GENERAL DYNAMICS
Fort Worth Division

FZM-4492-I
15 February 1966
Volume I

S U M M A R Y

SATURN IN-FLIGHT EXPERIMENTAL PAYLOAD STUDY

CONTRACT NAS8-20236

Prepared for the

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FOREWORD

This document is a summary of the results of the Saturn In-Flight Experimental Payload Study. The analysis was performed by the Fort Worth Division of General Dynamics for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under Contract No. NAS8-20236. The study was established by the Advanced Systems Office of NASA-MSFC as part of an effort to provide for the orderly and economic utilization of space vehicle hardware in the tasks devoted to the accumulation of scientific data.

The complete results of this study are contained in the following volumes:

Volume I - Summary

Volume II - Technical Report: Design of In-Flight Experiments

Volume III - Technical Report: Computer Program Development and Methodology

Volume IV - Utilization Instructions

This study was performed during the period beginning July 1965 and ending February 1966. The general guidelines of the study were set forth by NASA-MSFC in RFQ DCN 1-5-23-00009-01 and RFQ DCN 1-5-23-00010-01, and the Fort Worth Division has based the study effort on these guidelines in order to obtain the results described herein.

A C K N O W L E D G M E N T S

The technical direction and assistance of W. H. Stafford, Study Technical Supervisor, NASA-MSFC-ASO, is gratefully acknowledged. The following personnel at the Fort Worth Division of General Dynamics have made significant Technical contributions to this report:

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1.0 INTRODUCTION

1.1 GENERAL

This document is a summary of the results of the Saturn In-Flight Experimental Payload Study. The study was performed by the Fort Worth Division of General Dynamics for the George C. Marshall Space Flight Center.

By the utilization of the secondary payload capability of the Saturn family of launch vehicles, NASA can provide an efficient means for conducting the large number of Earth orbital experiments that has been suggested by the scientific community. Since it is to be assumed that the mission of each launch vehicle is designed to attain specific objectives associated with the primary payload only, it is essential that the in-flight experiments and the launch vehicle be properly mated to provide for efficient utilization of the remaining mass and volume capability of the launch vehicle and for the accomplishment of a high percentage of the experiment data acquisition objectives. Because of the combination of numerous vehicles with varying missions and capabilities and a large number of experiments with varying requirements, the evaluation of the vehicle/experiment mating presents a significant management problem. The basic objective of the Saturn In-Flight Experimental Payload Study is to provide NASA with a management tool in the form of a computer program which can be used to make a rapid evaluation of numerous potentially attractive space experiments that constitute possible secondary in-flight payloads for the Saturn family of launch vehicles.

1.2 APPROACH

To attain this overall study objective, two major study tasks were specified: (1) an analysis of the physical characteristics of sensors and associated equipments for use as possible experimental payloads on Saturn-class vehicles and the mission effectiveness values of these experiments as a function of the initial elements and/or mission parameters of the deployed orbit, and (2) the development of a computerized methodology for the technical evaluation and rating of these potential in-flight experimental payloads.

The technical approach used throughout the study is based on the development of Program SEPTER, (Saturn Experimental Payload Technical Evaluation and Rating). Two fundamental criteria are employed in Program SEPTER to evaluate the experiments that are being considered for possible inclusion on a Saturn flight: (1) physical compatibility of the experiments with possible locations aboard the vehicle, and (2) experiment/mission effectiveness. Experiment/mission effectiveness is defined as the percent of the data acquisition objectives which would be attained by including a particular experiment on a given Saturn flight. The physical compatibility of an experiment package with a vehicle location refers, in this study, not only to mass/volume compatibility but also to compatibility with the thermal, acoustic, vibration, and electromagnetic environments.

The basic program plan shown in Figure 1-1 was developed by the Fort Worth Division of General Dynamics to achieve the objectives set for this study. The use of this approach permits (1) an analysis of the physical characteristics and mission sensitivity of experiments of in-flight payloads for Saturn-class vehicles, and (2) the determination of a computer methodology for the technical

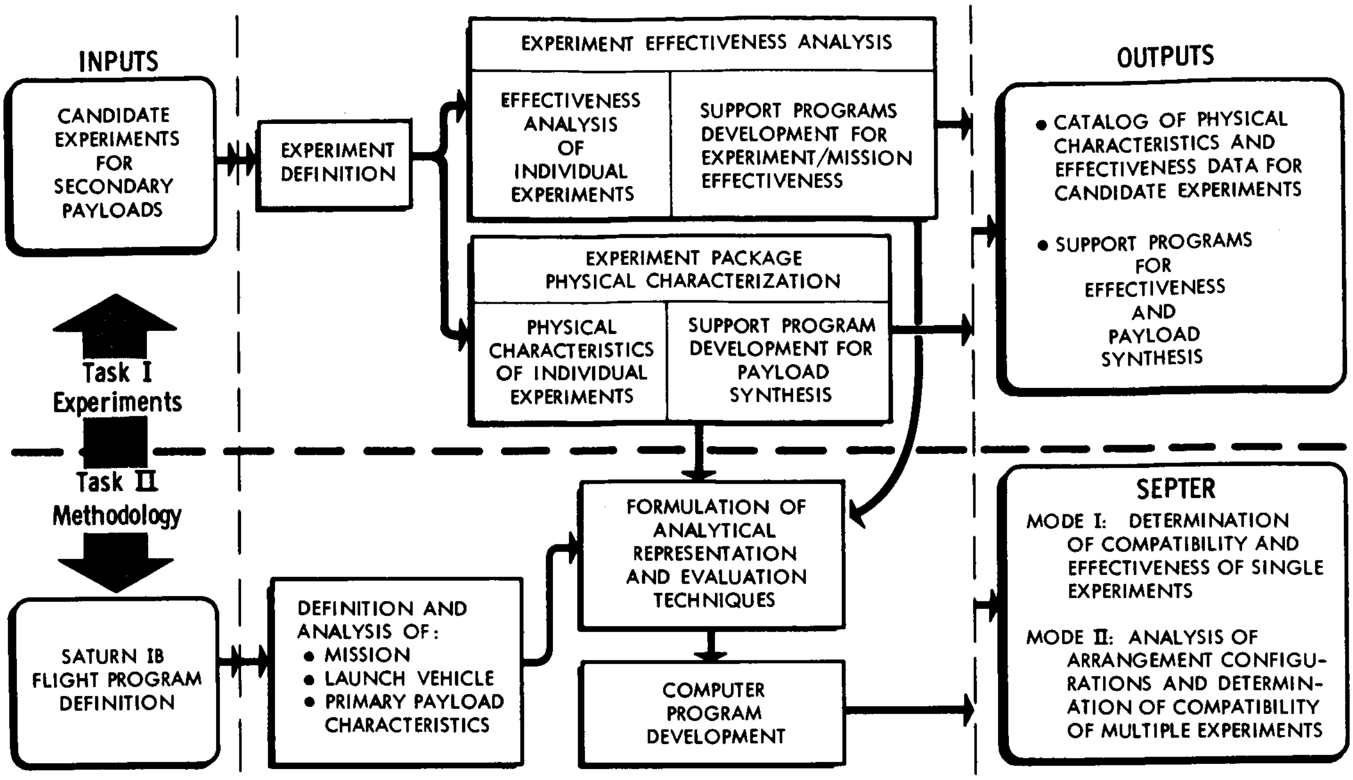


Figure 1-1 BASIC PROGRAM PLAN

evaluation and rating of these in-flight experimental payloads. The technical plan is divided into the individual study areas associated with the experiments-related task (Task I) and the computer methodology development task (Task II).

1.3 GUIDELINES AND GROUND RULES

A number of guidelines and ground rules were formulated at the beginning of the study in order to establish the overall study philosophy and to limit the scope of the experiment and vehicle analyses. The experiments considered in this study constitute secondary payloads in that the missions on which these experiments may be flown have been designed to attain the specified objectives associated with the primary payload. For example, the primary missions which were used in the mission characteristics library of the computer program are the Saturn IB/Apollo flight test missions. The basic Apollo spacecraft (Command Module, Service Module, and Lunar Excursion Module) is the primary payload, and any additional experimental packages carried on these flights are secondary payloads. Although other vehicle configurations will eventually be included in the launch vehicle/primary payload characteristics library, the Saturn IB/Apollo, including the Command Module, the Service Modules, and the Lunar Excursion Module, was chosen as the baseline configuration for this study.

The Fort Worth Division of General Dynamics acknowledges the prerogative and responsibility of NASA to define and approve in-flight experiments. However, in order to understand how the computer methodology may be affected by differences in (1) the physical characteristics of experiment packages and vehicle cavity locations, and (2) the requirements for realistic examples of experiment effectiveness, it was necessary for the Fort Worth Division to define a number of potentially attractive in-flight experiments. In establishing configuration designs for these experiment packages, primary emphasis was placed on self-contained packages; that is, consideration was not given to using the support capabilities of on-board equipments or to the possibility of sharing subsystems among experiments. The experiment packages were designed to assure that they do not in any way interfere with the primary payload. Furthermore, the package designs were based on the assumption that only a minimum of astronaut participation will be allowed, i.e., only to effect off-on switching, film retrieval, etc.

1.4 SUMMARY OF MAJOR STUDY ACCOMPLISHMENTS

The major tasks which have been accomplished as a result of this study effort are summarized below.

1. From the list of 85 experiments provided in NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights, 30 experiments were selected which were representative of the list and were compatible with the study ground rules. The following were accomplished in the case of each of these 30 experiments:

    a. The physical characteristics of the experiment sensors and the ancillary systems(attitude control, data automation, communications, electric power, and thermal control) were defined.

    b. The thermal, vibration, acoustic, and electromagnetic environmental requirements were established.

    c. Conceptual design drawings were prepared, and the mass, volume, and geometry of the experiment were determined.

    d. The requirements for deployment were defined.

    e. Preliminary reliability, development schedule, and cost analyses were performed.

2. The pertinent mission characteristics (trajectory parameters, sequence-of-events, and experimental payload possible deployment modes) of a typical Saturn IB/Apollo Earth-orbital mission were defined and analyzed.

3. A total of 53 cavities (potential payload locations) were identified on the Saturn IB/Apollo vehicle. The following were accomplished in the case of each of the 53 cavities:

    a. Isometric drawings were prepared showing the cavity shape and volume.

    b. The mass capacity was determined.

    c. The thermal, vibration, acoustic, and electromagnetic environments were established.

    d. The deployment capability was defined.

4. A methodology was developed for describing experiment and cavity volume/geometry by the use of standard geometric shapes (sphere, cylinder, and parallelepiped). Each experiment was represented by its total volume and the standard shape of its critical component. Each cavity was defined by its total volume and by its capacity to contain the standard shapes.

5. A methodology for describing experiment effectiveness as a function of the initial elements and/or mission parameters of the deployed orbit was developed, and parametric effectiveness analyses were performed on example experiments.

6. A computer program (SEPTER) was developed to evaluate and rate in-flight experimental payloads. The overall capabilities of this program are a result of the development of some unique and simplified methodologies which are reasonably accurate for the solution of generally complex problems. These methodologies include the following:

   a. The simulation of experimental payload deployment modes and the calculation of the orbital elements and/or mission parameters for the deployed orbit.

   b. The computation of experiment/mission effectiveness as a function of the initial orbital elements of the deployed orbit. A technique was developed in which three types of effectiveness factor relationships are utilized: (1) continuous function of two variables, (2) step function of two variables, and (3) continuous or step function of one variable. Two interpolation techniques are available.

   c. The determination of the experimental payload-mission/vehicle compatibility with numerous physical and operational criteria. A reasonably simple technique was developed for the determination of geometric compatibility between arbitrarily shaped cavities and experimental payloads represented by standard shapes.

   d. The determination of multiple experimental payload arrangements aboard a vehicle. A technique was developed which satisfies all constraints and directly searches for a non-unique "optimal" arrangement.

7. A computer program (DESIGN) for determining limited physical characteristics of arbitrary experiments was developed as a support program for Program SEPTER. DESIGN replaces the manual subsystem synthesis tasks of designing experimental payloads and provides "first-pass" estimates of mass and volume requirements.

2.0 OVERALL COMPUTER PROGRAM PHILOSOPHY AND LOGIC

Computer Program SEPTER provides NASA with a management tool with which to evaluate and rate numerous potentially rewarding space experiments that constitute possible secondary in-flight experimental payloads for the Saturn family of launch vehicles. Because SEPTER provides for the comprehensive manipulation of data, it is possible to continuously update the experimental payload evaluation and rating from the conceptual design stage through the fixed design stage, and in some cases up to the actual launch date.

2.1 CAPABILITIES AND UTILIZATION

The evaluation and rating of experimental payloads is based on two fundamental criteria employed in Program SEPTER:

1. Physical and operational compatibility of the experimental payloads with a given vehicle/primary payload and mission.

2. Experiment/mission effectiveness, i.e., percent accomplishment of data acquisition objectives.

SEPTER methodology is broad in scope yet sufficiently accurate to provide meaningful results. The primary objective was the development of a program methodology rather than the development of specific data for use in the program. Although specific data were developed, these data are only representative of the missions, vehicles, and payloads (both primary, i.e., the Saturn/Apollo configuration, and secondary) which can be handled with the computer program. Other data can be readily loaded for use in the program.

The overall capabilities and the method of utilization of Program SEPTER are illustrated in Figure 2-1. The program contains provisions for operating in two basic modes of analysis. In the Mode I operation, single experimental payloads are analyzed with respect to (1) physical compatibility with a specific vehicle location (e.g., volume/geometry, attachment mass, and environmental field criteria such as thermal, acoustic/vibration), (2) operational compatibility (e.g., launch date and deployment mode), and (3) effectiveness in accomplishing their data acquisition objectives. In the Mode II operation, multiple experimental payloads are analyzed for possible arrangements which satisfy all compatibility constraints and allow a near-maximum number of experimental payloads to be placed aboard the vehicle.

The mission/vehicle/primary payload data and the experimental payload data for SEPTER are stored in the form of libraries. This method of storage provides a high degree of flexibility in the use of the program. Various combinations of mission/vehicle/primary payload - experimental payloads may be selected at the user's discretion for use in SEPTER. Preliminary definitions may be readily updated. The libraries may be easily modified and expanded to include other spacecraft (e.g., Apollo Applications - LEM Lab, NASA Can, etc.), missions, and payloads.

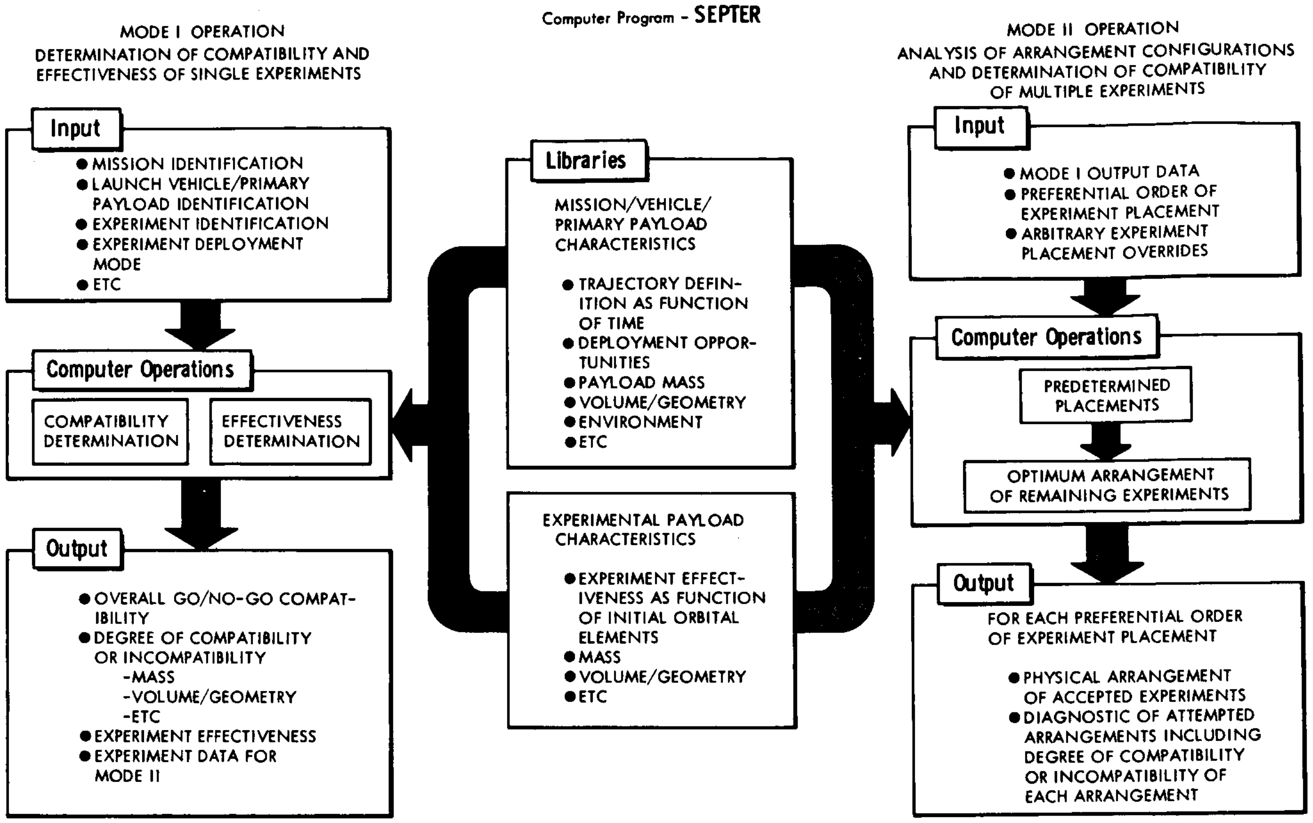


Figure 2-1 SATURN EXPERIMENTAL PAYLOAD TECHNICAL EVALUATION AND RATING

2.2 GENERAL CONSTRUCTION AND OPERATION

In Figure 2-2, the overall flow of calculations in Program SEPTER is shown schematically, including the types and forms of data inputs, the major areas of analyses (designated as subroutines), and the types and forms of output data for each mode of operation.

Mode 0 is not an analysis mode; its function is to perform unit conversions and compile binary library tapes of mission/vehicle/primary payload characteristics from card decks for direct input to Mode I. (The use of a binary library tape makes it possible to decrease computer running time and makes data conversion, and the storage of internal and external data other than card decks more efficient.)

Mode I is the operational mode in which the compatibility and effectiveness of a single experimental payload is analyzed. As shown in Figure 2-2, inputs consist of the mission/vehicle/primary payload library data (binary library tape), the experimental payload characteristics library data (card decks), and problem control data (card decks). Problem control data is used to identify tapes, select computational options, and specify overrides for the binary library tape input data. Depending upon the options selected, a limited number or all of the analysis subroutines shown in Figure 2-2 are utilized. For example, either the compatibility or the effectiveness analysis may be selected independently. Similarly, the overrides in the problem control may be used to specify, for example, a new launch date or a different excess payload capability. The output of Mode I is in the form of printed results and, if

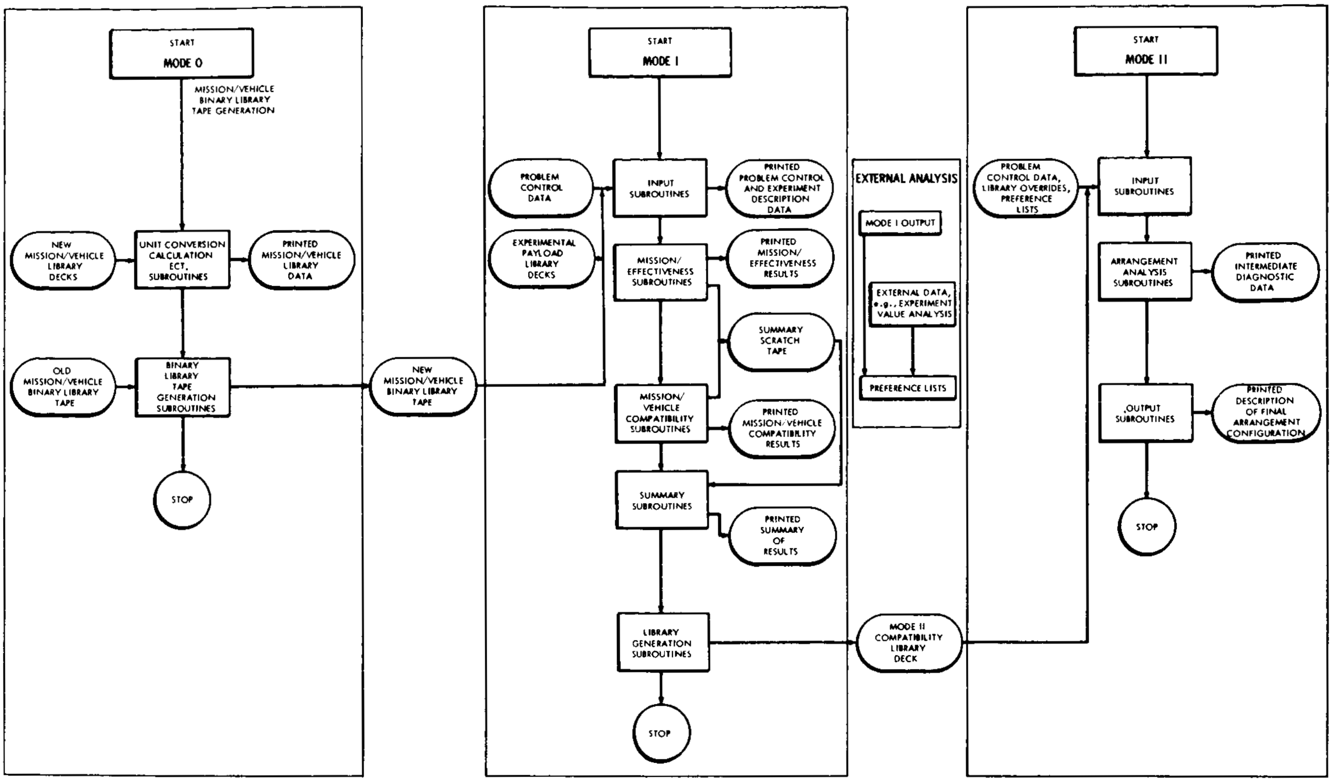


Figure 2-2 COMPUTATION FLOW DIAGRAM

specified, a library deck containing the required input for Mode II. This library deck consists of library data utilized in Mode I plus the computed compatibility results pertaining to individual experiments.

The external analysis required between the operation of Modes I and II consists in the formulation of preference lists which establish the desired order (priority) in which experiments are to be loaded aboard the vehicle.

Mode II is the operational mode of Program SEPTER which is used to analyze multiple experimental payload compatibility and arrangement configurations aboard the vehicle. The inputs in Mode II consist of a preference list, a compatibility library deck (from Mode I output), problem control data, and library overrides. These problem controls and overrides consist of, for example, predetermined placements for arbitrary experiments, deletion of cavities, and optional print-out of diagnostic data. The output of Mode II is in the form of printed results in which the accepted experimental payloads from the preference list and the cavities within which they have been placed according to the predetermined and optimal arrangement analyses are listed.

3.0 APPROACH TO DESIGN OF IN-FLIGHT EXPERIMENTS

It has been previously acknowledged that NASA has the prerogative and responsibility to define and approve in-flight experiments. It was necessary, however, in order to develop a workable computer methodology for the evaluation of such experiments, to define a number of experiments in order to establish typical requirements in terms of the physical characteristics of the packages and experiment data gathering objectives. The approach taken in defining these typical experiments is outlined in Figure 3-1.

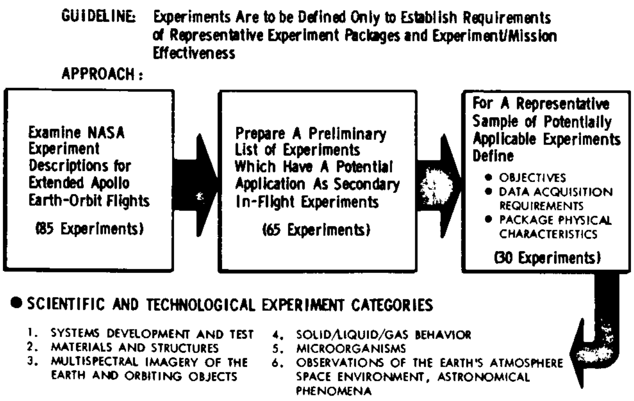


Figure 3-1 APPROACH TO DEFINITION OF EXPERIMENTS

The document NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights was reviewed, and the 85 experiments listed therein were examined to determine which of them might be performed as a secondary in-flight experiment. A number of the experiments were eliminated from consideration because of their incompatibility with the guidelines and ground rules established for the study. In a number of the experiments, for example, either a large payload capability or an excessive amount of astronaut participation was required. It was found that approximately 65 of the 85 experiments contained in this document could be defined as secondary in-flight experiments.

Each of these 65 experiments was examined in terms of the physical characteristics of the experiment package (size, mass, etc.), the operational requirements for the sensor (deployment, viewing, etc.), and the data gathering requirements related to elements of the orbit in which the sensor must be deployed. In this manner, 30 representative experiments were selected from the list of 65. This list of 30 was composed of 5 experiments in each of the six scientific and technological experiment categories indicated in Figure 3-1. The physical characteristics of the package, detailed objectives, and data acquisition requirements were determined for each of the 30 experiments.

Ancillary system characteristics were defined in the system detailed design on the basis of requirements related to the sensors, the experiment operations, and the environment. The physical characteristics of experiment sensors and ancillary systems were then used in achieving a configuration design for each experiment package. The results of the efforts on the individual in-flight experiment package designs have been expressed in terms of mass, volume/geometry (including critical dimensions), and other characteristics and have been stored on library magnetic tapes for use in the SEPTER computer program.

A support program, DESIGN, has been developed for the limited synthesis of arbitrary experiments. The approach used in the development of the computer methodology is shown in Figure 3-2. The characteristics of ancillary systems have been defined to satisfy the given input requirements by means of analytical representations either in the form of curve fits or characteristics stipulated for required components.

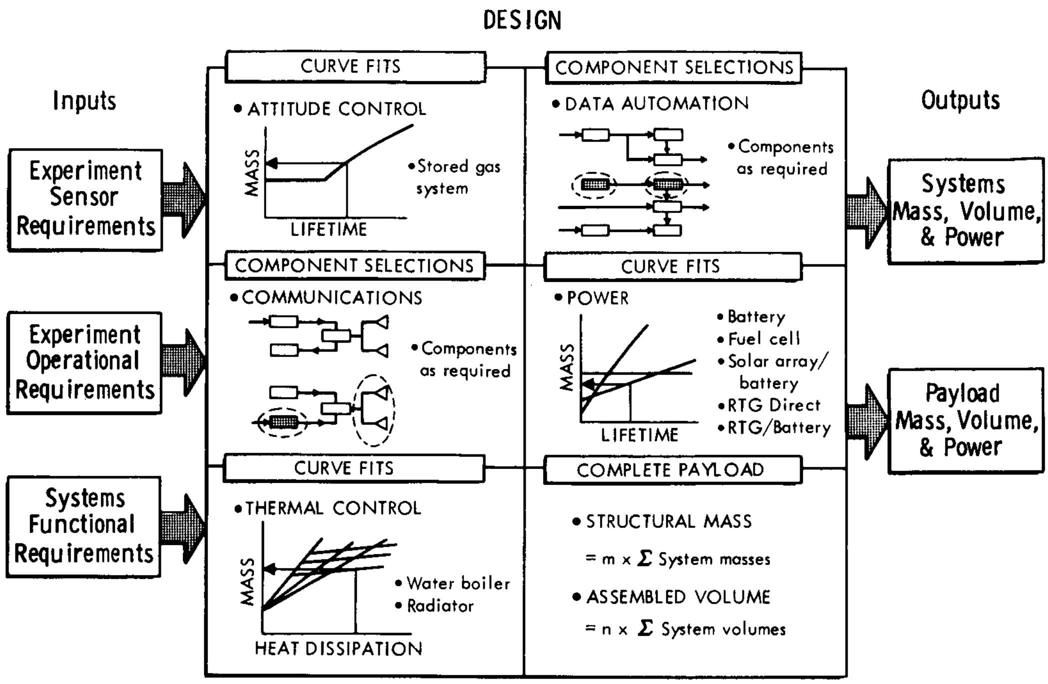


Figure 3-2 APPROACH TO LIMITED SYNTHESIS OF EXPERIMENTS

4.0 DEFINITION OF EXPERIMENTS

4.1 SELECTION OF A REPRESENTATIVE SAMPLE OF POTENTIALLY APPLICABLE EXPERIMENTS

The experiments which were selected for inclusion in the experiment characteristics library of Program SEPTER are listed in Table 4-1. These experiments are grouped under the six scientific and technological experiment categories which were established by the Fort Worth Division to ensure broad coverage of the scientific and technical disciplines outlined in the document NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights.

TABLE 4-1
LIST OF SELECTED IN-FLIGHT EXPERIMENTS

1. Systems Development and Test	SDT-1	RADIOISOTOPE-THERMOELECTRIC POWER SYSTEM INTEGRATION
	SDT-2	PERFORMANCE ASSESSMENT OF THIN-FILM SOLAR CELL ARRAYS
	SDT-3	PERFORMANCE ASSESSMENT OF SPACECRAFT NAVIGATION, GUIDANCE AND CONTROL HARDWARE AND TECHNIQUES
	SDT-4	CRYOGENIC PROPELLANT STORAGE SYSTEM PERFORMANCE
	SDT-5	LAUNCH OF UNMANNED SATELLITES AND PROBES
2. Materials and Structures	MS-1	DEGRADATION OF ORGANIC MATERIALS IN A SPACE ENVIRONMENT
	MS-2	BEHAVIOR OF LIQUID FILMS IN A SPACE ENVIRONMENT
	MS-3	VAPORIZATION RATE OF MOLTEN METALS
	MS-4	COLD WELDING OF METALS IN A SPACE ENVIRONMENT
	MS-5	SPRAY COATING AND SURFACE CONTAMINATION IN A SPACE ENVIRONMENT
3. Multispectral Imagery of the Earth and Orbiting Objects	MI-1	MULTISPECTRAL SURVEILLANCE OF EARTH
	MI-2	INFRARED LINE SCAN SURVEILLANCE OF EARTH
	MI-3	RADAR SURVEILLANCE OF EARTH
	MI-4	ELECTRONIC IMAGE MOTION STABILIZATION
	MI-5	SYNOPTIC EARTH CARTOGRAPHY
4. Solid/Liquid/Gas Behavior	SLG-1	BOILING IN ZERO-GRAVITY ENVIRONMENT
	SLG-2	NUCLEATE CONDENSATION IN ZERO-GRAVITY
	SLG-3	FORMATION OF SINGLE CRYSTALS
	SLG-4	SEGREGATION OF IMMISCIBLE LIQUIDS UNDER ZERO-GRAVITY CONDITIONS
	SLG-5	ZERO-GRAVITY COMBUSTION
5. Microorganisms	M-1	SOFT CAPTURE, ENUMERATION, AND IDENTIFICATION OF SPACE-BORNE MICROORGANISMS
	M-2	EFFECTS OF SPACE FLIGHT ON MORPHOLOGY, GROWTH, AND LIQUID/GAS SEPARATION IN MICROORGANISMS
	M-3	INHERENT MUTATION RATES IN MICROORGANISMS AND EFFECTS OF EXTENDED SPACE FLIGHT ON THE EXPRESSION OF THE MUTATION
	M-4	DETERMINATION OF THE MIGRATION OF MICROORGANISMS IN A SPACECRAFT ENVIRONMENT
	M-5	PRODUCTION OF NUTRIENTS BY CERTAIN MICROORGANISMS WHILE IN SPACE FLIGHT
6. Observations of the Earth's Atmosphere, the Space Environment, and Astronomical Phenomena	OEA-1	RADIATION ENVIRONMENT MONITORING
	OEA-2	STUDY OF MAGNETIC FIELD LINES
	OEA-3	TEST OF PROTOTYPE STAR TRACKER
	OEA-4	COSMIC RAY EMULSION EXPERIMENT
	OEA-5	EMISSION LINE RADIOMETRY

It should be noted that these six categories were selected primarily for convenience in defining typical experiments; consequently, they are only a general indication of the broad range of experiment categories that are actually covered by the selected 30 experiments. The relationship of the 30 selected experiments to NASA's major scientific and technical areas of interest is indicated in Table 4-2. This table gives some idea of the scope of the experiments to be contained in the experiment characteristics library of Program SEPTER.

TABLE 4-2

RELATIONSHIP OF THE THIRTY EXPERIMENTS SELECTED BY GENERAL DYNAMICS FOR THE STUDY TO THE NASA MAJOR SCIENTIFIC/TECHNICAL AREAS OF INTEREST

NASA	General Dynamics
Space Science	
1. PHYSICAL SCIENCES	OEA-1, OEA-2, OEA-4, SLG-1, SLG-2, SLG-3
2. BIOSCIENCE	M-1, M-2, M-3
3. ASTRONOMY/ASTROPHYSICS	OEA-5
Support for Space Operations	
4. BIOMEDICINE/BEHAVIOR	NONE
5. ADVANCED TECHNOLOGY AND SUPPORTING RESEARCH	SDT-3, SDT-4
6. EXTRAVEHICULAR ENGINEERING ACTIVITIES	MS-1, MS-2, MS-3, MS-4, MS-5, SDT-2, SDT-5
7. OPERATIONS TECHNIQUES AND ADVANCED MISSION SPACECRAFT SUBSYSTEMS	SDT-1
Earth Oriented Applications	
8. ATMOSPHERIC SCIENCE AND TECHNOLOGY	OEA-3
• EARTH SCIENCES AND RESOURCES	
9. AGRICULTURE/FORESTRY	SUPPORTED BY MI-1, MI-2, MI-3, MI-4 AND MI-5
10. GEOLOGY/HYDROLOGY	
11. OCEANOGRAPHY/MARINE TECHNOLOGY	
12. GEOGRAPHY	
13. COMMUNICATIONS AND NAVIGATION/TRAFFIC CONTROL	NONE

Only two of the 13 major scientific/technical areas are not covered by at least one of the 30 experiments. These areas are (1) Biomedicine/Behavior, and (2) Communications and Navigation/Traffic Control. The Biomedicine/Behavior area is not covered because the experiments in this category are not compatible with the study ground rule concerning minimum astronaut participation. A number of experiments in the area of Communications and Navigation/Traffic Control listed in <u>NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights</u> are suitable for consideration as potential secondary experiments (e.g., measurement of radio frequency radiation, wide-bandwidth transmission in space, deployment of RF reflective structures, etc.) and were listed among the 65 applicable experiments. These particular experiments were not considered further because of the limited scope of the study.

4.2 SUMMARY OF SELECTED EXPERIMENTS AND THEIR REQUIREMENTS

A summary of sensor requirements for the 30 selected experiments is presented in Table 4-3. From an examination of the table it can be seen that a considerable variety in requirements has been encountered in choosing these representative experiments. For example, duty cycle requirements range from a few minutes per day to continuous operation. The in-flight duration required varies from 2 hours to 1 year. As might be expected, the requirements for accuracy, resolution, and data recovery also tend to be peculiar to each particular experiment.

TABLE 4-3
SUMMARY OF SENSOR REQUIREMENTS

Category		Sensors	Duty Cycle	In-Flight Duration	Accuracy	Data Recovery: Telemetry	Data Recovery: Capsule
1. Systems Development and Test	SDT-1	RTEG, CO_2 CONCENTRATOR, H_2O EVAPORATOR	CONTINUOUS	60 DAYS	————	√	
	SDT-2	SOLAR COLLECTORS STANDARD CELLS	10-30 MINUTES EACH WEEK	1 YEAR	$\pm$0.5 VOLT $\pm$2°F TEMP $\pm$0.1 AMPERE	√	
	SDT-3	IMU, TRACKERS, HORIZON SCANNER, COMPUTER	30 MINUTE PERIODS 10-30 HOURS OPERATION	14 DAYS	POSITION 0.01 N.MI. ACCEL 10^{-6}g VELOCITY 0.01 FPS ANGLE 1 SEC.	√	
	SDT-4	SUN SENSOR INSTRUMENTATION: TEMP, PRESS, FLOW	CONTINUOUS	4 DAYS	POINTING $\pm$3 DEGREES	√	
	SDT-5	ENERGETIC PARTICLES EXPLORER	————		————		
2. Materials and Structures	MS-1	SUN SENSOR THERMOCOUPLES	2 HOURS EVERY 10 HOURS	30 DAYS	TEMP $\pm$3°R		√
	MS-2	LIQUID FILM SENSOR	10 MINUTES EACH APOGEE	27 HOURS	————	√	
	MS-3	THERMOCOUPLES VAPORIZATION SENSOR	18 PERIODS 20 MIN. TO 4 HOURS	29 HOURS	TEMP $\pm$10°F	√	
	MS-4	COLD WELD SENSOR	10 PERIODS 4 HOURS	15 HOURS	————	√	
	MS-5	SPRAY COATING SENSORS	3 PERIODS 2 HOURS & 10 MIN	2.5 HOURS	————	√	
3. Multispectral Imagery of the Earth and Orbiting Objects	MI-1	PHOTOGRAPHIC CAMERAS, SPECTRORADIOMETERS, V/h SENSOR	40 DATA RUNS 1 MINUTE EACH	2 WEEKS	POINTING $\pm$1.5 DEG	√	√
	MI-2	IR LINE SCANNER	15-20 DATA RUNS 3-4 MINUTES EACH	2 WEEKS	POINTING $\pm$2 DEG VERT, $\pm$3 DEG AZIM	√	
	MI-3	SIDE LOOKING RADAR	1 HOUR PER DAY	2 WEEKS	30 METERS GROUND RESOLUTION (185 Km ORBIT)	√	
	MI-4	CAMERA, TELESCOPE FINDER/TRACKER	50 PER DAY 20 SECONDS EACH	1 WEEK	12.2 METERS GROUND RESOLUTION (185 Km ORBIT)	√	
	MI-5	CAMERA V/h SENSOR	5 RUNS PER DAY 11 MINUTES EACH	20 DAYS	15 m. HORIZ GRND RESOLUTION 25 m. VERT GRND RESOLUTION TIME 0.01 SEC		√
4. Solid/Liquid/Gas Behavior	SLG-1	HI-SPEED CAMERA	1 OPERATIONAL CYCLE	2 HOURS	TEMPERATURE $\pm$1°K PRESSURE $\pm$2%	√	√
	SLG-2	PHOTOMETER	1 OPERATIONAL CYCLE	24 HOURS	TEMP 0.2°C TIME 0.1 SEC PRESS 0.1 psi	√	
	SLG-3	SOLAR COLLECTOR PHOTOCELL PYROMETER	EACH ORBIT	15 HOURS	TEMP $\pm$20°C TIME 1 SEC PRESS 1×10^{-1} torr		√
	SLG-4	CAMERAS	8 TEST CELLS 6-10 MINUTES EACH	2 HOURS	0.1 mm DROPLET DIAMETER		√
	SLG-5	PRESSURE TRANSDUCERS, THERMOCOUPLES, GAS CHROMATOGRAPH	1 OPERATIONAL CYCLE	35 HOURS	1 mg DROPLET MASS, TIME 0.1 SEC TEMP 1°C PRESS 0.1 psi	√	
5. Microorganisms	M-1	MICROSCOPE TV CAMERA	————	14 DAYS	MICRON RESOLUTION	√	
	M-2	SPECTROPHOTOMETER MICROSCOPE, GAS CHROMATOGRAPH	OBSERVATIONS EVERY 4-6 HOURS	3.5 DAYS	MICRON RESOLUTION	√	
	M-3	MICROSCOPE TV CAMERA	1 HOUR EVERY 2 DAYS	10 DAYS	100 MICRONS RESOLUTION	√	
	M-4	MICROSCOPE TV CAMERA	EVERY 2 DAYS	30 DAYS	10X MAGNIFICATION	√	
	M-5	PHOTOMETER	EVERY HOUR FOR 5 DAYS THEN ONCE EVERY 2 DAYS	15 DAYS	TYPICAL PHOTOMETRIC RESOLUTION	√	
6. Observations of the Earth's Atmosphere, the Space Environment, and Astronomical Phenomena	OEA-1	SPECTROMETERS DETECTORS	CONTINUOUS	3.5 DAYS	57 DEG RESOLUTION	√	
	OEA-2	ELECTRON GUN MAGNETOMETER	CONTINUOUS	3.1 DAYS (50 ORBITS)	POINTING $\pm$5 DEG	√	
	OEA-3	2 STAR TRACKERS INERTIAL PLATFORM	6 - 1 MIN. READINGS EACH 4 ORBITS	5 DAYS	ANGLE RESOLUTION 1 SEC POINTING 0.5 DEG EA. AXIS	√	
	OEA-4	NUCLEAR EMULSION PACKAGE	1 OPERATIONAL CYCLE	5 DAYS	HALF CONE ANGLE 40 DEG POINTING $\pm$1 DEG		√
	OEA-5	INTERFEROMETER TELESCOPE	20 MINUTES EVERY 2 DAYS	14 DAYS	5%	√	

5.0 DEFINITION OF ANCILLARY SYSTEMS

5.1 GENERAL

The definitions of experiment sensor requirements were used to determine the functional requirements to be specified for ancillary systems such as attitude control, data automation, communications, power, and thermal control. An effort was made in the case of each experiment to employ the most promising system concepts possible with state-of-the-art components. Experiment package configurations were designed to ensure design feasibility, to obtain the required physical characteristics, and to allow interpretation of the experiment package/launch vehicle interface.

Power requirements, sensor pointing requirements, and the necessity, in some cases, of physically recovering a portion of the experiment package influenced the definition of the physical characteristics of the overall experiment package. Other analyses have been made, and other requirements have been delineated for each experiment package to produce the total required outputs to be stored on the experiment characteristics library tape. The required outputs include information on such characteristics as mass, volume/geometry, environmental data, reliability, deployment requirements, development time, and cost.

The following sections contain definitions of the ancillary systems to be used in each experiment for (1) attitude control, (2) data automation, (3) communications, (4) electric power, and (5) thermal control.

5.2 ATTITUDE CONTROL

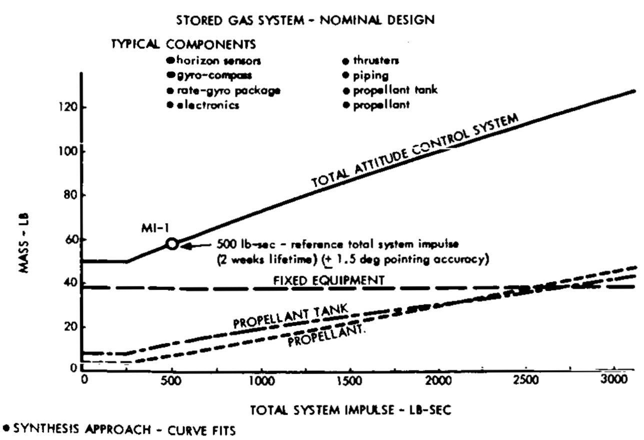


Figure 5-1 ATTITUDE CONTROL

To simplify the task of determining the physical characteristics of an attitude control system for each experiment in which stabilization is required, parametric design data were generated by the use of experiment MI-1 (Multispectral Surveillance of Earth) as a reference example. This reference experiment required attitude stabilization for a lifetime of 2 weeks with a pointing accuracy of ±1.5 degrees. Satisfaction of these requirements resulted in the choice of a 500-lb-sec impulse system having a mass of 58 pounds, a volume of 1565 cubic inches, and a power requirement of approximately 50 watts.

As shown in Figure 5-1, the mass of the system is a function of the required total system impulse which in turn depends on lifetime and pointing accuracy requirements. The mass of propellant and the propellant tank mass vary as a function of total system impulse. The mass of remaining items is constant because these items are essentially fixed equipment.

5.3 DATA AUTOMATION

The data automation system is used to provide the optimum interface between the sensors and the communications system. Four types of data are processed by this system: scientific and engineering, television, infrared, and photographic. The duty cycles of all data automation equipment are controlled by a programmer.

Scientific and engineering information is routed through an encoder which multiplexes, conditions, digitizes, and formats both digital and analog data. This information is then (1) placed on a magnetic tape recorder or (2) processed and compressed by a digital computer before being recorded. The data is then available for transfer to the communications system whenever station contact is accomplished.

Television coverage is provided by one or more TV cameras whose output is placed on a video tape recorder for replay through the communications system whenever a station is in sight. Infrared data from the IR scanners is routed directly to the video tape recorder for delayed readout.

The pictures from the cameras are received by the automatic film processors. After processing, individual frames are scanned by a flying-spot scanner, the output signals from which are placed on the video tape recorder. This information, like the television and infrared data, is then available for transmission whenever readout is desired.

TABLE 5-1
DATA AUTOMATION

Component	Mass (Lbs)	Volume (In^3)	Peak Pwr (W)
TAPE PROGRAMMER, 13 CHANNELS, 282 MINUTES	10	192	10
DIGITAL PROGRAMMER, 512 WORDS, 32-BITS EACH	15	320	12
ENCODER, 98 ANALOG, 2 DIGITAL CHANNELS	2	33	2
COMPUTER	20	896	50
ANALOG TAPE RECORDER	15	715	45
SMALL DIGITAL TAPE RECORDER	5	90	3
LARGE DIGITAL TAPE RECORDER	10	420	15
TV CAMERA	20	784	20
VIDEO TAPE RECORDER	96	8100	350
FULL FILM PROCESSOR	80	4563	1335
FLYING-SPOT SCANNER	45	1728	150
PARTIAL FILM PROCESSOR	40	432	1000
DATA RECOVERY CAPSULE	80	4936(18" DIA)	--

In some cases, video data may be transmitted in real time and the use of an on-board video tape recorder will not be required. Similarly, it may be desirable to recover film camera data physically by means of a re-entry capsule rather than use on-board film processing.

The physical characteristics of components which might be selected for use in a data automation system are listed in Table 5-1. Total system characteristics are determined by a summation of characteristics for the components selected to satisfy the requirements of individual experiments.

5.4 COMMUNICATIONS

The communications system used in conjunction with each experiment is design limited to meet only those requirements peculiar to that experiment. The UHF equipment includes a command receiver and a digital or an analog transmitter. The S-Band equipment includes a transmitter and a transponder. These units are considered to be standard, present-day, state-of-the-art equipment.

The communication capability has been divided into the functions to be performed. The information received from Earth will consist of commands to be used by the satellite programmer for starting and stopping the experiment and turning ancillary equipment on and off. Information to be relayed to Earth will consist of video, analog, and digital data obtained from the on-board sensors. The type of data will determine the number and type of output transmitters required and will vary with each experiment. The satellite construction and stability requirements affect the number of antennas needed on a specific experimental satellite. A line-of-sight capability of communication with the Earth station is to be maintained at all times. Flush-mounted antennas are to be considered whenever mission requirements will permit; however, turnstile-type antennas appear to offer a better omnidirectional type of coverage, and fewer antennas are required if this type is used.

TABLE 5-2
COMMUNICATIONS

Component	Mass (Lbs)	Volume (In.^{3})	Peak Pwr (W)
S-BAND TRANSPONDER	4.5	60	35
COMMAND RECEIVER	2.5	40	6
UHF CIRCULATOR	1.0	50	--
UHF ANALOG TRANSMITTER (PAM/FM/FM)	2.5	40	10
UHF DIGITAL TRANSMITTER (PCM/FM)	2.5	40	10
S-BAND TV TRANSMITTER	7.0	93	50
S-BAND ANTENNA SWITCH	0.8	17	--
S-BAND ANTENNAS (STABILIZED)	1.5	62	--
(UNSTABILIZED)	2.0	74	--
UHF ANTENNAS (STABILIZED)	1.0	62	--
(UNSTABILIZED)	1.5	74	--

Table 5-2 contains a list of the characteristics of each unit of the communications system. The mass is given in pounds, volume in cubic inches, and input power in watts. The final antenna design is considered to be a function of satisfying satellite physical and attitude control requirement; therefore, antenna size and type are selected only after mission requirements are completed. However, typical values are listed in the table.

The peak power values shown are typical of present-day state-of-the-art components. By 1970 it is anticipated that power requirements for an S-band transponder and an S-band TV transmitter will be 10 and 15 watts respectively.

5.5 ELECTRIC POWER

Candidate electric power sources suitable for the support of orbiting experiments include batteries, fuel cells, solar cells, and RTG (Radioisotope Thermal Electric Generator). Of these, batteries are the first choice because they are essentially off-the-shelf devices which are availalbe at a reasonable cost.

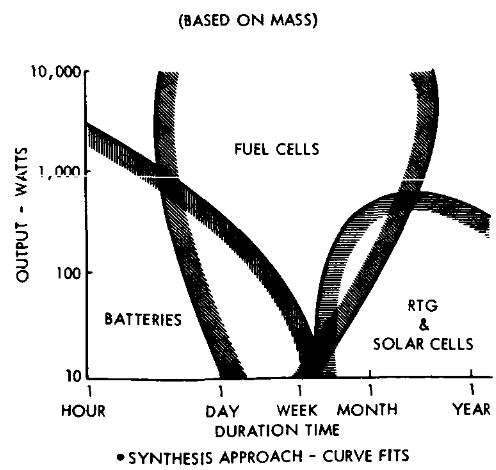


Figure 5-2 OPTIMUM POWER SOURCES

The well-known plot of power output versus duration of operation is shown in Figure 5-2. The data is presented in terms of regions of optimum power sources, i.e., minimum mass systems.

The power and the energy requirements of most of the secondary experiments analyzed in this study are such that they can be met by the use of batteries. Detailed analysis is needed in the case of experiments which cannot be powered by batteries. A notable example is the MI-3 experiment, Radar Surveillance of Earth.

For this experiment, a relatively high power requirement must be satisfied for one hour, but the demand is low at other times. Hence, consideration was given to using batteries in conjunction with (1) fuel cells or (2) RTG converters to satisfy average power requirements over the 24-hour period. Six power systems were analyzed in terms of mass as a function of days of opertion. It was found that the battery system is lighter for periods of operation up to 5 days. For periods from 5 to 25 days, the fuel cell is more advantageous. Beyond 25 days, the solar array and battery system is lighter.

5.6 THERMAL CONTROL

The proper operation of the individual experiments will be dependent upon the inclusion of an adequate thermal control system within the experiment package design. A thermal analysis of each experiment was performed in order to size the system in terms of mass, volume, and power requirements. The basic guidelines used to define the thermal control system were the following: (1) the system must not interfere with the primary payload, (2) the experiment must be self-contained, and (3) passive thermal control should be used if possible.

The selection of a thermal control concept for a particular experiment was based on a consideration of the thermal requirements for the experiment (allowable temperature range, heat dissipation), the physical characteristics of the experiment, the probable thermal environment, and the relationship between the stored and operating periods and the mission phases (prelaunch, launch, and orbit). Generally speaking, thermal control concepts may be categorized as passive, semi-active, or active. The use of active systems allows a greater degree of thermal control at the expense of reliability. Combinations of concepts, e.g., insulation used in conjunction with a heater and thermostat, may be used to advantage in certain situations.

After the thermal control concept was selected, the performance of the thermal control system was analyzed in relationship to the experiment thermal requirements and the probable thermal environment. Variations in the experiment thermal requirements and the thermal environment over the mission phases (prelaunch, launch, and orbit) were taken into account to ensure adequate performance throughout the mission.

5.7 ANCILLARY SYSTEMS MASS SUMMARY

A summary of the ancillary systems masses which are used in the 30 sample experiments considered in the study is presented in Table 5-3. The masses required for attitude control, data automation, communications, electric power, and thermal control systems are shown.

TABLE 5-3
ANCILLARY SYSTEMS MASS SUMMARY
MASS - POUNDS

	SDT-1	SDT-2	SDT-3	SDT-4	SDT-5	MS-1	MS-2	MS-3	MS-4	MS-5	MI-1	MI-2	MI-3	MI-4	MI-5
Attitude Control	118	☒	52	50	☒	95	*	*	*	*	58	58	124	96	124
Data Automation	29	2	18	24	☒	323	23	28	28	28	236	120	271	121	1023
Communications	12	7	12	12	☒	10	12	12	11	11	12	17	17	16	9
Electric Power	336	☒	222	211	☒	179	41	82	60	50	405	378	445	234	321
Thermal Control	24	3	5	10	2	2	1	1	3	2	8	36	36	19	20

	SLG-1	SLG-2	SLG-3	SLG-4	SLG-5	M-1	M-2	M-3	M-4	M-5	OEA-1	OEA-2	OEA-3	OEA-4	OEA-5
Attitude Control	*	*	42	*	*	☒	*	☒	☒	☒	*	50	43	50	58
Data Automation	110	29	329	100	19	10	12	20	10	29	36	13	19	101	29
Communications	4	12	9	☒	12	17	17	17	17	12	4	7	7	9	12
Electric Power	27	97	74	34	72	94	38	100	57	124	183	64	130	137	314
Thermal Control	5	3	16	9	3	18	1	6	7	4	1	5	4	4	28

* REMAINS ON-BOARD VEHICLE   ☒ NO REQUIREMENT

6.0 EXPERIMENT ENVIRONMENTAL REQUIREMENTS

6.1 GENERAL

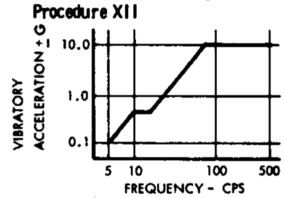
The following environments were investigated for applicability and significance to this program: thermal, acoustic, vibration, shock, acceleration, humidity, pressure/vacuum, electromagnetic, radiation, and contamination (dust, fungus, salt spray, etc.). Because this study was undertaken to develop a computer methodology, the environments that would not significantly contribute to this methodology development were eliminated from further consideration. In general, those environments were eliminated from consideration which were not peculiar to a specific launch vehicle or a specific location on the vehicle. The following environments were found to be significant to the development of the program: thermal, vibration, acoustic, and electromagnetic. The definition of requirements in the case of each of these environments is shown in Figure 6-1.

THERMAL

- Maximum and Minimum Temperature
- Heat Dissipation Rate
- Total Heat Dissipation

VIBRATION

- Assume Qualified Per Spec MIL-E-5272C Procedure XII



ACOUSTICS

- Noise Tolerance of 150 db Overall Per MIL Standard 810 Acoustical Test Method Grade B

ELECTROMAGNETIC

- Transmitter Signal Receiver Sensitivity

Figure 6-1 EXPERIMENT ENVIRONMENTAL REQUIREMENTS

TABLE 6-1
EXPERIMENT THERMAL ENVIRONMENT

EXPERIMENT	Prelaunch T_EMAX (°F)	Prelaunch T_EMIN (°F)	Prelaunch $\dot{Q}_E$ (BTU/HR)	Prelaunch Q_E (BTU)	Launch T_EMAX (°F)	Launch T_EMIN (°F)	Launch $\dot{Q}_E$ (BTU/HR)	Launch Q_E (BTU)	Orbit T_EMAX (°F)	Orbit T_EMIN (°F)	Orbit $\dot{Q}_E$ (BTU/HR)	Orbit Q_E (BTU)
Self-Contrained Experiments												
SDT-1	100	0	0	0	250	0	0	0		EJECTED		
SDT-2	100	0	0	0	250	0	0	0		EJECTED		
SDT-3	80	35	0	0	250	35	0	0		EJECTED		
SDT-4	80	0	0	0	250	0	0	0		EJECTED		
SDT-5	100	0	0	0	250	0	0	0		EJECTED		
MS-1	75	35	0	0	240	35	0	0		EJECTED		
MS-2	75	14	27.3	N/A	260	0	27.3	N/A	65	-50	225	N/A
MS-3	75	14	27.3	N/A	240	0	27.3	N/A	65	-50	232	N/A
MS-4	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	191	N/A
MS-5	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	198	N/A
MI-1	80	30	0	0	300	30	0	0		EJECTED		
MI-2	100	20	0	0	400	20	0	0		EJECTED		
MI-3	100	0	0	0	400	0	0	0		EJECTED		
MI-4	100	0	0	0	400	0	0	0		EJECTED		
MI-5	90	10	0	0	400	10	0	0		EJECTED		
SLG-1	90	20	0	0	250	20	0	0	0	-50	1970	N/A
SLG-2	212	32	0	0	350	32	0	0	75	0	150	N/A
SLG-3	75	14	0	0	300	0	0	0		EJECTED		
SLG-4	75	35	0	0	250	35	0	0	65	20	392	N/A
SLG-5	75	14	0	0	250	0	0	0	75	-50	239	N/A
M-1	100	25	0	0	250	25	0	0		EJECTED		
M-2	80	0	17.1	N/A	250	0	17.1	N/A	80	0	17.1	N/A
M-3	80	0	20	N/A	250	0	20	N/A		EJECTED		
M-4	85	0	3.5	N/A	200	0	3.5	N/A		EJECTED		
M-5	90	0	3.5	N/A	250	0	3.5	N/A		EJECTED		
OEA-1	75	25	0	0	250	25	0	0	60	-50	394	N/A
OEA-2	100	0	0	0	200	0	0	0		EJECTED		
OEA-3	80	35	0	0	250	35	0	0		EJECTED		
OEA-4	75	35	0	0	250	35	0	0		EJECTED		
OEA-5	100	0	0	0	250	0	0	0		EJECTED		
Vehicle-Dependent Experiments												
MS-2A	75	14	27.3	N/A	260	0	27.3	N/A	65	-50	202	N/A
MS-3A	75	14	27.3	N/A	240	0	27.3	N/A	65	-50	196	N/A
MS-4A	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	181	N/A
MS-5A	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	132	N/A
SLG-1A	90	20	0	0	250	20	0	0	0	-50	1960	N/A
SLG-2A	212	32	0	0	350	32	0	0	75	0	112	N/A
SLG-4A	75	35	0	0	250	35	0	0	65	20	392	N/A
SLG-5A	75	14	0	0	250	0	0	0	75	-50	172	N/A
M-2A	80	0	17.1	N/A	250	0	17.1	N/A	80	0	14	N/A
OEA-1A	75	25	0	0	250	25	0	0	60	-50	362	N/A

T_E - TIME-SPACE AVERAGED SINK TEMPERATURE
$\dot{Q}_E$ - HEAT DISSIPATION RATE
Q_E - TOTAL SHORT-PERIOD HEAT DISSIPATION

6.2 THERMAL

The thermal environment for each of the experiments was defined by three parameters:

1. Maximum and minimum allowable time-space averaged temperature
2. Heat dissipation rate
3. Total short period heat dissipation.

These parameters (refer to Table 6.1) were determined in the case of each of three mission phases: launch, prelaunch, and orbit. Those experiments that are ejected from the spacecraft must be compatible with an orbital operational environment not associated with the spacecraft, and consequently, no thermal compatibility checks will be made in the orbit mission phase for ejected experiments. The time-space averaged temperatures are the maximum and minimum temperatures to which the components of the experiment can be subjected without resulting in malfunctions.

6.3 VIBRATION

Because the experiment vibration tolerance is very difficult to determine by analysis, the only meaningful vibration tolerance levels are those levels to which the experiment components have been qualified by testing. Two types of specification can be used in describing the vibration tolerance: random and sinusoidal. Because many off-the-shelf components have not been qualified to the random vibration specification, only sinusoidal vibration levels were considered in the compatibility checks. A vibration tolerance level which applies to the majority of off-the-shelf components was assigned to all experiments in order to satisfy the requirement of the computer program. The maximum sinusoidal vibration level, indicated in Figure 6-1, is per MIL-E-5272C, Procedure XII.

6.4 ACOUSTICS

The same difficulty encountered in defining the vibration tolerance of the experiments is encountered in defining the acoustical noise tolerance. The tolerance levels assigned to experiments can only be as high as the levels to which the experiment components have been qualified by testing. Because off-the-shelf components were used whenever possible in the experiment definitions, a maximum noise tolerance of 150 db overall was assigned to all experiments. This value is per MIL Std 810 "Acoustical Test Method, Grade B".

6.5 ELECTROMAGNETIC

Electromagnetic compatibility can be defined as the ability of each component in an integrated system to perform its design function without interfering with the performance of the design function of any other component in the system. The following are the basic parameters which determine if one component will interfere with the function of another:

1. The level and bandwidth of the signal a component is capable of emitting (transmitter signal)

2. The level and bandwidth of the signal to which a component is capable of responding (receiver sensitivity)

3. "Coincident time interval" or the occurrence of simultaneous operation of components whose parameters, (1) and (2), overlap

4. Amount of isolation between components.

TABLE 6-2
ELECTROMAGNETIC ENVIRONMENT

Applicable Curves	Experiments									
	MS -2	MS -3	MS -4	MS -5	SLG -1	SLG -2	SLG -4	SLG -5	M -2	OEA -1
NARROWBAND SIGNAL #1	√	√	√	√	√	√		√		√
NARROWBAND SIGNAL #2									√	
BROADBAND SIGNAL #1	√	√	√	√		√		√	√	
BROADBAND SIGNAL #2					√		√			√
NARROWBAND SENSITIVITY	√	√	√	√	√	√		√	√	√
BROADBAND SENSITIVITY	√	√	√	√	√	√	√	√	√	√

In order to provide for an electromagnetic compatibility check between the experiments and the launch vehicle, the transmitter signal and the receiver sensitivity were defined for the selected experiments. Because of the isolation provided by the distance between the ejected experiments and the launch vehicle, only the ten experiments that remain aboard the launch vehicle were analyzed for the electromagnetic compatibility parameters. As shown in Table 6-2 and Figure 6-2, each experiment is described by a narrowband and a broadband transmitter signal and a narrowband and a broadband receiver sensitivity.

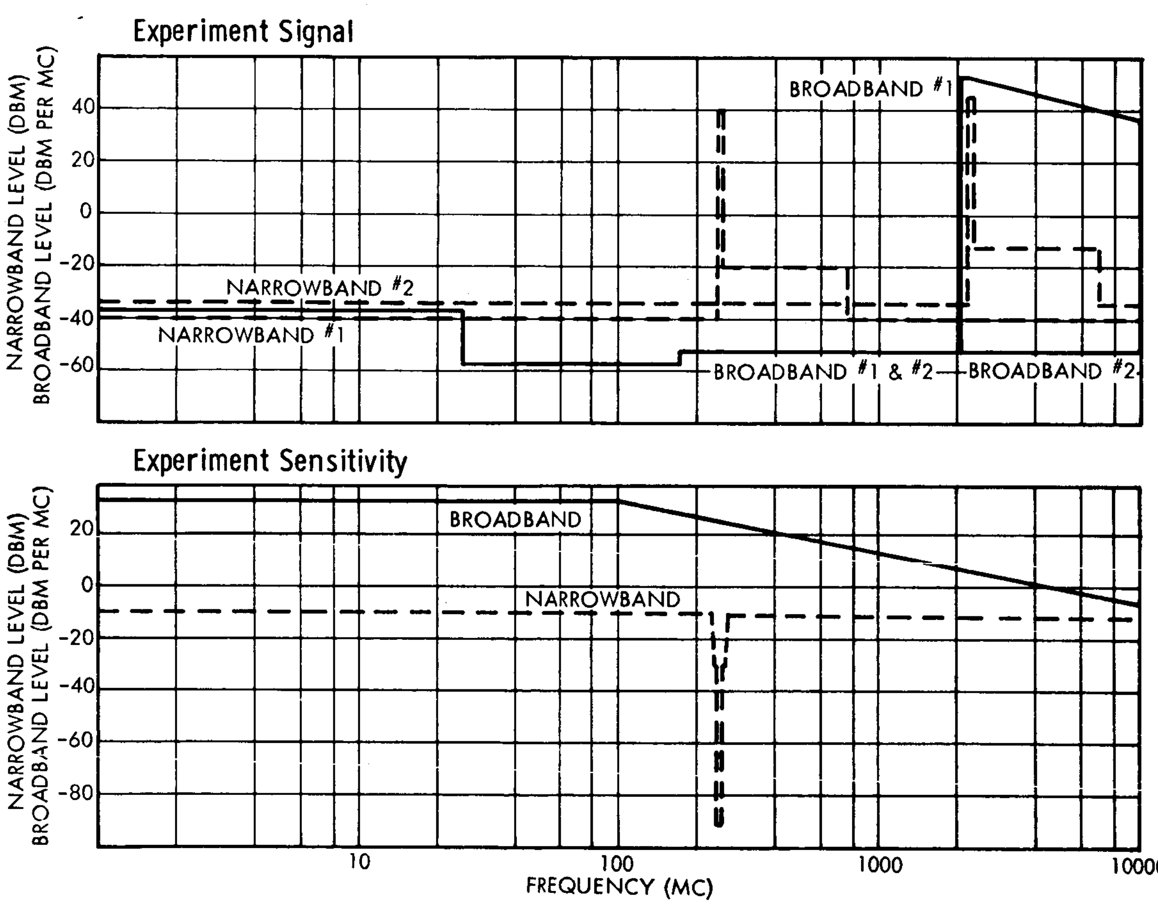


Figure 6-2 EXPERIMENT ELECTROMAGNETIC ENVIRONMENT

7.0 CONCEPTUAL DESIGN OF EXPERIMENTS

7.1 GENERAL

Experiment conceptual design drawings and mass and volume analyses were made for each of the thirty experiments selected for use in this study. In these drawings, the experiment package shape, the arrangement of components, the total package volume, the volume of the basic components, the total package mass, and the critical component shape are shown.

In establishing the configuration designs, primary emphasis was placed on self-contained experiment packages, that is, no consideration was given to the support capabilities of the vehicle or other experiments. However, to obtain a broader spectrum of data for use in the computer program checkout, the pertinent characteristics of certain vehicle-dependent experiments were also formulated. A vehicle-dependent experiment is defined as a self-contained experiment exclusive of power and communications systems and is indicated by an "A" after the basic experiment number. The experiments that were considered on both a self-contained and a vehicle-dependent basis are those ten experiments which remain aboard the launch vehicle and are not ejected as separate satellites. The mass and volume of these vehicle-dependent experiments are presented in Table 7-1.

TABLE 7-1
MASS AND VOLUME SUMMARY - VEHICLE-DEPENDENT EXPERIMENTS

EXPERIMENT	TOTAL INSTALLED MASS (LBS)	BASIC COMPONENT VOLUME (IN^3)	INSTALLED VOLUME (IN^3)
MS-2A	50	972	1,594
MS-3A	60	1,229	1,635
MS-4A	102	2,998	3,807
MS-5A	74	2,004	2,786
SLG-1A	160	6,914	12,722
SLG-2A	86	1,620	3,694
SLG-4A	167	4,220	9,031
SLG-5A	69	3,760	5,226
M-2A	119	2,488	5,349
OEA-1A	121	4,294	6,785

7.2 CONFIGURATION DESCRIPTION

The conceptual designs of the thirty self-contained experiments are summarized on the following pages by isometric drawings and mass and volume statements. The experiments that are shown with the Earth and sky in the background are designed to be ejected as separate satellites; those shown with no background are designed for operation aboard the launch vehicle.

7.2.1 Category I - Systems Development and Tests

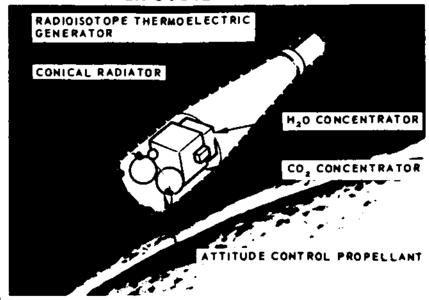
All of the experiments in Category I, Figure 7-1, are designed to be ejected as separate satellites. One experiment, the Energetic Particles Explorer, has already been developed, and therefore is considered a fixed hardware configuration. The experiment masses range from 58 pounds for SDT-2 to

Figure 7-1

CATEGORY I EXPERIMENT SUMMARY

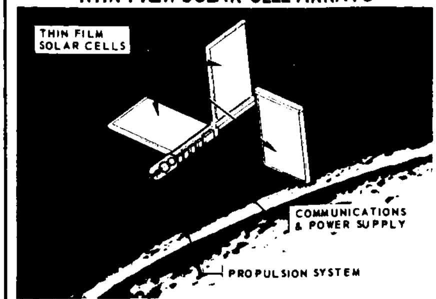
SYSTEMS DEVELOPMENT AND TEST

SDT-1
RADIOISOTOPE-THERMOELECTRIC POWER SYSTEM INTEGRATION



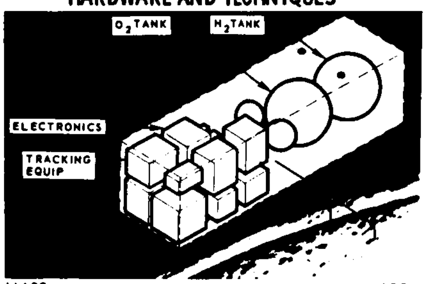
MASS	1031 LBS.
VOLUME (BASIC COMPONENT)	50,900 $IN.^3$
VOLUME (PACKAGED)	122,000 $IN.^3$

SDT-2
PERFORMANCE ASSESSMENT OF THIN FILM SOLAR CELL ARRAYS



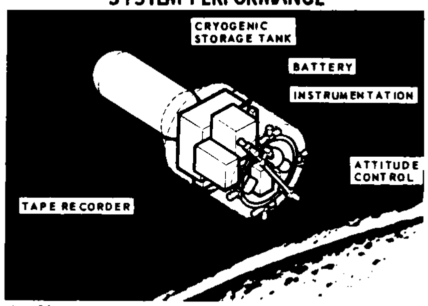
MASS	58 LBS.
VOLUME (BASIC COMPONENT)	2,504 $IN.^3$
VOLUME (PACKAGED)	3,689 $IN.^3$

SDT-3
PERFORMANCE ASSESSMENT OF SPACECRAFT NAVIGATION, GUIDANCE AND CONTROL HARDWARE AND TECHNIQUES



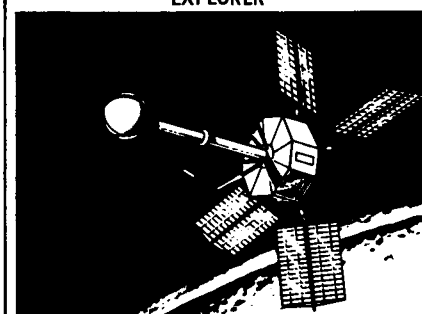
MASS	682 LBS.
VOLUME (BASIC COMPONENT)	25,803 $IN.^3$
VOLUME (PACKAGED)	72,900 $IN.^3$

SDT-4
CRYOGENIC PROPELLANT STORAGE SYSTEM PERFORMANCE



MASS	487 LBS.
VOLUME (BASIC COMPONENT)	22,465 $IN.^3$
VOLUME (PACKAGED)	47,595 $IN.^3$

SDT-5
ENERGETIC PARTICLES EXPLORER



MASS	440 LBS.
VOLUME (PACKAGED)	30,300 $IN.^3$

1,031 pounds for SDT-1. The required volumes for the same experiments are 3,689 and 122,000 cubic inches respectively.

7.2.2 Category II - Materials and Structures

Experiment MS-1, The Degradation of Organic Materials in Space Environment, is the only experiment in this category to be ejected as a separate satellite. It also has the largest mass, 795 pounds, and volume, 73,500 cubic inches. The remainder of the experiments are designed to remain aboard the launch vehicle. Their masses range from 103 to 173 pounds and volumes from 2,988 to 5,500 cubic inches. The Category II experiments are shown in Figure 7-2.

7.2.3 Category III - Multispectral Imagery of the Earth and Orbiting Objects

Because of the long life-time requirements, all of the experiments in Category III were designed as separate satellites, Figure 7-3. The Synoptic Earth Cartography experiment has the largest mass (2,812 pounds) and volume (469,800 cubic inches). The masses of the other experiments vary between 798 and 1,434 pounds, and the volumes vary between 39,468 and 139,000 cubic inches.

7.2.4 Category IV - Solid/Liquid/Gas Behavior

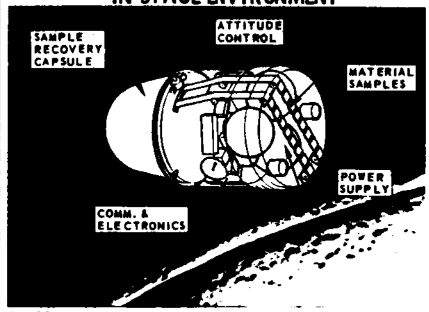
Only one experiment in Category IV, The Formation of Single Crystals, was designed to be ejected as a separate satellite. The remaining four are contained aboard the launch vehicle for the entire mission. Physical data recovery by a recoverable capsule is used in the ejected experiment and in two of the

Figure 7-2

CATEGORY II EXPERIMENT SUMMARY

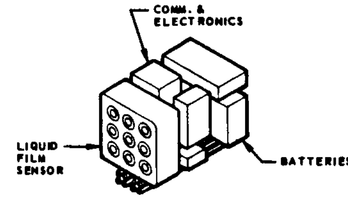
MATERIALS & STRUCTURES

MS-1 DEGRADATION OF ORGANIC MATERIALS IN SPACE ENVIRONMENT



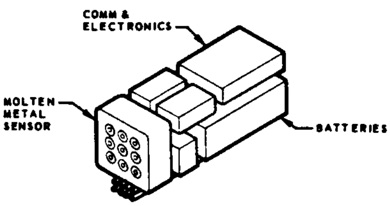
MASS	795 LBS.
VOLUME (BASIC COMPONENT)	46,200 $IN.^3$
VOLUME (PACKAGED)	73,500 $IN.^3$

MS-2 BEHAVIOR OF LIQUID FILMS IN SPACE ENVIRONMENT



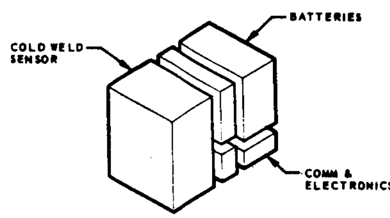
MASS	103 LBS.
VOLUME (BASIC COMPONENT)	1,827 $IN.^3$
VOLUME (PACKAGED)	2,988 $IN.^3$

MS-3 VAPORIZATION RATE OF MOLTEN METALS



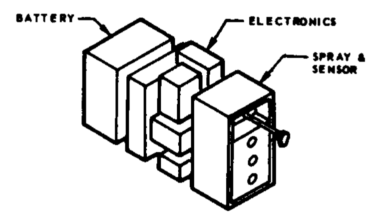
MASS	154 LBS.
VOLUME (BASIC COMPONENT)	2,809 $IN.^3$
VOLUME (PACKAGED)	3,750 $IN.^3$

MS-4 COLD WELDING OF METALS IN SPACE ENVIRONMENT



MASS	173 LBS.
VOLUME (BASIC COMPONENT)	4,344 $IN.^3$
VOLUME (PACKAGED)	5,500 $IN.^3$

MS-5 SPRAY COATING AND SURFACE CONTAMINATION



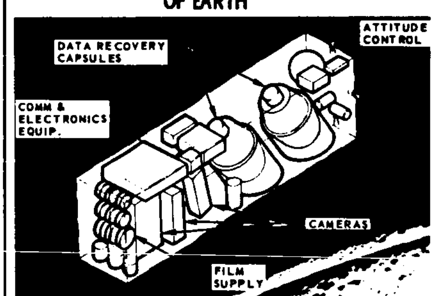
MASS	135 LBS.
VOLUME (BASIC COMPONENT)	3,350 $IN.^3$
VOLUME (PACKAGED)	4,650 $IN.^3$

Figure 7-3

CATEGORY III EXPERIMENT SUMMARY

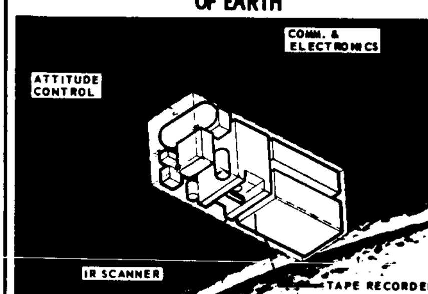
MULTISPECTRAL IMAGERY OF THE EARTH AND ORBITING OBJECTS

MI-1 MULTISPECTRAL SURVEILLANCE OF EARTH



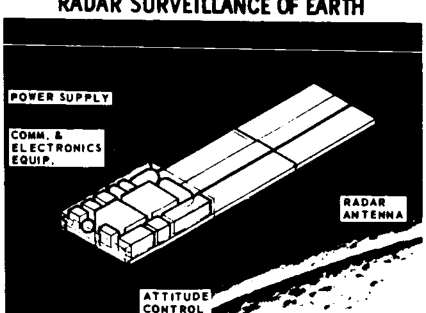
MASS	1,082 LBS.
VOLUME (BASIC COMPONENT)	31,670 $IN.^3$
VOLUME (PACKAGED)	81,200 $IN.^3$

MI-2 INFRARED LINE SCAN SURVEILLANCE OF EARTH



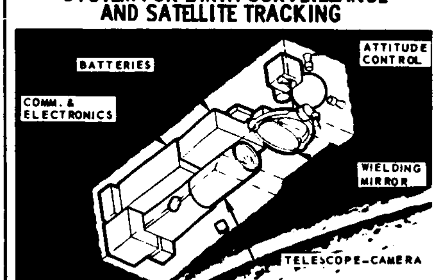
MASS	896 LBS.
VOLUME (BASIC COMPONENT)	21,000 $IN.^3$
VOLUME (PACKAGED)	39,468 $IN.^3$

MI-3 RADAR SURVEILLANCE OF EARTH



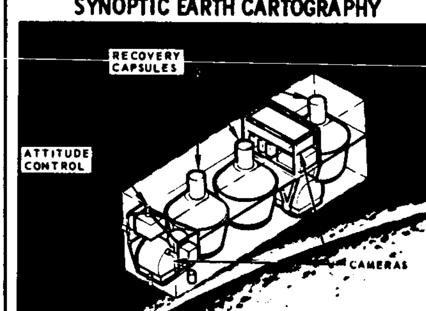
MASS	1,434 LBS.
VOLUME (BASIC COMPONENT)	98,000 $IN.^3$
VOLUME (PACKAGED)	139,000 $IN.^3$

MI-4 ELECTRONIC IMAGE MOTION STABILIZATION SYSTEM FOR EARTH SURVEILLANCE AND SATELLITE TRACKING



MASS	798 LBS.
VOLUME (BASIC COMPONENT)	27,110 $IN.^3$
VOLUME (PACKAGED)	79,083 $IN.^3$

MI-5 SYNOPTIC EARTH CARTOGRAPHY



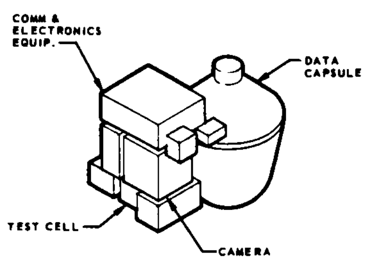
MASS	2,812 LBS.
VOLUME (BASIC COMPONENT)	200,903 $IN.^3$
VOLUME (PACKAGED)	469,800 $IN.^3$

contained experiments. The masses of the contained experiments vary from 153 to 201 pounds; the ejected experiment has a mass of 713 pounds. The volume requirement ranges from approximately 7,100 to 13,500 cubic inches for the contained experiments to 102,993 cubic inches for the ejected experiment. The Category IV experiments are shown in Figure 7-4.

Figure 7-4
CATEGORY IV EXPERIMENT SUMMARY

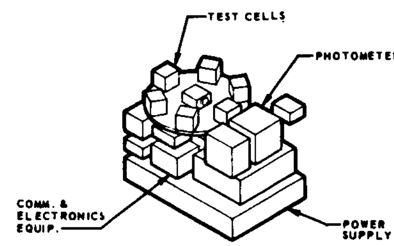
SOLID/LIQUID/GAS BEHAVIOR

SLG-1
BOILING IN ZERO-G ENVIRONMENT



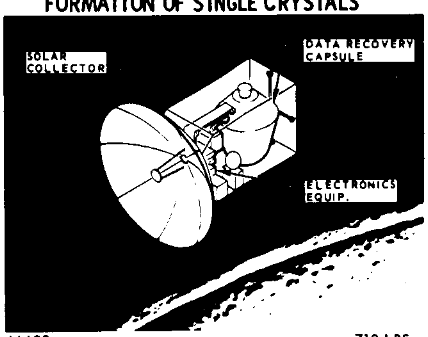
MASS	191 LBS.
VOLUME (BASIC COMPONENTS)	7,338 IN.^{3}
VOLUME (PACKAGED)	13,500 IN.^{3}

SLG-2
NUCLEATE CONDENSATION IN ZERO-GRAVITY



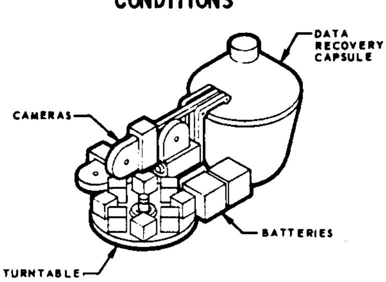
MASS	195 LBS.
VOLUME (BASIC COMPONENTS)	3,450 IN.^{3}
VOLUME (PACKAGED)	7,862 IN.^{3}

SLG-3
FORMATION OF SINGLE CRYSTALS



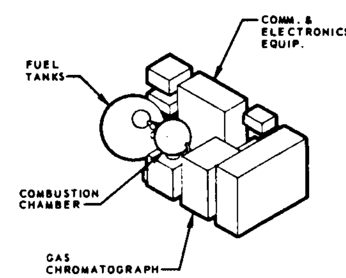
MASS	713 LBS.
VOLUME (BASIC COMPONENTS)	43,950 IN.^{3}
VOLUME (PACKAGED)	102,993 IN.^{3}

SLG-4
SEGREGATION OF IMMISCIBLE LIQUIDS UNDER ZERO GRAVITY CONDITIONS



MASS	201 LBS.
VOLUME (BASIC COMPONENTS)	4,720 IN.^{3}
VOLUME (PACKAGED)	10,090 IN.^{3}

SLG-5
ZERO-G COMBUSTION



MASS	153 LBS.
VOLUME (BASIC COMPONENTS)	5,130 IN.^{3}
VOLUME (PACKAGED)	7,100 IN.^{3}

7.2.5 Category V - Microorganisms

Four of the experimental payloads in Category V are designed to be ejected from the launch vehicle as separate orbiting satellites, Figure 7-5. Experiment M-2 is the only one designed to be contained in the launch vehicle. The mass requirements for this category are all between 135 pounds and 233 pounds. Volume requirements vary from 6,916 to 17,400 cubic inches.

7.2.6 Category VI - Observation of the Earth's Atmosphere, the Space Environment, and Astronomical Phenomena

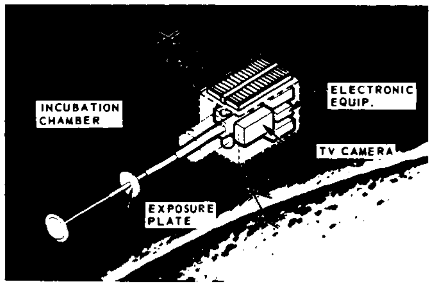
As a result of the duration requirements for Category VI, only one experiment, OEA-1, is designed to remain aboard the launch vehicle. The other four experiments are designed to be ejected as separate attitude-stabilized satellites for longer duration missions. Mass requirements for this category of experiments will vary from 308 to 691 pounds and the volume requirements from 11,600 to 31,870 cubic inches. Category VI experiments are shown in Figure 7-6.

Figure 7-5

CATEGORY V EXPERIMENT SUMMARY

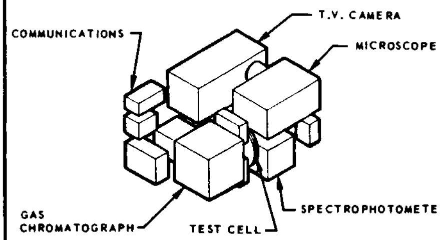
MICROORGANISMS

M-1
SOFT CAPTURE ENUMERATION AND IDENTIFICATION OF SPACE - BORNE MICROORGANISMS



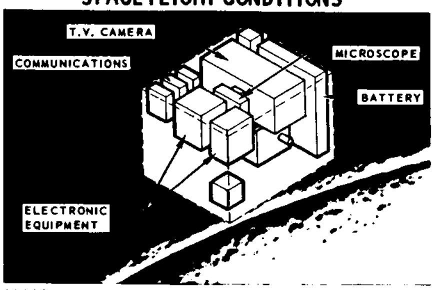
MASS	233 LBS.
VOLUME (BASIC COMPONENT)	7,427 IN.^{3}
VOLUME (PACKAGED)	17,400 IN.^{3}

M-2
EFFECTS OF SPACE FLIGHT ON MORPHOLOGY, GROWTH, AND LIQUID/GAS SEPARATION IN MICROORGANISMS



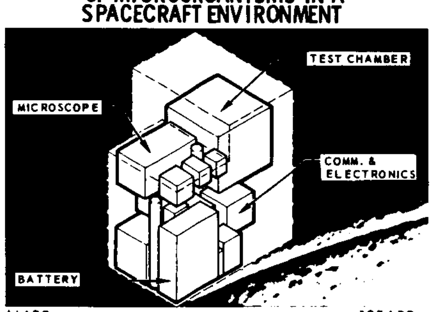
MASS	165 LBS.
VOLUME (BASIC COMPONENT)	3,218 IN.^{3}
VOLUME (PACKAGED)	6,916 IN.^{3}

M-3
GENETIC EFFECTS IN MICROORGANISMS AND MUTATION RATE IN EXTENDED SPACE FLIGHT CONDITIONS



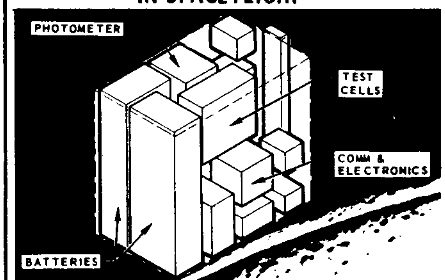
MASS	205 LBS.
VOLUME (BASIC COMPONENT)	3,500 IN.^{3}
VOLUME (PACKAGED)	8,700 IN.^{3}

M-4
DETERMINATION OF THE MIGRATION OF MICROORGANISMS IN A SPACECRAFT ENVIRONMENT



MASS	135 LBS.
VOLUME (BASIC COMPONENT)	4,117 IN.^{3}
VOLUME (PACKAGED)	13,696 IN.^{3}

M-5
PRODUCTION OF NUTRIENTS BY CERTAIN MICROORGANISMS WHILE IN SPACE FLIGHT



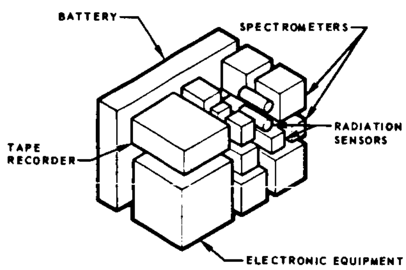
MASS	230 LBS.
VOLUME (BASIC COMPONENT)	4,299 IN.^{3}
VOLUME (PACKAGED)	8,757 IN.^{3}

Figure 7-6

CATEGORY VI EXPERIMENT SUMMARY

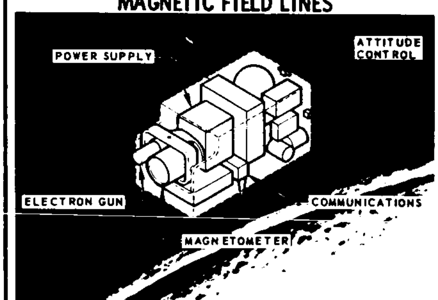
OBSERVATION OF THE EARTH'S ATMOSPHERE, THE SPACE ENVIRONMENT & ASTRONOMICAL PHENOMENA

OEA-1
RADIATION ENVIRONMENT MONITORING



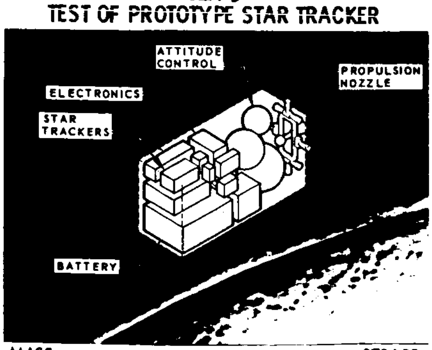
MASS	308 LBS.
VOLUME (BASIC COMPONENT)	7,333 IN.^{3}
VOLUME (PACKAGED)	11,600 IN.^{3}

OEA-2
STUDY OF MAGNETIC FIELD LINES



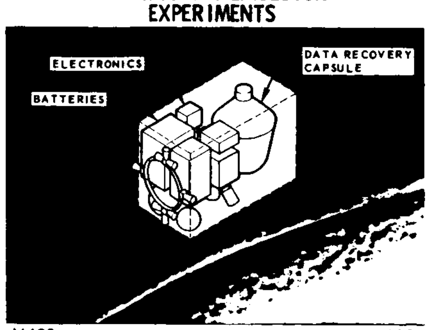
MASS	312 LBS.
VOLUME (BASIC COMPONENT)	7,700 IN.^{3}
VOLUME (PACKAGED)	19,600 IN.^{3}

OEA-3
TEST OF PROTOTYPE STAR TRACKER



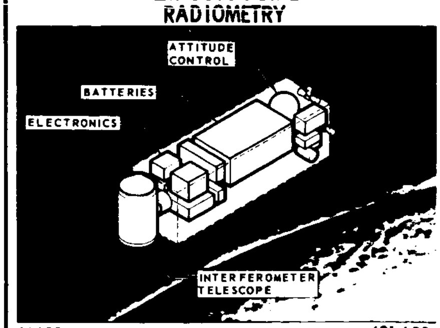
MASS	378 LBS.
VOLUME (BASIC COMPONENT)	6,838 IN.^{3}
VOLUME (PACKAGED)	17,512 IN.^{3}

OEA-4
COSMIC RAY EMULSION EXPERIMENTS



MASS	401 LBS.
VOLUME (BASIC COMPONENT)	8,818 IN.^{3}
VOLUME (PACKAGED)	22,140 IN.^{3}

OEA-5
EMISSION LINE RADIOMETRY



MASS	691 LBS.
VOLUME (BASIC COMPONENT)	12,456 IN.^{3}
VOLUME (PACKAGED)	31,870 IN.^{3}

8.0 RELIABILITY, DEVELOPMENT SCHEDULE, AND COST ANALYSES FOR EXPERIMENTS

8.1 RELIABILITY ANALYSIS

Preliminary predictions of reliabilities for the case of the selected experiments have been accomplished, and the results are presented in Table 8-1. The predicted reliabilities for these experiments range from 0.62 for SDT-1 to 0.98 for MS-2, MS-4, MS-5, SLG-1, and SLG-4.

TABLE 8-1
SUMMARY OF RELIABILITY, DEVELOPMENT SCHEDULE, AND COST ANALYSES

	SDT-1	SDT-2	SDT-3	SDT-4	SDT-5	MS-1	MS-2	MS-3	MS-4	MS-5	MI-1	MI-2	MI-3	MI-4	MI-5	SLG-1	SLG-2	SLG-3	SLG-4	SLG-5	M-1	M-2	M-3	M-4	M-5	OEA-1	OEA-2	OEA-3	OEA-4	OEA-5
Reliability	.62	.68	.92	.94	-	.83	.98	.96	.98	.98	.86	.84	.75	.89	.81	.98	.97	.95	.98	.96	.92	.96	.90	.90	.85	.87	.93	.91	.94	.93
Development Schedule - years	3.0	.75	3.0	1.5	-	1.5	.75	.75	.75	1.0	2.25	2.25	3.0	2.25	2.25	1.0	.75	1.5	1.0	1.0	1.0	.75	.75	.75	.75	.75	2.25	2.25	1.5	1.5
Cost - thousands of dollars	1764	288	1272	992	-	1201	448	545	631	659	1523	1808	3343	1523	1806	433	641	643	442	434	521	593	685	458	682	1532	1148	1067	993	978

The reliability of an experiment is defined as the probability that the experiment will be successfully performed. In this analysis, the partial completion of an experiment was not considered to be successful performance, e.g., if an experimenter specified that data were to be obtained by means of three sensors, all three sensors were assumed to be required for the successful performance of the experiment.

The predicted reliabilities for the case of each experiment were based on the following assumptions:

1. The equipment will be designed for protection against radiation, heat, and other known adverse space environment characteristics.

2. The equipment used in the experiment will be in a non-operative state during the boost phase.

3. The duration and operational phases of the experiments will be effected as suggested in each experiment write-up.

In the derivation of experiment reliabilities, space environment failure rates were used when they were available. In situations where no space-type failure rates were available, aircraft failure rates modified by an aircraft-to-space environmental factor were utilized.

8.2 DEVELOPMENT SCHEDULE ANALYSIS

It is expected that development time will have a decided influence on the final selection of the experiments which make up the Saturn IB/Apollo secondary payloads. Accordingly, a preliminary analysis of experiment development schedules was performed in this study. The resulting time span estimates have been incorporated into the SEPTER model and are available as outputs which may be used in screening the experiments on the basis of timing compatibility.

The technique employed in estimating the design/development spans may be summarized in two analytical steps. First, the experiments were ranked and categorized by considering the following in the case of each experiment:

1. The expected difficulty in designing the individual pieces of equipment

2. The overall complexity of the integrated experimental and ancillary equipment

3. The extensiveness of the implied test programs.

Secondly, span times were estimated for each of the experiment categories by relating the experiment equipment to comparable equipment with known development spans. The results are tabulated in Table 8-1.

8.3 COST ANALYSIS

Costs which have been established for each experiment package are summarized in Table 8-1. These costs vary from $288,000 for SDT-2 to $3,343,000 for MI-3. The cost analysis was performed in the following steps:

1. Cost categories were established.

2. Estimating procedures were determined.

3. A historical data collection was made.

4. Costs were calculated and totaled.

Cost categories were determined largely from weight statements. Some simplifications were made for the sake of consistency and comparability of the various systems. In addition, the costs of development and system integration were estimated. Ancillary subsystems were considered to be recurring cost categories while the development and integration costs were considered to be nonrecurring. The categories are summarized as follows:

1. Data Handling

2. Power

3. Structure

4. Communications

5. Attitude Control

6. Experiment

7. Development and Integration.

Estimating procedures were determined separately in the case of recurring costs and nonrecurring costs. Recurring costs were estimated on a cost per pound basis for cost categories 1 through 5. A separate factor was determined for each category. Category No. 6, Experiment, was estimated by the use of weighting factors applied to a baseline estimate. The weighting factors were a combination of factors based on the subjective analysis of the experiment description and the relative costs of various scientific instruments. Nonrecurring costs were estimated from a weighting factor applied to a subjective baseline. The weighting factor was a combination of three factors (complexity, data recovery, and state-of-the-art) which were determined through an analysis of the experiment description.

There were not enough historical data available to form the basis for a pure statistical analysis. However, some information was available for establishing general guidelines. Most important was the IBM study on Extended Apollo Earth-Orbit Flights. Budgetary information was also available in gross form on Explorer XIV, OAO, OGO, Syncom, Relay and Nimbus.

Costs were calculated as explained in the estimating procedures. It was assumed that three payloads would be built in order to satisfy the requirements of control experiments, system checkout, and the mission itself. No production learning was assumed.

9.0 E X P E R I M E N T D E S C R I P T I O N M E T H O D O L O G Y

9.1 GENERAL

The experiment descriptions required for the experimental payload characteristics library were developed during the experiment design effort. Those descriptions relating to experiment mass, volume, and environment were used directly as inputs to the library. However, it was necessary to develop a specialized methodology for representing experiment geometry and deployment requirements in the library.

9.2 GEOMETRY

In order to develop a methodology for representing experiment geometry, two classes of experiments, fixed geometry and amorphous geometry, were defined. The fixed-geometry experiments are defined in terms of finalized designs whose geometry cannot be modified. The amorphous-geometry experiments are those in which the configuration is not fixed and which are amenable to numerous design concepts. The shape and volume of the fixed-geometry experiments are represented by the standard shape envelope which will most efficiently contain the entire experiment as illustrated in Figure 9-1. The standard shapes selected for this representation are the sphere, the cylinder, and the rectangular parallelepiped. The shape of the amorphous-geometry experiments is represented by the standard shape envelope which will contain the experiment critical component as shown in Figure 9-2. The critical component is an envelope of such size and shape that it will contain, in turn, each of the undistortable component in the experiment package. The critical component, then, can be either the largest undistortable component in the experiment or a composite of several undistortable components. The volume of the amorphous-geometry experiments is represented by the minimum volume required for practical.

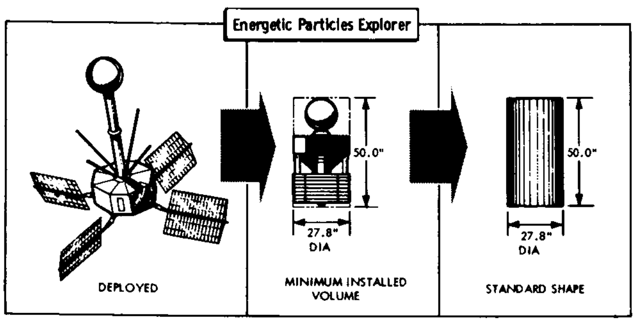


Figure 9-1 REPRESENTATION OF FIXED-GEOMETRY EXPERIMENTS

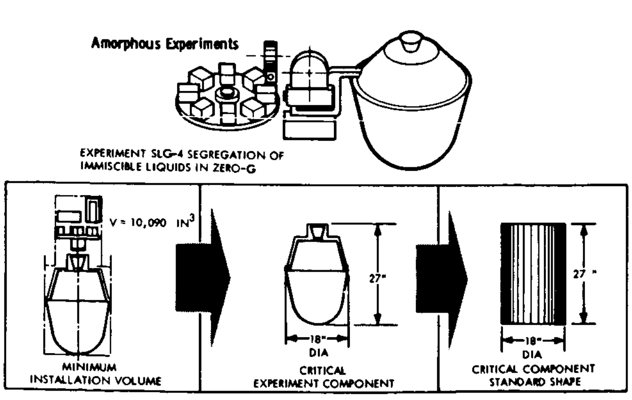


Figure 9-2 REPRESENTATION OF AMORPHOUS-GEOMETRY EXPERIMENTS

9.3 DEPLOYMENT REQUIREMENT

Deployment requirements are those experiment requirements which are contingent on proper installation of the experiment relative to the launch vehicle. These requirements have been defined by six deployment modes, Mode 0 through Mode 5, as shown in Figure 9-3. The modes were devised so that it is necessary to describe each experiment by only one deployment mode, Table 9-1.

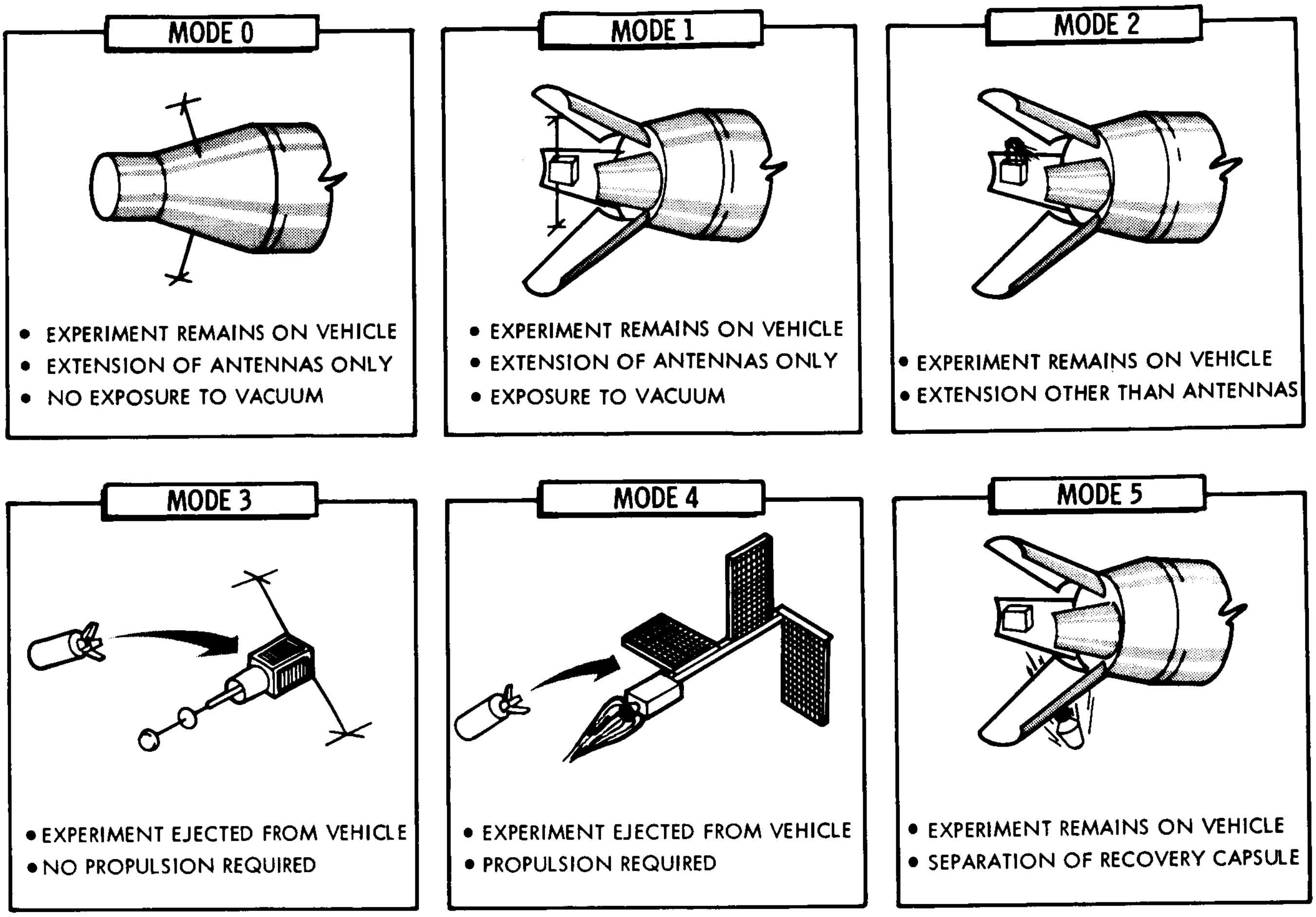


Figure 9-3 DEPLOYMENT MODE DEFINITION

TABLE 9-1
EXPERIMENT DEPLOYMENT MODES

Experiment	Mode	Experiment	Mode
SDT-1	3	SLG-1 & -1A	5
SDT-2	4	SLG-2 & -2A	0
SDT-3	3	SLG-3	3
SDT-4	3	SLG-4 & -4A	5
SDT-5	4	SLG-5 & -5A	0
MS-1	3	M-1	3
MS-2 & -2A	1	M-2 & -2A	0
MS-3 & -3A	1	M-3	3
MS-4 & -4A	1	M-4	3
MS-5 & -5A	2	M-5	3
MI-1	3	OEA-1 & -1A	2
MI-2	3	OEA-2	3
MI-3	3	OEA-3	4
MI-4	3	OEA-4	3
MI-5	3	OEA-5	3

10.0 PROGRAM DESIGN METHODOLOGY

10.1 GENERAL

A computer program for determining the limited physical characteristics of arbitrary experiments was developed as a support program for computer Program SEPTER. This support program DESIGN, will provide gross inputs to SEPTER rather than detailed design data on experimental payloads. DESIGN replaces the manual system synthesis tasks of the manual synthesis technique for designing experimental payloads and provides "first-pass" estimates of the mass and volume of the complete experimental payload.

The methodology used in DESIGN was formulated on the basis of parametric data and design experience obtained during the initial phase of the study. The output characteristics of ancillary systems have been defined to satisfy given experiment input requirements. These characteristics are defined by means of analytical representations either in the form of curve fits (equations) or characteristics stipulated for required components.

10.2 CONSTRUCTION AND OPERATION

The overall construction and operation of Program DESIGN is presented in Figure 10-1. The input to the program includes (1) the experiment sensor(s) requirements, (2) the experiment operational requirements, and (3) the systems functional requirements. The primary output of the program includes data on the mass and volume of each of the systems and the mass and volume of the experimental payload package. Pertinent power/energy characteristics are also included in the output.

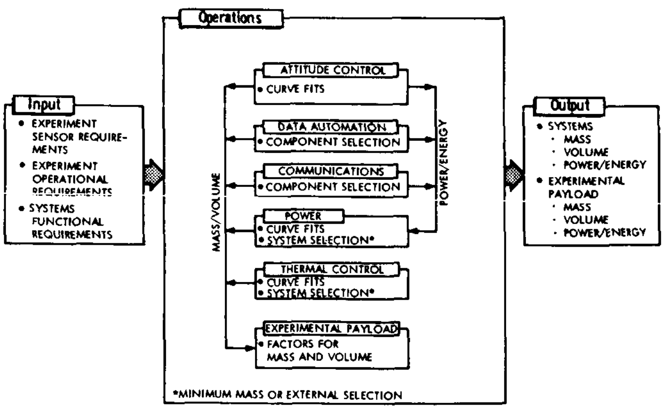


Figure 10-1 COMPUTER PROGRAM DESIGN CONSTRUCTION AND OPERATION

The input to the program consists of (1) coded answers to a number of yes-or-no questions concerning the experiment and (2) numerical data relative to the experiment. The experiment sensor(s) mass, volume, and power requirements are defined external to the program and are provided as a direct input to the program. The operational requirements are defined in terms of the duration of the mission and whether or not the experimental payload is deployed (ejected from the vehicle). The systems functional requirements are defined by numerical data (e.g., sensor pointing accuracy and number of TV cameras) and questions relative to each of the systems.

The program utilizes the input data to select the systems and to define the characteristics of those systems that are required. If stabilization of an experimental payload is required, an attitude control system is sized on the basis of the mission duration and sensor pointing accuracy requirements. If a data automation system is required, a series of coded answers to yes-or-no questions are utilized to select the required components for this system. If a communication system is required, the components of this system are determined by the same method that is used for the data automation system.

```
                    D E S I G N

                 EXPERIMENT SDT- 4


ATTITUDE CONTROL SYSTEM SYNTHESIS

   COMPONENTS                          MASS      VOL
                                       (LB)    (CU.IN.)

      FIXED EQUIPMENT                  38.0     660.0
      PROPELLANT                        3.8
      PROPELLANT TANKS                  7.9     454.7

                     TOTALS            49.7    1114.7


DATA AUTOMATION SYSTEM SYNTHESIS

   COMPONENTS                          MASS      VOL      POWER
                                       (LB)    (CU.IN.)  (WATTS)

      ENCODER                           2.0      33.0       2.0
      ANALOG TAPE RECORDER             15.0     715.0      45.0

                     TOTALS            17.0     748.0      47.0

COMMUNICATIONS SYSTEM SYNTHESIS

   COMPONENTS                          MASS      VOL      POWER
                                       (LB)    (CU.IN.)  (WATTS)

      S-BAND TRANSPONDER                4.5      60.0      35.0
      COMMAND RECEIVER                  2.5      40.0       6.0
      UHF CIRCULATOR                    1.0      50.0       0.
      UHF DIGITAL TRANSMITTER           2.5      40.0      10.0
      S-BAND ANTENNAS-  STABILIZED      1.5      62.0       0.
      UHF ANTENNAS-  STABILIZED         1.0      62.0       0.

                     TOTALS            13.0     314.0      51.0


POWER SYSTEM SYNTHESIS

            MISSION DURATION(DAYS)=  4.0

                                       PEAK       DAILY        TOTAL
   SYSTEMS                             POWER      ENERGY       ENERGY
                                      (WATTS) (WATT-HR/DAY)(WATT-HR)

      EXPERIMENT SENSORS                6.0      144.0        576.0
      ATTITUDE CONTROL                 50.0     1200.0       4800.0
      DATA AUTOMATION                  47.0     1128.0       4512.0
      COMMUNICATIONS                   51.0      816.0       3264.0
      THERMAL CONTROL                   0.         0.           0.

   TOTAL POWER SYSTEM REQMT.          154.0     3288.0      13152.0

          TYPE OF POWER SYSTEM-BATTERY

          MASS OF POWER SYSTEM (LB)                  232.7

          VOLUME OF POWER SYSTEM (CU.IN.)           3926.4

          EXPOSED AREA OF POWER SYSTEM (SQ.IN.)      N/A

THERMAL CONTROL SYSTEM SYNTHESIS

          TYPE OF SYSTEM-PASSIVE

    HEAT DISSIPATION PARAMETER (0.014 WATTS/SQ.IN.) IS LESS THAN
    CRITICAL VALUE (0.062 WATTS/SQ.IN.)

    MASS AND VOLUME ASSUMED TO BE NEGLIGIBLE


EXPERIMENTAL PAYLOAD SUMMARY

    MASS SUMMARY

            SENSORS (LB)                               84.0
            SYSTEMS (LB)                              312.4
            STRUCTURES (LB)                            59.5
            EJECTION EQUIPMENT (LB)                    45.6

        TOTAL MASS OF EXPERIMENT (LB)                 501.4

    VOLUME SUMMARY

            SENSORS (CU.IN.)                        15147.0
            SYSTEMS (CU.IN.)                         6103.1

          TOTAL-BASIC COMPONENTS (CU.IN.)           21250.1

        TOTAL VOLUME OF EXPERIMENT (CU.IN.)         63750.3
```

Figure 10-2 PROGRAM DESIGN - EXAMPLE OUTPUT DATA

The program utilizes the mission duration and the pertinent power/energy requirements of the various systems and sensors to determine the overall power/energy requirements of the experimental payload package. These overall power/energy requirements are computed by optionally specifying (1) continuous operations for the mission duration, (2) duty cycles (hours of operation per day), or (3) average daily energy requirements. If an attitude control system is required, it is assumed that it will operate continuously at peak power. The selection of the type of power system can be specified as input or selected by the program on the basis of minimum mass. The thermal control system is assumed to be a cooling system, and the cooling requirements are evaluated on the basis of the total thermal inertia of the experimental payload mass and the radiative heat rejection capability of the experimental payload envelope area. If these evaluations indicate that an active system is required, the selection of a system is based on minimum mass. If a passive system is indicated, the mass and volume requirements are assumed to be negligible (within the overall accuracy of the program).

The approximate total mass of the experimental payload is determined by summing the masses of the systems and sensors and applying a factor (based on manual design techniques and experience) to the sum to determine the structural mass and ejection equipment mass (if deployment is required). The approximate total volume of the experimental payload is determined by summing the volumes of the systems and multiplying this sum by another factor based on manual design experience. One value is applied for onboard experimental payloads, and a different value is applied for deployed payloads. An example of the output from Program DESIGN is given in Figure 10-2.

11.0 DEFINITION AND ANALYSIS OF MISSION CHARACTERISTICS

11.1 GENERAL

A preliminary task in the overall development of Program SEPTER was to define and analyze the pertinent mission characteristics of individual Saturn missions. This preliminary task had to be accomplished in order to (1) obtain representative data for the Mission/Vehicle/Primary Payload Characteristics Library, and (2) to formulate the program logic for representative deployment modes and the calculation of orbital elements and mission parameters for any deployment mode and deployment time.

11.2 SATURN MISSION TYPES

Initially, a survey was made to determine which mission types to consider for definition and analysis. The Saturn IB/Apollo vehicle/payload was used as the basic configuration for the study, and missions compatible with this configuration were investigated in particular. These missions consist primarily of the Earth-orbital, low-altitude, low-inclination type. Representative launch trajectory data for this mission type are included in the Mission/Vehicle/Primary Payload Characteristics Library of the program.

Other mission types, such as suborbital missions, were investigated for inclusion. Although missions of this type are not precluded by the program, their use for experimental payloads is considered to be limited because of (1) the short time duration of these missions, and (2) mission and vehicle constraints on experimental payload ejection.

11.3 PERTINENT MISSION CHARACTERISTICS

The mission characteristics that were found to be pertinent to the overall development of the computer program may be categorized as data of the following types: (1) trajectory parameters, (2) sequence-of-events, and (3) experimental payload possible deployment modes.

11.3.1 Trajectory Parameters

Time histories of the trajectory parameters of a typical Saturn IB/Apollo launch trajectory were obtained from nominal trajectory data for operational vehicle SA-207. Time histories of the parameters which are used to define the vehicle's Earth-relative position (latitude, longitude, and altitude) and its inertial velocity vector (velocity magnitude, flight path angle, and azimuth angle) are given in Figure 11-1. These six position and velocity parameters completely specify the vehicle's orbital elements at a given time. They are

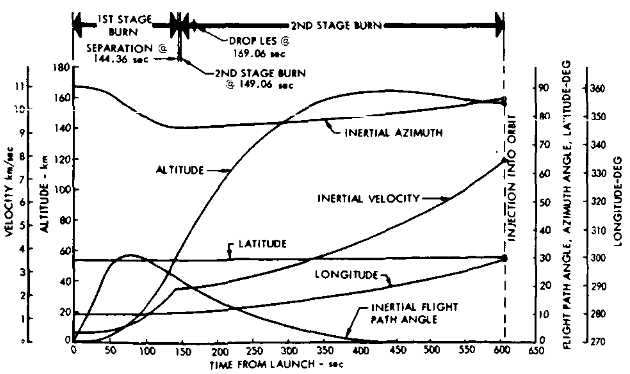


Figure 11-1 SATURN IB TYPICAL LAUNCH TRAJECTORY TIME HISTORY

also used for the determination of the orbital elements of an experimental payload for any deployment mode and deployment time.

11.3.2 Sequence-of-Events

Mission sequence-of-events data are required to identify (1) experimental payload deployment opportunities/constraints and possible modes, and (2) physical environments to which the experiments are subjected during various mission phases. The staging, jettisoning of hardware, separation of the payload from the vehicle, and the separation, transposition, and docking maneuvers of payload components are typical events which must be defined as a function of time in the mission.

Typical sequence-of-events data prior to injection of the primary payload are given along with the launch trajectory data in Figure 11-1. Representative sequence-of-events data subsequent to primary payload injection into orbit are given in Table 11-1 for a manned Apollo development mission. These data are approximate and were formulated primarily to provide representative data for the Mission/Vehicle/Primary Payload Characteristics Library.

TABLE 11-1
TYPICAL SEQUENCE-OF-EVENTS AFTER INJECTION INTO ORBIT

REFERENCE ORBIT: APOGEE - 215.8KM PERIGEE - 155.8KM INCLINATION - 30°

Orbital Operation	Time Interval - Min.	Time From Launch - Min.
• INJECTION OF PRIMARY PAYLOAD INTO ORBIT		10.8
• ROLL, PITCH, ETC. MANEUVERS	73.3	
• BEGIN TRANSPOSITION AND DOCKING MANEUVERS		84.1
INITIATE SEPARATION DEPLOY ADAPTER TRANSLATE CSM FORWARD PITCH CSM 180° TRANSLATE CSM BACK COMPLETE SOFT DOCKING	10	
• COMPLETE DOCKING MANEUVER		94.1
• COAST TO S-IVB JETTISON	3	97.1
• JETTISON S-IVB/IU/SLA	2	99.1
• COMPLETE DOCKING AND JETTISONING	1	100.1
• DEPLOYMENT OF EXPERIMENTAL PAYLOADS POSSIBLE	1	101.1

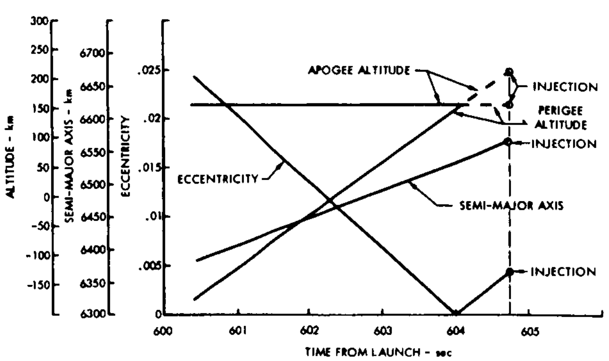


Figure 11-2 VARIATION OF ORBITAL ELEMENTS ALONG SATURN IB TYPICAL LAUNCH TRAJECTORY

11.3.3 Experiment Deployment

Mission characteristics defined and analyzed for the computer program included mission imposed deployment opportunities/constraints and possible deployment modes as a function of time in the mission. A secondary analysis was conducted to determine the effects of applying small propulsive velocity increments to the experimental payload at deployment time (for experimental payloads which require ejection).

A typical Saturn IB launch trajectory was analyzed in order to determine the feasibility and the limits of application of modes of experimental payload deployment to be used during the launch into orbit (up to primary payload injection). In Figure 11-2, the variation and sensitivity characteristics of some primary orbital elements are shown along a typical low-altitude, low inclination orbital launch trajectory as injection of the primary payload into its orbit is approached. In this example, the time during which an experimental payload could be non-propulsively ejected and attain an individual orbit is limited to approximately two seconds prior to the injection of the primary payload. The extreme sensitivity of perigee altitude to time before injection indicates that this mode of deployment (ejection without propulsion) is probably not desirable during the launch phase for this type of mission. In addition, for the Saturn/Apollo configuration, experimental payload locations defined in this study are not accessible for ejection until after the separation of vehicle/payload components.

During the primary mission orbital coast phase, experimental payloads may be deployed in various modes so that the maximum data acquisition objectives can be achieved. For many experiments, the orbit achieved by the primary payload may be adequate. Two basic modes of deployment are optional in this case, depending upon the physical environment required for the experiment: (1) the experimental payload remains fixed to the vehicle, and (2) the experimental payload is physically separated from the vehicle by some mechanism, e.g., a spring, which does not appreciably affect the orbital elements at the time of deployment.

For some experiments, the orbit achieved by the primary payload may be incompatible with the data acquisition objectives set for the experiment. The logical mode of deployment in this case would be one in which propulsion is applied to the experimental payload in order to attain a more compatible orbit.

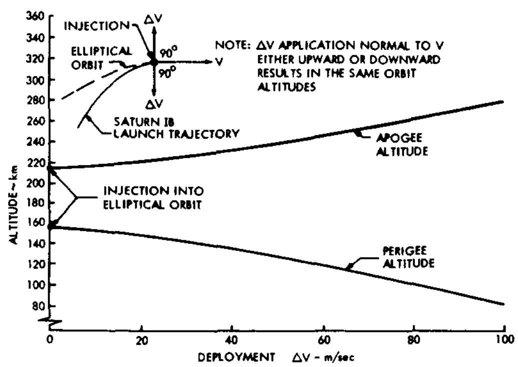


Figure 11-3 EFFECT OF EXAMPLE ΔV APPLICATION ON APOGEE AND PERIGEE ALTITUDES AT INJECTION OF PRIMARY PAYLOAD

A limited investigation was conducted to illustrate the effects of applying small impulsive velocity increments (ΔV's) to an experimental payload at deployment. Figure 11-3 shows the results relative to apogee and perigee altitudes when ΔV's are applied normal to the injection velocity vector in the plane of the orbit. The time of ΔV application was assumed to be at the instant of injection of the primary payload into its slightly elliptical orbit. The results show that an appreciable decrease in perigee altitude and increase in apogee altitude can be achieved with small ΔV's.

The data given in Figure 11-4 illustrate the major effects of applying small ΔV's in both in-plane and out-of-plane orthogonal directions to an experimental payload for deployment at some specified time in the orbital phase. The sketch in the upper right in Figure 11-4 defines the ΔV coordinate system. The major effects are summarized as follows:

1. Tangential ΔV affects the value of the semi-major axis and the eccentricity of the orbit. Semi-major axis changes are independent of the point of ΔV application in the orbit (i.e., time of deployment), but eccentricity is a function of the time of deployment (for an elliptical orbit).

2. Normal ΔV affects the value or orbit eccentricity. The effect is a function of deployment time.

3. Lateral ΔV affects the orbit inclination and the longitude of the ascending node. The effects are dependent on deployment time. Maximum change of inclination occurs at the nodes, and maximum nodal shift occurs at the maximum latitude point. No change of inclination occurs at maximum latitude, and no nodal shift occurs at the nodal point. As the point of ΔV application is moved from the node toward maximum latitude, there is less change in the inclination and more change in the longitude of the ascending node.

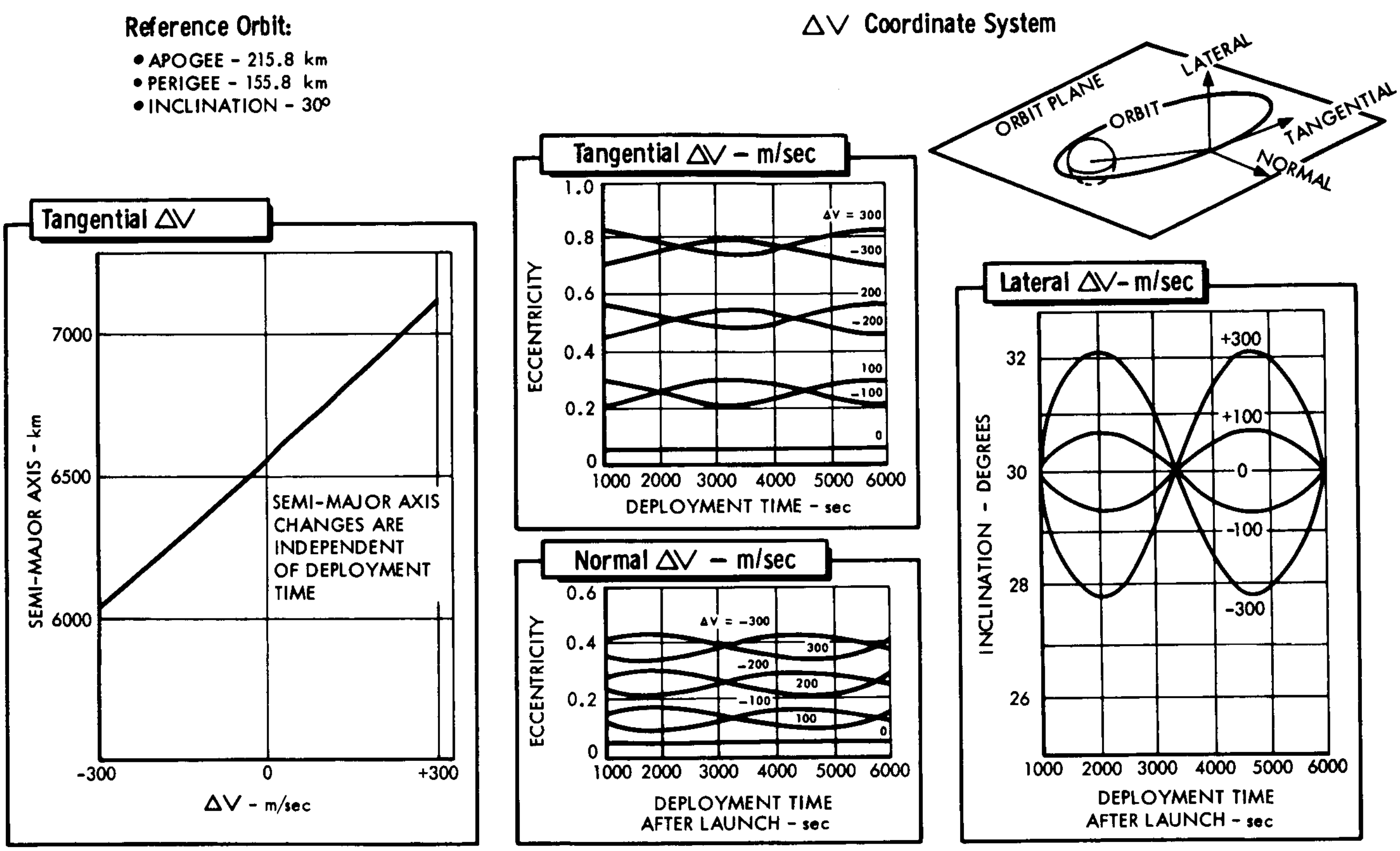


Figure 11-4 EFFECTS OF DEPLOYMENT ΔV COMPONENTS ON ORBITAL ELEMENTS

12.0 EXPERIMENT CAVITY DESCRIPTION METHODOLOGY

12.1 GENERAL

The baseline configuration selected for use in this study is the Saturn IB/Apollo with Command Module, Service Module, and Lunar Excursion Module. The areas considered in this study as potential experiment locations are (1) the LEM adapter fairing, (2) the Instrument Unit, and (3) the S-IVB Stage forward skirt. To simplify the descriptions of the location, these areas are subdivided into seven zones as shown in Figure 12-1. The zones are numbered in sequence beginning with the zone nearest the Service Module and progressing aft to the S-IVB Stage. Separate zones are provided for the Instrument Unit and the S-IVB cold panels.

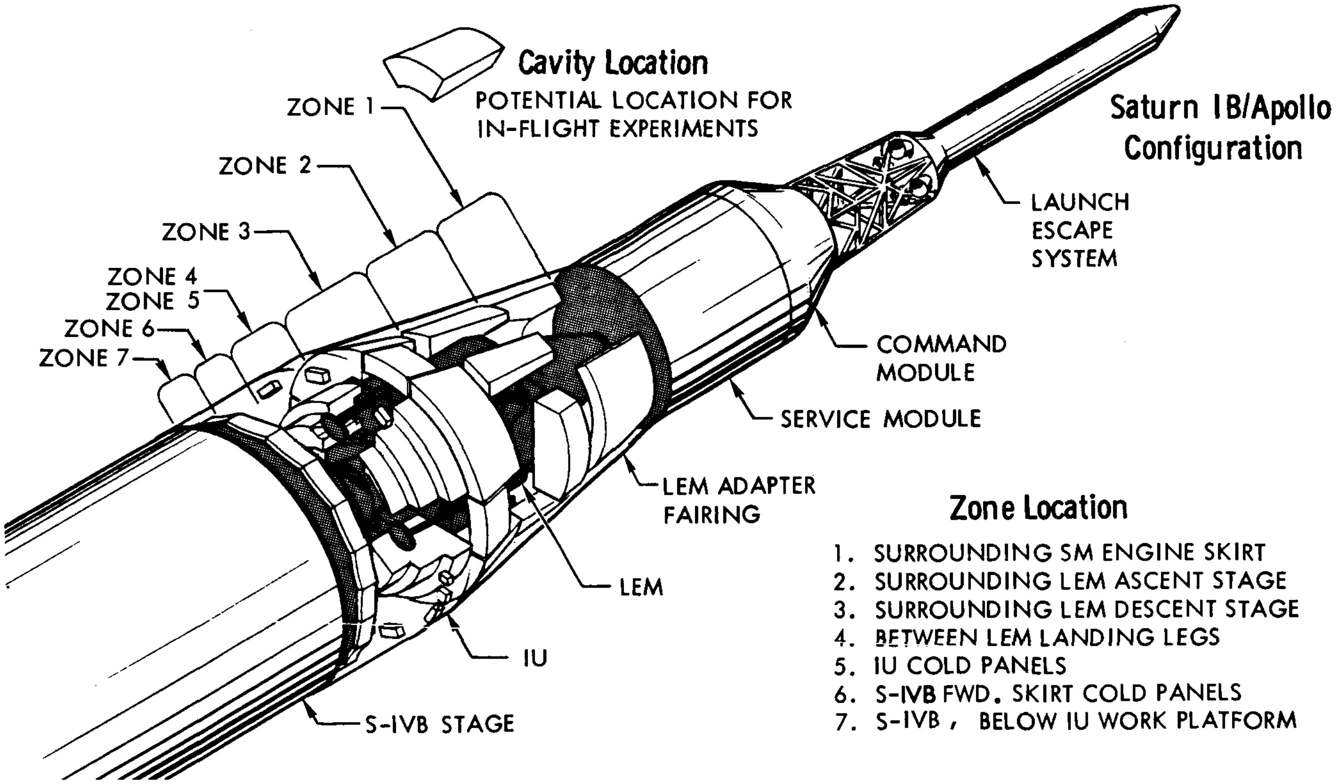


Figure 12-1 IN-FLIGHT EXPERIMENT CAVITY LOCATIONS

Each zone contains several individual cavitites which are the potential experiment locations. The zones were divided in such a way that the cavities contained in each zone would be similar in terms of location, environment, and installation requirements. The cavity shapes were obtained by providing the following clearances from the Saturn IB/Apollo vehicle:

1. A six-inch clearance was allowed between each cavity and the LEM, Service Module, S-IVB Tank, and all Work Platforms to provide adequate space for installation and maintenance of the experiment package.

2. A three-inch clearance was provided between each cavity and the LEM Adapter Fairing to provide for mounting structure.

3. Adequate clearance was provided for extraction of the LEM from the Adapter Fairing during an orbital mission.

4. Direct attachment to the cold panels in the Instrument Unit and the S-IVB Stage forward skirt was assumed.

5. Clearances from existing components on the cold panels were per NASA Report, Preliminary Definition of Saturn Instrument Unit and S-IVB Support Capabilities, dated 15 April 1965.

The cavities in each zone are identified as separate dash numbers of that zone. Cavities are numbered clockwise looking forward on the vehicle with the numbers beginning at position 1 which is the down position in Earth orbit. The numbering of the cavities in zone 4 is illustrated in Figure 12-2.

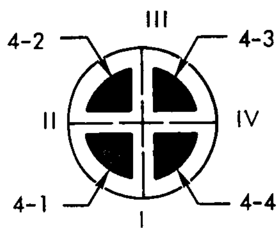


Figure 12-2 ZONE 4 CAVITIES

A total of 53 cavities were defined as potential payload locations, and each of these cavities was described in terms of the following:

1. Volume/Geometry

2. Mass capacity

3. Environment

4. Deployment capability.

The V, R, and L axis system is used in defining the dimensions and orientation of the cavities. The V axis is generally parallel to the vehicle longitudinal axis, the R axis is generally normal to the external contour of the vehicle, and the L axis is 90 degrees to both the V and R axes.

12.2 VOLUME/GEOMETRY

The geometry of each cavity was defined in terms of its capacity to contain the standard geometrical shapes, the parallelepiped, the sphere, and the cylinder. Definition of the capacity of the standard shapes was complicated by the fact that most of the cavities were tapered. The capacities shown in Figure 12-3 were defined in the following manner:

1. Parallelepiped - The axis system V, R, and L was used in describing the dimensions of the parallelepipeds. An analysis of cavity drawings yielded the type of curves shown in Figure 12-3 in which the L dimension is a function of the R dimension for various values of V.

2. Sphere - Layouts were made for each cavity to show the maximum size of sphere that the cavity would contain.

3. Cylinder - Definition of the cylindrical capacity was accomplished for a cylinder oriented with its longitudinal axis along each of the V, L, and R axes. An analysis of the cavity drawings yielded the type of

EACH CAVITY DEFINED BY CAPACITY TO CONTAIN CERTAIN STANDARD GEOMETRICAL SHAPES

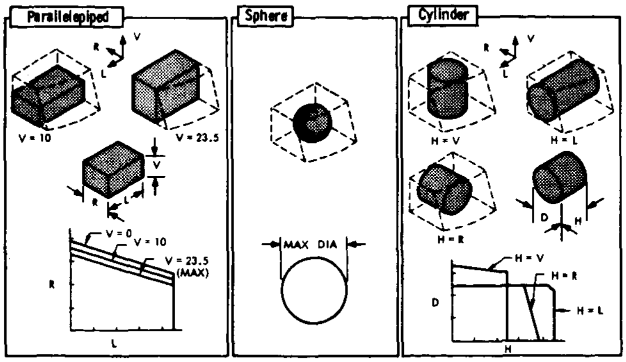


Figure 12-3 CAVITY GEOMETRY

curves shown. The diameter D is plotted as a function of the cylindrical length H for each of the three orientations with H equal to V, R, and L. The volume, geometry, and mass capacity of the 53 cavities are presented in Figure 12-4.

12.3 MASS CAPACITY

The mass capacities shown in Figure 12-4 for cavities in zones 1, 2, 3, 4, and 7 are predicated on a Mode I operation in which only one experiment at a time is considered. The 1,000-pound capacity of the cavities in zones 1, 2, and 3 represents the total load-carrying capability of each segment of the LEM adapter fairing. For a Mode II operation the capacity of a cavity would be equal to 1,000 pounds less the mass of the experiments already located on the particular adapter fairing segment. The mass capacity of cavities in zones 4 and 7 is based on values obtained from the "Saturn IB Payload Planner's Guide," Douglas Report SM-47010, Douglas Missile and Space Systems Division, June 1965. These values of 2,500 and 1,000 pounds represent the total load-carrying capability of those areas of the vehicle. The values for a Mode II operation can be obtained by the method previously discussed. The mass capacities of cavities in zones 5 and 6 are based on the load-carrying capability of the individual cold panels and are applicable to both Mode I and Mode II.

12.4 DEPLOYMENT CAPABILITY

The "cavity deployment capability" refers to the ability of a cavity to contain experiments that require exposure to vacuum, extension of an experiment component from the launch vehicle, separation of the experiment payload from the launch vehicle, or separation of a data recovery capsule. This capability is limited by the launch vehicle configuration and by the location of the cavity on the vehicle. In Section 9, Figure 9-3, the experiment requirements are described by six deployment models, Mode 0 through Mode 5. Table 12-1 presents the experiment deployment modes that each cavity is capable of containing, and the maximum angle at which a component can be deployed from an experiment in the cavity. As shown in Figure 9-3, all of the modes except Mode 0 require a vehicle configuration in which the Apollo payload has separated and the LEM adapter fairings are in the open position.

12.5 ENVIRONMENT

12.5.1 Thermal

The thermal environment associated with each cavity was defined, Table 12-2, in terms of the maximum allowable rate of heat dissipation, the maximum total short-period heat dissipation, and the time-space averaged sink temperature. These parameters are dependent on the mission phase, and a separate specification is required for each phase - prelaunch, launch, and orbit.

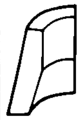
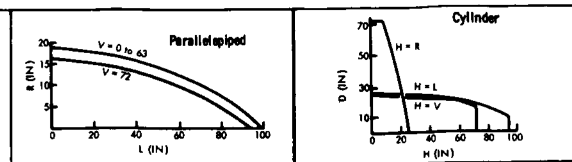
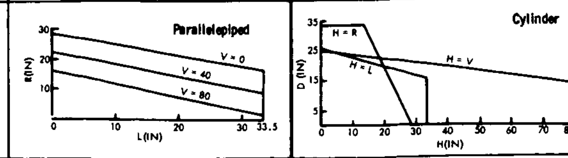
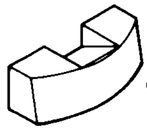
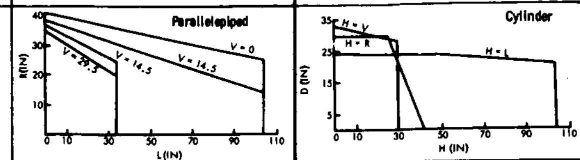
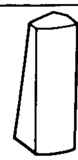
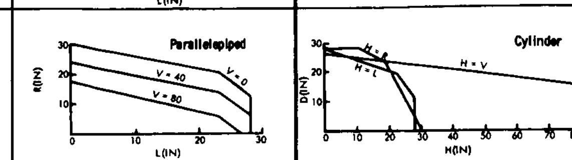
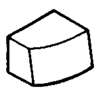
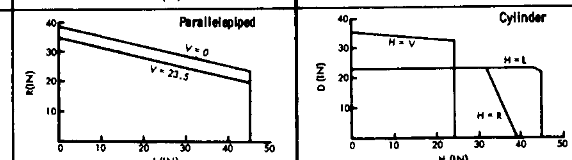
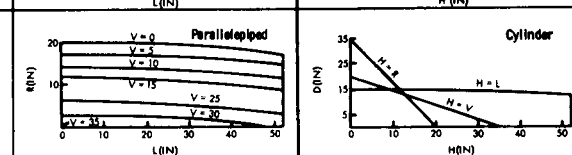
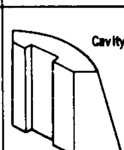
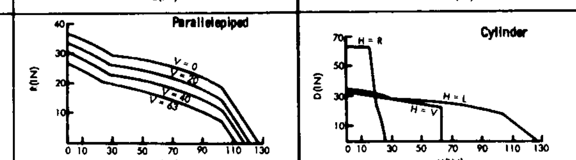
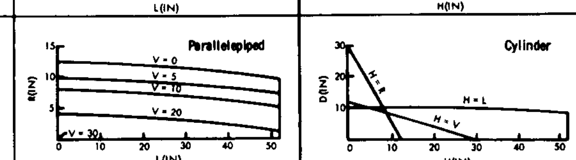
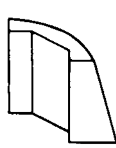
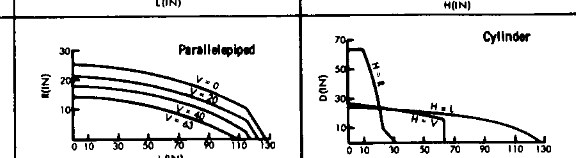
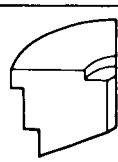
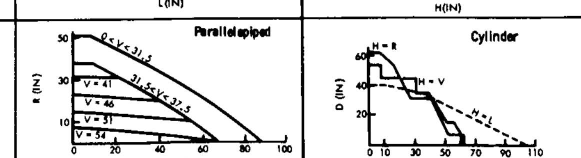
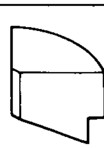
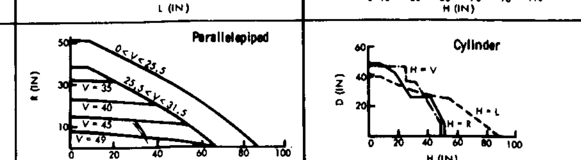
ZONE 1	 Cavity 1-1, 1-2, & 1-3	• Volume — 163,966 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 24.5 in	 Parallelepiped / Cylinder
ZONE 2	 Cavity 2-1 & 2-7	• Volume — 42,560 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 21.0 in	 Parallelepiped / Cylinder
ZONE 2	 Cavity 2-2	• Volume — 92,030 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 29.5 in	 Parallelepiped / Cylinder
ZONE 2	 Cavity 2-3 & 2-4	• Volume — 43,300 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 25.0 in	 Parallelepiped / Cylinder
ZONE 2	 Cavity 2-5 & 2-6	• Volume — 30,000 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 23.5 in	 Parallelepiped / Cylinder
ZONE 3	 Cavity 3-1, 3-3, 3-7	• Volume — 18,200 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 15.0 in	 Parallelepiped / Cylinder
ZONE 3	 Cavity 3-2, 3-4, & 3-6	• Volume — 171,039 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 28.0 in	 Parallelepiped / Cylinder
ZONE 3	 Cavity 3-5	• Volume — 9,350 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 10.0 in	 Parallelepiped / Cylinder
ZONE 3	 Cavity 3-8	• Volume — 132,500 in^3 • Mass Capacity — 1,000 lbs • Sphere Diameter — 22.5 in	 Parallelepiped / Cylinder
ZONE 4	 Cavity 4-1, 4-2, & 4-3	• Volume — 138,080 in^3 • Mass Capacity — 2,500 lbs • Sphere Diameter — 38.5 in	 Parallelepiped / Cylinder
ZONE 4	 Cavity 4-4	• Volume — 94,280 in^3 • Mass Capacity — 2,500 lbs • Sphere Diameter — 38.5 in	 Parallelepiped / Cylinder

Figure 12-4 CAVITY VOLUME, GEOMETRY, AND MASS CAPACITY SUMMARY

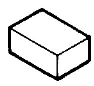
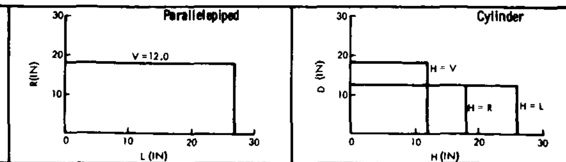
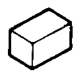
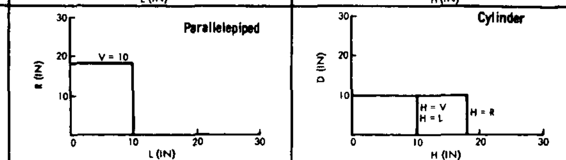
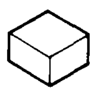
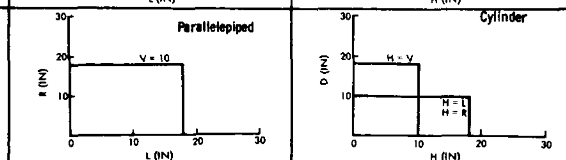
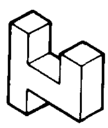
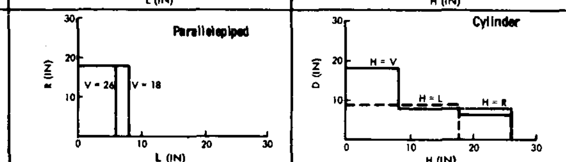
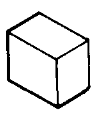
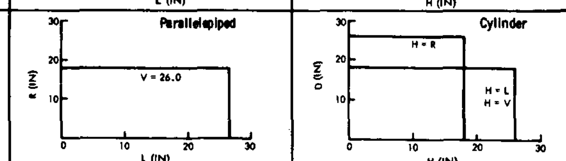
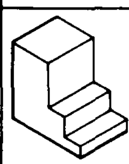
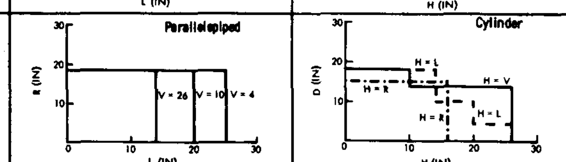
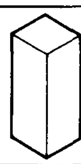
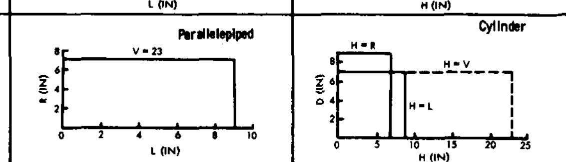
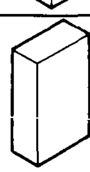
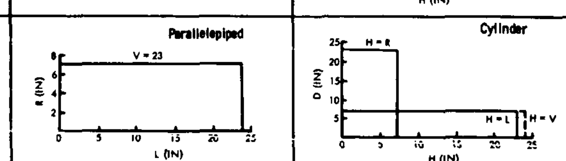
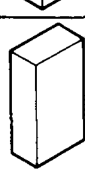
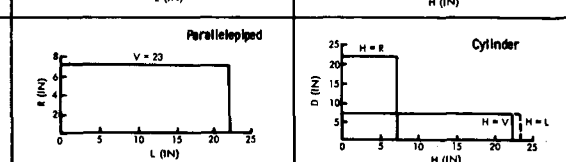
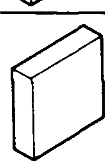
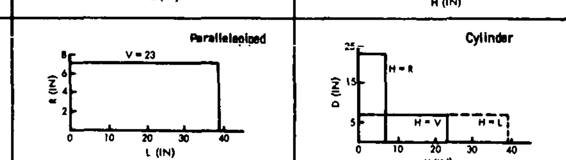
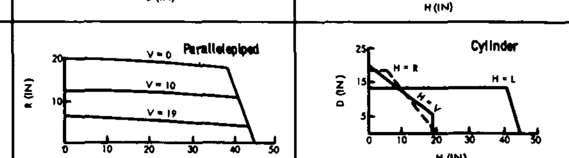
ZONE 5	 Cavity 5-1 & 5-6	● Volume ______ 5,616 in³ ● 5-1 Mass Capacity ______ 150 lbs ● 5-6 Mass Capacity ______ 297 lbs ● Sphere Diameter ______ 12.0 in	
ZONE 5	 Cavity 5-2	● Volume ______ 1,800 in³ ● Mass Capacity ______ 276 lbs ● Sphere Diameter ______ 10.0 in	
ZONE 5	 Cavity 5-3	● Volume ______ 3,240 in³ ● Mass Capacity ______ 280 lbs ● Sphere Diameter ______ 10.0 in	
ZONE 5	 Cavity 5-4	● Volume ______ 6,950 in³ ● Mass Capacity ______ 208 lbs ● Sphere Diameter ______ 9.3 in	
ZONE 5	 Cavity 5-5 & 5-7	● Volume ______ 12,186 in³ ● Mass Capacity ______ 330 lbs ● Sphere Diameter ______ 18.0 in	
ZONE 5	 Cavity 5-8	● Volume ______ 8,064 in³ ● Mass Capacity ______ 286 lbs ● Sphere Diameter ______ 14.5 in	
ZONE 6	 Cavity 6-1	● Volume 1,449 in³ ● Mass Capacity 50 lbs ● Sphere Diameter 7.0 in	
ZONE 6	 Cavity 6-2	● Volume 3,864 in³ ● Mass Capacity 130 lbs ● Sphere Diameter 7.0 in	
ZONE 6	 Cavity 6-3 & 6-5	● Volume 3,542 in³ ● Mass Capacity 125 lbs ● Sphere Diameter 7.0 in	
ZONE 6	 Cavity 6-4	● Volume 6,279 in³ ● Mass Capacity 150 lbs ● Sphere Diameter 7.0 in	
ZONE 7	 Cavity 7-1 thru 7-18	● Volume ______ 9,610 in³ ● Mass Capacity ______ 1,000 lbs ● Sphere Diameter ______ 13.2 in	

Figure 12-4 CAVITY VOLUME, GEOMETRY, AND MASS CAPACITY SUMMARY (Cont'd)

12.5.2 Vibration and Acoustics

The vibration and acoustic environments associated with each cavity were defined in terms of sinusoidal vibration levels and a maximum overall sound pressure level as shown in Figure 12-5. The actual values used in the definition were extracted from the "Saturn IB Payload Planners' Guide". These values represent the maximum environments to which components contained in these cavities would be subjected.

12.5.3 Electromagnetic

The electromagnetic environment of the launch vehicle was described by a narrowband and broadband transmitter signal and a narrowband and broadband receiver sensitivity as shown in Figure 12-6.

TABLE 12-1
CAVITY POSSIBLE DEPLOYMENT MODES AND DIRECTIONS

• DIRECTION MEASURED FROM VEHICLE • FOR MODE 4 DEPLOYMENT ONLY
+ θ UP, - DOWN + ϕ RIGHT, - LEFT

Cavity No.	Possible Deployment Modes	Direction (Degrees)	
		θ	ϕ
1-1	5, 4, 3, 2, 1, 0	+72 1/2 TO -72 1/2	+81 TO -35
1-2	4, 3, 2, 1, 0	+35 TO -81	+72 1/2 TO -72 1/2
1-3	5, 4, 3, 2, 1, 0	+72 1/2 TO -72 1/2	+35 TO -81
2-1	4, 3, 2, 1, 0	+64 TO -35	+56 TO -49 1/2
2-2	5, 4, 3, 2, 1, 0	+58 TO -58	+70 1/2 TO -35
2-3	4, 3, 2, 1, 0	+35 TO -64	+56 TO -49 1/2
2-4	4, 3, 2, 1, 0	+35 TO -64	+49 1/2 TO -56
2-5	5, 4, 3, 2, 1, 0	+58 TO -62	+35 TO -70 1/2
2-6	5, 4, 3, 2, 1, 0	+62 TO -58	+35 TO -70 1/2
2-7	4, 3, 2, 1, 0	+64 TO -35	+49 1/2 TO -56
3-1	4, 3, 2, 1, 0	+53 TO +9	+41 1/2 TO -41 1/2
3-2	4, 3, 2, 1, 0	+47 TO +9	+47 TO +9
3-3	4, 3, 2, 1, 0	+41 1/2 TO -41 1/2	+53 TO +9
3-4	4, 3, 2, 1, 0	-9 TO -47	+47 TO +9
3-5	4, 3, 2, 1, 0	-9 TO -54	+41 1/2 TO -41 1/2
3-6	4, 3, 2, 1, 0	-9 TO -47	-9 TO -47
3-7	4, 3, 2, 1, 0	+41 1/2 TO -41 1/2	-9 TO -53
3-8	4, 3, 2, 1, 0	+47 TO +9	-9 TO -47
4-1	4, 3, 2, 1, 0	+41 TO 0	+41 TO 0
4-2	4, 3, 2, 1, 0	0 TO -41	+41 TO 0
4-3	4, 3, 2, 1, 0	0 TO -41	0 TO -41
4-4	4, 3, 2, 1, 0	+41 TO 0	0 TO -41
Zones 5, 6 & 7	1, 0	0	0

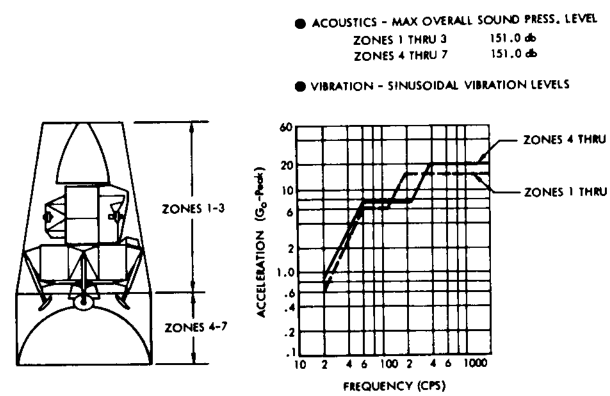


Figure 12-5 CAVITY VIBRATION AND ACOUSTIC ENVIRONMENT

TABLE 12-2
CAVITY THERMAL ENVIRONMENT

Parameter	Mission Phase		
	Prelaunch	Launch	Orbit
• MAX ALLOWABLE RATE OF HEAT DISSIPATION - ALL CAVITIES (BTU/HR)	200	100	300
• MAX TOTAL SHORT PERIOD HEAT DISSIPATION - ALL CAVITIES (BTU)	17	17	17
• TIME - SPACE AVERAGED TEMP (°F)			
CAVITIES 1-1 THRU 1-3	35 - 75	170 - 230	25 - 65
2-1 THRU 2-7	35 - 75	200 - 240	-45 - 30
3-1, 3-3, 3-5, & 3-7	35 - 75	180 - 220	35 - 65
3-2, 3-4, 3-6, & 3-8	35 - 75	190 - 230	30 - 65
4-1 THRU 4-4	35 - 75	25 - 65	25 - 65
5-1 THRU 5-4	35 - 75	15 - 55	-105 - 55
6-1 THRU 6-5	35 - 75	15 - 55	-105 - 55
7-1 THRU 7-18	35 - 75	140 - 180	100 - 140

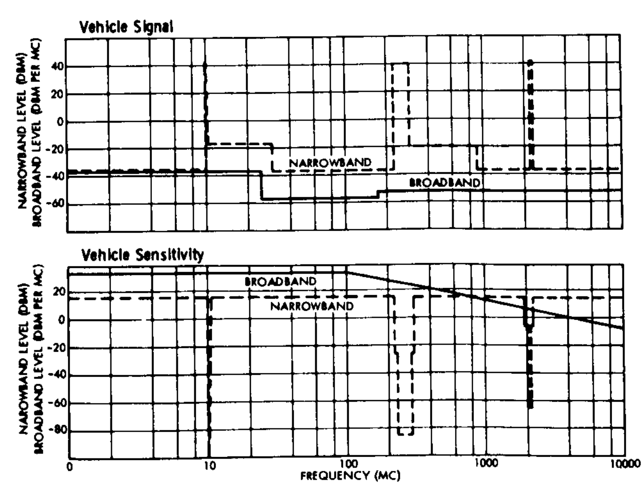


Figure 12-6 VEHICLE ELECTROMAGNETIC ENVIRONMENT

13.0 EXPERIMENT/MISSION EFFECTIVENESS ANALYSIS

13.1 EXTERNAL ANALYSIS APPROACH

Experiment effectiveness is defined as the percent of data acquisition objectives accomplished. For the type of experimental payloads considered in this study (i.e., specified components), effectiveness is primarily a function of the initial elements of the deployed orbit (experiment/mission effectiveness) Other factors (e.g., payload location, angular rates, reliability, etc.) are recognized as potentially significant, but generally secondary, influences. The determination of experiment effectiveness as a function of the initial orbital elements is accomplished by the use of methods (including auxiliary computer procedures) which are not included in Program SEPTER.

The experiment design orbit is defined as that initial orbit which yields maximum effectiveness. In some cases, the experiment effectiveness of the design orbit is less than 100 percent. This implies that no single orbit can be used to attain all the data acquisition objectives of the scientific program.

Each experiment is first analyzed in order to define its data acquisition objectives and to determine the orbital elements which affect these objectives. The trajectory/mission data necessary to relate the experiment effectiveness to the initial orbital elements are then generated by the use of auxiliary computer procedures. For some experiments, the experimenter and the mission analyst may have to perform a rather extensive analysis in order to arrive at meaningful effectiveness relationships. After these relationships have been established, effectiveness data are prepared in a form acceptable for inclusion in the Experimental Payload Characteristics Library of Program SEPTER.

13.2 EXAMPLE EFFECTIVENESS ANALYSES

The extent and complexity of experiment effectiveness analysis vary considerably with the various experiments which are being considered. In order to demonstrate this variation and to illustrate the actual analyses which must be performed to obtain effectiveness data, three example experiments are presented in the following subsections.

13.2.1 Example Effectiveness Analysis of Experiment MS-3, Vaporization of Molten Metals

This vaporization experiment provides an illustration of one of the more complex analyses required to obtain effectiveness relationships. The data acquisition objective set for this experiment is to determine the vaporization characteristics of various molten metals in near-Earth orbits. The purpose of this data acquisition objective is to verify the predicted (calculated from theory) vaporization rates of metals in a very low atmospheric pressure environment. To obtain 100 percent effectiveness, the experimenter specified that the atmospheric pressure must not exceed 10^{-7} millimeters of Hg for the duration of the experiment (29 hours). The basic effectiveness variables are, therefore, atmospheric pressure and time.

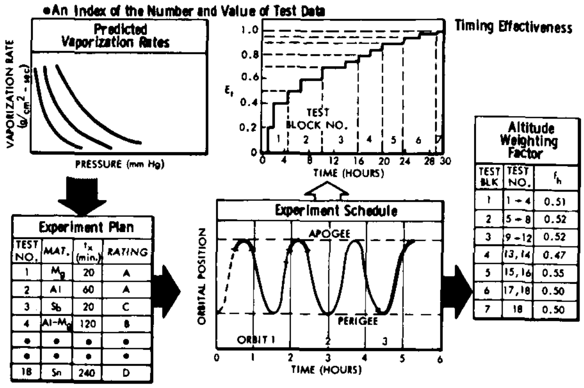


Figure 13-1 DETERMINATION OF TIMING EFFECTIVENESS FACTOR - EXPERIMENT MS-3

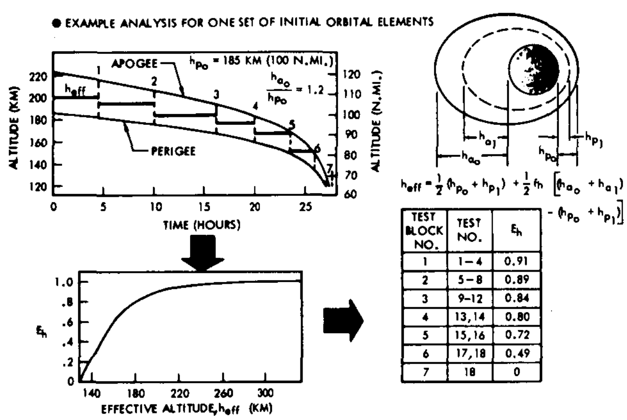


Figure 13-2 DETERMINATION OF ALTITUDE EFFECTIVENESS FACTOR - EXPERIMENT MS-3

To accomplish the data acquisition objective, the experimenter planned 18 vaporization tests, using eight selected materials. Each material was rated on the basis of the scientific and practical value of the data it could yield. In the experiment plan of Figure 13-1, ratings A, B, C, etc. designate decreasing yields. The individual experiments were then scheduled on the basis of this rating and the predicted vaporization rates in such a way that the amount and value of test data would decrease with time. The experiment schedule is partially illustrated in Figure 13-1.

By the use of the experiment schedule, the experimenter subjectively evaluated the data yield as a function of experiment duration to arrive at a "timing effectiveness" function E_t (t) as shown in the upper right graph of Figure 13-1. The timing effectiveness factor, Et, is an index to the amount and value of the test data accumulated at any time that the experiment is terminated. The experimenter also assigned altitude "weighting factors," f_h, to each of several blocks of tests, as shown in the listing at the extreme right in Figure 13-1. These factors were used for computing the "effective" altitude (as defined in Figure 13-2) of each block of tests. In the case of short tests, which occur near apogee, the factor is nearly 1.0; consequently, the effective altitude is weighted toward the average apogee altitude.

Because of the time variance of the altitude and, therefore, atmospheric pressure, the concept of effectiveness altitude was introduced to simplify the experiment effectiveness analysis. For a given test, the effective altitude was determined by adding the weighted average of the difference between the perigee and apogee altitudes to the average perigee altitude (see equation in Figure 13-2). The weighting factor, f_h, in the equation was arrived at subjectively by the experimenter. In the example experiment, significant decay of perigee and apogee altitude occurred during the experiment. Therefore, the tests were divided into blocks, and the effective altitude was determined for each block. After the experimenter determined the effective altitude, the altitude effectiveness factor, E_h, was defined subjectively as a function of the calculated effective altitude. This relationship is given in the lower left graph in Figure 13-2.

A sample analysis of an orbit with an initial perigee altitude of 185 kilometers (100 nautical miles) and an initial apogee/perigee altitude ratio of 1.2 is shown in Figure 13-2. The experiment was divided into seven test blocks, and the effective altitude was computed for each block. From the altitude effectiveness curve, E_h was determined and tabulated for each test block. From the tabulated data in Figure 13-2, it can be seen that the altitude effectiveness decreases rapidly after test 16 and becomes zero in the last test. The procedure illustrated in Figure 13-2 was repeated for a matrix of initial perigee altitudes and apogee/perigee altitude ratios to complete this portion of the analysis.

The final step in this example analysis was to compute the experiment effectiveness from the timing and altitude effectiveness factors, E_t and E_h. This computation was done by multiplying the sum of the product ($\Delta E_t \cdot E_h$) by an eccentricity factor, f_e. The quantity ΔE_t is the change in E_t over the duration of the test block, and E_h is the altitude effectiveness of the test block. The eccentricity factors were used to adjust the effectiveness relationship for a slight degradation of the data caused by the altitude variation that results from orbital eccentricity.

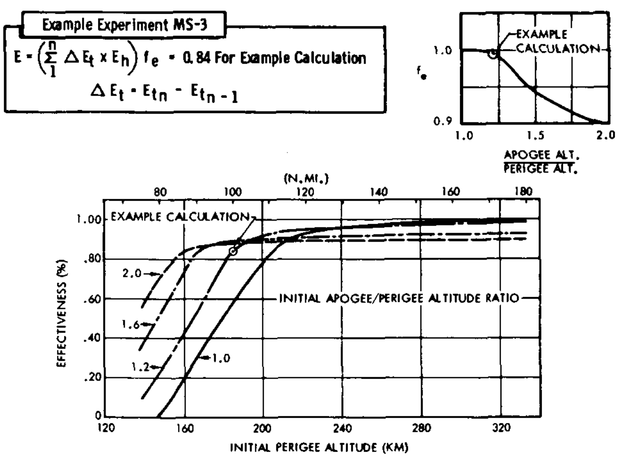


Figure 13-3 EFFECTIVENESS AS A FUNCTION OF THE INITIAL ORBITAL ELEMENTS - EXPERIMENT MS-3

Experiment effectiveness is shown in Figure 13-3 as a function of the initial orbital elements (i.e., perigee, altitude, and apogee/perigee altitude ratio). These are the data that are required for the effectiveness segment of the Experimental Payload Characteristics Library.

The design orbit for the example experiment is an initially circular orbit at an altitude of 333 kilometers (180 nautical miles). In this case, the maximum effectiveness for the design orbit is 100 percent. Because of the eccentricity factor, only circular orbits can achieve 100 percent effectiveness.

13.2.2 Example Effectiveness Analysis of Experiment SLG-2, Nucleate Condensation in Zero Gravity

A second experiment which illustrates a simpler and perhaps more typical effectiveness relationship is shown in Figure 13-4. The objective of experiment SLG-2 is to observe nucleate condensation in a zero-gravity environment. Data are recorded on magnetic tape and relayed back at four-hour intervals. In order to achieve 100 percent effectiveness, the drag acceleration must be less than 0.01 g for the duration of the experiment (24 hours). Should the experiment be terminated during one of the four-hour test intervals, the data recorded in that interval are considered unavailable. Therefore, the effectiveness variation with experiment duration is a series of step functions as shown in the upper left graph in Figure 13-4.

To relate experiment effectiveness to the initial perigee and apogee altitudes, the useful orbital lifetime was determined as a function of these elements (lower left graph). The useful orbital lifetime for this experiment is defined as the period of time when the drag acceleration is less than 0.01 g (i.e., perigee is less than about 104 kilometers). From the lifetime data, curves of perigee altitude versus apogee/perigee ratio were generated for 4-,

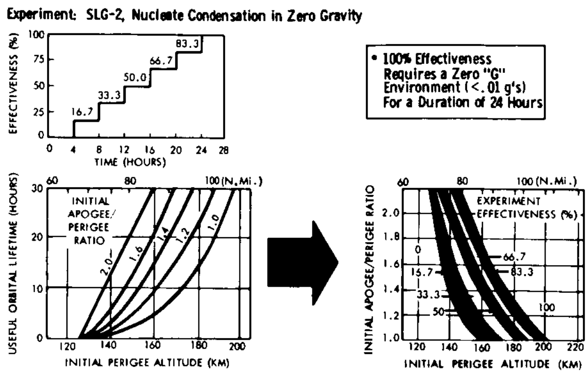


Figure 13-4 EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS - EXPERIMENT SLG-2

8-, 12-, 16-, 20-, and 24-hour orbital lifetimes corresponding to 16.7-, 33.3-, 50-, 66.7-, 83.3-, and 100-percent effectiveness values, respectively. Any initial set of perigee and apogee altitudes which provide an orbital lifetime of 24 hours will have an effectiveness of 100 percent as shown in the lower right graph in Figure 13-4.

13.2.3 Example Effectiveness Analysis of Experiment SDT-4, Cryogenic Propellant Storage System Performance

The objectives of experiment SDT-4 are to (1) evaluate the performance of certain thermal protection systems, (2) determine the degree of propellant stratification, and (3) evaluate the performance of an ullage orientation system for the reduction of propellant stratification.

The effectiveness of SDT-4 is dependent upon four parameters: (1) useful orbital lifetime, (2) mean drag acceleration of initial orbit (3) change in mean drag acceleration over the mission duration, and (4) initial inclination to the terminator. Parameters (1), (2) and (3) are determined by the atmospheric decay of orbit altitude and can be expressed in terms of the initial perigee and apogee altitude. Thus, the experiment effectiveness can be defined in terms of the initial perigee altitude, the initial apogee/perigee altitude ratio, and the initial inclination of the orbit to the terminator.

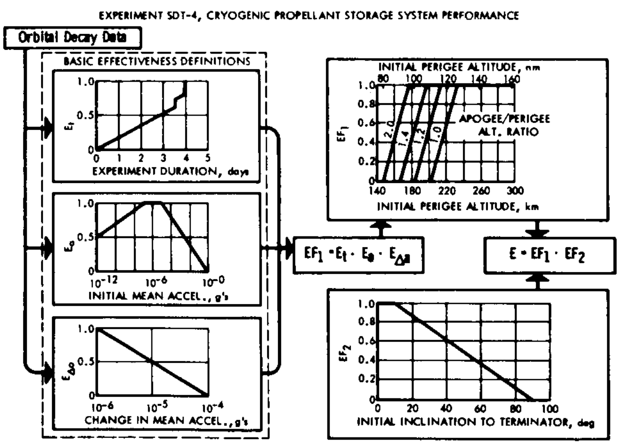


Figure 13-5 EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS - EXPERIMENT SDT-4

Experiment effectiveness can be expressed as the product of the timing factor E_t, the initial mean acceleration factor E_a, the mean acceleration change factor $E_{\Delta a}$, and the inclination to terminator factor E_{f2}. The basic effectiveness factor relationships (see Figure 13-5) were established by the experimenter after an analysis of the effects of mission duration and drag acceleration on experiment effectiveness.

If the useful orbit lifetime (defined as the time from perigee of the first orbit after deployment to apogee of the last complete orbit) is greater than 4 days, completion of the entire experiment is possible and $E_t = 1.0$. A mission of shorter duration degrades the effectiveness of the experiment. Likewise, the values of the initial mean acceleration (time average of drag acceleration over one orbit) less than 10^{-7} g's and greater than 10^{-5} g's and/or the values of the change in mean acceleration (difference between mean acceleration of the initial orbit and the last complete orbit) greater than 10^{-6} g's, result in a decrease in experiment effectiveness.

Orbital decay data were generated for a series of orbits with perigee altitudes between 148 and 296 km and apogee/perigee altitude ratios between 1.0 and 2.0. Based on these data the effectiveness factors E_t, E_a, and $E_{\Delta a}$ were determined for each orbit from the basic effectiveness relationships and then multiplied together to form a combined effectiveness factor EF_1. Factor EF_1 as a function of perigee altitude and apogee/perigee altitude ratio and factor EF_2 as a function of initial inclination to the terminator are shown in Figure 13-5. These data were loaded into the effectiveness segment of the Experimental Payload Characteristics Library. The factors EF_1 and EF_2 are obtained from these tables and multiplied together to form the absolute experiment effectiveness.

14.0 PROGRAM SEPTER

METHODOLOGY - MODE I

14.1 MODE I OPERATION

Mode I analysis consists in the determination of the physical and operational compatibility and the experiment/mission effectiveness of single experimental payloads in terms of a given vehicle/mission/primary payload combination. Compatibility criteria include only major items which are considered significant within the scope of the program philosophy. Experiment/mission effectiveness is computed as the percent of data acquisition objectives accomplished. Effectiveness relationships are determined externally in terms of the initial values of the orbital elements in which the experimental payload is deployed. In Mode I output, the overall GO/NO-GO compatibility, the degree of compatibility or incompatibility, experiment effectiveness data, and the experimental payload library data for input to Mode II of the program are defined.

14.2 LIBRARIES

The definition-type input data for SEPTER are stored and provided to the program in the form of libraries. These libraries are of two distinct types: Mission/Vehicle/Primary Payload Characteristics, and (2) Experimental Payload Characteristics. As is implied by the title, the Mission/Vehicle/Primary Payload Library contains: (1) mission data: mission identification code, launch date and time, launch trajectory, mission duration, and primary payload separation time, and (2) vehicle/primary payload data: vehicle identification, vehicle- and zone-dependent environmental definitions, excess payload capability, experimental payload location (cavity) identifications; and cavity environmental definitions, mass limits, volumes, geometric standard shapes capacities, and deployment modes.

The Experimental Payload Characteristics Library contains definition data for each experiment: identification code, availability, installation time, deployment modes and time, deployment velocity increments and angles (if applicable), environmental characteristics, standard shapes, dimensions and alignment (with vehicle), mass, volume, experiment/mission effectiveness, reliability, development time, and cost.

14.3 PROBLEM INPUT AND CONTROLS

The inputs for the operation of Mode I consist of library data and problem control data. The Mission/Vehicle/Primary Payload Characteristics Library data are provided from a binary library tape. The Experimental Payload Characteristics Library data are provided from card decks. Both types of libraries are required to run a problem. Problem control data (from card decks) are used to select computational options and specify overrides for the binary library tape input data.

Problem options and overrides provide operation and program utilization versatility. Problem options include the following:

1. Computation of experiment/mission effectiveness and compatibility for each experimental payload. Compatibility is computed with respect to each vehicle cavity as well as to the overall vehicle.

2. Computation of experiment/mission effectiveness data only

3. Computation of experimental payload compatibility data only

4. Generation of a Mode II compatibility library card deck

5. Exclusion of experimental payload cavities (specify how many and which ones)

Mission/Vehicle/Primary Payload Library data overrides include the following:

1. Launch date

2. Launch time

3. Vehicle-primary payload separation time (allowing ejection of experimental payloads)

4. Primary mission duration

5. Excess payload capability.

14.4 DEPLOYMENT

The various data acquisition objectives of possible experimental payloads require a broad spectrum of operating environments. With some experimental payloads almost all the data can be acquired with the payload remaining fixed inside the launch vehicle. Other payloads require separation from the vehicle, and still other types require injection into orbits other than that of the primary payload. The following six deployment modes are included in SEPTER to represent the various deployment requirements:

Mode 0 - Fixed. The experimental payload remains on the vehicle (mission profile), requires the extension of only an antenna, and does not require exposure to a vacuum.

Mode 1 - Fixed Exposed. The experimental payload remains on the vehicle (mission profile), requires the extension of only an antenna, and requires exposure to a vacuum.

Mode 2 - Extension. The experimental payload remains on the vehicle (mission profile) and requires extension of components other than an antenna.

Mode 3 - Separation. The experimental payload is separated from the vehicle. The orbital elements of the separated payload are assumed to be the same as those of the vehicle at the time of deployment.

Mode 4 - Propulsive Separation (ΔV). The experimental payload is separated from the vehicle, and propulsion is required to inject the payload into an orbit different from that of the primary payload.

Mode 5 - Recovery Capsule Separation. The experimental payload remains on the vehicle, but separation of one or more recovery capsules is required.

These six deployment modes have been illustrated previously in Figure 9-3. The coordinate systems used in the deployment methodology are illustrated and defined in Figure 14-1.

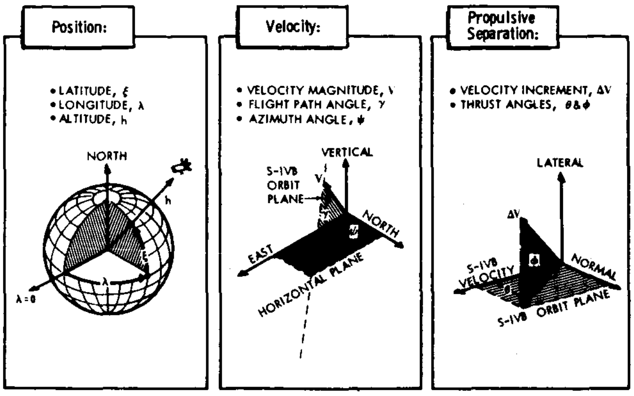


Figure 14-1 DEPLOYMENT METHODOLOGY COORDINATE SYSTEMS

The methodology used in the computer program to calculate the orbital elements and additional mission parameters of an experimental payload for any specified deployment mode and deployment time is illustrated in Figure 14-2. These data are made available to another subroutine (EFFECT) in the program to calculate experiment/mission effectiveness.

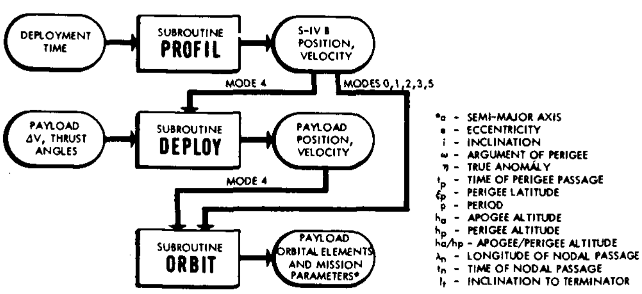


Figure 14-2 ORBITAL ELEMENTS LOGIC

14.5 EXPERIMENT/MISSION EFFECTIVENESS

The relationships between experiment effectiveness factors and the initial orbital elements and/or mission parameters are established during the external experiment effectiveness analysis discussed in Section 13.0. These effectiveness factors are loaded into the effectiveness segment of the Experimental Payload Characteristics Library in tables of one- and/or two-dimensional arrays. Effectiveness factors are obtained from each table by the use of a table "look up" procedure and are multiplied together to obtain the absolute value of experiment effectiveness (percent accomplishment of data acquisition objectives). The percentage of effectiveness relative to the maximum possible effectiveness (absolute effectiveness divided by maximum effectiveness), designated "normalized effectiveness," is also computed since, in general, the maximum effectiveness possible for a given experimental payload can be less than 100 percent.

Table look-up procedures are provided for the three types of effectiveness factor relationships shown in Figure 14-3. They are as follows: (1) effectiveness factor as a continuous function of two variables, (2) effectiveness factor as a step function of two variables, and (3) effectiveness factor as a function of one variable. Options are provided for either linear or fourth-order Lagrange interpolation.

14.6 EXPERIMENTAL PAYLOAD - MISSION/VEHICLE COMPATIBILITY

The computer methodology used to determine experimental payload compatibility with both the primary payload mission and the vehicle is based on two fundamental guidelines:

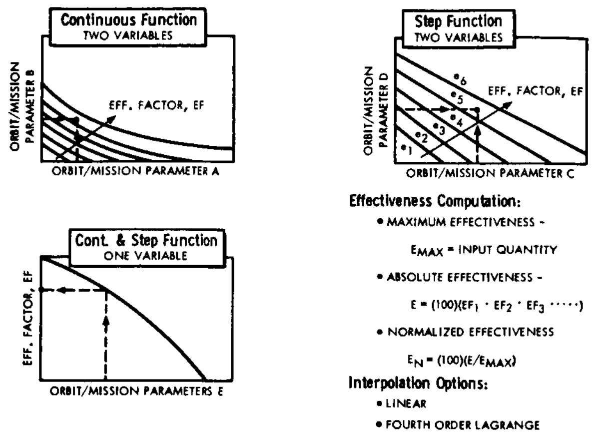


Figure 14-3 COMPUTATION OF EXPERIMENT/MISSION EFFECTIVENESS

1. The experimental payload must tolerate all mission and vehicle constraints and environments.

2. The experimental payload cannot significantly affect the performance of the primary mission/vehicle.

The compatibility criteria and the methodology used to determine compatibility are summarized in the following subsections.

14.6.1 Compatibility Criteria

The criteria used to determine experimental payload-mission/vehicle compatibility are classified as (1) physical, and (2) operational. The specific criteria in each category are summarized as follows:

PHYSICAL CRITERIA

1. Environmental (thermal, acoustic, vibration, and electromagnetic)
2. Cavity mass attachment limit
3. Volume/geometry (size, shape, and orientation).

OPERATIONAL CRITERIA

1. Experimental payload availability (availability date, launch date, and installation time)
2. Experimental payload deployment (mode and time).

In order for an experimental payload to be mission/vehicle compatible, it must satisfy the GO condition for the following listed criteria:

1. Thermal
2. Acoustic
3. Vibration
4. Mass attachment limit
5. Volume/geometry
6. Deployment mode
7. Deployment time.

The remaining criteria, i.e., experimental payload availability and electromagnetic interference (EMI), are included to indicate possible problem areas which should be investigated more thoroughly by external analysis.

The acoustic and vibration criteria are treated uniquely in the computer program in that a NO-GO condition is initially corrected to a GO-condition by the computation of a mass penalty. In the event that the computed mass penalties are obviously excessive, the NO-GO condition remains to affect the overall compatibility decision for the experimental payload.

14.6.2 Experiment Availability Compatibility

The experimental payload must obviously be available for installation in the vehicle prior to the launch date. This simple check, however, does not allow for the time required for installation and/or checkout of the experimental payload. An installation time must also be specified to provide for a more realistic compatibility check. Thus, in the compatibility check, the experimental payload availability data plus the required installation time (days) must precede the launch date.

The availability date/launch date compatibility determination in the computer program is not used to affect the overall experimental payload - mission/vehicle compatibility decision. If a GO condition is calculated, the number of "buffer" days are given in the output. In case of a NO-GO calculation, a warning statement is given in the output.

14.6.3 Deployment Compatibility

The deployment compatibility of an experimental payload with a vehicle cavity is a function of two criteria: (1) deployment mode, and (2) deployment time (if applicable). These are operational compatibility criteria. The overall GO/NO-GO compatibility determination in the computer program is affected by both of these criteria.

Each cavity is assigned the deployment mode(s) which it can accommodate. These modes are defined in subsection 14.4. Likewise, each experimental payload is assigned its required deployment mode. These assigned mode data are stored in the computer program libraries. Deployment mode compatibility is simply a check of the required mode with the available mode for the cavity from which the experimental payload must be deployed.

Deployment time compatibility is dependent on the assigned times during which a cavity is available for the specified deployment mode and the assigned time at which an experimental payload must be deployed in the mission. Only deployment modes which require the separation or ejection of an experimental payload from the vehicle are time-dependent.

14.6.4 Environmental Compatibility

Environmental compatibility is based on criteria which are either cavity dependent, vehicle-zone dependent, complete-vehicle dependent, and, in some cases, mission-phase dependent. The environmental criteria used in the computer program are: (1) thermal, (2) acoustic/vibration, and (3) electromagnetic.

Thermal environment compatibility is determined by comparing thermal parameter values pertaining to a specific cavity or vehicle zone with the corresponding parameter values associated with a given experimental payload. This comparison will yield GO/NO-GO decisions. Thermal compatibility requires a full set of GO decisions for the thermal parameters that are selected as meaningful for a given cavity-experimental payload. The three following thermal parameters are used for comparison in the program:

1. Time-space averaged sink temperature

2. Heat dissipation rate

3. Total short period heat dissipation.

Items (2) and (3) are optional. These parameters are defined for three mission phases: prelaunch, launch, and orbital. Compatibility checks are made only during the mission phases in which the experiment is aboard the vehicle. In certain situations the heat dissipation rate and the total short period heat dissipation thermal parameters are not mutually exclusive criteria. Therefore, the computer methodology provides an optional capability such that either one of the two parameters can be excluded from the compatibility checks. However, this optional capability does not preclude the use of both parameters in cases where they are both applicable. The overall methodology used in the computer program to determine thermal compatibility is illustrated in Figure 14-4.

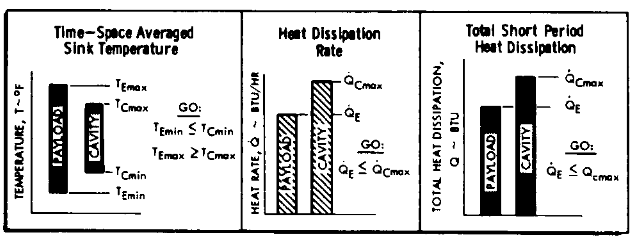


Figure 14-4 THERMAL ENVIRONMENT COMPATIBILITY

The acoustic/vibration environmental compatibility determination in the computer program does not yield a direct GO/NO-GO decision. Rather, the methodology determines whether a given experimental payload can survive the environmental noise level and vibration level induced during booster operation, and if a possible failure is indicated, a mass penalty is computed and added to the mass of the experimental payload in order to correct the tolerance deficiency. The experimental payload is disqualified by a NO-GO decision only if the calculated mass penalty is obviously excessive. The simple GO/NO-GO comparison methodology is not used in the program for two reasons: (1) the experimental payloads defined for Program SEPTER are not necessarily "fixed-design," and (2) preliminary acoustic/vibration data will seldom be accurate enough to justify the elimination of an experimental payload on this basis. Even if the actual tolerance levels of the payload fall below the ambient levels of the vehicle, it is reasonable to assume that the experimental payload components could be built to withstand the environmental levels by the addition of increased material gauges, isolation mountings, stiffeners, or dampers (mass penalties). Therefore, the compatibility methodology used in Program SEPTER consists in the assignment of ambient noise and vibration levels to vehicle zones and tolerance noise and vibration levels to the experimental payloads as in the case for a simple GO/NO-GO comparison concept. However, since it is reasonable to assume mass penalties for the correction of deficiencies in a "non-fixed-design" experimental payload, this feature is also a part of the methodology. Mass penalties are calculated for both noise and vibration deficiencies. The noise and vibration parameters and the methodology used in the compatibility checks are illustrated in Figure 14-5. The mass penalty

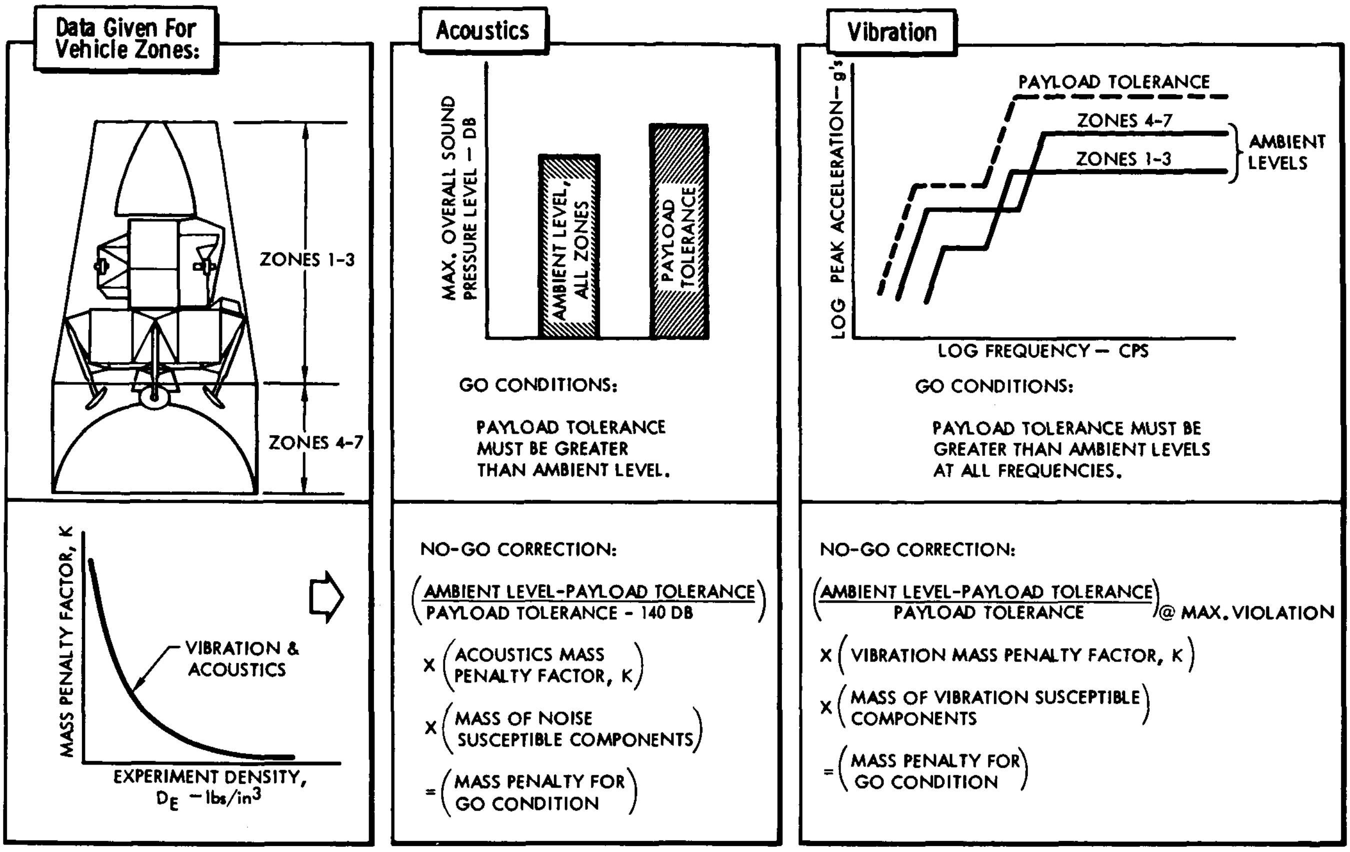


Figure 14-5 ACOUSTICS AND VIBRATION ENVIRONMENT COMPATIBILITY

methodology is illustrated by the data and equations given in the lower half of Figure 14-5. The compatibility methodology is based on the following assumptions:

1. If an experimental payload can survive the launch phase of a mission, it can survive all other phases.

2. The experimental payload does not operate during the launch phase.

3. Acoustical and vibration tolerance deficiencies can be corrected by the addition of mass to the experimental payload.

4. The mass required to correct a deficiency is proportional to the payload density, the percent deficiency, and the mass of susceptible components.

5. The correction of an acoustic tolerance deficiency does not correct a vibration tolerance deficiency and vice versa because the range of frequencies associated with each are different.

Electromagnetic compatibility can be defined as the ability of each piece of electrical/electronics equipment in an integrated system to perform its design function without interfering with the performance of the design function of any other piece of electrical/electronic equipment in the system. In the computer methodology, the Saturn vehicle/primary payload experimental payload is regarded as the system. The basic parameters which define electromagnetic compatibility are: (1) the level and bandwidth of the signal which the equipment is capable of emitting, (2) the level and bandwidth of the signal to which the equipment is capable of responding, (3) "coincident time interval" or the simultaneous operation of equipments whose parameters, items (1) and (2), overlap, and (4) the amount of isolation (insulation or separation) between interfering equipment. In order to determine electromagnetic compatibility for a definite GO/NO-GO decision, the given parameters would have to be accurately defined. Because the determination of the "coincident time interval" (requiring payload operations scheduling) and the amount of isolation between equipments (a function of numerous undefined variables) is beyond the scope of the present computer program, the "exact" methodology was replaced by a more simplified approach. However, because of the simplifying assumptions made, the compatibility checks are not sufficiently accurate or complete to justify a definite GO/NO-GO decision. Therefore, the output of the computer program only gives warning-type statements for indicated incompatibilities. Frequency ranges are given where incompatibilities may exist. This type of output is helpful in locating possible problem areas which can only be thoroughly analyzed external to the computer program. The simplifying assumptions made in the computer methodology are as follows:

1. All electromagnetic equipments operate simultaneously. (The present computer program does not include operations scheduling.)

2. No isolation exists between equipments aboard the vehicle.

3. An infinite amount of isolation exists between vehicle equipment and an ejected experimental payload equipment.

The computer methodology used to check electromagnetic compatibility for both narrowband and broadband signals is illustrated in Figure 14-6. Note that the amount of "overlap," i.e., signal level greater than sensitivity, is the amount of interference within a given frequency bandwidth and that in order for compatibility to exist isolation equivalent to the overlap must be provided.

14.6.5 Mass Attachment Compatibility

The determination of the mass compatibility of an experimental payload with the cavity in which it is placed is based on the mass attachment limit of the cavity. This limit is usually determined from structural analyses. The total mass of experimental payload includes any penalty masses which may have been computed and added for the correction of acoustic/vibration tolerance deficiencies.

The experimental payload-cavity mass attachment compatibility methodology in the computer program yields a GO/NO-GO decision for each cavity. However, the methodology which determines the compatibility of an experimental payload mass with the excess payload capability of the vehicle/primary payload does not affect the overall GO/NO-GO decision. In the event that a single experimental payload mass exceeds the total excess payload capability, a warning-type statement indicating the overload is given in the output. In this manner, all other compatibility criteria are analyzed in Mode I, and the final accumulative multiple payload mass compatibility is determined in Mode II.

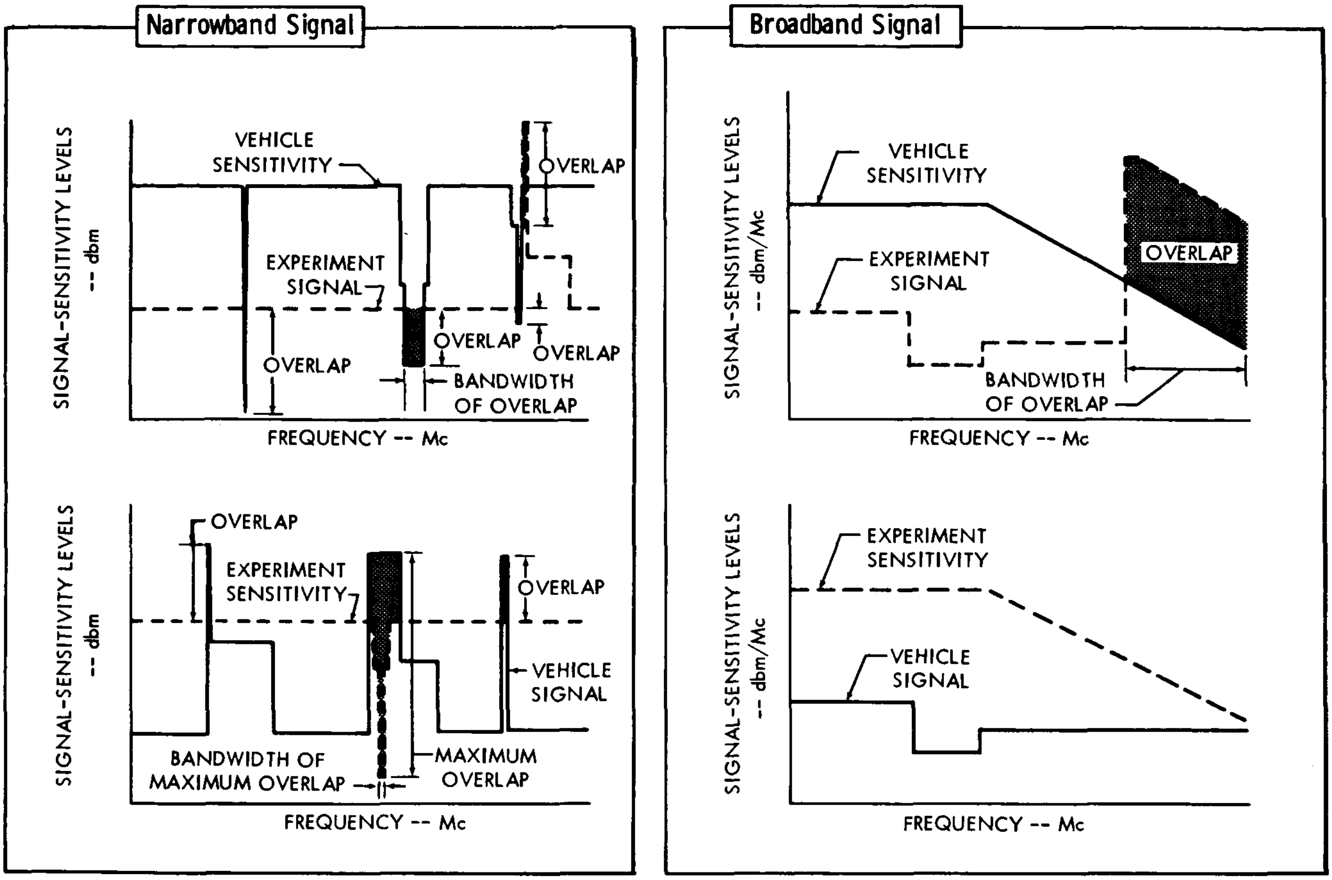


Figure 14-6 ELECTROMAGNETIC ENVIRONMENT COMPATIBILITY

14.6.6 Volume/Geometry Compatibility

The volume/geometry physical compatibility analysis consists in the determination of whether a given experimental payload can be contained within a given payload cavity. The formulation of a general "exact" methodology for the computer program would require an extremely complex computer program logic and in many cases prohibitive data storage capabilities. Therefore, a less general methodology, i.e., one restricted to standard shape representation for experimental payloads, is used. The actual sizes and shapes of the cavities are not represented in the program; instead, their capacities for several geometrical solids are stored in tabular form in the Mission/Vehicle/Primary Payload Characteristics Library. The experimental payloads are represented as either fixed in shape or amorphous. The fixed geometry representations are restricted to one of the standard shapes for which the cavity capacities have been analyzed, i.e., sphere, right circular cylinder, or rectangular parallelepiped. An amorphous geometry payload is treated as a fluid volume containing an undistortable component which is given a fixed geometry representation (standard shape envelope). The total volume of an amorphous geometry payload is composed of the sum of the volumes of the components multiplied by a "packaging" factor. These two concepts allow for the representation of experimental payloads which are (1) in the "off-the-shelf" or final design stages, and (2) those in the conceptual design stage and amenable to some rearranging of the components of the entire payload package.

The vehicle cavities are divided into two categories, rectilinear or tapered, according to the form of their capacities data. There are very simple methods available for representing the rectilinear capacities in a computer program. However, these methods are not applicable to the tapered cavity capacities. A single technique, which is reasonably simple and nearly exact, was utilized for representing both types of cavities. The method is general, efficient, and accurate.

14.6.6.1 <u>Sphere and Cylinder Capacities</u>

Since some experimental payloads may require a specified orientation in the vehicle, an orthogonal coordinate system is used in each cavity. In this system, the vertical axis is parallel to the vehicle longitudinal axis, the radial axis emanates from the center of the vehicle and passes through the cavity, and the lateral axis is perpendicular to the other two axes. The geometric capacities compatibility methodology is used to determine whether a sphere of given diameter or a cylinder of given diameter and length with its axes parallel to the vertical, radial, or lateral axes can be contained in a specified cavity.

A tapered and a rectilinear cavity and their sphere and cylinder capacities are shown in Figure 14-7. There is only one maximum diameter sphere which can

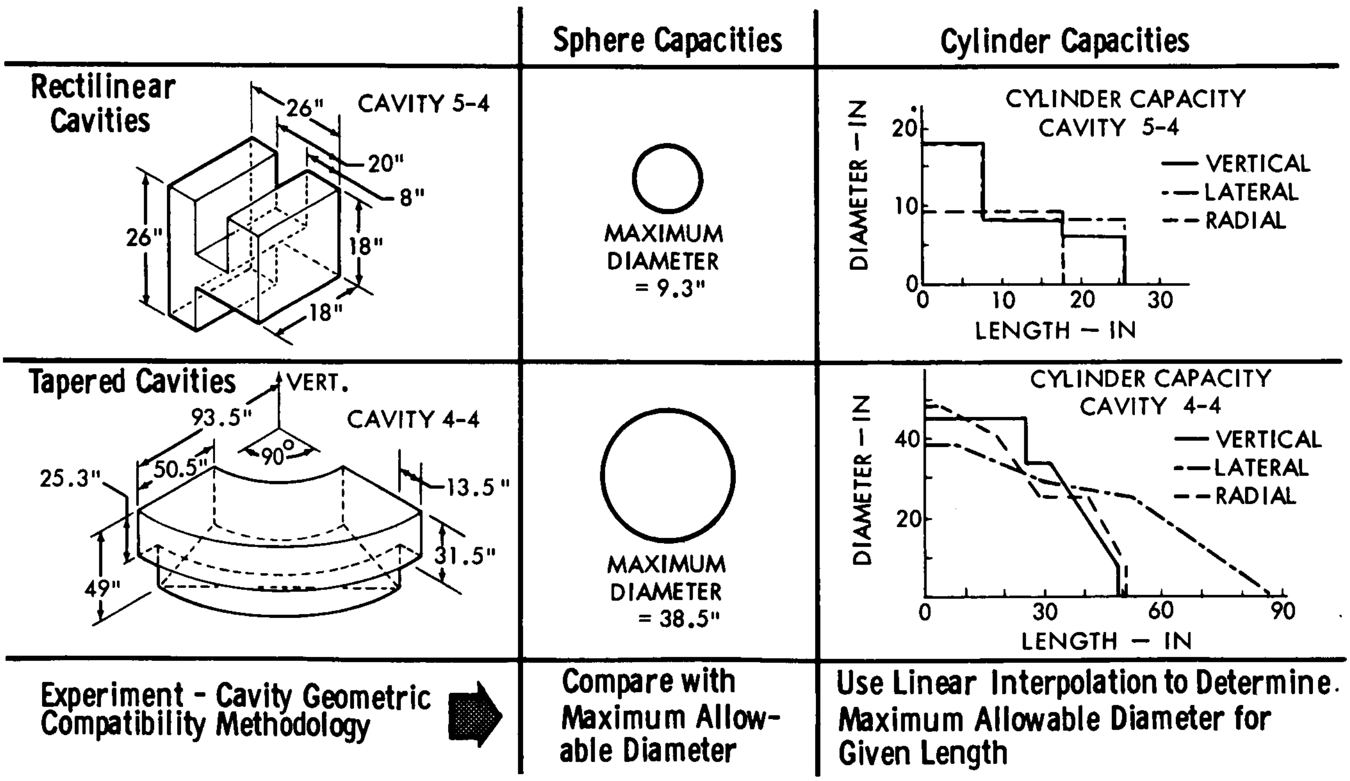


Figure 14-7 GEOMETRIC CAPACITIES COMPATIBILITY - SPHERE AND CYLINDER

be contained in any cavity, and there is obviously no need to specify an orientation. The methodology to determine the geometric compatibility of a spherical experimental payload consists in analytically checking whether the diameter of the payload is less than or equal to the maximum diameter of the sphere which the cavity will contain.

The cylinder capacity curves shown for a specified cavity in Figure 14-7 represent the maximum diameters of cylinders of given length and orientation

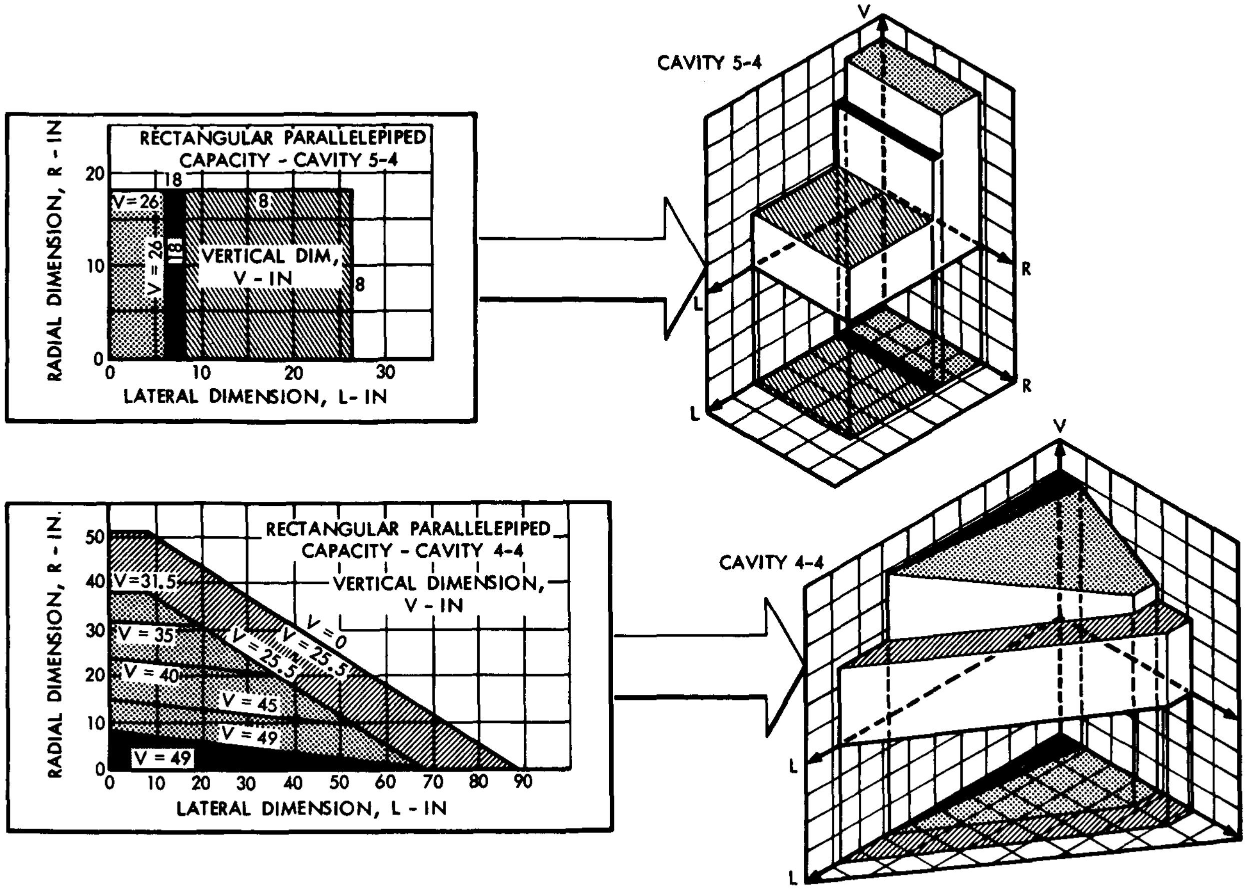


Figure 14-8 GEOMETRIC CAPACITY COMPATIBILITY - RECTANGULAR PARALLELEPIPED

which can be contained in the cavity. A difference between rectilinear and tapered cavities is now evident. The rectilinear cavity cylinder capacity curves consist of constant diameter steps, and the curves can be replaced by five distinct cylinders at least one of which will contain any cylinder that can be contained in the cavity. The tapered cavity cylinder capacity curves consist of sloped and curved lines which represent the simplest form in which the cylinder capacities for tapered cavities may be represented.

The methodology used to determine the geometric capacities compatibility of a cylindrical payload consists in the analytical interpolation,from these curves, of the maximum possible diameter corresponding to the length of the payload. The payload diameter must be less or equal to the maximum possible diameter for compatibility. If the orientation is specified, only one interpolation is required, but if no orientation is specified, as many as three interpolations may be required to determine whether a fit is possible.

14.6.6.2 Rectangular Parallelepiped Capacities

Because three dimensions are required to specify the size of a rectangular parallelepiped, the capacities for this standard shape payload are represented by surfaces. The rectangular parallelepiped capacity surfaces for the cavities given in Figure 14-7 are illustrated in Figure 14-8. These surfaces represent the maximum possible vertical dimension for each pair of lateral and radial dimensions which can be contained in the cavity. They are shown in Figure 14-8 in both contour and isometric form for clarity. Because each point on a surface corresponds to a rectangular parallelepiped and different orientations are given merely by the dimensions of the parallelepiped taken in different orders,

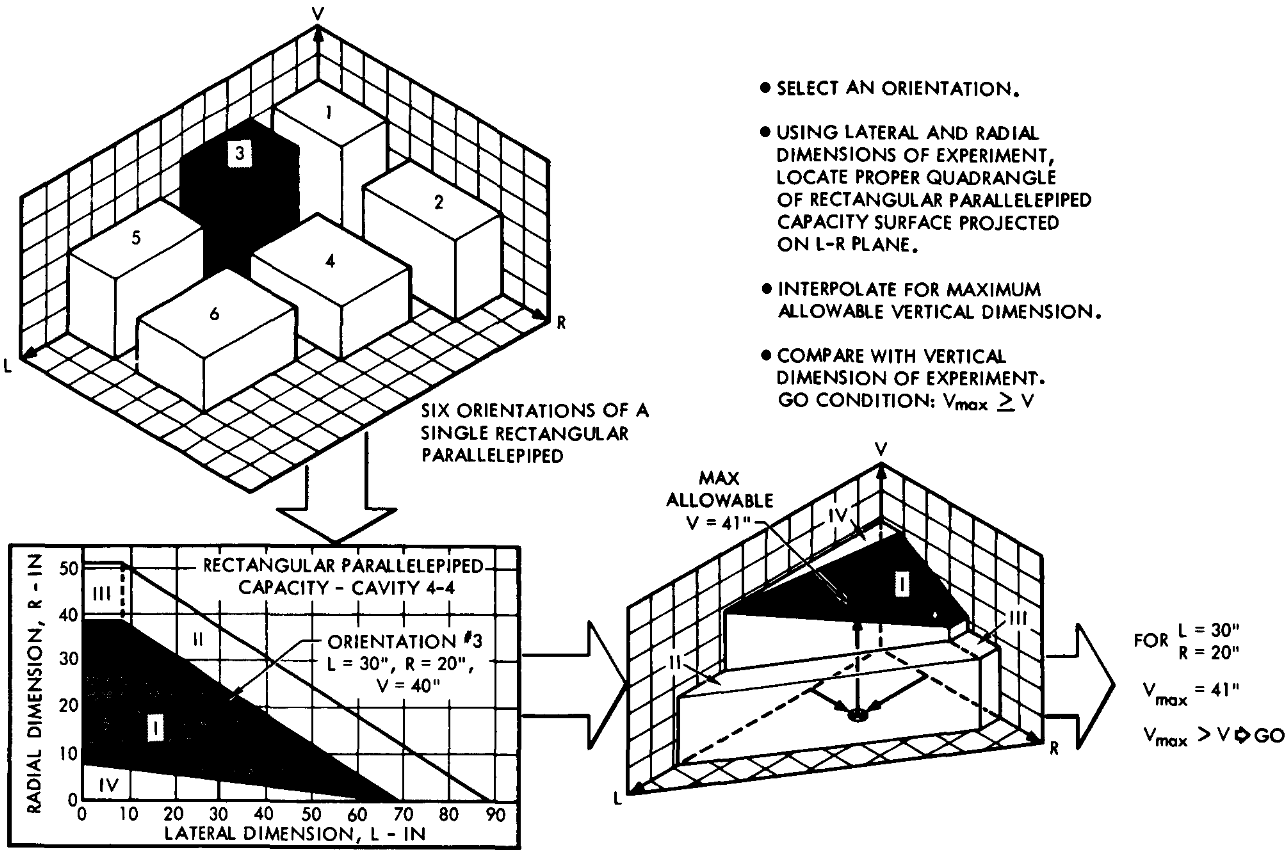


Figure 14-9 GEOMETRIC CAPACITY COMPATIBILITY - RECTANGULAR PARALLELEPIPED METHODOLOGY

all orientations (parallel to the L-R-V axes) are included on a single surface, which is generally discontinuous in the case of each cavity.

It is again noted in Figure 14-8 that there is a distinct difference between the forms of the rectilinear and tapered capacities. The rectangular parallelepiped capacity surfaces for rectilinear cavities are composed of planar horizontal rectangles (example cavity 5-4). These may be represented in a simpler manner. The surface in Figure 14-8 may be replaced by the dimensions for three rectangular parallelepipeds, at least one of which will contain any rectangular parallelepiped which can be contained in the cavity. The rectangular parallelepiped capacity surfaces for tapered cavities (example cavity 4-4) are composed of inclined planes and curved surfaces of irregular shape. They cannot be simplified further.

It is interesting to note that the rectangular parallelepiped capacity surface (shown in isometric form in Figure 14-8) is the original cavity distorted - beyond recognition in some cases - in such a way that its shape is simplified, its volume decreased, but its capacity for rectangular parallelepipeds maintained exactly.

The methodology used in the computer program to determine rectangular parallelepiped compatibility is illustrated in Figure 14-9. The six possible orientations of a given parallelepiped within a cavity in which its sides are parallel to the axes of the cavity are shown. The example given is a rectangular parallelepiped measuring 20 by 30 by 40 inches. If the orientation of a rectangular parallelepiped payload is not critical, each of the six orientations is tried until one is found which will allow the payload to be contained within the cavity. In some cases, there may be a required alignment for only one axis of the payload, e.g., a camera pointing outboard. There are nine possible alignments of this type, each corresponding to a distinct pair of orientations. The orientation pairs given in the following table apply to the

TABLE OF ORIENTATION PAIRS

PAYLOAD AXES

	20"	30"	40"
L	1, 2	3, 4	5, 6
R	3, 5	1, 6	2, 4
V	4, 6	2, 5	1, 3

example given in Figure 14-9. In this case, only two orientations must be tried in order to determine compatibility. When the alignment of two axes is specified, there is only one possible orientation to be checked for compatibility.

Once an orientation is selected, whether the orientation can be contained in a given cavity is determined in the following manner. The rectangular parallelepiped capacity surface is divided into plane quadrangles. This does not mean that the cavity is being approximated as having planar faces, because the rectangular parallelepiped capacity surfaces are very nearly planar even for cavities possessing extreme curvature in their faces. These plane quadrangles are then projected onto the L-R plane. An equation involving the coordinates of the vertices of the quadrangles is applied in the computer program to the L and R dimensions of the payload in order to determine which of the quadrangles in the L-R plane contains the point represented by these dimensions. This operation indicates which plane must be interpolated on to determine the maximum possible vertical dimensions corresponding to those lateral and radial dimensions. When this dimension is found, a comparison is made with the actual vertical dimension of the experimental payload. It must be less than or equal to the maximum possible vertical dimension to be contained in the cavity. If the payload cannot be contained and another orientation is possible, the procedure is repeated with the next orientation.

14.7 MODE I OUTPUT

The output of Mode I is in the form of printed results and, if specified, a compatibility library deck containing the required input for operation of Mode II.

Unless problem options specify otherwise, the printed output for each item consists of the following pages (examples given in the figures indicated):

1. Problem control input data (binary library tape overrides and problem options) (Figure 14-10)

2. Experimental payload description (operational and physical) input data (Figure 14-11)

3. Experiment/mission effectiveness array(s) input data (Figure 14-12)

4. Experiment/mission effectiveness data (orbital elements, mission parameters, and experiment effectiveness) (Figure 14-13)

5. Experimental payload/vehicle compatibility data (GO/NO-GO or, in some cases, warning-type statements for each vehicle-dependent criterion) (Figure 14-14)

6. Experimental payload/cavity compatibility data (GO/NO-GO for each cavity-dependent criterion and the final GO/NO-GO decision) (Figures 14-15 and 14-16). One page of output is given for each experimental payload-cavity combination.

7. Experimental payload summary data (compatibility and effectiveness) (Figure 14-17)

```
                    P R O G R A M   S E P T E R
           S A T U R N   E X P E R I M E N T A L   P A Y L O A D
     T E C H N I C A L   E V A L U A T I O N   A N D   R A T I N G

                            M O D E   I
                   S I N G L E   E X P E R I M E N T
   C O M P A T I B I L I T Y   A N D   E F F E C T I V E N E S S
                          A N A L Y S I S


                          FLIGHT SA-207
                  FROM LIBRARY TAPE 001437A01


                OVERRIDDEN MISSION/VEHICLE DATA

    NEW LAUNCH DATE                          31.0 MAR 1967
    NEW SOLAR DECLINATION                           4.2 DEG
    NEW LAUNCH TIME                                1000 EST
    NEW INCLINATION TO TERMINATOR                 152.8 DEG
    NEW PRIMARY PAYLOAD SEPARATION TIME           5300. SEC
    NEW PRIMARY MISSION DURATION                   7.0 DAYS
    NEW EXCESS PAYLOAD CAPABILITY                  1000. LB

                  CAVITIES TO BE DELETED
     1- 1    2- 4    6- 1

              FOR EACH EXPERIMENTAL PAYLOAD--
          MISSION/EFFECTIVENESS WILL BE DETERMINED
      MISSION/VEHICLE COMPATIBILITY WILL BE.DETERMINED

      A SEPTER-MODE II LIBRARY DECK WILL BE GENERATED
```

Figure 14-10 SEPTER - MODE I:
TITLE AND PROBLEM CONTROL DATA

```
               INFLIGHT EXPERIMENTAL PAYLOAD DESCRIPTION
                          EXPERIMENT  MS- 3

AVAILABLE  1.0 JAN 1967      INSTALLATION TIME   5.0 DAYS

DEPLOYMENT MODE  1

ELECTROMAGNETIC DATA
             BAND  LOW FREQ (MC)  HIGH FREQ (MC)  LEVEL (DBM)
  SENSITIVITY
              1        1.00          230.00         -10.0
              2      230.00          240.00         -30.0
              3      240.00          250.00         -90.0
              4      250.00          260.00         -30.0
              5      260.00        10000.00         -10.0
  SIGNALS
              1        1.00          240.00         -40.0
              2      240.00          250.00          40.0
              3      250.00          750.00         -20.0
              4      750.00        10000.00         -40.0

ACOUSTICS DATA
  NOISE TOLERANCE 150.0 DB     SUSCEPTIBLE COMPONENTS MASS   63.3 LB
  CORRECTIVE MASS PENALTIES MAY BE COMPUTED.

VIBRATIONS DATA
  CORNER FREQUENCIES  10.0,  75.0 CPS   TOLERANCE LEVELS  0.4, 10.0 G
  SUSCEPTIBLF COMPONENTS MASS   63.3 LB
  CORRECTIVE MASS PENALTIES MAY BE COMPUTED.

THERMAL DATA                                  PAD     LAUNCH     ORBIT
  LOW TEMPFRATURE TOLERANCE  (DEG-F)          14.0      0.        -50.0
  HIGH TEMPERATURE TOLERANCE (DEG-F)          75.0    240.0        65.0
  HEAT DISSIPATION RATE      (BTU/HR)         27.3     27.3       232.0
  TOTAL HEAT DISSIPATION          (BTU)

PAYLOAD MASS   154.3 LB,  VOLUME   3750. CU.IN.,  SHAPE  REC.PAR.
                                                  TYPE   AMORPHOUS

DIMENSIONS (IN)  LENGTH   3.0   WIDTH  10.0   HEIGHT  14.0
ALIGNMENT        NONE           NONE          NONE

DEVELOPMENT TIME  9.0 MO,  COST  $  545300.,  RELIABILITY  0.9600
```

Figure 14-11 SEPTER - MODE I:
IN-FLIGHT EXPERIMENTAL PAYLOAD DESCRIPTION DATA

MISSION EFFECTIVENESS ARRAY
EXPERIMENT MS- 3

MAXIMUM POSSIBLE EFFECTIVENESS 100.0 PERCENT

KEY	TABLE NO.	ORIGIN (X,Y)	SIZE (X)X(Y)	ELEMENTS (X)	(Y)	INTERP OPTION
	1	5, 1	7X 4	10	11	2

	1	2	3	4	5	6	7	8
1	2.00+00	7.10-01	8.64-01	8.80-01	8.90-01	8.93-01	8.97-01	9.00-01
2	1.60+00	5.20-01	8.30-01	8.90-01	9.10-01	9.20-01	9.27-01	9.30-01
3	1.20+00	2.40-01	5.30-01	8.40-01	9.50-01	9.68-01	9.80-01	9.90-01
4	1.00+00	3.00-02	3.10-01	5.80-01	9.30-01	9.75-01	9.90-01	1.00+00
5	0.	1.48+02	1.67+02	1.85+02	2.22+02	2.59+02	2.96+02	3.33+02
6	0.	0.	0.	0.	0.	0.	0.	0.
7	0.	0.	0.	0.	0.	0.	0.	0.
8	0.	0.	0.	0.	0.	0.	0.	0.
9	0.	0.	0.	0.	0.	0.	0.	0.
10	0.	0.	0.	0.	0.	0.	0.	0.
11	0.	0.	0.	0.	0.	0.	0.	0.
12	0.	0.	0.	0.	0.	0.	0.	0.
13	0.	0.	0.	0.	0.	0.	0.	0.
14	0.	0.	0.	0.	0.	0.	0.	0.
15	0.	0.	0.	0.	0.	0.	0.	0.
16	0.	0.	0.	0.	0.	0.	0.	0.
17	0.	0.	0.	0.	0.	0.	0.	0.
18	0.	0.	0.	0.	0.	0.	0.	0.
19	0.	0.	0.	0.	0.	0.	0.	0.
20	0.	0.	0.	0.	0.	0.	0.	0.
21	0.	0.	0.	0.	0.	0.	0.	0.
22	0.	0.	0.	0.	0.	0.	0.	0.
23	0.	0.	0.	0.	0.	0.	0.	0.
24	0.	0.	0.	0.	0.	0.	0.	0.
25	0.	0.	0.	0.	0.	0.	0.	0.

Figure 14-12 SEPTER - MODE I: MISSION EFFECTIVENESS ARRAY DATA

EXPERIMENTAL PAYLOAD/MISSION EFFECTIVENESS
EXPERIMENT MS- 3

DEPLOYMENT PARAMETERS

MODE	TIME (SEC)	THETA (DEG)	PHI (DEG)	DELTA V (KM/SEC)
0	604.77	-0.	-0.	-0.

MISSION PARAMETERS AND ORBITAL ELEMENTS

	SEMIMAJOR AXIS	=	6564.05	KM
	ECCENTRICITY	=	0.0046	-
	INCLINATION	=	30.00	DEG
	ARGUMENT OF PERIGEE	=	80.78	DEG
	TRUE ANOMALY	=	2.01	DEG
	TIME OF PERIGEE PASS.	=	5263.36	SEC
	PERIGEE LATITUDE	=	29.57	DEG
	PERIOD	=	5292.57	SEC
	APOGEE ALTITUDE	=	215.95	KM
**	PERIGEE ALTITUDE	=	155.81	KM
**	APOGEE/PERIGEE ALT.	=	1.39	-
	LONG. OF NODAL PASS.	=	200.38	DEG
	TIME OF NODAL PASSAGE	=	4083.44	SEC
	INCLINATION TO TERM.	=	140.59	DEG
	SOLAR DECLINATION	=	23.37	DEG
	LAUNCH MONTH	=	6	-
	LAUNCH YEAR	=	1967	-
	JULIAN DATE	=	2439656.5	-
	MISSION DURATION	=	14.00	DAYS
	LAUNCH TIME	=	9.00	HR

EFFECTIVENESS PARAMETERS

EFFECTIVENESS FACTORS

0.525

EXPERIMENT EFFECTIVENESS (PCT)

ABSOLUTE E	MAXIMUM EMAX	NORMALIZED E/EMAX
52.5	100.0	52.5

Figure 14-13 SEPTER - MODE I: EXPERIMENTAL PAYLOAD/MISSION EFFECTIVENESS, MISSION PARAMETERS AND ORBITAL ELEMENTS

EXPERIMENTAL PAYLOAD/VEHICLE COMPATIBILITY

EXPERIMENT MS- 3
VEHICLE SA-207

AVAILABILITY DATE/LAUNCH DATE BUFFER = 160. DAYS GO

ELECTROMAGNETIC INTERFERENCE

POSSIBLE EXPERIMENT-VEHICLE INTERFERENCE ON FOLLOWING BANDWIDTHS

9.90 TO 10.10 MC
230.00 TO 305.00 MC
2050.00 TO 2300.00 MC

POSSIBLE VEHICLE-EXPERIMENT INTERFERENCE ON FOLLOWING BANDWIDTHS

10.00 TO 10.50 MC
225.00 TO 295.00 MC
2200.00 TO 2300.00 MC

ACOUSTICS DATA

AC/VIB ZONE 1	MASS PENALTY REQUIRED =	0.9	GO
AC/VIB ZONE 2	MASS PENALTY REQUIRED =	0.9	GO

VIBRATIONS DATA

AC/VIB ZONE 1	MASS PENALTY REQUIRED =	5.5	GO
AC/VIB ZONE 2	MASS PENALTY REQUIRED =	9.2	GO

Figure 14-14 SEPTER - MODE I: EXPERIMENTAL PAYLOAD/VEHICLE COMPATIBILITY

EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

EXPERIMENT MS- 3
CAVITY 1- 1

DEPLOYMENT COMPATIBILITY

MODE GO *

ENVIRONMENTAL COMPATIBILITY

THERMAL
PAD GO
LAUNCH GO
ORBIT GO

ACOUSTIC CORRECTIVE MASS PENALTY OF 0.9 ADDED GO

VIBRATION CORRECTIVE MASS PENALTY OF 5.5 ADDED GO

OVERALL DECISION GO *

MASS COMPATIBILITY

STRUCTURAL LIMIT = 1000.0 TOTAL EXP.MASS= 160.8 GO *

GEOMETRIC COMPATIBILITY

AVAIL.VOL.=163966. REQ.VOL.= 3750. PCT.USED= 2.3 GO

STANDARD SHAPE--RECTANGULAR PARALLELEPIPED

L= 3.0	R= 10.0	V= 14.0	GO
L= 3.0	R= 14.0	V= 10.0	GO
L= 10.0	R= 3.0	V= 14.0	GO
L= 10.0	R= 14.0	V= 3.0	GO
L= 14.0	R= 3.0	V= 10.0	GO
L= 14.0	R= 10.0	V= 3.0	GO

OVERALL DECISION GO *

****
FINAL * GO * DECISION
****

Figure 14-15 SEPTER - MODE I: EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

```
          EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

                    EXPERIMENT  MI- 1
                        CAVITY   3- 6

DEPLOYMENT COMPATIBILITY

   MODE                                                 GO  *
   TIME                                                 GO  *

ENVIRONMENTAL COMPATIBILITY

   THERMAL
      PAD                                               GO
      LAUNCH                                            GO

   ACOUSTIC   CORRECTIVE MASS PENALTY OF   5.8 ADDED    GO

   VIBRATION  CORRECTIVE MASS PENALTY OF  35.0 ADDED    GO

                                OVERALL DECISION        GO  *

MASS COMPATIBILITY

   STRUCTURAL LIMIT = 1000.0  TOTAL EXP.MASS= 1122.5   NOGO *

GEOMETRIC COMPATIBILITY

    AVAIL.VOL.=171039.  REQ.VOL.= 81200.  PCT.USED=  47.5  GO

    STANDARD SHAPE--RECTANGULAR PARALLELEPIPED
                  L= 44.0  R= 22.0  V= 31.0             GO
                  L= 44.0  R= 31.0  V= 22.0            NOGO
                  L= 22.0  R= 44.C  V= 31.0            NOGO
                  L= 22.0  R= 31.0  V= 44.0            NOGO
                  L= 31.0  R= 44.0  V= 22.0            NOGO
                  L= 31.0  R= 22.0  V= 44.0             GO

                                OVERALL DECISION        GO  *

                       ********
                 FINAL * NOGO * DECISION
                       ********
```

Figure 14-16 SEPTER - MODE I:
EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

EXPERIMENTAL PAYLOAD/MISSION/VEHICLE COMPATIBILITY SUMMARY
EXPERIMENT MS- 3/FLIGHT SA-207

NORMALIZED MISSION EFFECTIVENESS 52.5 PERCENT   AVAILABILITY GO
POSSIBLE EMI YES

			EXPERIMENT/CAVITY COMPATIBILITY						
CAVITY	DEPL MODE	DEPL TIME	THERMAL	ACOUS	VIB	MASS ATTACH	VOLUME	GEOM	OVERALL
1- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
1- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
1- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 4	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 5	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 6	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 7	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 4	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 5	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 6	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 7	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 8	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 4	GO	N/A	GO	GO	GO	GO	GO	GO	GO
5- 1	GO	N/A	NOGO	GO	GO	NOGO	GO	GO	NOGO
5- 2	GO	N/A	NOGO	GO	GO	GO	NOGO	NOGO	NOGO
5- 3	GO	N/A	NOGO	GO	GO	GO	NOGO	NOGO	NOGO
5- 4	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 5	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 6	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 7	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 8	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
6- 1	GO	N/A	NOGO	GO	GO	NOGO	NOGO	NOGO	NOGO
6- 2	GO	N/A	NOGO	GO	GO	NOGO	GO	GO	NOGO
6- 3	GO	N/A	NOGO	GO	GO	NOGO	NOGO	NOGO	NOGO
6- 4	GO	N/A	NOGO	GO	GO	NOGO	GO	GO	NOGO
6- 5	GO	N/A	NOGO	GO	GO	NOGO	NOGO	NOGO	NOGO
7- 1	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 2	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 3	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 4	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 5	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO

Figure 14-17 SEPTER - MODE I:
EXPERIMENTAL PAYLOAD/MISSION/VEHICLE
COMPATIBILITY SUMMARY

15.0 PROGRAM SEPTER METHODOLOGY- MODE II

15.1 MODE II OPERATION

Mode II operation and analysis consists of the arrangement of multiple experimental payloads in the vehicle so that (1) a preferred order (priority) loading is used in the arrangement according to externally prepared preference list(s), (2) no payload-vehicle or payload-cavity or payload-payload incompatibilities exist, (3) the payload mass capability of the mission/vehicle is not exceeded, and (4) the near-maximum number of experimental payloads within the placement policy mechanics of the program are placed aboard the vehicle from the preference list. Mode II output, therefore, consists of an arrangement of experimental payloads aboard the vehicle in the preferred order with no incompatibilities.

15.2 LIBRARIES

In the Mode II operation, definition-type library data are supplied as a part of the Mode I output. These data are largely the same as those provided by the Mission/Vehicle/Primary Payload Characteristics Library and the Experimental Payload Characteristics Library for Mode I. In Mode II, however, some data are deleted (e.g., mission characteristics) and other data are added as a result of computations completed in the Mode I operation. The additional library data which are of most significance for the Mode II operation are those which specify the compatible cavities for each experimental payload. Other additional data are the mass penalties calculated as a result of acoustics/vibration deficiencies for each experimental payload.

15.3 EXTERNAL ANALYSIS

The external analysis required for the Mode II operation consists in compiling preference list(s). These lists are simply a tabulation of experimental payload identifications in a preferred order of loading in a given vehicle for a given mission. Although the compatibility and effectiveness output data of Mode I are obviously provided to assist the user in the formulation of preference lists, any additional data or methods of establishing priority may be used in arriving at a preference list. Several sets of preference lists may be formulated for a given set of experiments.

15.4 PROBLEM INPUT AND CONTROLS

Inputs for the operation of Mode II consists of a preference list, a compatibility library deck, and problem control data. The preference list (card deck) is prepared by the user of the program in order to establish the desired order (priority) in which experimental payloads are to be loaded aboard the vehicle. The compatibility library card deck generated as output from the Mode I operation, contains: (1) the vehicle, cavity, and experimental payload description data obtained from the Mission/Vehicle/Primary Payload Characteristics Library and the Experimental Payload Characteristics Library of Mode I, and (2) computed compatibility data from Mode I, i.e., the identification of all cavities with which an experimental payload is compatible (final GO decision), mass penalties for acoustics/vibration tolerance deficiencies (if any), and any library override data which may have been used in Mode I. Problem control and override data (from card decks) consist of, for example, predetermined placements for arbitrary experiments and overrides for library data.

15.5 MULTIPLE PAYLOAD ARRANGEMENT LOGIC

The purpose of the multiple payload arrangement analysis is to determine the arrangements of payloads in cavities throughout the vehicle in such a manner that no incompatibilities occur within any cavity and that the payloads are loaded in a preferred order. Arrangements which will allow the greatest number of payloads within the overall mass and volume limits of the vehicle are the desired result.

The arrangement analysis is an optimization problem, but optimization methods (except for complete enumeration, which is not feasible because of the extremely large number of possible arrangements) are not readily applicable Consideration of the problem indicates that optimal arrangements will not usually be unique. This conclusion is evident because the mass attachment limit for a cavity divided by its available volume is generally a smaller number than the densities of typical experiments. In addition, the sum of the cavity mass attachment limits is usually greater than the payload capability of the vehicle. This indicates that in loading the vehicle, mass limits will be encountered prior to volume limits. It is further implied that the maximum number of experiments which can be loaded will depend more on the payload capability of the vehicle than on the available volume or arrangement of the payloads in the vehicle. Consequently, an attempt to apply a true optimization process to the problem appears to involve a degree of effort not justified by the results desired from this study.

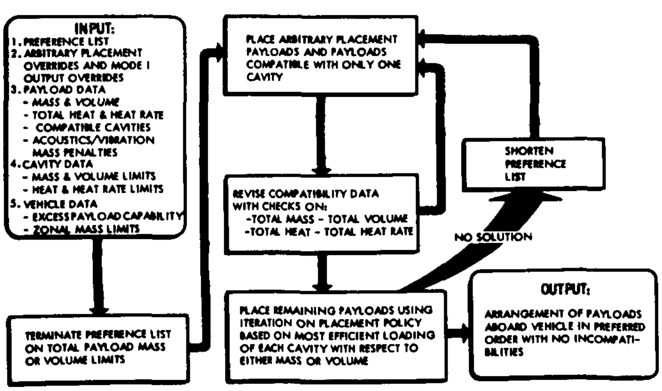


Figure 15-1 MULTIPLE PAYLOAD ARRANGEMENT LOGIC

In Figure 15-1, a basic outline of an alternate approach aimed at satisfying all constraints and directly searching for one of the non-unique "optimal" solutions or arrangements is shown. The approach is simple in concept, but its application is complex. It consists of three iterations, two of them contained within the third.

15.5.1 Compatibility Determined and Arbitrary Placements

The first iteration in Mode II consists in the placement of any payloads which have no choice of location. These placements may be arbitrary, as determined by an override option, or they may be compatibility-determined by Mode I operation. This procedure must be iterated because the placement of these payloads may limit or eliminate the choice of others. In addition, hidden incompatibilities within the preference list may be discovered. When this occurs, the lower priority incompatible payloads are dropped from the list. When all remaining payloads have a choice of locations, the first iteration is completed.

15.5.2 Optimal Arrangements

The second iteration consists of a computational block based upon an arbitrary set of rules for placement (placement policy) of experimental payloads. The placement policy is applied cavity-by-cavity in search of possible arrangements. Iteration is required in this search because the placement policy is arbitrary and is influenced by the order in which cavities are selected. If no solution is found, the third iteration is required because the preference list must be shortened, and the entire procedure is repeated.

15.6 MODE II OUTPUT

The output of Mode II is in the form of printed results consisting of the following types of data (unless problem options specify otherwise) for each problem:

1. Title page and mission identification data. Example results are shown in Figure 15-2.

2. Problem control and preference list data. Example results are shown in Figure 15-3.

3. Identification of incompatible experimental payloads dropped from the preference list. Example results are shown in Figure 15-4.

4. Compatibility array data. Example results are shown in Figure 15-5.

5. Description of multiple experimental payload arrangements by cavities:

   a. Identification of experimental payloads contained in each cavity.

   b. Total mass, volume, etc., contained in each cavity.

   c. Remaining mass, volume, etc., in each cavity.

   Example results are shown in Figure 15-6.

6. Summary table of experimental payload arrangements:

   a. Identification of the cavity in which each experimental payload is contained (if placed).

   b. Identification of experimental payloads that have not been placed in a cavity.

   c. Rank (priority) of each experimental payload in the preference list.

   d. Identification of vacant cavities.

   An example summary table is shown in Figure 15-7.

P R O G R A M   S E P T E R

S A T U R N   E X P E R I M E N T A L   P A Y L O A D
T E C H N I C A L   E V A L U A T I O N   A N D   R A T I N G

M O D E   I I
M U L T I P L E   E X P E R I M E N T
C O M P A T I B I L I T Y   A N D   A R R A N G E M E N T
A N A L Y S I S

FLIGHT SA-207
LAUNCH 15.0 JUN 1967
EXCESS PAYLOAD CAPABILITY   10000.0 LB

Figure 15-2 SEPTER - MODE II:
TITLE AND MISSION IDENTIFICATION DATA

PREFERENCE LIST NUMBER 1

THE PLACEMENT POLICY WILL BE BASED ON MASS. THE CAVITIES WILL BE REORDERED 22 TIMES BEFORE THE PREFERENCE LIST IS SHORTENED. A MAXIMUM OF 88 ARRANGEMENTS WILL BE ATTEMPTED. THE FOLLOWING IS THE PREFERRED ORDER OF PLACEMENT.

PREFERENCE	EXPERIMENT
1	SDT- 1
2	SDT- 2
3	SDT- 3
4	SDT- 4
5	SDT- 5
6	MS- 1
7	MS- 2
8	MS- 3
9	MS- 4
10	MS- 5
11	MI- 1
12	MI- 2
13	MI- 3
14	MI- 4
15	MI- 5
16	M- 1
17	M- 2
18	M- 3
19	M- 4
20	M- 5
21	OEA- 1
22	OEA- 2
23	OEA- 3
24	OEA- 4
25	OEA- 5
26	SLG- 1
27	SLG- 2
28	SLG- 3
29	SLG- 4
30	SLG- 5

Figure 15-3 SEPTER - MODE II:
PROBLEM CONTROL AND PREFERENCE LIST DATA

EXPERIMENT MS- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT MI- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT MI- 3 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT MI- 5 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT OEA- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT SLG- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT SLG- 3 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT SLG- 4 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

Figure 15-4 SEPTER - MODE II:
IDENTIFICATION OF INCOMPATIBLE EXPERIMENTAL PAYLOADS

COMPATIBILITY ARRAY

THIS COMPATIBILITY ARRAY WILL BE USED IN DETERMINING PLACEMENTS

(XX COMPATIBLE   00 INCOMPATIBLE)

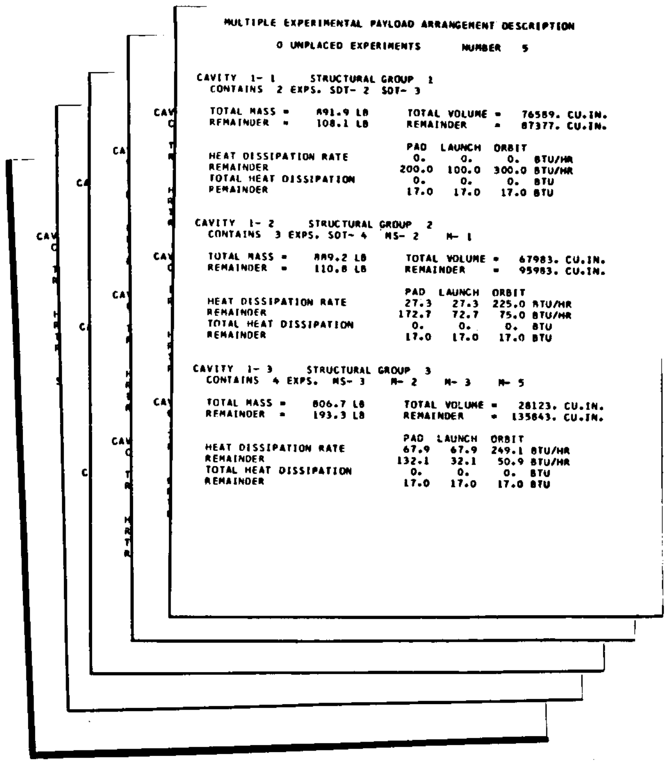
CAVITY ZONE AND NUMBER

		1	1	1	2	2	2	2	2	2	2	3	3	3	3	3	3	3	3	4	4	4	4
		1	2	3	1	2	3	4	5	6	7	1	2	3	4	5	6	7	8	1	2	3	4
PREF	EXP																						
1	SDT- 1	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	00
2	SDT- 2	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
3	SDT- 3	XX	XX	XX	00	XX	00	00	00	00	00	00	XX	00	XX	00	XX	00	XX	00	00	00	00
4	SDT- 4	XX	XX	XX	00	XX	00	00	00	00	00	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
5	SDT- 5	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	00
6	MS- 2	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
7	MS- 3	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
8	MS- 4	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
9	MS- 5	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
10	MI- 2	XX	XX	XX	XX	XX	XX	XX	00	00	XX	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
11	MI- 4	XX	XX	XX	00	XX	00	00	00	00	00	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
12	M- 1	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
13	M- 2	XX	XX	XX	00	00	00	00	00	00	00	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
14	M- 3	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
15	M- 4	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	XX
16	M- 5	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
17	OEA- 2	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	XX
18	OEA- 3	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	00	00	00	00
19	OEA- 4	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	00	XX	00	XX	00	XX	00	00	00	00
20	OEA- 5	XX	XX	XX	XX	XX	XX	XX	00	00	XX	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
21	SLG- 2	XX	XX	XX	00	00	00	00	00	00	00	XX	XX	XX	XX	XX	XX	XX	XX	00	00	00	00
22	SLG- 5	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX

28 INCOMPATIBLE CAVITIES

5- 1	5- 2	5- 3	5- 4	5- 5	5- 6	5- 7	5- 8
6- 1	6- 2	6- 3	6- 4	6- 5	7- 1	7- 2	7- 3
7- 4	7- 5	7- 6	7- 7	7- 8	7- 9	7-10	7-11
7-12	7-13	7-14	7-15				

Figure 15-5 SEPTER - MODE II:
COMPATIBILITY ARRAY - INCOMPATIBLE CAVITIES



```
MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT DESCRIPTION

          0 UNPLACED EXPERIMENTS          NUMBER   5

CAVITY  1- 1      STRUCTURAL GROUP  1
  CONTAINS  2 EXPS. SDT- 2  SDT- 3

  TOTAL MASS =     891.9 LB      TOTAL VOLUME =   76589. CU.IN.
  REMAINDER  =     108.1 LB      REMAINDER    =   87377. CU.IN.

                                 PAD   LAUNCH  ORBIT
  HEAT DISSIPATION RATE           0.      0.      0.  BTU/HR
  REMAINDER                     200.0   100.0   300.0 BTU/HR
  TOTAL HEAT DISSIPATION          0.      0.      0.  BTU
  REMAINDER                      17.0    17.0    17.0 BTU

CAVITY  1- 2      STRUCTURAL GROUP  2
  CONTAINS  3 EXPS. SDT- 4  MS- 2   M- 1

  TOTAL MASS =     889.2 LB      TOTAL VOLUME =   67983. CU.IN.
  REMAINDER  =     110.8 LB      REMAINDER    =   95983. CU.IN.

                                 PAD   LAUNCH  ORBIT
  HEAT DISSIPATION RATE          27.3    27.3   225.0 BTU/HR
  REMAINDER                     172.7    72.7    75.0 BTU/HR
  TOTAL HEAT DISSIPATION          0.      0.      0.  BTU
  REMAINDER                      17.0    17.0    17.0 BTU

CAVITY  1- 3      STRUCTURAL GROUP  3
  CONTAINS  4 EXPS.  MS- 3    M- 2    M- 3    M- 5

  TOTAL MASS =     806.7 LB      TOTAL VOLUME =   28123. CU.IN.
  REMAINDER  =     193.3 LB      REMAINDER    = 135843. CU.IN.

                                 PAD   LAUNCH  ORBIT
  HEAT DISSIPATION RATE          67.9    67.9   249.1 BTU/HR
  REMAINDER                     132.1    32.1    50.9 BTU/HR
  TOTAL HEAT DISSIPATION          0.      0.      0.  BTU
  REMAINDER                      17.0    17.0    17.0 BTU
```

Figure 15-6 SEPTER - MODE II: MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT DESCRIPTION

MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT SUMMARY 5

(XX CONTAINED IN -- NOT CONTAINED IN OO UNPLACED)

CAVITY ZONE AND NUMBER

PREF	EXP	1 1	1 2	1 3	2 1	2 5	2 7	3 1	3 2	3 4	3 6	3 8	4 1	4 2
1	SDT- 1	--	--	--	--	--	--	--	--	--	--	--	XX	--
2	SDT- 2	XX	--	--	--	--	--	--	--	--	--	--	--	--
3	SDT- 3	XX	--	--	--	--	--	--	--	--	--	--	--	--
4	SDT- 4	--	XX	--	--	--	--	--	--	--	--	--	--	--
5	SDT- 5	--	--	--	--	--	--	--	--	--	--	--	--	XX
6	MS- 2	--	XX	--	--	--	--	--	--	--	--	--	--	--
7	MS- 3	--	--	XX	--	--	--	--	--	--	--	--	--	--
8	MS- 4	--	--	--	XX	--	--	--	--	--	--	--	--	--
9	MS- 5	--	--	--	--	XX	--	--	--	--	--	--	--	--
10	MI- 2	--	--	--	--	--	--	--	XX	--	--	--	--	--
11	MI- 4	--	--	--	--	--	--	--	--	XX	--	--	--	--
12	M- 1	--	XX	--	--	--	--	--	--	--	--	--	--	--
13	M- 2	--	--	XX	--	--	--	--	--	--	--	--	--	--
14	M- 3	--	--	XX	--	--	--	--	--	--	--	--	--	--
15	M- 4	--	--	--	--	--	--	--	--	--	--	--	XX	--
16	M- 5	--	--	XX	--	--	--	--	--	--	--	--	--	--
17	OEA- 2	--	--	--	--	--	--	--	--	--	--	--	--	XX
18	OEA- 3	--	--	--	XX	--	--	--	--	--	--	--	--	--
19	OEA- 4	--	--	--	--	--	--	--	--	--	XX	--	--	--
20	OEA- 5	--	--	--	--	--	--	--	--	--	--	XX	--	--
21	SLG- 2	--	--	--	--	--	--	XX	--	--	--	--	--	--
22	SLG- 5	--	--	--	--	--	XX	--	--	--	--	--	--	--

37 VACANT CAVITIES

2- 2	2- 3	2- 4	2- 6	3- 3	3- 5	3- 7	4- 3
4- 4	5- 1	5- 2	5- 3	5- 4	5- 5	5- 6	5- 7
5- 8	6- 1	6- 2	6- 3	6- 4	6- 5	7- 1	7- 2
7- 3	7- 4	7- 5	7- 6	7- 7	7- 8	7- 9	7-10
7-11	7-12	7-13	7-14	7-15			

Figure 15-7 SEPTER - MODE II: MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT SUMMARY

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