SATURN IN-FLIGHT EXPERIMENTAL PAYLOAD STUDY

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MID-TERM PROGRESS REPORT

GENERAL DYNAMICS
Fort Worth Division

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SATURN IN-FLIGHT EXPERIMENTAL PAYLOAD STUDY

Contract NAS8-20236
Mid-Term Progress Report

Prepared for the

GEORGE C. MARSHALL SPACE FLIGHT CENTER

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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SUMMARY

This document is the mid-term progress report on the Saturn In-Flight Experimental Payload Study. This study is being conducted 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 NAS8-20236. During the first half of the program, the following tasks were accomplished.

- Thirty representative experiments from six different scientific and technological categories were selected and ancillary subsystems were defined for each.
- Conceptual design drawings were prepared, and each experiment was defined in terms of mass, operational, environmental, shape, and volume requirements.
- The experiment effectiveness was determined for an initial group of experiments as a function of initial orbital elements.
- A summary table of flight regimes and experiment operations was prepared for use in determining the orbital elements which influence experimental effectiveness.
- Forty-eight potential inflight experiment locations were selected and defined on the base-line Saturn IB/Apollo configuration.
- The Saturn IB/Apollo mission types were defined, and pertinent mission characteristics of individual Saturn missions were analyzed.
- Computer methodologies to be used in computer program SEPTER (Saturn Experimental Payload Technical Evaluation and Rating) were established for determination of experiment payload/cavity location compatibility criteria, for determination of in-flight experiment effectiveness, and for multiple payload arrangement compatibility analysis.
- Summary tables of experimental payloads and cavities characteristics were prepared for incorporation into computer programs library.
- The subroutines were coded for the determination of orbital elements for the modes of payload deployment to be considered.

The second portion of the study will be devoted to the following: (1) performance of experiment reliability, development schedule, and cost analyses, (2) synthesis of ancillary subsystem characteristics of arbitrary experiments, (3) preparation of computer program libraries, (4) coding of the subroutines for the determination of experimental payload/cavity compatibility criteria, (5) refinement and coding of the computer methodology for determination of inflight experiment effectiveness and multiple payload arrangement compatibility analysis, (6) assembly and checkout of program SEPTER, (7) development and coding of support program(s) for experiment/mission effectiveness, and (8) coding of the support program for arbitrary experiment synthesis.

CONTRIBUTORS

The following personnel at the Fort Worth Division of General Dynamics have made significant technical contribution to this report.

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K. A. Pinter	Definition and Analysis of Saturn Mission Characteristics
J. A. McCabe	Experiment Package Propulsion System Selection and Sizing

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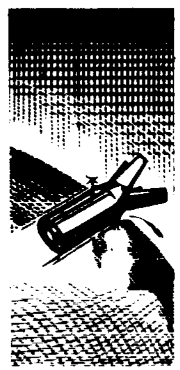

INTRODUCTION

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STUDY OBJECTIVES AND APPROACH

The basic objective set for the Saturn In-Flight Experimental Payload Study is to provide NASA with a computer program which can be used as a management tool to evaluate numerous potentially attractive space experiments that constitute possible secondary inflight payloads for the Saturn family of launch vehicles.

To attain this overall study objective, two major study tasks have been specified: (1) an analysis of (a) the physical characteristics of sensors and associated equipments for use as possible experimental payloads on Saturn-class vehicles and (b) the mission effectiveness values and sensitivities of these experiments relative to their deployment in an optimal orbit and (2) the development of a computerized methodology for the technical evaluation and rating of these potential inflight experimental payloads.

The technical approach used throughout the study is based on the development of Program SEPTER, where SEPTER is an acronym for Saturn Experimental Payload Technical Evaluation and Rating. There are two fundamental criteria which are employed in Program SEPTER in evaluating the experiments that are being considered for possible inclusion on a Saturn flight: (1) experiment/mission effectiveness and (2) physical compatibility of the experiments with possible locations aboard the vehicle. 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 in terms of the thermal, acoustic, vibration, and electromagnetic environments.

STUDY OBJECTIVES AND APPROACH

- **Basic Objective**

  TO PROVIDE NASA WITH A COMPUTER PROGRAM TO EVALUATE VARIOUS IN-FLIGHT EXPERIMENTAL PAYLOADS FOR POSSIBLE INCLUSION ON THE SATURN LAUNCH VEHICLES

- **Study Tasks**

  I - TO ANALYZE THE PHYSICAL CHARACTERISTICS AND MISSION SENSITIVITIES OF EXPERIMENTS FOR IN-FLIGHT PAYLOADS

  II - TO DEVELOP A COMPUTER METHODOLOGY FOR THE TECHNICAL EVALUATION AND RATING OF IN-FLIGHT EXPERIMENTAL PAYLOADS.

- **Approach:**

  DEVELOP PROGRAM **SEPTER** (**S**ATURN **E**XPERIMENTAL **P**AYLOAD **T**ECHNICAL **E**VALUATION AND **R**ATING)

- **Evaluation Criteria:**

  - PHYSICAL COMPATIBILITY
  - EXPERIMENT/MISSION EFFECTIVENESS

SATURN EXPERIMENTAL PAYLOAD TECHNICAL EVALUATION AND RATING

COMPUTER PROGRAM - SEPTER

The overall structure and the key concepts of Program SEPTER are shown on the facing page. This program will be developed in the study and will contain provisions for operating in two basic modes. In the Mode I operation, the compatibility and effectiveness of single experiments will be determined. In the Mode II operation, arrangement configurations and compatibility of multiple experiments will be analyzed, and desirable arrangement will be determined.

Data used in the single experiment or Mode I analysis consist of mission, launch vehicle/primary payload, and experiment identifications and associated information, such as experiment deployment mode, etc. These identifications will result in selections from the libraries of the mission profile, the potential experimental payload locations aboard the vehicle, and the experiments to be considered in the particular "mission." The program is then used to compare the libraries of mission/vehicle/primary payload characteristics and experiments characteristics with these data and to determine the compatibility and effectiveness of each individual experiment. The Mode I output consists of a listing of the experiments, along with information on their individual overall go/no-go compatibility, degree of compatibility or incompatibility and effectiveness.

The Mode I output data will be examined by NASA management, and a preferential order of experiment placement will be established. In addition to this preferential order listing, provisions are also being made to specify particular arbitrary locations for individual experiments.

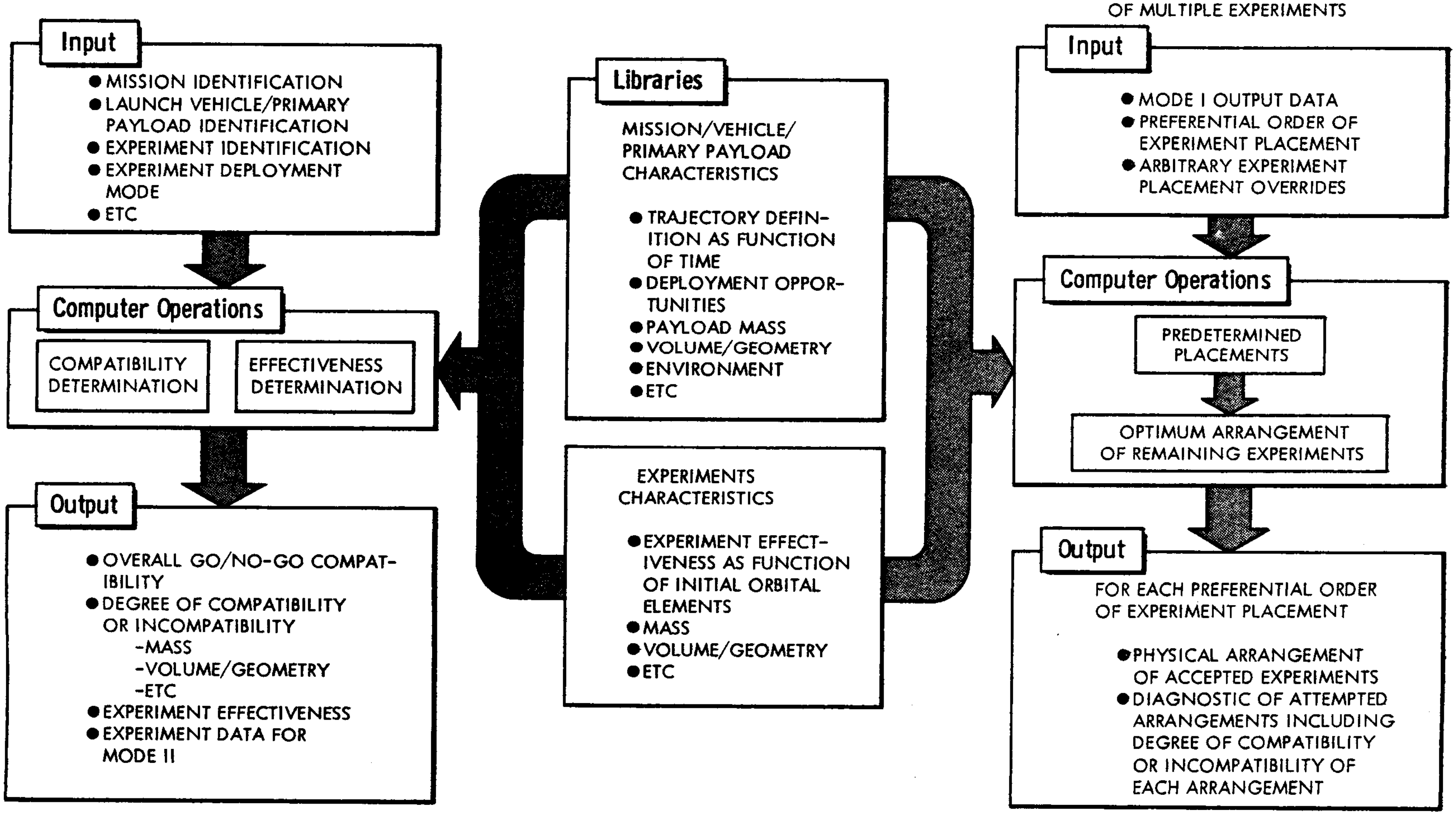
In the multiple experiment or Mode II operation, experiments for which predetermined placements have not been specified are arranged in the remaining cavity locations so as to maximize the number of experiments which can be accommodated.

SATURN EXPERIMENTAL PAYLOAD TECHNICAL EVALUATION AND RATING

Computer Program - SEPTER

MODE I OPERATION
DETERMINATION OF COMPATIBILITY AND EFFECTIVENESS OF SINGLE EXPERIMENTS

MODE II OPERATION
ANALYSIS OF ARRANGEMENT CONFIGURATIONS AND DETERMINATION OF COMPATIBILITY OF MULTIPLE EXPERIMENTS



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GUIDELINES AND GROUNDRULES

A number of guidelines and groundrules were specified 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. These guidelines and groundrules are briefly discussed in the following paragraphs.

The space experiments to be considered in the study represent secondary mission objectives; that is, these experiment packages are potential payloads, and they have been designed to attain specified objectives associated with the primary experimental payload. For example, the primary missions which are being 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, Lunar Excursion Module) is the primary payload. Additional experimental packages carried on these flights would then be secondary payloads.

The Fort Worth Division of General Dynamics acknowledges the prerogative and responsibility of NASA to define and approve inflight 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 inflight experiments. In establishing configuration designs for these experiment packages, primary emphasis was placed on self-contained packages, that is, consideration was not to be given to using the support capabilities of on-board equipments or to the possibility of sharing subsystems among experiments. The analysis of physical compatibility will be basically performed by considering completely self-contained packages; however, "self-contained packages" exclusive of power and communications subsystems, will also be treated. The experiment packages are to be designed to assure that they do not in any way interfere with the primary payload. Furthermore the package designs are to be 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.

Although other vehicle configurations should eventually be included in the launch vehicle/primary payload characteristics library, the Saturn IB/Apollo, including the Command Module, the Service Module, and the Lunar Excursion Module, was chosen as the baseline configuration.

GUIDELINES AND GROUND RULES

- Only "secondary" experimental payloads are to be considered, i. e., the mission has been designed for accomplishment of specific objectives associated with the primary experimental payload

- Experiments are to be defined only to establish representative experiment package and effectiveness requirements

- Primary emphasis is to be placed on analysis of self-contained experiment packages
  - PHYSICAL COMPATIBILITY ANALYSIS OPTIONS
    - SELF-CONTAINED
    - SELF-CONTAINED EXCLUSIVE OF POWER AND COMMUNICATION SUBSYSTEMS

- Non-interference of the secondary payload with the primary payload must be assured

- Only a minimum of astronaut participation will be allowed

- The baseline configuration is the Saturn IB/Apollo (CM, SM, LEM)

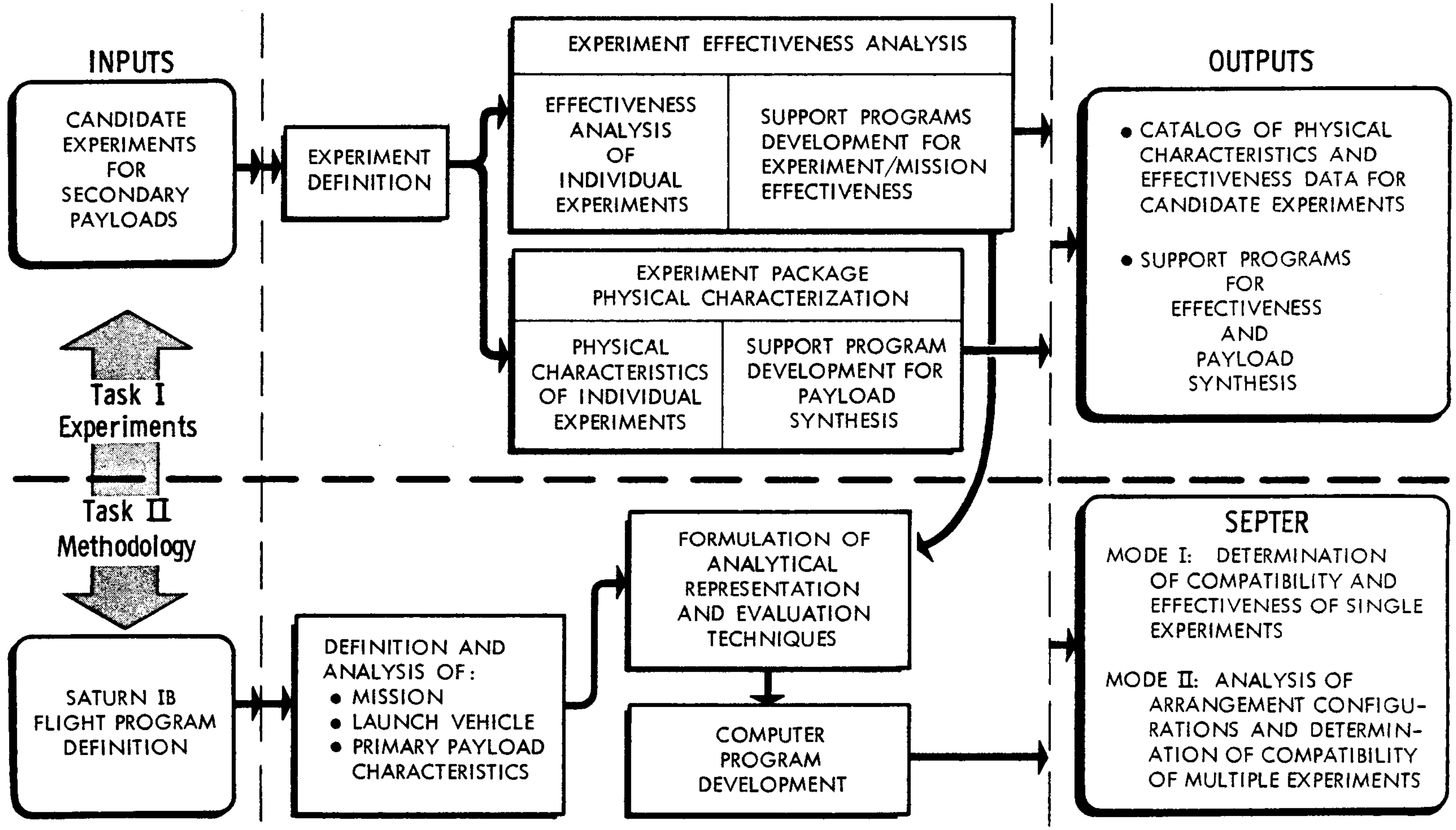
BASIC PROGRAM PLAN

The basic program plan shown on the facing page has been proposed by the Fort Worth Division of General Dynamics in order to achieve the objectives set for this study. The use of this approach will permit (1) an analysis of the physical characteristics and mission sensitivity of experiments of inflight payloads for Saturn-class vehicles and (2) the determination of a computer methodology for the technical evaluation and rating of these inflight experimental payloads. The proposed 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).

The Task I studies are to be devoted to (1) a thorough definition of the objectives, data acquisition requirements, and sensors for each of a group of representative experiments; (2) establishment of the physical characteristics of each individual experiment by synthesizing self-contained experiment packages on the basis of sensor requirements; (3) analysis of experiment effectiveness variations as a function of experiment-deployment orbital elements; and (4) computer mechanization of the computation of effectiveness values and physical characteristics.

The Task II studies are to be devoted to (1) a definition and analysis of the relevant mission, vehicle, and primary payload characteristics of the vehicle configurations to be considered; (2) development of analytical representations to allow the computer program to evaluate and rate single-experiment compatibility/effectiveness and to analyze multiple arrangement/compatibility; and (3) formulation of the computer program logic, the input and output data requirements and formats, the library data formats, and the options and modes of operation which, when combined with the analytical representations, will yield the operable computer program.

BASIC PROGRAM PLAN



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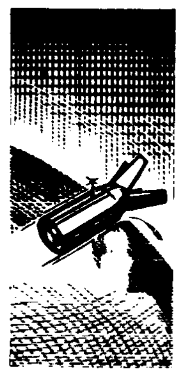
Part II



IN-FLIGHT EXPERIMENT DEFINITIONS

DEFINITION OF EXPERIMENTS

As has been indicated previously, the Fort Worth Division of General Dynamics recognized NASA's responsibility for the definition of potentially attractive space experiments to be carried aboard the Saturn-class of vehicles. However, in order to develop a workable computer methodology for the evaluation of such experiments, it was necessary 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 to define these example experiments is outlined on the facing page.

The document "NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights" was reviewed and the 85 experiments listed in this report were examined to determine which of the experiments had a potential application as a secondary inflight experiment. A number of the experiment "titles" were eliminated from consideration because of their incompatibility with the guidelines and groundrules established for the study. For example, in a number of the experiments, either a large payload capability or an excessive amount of astronaut participation was required. It was found that approximately 65 of the 85 experiment "titles" contained in this document could be defined as secondary inflight experiments.

Each of these 65 experiments were 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 experiments were selected from the list of 65 as representative. This list contained 5 experiments in each of the six scientific and technological experiment categories indicated on the facing page. Detailed objectives, data acquisition requirements, and package physical characteristics were determined for each of the 30 experiments.

DEFINITION OF EXPERIMENTS

GUIDELINE: Experiments Are to be Defined Only to Establish Requirements of Representative Experiment Packages and Experiment/Mission Effectiveness

APPROACH:

Examine NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights

(85 Experiments)

→

Prepare A Preliminary List of Experiments Which Have A Potential Application As Secondary In-Flight Experiments

(65 Experiments)

→

For A Representative Sample of Potentially Applicable Experiments Define

- OBJECTIVES
- DATA ACQUISITION REQUIREMENTS
- PACKAGE PHYSICAL CHARACTERISTICS

(30 Experiments)

- **SCIENTIFIC AND TECHNOLOGICAL EXPERIMENT CATEGORIES**

1. SYSTEMS DEVELOPMENT AND TEST
2. MATERIALS AND STRUCTURES
3. MULTISPECTRAL IMAGERY OF THE EARTH AND ORBITING OBJECTS
4. SOLID/LIQUID/GAS BEHAVIOR
5. MICROORGANISMS
6. OBSERVATIONS OF THE EARTH'S ATMOSPHERE, SPACE ENVIRONMENT, ASTRONOMICAL OBSERVATIONS

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LIST OF SELECTED IN-FLIGHT EXPERIMENTS

The experiments which were selected for inclusion in the experiment characteristics library of Program SEPTER are listed on the following three charts. These experiments are grouped under the six Scientific and Technological Experiment Categories which were established by the Fort Worth Division to insure broad coverage of the scientific and technical disciplines outlined in the document "NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights." These categories are listed below.

1. Systems Development and Test
2. Materials and Structures
3. Multi-Spectral Imagery of the Earth and Orbiting Objects
4. Solid/Liquid/Gas Behavior
5. Microorganisms
6. Observations of the Earth's Atmosphere, the Space Environment, and Astronomical Observation

It should be emphasized that the scientific and technological categories contained in "NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights" which have not been included here were eliminated from consideration as potential secondary in-flight experiments only because they did not conform to the guidelines and groundrules established for this particular study.

LIST OF SELECTED IN-FLIGHT EXPERIMENTS - 1

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

LIST OF SELECTED IN-FLIGHT EXPERIMENTS - 2

3. Multi-Spectral Imagery of the Earth and Orbiting Objects

MI-1. MULTI-SPECTRAL 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

LIST OF SELECTED IN-FLIGHT EXPERIMENTS - 3

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 Observations

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

RELATIONSHIP OF THE 30 EXPERIMENTS SELECTED BY GENERAL DYNAMICS FOR STUDY TO THE NASA MAJOR SCIENTIFIC/TECHNICAL AREAS OF INTEREST

As indicated on the previous three charts, the experiments considered in the study as being representative of potential inflight application have been grouped under six broad scientific and technological categories:

1. Systems Development and Test (SDT)
2. Materials and Structures (MS)
3. Multispectral Imagery of the Earth and Orbiting Objects (MI)
4. Solid/Liquid/Gas Behavior (SLG)
5. Microorganisms (M)
6. Observations of the Earth's Atmosphere, the Space Environment, and Astronomical Observations (OEA).

It should be noted that these six categories were selected primarily for convenience in defining typical experiments and are only grossly indicative of the broad range of experiment categories that are actually covered by the selected 30 experiments. As an indication of the scope of the experiments which will be contained in the experiment characteristics library of Program SEPTER, the relationship of the experiments to the NASA's major scientific/technological areas that encompass potential in-flight experiments is shown on the facing chart.

Only two of the 13 major scientific/technical areas are not covered by at least one of the 30 experiments. These two areas are Biomedicine/Behavior and Communications and Navigation/Traffic Control. The Biomedicine/Behavior area is not covered because the experiments in this category are not compatible with the study groundrule concerning minimum astronaut participation. A number of experiments in the area of Communications and Navigation/Traffic Control which are listed in "NASA Experiment Descriptions for Extended Apollo Earth-orbit Flights" 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 accordingly listed among the 65 applicable experiments. These particular experiments were not considered further because of the limited scope of the study.

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

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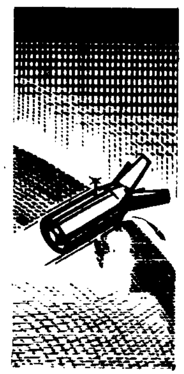
Part III



IN-FLIGHT EXPERIMENT PACKAGE DESIGN

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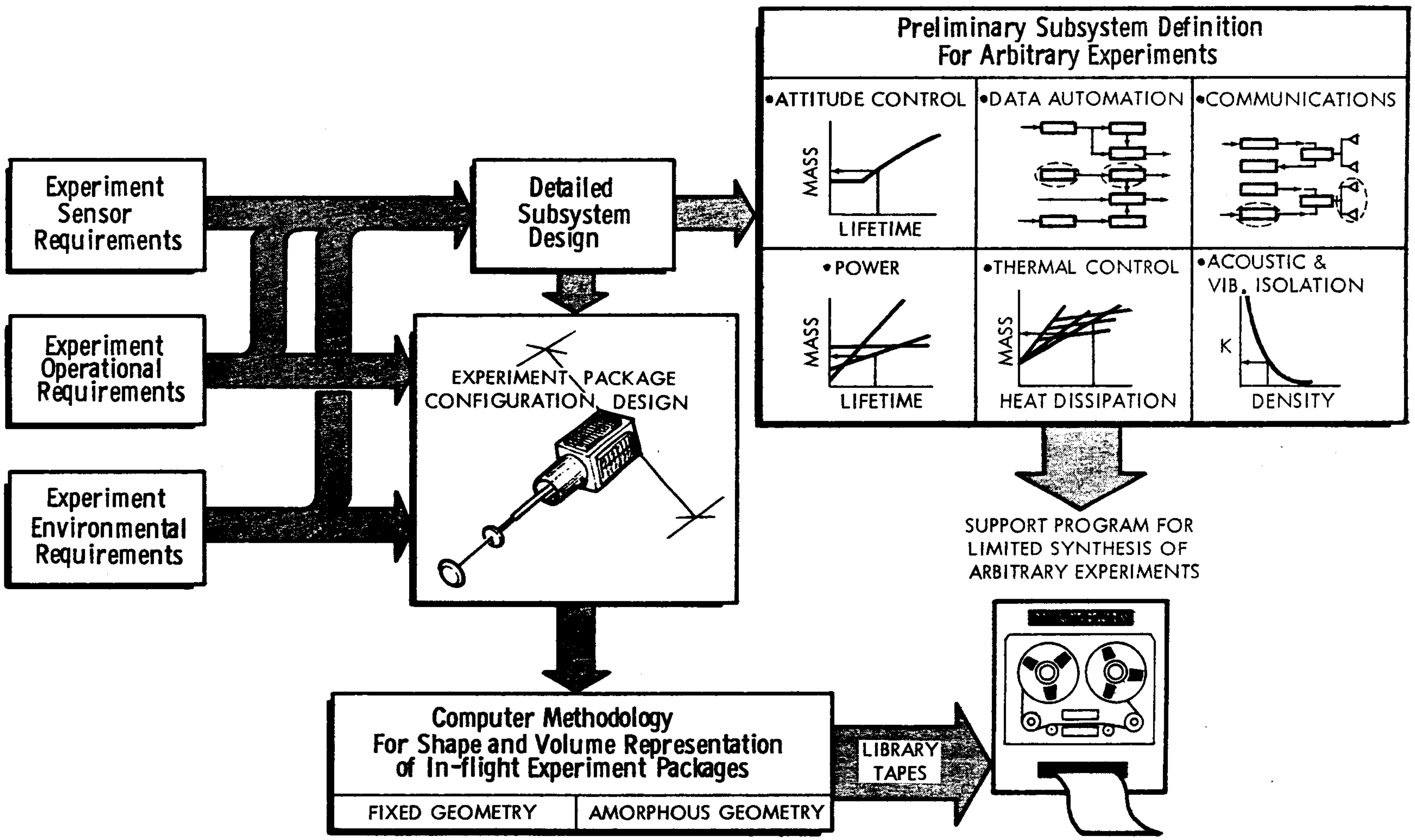
APPROACH TO IN-FLIGHT EXPERIMENT PACKAGE DESIGN

Thirty inflight experiment packages have been designed by means of a straight forward approach based on a minimum of iteration. Package design began with the experimenters who defined the sensor requirements for five experiments in each of the six selected scientific and technological categories of experiment. Ancillary subsystem characteristics were defined in the subsystem detailed design on the basis of requirements related to the sensors, the experiment operations, and the environment. Satisfaction of these requirements influenced the consideration of attitude control, data automation, communications, power, and environmental control. The physical characteristics of experiment sensors and ancillary subsystems were then used in achieving a configuration design for each experiment package. The results of the efforts on the individual inflight experiment package designs have been expressed in terms of mass, volume/geometry (including critical dimensions), and other characteristics. The experiment package characteristics will be stored on library magnetic tapes for use in the SEPTER computer program.

A future task is the development of a support program for limited synthesis of arbitrary experiments. An attempt will be made to provide a methodology for synthesizing complete packages for arbitrary experiments. The characteristics of ancillary subsystems will be defined to satisfy the given input requirements by means of analytical representations in the form of curve fits or characteristics stipulated for required components.

APPROACH TO IN-FLIGHT EXPERIMENT PACKAGE DESIGN



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SUMMARY OF SENSOR REQUIREMENTS

Each experimenter was asked to define the sensor requirements for each experiment submitted. On the following three charts, these requirements are summarized in the form of the duty cycle, the inflight duration of the experiment, the accuracy or resolution required (including the pointing accuracy, where applicable), and an indication of whether results were to be telemetered or physically recovered in a re-entry capsule.

Six categories of experiments are considered:

1. Systems Development and Test
2. Materials and Structures
3. Multispectral Imagery of the Earth and Orbiting Objects
4. Solid/Liquid/Gas Behavior
5. Microorganisms
6. Observations of the Earth's Atmosphere, the Space Environment, and Astronomical Observations.

SUMMARY OF SENSOR REQUIREMENTS - 1

Sensors	Duty Cycle	In-Flight Duration	Accuracy	Data Recovery	
				Telemetry	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$0.1 AMPERE $\pm 2^{\circ}$F TEMP	√	
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 $\pm 3^{\circ}$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 $\pm 10^{\circ}$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	———	√	

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SUMMARY OF SENSOR REQUIREMENTS - 2

Sensors	Duty Cycle	In-Flight Duration	Accuracy	Data Recovery	
				Telemetry	Capsule
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 x 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	√	

FW/65/274/6820

SUMMARY OF SENSOR REQUIREMENTS - 3

	Sensors	Duty Cycle	In-Flight Duration	Accuracy	Data Recovery	
					Telemetry	Capsule
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 Observations						
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%	√	

APPROACH TO ANCILLARY SUBSYSTEM DEFINITION

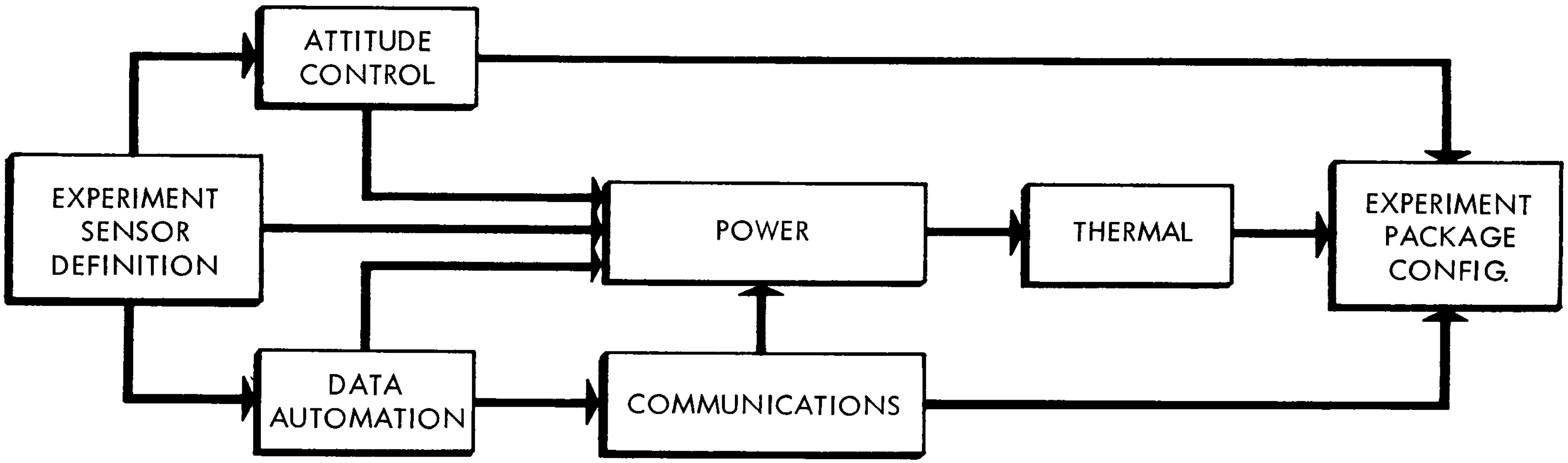
The basic approach used in the synthesis of self-contained experiment packages is illustrated on the accompanying chart. The definitions of experiment sensor requirements were used to determine the functional requirements to be specified for ancillary subsystems, such as attitude control, data automation, communications, power, and thermal control. In the case of each experiment, an effort was made to employ the most promising system concepts through the use of state-of-the-art components. Experiment package configuration efforts were undertaken to assure 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 overall experiment package physical characteristics. Other analyses have been made, and other requirements have been and are being delineated for each experiment package to produce the total required outputs as depicted on the chart.

APPROACH TO ANCILLARY SUBSYSTEM DEFINITION

Detailed Subsystem Design

Objective: Synthesize Self-contained Experiment Packages to Meet Sensor and Data Acquisition Requirements by Utilizing State-of-the-Art Components



Outputs: Experiment Package Characteristics

- MASS
- VOLUME/GEOMETRY
- THERMAL, VIBRATION, ELECTROMAGNETIC ENVIRONMENT
- COST
- RELIABILITY
- DEPLOYMENT REQUIREMENTS
- DEVELOPMENT TIME

FW/65/274-6822

ATTITUDE CONTROL

The use of an attitude determination and control system enables the orientation of satellite body axes to be known in relation to some primary coordinate frame of reference and to be aligned precisely for sensor pointing. Normally, the design of attitude control, torque-producing mechanisms is based on a thorough knowledge of satellite size and configuration, mission duration, sensors, disturbance torques, and other factors. Candidate concepts for stabilization include the use of gas jets, spinning, gravity gradient, and (possibly) reaction wheels.

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 using experiment MI-1 (Multi-spectral Surveillance of Earth) as a reference example. A stored gas system was chosen on the basis of simplicity and reliability. It is assumed that the function of the attitude control system is to maintain the vehicle in one of two desired attitudes throughout its specified lifetime (i.e., Earth or solar orientation). The vehicle is assumed to be ejected from its booster in this attitude. Initially, the attitude control system must remove ejection disturbances by driving the rates about all three body axes to zero. The body-mounted rate gyros furnish signals for this purpose and also provide damping in the control loop. Soon after ejection, the horizon sensors, in earth-pointing situations, are activated to provide attitude error signals in the pitch and roll portions of the system. Yaw attitude error signals are provided by the control moment gyro, which is an inertial gyro-compassing device. This system provides control for all orbit inclinations from equatorial to polar. For orbits in the equatorial plane, the control moment gyro could be replaced by a Polaris star sensor in order to reduce system mass. For orientation with respect to the sun, the horizon sensors could be replaced by sun sensors.

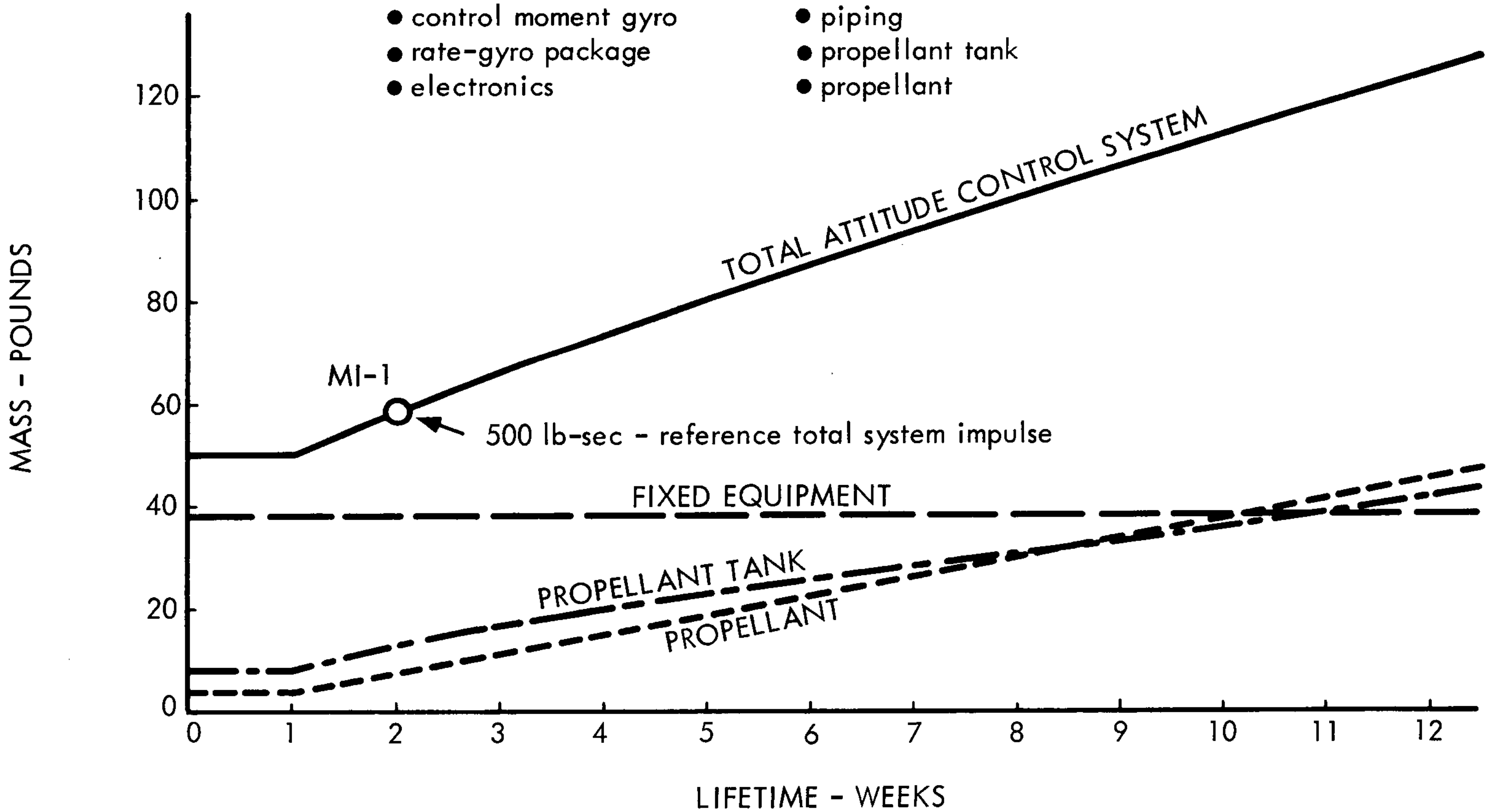
The mass of propellant and the propellant tank mass vary as a function of lifetime requirements. The mass of remaining items is constant since these items are essentially fixed equipment.

ATTITUDE CONTROL

STORED GAS SYSTEM - NOMINAL DESIGN

TYPICAL COMPONENTS

- horizon sensors
- control moment gyro
- rate-gyro package
- electronics
- thrusters
- piping
- propellant tank
- propellant



- SYNTHESIS APPROACH - CURVE FITS

FW/65/274-6823

DATA AUTOMATION

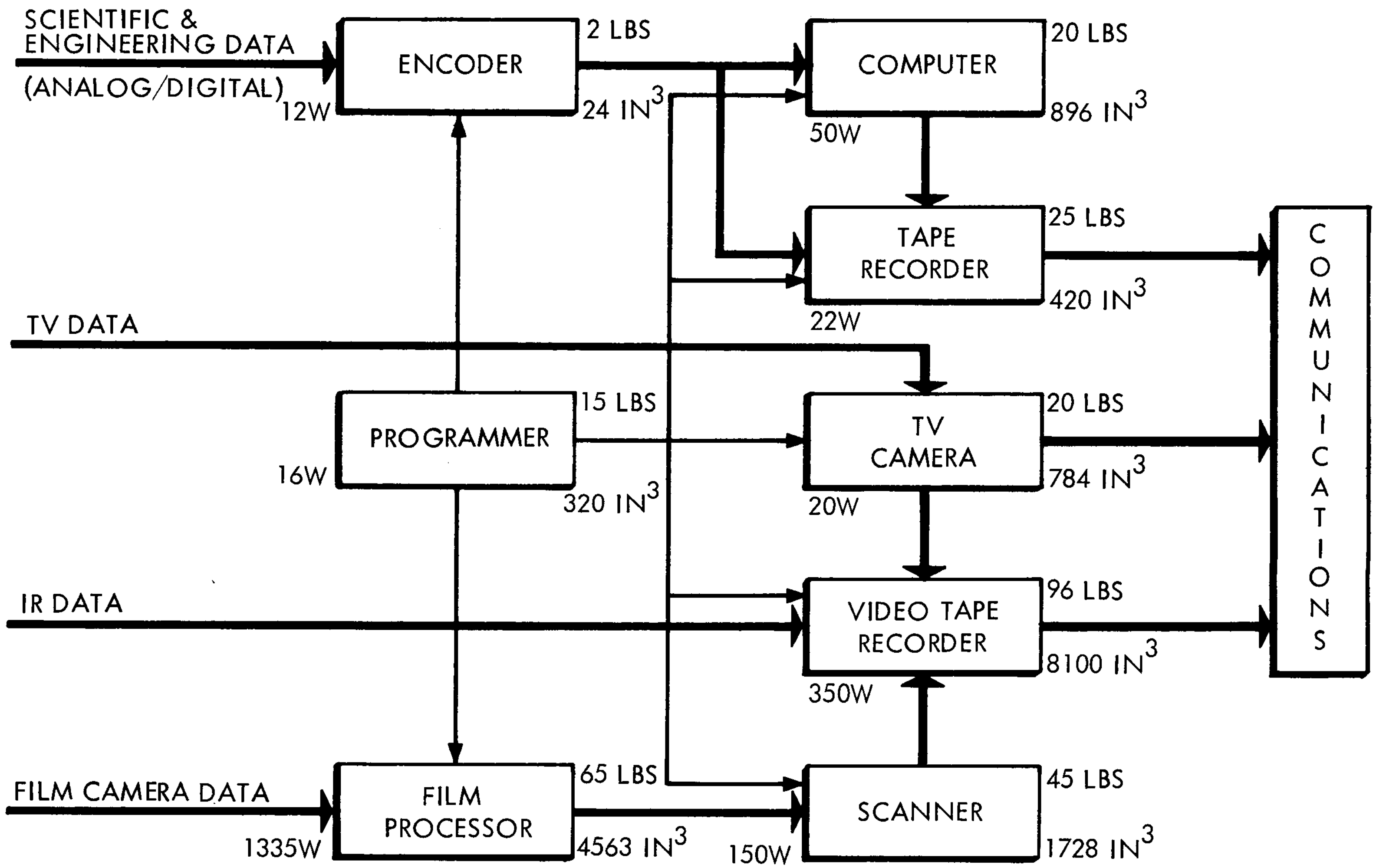
The data automation subsystem is used to provide the optimum interface between the sensors and the communications subsystem. Four types of data are processed by this subsystem: scientific and engineering, television, infrared, and photographic. The duty cycles of all data automation equipment are controlled 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 subsystem 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 though the communications subsystem whenever a station is in sight. Infrared data from the IR scanners is routed directly to the video tape recorder for delayed readout.

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

DATA AUTOMATION



- SUMMATION OF MASS, VOLUME, AND POWER FOR INDIVIDUAL EXPERIMENTS
- SYNTHESIS APPROACH - INDIVIDUAL COMPONENT SELECTION

FW/65/274-6824

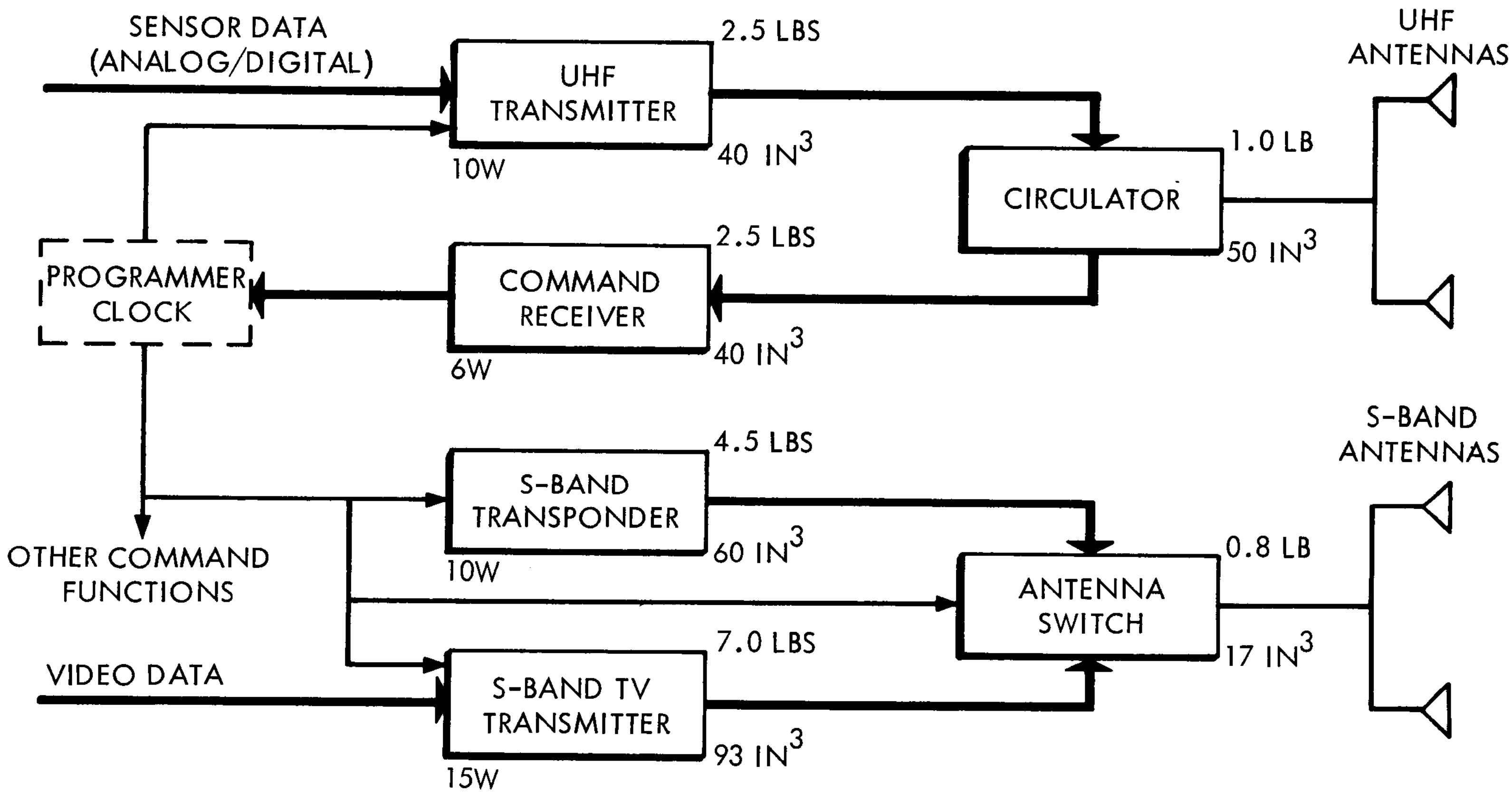
COMMUNICATIONS

The communications system used in conjunction with each experiment is limited in design to meet only those requirements particular 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 equipment and within the present-day state-of-the-art.

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 is to be maintained at all times with the Earth station. 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.

The input power in watts, mass in pounds, and volume in cubic inches are shown in the facing block diagram for each unit of the communications system. The programmer clock unit is considered to be a part of the data handling system; consequently, no values are shown. In addition, the final antenna design is considered to be a function of satisfying satellite physical and attitude control requirements; therefore, antenna size and type are selected only after mission requirements are completed.

COMMUNICATIONS



- SUMMATION OF MASS, VOLUME, AND POWER FOR INDIVIDUAL EXPERIMENTS
- SYNTHESIS APPROACH - INDIVIDUAL COMPONENT SELECTION

FW/65/274/6825

POWER

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

The left side of the facing figure is the well-known plot of power output versus duration of operation. The data is presented in terms of an optimum power source as a function of the mass of the source. In the major portion of the secondary experiments analyzed in this study, the power and the energy requirements either fall within the battery area or near its boundary.

Detailed analysis is needed in the case of experiments which fall well out of the battery area. A notable example is the MI-3 experiment, Radar Surveillance of Earth. As shown in the lower left tabulation, a relatively high power requirement must be satisfied for one hour; 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 are analyzed in terms of mass as a function of days operation at the lower right:

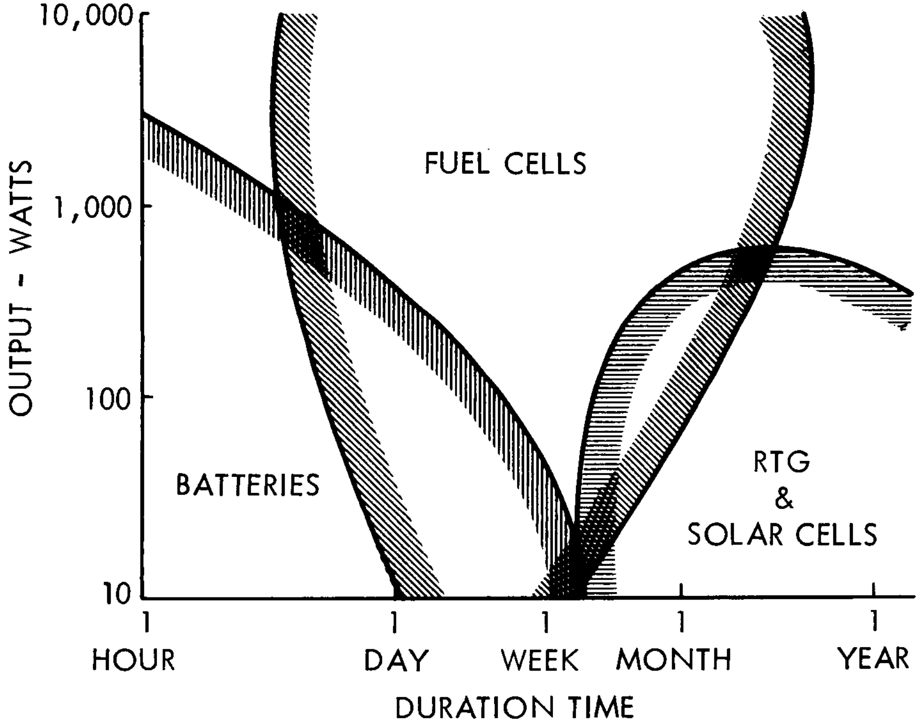
Item	Watts	Hours/Day
Sensor	1700	1
Data Automation	316	2
Data Transmission	15	1
Beacon & Command Receiver	16	24
Attitude Control	50	24

Power System	Mass in Pounds (d = Days Operated)
Silver Zinc Pri. Bat.	60 + 63 d
Solar Array & Bat.	504
RTG direct	1045
RTG & Battery	610
Fuel Cell Direct	300 + 6 d
Fuel Cell & Battery	500 + 9 d

The right side of the facing figure represents a generalization of the analysis reported above. For periods up to 5 days, the battery is lighter; from 5 to 25 days, the fuel cell is more advantageous. Beyond 25 days, the solar array is lighter.

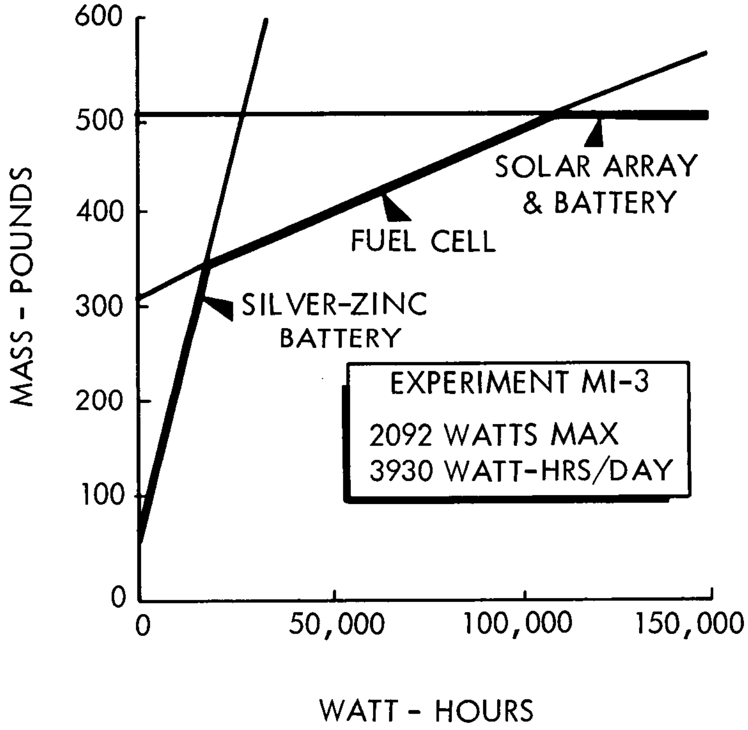
POWER

Optimum Power Sources
(BASED ON MASS)



• SYNTHESIS APPROACH - CURVE FITS

Typical Power System Selection



FW/65/274 6826

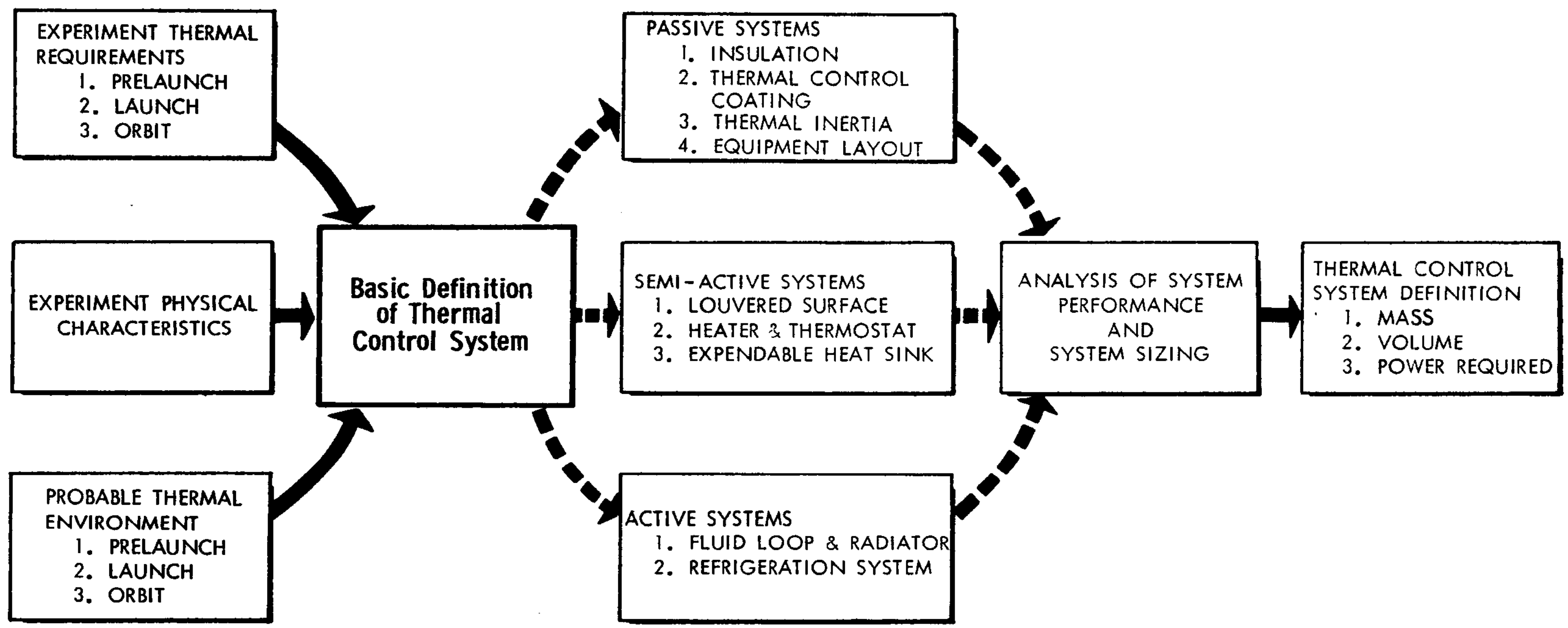
APPROACH TO THERMAL CONTROL SYSTEM DESIGN

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 to size the system in terms of the 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.

Selection of a thermal control concept for a particular experiment was based on consideration of the experiment thermal requirements (allowable temperature range, heat dissipation), the physical characteristics of the experiment, the probable thermal environment, and the relationship between the experiment's 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. Examples of specific concepts within each category are included on the facing chart. In certain situations, combinations of concepts, e.g., insulation used in conjunction with a heater and thermostat, may be used to advantage.

With the thermal control concept selected, analysis of thermal control system performance was undertaken with respect 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 insure adequate performance throughout the mission.

APPROACH TO THERMAL CONTROL SYSTEM DESIGN



- Synthesis Approach - Curve Fit

FW/65/274-6827

RESULTS OF ANCILLARY SUBSYSTEM DEFINITION

The ancillary subsystems required for the thirty representative experiments are indicated on the facing chart in terms of applicable components or systems. It was determined that ten experiments were best suited to be undertaken on board the SIVB stage. The others will be undertaken in separate satellites after ejection from the primary vehicle. Of the twenty ejected satellites, an attitude control system is needed in fourteen, and six may be allowed to tumble without attitude stabilization.

Three of the experiments are better suited to a fuel cell electric power system than to a battery system. The selection of components for data automation and communications appears to be self-explanatory. In four experiments, television cameras are needed, and these experiments are defined so that TV data may be telemetered directly to an Earth-based station without the need for data storage on board the vehicle.

RESULTS OF ANCILLARY SUBSYSTEM DEFINITION

	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
Attitude Control - Synthesis Approach/Curve Fits																														
SYSTEM REQ'D	◆		◆	◆		◆	*	*	*	*	◆	◆	◆	◆	◆	*	*	◆	*	*		*				*	◆	◆	◆	◆
Data Automation - Synthesis Approach/Individual Component Selection																														
PROGRAMMER	◆					◆	◆	◆			◆	◆	◆	◆	◆		◆	◆	◆				◆		◆					◆
ENCODER	◆	◆	◆	◆		◆	◆	◆	◆	◆					◆	◆	◆	◆		◆		◆			◆	◆		◆		◆
TAPE RECORDER	◆		◆	◆		◆	◆	◆	◆		◆				◆	◆	◆	◆		◆					◆	◆	◆	◆		◆
COMPUTER			◆																							◆				
TV CAMERA																					◆	◆	◆	◆						
VIDEO TAPE REC'D'R												◆	◆	◆																
Communications - Synthesis Approach/Individual Component Selection																														
UHF TRANSMITTER	◆	◆	◆	◆			◆	◆	◆	◆	◆					◆	◆			◆					◆	◆	◆	◆		◆
COMMAND REC'V'R	◆	◆	◆	◆			◆	◆	◆	◆	◆	◆	◆	◆		◆	◆			◆	◆	◆	◆	◆	◆	◆	◆	◆		◆
UHF CIRCULATOR	◆	◆	◆	◆			◆	◆	◆	◆	◆					◆	◆			◆					◆	◆	◆	◆		◆
UHF ANTENNA	◆	◆	◆	◆			◆	◆	◆	◆	◆	◆	◆	◆		◆	◆			◆	◆	◆	◆	◆	◆	◆	◆	◆		◆
TRANSPONDER	◆		◆	◆		◆	◆	◆	◆	◆	◆	◆	◆	◆	◆		◆	◆		◆	◆	◆	◆	◆	◆				◆	◆
TV TRANSMITTER												◆	◆	◆							◆	◆	◆	◆						
ANTENNA SWITCH												◆	◆	◆							◆	◆	◆	◆						
S-BAND ANTENNA	◆		◆	◆		◆	◆	◆	◆	◆	◆	◆	◆	◆	◆		◆	◆		◆	◆	◆	◆	◆	◆				◆	◆
Power - Synthesis Approach/Curve Fits																														
BATTERIES		◆		◆			◆	◆	◆	◆	◆	◆		◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆
FUEL CELLS			◆			◆							◆																	

* REMAINS ON SIVB

FW/65/274-6828

EXPERIMENT OPERATION REQUIREMENTS

During a mission, certain operations must be performed in order to assure success of any experiment. Of particular interest in this study are those requirements which are contingent on proper installation of the experiment relative to the launch vehicle so that proper operation will be achieved. In this study the operational requirements for each of the experiments is defined as follows:

1. <u>View Requirements</u> for experiments based on the use of cameras or other sensors which must view outside the launch vehicle envelope:

    a. Field of view or aperture required
    b. Viewing direction required
    c. Timing requirements

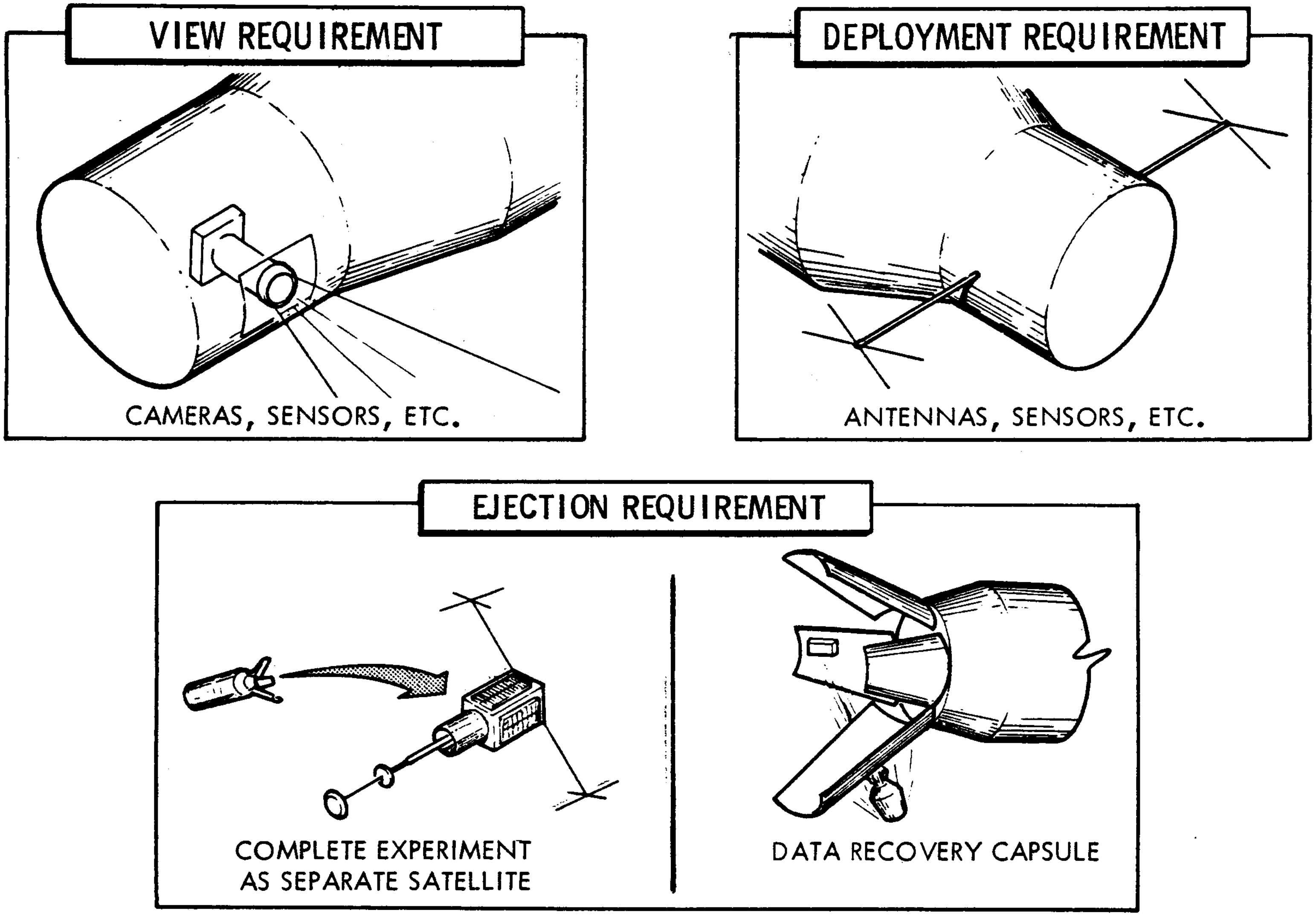
2. <u>Deployment Requirements</u> for experiments in which antennas, sensor components, etc., must be erected outside of the launch vehicle:

    a. Size of deployed object
    b. Final deployed direction required
    c. Timing requirements

3. <u>Ejection Requirements</u> for experiments in which the entire experiment or experiment component parts, such as a data recovery capsule, must be separated from the launch vehicle:

    a. Direction required
    b. Aperture required
    c. Separation velocity required
    d. Timing required

EXPERIMENT OPERATION REQUIREMENTS



FW/65/274-6829

EXPERIMENT ENVIRONMENTAL REQUIREMENTS

Data on experiment environmental requirements will be used in performing environmental compatibility checks and in determining environmental control system mass penalties.

The experiment environmental requirements will be defined as follows:

1. Thermal

    1. A maximum and minimum temperature limit based on a critical component used in the experiment

    2. The maximum heat dissipation rate of the experiment

    3. The total heat dissipation from the experiment

2. Vibrations - Assumed qualification of all components per Spec MIL-E-5272C procedure XII

3. Accoustics - Assumed qualification of all components noise tolerance of 150 db per MIL Standard 810 Acoustical Test Method Grade B which is highest level required for jet aircraft

4. Electromagnetic - Not defined to date. To be accomplished during the second half of the study.

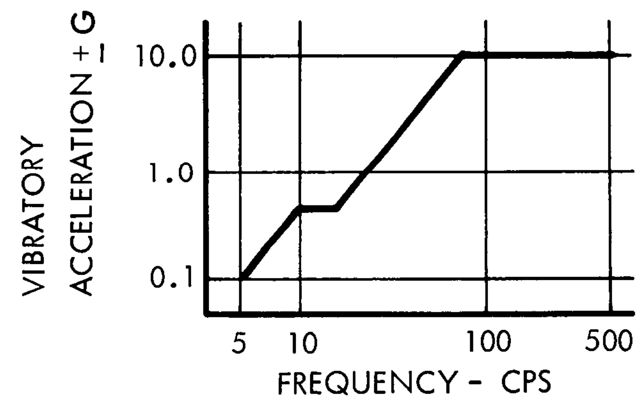
EXPERIMENT ENVIRONMENTAL REQUIREMENTS

THERMAL

- Maximum and Minimum Temperature
- Heat Disipation 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 Accoustical Test Method Grade B

ELECTROMAGNETIC

- To Be Defined Later in Program

FW/65/274-6830

EXPERIMENT SUMMARY I, II, & III

Experiment conceptual design drawings and weight and volume analysis were made for each of the 30 experiments selected for use in this study. This design effort was made necessary for three reasons; first, to show the typical characteristics of the six scientific and technological experiment categories selected; second, to assure design feasibility; and third, to interpret the experiment/launch vehicle interface. The experiment configurations are summarized in their respective categories on the following six charts. The total experiment mass, basic component volume, and the packaged or installed volume are also listed. The mass and volume includes any installation or ejection hardware required for packaging the experiment aboard the launch vehicle. The basic component volume was listed in order to indicate the packaging factor requirements for each experiment category. The packaging factor can be found by dividing the packaged volume by the basic component volume ($F_p = V_p/V_{BC}$). As will be noted in the charts, some of the experiments were designed for operation aboard the Saturn IB/Apollo vehicle while others were designed to be ejected as separate satellites. This difference in design was dictated mainly by orbital lifetime requirements.

I Systems Development and Testing

All of the experiments in Category I were designed to be ejected as separate satellites. One experiment, the Energetic Particles Explorer, is a fixed hardware configuration. The experiment masses range from 47 pounds (Experiment SDT-2, Performance Assessment of Solar Cell Arrays) to 9111 pounds (Experiment SDT-1, The Radioisotope-Thermoelectric experiments). For the same experiments, the installation volumes required are 3700 and 212,000 cubic inches, respectively.

II Materials and Structures

Experiment MS-1, The Degradation of Organic Materials in Space Environment, was the only experiment in Category II designed to be ejected as a separate satellite. MS-1 also has the largest mass and volume (822. pounds and 73,500 cubic inches, respectively). The remainder of the experiments are designed to be contained aboard the launch vehicle. These masses range from 100 to 150 pounds and volumes from 3,000 to 5,000 cubic inches.

III Multiplespectral Imagery of the Earth and Orbiting Objects

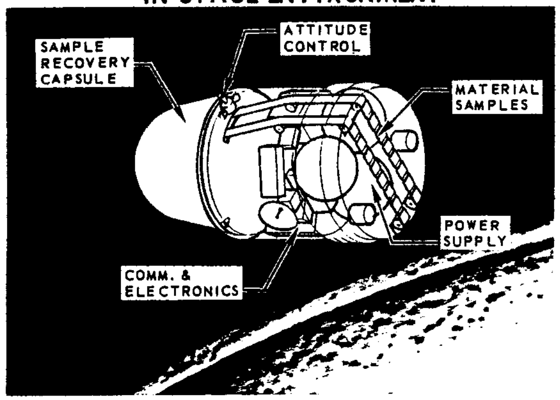
Because of the long life time requirements, all of the experiments in Category III were designed as separate satellites. Four have masses between 700 and 1200 pounds. The synoptic Earth cartography experiment has the largest mass of 2850 pounds. The experiment volume requirements vary from about 40,000 to 470,000 cubic inches.

EXPERIMENT SUMMARY

II

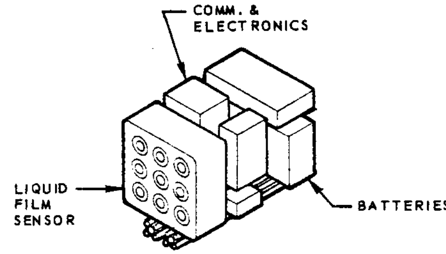
MATERIALS & STRUCTURES

MS-1
DEGRADATION OF ORGANIC MATERIALS IN SPACE ENVIRONMENT



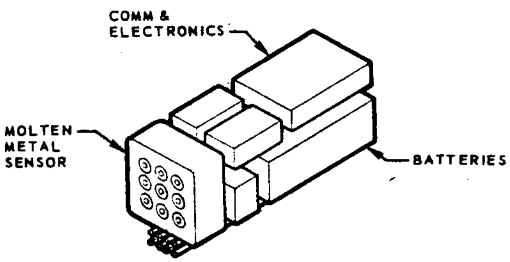
MASS	822 LBS.
VOLUME (BASIC COMPONENT)	46,200 IN.^{3}
VOLUME (PACKAGED)	73,500 IN.^{3}

MS-2
BEHAVIOR OF LIQUID FILMS IN SPACE ENVIRONMENT



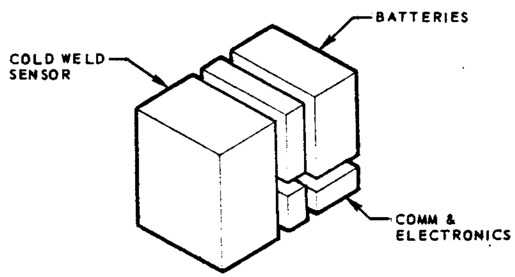
MASS	97 LBS.
VOLUME (BASIC COMPONENT)	1,827 IN.^{3}
VOLUME (PACKAGED)	2,988 IN.^{3}

MS-3
VAPORIZATION RATE OF MOLTEN METALS



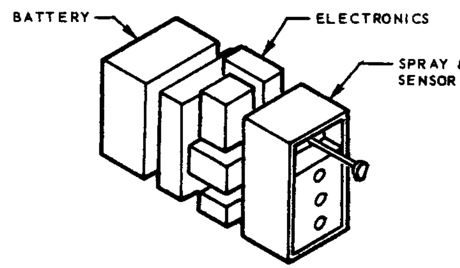
MASS	147 LBS.
VOLUME (BASIC COMPONENT)	2,809 IN.^{3}
VOLUME (PACKAGED)	3,750 IN.^{3}

MS-4
COLD WELDING OF METALS IN SPACE ENVIRONMENT



MASS	120 LBS.
VOLUME (BASIC COMPONENT)	4,344 IN.^{3}
VOLUME (PACKAGED)	5,500 IN.^{3}

MS-5
SPRAY COATING AND SURFACE CONTAMINATION



MASS	110 LBS.
VOLUME (BASIC COMPONENT)	3,624 IN.^{3}
VOLUME (PACKAGED)	4,650 IN.^{3}

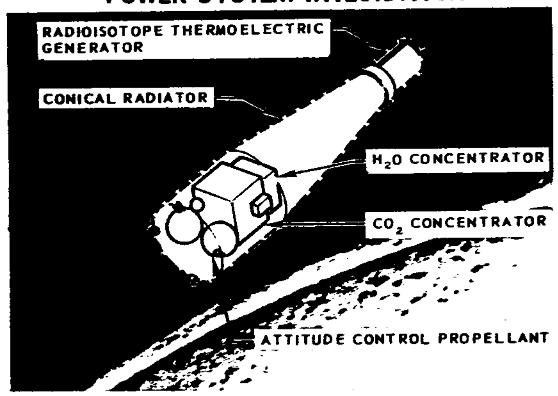
FW/65/274-6832

EXPERIMENT SUMMARY

I

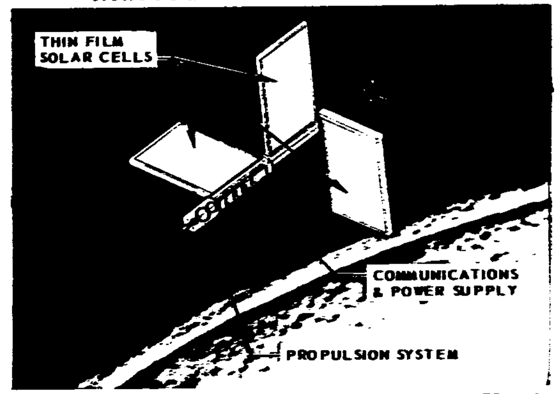
SYSTEMS DEVELOPMENT AND TESTING

SDT-1
RADIOISOTOPE-THERMOELECTRIC POWER SYSTEM INTEGRATION



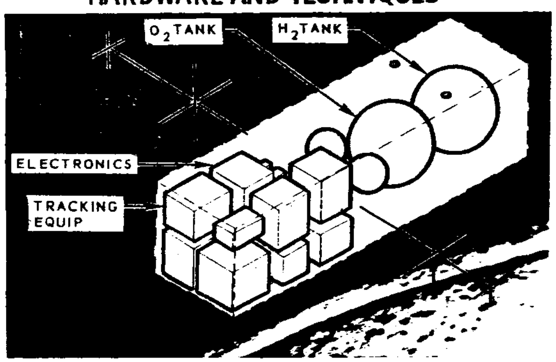
MASS	936 LBS.
VOLUME (BASIC COMPONENT)	50,900 IN.^{3}
VOLUME (PACKAGED)	122,000 IN.^{3}

SDT-2
PERFORMANCE ASSESSMENT OF THIN FILM SOLAR CELL ARRAYS



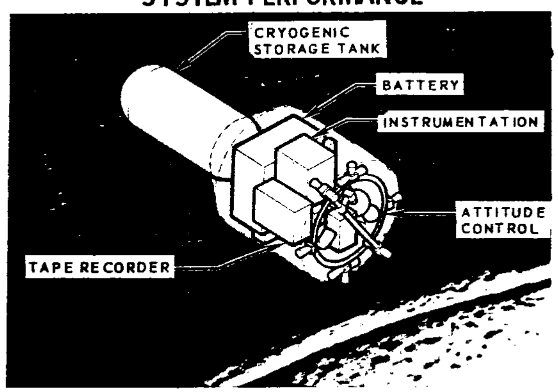
MASS	50 LBS.
VOLUME (BASIC COMPONENT)	2,504 IN.^{3}
VOLUME (PACKAGED)	3,689 IN.^{3}

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



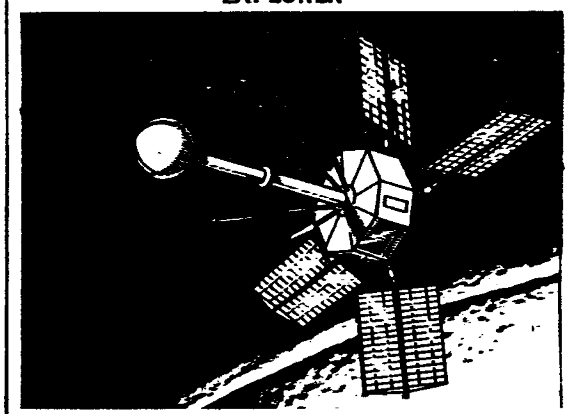
MASS	680 LBS.
VOLUME (BASIC COMPONENT)	25,803 IN.^{3}
VOLUME (PACKAGED)	72,900 IN.^{3}

SDT-4
CRYOGENIC PROPELLANT STORAGE SYSTEM PERFORMANCE



MASS	466 LBS.
VOLUME (BASIC COMPONENT)	22,465 IN.^{3}
VOLUME (PACKAGED)	47,595 IN.^{3}

SDT-5
ENERGETIC PARTICLES EXPLORER



MASS	400 LBS.
VOLUME (PACKAGED)	30,300 IN.^{3}

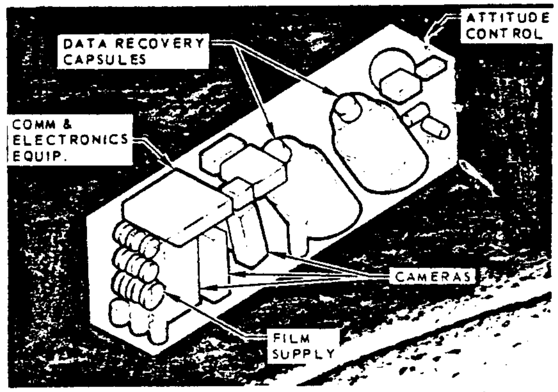
FW/65/274/6831

EXPERIMENT SUMMARY

III

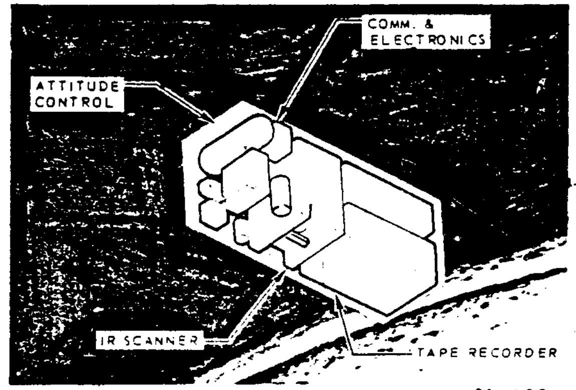
MULTISPECTRAL IMAGERY
OF THE EARTH
AND ORBITING OBJECTS

MI-1
MULTISPECTRAL SURVEILLANCE OF EARTH



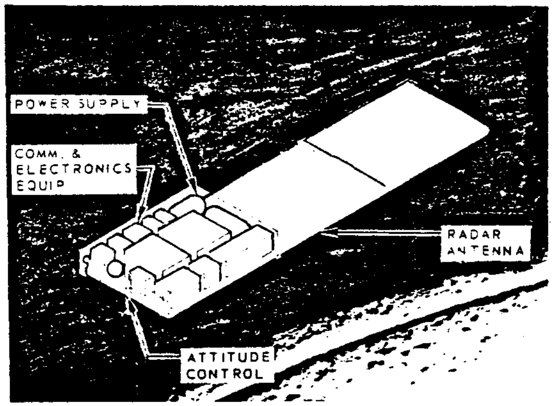
MASS	1,000 LBS.
VOLUME (BASIC COMPONENT)	31,570 IN.^{3}
VOLUME (PACKAGED)	81,200 IN.^{3}

MI-2
INFRARED LINE SCAN SURVEILLANCE OF EARTH



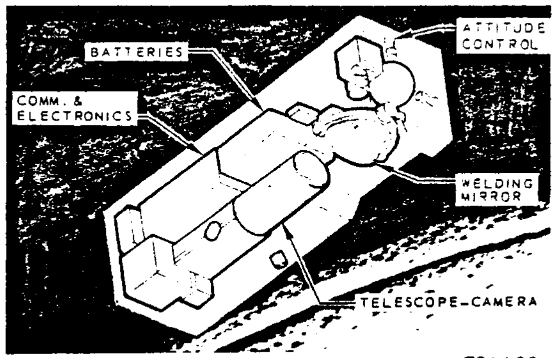
MASS	914 LBS.
VOLUME (BASIC COMPONENT)	21,000 IN.^{3}
VOLUME (PACKAGED)	39,468 IN.^{3}

MI-3
RADAR SURVEILLANCE OF EARTH



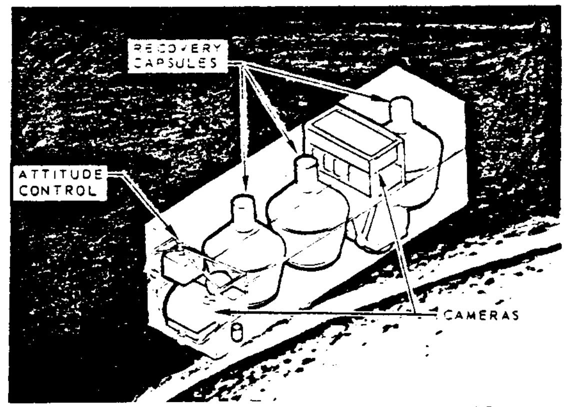
MASS	1,307 LBS.
VOLUME (BASIC COMPONENT)	98,000 IN.^{3}
VOLUME (PACKAGED)	139,000 IN.^{3}

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



MASS	796 LBS.
VOLUME (BASIC COMPONENT)	27,110 IN.^{3}
VOLUME (PACKAGED)	79,083 IN.^{3}

MI-5
SYNOPTIC EARTH CARTOGRAPHY



MASS	2,850 LBS.
VOLUME (BASIC COMPONENT)	200,903 IN.^{3}
VOLUME (PACKAGED)	469,800 IN.^{3}

FW/65/274-6833

EXPERIMENT SUMMARY IV, V, & VI

IV Solid/Liquid/Gas Behavior

Only one experiment in Category IV, The Formation of Single Crystals, number SLG-3, was designed to be ejected as a separate satellite. The remaining four are contained aboard the Saturn IB/Apollo for the entire mission. Physical data recovery by a recoverable capsule is used in the ejected experiment and in two of the contained experiments. Masses of the contained experiments in this category vary from 143 pounds to 190 pounds. The ejected experiment SLG-3 has a mass of 598 pounds. Volume requirements range from about 7,000 to 14,000 cubic inches for the contained experiments; 103,000 cubic inches are needed for the ejected experiment.

V Microorganisms

Four of the experimental payloads in Category V are designed to be ejected from the Saturn IB/Apollo as separated orbiting satellites. Experiment M-2, Effects of Space Flight on Morphology, Growth, and Liquid/Gas Separation in Microorganisms, is the only one designed to be contained in the launch vehicle. The total mass requirements for this category are all between 129 pounds and 237 pounds. Volume requirements vary from 6,900 to 17,400 cubic inches.

VI Observations of the Earth's Atmosphere, The Space Environment, and Astronomical Observations

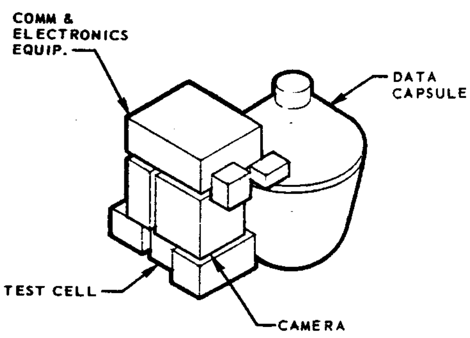
As a result of duration requirements for Category VI, only one experiment OEA-1, Radiation Environment Monitoring, is designed to remain aboard the Saturn IB/Apollo. The other four experiments are designed to be ejected as separate attitude-stabilized satellites for longer duration missions. Mass and volume requirements for this category of experiments will vary from approximately 300 to 600 pounds and 11,000 to 32,000 cubic inches, respectively.

EXPERIMENT SUMMARY

IV

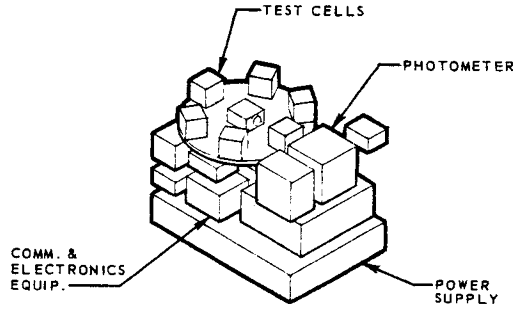
SOLID/LIQUID/GAS BEHAVIOR

SLG-1
BOILING IN ZERO-G ENVIRONMENT



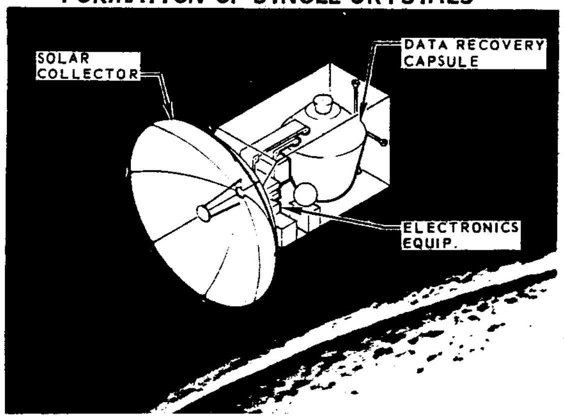
MASS	182 LBS.
VOLUME (BASIC COMPONENTS)	7,338 IN.^{3}
VOLUME (PACKAGED)	13,500 IN.^{3}

SLG-2
NUCLEATE CONDENSATION IN ZERO-GRAVITY



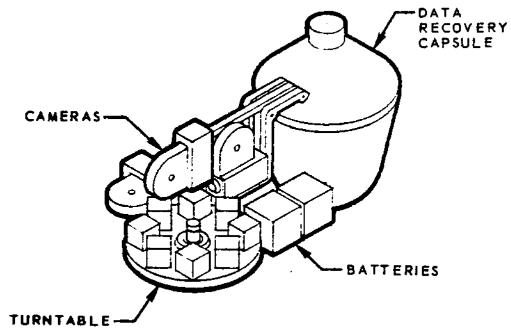
MASS	186 LBS.
VOLUME (BASIC COMPONENTS)	3,450 IN.^{3}
VOLUME (PACKAGED)	7,862 IN.^{3}

SLG-3
FORMATION OF SINGLE CRYSTALS



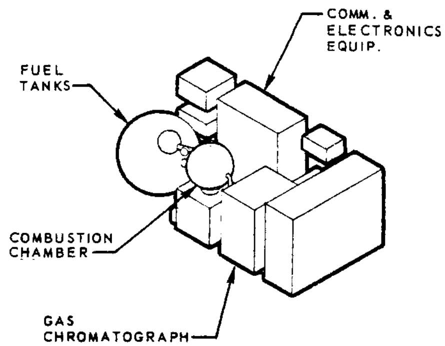
MASS	589 LBS.
VOLUME (BASIC COMPONENTS)	43,950 IN.^{3}
VOLUME (PACKAGED)	102,993 IN.^{3}

SLG-4
SEGREGATION OF IMMISCIBLE LIQUIDS UNDER ZERO GRAVITY CONDITIONS



MASS	190 LBS.
VOLUME (BASIC COMPONENTS)	4,720 IN.^{3}
VOLUME (PACKAGED)	10,090 IN.^{3}

SLG-5
ZERO-G COMBUSTION



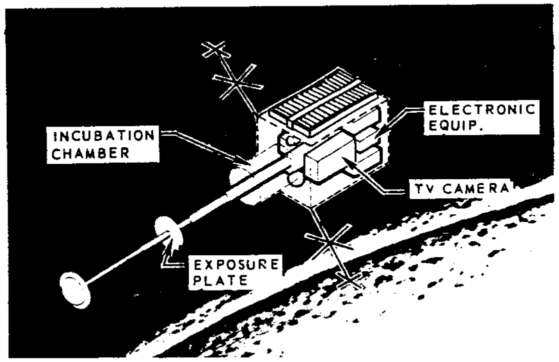
MASS	146 LBS.
VOLUME (BASIC COMPONENTS)	5,130 IN.^{3}
VOLUME (PACKAGED)	7,100 IN.^{3}

EXPERIMENT SUMMARY

V

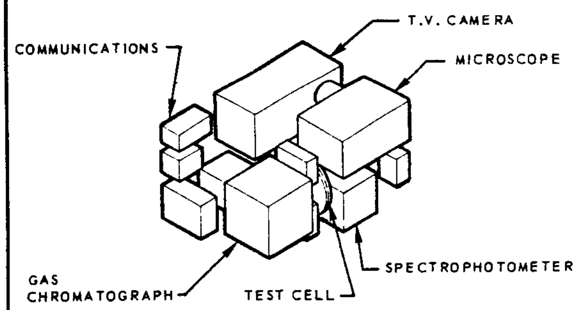
MICROORGANISMS

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



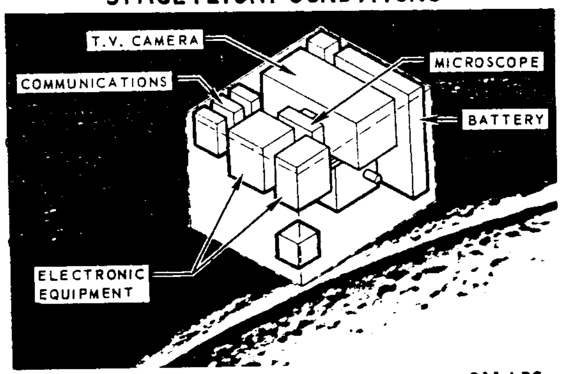
MASS	224 LBS.
VOLUME (BASIC COMPONENT)	7,320 IN.^{3}
VOLUME (PACKAGED)	17,400 IN.^{3}

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



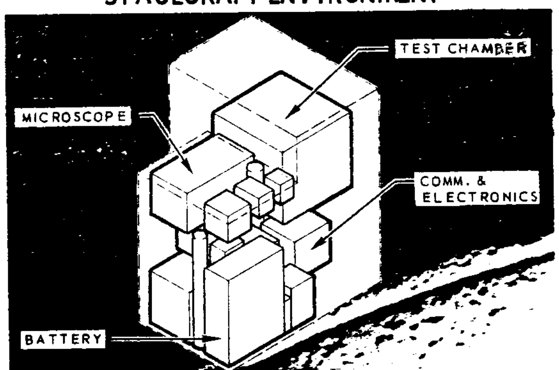
MASS	160 LBS.
VOLUME (BASIC COMPONENT)	3,218 IN.^{3}
VOLUME (PACKAGED)	6,916 IN.^{3}

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



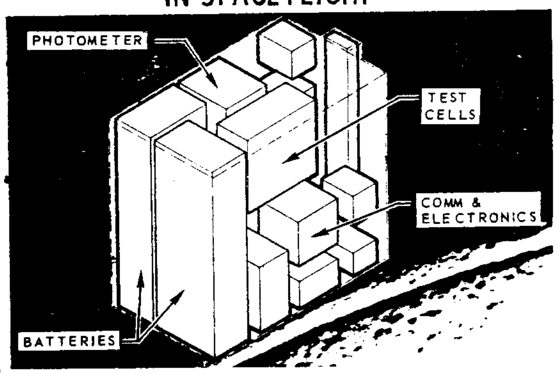
MASS	211 LBS.
VOLUME (BASIC COMPONENT)	3,500 IN.^{3}
VOLUME (PACKAGED)	8,700 IN.^{3}

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



MASS	137 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	241 LBS.
VOLUME (BASIC COMPONENT)	4,299 IN.^{3}
VOLUME (PACKAGED)	8,757 IN.^{3}

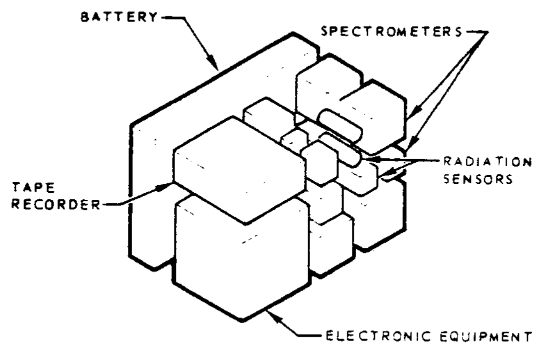
FW/65/274-6835

EXPERIMENT SUMMARY

VI

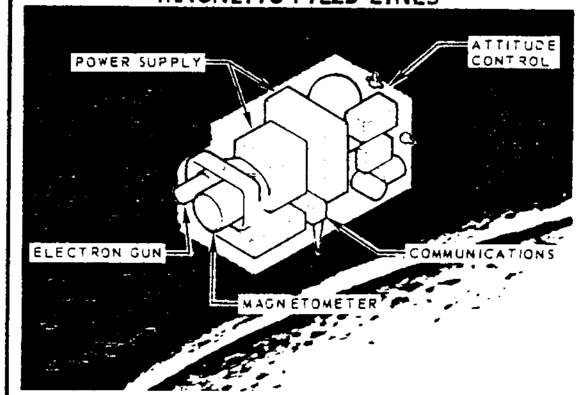
OBSERVATION OF THE EARTH'S ATMOSPHERE, THE SPACE ENVIRONMENT & ASTRONOMICAL OBSERVATION

OEA-1
RADIATION ENVIRONMENT MONITORING



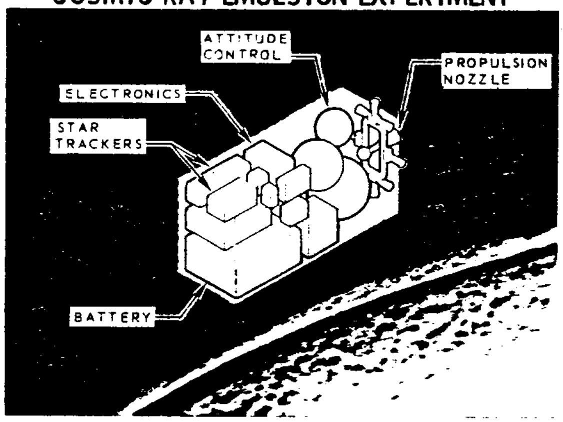
MASS	303 LBS.
VOLUME (BASIC COMPONENT)	7,333 IN.^{3}
VOLUME (PACKAGED)	11,600 IN.^{3}

OEA-2
STUDY OF MAGNETIC FIELD LINES



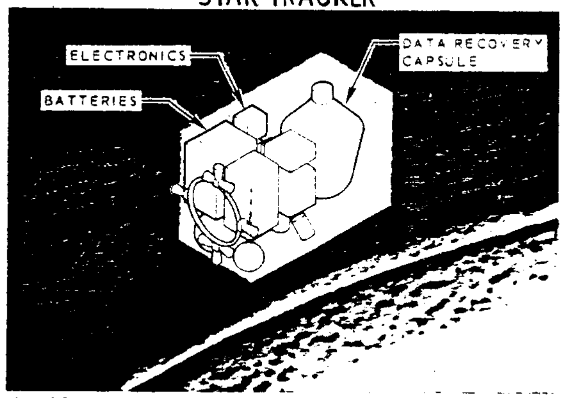
MASS	324 LBS.
VOLUME (BASIC COMPONENT)	7,700 IN.^{3}
VOLUME (PACKAGED)	19,600 IN.^{3}

OEA-3
COSMIC RAY EMULSION EXPERIMENT



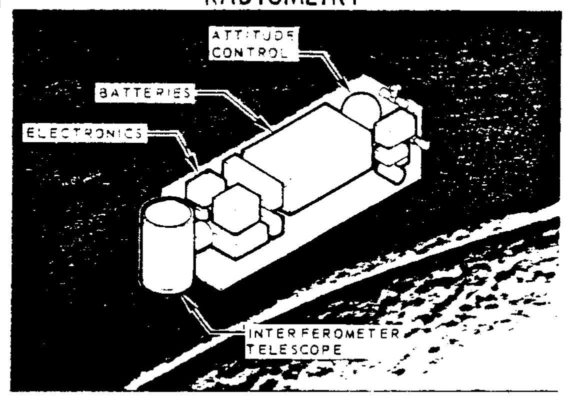
MASS	412 LBS.
VOLUME (BASIC COMPONENT)	8,818 IN.^{3}
VOLUME (PACKAGED)	22,140 IN.^{3}

OEA-4
TEST OF PROTOTYPE STAR TRACKER



MASS	340 LBS.
VOLUME (BASIC COMPONENT)	6,838 IN.^{3}
VOLUME (PACKAGED)	17,512 IN.^{3}

OEA-5
EMISSION LINE RADIOMETRY



MASS	609.8 LBS.
VOLUME (BASIC COMPONENT)	12,128 IN.^{3}
VOLUME (PACKAGED)	31,870 IN.^{3}

FW/65/274-6836

APPROACH TO DEVELOPMENT OF COMPUTER METHODOLOGY FOR EXPERIMENT SHAPE AND VOLUME REPRESENTATION

Two methods have been developed for representing experiment shape and volume, one method for the fixed geometry experiments and a second for amorphous geometry experiments. The fixed geometry experiments are defined in terms of finalized designs whose geometry cannot be modified. The amorphous experiments are those in which the configuration is not fixed and are amendable to numerous design concepts.

Fixed Geometry Experiments

The shape and volume of the fixed geometry experiments will be represented by a standard shape envelope which will most efficiently contain the entire experiment. The standard shapes selected for this representation are the sphere, the cylinder, and the rectangular parallelepiped.

Amorphous Geometry Experiments

The volume of the amorphous geometry experiment will be represented by a minimum volume for practical installation. This volume can be found in one of two ways: first, by making conceptual design drawings, as in the case of the 30 experiments previously summarized and, second, by multiplying the basic component volumes by a packaging factor. Packaging factors were determined for the six categories of experiments previously summarized and will be used in the synthesis of arbitrary experiments.

The shape of the amorphous geometry experiment will be represented by the standard shape envelope which will contain the largest undistortable component used in the experiment. This standard shape then actually represents the minimum dimensions to which the experiment total volume can be conformed. The standard shapes used for this representation are the same as used in the fixed experiments.

APPROACH TO DEVELOPMENT OF COMPUTER METHODOLOGY FOR EXPERIMENT SHAPE AND VOLUME REPRESENTATION

Fixed Geometry Experiments

Finalized Designs

DEFINED BY:

- STANDARD SHAPE ENVELOPE WHICH WILL CONTAIN THE ENTIRE EXPERIMENT

Amorphous Geometry Experiments

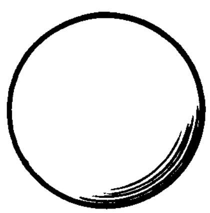
Amendable To Numerous Design Concepts

DEFINED BY:

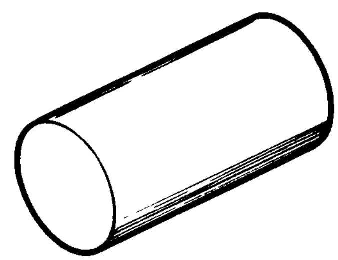
- MINIMUM PRACTICAL INSTALLATION VOLUME &
- STANDARD SHAPE ENVELOPE WHICH WILL CONTAIN THE LARGEST UNDISTORTABLE COMPONENT

Standard Shapes

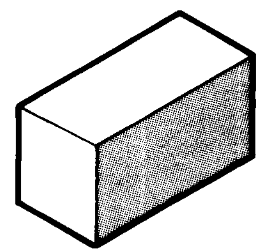
SPHERE



CYLINDER



RECTANGULAR PARALLELEPIPED



FW/65/274/6838

EXPERIMENT SHAPE AND VOLUME REQUIREMENTS

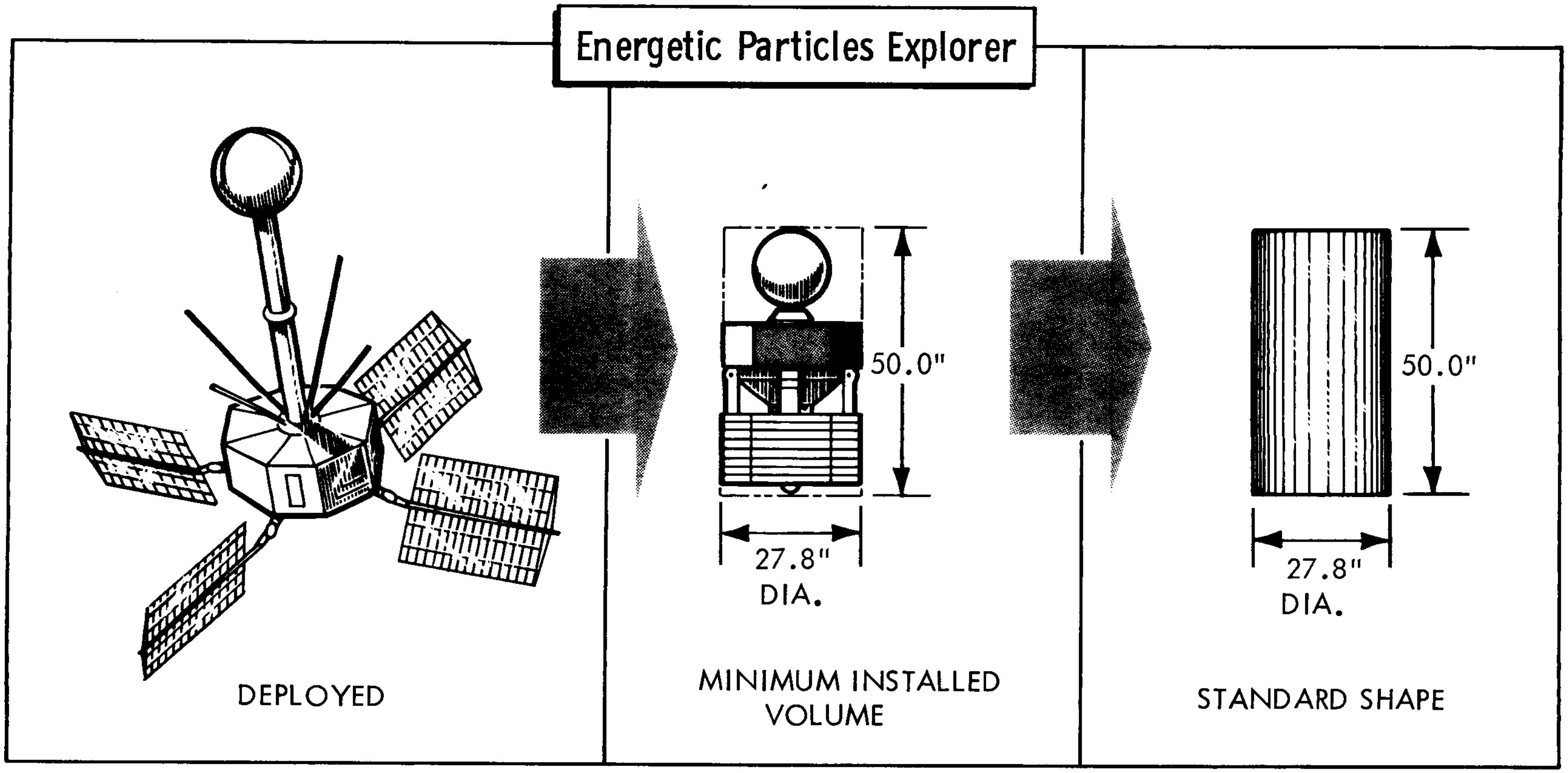
FIXED GEOMETRY EXPERIMENTS

On the facing chart is shown the energetic particle explorer satellite in both the deployed and installed configuration. This is considered a designed hardware item and therefore represents fixed geometry.

The shape and volume required for the installed configuration can most efficiently be represented by the standard shape cylinder which is 27.8 inches in diameter and 50.0 inches long.

EXPERIMENT SHAPE AND VOLUME REQUIREMENTS

Fixed Experiments



FW/65/274-6837

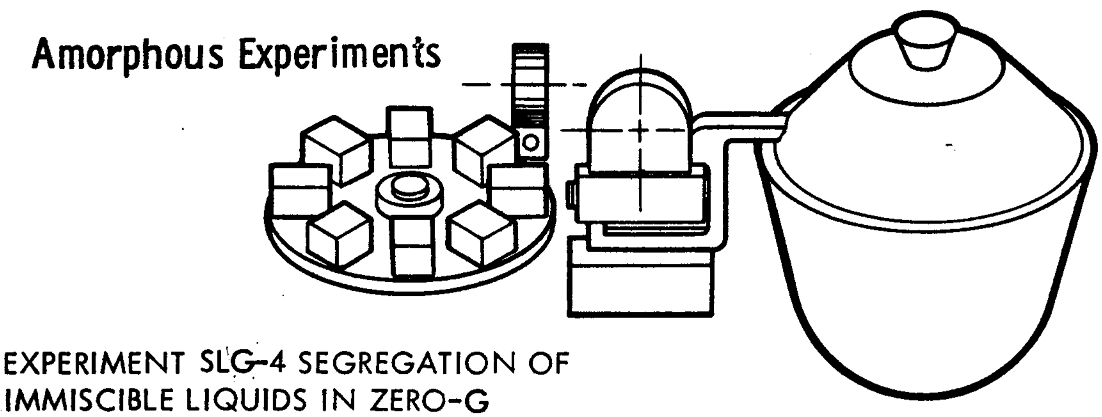
EXPERIMENT SHAPE AND VOLUME REQUIREMENTS

AMORPHOUS GEOMETRY EXPERIMENTS

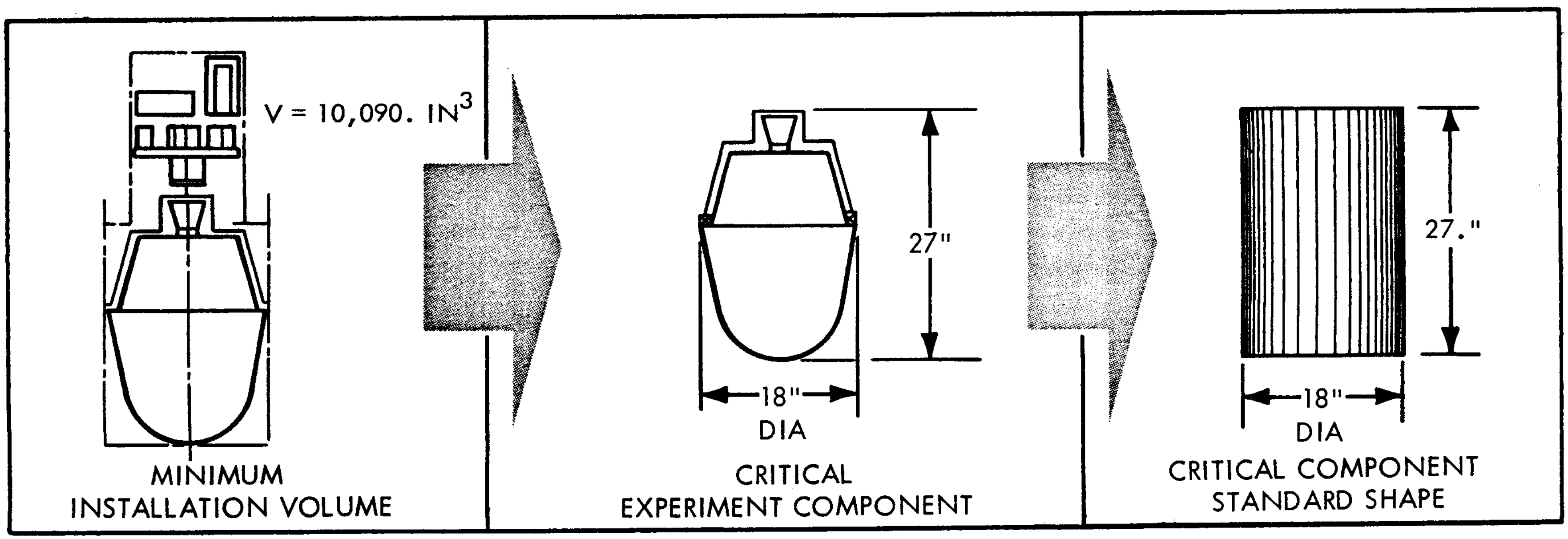
On the facing chart is shown the experiment SLG-4, Segregation of Immiscible Fluids in Zero-G. This experiment is made up of a number of components which can be arranged in a number of different ways in which the experiment is still functional. Shown also is another arrangement that represents the minimum practical volume into which the experiment can be packaged. The experiment packaged volume was computed from this packaging arrangement.

Even though the experiment is considered as an amorphous configuration, there are some critical limiting dimensions to which the experiment can be conformed. The critical component is the data recovery capsule which is 18.0 inches in diameter and 27.0 inches long. The standard shape which most efficiently represents this critical component is a cylinder of the same dimensions.

EXPERIMENT SHAPE AND VOLUME REQUIREMENTS

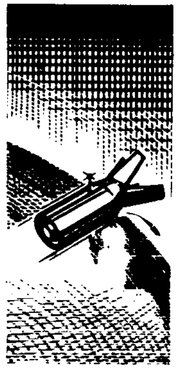


EXPERIMENT SLG-4 SEGREGATION OF
IMMISCIBLE LIQUIDS IN ZERO-G



FW/65/274-6839

Part IV



IN- FLIGHT EXPERIMENTAL PAYLOAD EFFECTIVENESS ANALYSIS

FW/65/274-6840

EXPERIMENT/MISSION EFFECTIVENESS
External Analysis

Experiment effectiveness is defined as the percent of accomplishment of data acquisition objectives. For the type of experiment 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., package 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 procedures not included in the SEPTER computer 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.

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 condition implies that no single orbit can be used to attain all the data acquisition objectives of the scientific program. Finally, the effectiveness data are prepared for inclusion in the effectiveness section of the Experiment Characteristics Library.

EXPERIMENT/MISSION EFFECTIVENESS
External Analysis

Definitions:

- EXPERIMENT EFFECTIVENESS: PERCENT ACCOMPLISHMENT OF DATA ACQUISITION OBJECTIVES

- EXPERIMENT DESIGN ORBIT: THE INITIAL ORBIT WHICH YIELDS MAXIMUM EXPERIMENT EFFECTIVENESS

Effectiveness Analysis:

- DEFINE DATA ACQUISITION OBJECTIVES
- DETERMINE ORBITAL ELEMENTS WHICH AFFECT EXPERIMENT EFFECTIVENESS
- GENERATE DATA NECESSARY TO EXPRESS EFFECTIVENESS AS A FUNCTION OF THE INITIAL ORBITAL ELEMENTS
- DETERMINE THE INITIAL ORBITAL ELEMENTS WHICH YIELD MAXIMUM EFFECTIVENESS (EXPERIMENT DESIGN ORBIT)
- PREPARE **SEPTER** EFFECTIVENESS LIBRARY

DETERMINATION OF ORBITAL ELEMENTS WHICH INFLUENCE EXPERIMENT EFFECTIVENESS

The data acquisition objectives, the flight regime, and the operational aspects of each experiment are analyzed to identify the orbital elements which influence effectiveness. The important flight regime and operations characteristics of representative experiments (one from each of the six experiment categories listed under "Inflight Experiment Definitions") are summarized on the facing page. Experiment effectiveness of each experiment can be expressed as a function of the initial orbital elements shown in the last column.

DETERMINATION OF ORBITAL ELEMENTS WHICH INFLUENCE EXPERIMENT EFFECTIVENESS

IDENTIFICATION		TITLE	FLIGHT REGIME	OPERATIONS					ORBITAL ELEMENTS AFFECTING EFFECTIVENESS
CATE-GORY	NO.			DURATION	DUTY CYCLE	ATTITUDE	DATA RATE	DEPLOYMENT MODE	
SDT	4	CRYOGENIC PROPELLANT STORAGE SYSTEM PERFORMANCE	NO RESTRICTIONS	3 TO 4 DAYS	CONTIN-UOUS AFTER INJECTION	TOWARD SUN $\pm 3^\circ$	500-1000 BITS/ ORBIT	SEPARA-TION	• PERIGEE ALTITUDE • APOGEE ALTITUDE • INCLINATION • RT. ASCENSION OF ASCENDING NODE
MS	3	VAPORIZATION RATE OF MOLTEN METALS	• APPROXIMATELY CIRCULAR ORBIT • PERIGEE $>$ 148 KM	29 HRS	0.3 TO 4 HR RUNS	LOW TUM-BLING RATE	26 HR TAPE	EXTENSION	• PERIGEE ALTITUDE • APOGEE ALTITUDE
MI	1	MULTI-SPECTRAL SURVEILLANCE OF EARTH	• ALTITUDE RANGE OVER LAND: 185-250 KM OVER WATER: $<$ 370 KM • SUN ANGLE $> 30^\circ$ ABOVE HOR • INCLINATION DEPENDS ON TARGET	2 TO 14 DAYS	40 1 MIN RUNS	UP TO $\pm 30^\circ$ FROM VERTICAL $\pm 1.5^\circ$ POINTING ACCURACY	DATA EVERY 3 SEC	SEPARA-TION	• PERIGEE ALTITUDE • APOGEE ALTITUDE • INCLINATION • RT. ASCENSION OF ASCENDING NODE
SLG	1	BOILING IN ZERO-GRAVITY ENVIRONMENT	• ACCELERATION $< 10^{-4}$ g's	2 HRS	ONE 2 HR CYCLE OF 10 RUNS	ANTENNA ANGLE $\pm 20^\circ$ FROM DOWN POSITION	10,000 FRAMES /SEC 10 FRAMES / RUN	• EXTENSION • PROPUL-SIVE SEPARA-TION	• PERIGEE ALTITUDE • APOGEE ALTITUDE
M	1	SOFT CAPTURE, ENUMERATION AND IDENTIFICATION OF SPACE-BORNE MICROORGANISMS	NO RESTRICTIONS	14 DAYS	SEVERAL DAYS	PLATES NORMAL TO FLIGHT PATH	100-500 PICTURES/ MISSION	SEPARA-TION	• PERIGEE ALTITUDE • APOGEE ALTITUDE
OEA	2	STUDY OF MAGNETIC FIELD LINES	• ALTITUDE $<$ 370 KM • INCLINATION $\approx 90^\circ$	50 ORBITS	15 MIN/ RUN 50 RUNS	ALONG FIELD LINES $\pm 5^\circ$	4000 BITS/ RUN	SEPARA-TION	• PERIGEE ALTITUDE • APOGEE ALTITUDE • INCLINATION

FW/65/274-6842

EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS
MS-3 VAPORIZATION OF MOLTEN METALS

The extent and complexity of the experiment effectiveness analyses vary considerably with the various experiments which are being considered. In the following charts, the effectiveness analyses of two experiments are reviewed. The first experiment, MS-3, Vaporization of Molten Metals, provides an illustration of one of the more complex effectiveness relationships.

The objective of the experiment is to determine the vaporization characteristics of molten metals in near Earth orbits. To obtain 100 percent effectiveness, the experimenter has 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.

EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS

- Experiment: MS-3, Vaporization of Molten Metals

- Data Acquisition Objective: Determine Vaporization Characteristics of Molten Metals in Near Earth Orbits

- 100% Effectiveness Requirements: Atmospheric Pressure Environment of Less Than 10^{-7} mmHg ($\approx 1.9 \times 10^{-9}$ psi) for an Experiment Duration of 29 Hours

- Effectiveness Variables: Atmospheric Pressure, Time

FW/65/274-6843

DETERMINATION OF TIMING EFFECTIVENESS FACTOR
EXAMPLE EXPERIMENT MS-3

Vaporization rates of various metals have been previously calculated from theory as illustrated in the upper left graph. The purpose of this experiment is to verify these predicted rates at pressures less than about 10^{-7} millimeters of Hg. To accomplish this objective, the experimenter planned 18 individual vaporization tests by using eight selected materials. Each material was rated on the basis of the scientific and practical value of the data it could yield (e.g., in the experiment plan, 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, so that the amount and value of test data would decrease with time.

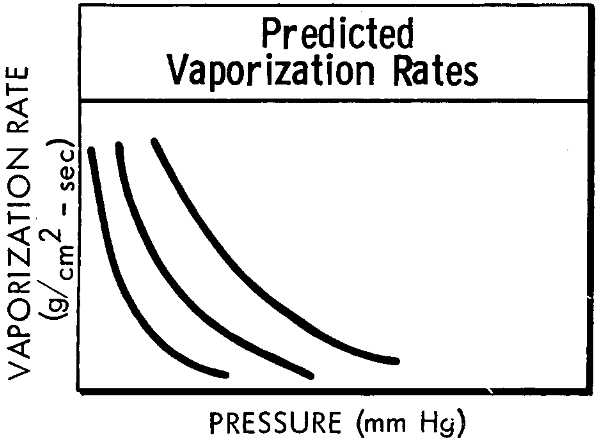
From the experiment schedule (partially illustrated in the facing chart), the experimenter subjectively evaluated the data yield as a function of experiment duration to arrive at a timing effectiveness function $E_t(t)$, shown in the upper right graph. The timing effectiveness factor, E_t, is then an index of the amount and value of the test data accumulated at any time that the experiment is terminated.

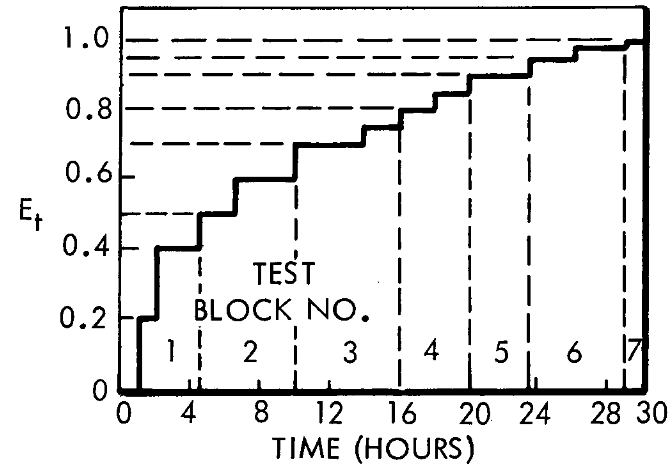
The experimenter also assigned altitude "weighting factors", f_h, to each of several blocks of tests, as shown in the table at the extreme right. These factors were used for computing the "effective" altitude (as defined in the next chart) of each block of tests. In the case of short tests, which occur near apogee, the factor is nearly 1.0, consequently the effective altitude is weighted toward the average apogee altitude.

DETERMINATION OF TIMING EFFECTIVENESS FACTOR

Example Experiment MS-3

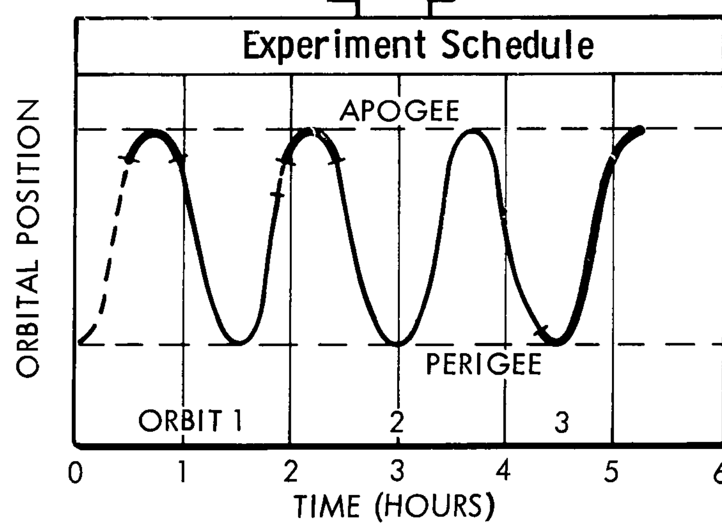
- An Index of the Number and Value of Test Data





Timing Effectiveness

Experiment Plan			
TEST NO.	MAT.	t_x	RATING
1	M_g	20	A
2	Al	60	A
3	Sb	20	C
4	Al-M_g	120	B
•	•	•	•



Altitude Weighting Factor		
TEST BLK	TEST NO.	f_h
1	1 → 4	0.51
2	5 → 8	0.52
3	9 → 12	0.52
4	13, 14	0.47
5	15, 16	0.55
6	17, 18	0.50
7	18	0.50

FW/65/274-6844

DETERMINATION OF ALTITUDE EFFECTIVENESS FACTOR
EXAMPLE EXPERIMENT MS-3

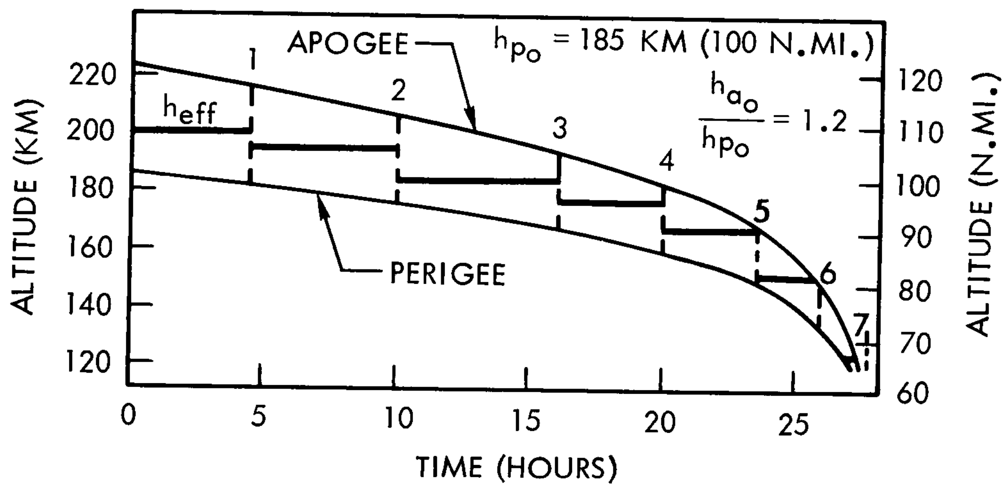
Because of the time variance of altitude (and therefore atmospheric pressure), the concept of effective altitude was introduced to simplify the experiment effectiveness analysis. For a given test, the effective altitude is defined as the sum of the perigee altitude and the weighted average of perigee and apogee altitudes. The weighting factor, f_h, was arrived at subjectively by the experimenter, as stated previously. When significant decay of perigee and apogee altitude occurs during the experiment, the tests are divided into blocks and an effective altitude is determined for each block. The effective altitude is determined by adding the weighted average of the difference between the perigee and the apogee altitudes to the average perigee altitude. (See equation on facing page.) The experimenter then defines an altitude effectiveness factor, E_h, which is a function of the calculated effective altitude. This relationship is plotted in the lower left graph on the facing page.

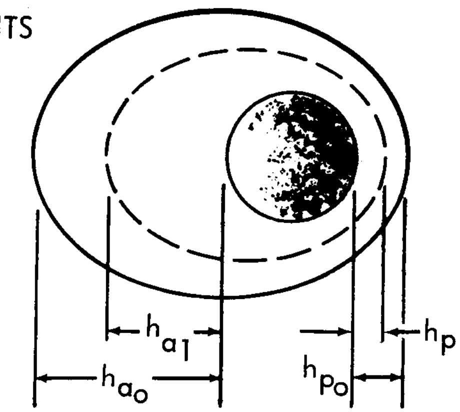
Also shown on the facing page is an example analysis of an orbit with an initial perigee altitude of 185 kilometers (100 nautical miles) and an initial apogee/perigee altitude ratio of 1.2. The experiment was divided into 7 test blocks, and the effective altitude was computed for each block. From the altitude effectiveness curve, E_h was determined and tabulated for each test block. From the tabulated data, it can be seen that the altitude effectiveness decreases rapidly after test 16 and becomes zero for the last test.

The procedure illustrated in this example was repeated for a matrix of initial perigee altitudes and apogee/perigee altitude ratios to complete this portion of the analysis.

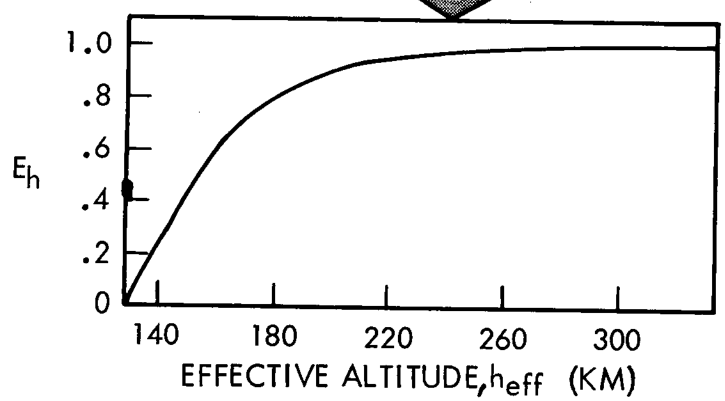
DETERMINATION OF ALTITUDE EFFECTIVENESS FACTOR
EXAMPLE EXPERIMENT MS-3

- EXAMPLE ANALYSIS FOR ONE SET OF INITIAL ORBITAL ELEMENTS





$$h_{eff} = \frac{1}{2}(h_{p_o} + h_{p_1}) + \frac{1}{2} f_h \left[(h_{a_o} + h_{a_1}) - (h_{p_o} + h_{p_1})\right]$$



TEST BLOCK NO.	TEST NO.	E_h
1	1–4	0.91
2	5–8	0.89
3	9–12	0.84
4	13,14	0.80
5	15,16	0.72
6	17,18	0.49
7	18	0

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

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

Experiment effectiveness is shown on the facing page as a function of the initial orbital elements (i.e., perigee altitude and apogee/perigee altitude ratio). These data, will be loaded into the effectiveness segment of the Experiment Characteristics Library.

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

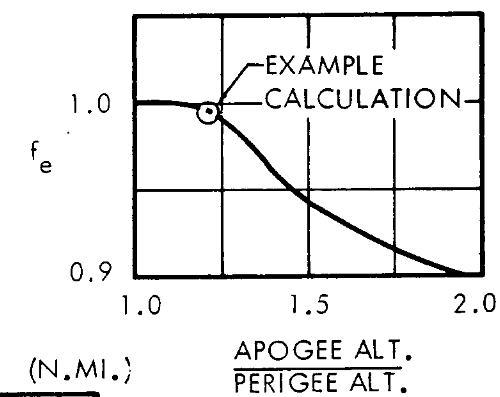
A launch-vehicle-related factor, tumbling rate, has a secondary influence on the overall effectiveness of experiment MS-3. This factor is significant only at very high tumbling rates and its value was taken as 1.0.

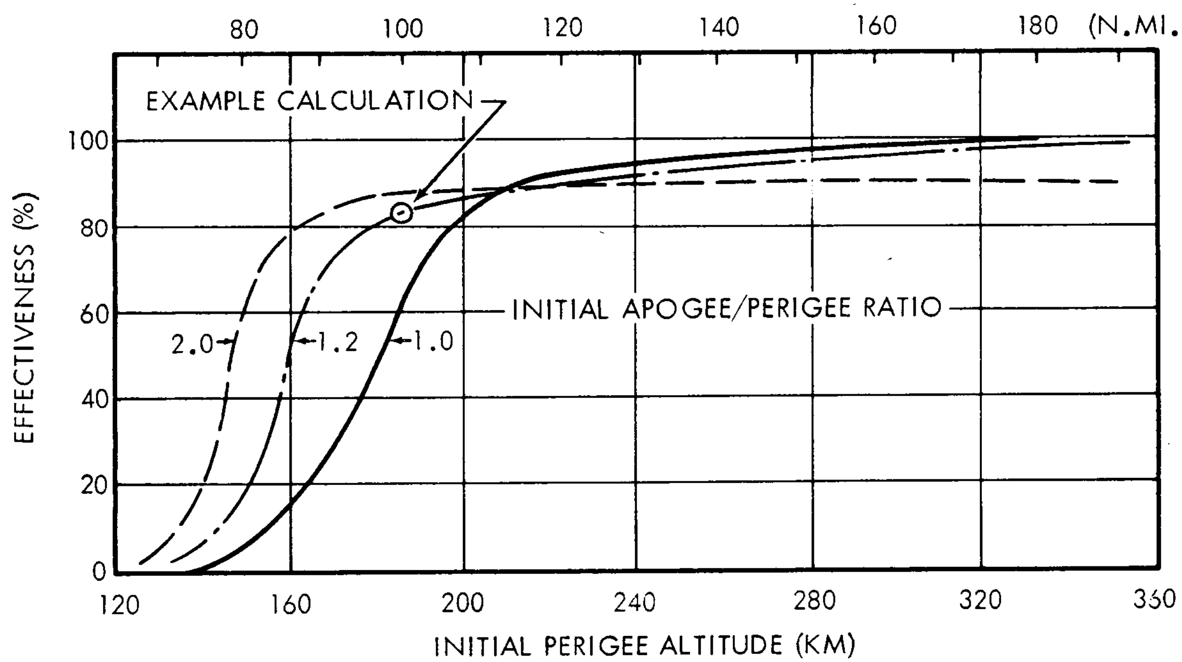
EFFECTIVENESS AS A FUNCTION OF THE INITIAL ORBITAL ELEMENTS

Example Experiment MS-3

$$E = \left(\sum_{1}^{n} \Delta E_t \times E_h\right) f_e = 0.84 \text{ For Example Calculation}$$

$$\Delta E_t = E_{t_n} - E_{t_{n-1}}$$





FW/65/274-6846

EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS

SLG-2, NUCLEATE CONDENSATION IN ZERO GRAVITY

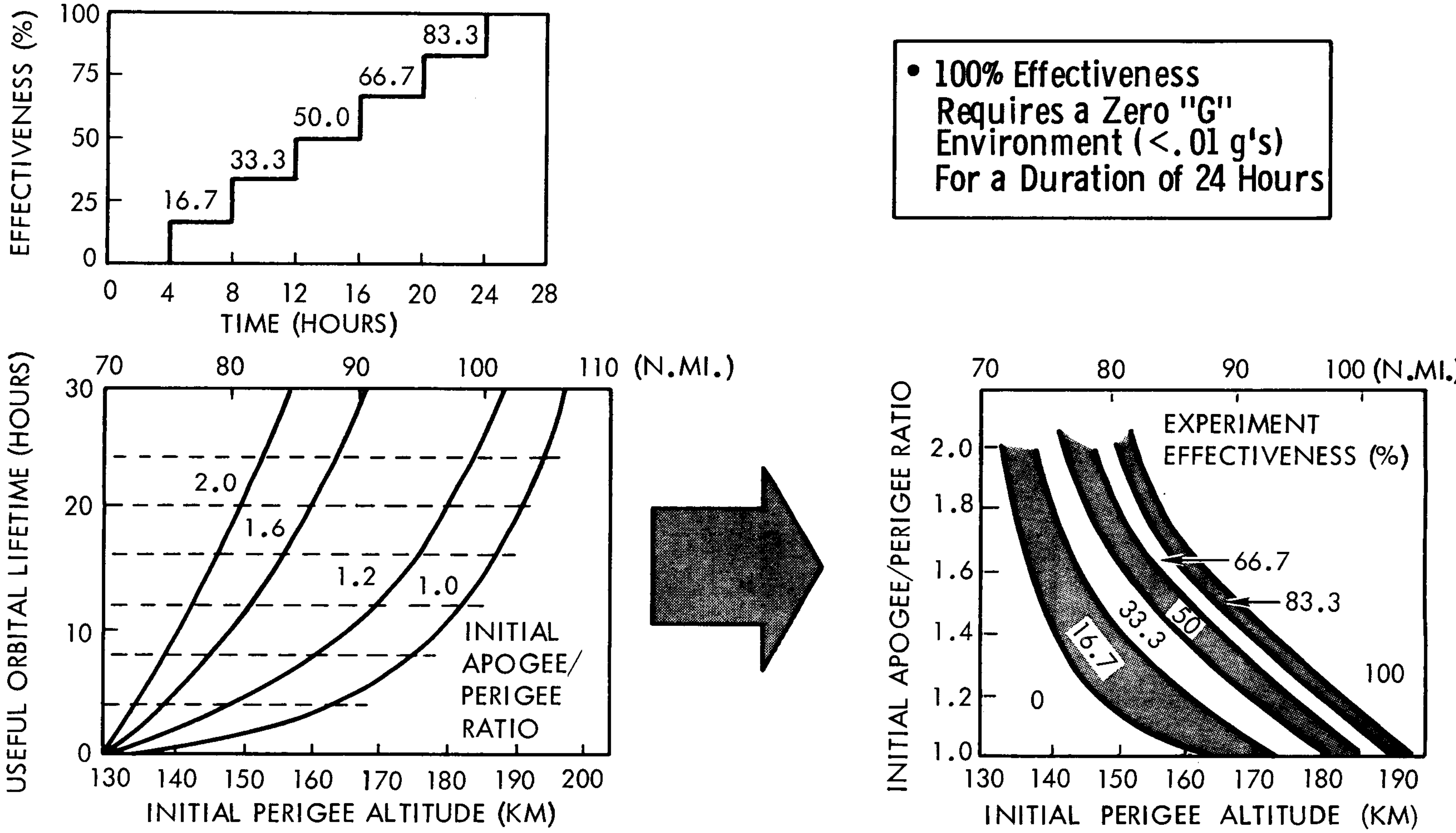
A second example of an experiment effectiveness analysis is presented to illustrate a simpler and perhaps more typical effectiveness relationship. The objective of experiment SLG-2 is to observe nucleate condensation in a zero-gravity environment. Data are recorded on magnetic tape and relayed back at four-hour intervals. To achieve 100-percent effectiveness, the drag acceleration must be less than .01 g for the duration of the experiment (24 hours). Should the experiment terminate 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.

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 .01 g (i.e., perigee is less than about 130 kilometers). From the lifetime data, perigee versus apogee/perigee altitude curves were generated for 4-, 8-, 12-, 16-, 20-, and 24-hour orbital lifetimes (corresponding to 16.7-, 33.3-, 50-, 66.7-, 83.3-, and 100-percent effectiveness, 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.

EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS

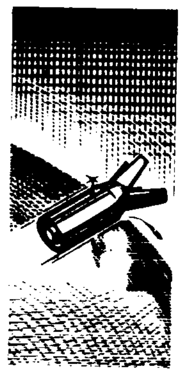
Experiment: SLG-2, Nucleate Condensation in Zero Gravity



- 100% Effectiveness Requires a Zero "G" Environment (<.01 g's) For a Duration of 24 Hours

FW/65/274-6847

Part V



IN-FLIGHT EXPERIMENT CAVITY LOCATIONS

FW/65/274-6848

IN-FLIGHT EXPERIMENT CAVITY LOCATIONS

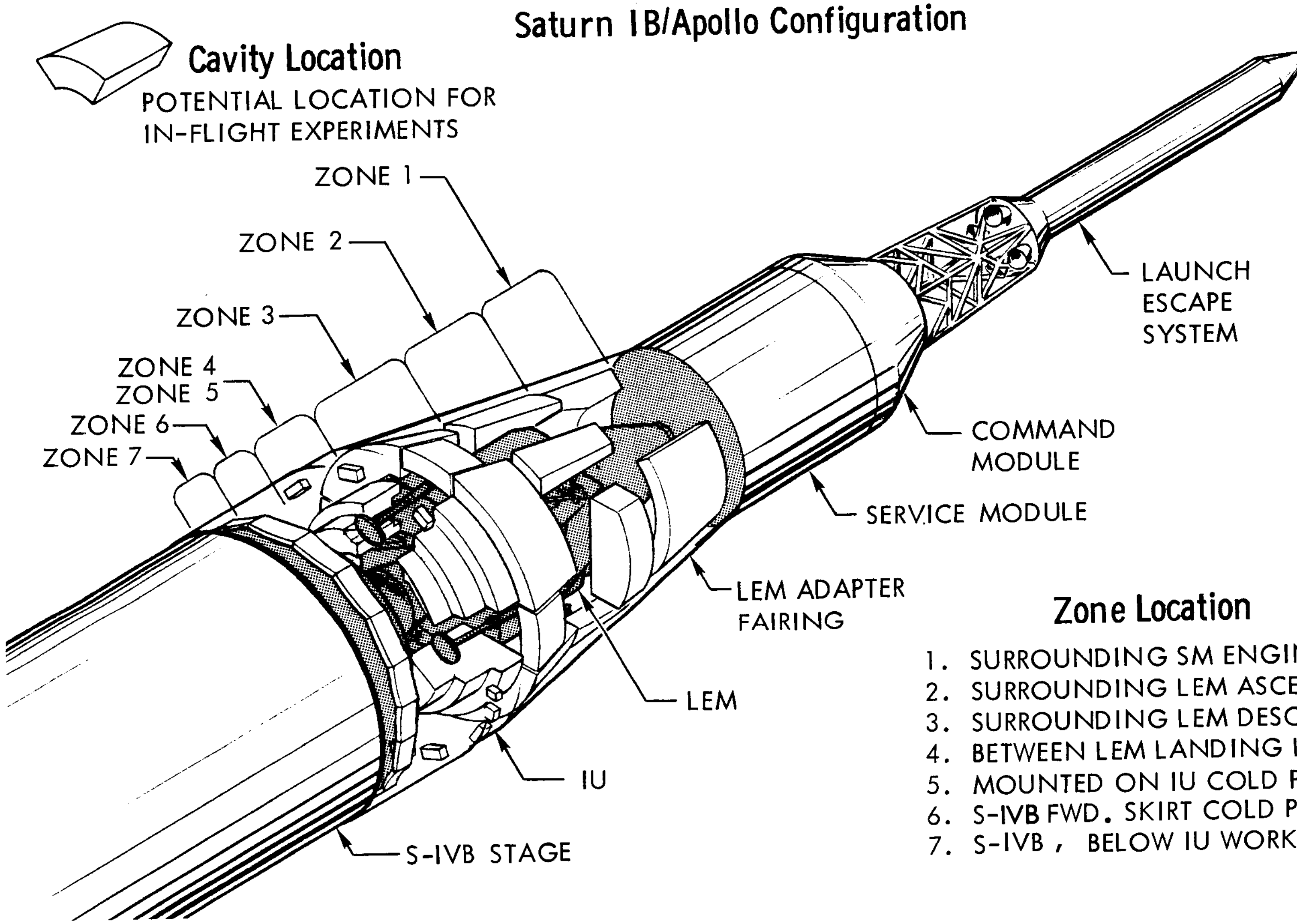
The baseline configuration selected for use in this study is the Saturn IB/Apollo with Command Module, Service Module and LEM. The areas considered in this study as potential experiment locations are (1) the LEM adapter fairing, (2) the Instrument Unit, and (3) the SIVB Stage forward skirt. To simplify the descriptions of the location, these areas have been broken down into 7 zones. Each zone contains several individual cavities which are the potential locations for experiments. The locations of the 7 zones are:

Zone 1 - Surrounding Service Module Engine
Zone 2 - Surrounding LEM Ascent Stage
Zone 3 - Surrounding LEM Descent Stage
Zone 4 - Between LEM Landing Legs
Zone 5 - Mounted on Instrument Unit Cold Panels
Zone 6 - Mounted on SIVB Stage Forward Skirt Cold Panels
Zone 7 - Mounted on SIVB Stage Forward Skirt Below IU Work Platform.

Cavities for all zones have been defined with exception of Zone 6. The information necessary to complete this definition is not available at this time but will be available during the second half of the study.

IN-FLIGHT EXPERIMENT CAVITY LOCATIONS

Saturn IB/Apollo Configuration



Zone Location

1. SURROUNDING SM ENGINE SKIRT
2. SURROUNDING LEM ASCENT STAGE
3. SURROUNDING LEM DESCENT STAGE
4. BETWEEN LEM LANDING LEGS
5. MOUNTED ON IU COLD PANELS
6. S-IVB FWD. SKIRT COLD PANELS
7. S-IVB , BELOW IU WORK PLATFORM

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SELECTION AND IDENTIFICATION OF CAVITY LOCATIONS

The zone divisions, as shown on the previous chart, were made in order that the cavities contained in each zone would exhibit a similarity in terms of locations, environments, and installation requirements. The cavity shapes were obtained by providing the following clearances from the Saturn IB/Apollo vehicle components:

1. A six-inch clearance was allowed between each cavity and the LEM, Service Module, SIVB Tank, and all Work Platform Locations to provide adequate space for installation and maintenance.

2. A three-inch clearance between each cavity and the LEM adapter fairing was provided for additional mounting structure.

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

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

5. Required clearances from components on cold panels was dictated by NASA Report, Preliminary Definition of Saturn Instrument Unit and SIVB Support Capabilities, dated 15 April 1965.

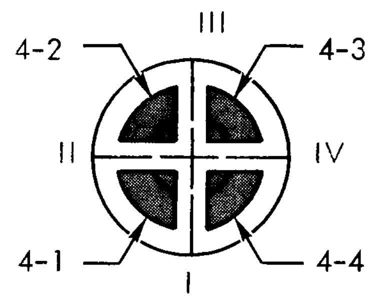
The zones were numbered in sequence, beginning with the zone nearest the Service Module and progressing aft to the SIVB Stage. Separate zones were provided for the Instrument Unit and SIVB cold panels. The cavities in each zone are identified as separate dash numbers of that zone. Cavities are numbered beginning at position 1 which will be down in Earth orbit and continue clockwise looking forward on the vehicle.

SELECTION AND IDENTIFICATION OF CAVITY LOCATIONS

- **Zone Selection**
  - LOCATION
  - ENVIRONMENT
  - INSTALLATION REQUIREMENTS

- **Cavity Selection**
  - SIX INCH CLEARANCE FROM LEM, SM, S-IVB TANK, AND WORK PLATFORMS
  - THREE INCHES FROM LEM ADAPTER FAIRING FOR MOUNTING STRUCTURE
  - CLEARANCE FOR EXTRACTION OF LEM
  - DIRECT ATTACHMENT TO IU AND S-IVB COLD PANELS
  - CLEARANCE FROM ELECTRONIC EQUIPMENT ON COLD PANELS PER NASA REPORT

- **Cavities Identified by Zone Number and Dash Number**
  - ZONE NUMBERS
    - FORWARD TO AFT SEQUENCE
    - SEPARATE ZONES FOR IU AND S-IVB COLD PANELS
  - DASH NUMBERS
    - BEGIN AT POSITION I
    - PROGRESS CLOCKWISE LOOKING FORWARD



ZONE 4 CAVITIES

FW/65/274-6850

CAVITY DESCRIPTION

In the six zones defined to date, a total of 48 cavities are available as possible experiment locations. During the first half of the study, each of these cavities has been described as follows:

1. An isometric drawing was made to provide dimensions and orientation.

2. The total volume was computed.

3. The shape was defined on the basis of its capacity to contain the three standard shapes: cylinders, spheres, and parallelepipeds.

4. The thermal environment was defined in terms of (a) the maximum and minimum temperatures, (b) the maximum heat dissipation rate, and (c) the total heat dissipation capability.

5. The structural loads and criteria were defined in terms of (a) the maximum overall sound pressure level, (b) the maximum acceleration, (c) the sinusoidal vibration level, and (d) the load-carrying capability of the adjacent launch vehicle support structure.

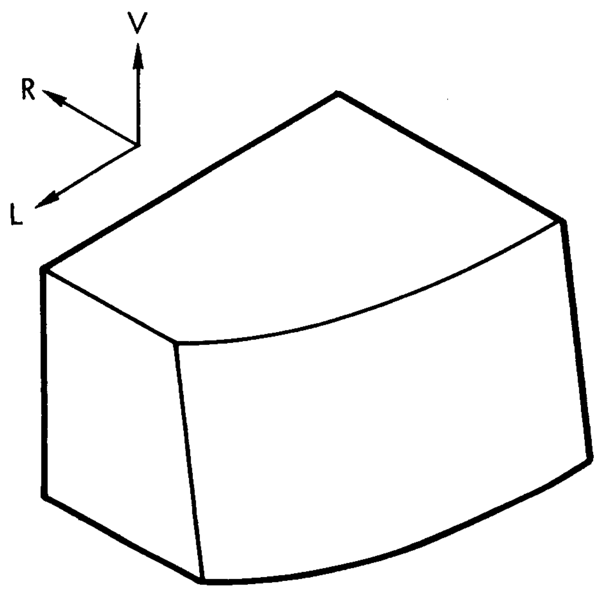
6. The operational capabilities were used to describe the capability of the cavity to contain an experiment in which ejection or deployment is required or a view externally from the vehicle is required.

7. The description of the electromagnetic environment of the cavities will be completed during the second half of the study.

The system of axis used in defining dimensions and orientation of the cavities is defined below:

V axis - Generally parallel to vehicle longitudinal axis
R axis - Generally normal to the external surface of the vehicle
L axis - 90 degrees to the V and R axis.

CAVITY DESCRIPTION



TYPICAL CAVITY NO. 2-5

V AXIS - GENERALLY PARALLEL TO VEHICLE LONGITUDINAL AXIS

R AXIS - GENERALLY NORMAL TO EXTERNAL SURFACE OF VEHICLE

L AXIS - 90° TO V AND R AXIS

- **Geometrical Description of Shape**
- **Total Volume**
- **Standard Shapes Capacity**
  - CYLINDER
  - SPHERE
  - PARALLELEPIPED
- **Thermal Environment**
  - MAX AND MIN AVERAGE TEMP
  - MAX HEAT DISSIPATION RATE
  - TOTAL HEAT DISSIPATION
- **Structural Loads and Criteria**
  - MAX OVERALL SOUND PRESSURE LEVEL
  - MAX VEHICLE ACCELERATION
  - SINUSOIDAL VIBRATION LEVEL
  - LOAD CARRYING CAPABILITY OF SUPPORT STRUCT
- **Operation Capability**
  - EJECTION
  - DEPLOYMENT
  - VIEW
- **Electromagnetic Environment**

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STANDARD SHAPES CAPACITY

The shape of each cavity has been defined in terms of its capacity to contain the standard geometrical shapes, the parallelepiped, the sphere, and the cylinder. This approach is compatible with the method described earlier for the representation of experiment geometry by the same standard shapes. Definition of the capacity of the standard shapes was complicated by the fact that most of the cavities were tapered. The capacities were defined in the following manner:

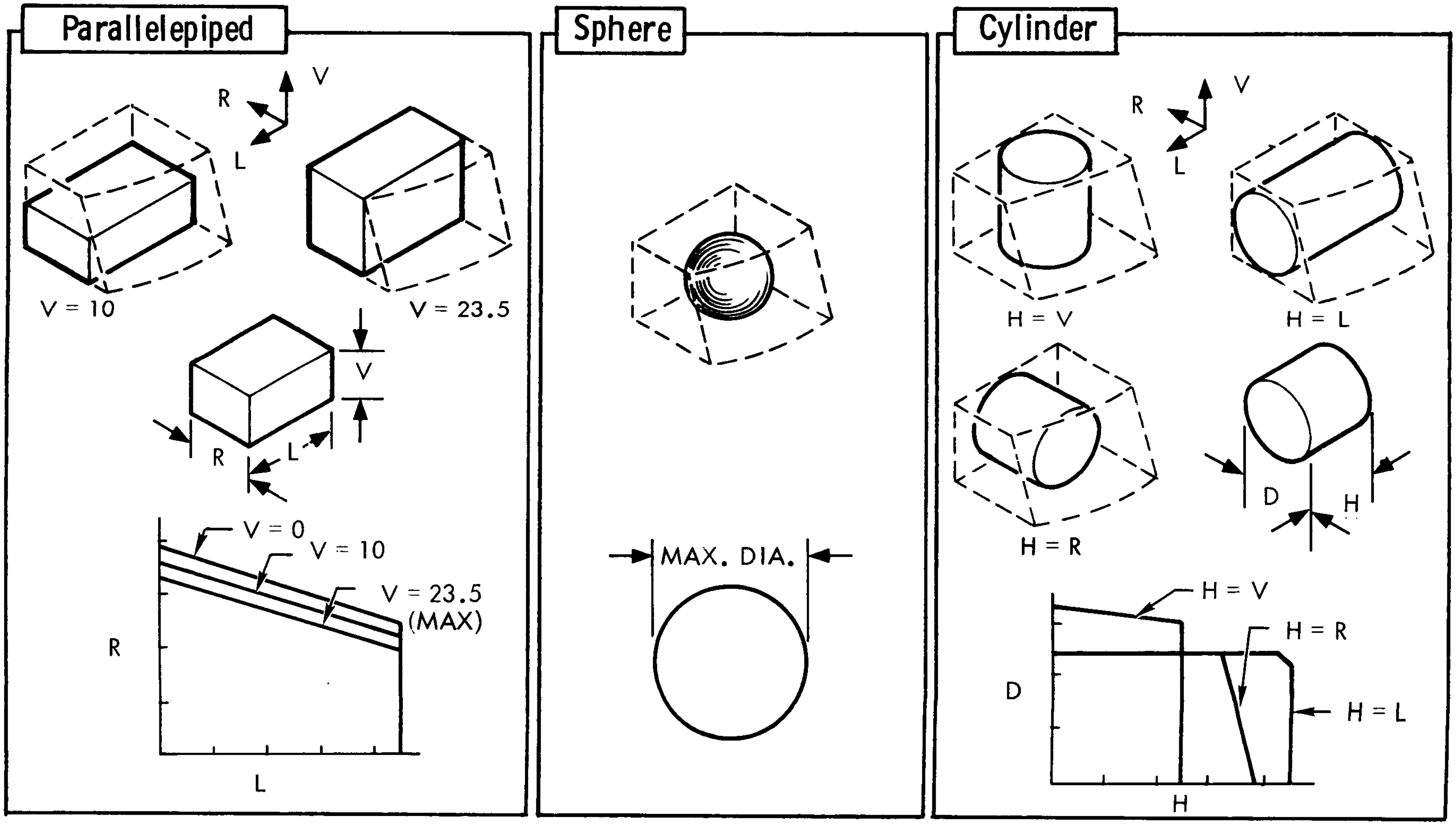
1. Parallelepiped - The axis system R, L, and V described in the previous chart are used in describing the dimensions of the parallelepipeds. Analysis of cavity drawings yielded the type of curves shown on the facing charts where the L dimension was a function of the R dimension for various values of 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. Analysis of cavity drawings yielded the type of curves shown in the facing chart. The diameter D was plotted as a function of the cylindrical length H for each of the three orientations with H equal to V, R, and L.

STANDARD SHAPES CAPACITY

EACH CAVITY DEFINED BY CAPACITY TO CONTAIN CERTAIN STANDARD GEOMETRICAL SHAPES



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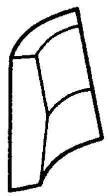
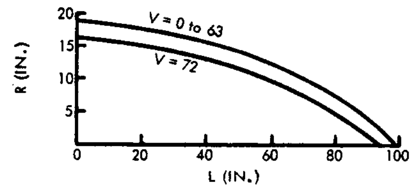
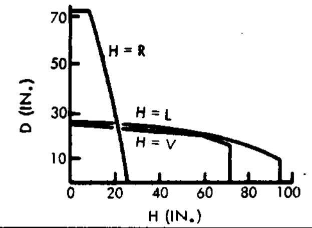
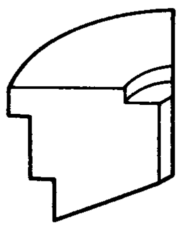
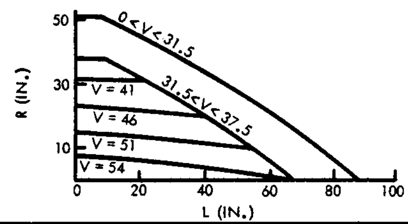
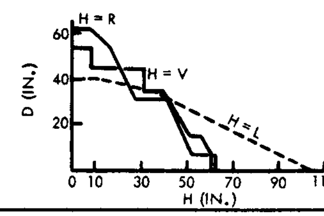
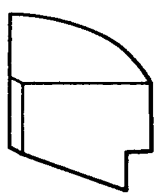
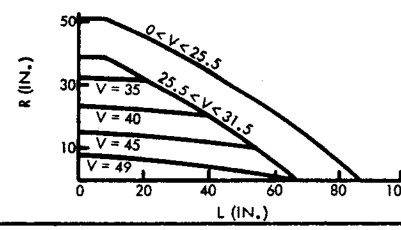
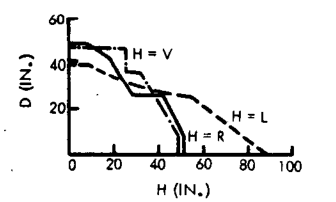
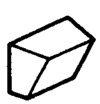
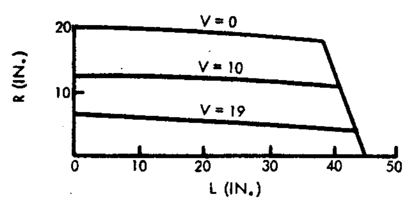
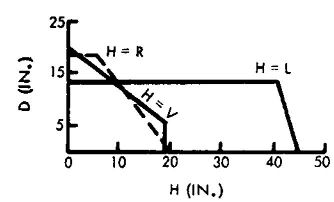
STANDARD SHAPES CAPACITY AND VOLUME SUMMARY

The capacity of the standard shapes described on the previous chart, a mass capacity, and a total volume have been determined for each of the 48 cavities defined as possible experiment locations. The information necessary to define zone 6 cavities has not yet been obtained. These cavities will be described during the second half of the study.

The mass capacity of cavities in Zones 1, 2, and 3 is based on a preliminary analysis of the load-carrying capability of each segment of the LEM adapter fairing. 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, dated June 1965. The mass capacity of cavities in Zone 5 is in agreement with MSFC document "Preliminary Definition of Saturn Instrument Unit and S-IVB Support Capabilities for Extended Apollo Earth-Orbit Experiments," dated April 15, 1965. All the mass capacities shown on the following charts are predicated on a Mode I operation in which only a single experiment per vehicle is considered. Mass capacities will be revised during the second half of the study to provide for Mode II operation.

STANDARD SHAPES CAPACITY AND VOLUME SUMMARY
ZONES 1, 4, AND 7

Cavity 1-1, 1-2, & 1-3 	● Volume ———— 163,966 in^3 ● Mass Capacity ———— 1,000 lbs ● Sphere Diameter ———— 24.5 in	Parallelepiped 	Cylinder 
Cavity 4-1, 4-2, & 4-3 	● Volume ———— 138,080 in^3 ● Mass Capacity ———— 2,500 lbs ● Sphere Diameter ———— 35.0 in	Parallelepiped 	Cylinder 
Cavity 4-4 	● Volume ———— 94,280 in^3 ● Mass Capacity ———— 2,500 lbs ● Sphere Diameter ———— 38.5 in	Parallelepiped 	Cylinder 
Cavity 7-1 thru 7-18 	● Volume ———— 9,610 in^3 ● Mass Capacity ———— 1,000 lbs ● Sphere Diameter ———— 13.2 in	Parallelepiped 	Cylinder 

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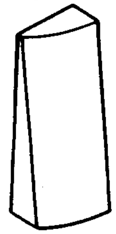
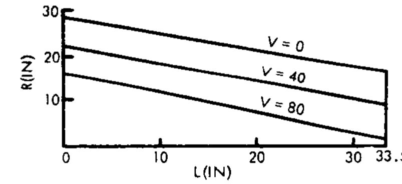
STANDARD SHAPES CAPACITY AND VOLUME SUMMARY
ZONE 2

Cavity 2-1 & 2-7

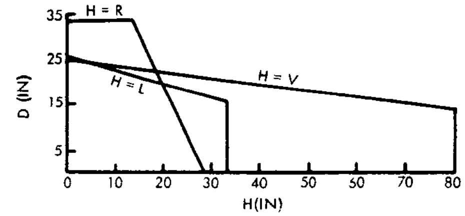


- Volume ———— 42,560 in^3
- Mass Capacity ———— 1,000 lbs
- Sphere Diameter ———— 21.0 in

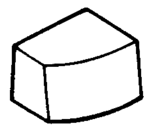
Parallelepiped



Cylinder

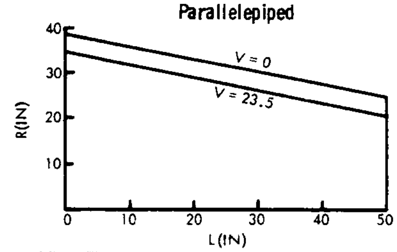


Cavity 2-5 & 2-6

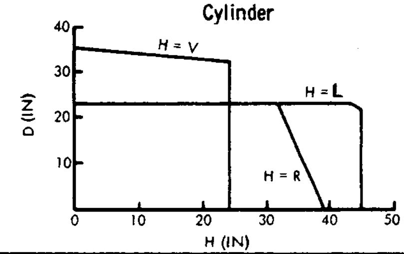


- Volume ———— 30,000 in^3
- Mass Capacity ———— 1,000 lbs
- Sphere Diameter ———— 23.5 in

Parallelepiped



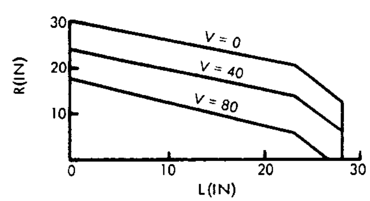
Cylinder



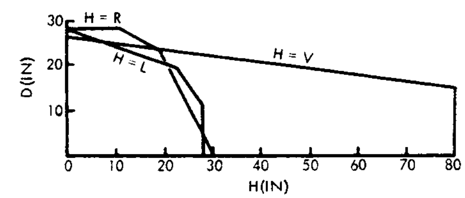
Cavity 2-3 & 2-4

- Volume ———— 43,300 in^3
- Mass Capacity ———— 1,000 lbs
- Sphere Diameter ———— 25.0 in

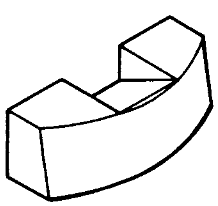
Parallelepiped



Cylinder

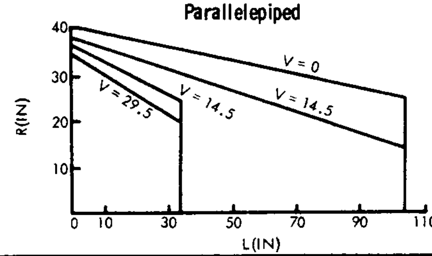


Cavity 2-2

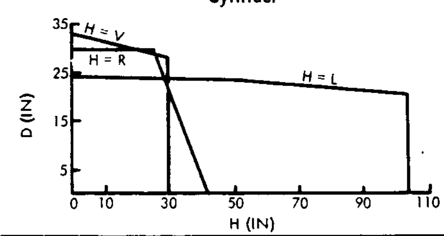


- Volume ———— 92,030 in^3
- Mass Capacity ———— 1,000 lbs
- Sphere Diameter ———— 29.5 in

Parallelepiped



Cylinder



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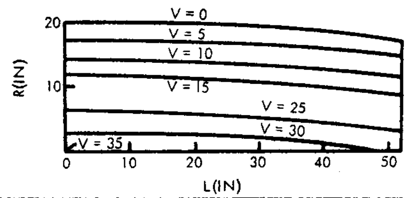
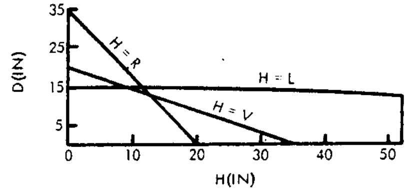
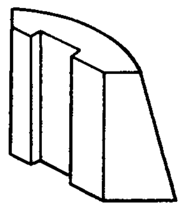
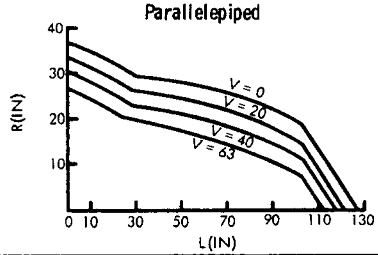
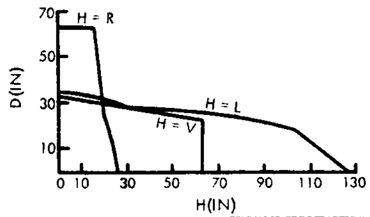
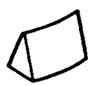
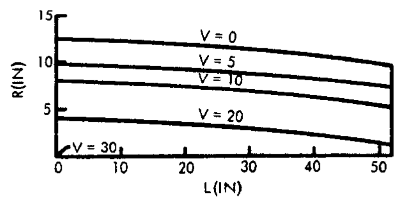
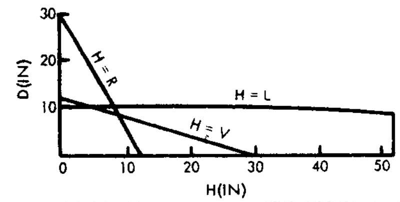
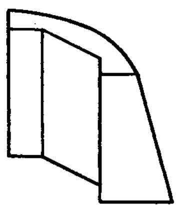
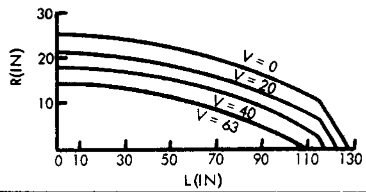
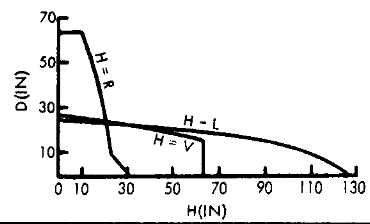
STANDARD SHAPES CAPACITY AND VOLUME SUMMARY
ZONE 3

		Parallelepiped	Cylinder
Cavity 3-1, 3-3, 3-7 	● Volume ———— 18, 200 in^3 ● Mass Capacity ——— 1, 000 lbs ● Sphere Diameter ——— 15. 0 in		
Cavity 3-2, 3-4, & 3-6 	● Volume ———— 171, 039 in^3 ● Mass Capacity ——— 1, 000 lbs ● Sphere Diameter ——— 28. 0 in	Parallelepiped 	Cylinder 
Cavity 3-5 	● Volume ———— 9, 350 in^3 ● Mass Capacity ——— 1, 000 lbs ● Sphere Diameter ——— 10. 0 in	Parallelepiped 	Cylinder 
Cavity 3-8 	● Volume ———— 132, 500 in^3 ● Mass Capacity ——— 1, 000 lbs ● Sphere Diameter ——— 22. 0 in	Parallelepiped 	Cylinder 

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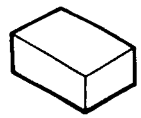
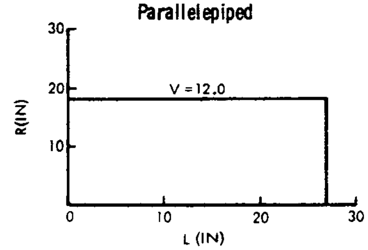
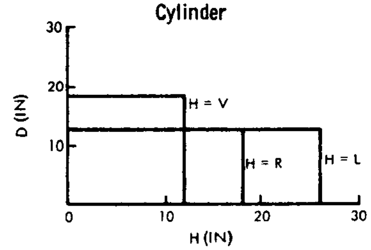
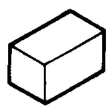
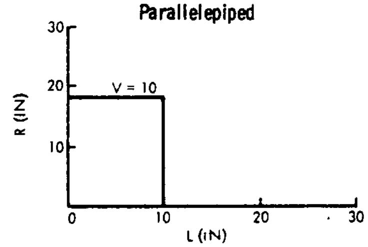
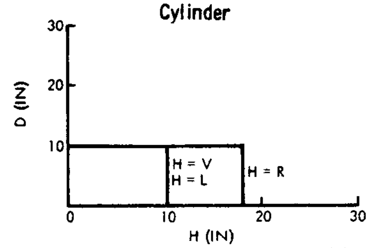
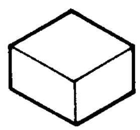
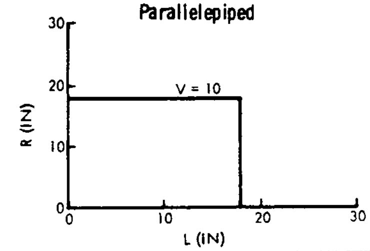
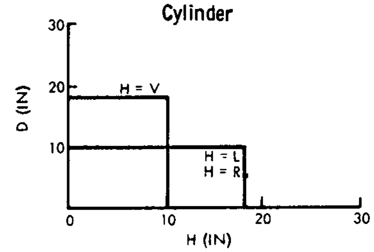
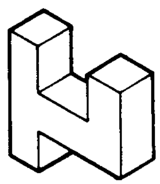
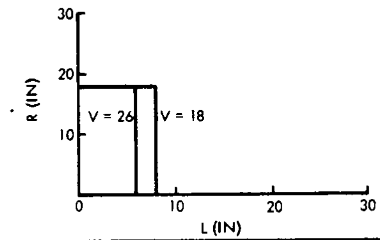
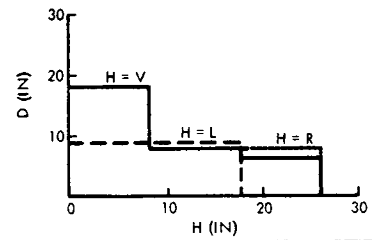
STANDARD SHAPES CAPACITY AND VOLUME SUMMARY
ZONE 5

Cavity 5-1 & 5-6 	● Volume ______ 5, 616 in^3 ● 5-1 Mass Capacity ______ 150 lbs ● 5-6 Mass Capacity ______ 297 lbs ● Sphere Diameter ______ 12.0 in	Parallelepiped 	Cylinder 
Cavity 5-2 	● Volume ______ 1, 800 in^3 ● Mass Capacity ______ 276 lbs ● Sphere Diameter ______ 10.0 in	Parallelepiped 	Cylinder 
Cavity 5-3 	● Volume ______ 3, 240 in^3 ● Mass Capacity ______ 280 lbs ● Sphere Diameter ______ 10.0 in	Parallelepiped 	Cylinder 
Cavity 5-4 	● Volume ______ 6, 950 in^3 ● Mass Capacity ______ 208 lbs ● Sphere Diameter ______ 9.3 in	Parallelepiped 	Cylinder 

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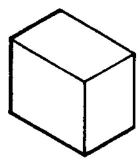
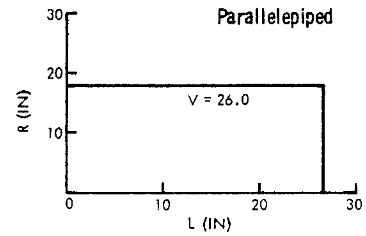
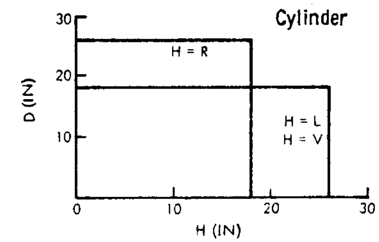
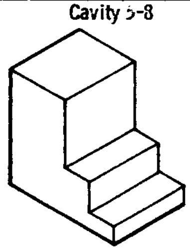
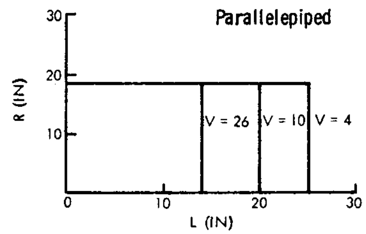
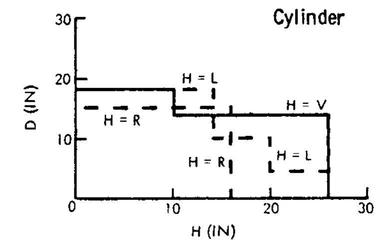
STANDARD SHAPES CAPACITY AND VOLUME SUMMARY
ZONE 5 (CONT'D)

Cavity 5-5 & 5-7 	● Volume ________ 12,168 in^3 ● Mass Capacity ______ 330 lbs ● Sphere Diameter _____ 18.0 in		
Cavity 5-8 	● Volume ________ 8,064 in^3 ● Mass Capacity ______ 286 lbs ● Sphere Diameter _____ 14.5 in		

NOTE: ZONE 6 SUMMARY TO BE ADDED WHEN INFORMATION IS AVAILABLE

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CAVITY OPERATIONAL CAPABILITIES

The cavity operational capability is defined in this study as the ability of a cavity to provide the required deployment, ejection, or view for an experiment. This description of cavities is compatible with the experiment operational requirements previously discussed in another section. The operational capabilities of the cavities are limited, first, by the configuration of the launch vehicle and, second, by the location of the cavity on the launch vehicle.

The minimum operational capabilities will occur when the Saturn IB/Apollo has the configuration shown on the left of the facing page. The maximum operational capabilities will occur when the Apollo payload has separated, and the configuration is that shown on the right of the facing page.

The base line configuration to be described in this study will be the configuration shown on the right of the facing page. For each cavity, the capabilities are described as follows:

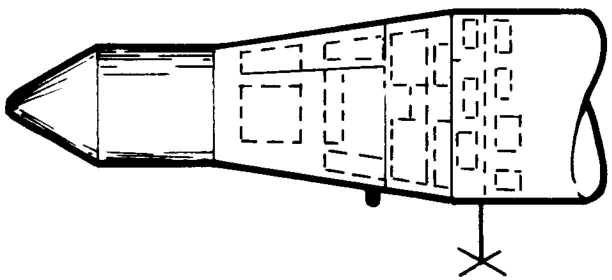
1. Directions available for deployment, ejection, or view.

2. Size limitations set on objects ejected or deployed and view aperture.

3. Times available during mission for deployment, ejection, or view.

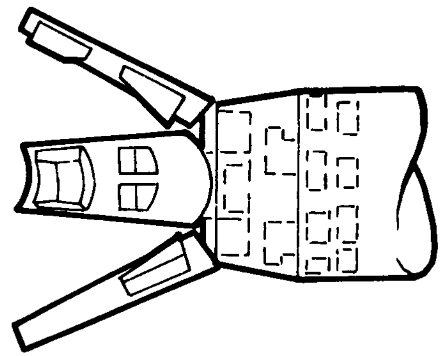
CAVITY OPERATIONAL CAPABILITIES

Minimum Operational Capability



NO APOLLO PAYLOAD SEPARATION

Maximum Operational Capability



COMPLETE SEPARATION OF APOLLO PAYLOAD

Definition

Direction Possibility For:
Time Available For:
Size Limitation For:

- Deployment
- Ejection
- View

THERMAL ENVIRONMENT

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

The maximum allowable rate of heat dissipation for all cavities varies from a maximum of 200 BTU per hour for prelaunch to a minimum of 100 BTU per hour during launch. The maximum total short period heat dissipation for all cavities and all mission phases is 17 BTU. The time-space averaged temperature varies from -105^{o}F in cavities 5-1 through 5-4 during orbit to +240^{o}F in cavities 2-1 through 2-7 during launch.

THERMAL ENVIRONMENT

PARAMETER	MISSION PHASE		
	PRELAUNCH	LAUNCH	ORBIT
● Max Allowable Rate of Heat Dissipation - All Cavities (BTU/Hr)	200	100	150
● 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
7-1 Thru 7-18	35 - 75	140 - 180	100 - 140

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ACOUSTIC AND VIBRATION ENVIRONMENT

The acoustic and vibration environments associated with each cavity have been defined in terms of a maximum overall sound pressure level and sinusoidal vibration levels. The actual values used in these definitions were extracted from the "Saturn IB Payload Planner's Guide," Douglas Report SM-47010, Douglas Missile and Space Systems Division, dated June 1965. These values represent the maximum environments to which components contained in the cavities would be subjected.

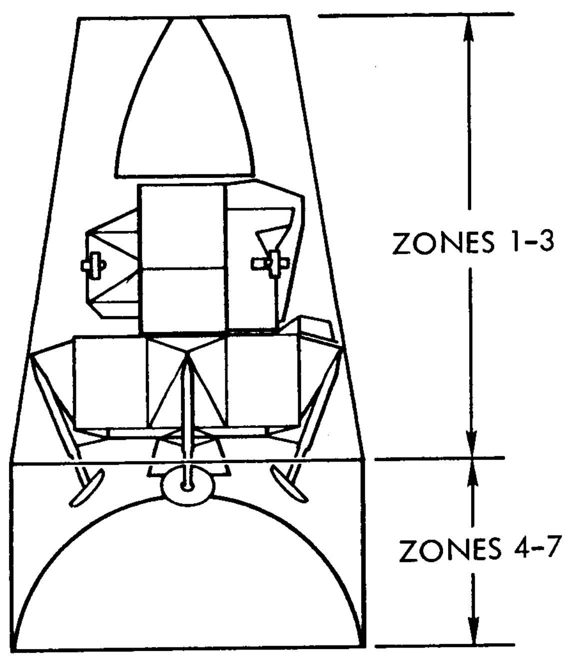
The maximum overall sound pressure level of all cavities is 151.0 db. The vibration level assigned to cavities in Zones 1 through 3 varies from .6 g at 20 cycles per second to 16 g at 200 cycles per second. The vibration level for cavities in Zones 4 through 7 varies from .9 g at 20 cycles per second to 20 g at 400 cycles per second.

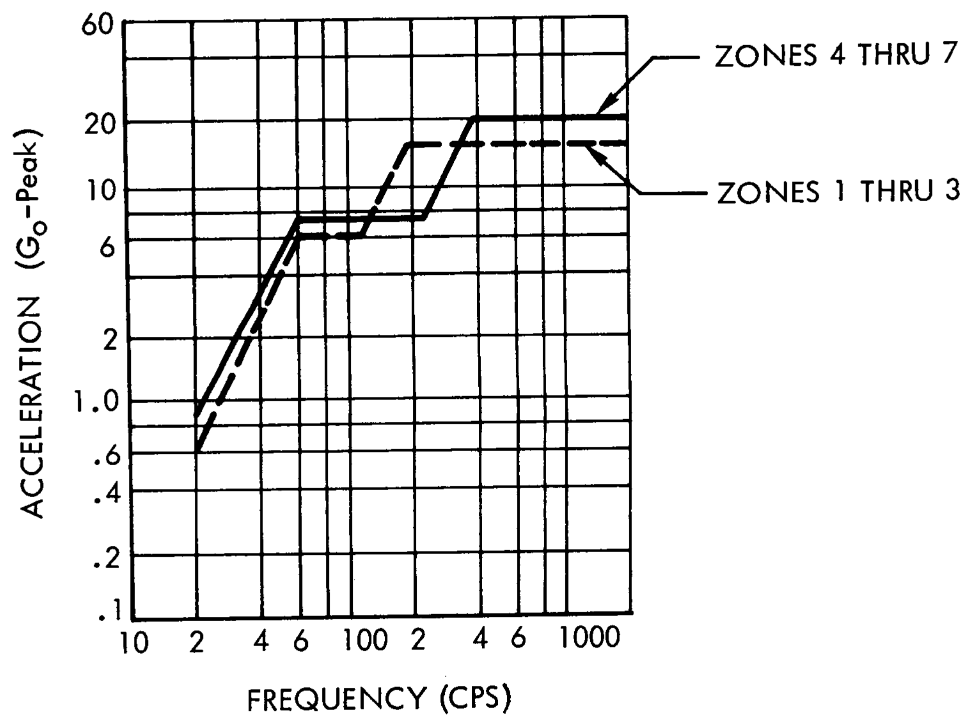
ACOUSTIC AND VIBRATION ENVIRONMENT

- ACOUSTICS - MAX OVERALL SOUND PRESS. LEVEL

ZONES 1 THRU 3	151.0 db
ZONES 4 THRU 7	151.0 db

- VIBRATION - SINUSOIDAL VIBRATION LEVELS





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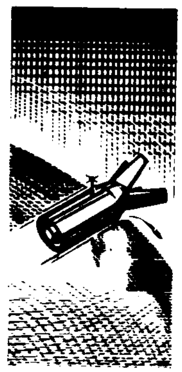
Part VI



DEFINITION AND ANALYSIS OF MISSION CHARACTERISTICS

DEFINITION AND ANALYSIS OF MISSION CHARACTERISTICS

A preliminary task in the overall development of program SEPTER was to define and analyze the pertinent mission characteristics of individual Saturn missions for inclusion in the mission characteristics libraries.

Initially, a survey was made to determine which types of missions to consider. Since the Saturn IB/Apollo vehicle/payload is being used as the basic configuration for the study, the Saturn IB scheduled flight program was investigated in particular. Representative missions were found to include both Earth orbital and suborbital (with and without powered re-entry). However, in this study, primary emphasis was placed on the definition and analysis of Earth orbital, low-altitude, low-inclination missions compatible with the Saturn/Apollo program. Suborbital missions were investigated for possible inclusion in the program, but since the time duration for nondeployable experiments is short on suborbital missions, their utilization for inflight experiments is considered of secondary importance.

The mission characteristics that were found to be pertinent to the development of the program are listed on the facing page. These characteristics were defined and analyzed as they are presented on the subsequent charts and in the following discussions.

DEFINITION AND ANALYSIS OF MISSION CHARACTERISTICS

Saturn Mission Types Considered

- EARTH ORBITAL
  - LOW-ALTITUDE
  - LOW-INCLINATION
- SUBORBITAL
  - REENTRY

Pertinent Mission Characteristics

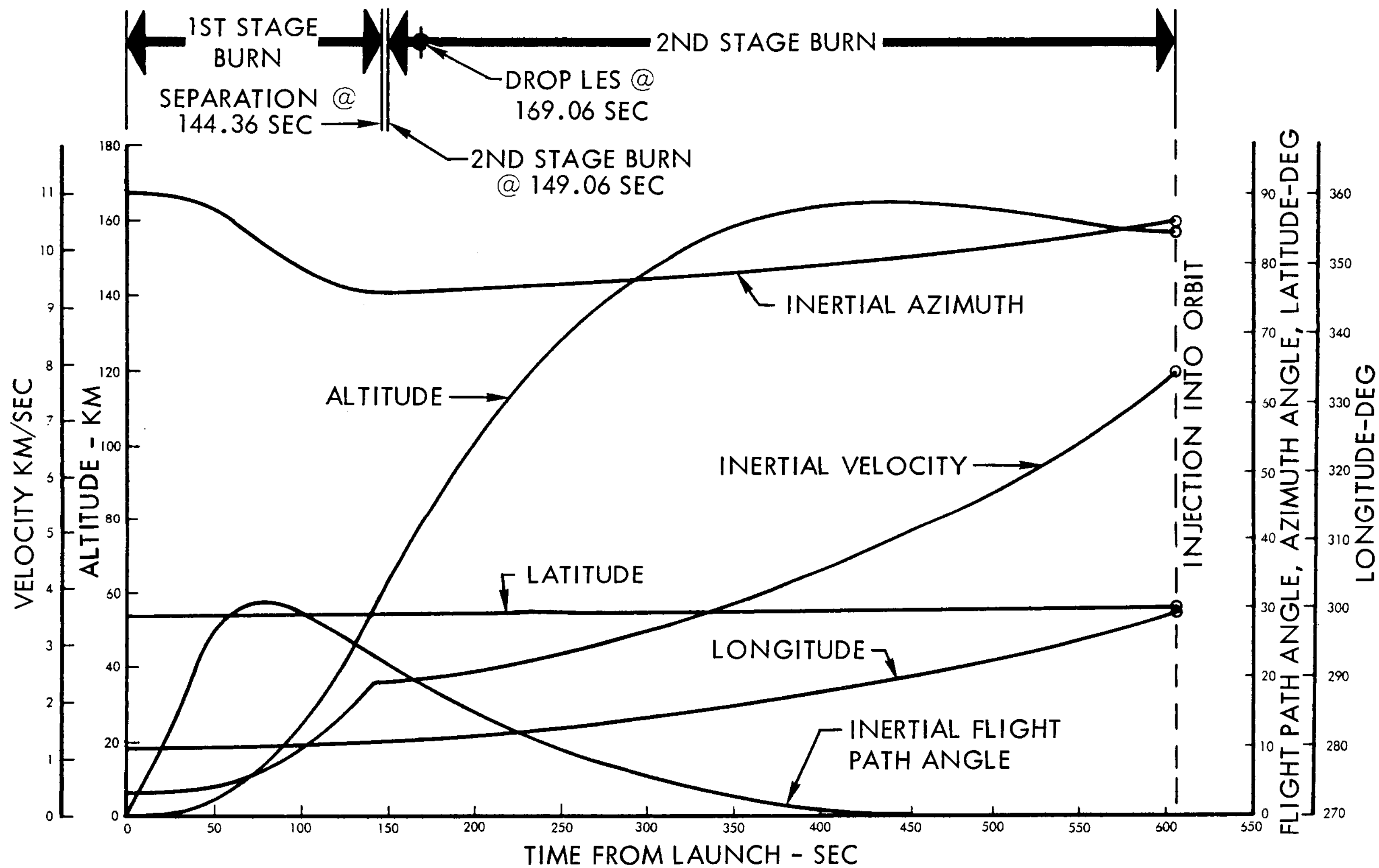
- TIME HISTORIES OF TRAJECTORY PARAMETERS
- SEQUENCE OF EVENTS
- DEPLOYMENT OPPORTUNITIES/CONSTRAINTS
  - BEFORE INJECTION OF PRIMARY PAYLOAD
  - AFTER INJECTION OF PRIMARY PAYLOAD
- VARIATION OF ORBITAL ELEMENTS ALONG LAUNCH TRAJECTORY
- EFFECT OF OPTIONAL ΔV APPLICATION AT EXPERIMENT DEPLOYMENT

SATURN IB TYPICAL LAUNCH TRAJECTORY TIME HISTORY

Time histories of the trajectory parameters of a typical Saturn IB launch trajectory are shown on the facing page. These parameters 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). These six parameters are a complete specification of the orbital elements at a given time, and they can be used to determine the orbital elements of an experiment payload deployed prior to, at, or a short time after injection of the primary payload.

Also shown on the facing chart are sequence-of-events data which are used to define experiment deployment opportunities or constraints along the launch trajectory. Staging and jettisoning of hardware times are of particular importance in analyzing deployment opportunities/constraints. In addition, vehicle position and orientation data may be required for experiment payloads with a specified requirement for viewing angles and/or deployment angles.

SATURN IB TYPICAL LAUNCH TRAJECTORY TIME HISTORY



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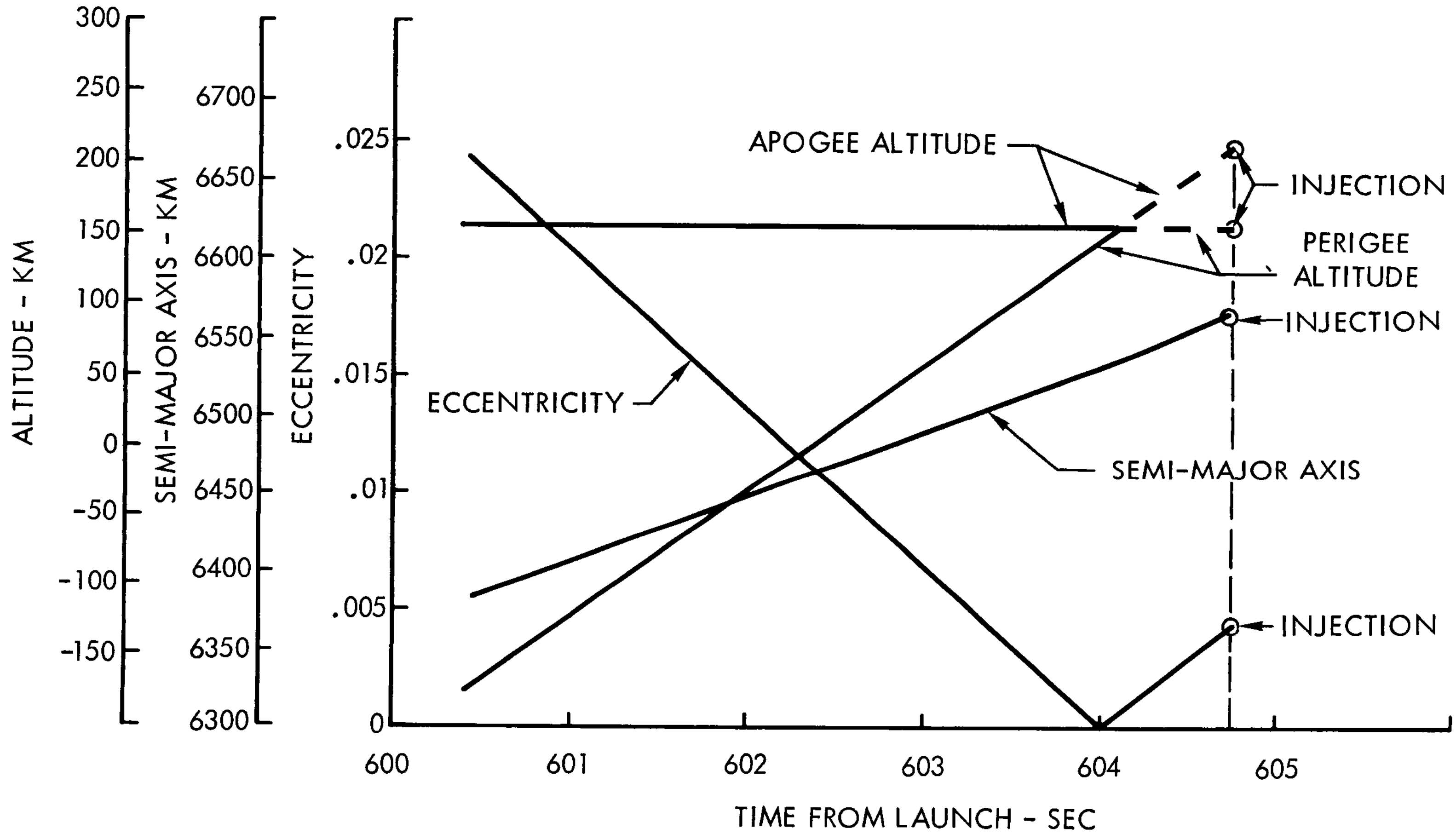
VARIATION OF ORBITAL ELEMENTS ALONG SATURN IB TYPICAL LAUNCH TRAJECTORY

The data on the facing page illustrate the variation and sensitivity characteristics of some primary orbital parameters along a typical Saturn IB launch trajectory as injection into orbit is approached. These data illustrate the orbital elements that an experiment payload would attain if it were deployed without an impulsive ΔV during the launch phase of an orbital mission. In this example, the time during which an experiment could be deployed and attain an individual orbit is limited to less than two seconds prior to the injection of the primary payload into its orbit. The extreme sensitivity of perigee altitude to time before injection indicates the limited application of deployment during the launch phase. The problem is further complicated by factors other than trajectory parameters, e.g., physical separation from the vehicle from locations that are not readily accessible for deployment prior to injection.

VARIATION OF ORBITAL ELEMENTS ALONG SATURN IB TYPICAL LAUNCH TRAJECTORY



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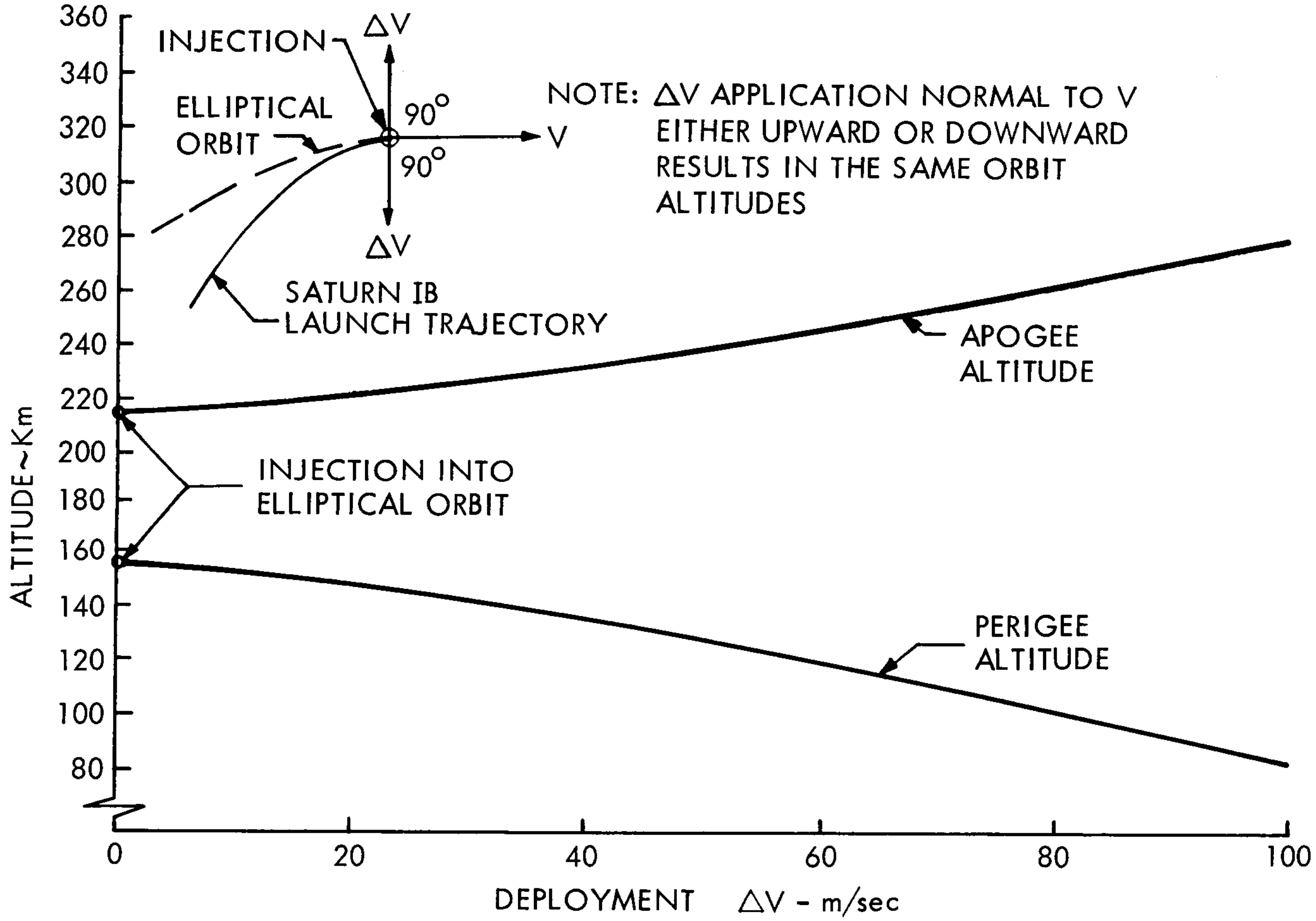
EFFECT OF EXAMPLE ΔV APPLICATION ON APOGEE AND PERIGEE ALTITUDES

AT INJECTION OF PRIMARY PAYLOAD

In the case of many experiments, the orbit achieved by the primary payload may not be compatible with the data acquisition objectives of the experiment. Obviously, a more compatible orbit can be attained by the application of propulsion to the experiment at or after injection. In this study, the application of impulsive ΔV is used to simulate propulsive deployment.

The data shown on the next page illustrate the effect on apogee and perigee altitudes of the propulsive deployment of an experiment payload at injection of the primary payload. Injection conditions are those of the typical launch trajectory shown on the previous charts. As illustrated in the diagram, the ΔV application is considered normal to the velocity vector, in the plane of the initial orbit. It is noted that, in this example of ΔV application, the perigee altitude is decreased; the apogee altitude is increased, and the resulting orbit is more eccentric.

EFFECT OF EXAMPLE ΔV APPLICATION ON APOGEE & PERIGEE ALTITUDES AT INJECTION OF PRIMARY PAYLOAD



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EFFECTS OF DEPLOYMENT ΔV COMPONENTS ON ORBITAL ELEMENTS

In most cases, deployment of an experimental payload before or at injection of the primary payload is undesirable and/or physically impossible. In particular, in the case of the payload cavities that have been defined for the example configuration for the study (Saturn IB/Apollo), deployment must occur after injection because of physical restrictions. Experimental payload deployment cannot occur until the Apollo CM-SM is separated from the LEM adapter and the adapter panels are deployed.

The chart on the facing page is an illustration of the major effects on orbital elements created by small impulsive velocities applied in three orthogonal directions after injection of the primary payload. The tangential direction lies along the velocity vector, the normal direction is perpendicular to the tangential direction and in the plane of the orbit, and the lateral direction completes the right-handed cartesian coordinate system.

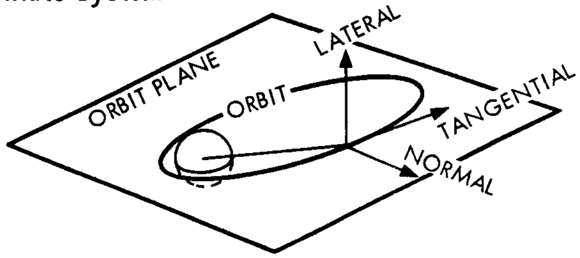
The data on the charts are based on a 155.8 by 215.8 kilometer elliptical orbit with an inclination of 30 degrees and impulsive velocity increments of less than 300 meters per second. The major effects are the following. The value of the semimajor axis of the orbit changes only as a function of variations in the tangential component of impulsive velocity and is independent of deployment time after injection. The value of eccentricity is dependent on both the tangential and the normal components of impulsive velocity. The value of eccentricity is also dependent on the position in the orbit (deployment time) and on the magnitude of the impulsive velocity. The orbital inclination change is dependent only on the lateral component of impulsive velocity. Also, the change in inclination is dependent on the deployment time after launch, and maximum change will occur at the nodes and negligible change at the position of maximum latitude.

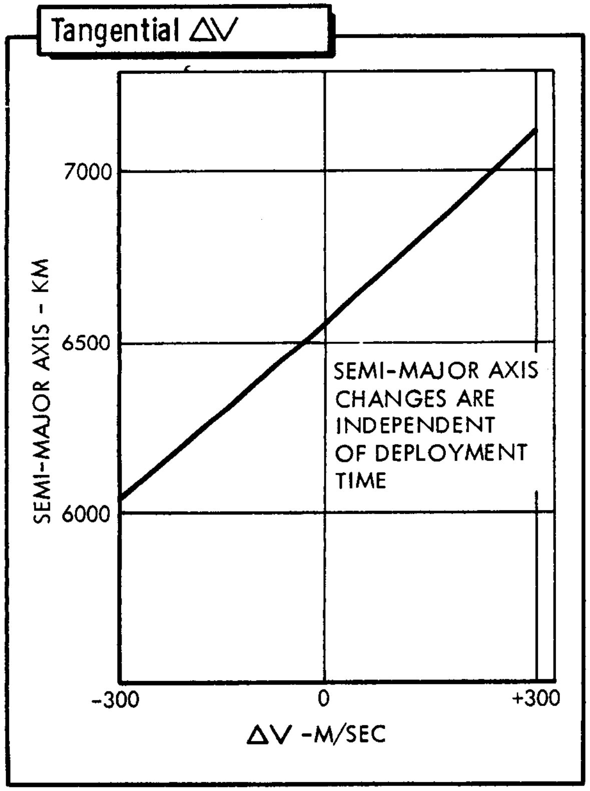
EFFECTS OF DEPLOYMENT ΔV COMPONENTS ON ORBITAL ELEMENTS

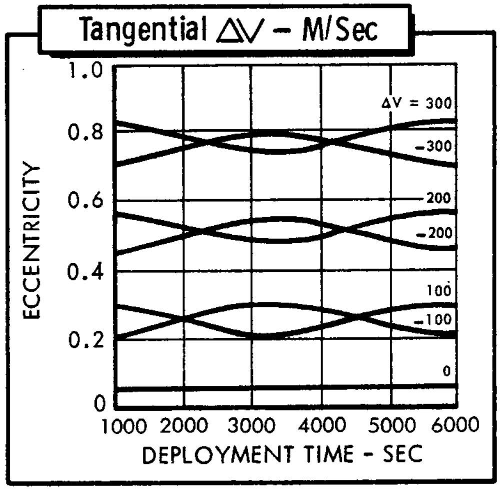
Reference Orbit:

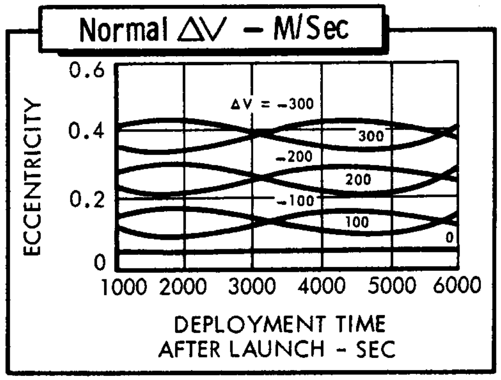
- APOGEE - 215.8 KM
- PERIGEE - 155.8 KM
- INCLINATION - 30°

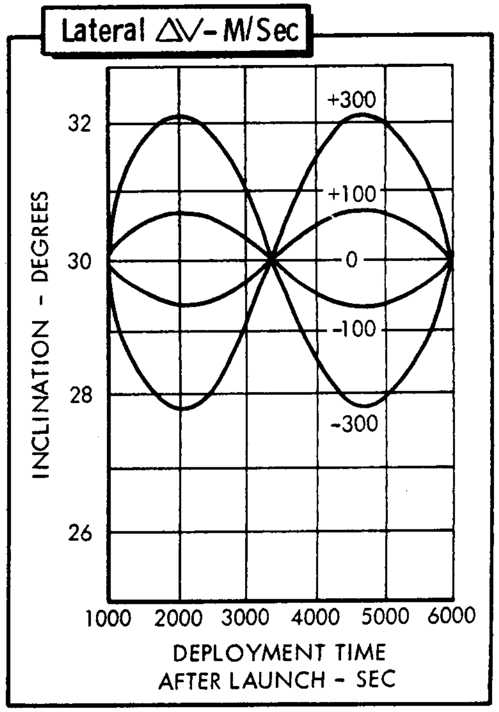
ΔV Coordinate System











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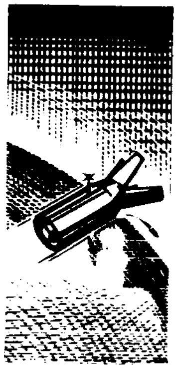
Part VII



COMPUTER PROGRAM DEVELOPMENT AND METHODOLOGY

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PROGRAM SEPTER COMPUTATION FLOW DIAGRAM

The chart on the facing page is a general flow diagram of program SEPTER (Saturn Experimental Payload Technical Evaluation and Rating), showing the major areas of analysis and flow of calculations. The major areas of analysis are described here with regard to overall program philosophy and logic. More detailed descriptions of the methodology used in the individual areas of analysis are presented in subsequent charts and discussions.

Program SEPTER philosophy and logic may be generally divided on the basis of two modes of operation and analysis: Mode I and Mode II.

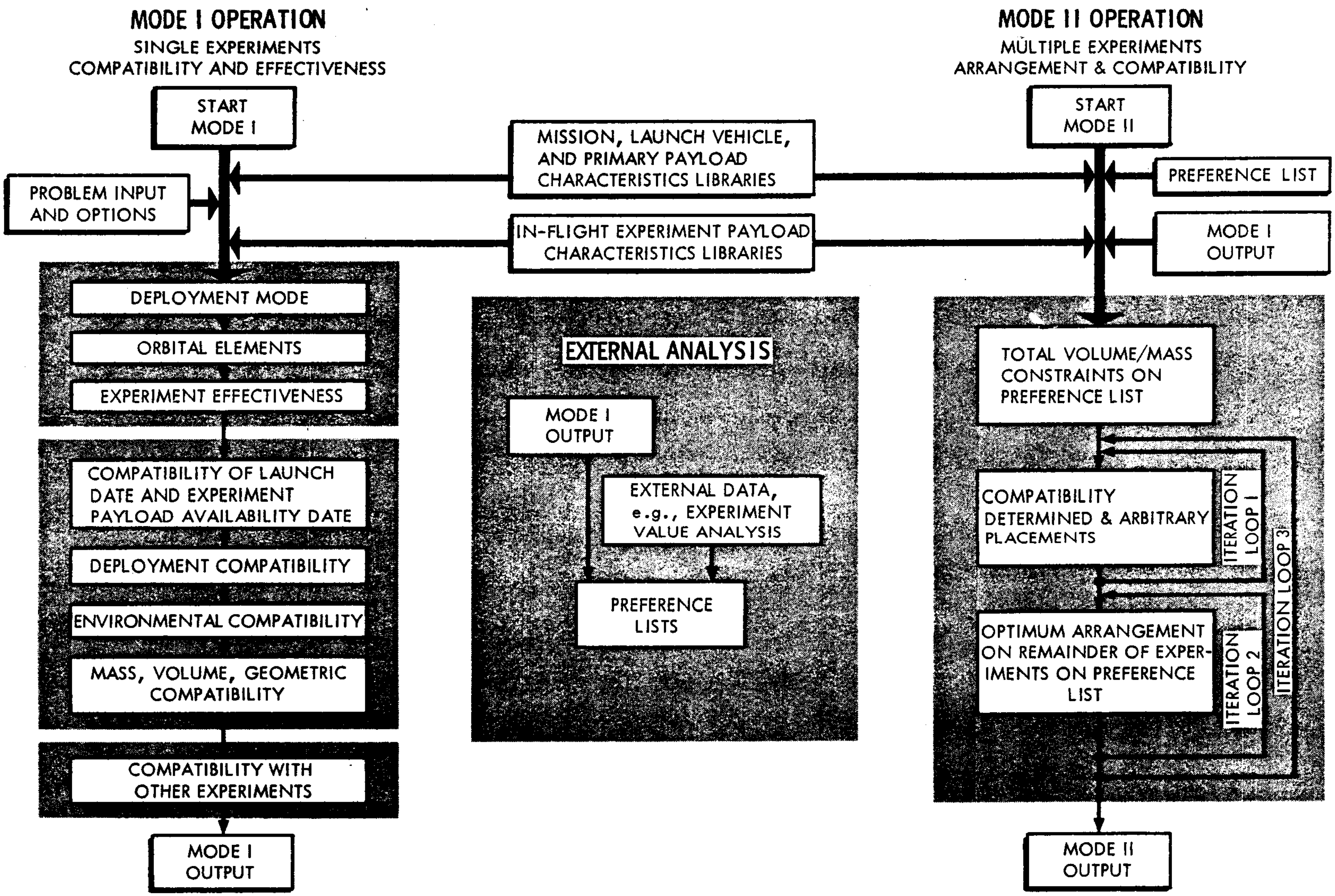
Mode I analysis consists of the determination of the compatibility and effectiveness of single experiment payloads in terms of a given mission/vehicle/primary payload combination. Compatibility criteria include only major items which are considered significant within the scope of the program philosophy.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

Preference lists are compiled outside of the program by the user. These lists are simply a listing of experiments in a preferred order of loading in a given vehicle for a given mission. Although the effectiveness and compatibility output data of Mode I are obviously provided to assist 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.

Mode II analysis consists of the arrangement of multiple experimental payloads in the vehicle so that (1) a preferred order of selection is used in the arrangement (according to the preference list), (2) no incompatibilities exist (either payload-cavity or payload-payload), (3) the payload mass capability of the mission/vehicle is not exceeded, and (4) the maximum number (within the placement policy mechanics) of experimental payloads are placed aboard the vehicle from the preference list.

The definition-type input data for SEPTER are stored and provided in the form of libraries. These libraries are of two distinct types: (1) mission/vehicle/primary payload characteristics and (2) experimental payload characteristics. Data from these libraries are used in both Mode I and Mode II analyses.

PROGRAM SEPTER COMPUTATION FLOW DIAGRAM



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DEPLOYMENT METHODOLOGY

Deployment Modes - Orbital Elements Logic

Since some experimental payloads must be separated from the vehicle for operation and others are more effective when they are ejected into orbits other than that of the mission profile, the following four modes of deployment are included in SEPTER:

1. Launch vehicle fixed. The experimental payload is not deployed and remains on the mission profile at all times.

2. Experiment equipment extension. Some piece of equipment (such as an antenna) is extended from the experimental payload, and there is no separation of the payload and launch vehicle.

3. Separation from launch vehicle. The experimental payload remains on the mission profile until deployment. At deployment, it is separated from the vehicle with negligible effects on its orbital elements. Subsequent to deployment, the orbital elements of the vehicle and ejected payload may differ as a result of perturbative forces, e.g., atmospheric drag.

4. Propulsive separation (ΔV). The experiment payload remains on the mission profile until deployment. Program input includes the specification of an impulsive velocity increment, two orientation angles (in-plane and out-of-plane), and a deployment time to effect a change in orbital elements from the mission profile.

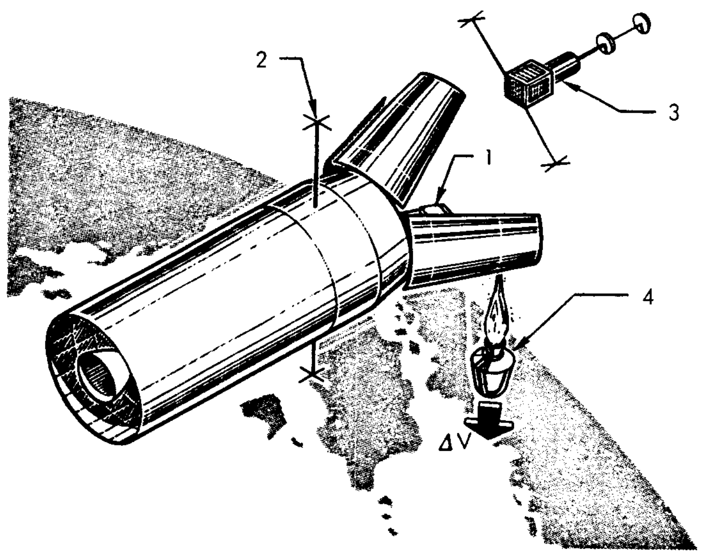
Subroutines PROFIL, DEPLOY, and ORBIT are used to calculate the mission profile parameters and orbital elements of the experimental payload at deployment.

The position and velocity of the vehicle at the deployment time are computed in subroutine PROFIL. These data are obtained from the Mission Characteristics Library. In the case of modes 1, 2, and 3, the position and velocity data are used in subroutine ORBIT to compute the initial orbital elements of the payload. When mode 4 is specified, the velocity vector is modified by use of subroutine DEPLOY before calculation of the initial orbital elements.

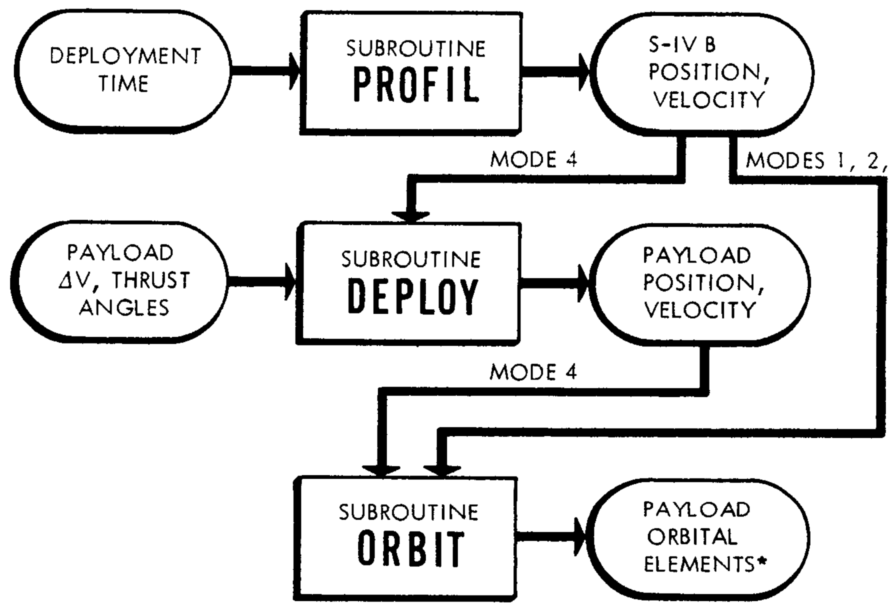
DEPLOYMENT METHODOLOGY

Deployment Modes

1. LAUNCH VEHICLE FIXED (MISSION PROFILE)
2. EXPERIMENT EQUIPMENT EXTENSION (MISSION PROFILE)
3. SEPARATION FROM LAUNCH VEHICLE (M/C_DA VARIATION)
4. PROPULSIVE SEPARATION (ΔV)



Orbital Elements Logic



* a - SEMI-MAJOR AXIS
e - ECCENTRICITY
i - INCLINATION
ω - ARGUMENT OF PERIGEE
λ_n - LONGITUDE OF NODAL PASSAGE
t_p - TIME OF PERIGEE PASSAGE
h_a - APOGEE ALTITUDE
h_p - PERIGEE ALTITUDE
η - TRUE ANOMALY
t_n - TIME OF NODAL PASSAGE
P - PERIOD

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DEPLOYMENT METHODOLOGY

Coordinate Systems

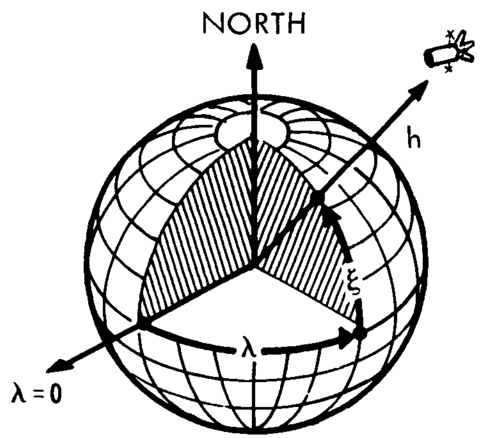
The coordinate systems used in the deployment methodology are shown on the chart on the facing page. Position is specified in terms of altitude, longitude, and latitude with respect to a sperical, rotating Earth. Velocity is defined by inertial speed, flight path angle, and azimuth.

The coordinate system in which the ΔV is defined for propulsive separation has an axis tangential to the velocity vector at the time of deployment, an axis normal to the velocity vector in the orbit plane of the vehicle, and an axis normal to the orbit plane of the vehicle. The in-plane thrust angle, θ, is measured from the velocity direction to the projection of the ΔV vector on the orbital plane. The out-of-plane angle, ϕ, is measured from the orbital plane to the ΔV vector.

DEPLOYMENT METHODOLOGY COORDINATE SYSTEMS

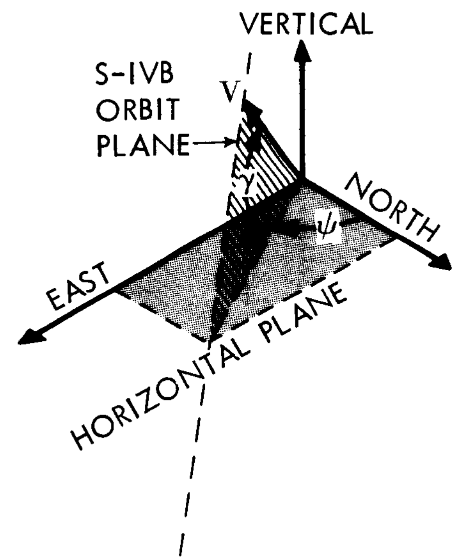
Position:

- LATITUDE, ξ
- LONGITUDE, λ
- ALTITUDE, h



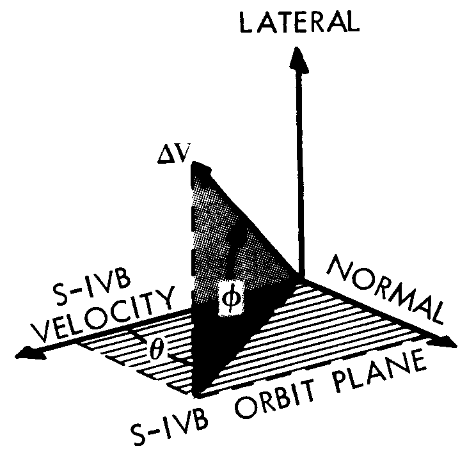
Velocity:

- VELOCITY MAGNITUDE, V
- FLIGHT PATH ANGLE, γ
- AZIMUTH ANGLE, ψ



Propulsive Separation:

- VELOCITY INCREMENT, ΔV
- THRUST ANGLES, θ & ϕ



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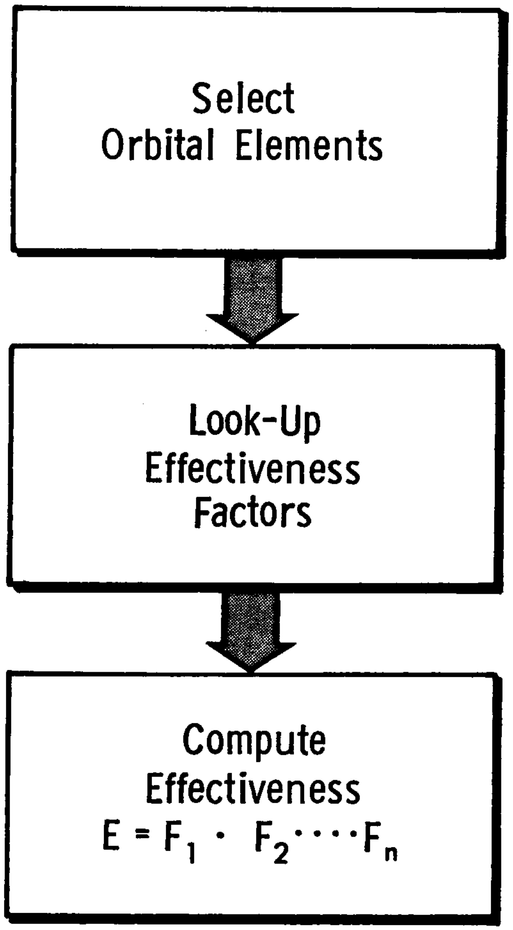
EXPERIMENT/MISSION EFFECTIVENESS

As discussed under "In-Flight Experimental Payload Effectiveness Analysis," the effectiveness of each experiment is analyzed as a function of the initial orbital elements of the deployment orbit. The relationships between experiment effectiveness and the initial orbital elements are established during the external experiment effectiveness analysis. These effectiveness factors are loaded into the effectiveness segment of the Experiment Characteristics Library in tables of one- and two-dimensional arrays.

From the input controls, the program selects the required initial orbital element(s) for each table. Effectiveness factors are then obtained from each array by use of a table "look-up" procedure and are multiplied together to obtain the final value of experiment effectiveness.

EXPERIMENT/MISSION EFFECTIVENESS



- Required orbital elements for effectiveness factor tables are selected according to input controls
- Effectiveness factors are obtaind from one- and two-dimensional library arrays by table look-up procedure
- Effectiveness factors are multiplied to obtain effectiveness

EXPERIMENTAL PAYLOAD-MISSION/VEHICLE COMPATIBILITY

Two basic ground rules are applied to checks of experimental payload-mission/vehicle compatibility:

1. The experimental payload must tolerate all mission/vehicle constraints.
2. The experimental payload cannot significantly affect primary mission/vehicle performance.

The criteria for compatibility are divided into five general categories, as listed on the facing chart. For an experimental payload to be mission/vehicle compatible, it must satisfy the GO-condition in each category:

1. The experiment must be available prior to the mission launch date.
2. If the experimental payload is deployed, it must be placed in a cavity from which the required deployment mode is possible (e.g., thrust directions, deployment time, etc.)
3. The experimental payload must be able to tolerate the ambient environmental levels without adversely affecting the vehicle or the primary payload. Thermal and acoustics/vibration environmental compatibility aspects are considered in detail in subsequent charts and discussions. Electromagnetic and radiation environmental compatibility aspects are currently being investigated for applicability. The remaining environmental criteria were considered but eliminated from this study. The basis for elimination of these criteria is the assumption that any experimental payloads considered in the present analysis would be designed to survive normal launch pad, boost, and orbit environments not peculiar to a specific vehicle.
4. The experimental payload mass must be less than or equal to the mass attachment limit of the cavity in which it is placed.
5. Volume/geometry compatibility must be established. These criteria are considered in detail in subsequent charts and discussions.

EXPERIMENTAL PAYLOAD - MISSION/VEHICLE COMPATIBILITY

Compatibility Critieria Considered:

- EXPERIMENT AVAILABILITY DATE/LAUNCH DATE
- DEPLOYMENT
  - MODE
  - DIRECTION
  - TIME
- ENVIRONMENTAL
  - THERMAL
  - ACOUSTICS/VIBRATION
  - ELECTROMAGNETIC
  - RADIATION
  - ACCELERATION/SHOCK *
  - HUMIDITY *
  - PRESSURE/VACUUM *
  - CONTAMINATION (DUST, FUNGUS, GASES, SALT SPRAY) *
- MASS ATTACHMENT
- VOLUME/GEOMETRY
  - SIZE
  - SHAPE
  - ORIENTATION

*NONAPPLICABLE CRITERIA

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THERMAL ENVIRONMENT COMPATIBILITY

Thermal environment compatibility is determined by comparing certain parameters pertaining to a specific cavity with the corresponding parameters associated with a particular experiment (Mode I) or combination of experiments (Mode II) The basic guideline is that to be thermally compatible the experiment must tolerate the cavity thermal environment without affecting the primary payload. This comparison of parameters will yield go or no-go decisions on the basis of the compatibility of the given experiment(s) and cavity with respect to the individual parameters. Thermal compatibility will require a full set of go decisions.

Three parameters will be compared during Mode I operation: (1) the time-space averaged sink temperature (2) the heat dissipation rate, and (3) the total heat dissipation. In Mode II, two parameters will be compared: (1) the heat dissipation rate, and (2) the total heat dissipation. Thermal compatibility will be based on separate comparisons for each mission phase--prelaunch, launch, and orbit. In the case of experiments in which packages are ejected from the vehicle into a separate orbit, the compatibility checks will be made only during the mission phases where the experiment is aboard the vehicle.

In certain situations, the rate of heat dissipation and total heat dissipation should not be treated as mutually exclusive criteria. If the heat dissipation rate exceeds the allowable rate, but only for a short time interval, the total heat dissipation may be well below the tolerable level and may be a more meaningful basis of comparison. Thus, the capability to override the heat dissipation rate comparison will be included in the program logic.

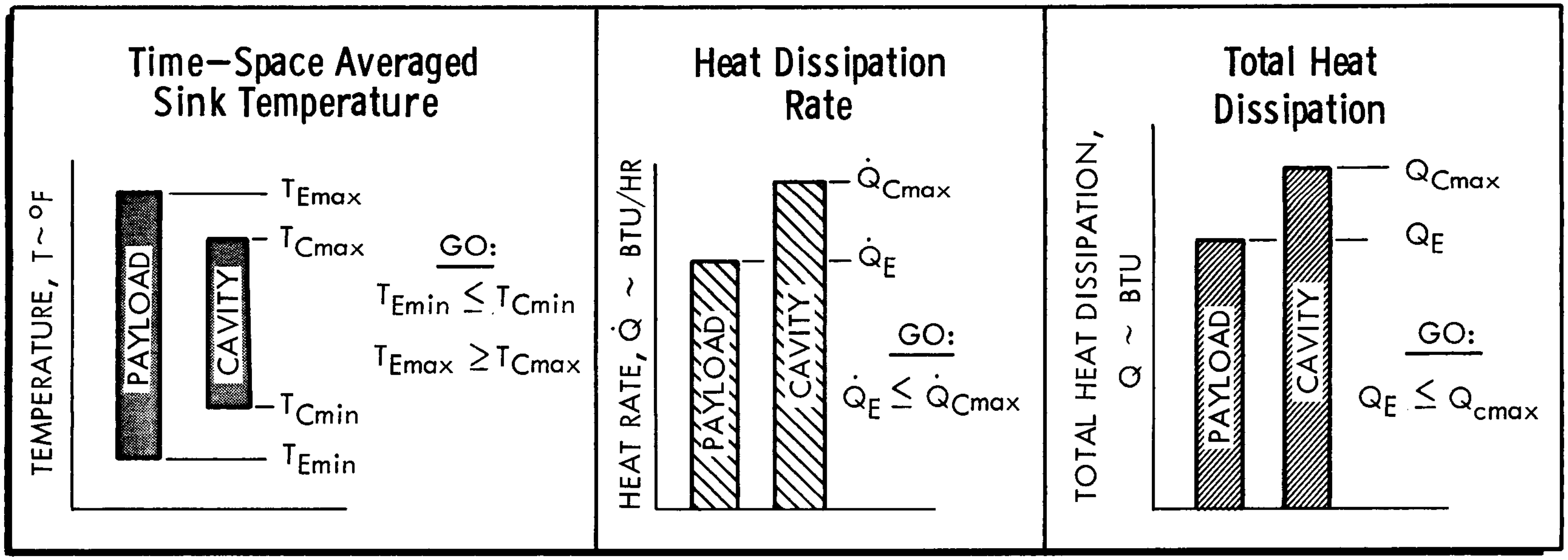
THERMAL ENVIRONMENT COMPATIBILITY

Guideline:

- EXPERIMENT MUST TOLERATE CAVITY THERMAL ENVIRONMENT WITHOUT AFFECTING THE PRIMARY PAYLOAD AND VEHICLE

Approach:

- DEFINE CAVITY THERMAL ENVIRONMENT FOR EACH MISSION PHASE (e.g., PRE-LAUNCH, LAUNCH, ORBIT)
- DEFINE LIMITS OF ACCEPTANCE FOR EXPERIMENT PAYLOAD THERMAL PARAMETERS
- DEFINE ANALYTICAL METHODOLOGY FOR THE DETERMINATION OF THERMAL COMPATIBILITY



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ACOUSTICS AND VIBRATIONS ENVIRONMENT COMPATIBILITY

Acoustic and vibrational compatibility checks consist in determining whether a payload can survive the environmental criteria of the vehicle. If a possible failure is indicated, a mass penalty is computed and added to the mass of the payload in order to correct the deficiency rather than disqualify the payload completely.

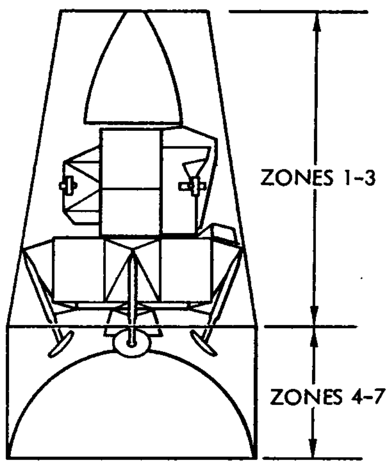
The parameters used in the compatibility checks are shown on the facing page. The ambient levels of these parameters are given for vehicle zones. The corresponding tolerances are given for each experimental payload. The computer methodology used to correct a deficiency (NO-GO condition) to a GO condition by the addition of a mass penalty is illustrated by the data and equations given in the lower half of the facing chart.

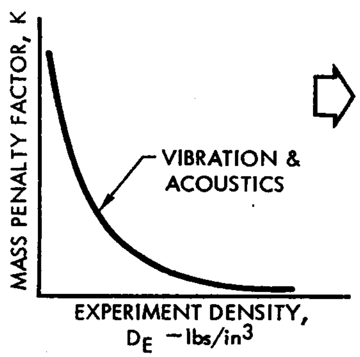
The analysis of these compatibility checks is based on the following assumptions:

- If a payload can survive the launch, it can survive all other mission phases.
- The payload does not operate during launch.
- Acoustic and vibration tolerance deficiencies can be corrected by the addition of mass (in the form of heavier material, isolation mountings, stiffeners, or dampers) to the payload.
- The mass required is proportional to payload density, percent deficiency, and the mass of susceptible components.
- The correction of an acoustic tolerance deficiency does not correct a vibration tolerance deficiency and vice-versa, because they apply to different frequency ranges.

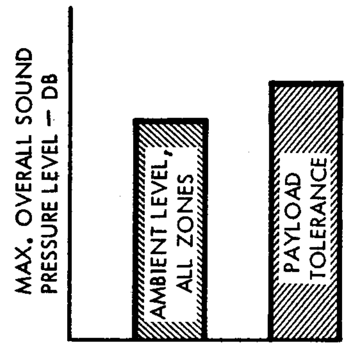
ACOUSTICS AND VIBRATION ENVIRONMENT COMPATIBILITY

Data Given For Vehicle Zones:





Acoustics



GO CONDITIONS:

PAYLOAD TOLERANCE MUST BE GREATER THAN AMBIENT LEVEL.

NO-GO CORRECTION:

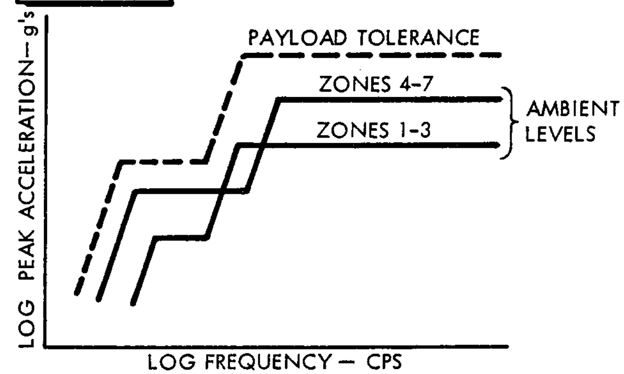
$$\left(\frac{\text{AMBIENT LEVEL-PAYLOAD TOLERANCE}}{\text{PAYLOAD TOLERANCE - 140 DB}}\right)$$

X (ACOUSTICS MASS PENALTY FACTOR, K)

X (MASS OF NOISE SUSCEPTIBLE COMPONENTS)

= (MASS PENALTY FOR GO CONDITION)

Vibration



GO CONDITIONS:

PAYLOAD TOLERANCE MUST BE GREATER THAN AMBIENT LEVELS AT ALL FREQUENCIES.

NO-GO CORRECTION:

$$\left(\frac{\text{AMBIENT LEVEL-PAYLOAD TOLERANCE}}{\text{PAYLOAD TOLERANCE}}\right)_{\text{@ MAX. VIOLATION}}$$

X (VIBRATION MASS PENALTY FACTOR, K)

X (MASS OF VIBRATION SUSCEPTIBLE COMPONENTS)

= (MASS PENALTY FOR GO CONDITION)

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VOLUME/GEOMETRY COMPATIBILITY

The purpose of the volume/geometry analysis is to determine whether a given experiment package can be physically contained within a given payload cavity.

Rather than attempting to formulate an "exact" approach, which would require extremely complex computer logic and prohibitive data storage capabilities, a less direct method is used. The actual sizes and shapes of the cavities will not be represented within the computer program; instead, their capacities for several geometrical solids are stored in tabular form. The experimental payloads, in turn, are treated 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 package is composed of the sum of the volumes of the components multiplied by a packaging factor.

These two concepts allow for the handling of experimental payloads which are (1) in the "off-the-shelf" or final design stages, and (2) those in the conceptual design stage, amenable to some rearranging of the components of the entire package.

The cavities are divided into two categories, rectilinear or tapered, according to the form of their capacities data. There are very simple methods available for handling the rectilinear capacities in a computer program, but these methods are not applicable to the tapered cavity capacities. A reasonably simple and nearly exact technique is being utilized for handling both types of cavities. The method is general, efficient, and accurate. The subsequent charts and discussions illustrate this method.

VOLUME/GEOMETRY COMPATIBILITY

Methodology Approach:

- TWO TYPES OF REPRESENTATIONS FOR EXPERIMENT PACKAGES
  - FIXED GEOMETRY - OFF-THE-SHELF DESIGN, STANDARD SHAPE ENVELOPE OF ENTIRE PACKAGE
  - AMORPHOUS GEOMETRY - CONCEPTUAL DESIGN, STANDARD SHAPE ENVELOPE OF CRITICAL COMPONENT ONLY
- EXACT VOLUME/GEOMETRIC REPRESENTATION FOR CAVITIES
  - RECTILINEAR (COMPOSED OF ONE OR MORE RECTANGULAR PARALLELEPIPEDS INTERSECTING AT RIGHT ANGLES)
  - TAPERED (COMPOSED OF TAPERED OR CURVED SIDES)
- ANALYTICAL COMPARISON OF EXPERIMENT PACKAGE-CAVITY DIMENSIONS AND VOLUME

Methodology Features:

- PRACTICAL AND REASONABLY ACCURATE APPROACH TO PROBLEM
  - STRAIGHT FORWARD EXTERNAL PROCEDURES FOR EXPERIMENT AND CAVITY DEFINITIONS
  - REASONABLE PROGRAMMING AND DATA STORAGE REQUIREMENTS
  - REASONABLE COMPUTATION TIME REQUIREMENTS

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GEOMETRIC CAPACITIES COMPATIBILITY - 1
Sphere and Cylinder Capacities

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 purpose of the geometric capacities compatibility analysis is to determine whether a sphere of given diameter, a cylinder of given diameter and length with its axis parallel to the vertical, radial, or lateral axes, or a rectangular parallelepiped of given vertical, radial, and lateral dimensions can be contained within a given cavity.

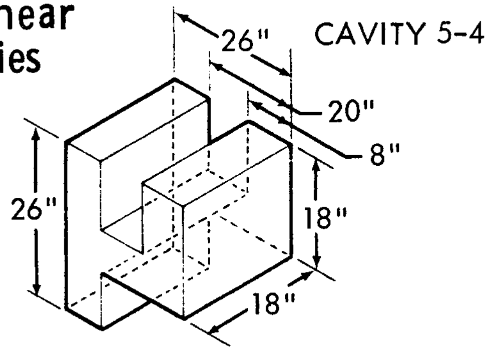
The chart on the facing page shows a rectilinear and a tapered cavity and their sphere and cylinder capacities. For any cavity, there is only one maximum diameter sphere which it will contain. There is obviously no necessity to specify an orientation in the case of a sphere. Therefore, the technique used to determine geometric compatibility in the case of a spherical payload consists in checking the diameter of the spherical payload to determine if it is less than or equal to the maximum diameter sphere which the cavity will contain. This same technique could be used on any standard shape the size of which can be specified with one dimension, e.g., cubes and hemispheres.

The cylinder capacity curves represent the maximum diameters of cylinders of given length and orientation which can be contained in the cavity. A difference between rectilinear and tapered cavities is now evident. The rectilinear cavity cylinder capacity curves consist in constant diameter steps. The curves in the example 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, and these represent the simplest form in which the cylinder capacities for tapered cavities may be represented.

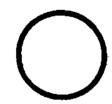
The technique used to determine the geometric capacities compatibility for a cylindrical payload is to interpolate, on these curves, for the maximum possible diameter corresponding to the length of the payload. The payload diameter must be less than or equal to the maximum possible diameter for compatibility. If the orientation is specified, only one interpolation is required; but if the orientation is not specified, as many as three interpolations may be necessary to determine whether a fit exists. This technique can also be used on other standard shapes whose size can be specified with two dimensions, such as right circular cones and square-base pyramids.

GEOMETRIC CAPACITIES COMPATIBILITY - 1

Rectilinear Cavities

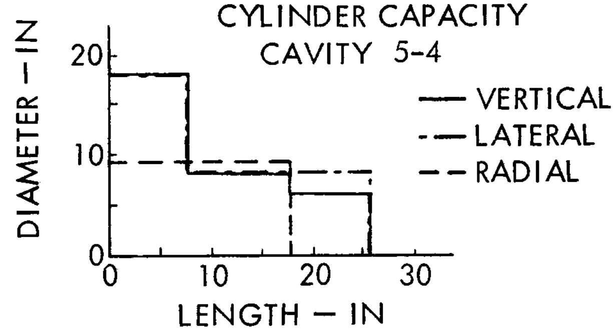


Sphere Capacities

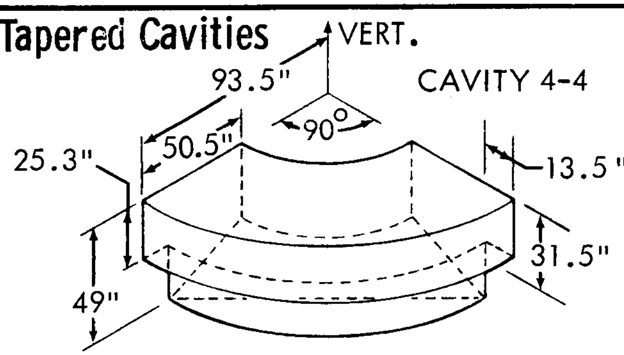


MAXIMUM
DIAMETER
= 9.3"

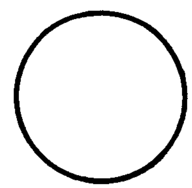
Cylinder Capacities



Tapered Cavities

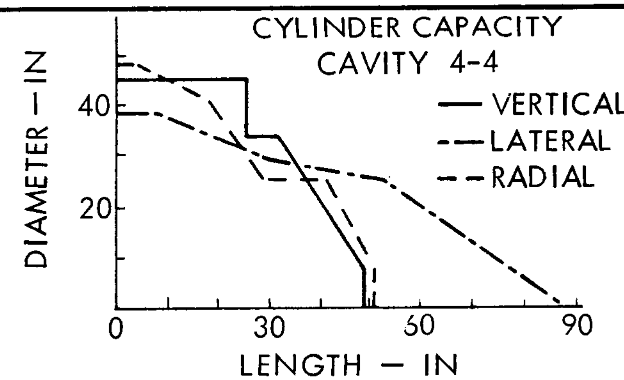


Sphere Capacities



MAXIMUM
DIAMETER
= 38.5"

Cylinder Capacities



Experiment - Cavity Geometric Compatibility Methodology

Compare with Maximum Allowable Diameter

Use Linear Interpolation to Determine Maximum Allowable Diameter for Given Length

GEOMETRIC CAPACITIES COMPATIBILITY - 2

Rectangular Parallelepiped Capacities

Since three dimensions are required to specify the size of a rectangular parallelepiped, the capacities for this type of payload are represented by surfaces. The rectangular parallelepiped capacity surfaces for the cavities on the preceding chart are shown on the facing page. 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 both contour and isometric form for clarity.

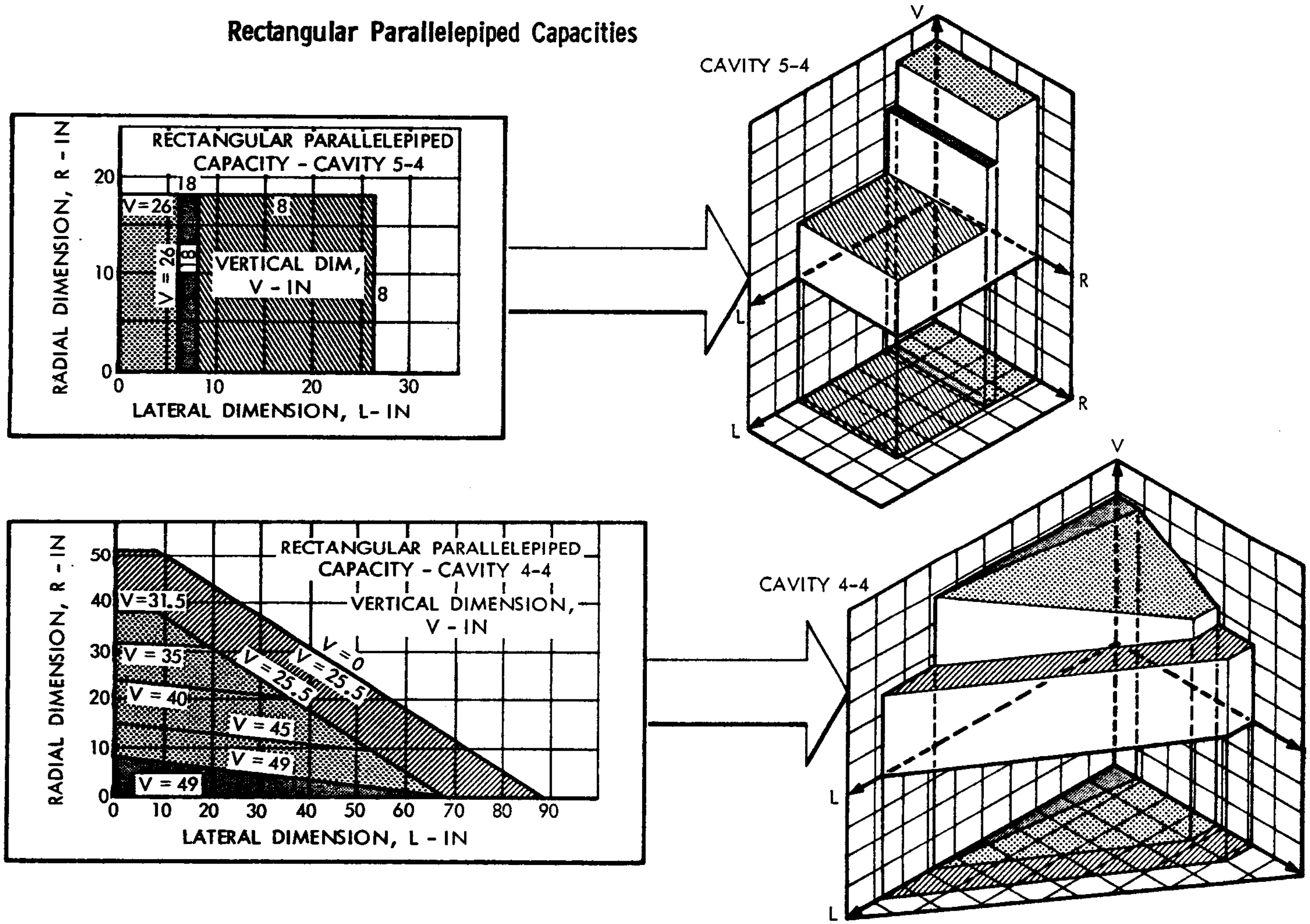
Since each point on this surface corresponds to a rectangular parallelepiped and different orientations are given merely by its dimensions taken in different orders, all orientations (parallel to the L-R-V axes) are included on a single surface, which is generally discontinuous, for each cavity.

Again, 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. These may also be represented in a simpler way. The surface on the chart 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 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, as shown in isometric form, is the original cavity distorted --beyond recognition in some cases --in such a way that its shape is simplified, its volume is decreased, but its capacity for rectangular parallelepipeds is maintained exactly.

GEOMETRIC CAPACITIES COMPATIBILITY - 2

Rectangular Parallelepiped Capacities



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GEOMETRIC CAPACITIES COMPATIBILITY - 3
Rectangular Parallelepiped Methodology

There are six possible orientations of a given rectangular parallelepiped within a cavity in which its sides are parallel to the axes of the cavity. The example shown on the chart 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 other cases, there is a desirable 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 shown in the following table apply to the example on the facing page. 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.

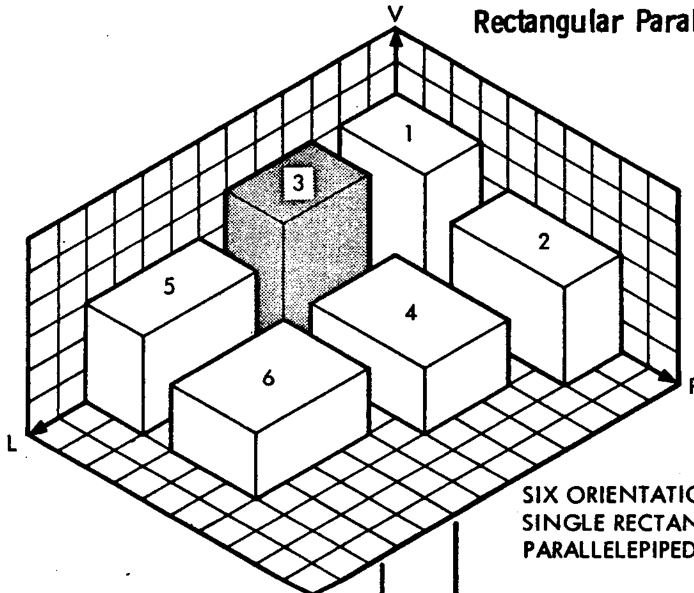
	Payload Axis		
	20"	30"	40"
L	1,2	3,4	5,6
R	3,5	1,6	2,4
V	4,6	2,5	1,3

Once an orientation is selected, it is determined whether it can be contained in the cavity in the following manner. The rectangular parallelepiped capacity surface is divided into plane quadrangles, which provide a very good approximation. 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 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 dimension corresponding to those lateral and radial dimensions. When this dimension is found, a comparison is made with the actual vertical dimension of the 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.

This technique described can be extended for use on other standard shapes the size of which can be specified with three dimensions, such as truncated right circular cones and rectangular-base pyramids.

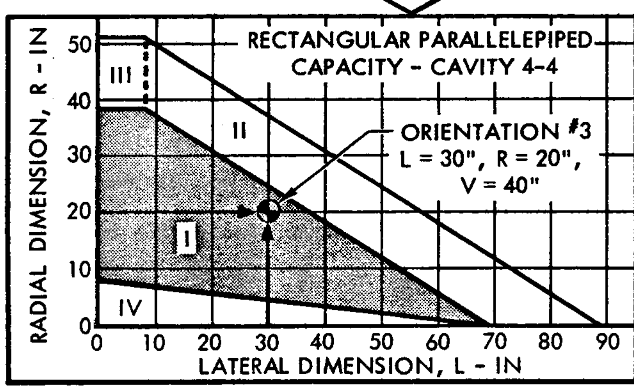
GEOMETRIC CAPACITIES COMPATIBILITY - 3

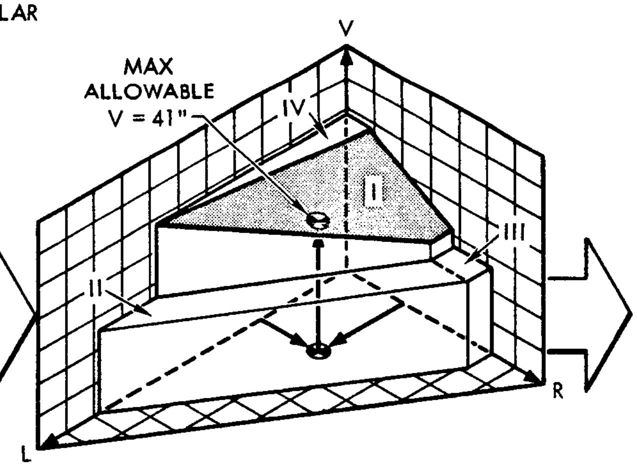
Rectangular Parallelepiped Methodology



SIX ORIENTATIONS OF A SINGLE RECTANGULAR PARALLELEPIPED

- SELECT AN ORIENTATION.
- USING LATERAL AND RADIAL DIMENSIONS OF EXPERIMENT, LOCATE PROPER QUADRANGLE OF RECTANGULAR PARALLELEPIPED CAPACITY SURFACE PROJECTED ON L-R PLANE.
- INTERPOLATE FOR MAXIMUM ALLOWABLE VERTICAL DIMENSION.
- COMPARE WITH VERTICAL DIMENSION OF EXPERIMENT. GO CONDITION: $V_{max} \geq V$





FOR $L = 30"$, $R = 20"$

$V_{max} = 41"$

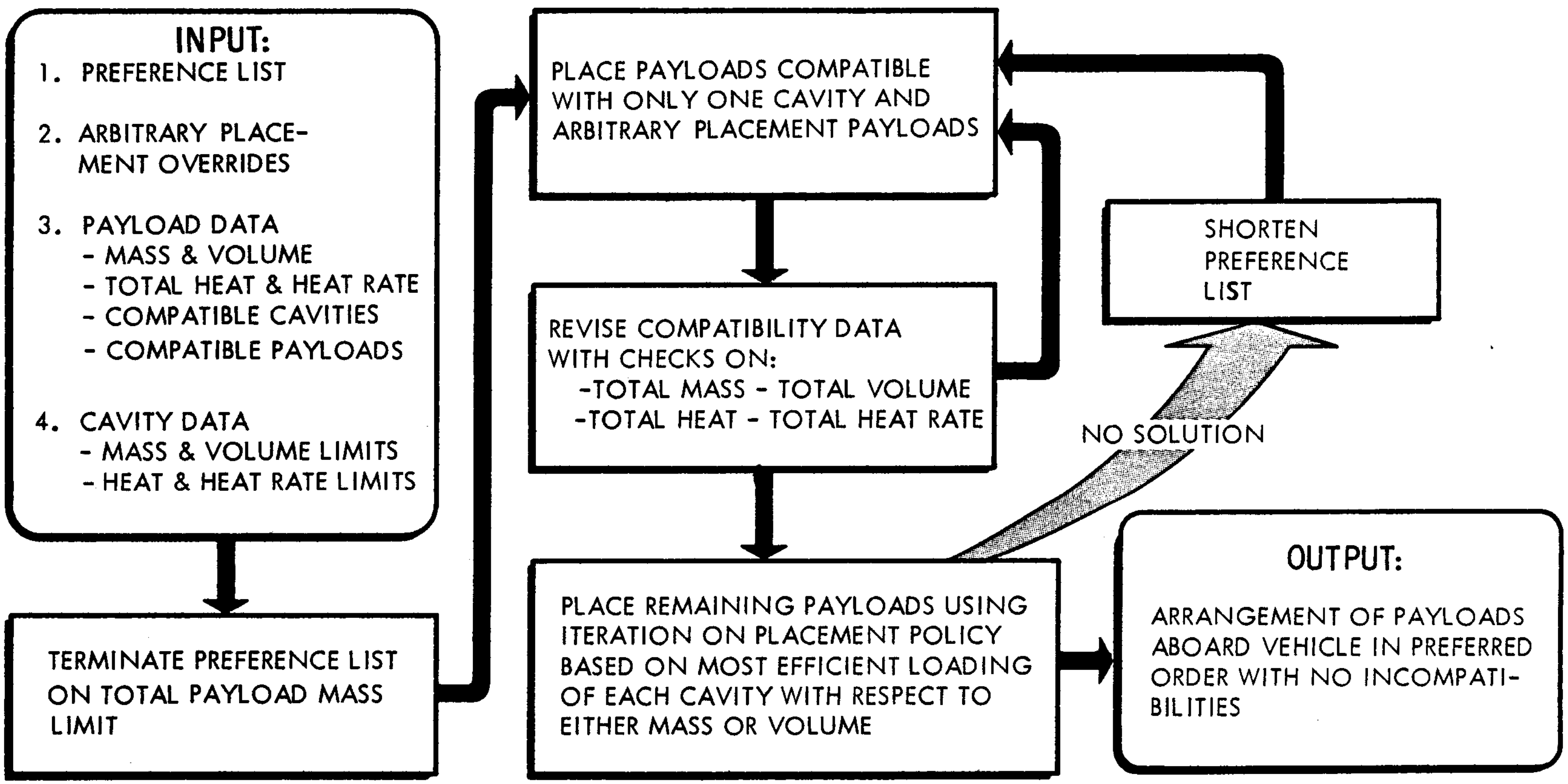
$V_{max} > V \Rightarrow$ GO

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. This 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. In view of this, an attempt to apply a true optimization process to the problem appears to involve a degree of effort not justified by the results desired for this study.

In the chart on the facing page, 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. The first iteration consists in the placement of any payloads which have no choice of location. These placements may be arbitrary (override option) or compatibility-determined. It is necessary to iterate this procedure 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 second iteration is initiated. This computational block consists in an arbitrary set of rules for placement (placement policy) which is applied cavity-by-cavity in search of possible arrangements. Iteration is required here because the placement policy is arbitrary and is influenced by the order in which the cavities are taken. If no solution is found, the third iteration is required because the preference list must be shortened, and the entire procedure is repeated.

MULTIPLE PAYLOAD ARRANGEMENT LOGIC



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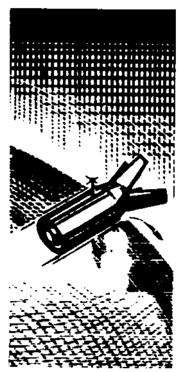
Part VIII



SUMMARY

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SUMMARY - 1

Tasks Accomplished During The First Half of Study

- DESIGN OF IN-FLIGHT EXPERIMENT PACKAGES
  - SELECTION AND DEFINITION OF REPRESENTATIVE EXPERIMENTS
  - DEFINITION OF ANCILLARY SUBSYSTEMS
    - EXPERIMENT SENSORS
    - ATTITUDE CONTROL
    - DATA AUTOMATION
    - COMMUNICATIONS
    - POWER SUPPLY
    - ENVIRONMENTAL CONTROL
  - DEFINITION OF PACKAGE CONFIGURATIONS
    - CONCEPTUAL DESIGN DRAWINGS AND MASS SUMMARIES
    - OPERATIONAL REQUIREMENTS
    - ENVIRONMENTAL REQUIREMENTS
    - SHAPE AND VOLUME REQUIREMENTS
- ANALYSIS OF IN-FLIGHT EXPERIMENTAL PAYLOAD EFFECTIVENESS
  - PREPARATION OF SUMMARY TABLE OF FLIGHT REGIMES AND EXPERIMENT OPERATIONS TO BE USED IN DETERMINING THE ORBITAL ELEMENTS INFLUENCING EXPERIMENT EFFECTIVENESS
  - ANALYSIS AND DETERMINATION OF EXPERIMENT EFFECTIVENESS AS A FUNCTION OF INITIAL ORBITAL ELEMENTS FOR AN INITIAL GROUP OF EXPERIMENTS

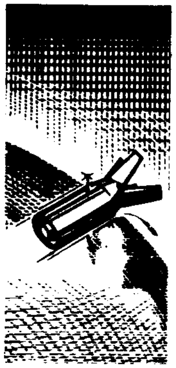
SUMMARY - 2

- DETERMINATION OF IN-FLIGHT EXPERIMENT CAVITY LOCATIONS
  - SELECTION AND IDENTIFICATION OF CAVITIES
  - DESCRIPTION OF CAVITIES (WITH EXCEPTION OF SIVB FORWARD SKIRT COLD PLATE AREA)
    - GEOMETRICAL REPRESENTATIONS
    - STANDARD SHAPE, VOLUME, AND MASS CAPACITIES
    - THERMAL ENVIRONMENTS
    - ACOUSTIC AND VIBRATION ENVIRONMENTS
    - OPERATIONAL CAPABILITIES
- DEFINITION AND ANALYSIS OF MISSION CHARACTERISTICS
  - DEFINITION OF SATURN IB/APOLLO MISSION TYPES
    - EARTH ORBITAL
    - SUBORBITAL
  - ANALYSIS OF PERTINENT MISSION CHARACTERISTICS OF INDIVIDUAL SATURN MISSIONS
    - ANALYSIS OF TIME HISTORIES OF TRAJECTORY PARAMETERS
    - ANALYSIS OF SEQUENCE OF EVENTS
    - DETERMINATION OF DEPLOYMENT OPPORTUNITIES/CONSTRAINTS
    - DETERMINATION OF VARIATION OF ORBITAL ELEMENTS ALONG LAUNCH TRAJECTORY
    - DETERMINATION OF EFFECT OF OPTIONAL ΔV APPLICATION AT EXPERIMENT DEPLOYMENT

SUMMARY - 3

- COMPUTER PROGRAM DEVELOPMENT AND METHODOLOGY
  - PREPARATION OF COMPUTATION FLOW DIAGRAM
  - CODING OF SUBROUTINES FOR THE DETERMINATION OF ORBITAL ELEMENTS FOR MODES OF PAYLOAD DEPLOYMENT TO BE CONSIDERED
  - PREPARATION OF SUMMARY TABLES OF VOLUME AND GEOMETRIC CHARACTERISTICS OF EXPERIMENTAL PAYLOADS AND CAVITY LOCATIONS FOR INCORPORATION INTO COMPUTER PROGRAM LIBRARIES
  - ESTABLISHMENT OF THE COMPUTER METHODOLOGY FOR THE DETERMINATION OF EXPERIMENT PAYLOAD/CAVITY LOCATION COMPATIBILITY CRITERIA
    - ENVIRONMENTAL: THERMAL, ACOUSTIC, AND VIBRATION
    - VOLUME/GEOMETRY
  - ESTABLISHMENT OF THE BASIC COMPUTER METHODOLOGY FOR DETERMINATION OF IN-FLIGHT EXPERIMENT EFFECTIVENESS
  - ESTABLISHMENT OF THE BASIC COMPUTER METHODOLOGY FOR MULTIPLE PAYLOAD ARRANGEMENT COMPATIBILITY ANALYSIS.

Part IX



FUTURE STUDY TASKS

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FUTURE STUDY TASKS - 1

Tasks to be Accomplished During Second Half of Study

- DESIGN OF IN-FLIGHT EXPERIMENT PACKAGES
  - ESTABLISHMENT OF ELECTROMAGNETIC ENVIRONMENTAL REQUIREMENTS
  - PERFORMANCE OF RELIABILITY ANALYSIS
  - ESTABLISHMENT OF DEVELOPMENT TIME SCHEDULE
  - PERFORMANCE OF COST ANALYSIS
  - SYNTHESIS OF SUBSYSTEM CHARACTERISTICS FOR ARBITRARY EXPERIMENTS
- ANALYSIS OF IN-FLIGHT EXPERIMENTAL PAYLOAD EFFECTIVENESS
  - COMPLETION OF DETERMINATION OF ORBITAL ELEMENTS WHICH INFLUENCE EXPERIMENT EFFECTIVENESS
  - ANALYSIS AND DETERMINATION OF EXPERIMENT EFFECTIVENESS AS A FUNCTION OF INITIAL ORBITAL ELEMENTS FOR OTHER EXPERIMENT TYPES
- DETERMINATION OF IN-FLIGHT EXPERIMENT CAVITY LOCATIONS
  - DESCRIPTION OF ZONE 6 CAVITIES (SIVB FORWARD SKIRT)
- COMPUTER PROGRAM DEVELOPMENT AND METHODOLOGY
  - PREPARATION OF COMPUTER PROGRAM LIBRARIES
    - MISSION/VEHICLE/PRIMARY PAYLOAD CHARACTERISTICS
    - EXPERIMENTS CHARACTERISTICS

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FUTURE STUDY TASKS - 2

- CODING OF SUBROUTINES FOR THE DETERMINATION OF EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY CRITERIA
  - ENVIRONMENTAL - THERMAL, ACOUSTIC, VIBRATION AND OTHERS AS FOUND PRACTICAL
  - VOLUME/GEOMETRY
- REFINEMENT AND CODING OF THE COMPUTER METHODOLOGY FOR DETERMINATION OF IN-FLIGHT EXPERIMENT EFFECTIVENESS
- REFINEMENT AND CODING OF THE COMPUTER METHODOLOGY FOR MULTIPLE PAYLOAD ARRANGEMENT COMPATIBILITY ANALYSIS
- ASSEMBLY AND CHECKOUT OF COMPUTER PROGRAM SEPTER
- DEVELOPMENT AND CODING OF SUPPORT PROGRAM(S) FOR EXPERIMENT/MISSION EFFECTIVENESS
- CODING OF SUPPORT PROGRAM FOR ARBITRARY EXPERIMENT SYNTHESIS