mission operations.
4. Flight Failures Which Escaped Detection by the Intended Ground Tests These failures show 4 general sources: 1) Incomplete environmental exposures. 2) Manufacturing defects which were difficult to detect, or were introduced by change. 3) Defects which would have occurred in more prolonged tests. 4) Defects or conditions masked by equipment or the operating environment. The latter of this is the most difficult to overcome. Available data and experience often points to sources outside the spacecraft.	4. Elimination or reduction of this class of failure requires effort in the following areas; • Detail review of environmental levels coupled with provision for extended or margin testing. • Detail review of simulation required and method of achieving, particularly for disturbance sources originating outside the spacecraft. • Presume that the failure originates in the spacecraft until conclusive data identifies an external origin. • Control of retest associated with replacement of components during the course of a test.

SIGNIFICANT OBSERVATIONS AND CONCLUSIONS	RECOMMENDATIONS
B. HARDWARE FAILURES AND CAUSES	
1. Tape Recorders Long term operation of tape recorders has been difficult. Problems have included: - Head contamination - Flutter - Drive mechanisms - Control - Tape stability against ageing, temperature and humidity	1. Note the experience details in Table 5-5.5, including solutions to the problems.
2. Rate gyros The gyros used on these space programs have a number of inherent failure mechanisms such as fluid contamination, outgassing, bearing friction, and dimensional changes. There are techniques for eliminating or minimizing each of these, but the improvements tend to introduce new problems.	2. The gyro design for a new program should be carefully studied to determine how the known prevalent failure mechanisms have been minimized.
3. Vibration Sensitivity All four spacecraft have shown effects of dynamic sensitivity. These involved inadequate attachment, overstress due to resonance, etc.	3. Perform exploratory model surveys of spacecraft structural response at component mounting locations to define transmissibility at a time during the project sufficiently early to aid the design of components.
4. Corona and Arcing High voltage breakdown due to arcing or corona have occurred or threatened all four spacecraft.	4. A design review should consider these effects and tests should be performed in deep space vacuum to the maximum extent possible. Do not assume "sealed" components are in fact sealed. Evaluate possible effects of internal outgassing very thoroughly.
5. Ringing and Oscillations These have been a persistent source of failures. They include effects within components, effects across interfaces and loop stabilities. The presence of beat frequencies and harmonics have circumvented redundancies and eliminated isolation margins to the extent of significant failures in flight.	5. Testing must include operation at extremes of voltage and signal levels with documented results early in the development phase of components, subsystems, and system.

TABLE 8-1 STUDY CONCLUSIONS AND RECOMMENDATIONS (Contd.)

SIGNIFICANT OBSERVATIONS AND CONCLUSIONS	RECOMMENDATIONS
6. Interface Compatibility Signals across interface between components have been a persistant problem area. These problems arise when spurious bits, pulses and transients are accepted into the main signal stream. Component noise rejection criteria were not adequately specified in these programs.	6. Noise rejection criteria should be adequately specified and demonstrated in new programs.
7. Polarity reversals have caused major problems in flight and delays in test.	7. Specific responsibility and procedures should be assigned to verify polarities in flight equipment, test setups at design reviews and test data reviews.
8. Pyrotechnically initiated devices are sensitive to relatively small changes in the pyro charge which occur during resizing or process changes.	8. Verify satisfactory operation of pyrotechnically initiated devices, under extreme conditions (3σ) of initiation.
C. TESTING CONSIDERATIONS	
1. EMI EMI control has been difficult. All projects followed present specifications toward establishing a 6 db margin, however EMI problems have persisted into flight.	1. EMI 1) The use of 12 db margin is suggested. 2) An EMI design review would be helpful in reducing the problem. 3) It is desirable to institute a high level EMI review board, and assign design analysts to explore the problem in depth. 4) Specifications should be definitized for components and interfaces across subsystems to prescribe EMI controls and verification tests.
2. Test Procedures Complex test setups and lengthy procedures caused test problems and delays.	2. Test Procedures It is desirable to limit the scope of individual tests to facilitate their conduct and controls. Procedures review and control must be given more attention for complex systems testing.

TABLE 8-1 STUDY CONCLUSIONS AND RECOMMENDATIONS (Concluded)

SIGNIFICANT OBSERVATIONS AND CONCLUSIONS	RECOMMENDATIONS
D. FLIGHT OPERATIONS	D. FLIGHT OPERATIONS
All the long life spacecraft have had stressful flight control and real-time failure diagnostic problems despite capability in the hardware.	Plan operational workarounds to handle degraded modes of the system especially for failure of non-redundant elements.
E. FAILURE REPORTING, CONTINUITY OF EXPERIENCE	E. FAILURE REPORTING, CONTINUITY OF EXPERIENCE
Future space programs should, at their outset, make studies similar to this such that experience can be effectively used. Repetitive or anomalous flight problems generally have a source design weakness or an unknown operational environment. Fail-safe concept of flight operation is not always possible due to equipment limitations.	Feedback of past problems and failure experience into design should progress from the initial preliminary design review to the final critical design review and should be a formal process. Continuation of such inputs throughout testing and flight operations is needed to ensure the maximum return of problem experience.

BIBLIOGRAPHY

The following listed sources provided the data used for this study. Principal sources are indicated by an asterisk (*).

1. LUNAR ORBITER:

* Analysis of Flight Problems (Boeing Company, March 1968)	D2-100807-1
* Lunar Orbiter Test Program (NASA-LRC, June 1967)	LWP-442
* Lunar Orbiter Failure Summary (NASA-LRC, Feb. 1968)	LWP-552
* Failure Summary Descriptions (NASA-LRC, 1967)	Unpublished
* Summary Notes, Diagrams (NASA, LRC, 1967)	Unpublished
Lunar Orbiter Flight Reports (The Boeing Company, 1968)	CR Reports for 5 Flights

2. MARINER '69:

Space Program Summaries (JPL, 1967-1969)	37-47 thru 60, Inclusive
* MM '69 Notable Problem/Failure Reports (JPL, Feb. 1969)	605-173
* Mariner Mars 1969 Handbook (JPL - July 1969)	605-211
* MM '69 EMC P/FR Study for Updating M '71 Constraints (JPL, Feb. 1970)	
* Mariner Mars Environmental Test Result Summary (JPL, Jan. 1970)	605-240

BIBLIOGRAPHY (continued)

* Appendix 1, Environmental Test Result Summaries (JPL, 1970)	
* Summary Listing of All Problem Failure Reports MM-69 (JPL, Feb. 1970)	Unpublished
Mariner Mars Parts Control Document Section 4 (JPL, No Date)	605-13

3. BIOSATELLITE:

* Biosatellite Failure Summary Report (GE-RESD, May 1969)	69SD701
* Biosatellite Final Project Report (GE-RESD, Dec. 1969)	69SD5225
* Biosatellite 3 Flight Evaluation Report (GE-RESD, Oct. 69)	69SD845
Qualification Test Report Vehicle 203 (Primate) (GE-RESD, May 1969)	-
* Internal Environmental and Component Qualification Test Requirements (GE-RESD, Dec. 1968)	S0020-02-0004-F
* Biosatellite Primate Mission Systems Acceptance Test Specification (GE-RESD, Feb. 1969)	S0040-01-0015-F
Electromagnetic Compatibility Test Specification, Biosatellite (GE-RESD, Feb. 1967)	S0010-09-0004-B
Notes and Letters from the Biosatellite Project	Unpublished

BIBLIOGRAPHY (concluded)

4. NIMBUS B:

* Failure Reports Nimbus B (GE-RESD, 1968-1969)	Unpublished
Environmental Test Requirements for Prototype and Flight Acceptance Testing of Nimbus B and D Components (GE-SD, Nov. 1967)	N17-B-7445 Rev. C
* Excerpts from the Nimbus Monthly Report (GSFC, June, 1970)	-
* Nimbus Spacecraft Log Books (GE-SD, May 1969)	-
* Summary Document Nimbus B Reliability Assessments and Failure Mode Analysis (ORI to GSFC, Feb. 1968)	Tech. Report 469

5. VIKING LANDER:

* Viking '75, Notes from Presentations (LRC, March 1970)	Unpublished
* Master Integrated Test Plan	
Summary Volume	PL 3710000
Appendix A (MMC, Jan. 1970)	PL 3710001
* Test Methods and Controls (MMC, Jan. 1970)	83710010605
Environmental Specification (MMC, Nov. 1969)	QR-3701033
* Viking Mission Definition No. 3 (LRC, Mar. 1970)	M75-123-0

NASA-Langley, 1971