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FINAL REPORT

ORBITING EXPERIMENT FOR STUDY OF EXTENDED WEIGHTLESSNESS

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SUMMARY

A study was performed to arrive at a preliminary design, mission analysis, and development plan for a system to orbit two 13-lb female Rhesus monkeys for a period of six months to one year for the purpose of determining the psychological and physiological effects of extended weightlessness. The experiment-carrying spacecraft is planned to be placed in orbit by a vehicle of the Apollo Application Program (AAP). Retrieval and recovery of the animals is to be effected by the command module of a subsequent AAP flight.

Candidate AAP vehicle configurations were reviewed and a preliminary mission analysis was performed. As a result of these tasks, it was determined that AAP Cluster B mission profiles were compatible with the primate experiment objectives. Flight 216 was selected as the launch vehicle for the spacecraft and Flight 221 as the retrieving mission. On the basis of these findings and the experiment requirements, design criteria and system requirements for the payload and spacecraft were finalized.

Trade-off analyses were conducted to arrive at the preferred concepts for all subsystems. These concepts were then translated into preliminary designs of payload and spacecraft subsystems.

A preliminary reliability assessment was made which indicates an overall subsystem reliability of 0.942 for the one-year mission. The designs describe a system which includes the following key features:

- A thermal and atmosphere control subsystem using an active thermal control loop, stored high-pressure gas, partial pressure sensing and control, and lithium hydroxide for carbon dioxide removal
- A passive waste management subsystem for feces, urine, and other animal wastes
- A retrieval subsystem with provisions for retrieving and preserving a dead animal
- A mass measurement subsystem based on measuring the radiation attenuation through the animal
- A data management subsystem utilizing government furnished implanted biosensors and modified Apollo telemetry equipment
- A solar cell electrical power subsystem
- A cold-gas, sun-seeking attitude control subsystem

A final mission analysis was performed to develop a plan for prelaunch, launch, on-orbit, and recovery operations. A development plan was also prepared outlining the steps, time spans, facilities, and equipment required for continuation of the program from the end of this study phase through flight.

The conclusions of this study are that the program objectives are achievable with designs that are essentially within the scope of existing technology and that the desired flight date of mid-1970 can be met.

INTRODUCTION

This introduction discusses the program objectives and approach and the study objectives and approach.

Program Objectives and Approach

The objectives of the Orbiting Experiment for Study of Extended Weightlessness Program are to provide:

- Physiological and psychological data applicable to extended manned space flights particularly in regard to the effects of extended weightlessness
- Scaling factors for long-term life support requirements in the weightless environment
- Long-term life support component experience

To this end, the primary goal of the program is to place two primates in orbit for a duration of six months to one year and to recover them alive at the end of the mission. Experiment design is the responsibility of NASA OART. The animals including implanted transensors will be provided to the contractor as Government Furnished Equipment.

The Orbiting Primate Experiment (OPE) will be carried as part of an independent spacecraft which is completely self-sustaining between the time of orbit injection and animal retrieval. A preselected Saturn/Apollo Applications (S/AAP) flight will be used as the launch vehicle. At the end of the 6 to 12 months mission duration, the experimental animals will be returned to earth by the Apollo Command Module (ACM) of a subsequent S/AAP flight. The OPE will be integrated into the overall S/AAP mission planning which is now in progress. The experimental animals to be used are two female Rhesus (Macaca mulatta) monkeys, each approximately 13 lb in weight.

Physiological, psychological, and environmental data will be obtained by telemetry during the flight and by histochemical and histopathological examination of the animals after recovery. Long-term life support requirements will be obtained by relating telemetered measurements of food, water and oxygen consumption, CO_2 production, and cabin environmental data. Information on life support equipment components will be obtained by monitoring the telemetered performance of key components for the duration of the mission.

Study Objectives and Approach

The objectives of the OPE study reported herein have been to:

- Investigate the requirements which the OPE imposes on the S/AAP missions
- Investigate the requirements which the S/AAP missions impose on OPE hardware and operations
- Translate the above requirements into criteria for hardware design and operations
- Conduct tradeoff studies to examine all significant approaches to meeting the requirements and select a preferred concept
- Provide a preliminary design definition of a flight system which will fulfill the mission objectives and conform to the preferred concept
- Derive a development plan for a logical progression of the program from the conclusion of this study through the conduct of the flight mission

The Lockheed approach to meeting these objectives is shown in the work flow diagram of Fig. 1. In general, the plan consisted of transforming NASA experiment requirements and the results of a preliminary mission analysis and review of S/AAP configurations into a set of spacecraft requirements and design criteria. These, then, formed the basis for conducting tradeoff analyses of various design approaches for all subsystems and the spacecraft itself. The output of these analyses yielded the preferred concept. After approval of these selected approaches by NASA following the midterm review of 28 June 1967, the concepts were translated into spacecraft and subsystem preliminary designs as well as drawings and specifications for a laboratory test model of the payload. As the design definition progressed, major S/AAP interfaces were identified and resolved and a final mission analysis was developed. As a final step, the preliminary design definition was used as a basis for formulating a development plan for continuation of the program through the flight phase.

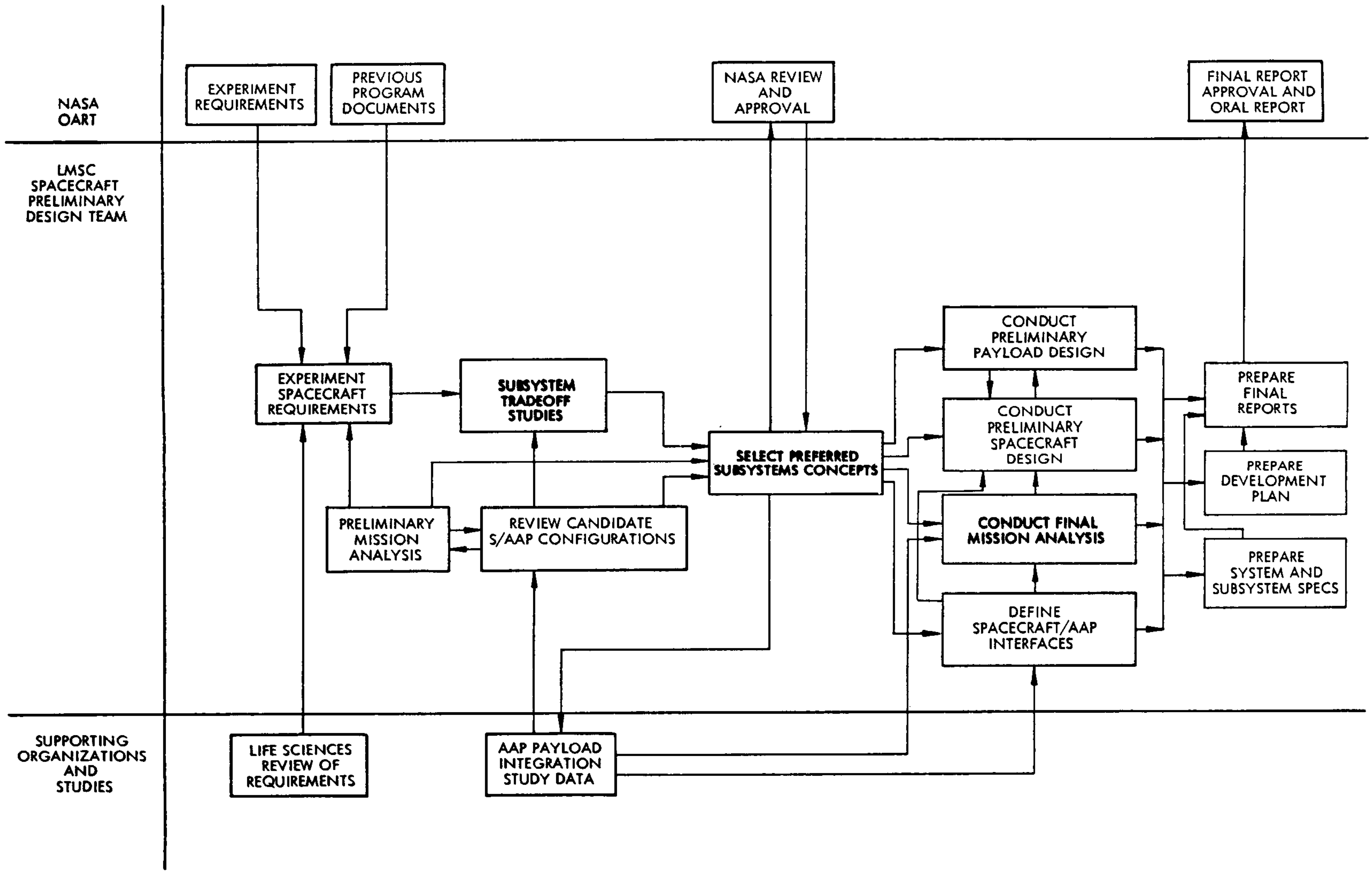

Fig. 1 Work Flow Diagram

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DESIGN CRITERIA AND SYSTEM REQUIREMENTS

This section discusses the OPE design guidelines which were formulated early in the study phase as Task 2 under the contract. It reflects requirements established by the contract Statement of Work, discussions with NASA/LRC personnel and the Principal Investigator, and portions of the results of Task 3 (Preliminary Mission Analysis), and Task 4 (Review of S/AAP Configurations).

Design Objectives

Mission-oriented objectives for the design of the spacecraft and payload were established by four major program phases:

- Preflight Operations
- Launch and Launch Operations
- In-Orbit Operations
- Recovery and Postflight Operations

Preflight operations. – Preflight operations are discussed in the following paragraphs.

Safety: The spacecraft design and spacecraft operations at ETR will comply with ETR range-safety requirements and approved KSC prelaunch procedures.

Animal insertion: The KSC nominal prelaunch procedures for S/AAP flights are being used as interim data for the preliminary definition of OPE prelaunch preparations for flight. Their evaluation has shown that the SLA is normally closed 24 days prior to launch. To provide the means for animal examination and bio-instrumentation adjustment up to the launch day, the animal canisters will be designed to allow insertion of the animals into the payload by T-14 days. No additional access provisions in the SLA will be required. Animal simulators will be used for portions of the spacecraft all-systems checks extending from the T-14 day to the launch day.

Access: A major factor for efficient prelaunch operations and high reliability is adequate access to spacecraft systems and subsystems which must be serviced, checked-out, or exchanged during the remaining days prior to the launch day. Access to these equipments without removal or disconnection of other associated equipments is therefore required.

AGE installation: Mechanical and electrical AGE installed near the spacecraft will be required to perform the spacecraft all-systems checks necessary to verify launch readiness.

Launch and launch operations. – Launch and launch operations are discussed in the following paragraphs.

Launch: The final design of the spacecraft will be based upon the weight, volume, and environmental constraints presented below.

Launch weight: An estimate for the maximum allowed weight of the spacecraft is derived for use as a design guideline. It is based upon the planned S/AAP flights, and their experimental-payload capabilities, presented by MSFC in NASA-MSFC "Guidelines for Payload Integration, Phase "D" Proposal, Revision "A", March 7, 1967 (Ref. 1).

The S/AAP 200-series earth orbit missions, the weights of experiments planned for these flights, and a description of these experiments are reproduced from the MSFC document in Tables 1 through 3. The following entries on these tables are highlighted:

- Table 1. Launch of the OPE spacecraft aboard the SLA on Mission 216 or 217, and recovery aboard the CM on Mission 221, would be required to fulfill the requirement for orbit durations of 6 to 12 months.

- Table 2. With respect to Missions 216 and 217, only the former has the payload capability to accept additional experiments greater than a few hundred pounds.

- Table 3. One of the experiments planned for flight is Experiment T009, "Primates in Long Term Zero G."

From Table 2, the total allocated weight of experiments is 200 lb. Assuming no weight growth of these experiments, the payload capability left is 7,279 lb. Assuming that the OPE spacecraft might be added as another experiment, a rough estimate of its maximum allowed weight is derived as follows:

- 200 lb = allocated weight of other-than-OPE experiments
- 400 lb = estimated, potential growth weight of other-than-OPE experiments
- 7,279 lb - 400 lb = 6,879 lb
- 6,879 lb - 20% (6,879) ≅ 5,500 lb

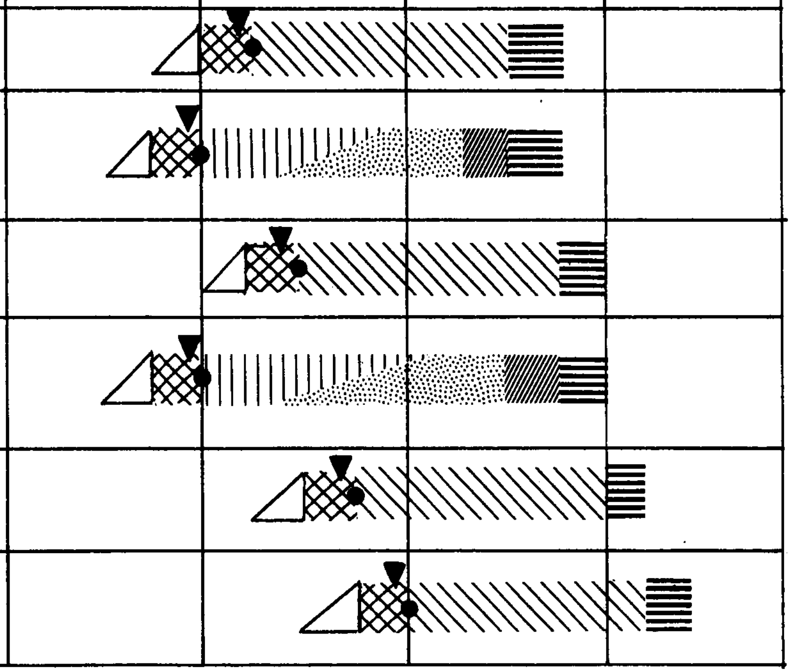
The 6,879-lb figure represents an estimated maximum weight of OPE allowed on the launch day. The 5,500-lb figure allows for a potential weight growth and a contingency margin. The target weight of the OPE spacecraft is established as 5,500 lb.

Launch volume: The available space for experiments inside the SLA is shown in Fig. 2. The available volume for experiments (payload envelope) is 4,525 ft^3. The locations of experiments other than the OPE spacecraft within this volume, their individual dimensions, and attachment structure are not known.

TABLE 1

AAP SELECTED MISSION – SCHEDULE

Near Earth Cluster "B" Mission		AAP Schedule			
Launch Vehicle	Planned Payload and Carriers	1968	1969	1970	1971
216	CSM				
217	OWS-2 AM/MDA				
218	XCSM				
219	LM-A/ ATM-B				
220	XCSM				
221	X or L CSM				



AAP Concept Des. and Integ

NASA Flt. Directive

Mission Flt. Go-ahead

Sys. Anal.

KSC Ops.

Exp. Support

Des. & Dev.

Fab. & Mod.

Sys. Test

TABLE 2

MISSIONS 216 AND 217, PAYLOAD WEIGHT CAPABILITIES

MISSION:	216	217
CARRIER:	CSM	OWS NO. 2 AM/MDA

TOTAL PAYLOAD CAPABILITY	40,300	27,900
PAYLOAD ALLOCATED	33,021	27,667
SLA	3,950	-
Adapter/shroud	-	4,100
Nose Cap	-	1,067
CSM	25,262	-
Expendables/spares	-	2,000
SPS Propellant	3,109	-
AM	-	8,705
MDA	-	3,595
Stage Mod.	300	2,500
Solar Panel	-	4,100
Booster Yaw	-	100
Experiments	200	1,500
Support Rack	200	—
AVAILABLE PAYLOAD WEIGHT	7,279	233

TABLE 3

MISSIONS 216-221, PLANNED EXPERIMENTS

Experiment Designation	Experiment Name
–	Biomedical Laboratory (Early Version)
T005	Fusible Material Radiator
T006	Vision Test Equipment Evaluation
T007	Human Transfer Functions
T009	Primates in Long Term Zero "g"
T014	Orbital Horizon Definition
T015	Meteoroid Composition
–	Multi-Sphere Satellite (MSFC 47)
M433	Satellite Recovery
–	Behavior of Particular Material (MSFC 54)
–	Liquid Drop Dynamics (MSFC 26)
M426	Condensing Heat Transfer
–	Fluid Density Gradient (MSFC 43)
–	Project Thermo
–	ATM-B
–	Application "B," Natural Resources
–	Electromagnetic Radiation
–	Infrared Spectrometer Radiometer
–	Earth Albedo Measurements
–	Cosmic Ray Neutron Albedo Spectrum
–	Ultraviolet Remote Sensing Measurement

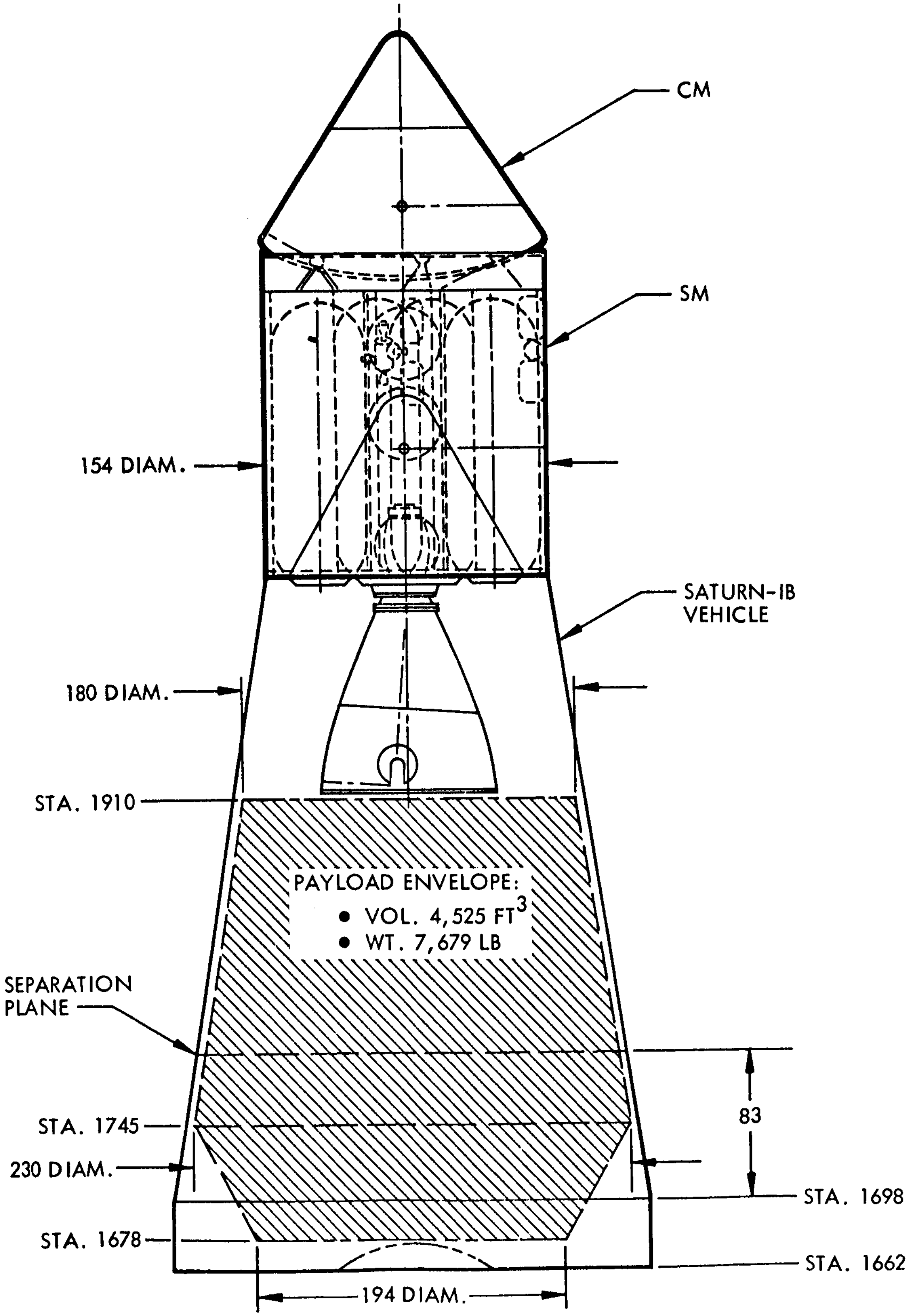


Fig. 2 Available Space Inside LM Adapter

Fig. 3 compares the space required by the OPE with the volume occupied by other prime AAP payloads on other flights. It is clear that any single flight can accommodate only one of the three prime payloads shown. The selection of ascent Mission 216 is based on this comparison.

Launch environment: The design of the spacecraft will be designed for successful operation of necessary functions during and after exposure to the launch environments delineated below.

1. Vibration. The booster vibration phenomena involving resonant frequencies of the total vehicle and short-time instabilities of the propulsion system may influence spacecraft vibration during powered flight. Coupling between low-frequency modes and steady-state oscillations will be assumed. The sinusoidal and random vibration amplitude-frequency-time histories presented in Figs. 4 and 5 will be used.

2. Acoustic noise. During liftoff and transonic flight, acoustic pressure fields surrounding the vehicle will occur, causing random vibrations of structural members and systems of the spacecraft. Noise will be generated during liftoff by combustion and aerodynamic shearing forces associated with the Saturn's rocket engine exhaust. Noise amplification due to ground reflection will occur. The magnitude and character of the noise field affecting the spacecraft will therefore depend upon the rocket-engine characteristics, distance of the spacecraft from the noise source, and factors affecting the reflection of exhaust gases. The acoustic noise amplitude-frequency-time histories presented in Fig. 6 will be used.

3. Acceleration and dynamic pressure. The acceleration-time and pressure-time histories for a nominal trajectory of the S-IB vehicle, shown in Figs. 7 and 8 will be used.

4. Velocity. The velocity-time histories for a nominal trajectory of the S-IB vehicle, shown in Fig. 9, will be used.

In-orbit environment: In-orbit environment covering micrometeoroid data and nuclear radiation is discussed below.

1. Micrometeoroid. Analysis of impact data obtained from space vehicles indicates the presence of a higher concentration of meteoric dust near the earth as compared with interplanetary space. Three characteristic zones to the earth's particle belt have been suggested by meteorologists. The zone nearest earth covers the altitude range from 100 to 400 km. Its characteristics are as follows:

- Flux impact, 0.1 to 1.0/m^2-sec
- Flux mass, 10^{-13} to 10^{-12} gm/cm^2-sec
- Particle concentration, 4×10^{-11} to $4 \times 10^{-12}/cm^3$
- Density, 4×10^{-19} to $4 \times 10^{-18} gm/cm^3$

The spacecraft will be designed against meteoroid penetrations which may degrade the operation of necessary functions. Averages of the above characteristics will be used.

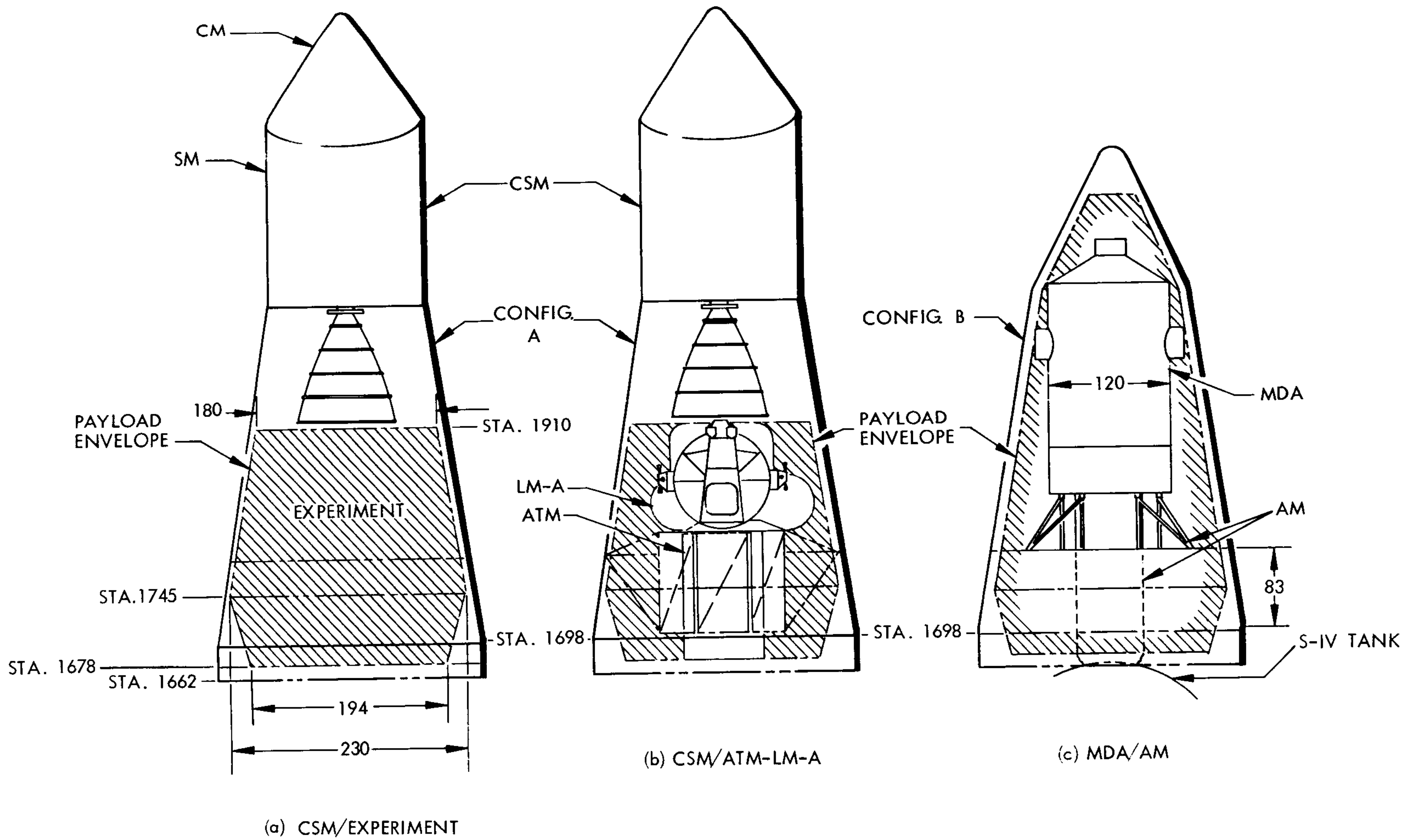
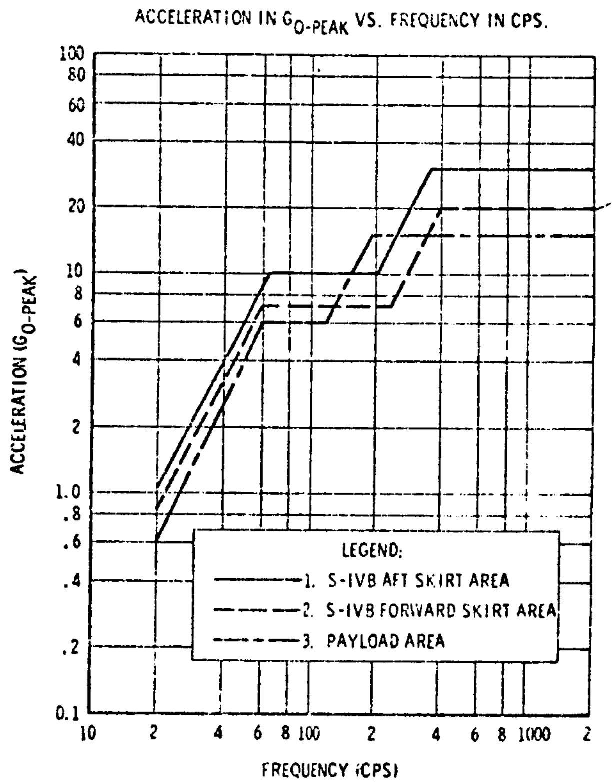


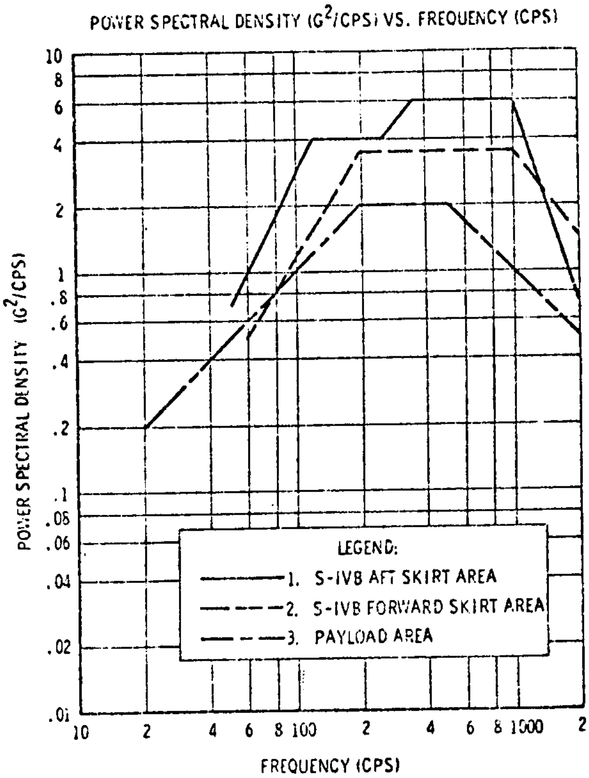
Fig. 3 S/AAP Launch Configurations



NOTE:

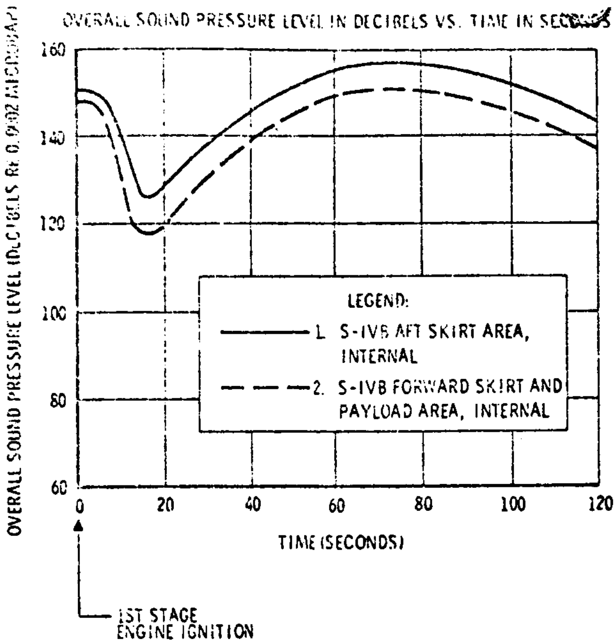
1. THE VIBRATION INPUT IS ASSUMED TO BE APPLIED IN EACH OF THREE MUTUALLY PERPENDICULAR DIRECTIONS.
2. THE LOGARITHMIC SWEEP RATE IS ASSUMED TO BE ONE OCTAVE PER MINUTE OVER THE FREQUENCY RANGE FROM 20 TO 2000 AND BACK TO 20 CPS.

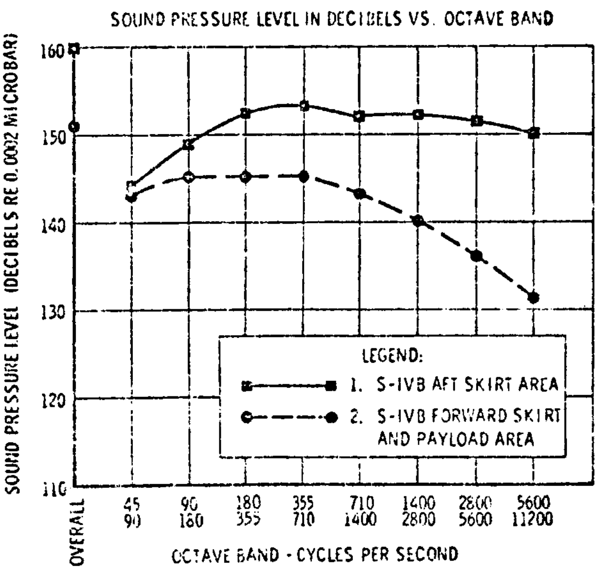
Fig. 4 Design Specification for Sinusoidal Vibration



NOTE.
1. AMPLITUDE DISTRIBUTION IS ASSUMED GAUSSIAN
2. DURATION IS ASSUMED TO BE TWELVE MINUTES FOR EACH OF THREE MUTUALLY PERPENDICULAR DIRECTIONS

Fig. 5 Design Specification for Random Vibration





NOTE:

1. THE TIME DURATION IS ASSUMED TO BE EIGHTEEN MINUTES.
2. THE DIFFUSED SOUND FIELD OF RANDOM NOISE IS ASSUMED TO HAVE A GAUSSIAN AMPLITUDE DISTRIBUTION.

Fig. 6 Saturn IB Payload Acoustics

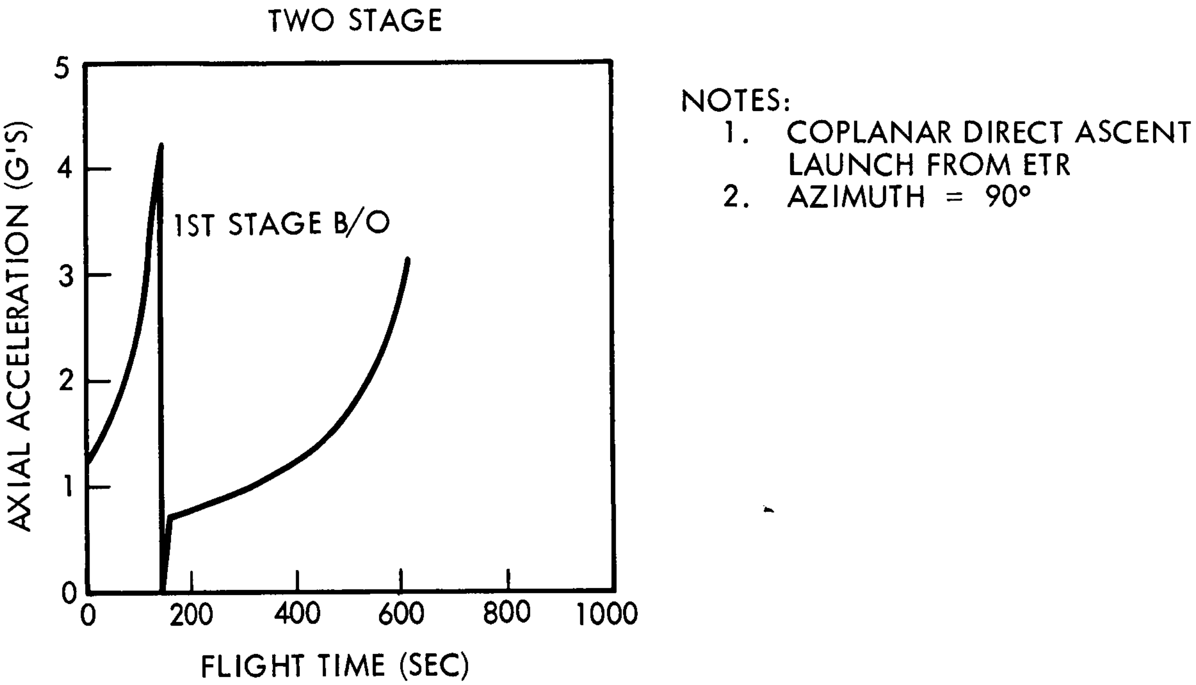


Fig. 7 Saturn IB Axial Acceleration vs. Flight Time

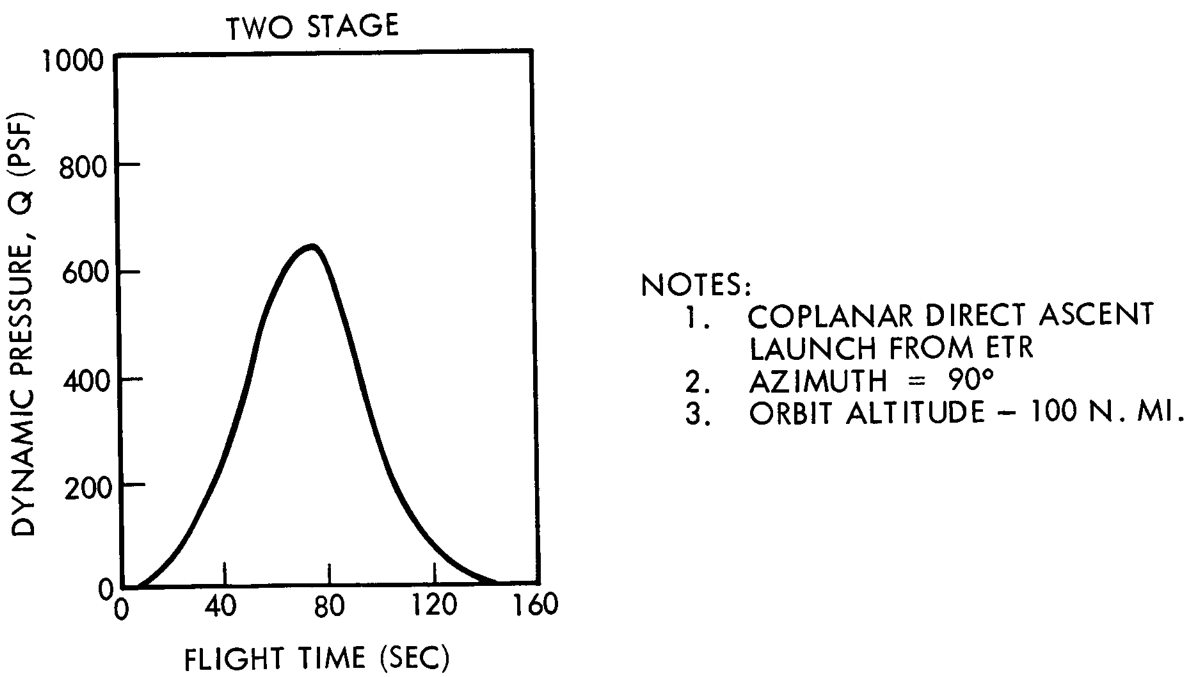


Fig. 8 Saturn IB Dynamic Pressure vs. Flight Time

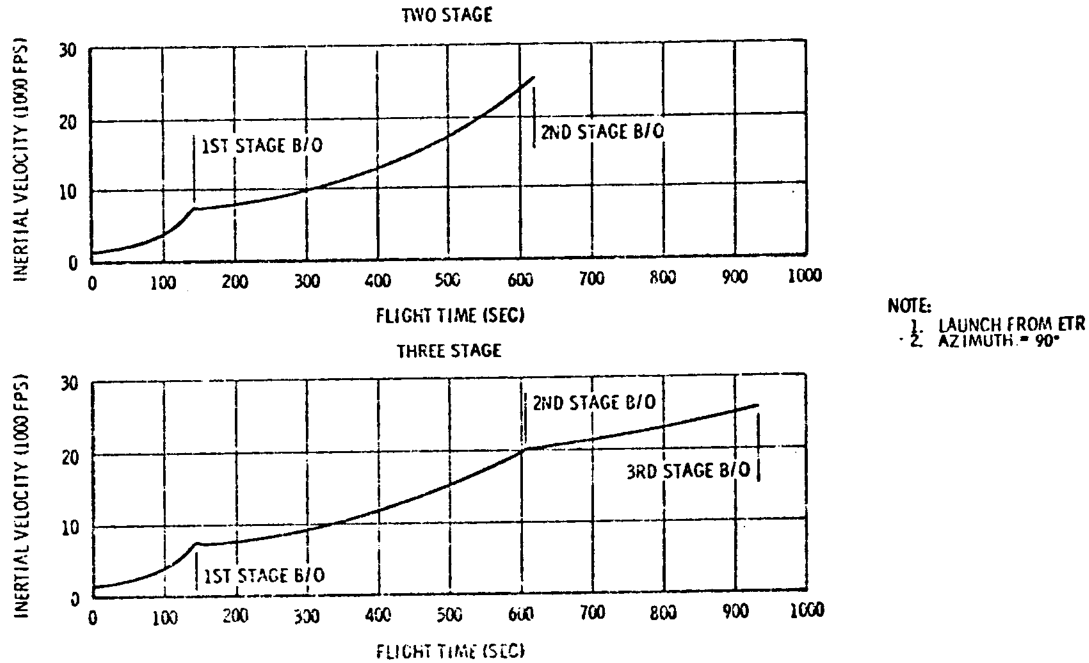


Fig. 9 Saturn IB Velocities

2. Nuclear radiation. The penetrating radiation environment for a low-altitude, low-inclination, circular earth orbit will be considered. This orbit is protected from solar flare plasma by the geomagnetic field. Trapped artificial electrons form the major part of the electron dose. The following integrated mission fluxes for these orbits will be used:

- Protons, 4 Mev, 6×10^4 p/cm^2
- Electrons, 0.5 Mev, 5×10^8 e/cm^2

Ascent sensing: Knowledge of the acceleration, vibration, and noise levels actually experienced by the spacecraft (equipment and animals) during powered flight is necessary to assure that the experimental results will be meaningful. This is so because of uncertainties concerning the expected severity of, and responses from, these levels. The exposure of the animals to levels more severe than expected can degrade (or preclude the acquisition of) the experimental data. Knowledge of the acceleration, vibration, and noise is used to interpret the test results if in wide disagreement with predicted results.

In-flight measurements of vibration, noise and acceleration inside of, and as affected by, the SLA are therefore required. The continuous measurement of these environments during powered flight, to determine local conditions at several points common to the LM adapter and the spacecraft structure, are specifically required. The measurement of energy levels in discrete ranges of frequencies, and not complete vibration and noise spectra, would probably suffice.

Criteria: The orbits chosen for the OPE spacecraft must meet the following mission requirements:

- The orbital lifetimes must be within the mission duration range of 6 to 12 months.

- The spacecraft must be stabilized to within 0.001 g steady-state and 0.01 g transient accelerations.

- A complete readout of basic physiological data including housekeeping and TV data will be collected for a 5-minute period four times per day, to be transmitted to earth once per day upon command. This is a minimum.

It is also desirable for the orbits chosen to have the following characteristics:

- At the end of the on-orbit mission, a spacecraft orbit altitude near the Mission 221 CM orbit altitude is required in order to minimize the CM propulsion required to execute the rendezvous sequence.

- For a given time of launch (year, quarter) and initial orbit inclination angle, it is desirable to maximize solar-cell efficiency through a predetermined orientation of the spacecraft relative to the sun.

- The ground traces associated with the on-orbit altitudes should result in the maximum utilization of the Apollo ground stations without modification.

- The initial orbit-inclination angle selected to maximize spacecraft/ground-station contact times should be approximately the same as that tentatively planned for Mission 216 (29 degrees).

- The initial orbit altitude selected to meet the spacecraft mission-duration requirements should be the same as that tentatively planned for Mission 216 (260 n. mi.).

In-orbit operations. — In-orbit operations covering communications, setup and calibration, spacecraft control center, and astronaut EVA are discussed in the following paragraphs.

Communications: The capability of remote monitoring and control of payload status data is required. The spacecraft will use a communications subsystem to transmit wideband 5 Mc data to earth. A ground-station network capability of a minimum of two contacts per orbit revolution for 12 months to receive this data is required. The airborne Data Management Subsystem must be compatible with the configuration and capabilities of the Apollo ground station network (MFSN). Television cameras will be used aboard the spacecraft to observe animal activity and must be tape recorded at the acquiring stations.

Setup and calibration, operation: Checkout of the spacecraft (payload and critical systems) at periodic intervals during the 12-month orbital flight is required to verify satisfactory equipment operation. External artificial stimuli or fault isolation procedures which are built into the spacecraft are required.

Spacecraft operations control center: The transmission of all experimental data, once acquired by or returned to earth, in near real-time to a central control center for processing and to facilitate timely decisions on contingency command procedures is required. This requirement is especially stringent with respect to lifecell data.

Astronaut EVA: Following CM/spacecraft rendezvous after completion of the spacecraft in-orbit phase, astronaut extra-vehicular activity is required to retrieve the primate-retrieval canisters. The crew skills required fall into the following categories: observation skills (cognitive-perceptual) and actual mechanical manipulation (motor).

Recovery operations. — Recovery operations are discussed in the following paragraphs.

Recovery weight and volume: The weight and volume of the animal retrieval canisters must be compatible with the return capabilities of the retrieving CM. Any required rearrangement or modification of the CM to achieve this must be kept to a minimum level without compromise to AAP mission objectives.

Postflight disposition: Retrieval of the monkeys soon after earth landing and return (alive or dead) to a postflight examination laboratory is required.

Life Support and Environmental Requirements

Environmental requirements. – The conditions which will be maintained within the lifecell are as follows:

Temperature	75 ± 3°F
Total Pressure	14.7 ± 2 psia
Oxygen Partial Pressure	18 – 30%*
Carbon Dioxide Partial Pressure	1% * maximum
Methane Partial Pressure	4.6% * maximum
Trace Contaminants	0.1 ACGIH TLV **
Relative Humidity	50 ± 10% 30 to 85% for 1 to 2 days
Air Velocity	30 ft/min
Air Flow (Two Cages)	424 CFM (1908 lb/hr)
Leakage Rate	0.3 lb/day maximum 0.07 lb/day minimum

Metabolic requirements. – The animal basal metabolic rate was determined to be 291 K cal/day or 1,155 Btu/day. This value was derived from information presented in Ref. 2.

The data given for the Rhesus primate is:

$$\text{Basal metabolism} = 48.4 \text{ K cal/kg/day}$$

For a 6 Kg animal, basal metabolic rate is:

$$\text{BMR} = (6)(48.4) = 291 \text{ K cal/day} = 1{,}155 \text{ Btu/day}$$

The animals are presumed to be functioning at an average metabolic rate of 1.5 × basal, or 1,735 Btu/day.

The CIBA food pellet selected to sustain the animal is spherical in shape and approximately 7/16 inch in diameter. The composition of the pellet is as indicated

*of 14.7 psia

**American Conference of Government and Industrial Hygienists Threshold Limit Value (1965) in the following listing:

	Percent
Sucrose	14
Banana flake	20
Soybean flour	30
Dried egg white	10
Non-fat dry milk	8
Dried whole egg	4
Wheat germ	5
Brewers yeast	2
Sodium saccarin	1
Sodium sucrose	1
Calcium lactate	0.70
Iodized salt	0.01
Other	0.04
Vitamins	
A – D $B_1 - B_6$ Pantothenic acid Nicotinic acid	0.25
C Binder	4

The carbon dioxide production rate, oxygen, and food required to sustain body weight at the established levels is determined from the data listed in Table 4.

TABLE 4

CONSUMPTION OF O_2 AND PRODUCTION OF CO_2 AND HEAT IN METABOLISM OF FOOD PELLET

SUBSTANCE	Composition (gm/1 gm pellet)	O_2 Required (cc)	CO_2 Produced (cc)	Heat Produced	
				K-cal	BTU
Carbohydrate	0.56	465.0	465.0	2.35	9.40
Fat	0.04	80.6	57.3	0.38	1.52
Protein	0.26	248.0	201.0	1.14	4.56
Roughage	0.14	—	—	—	—
Total	1.00	793.6	723.3	3.87	15.48

The daily food pellet requirements are found as follows:

$$\text{Average daily metabolic rate} = \text{pellets/day} \times 15.48 \text{ Btu/pellet}$$

$$n = \text{pellets/day} = \frac{1735}{15.48} = 112 \text{ (for one animal)}$$

The oxygen required to metabolize this quantity of food and the carbon dioxide produced are determined as follows:

$$O_2 \text{ required} = 793.6\,n = 793.6 \times 112 = 88{,}000 \text{ cc/day}$$

For standard temperature and pressure conditions (STP) 88,000 cc/day = 0.260 lb/day. CO_2 produced = 723.3 n = 81,200 cc/day. For standard temperature and pressure conditions (STP) 81,200 cc/day = 0.335 lb/day.

The animal water balance is given as follows:

	gm/day
Water Intake	420
Metabolic Production	72
Total	492
Urine Output	318
Insensible Loss	174
Total	492

For the 379-day mission (365 days + 14 days on pad) a total of 93,386 one-gram food pellets and 772 lb of water are required. These quantities include a 10 percent contingency margin as shown in Fig. 10 and 11. A total of 197 lb of metabolic oxygen will be required for the mission, as indicated in Fig. 12.

Preservation Requirements. — Although one of the key program objectives is to recover both primates alive, it is considered essential to design into the payload the means for primate preservation should his death from natural, or flight-imposed, causes occur.

The anthropometric data presented in Figs. 13a and 13b will be used to determine the preservation-design requirements and to size the primate retrieval canisters.

Data requirements. — The following implanted physiological sensors will be included:

- Electrocardiogram
- Respiration
- Body Temperature

Other primate-oriented sensors will include the following:

- Primate Activity
- Vocalization
- Mass Measurement
- Psychomotor Performance
- High- and Low-Resolution Video

No exceptions to the contract data requirements are taken.

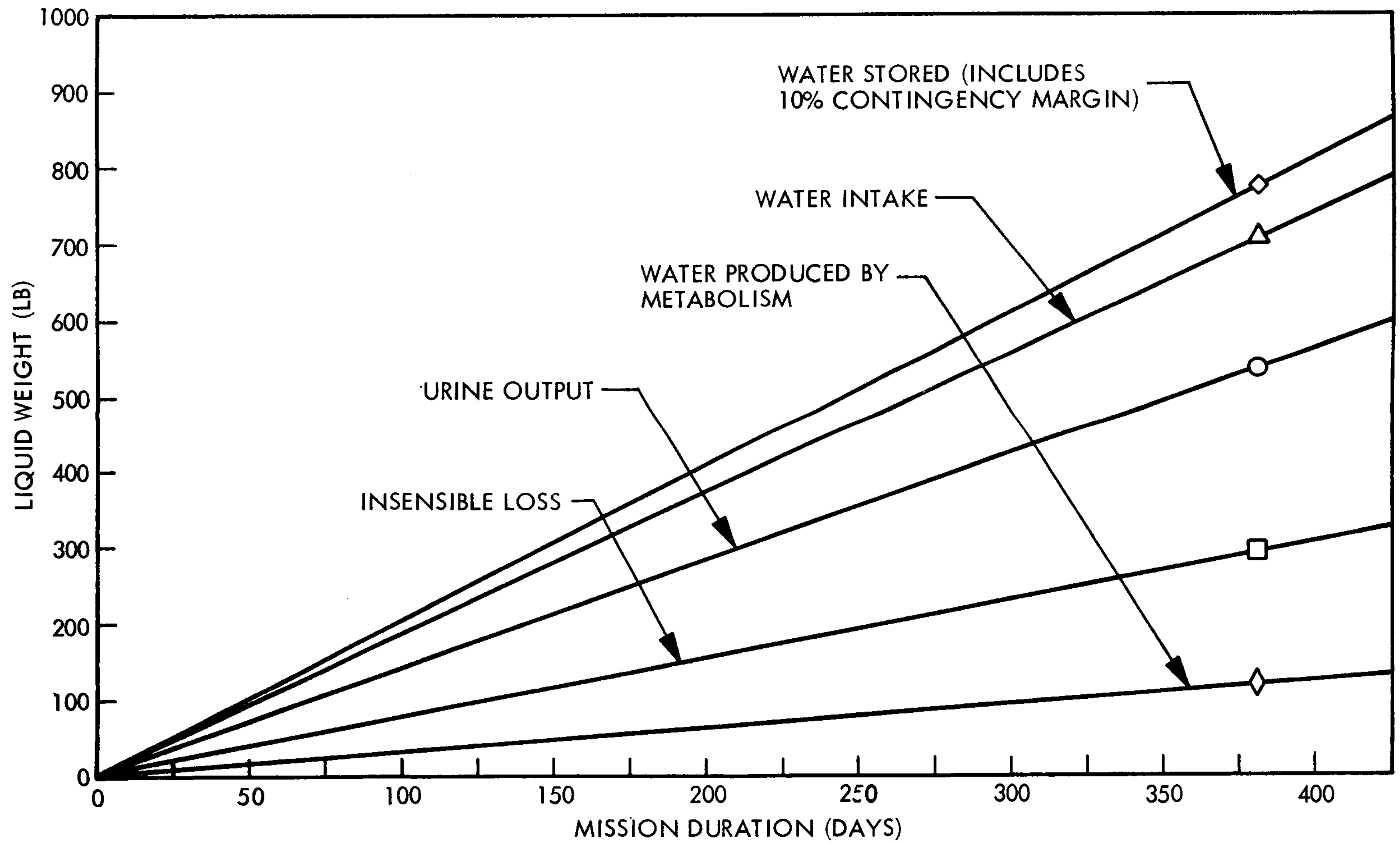


Fig. 10 Water Balance for Two 6.0 kg Rhesus Primates

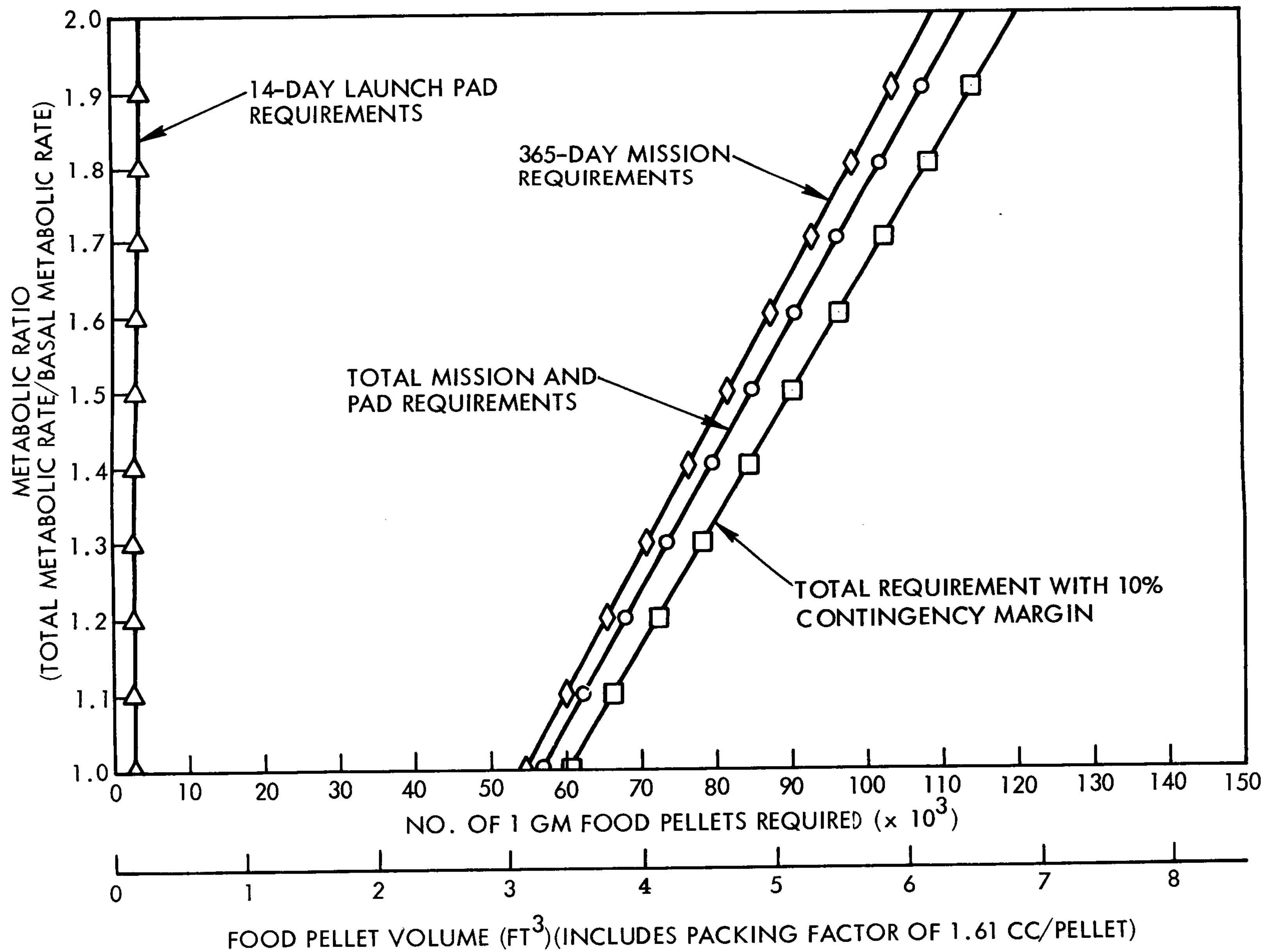


Fig. 11 Food Pellet Requirements for Two 6.0 kg Rhesus Primates (Basal Metabolic Rate – 48.2 Btu/hr)

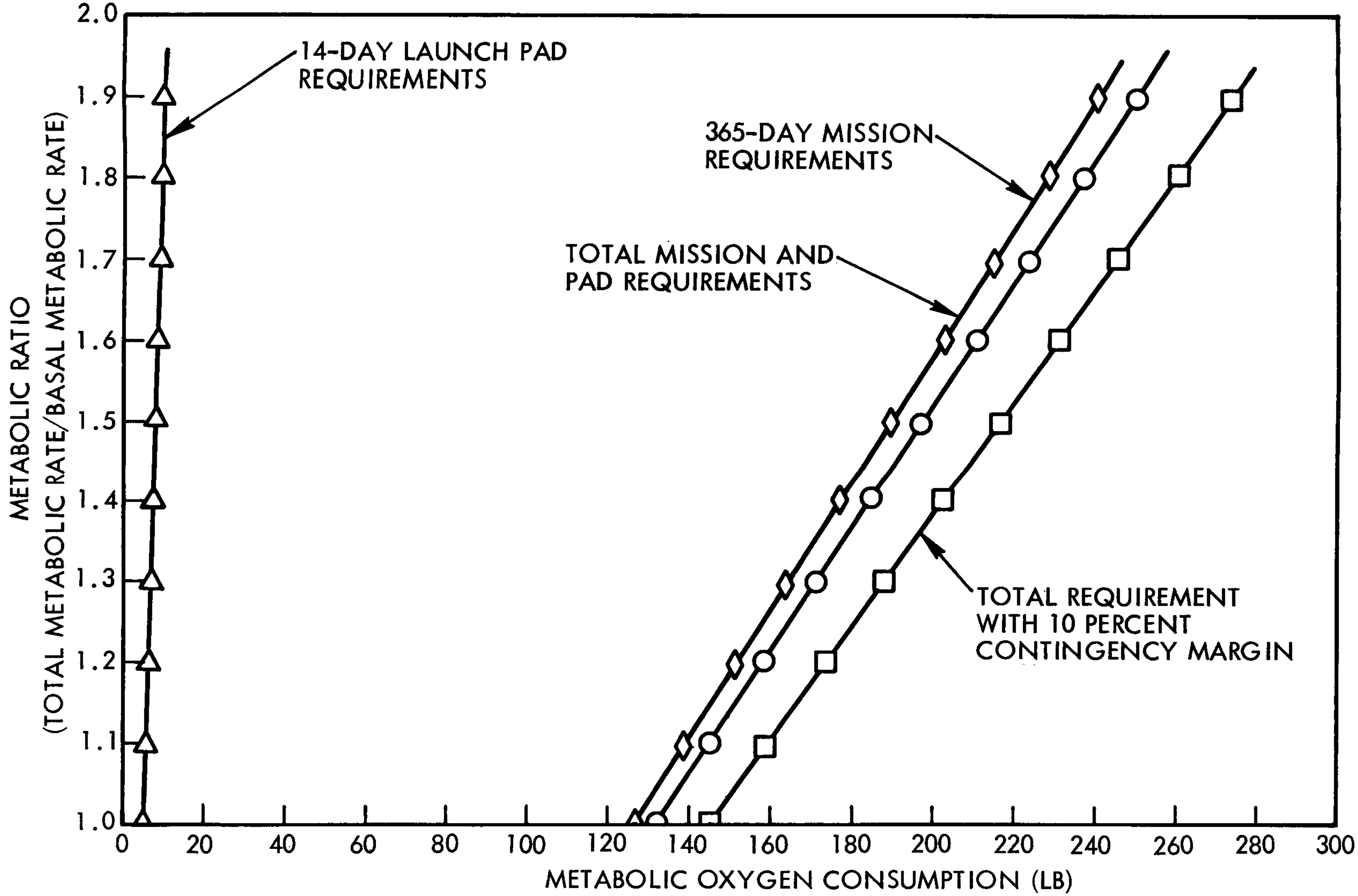


Fig. 12 Metabolic Oxygen Requirements for Two 6.0 kg Rhesus Primates (Basal Metabolic Rate – 48.2 Btu/hr)

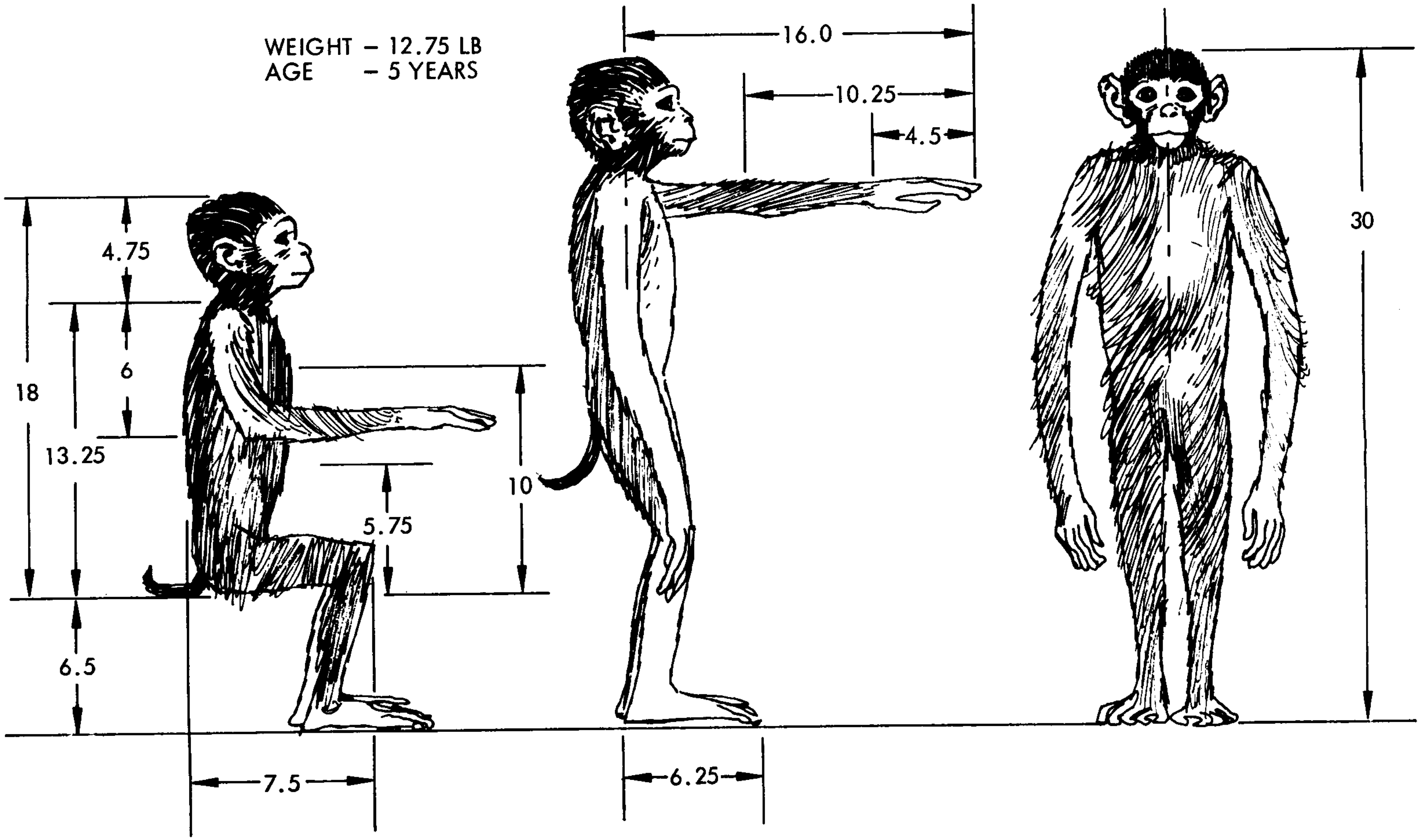


Fig. 13a Rhesus Anthropometric Linear Dimensions

CIRCUMFERENCE		
NO.	ITEM	INCHES
1	HEAD	11.5
2	NECK	9.5
3	SHOULDER	16.0
4	CHEST	15.0
5	WAIST (HIP)	11.5
6	BICEPS	6.25
7	WRIST	3.25
8	THIGH (MID SHAFT)	9.00
9	CALF	5.75
10	ANKLE	3.60

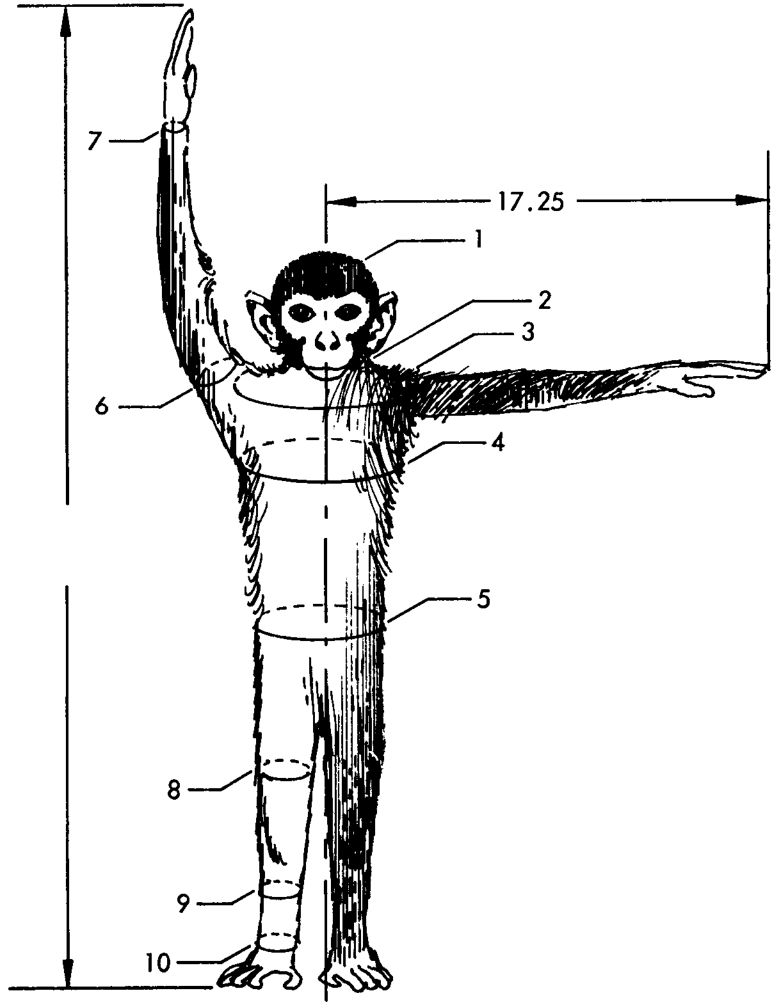


Fig. 13b Rhesus Anthropometric Circumferential Dimensions

Behavioral program. – A behavioral program and task panel is required to monitor animal pyschological performance. This equipment will be used as the primary enabling source for supplying food and water to the animals. The criteria for the design of this equipment will be based on the requirements of the Principal Investigator as expressed through NASA/LRC.

Spacecraft Requirements

Spacecraft requirements for spaceframe, guidance and control, electrical power, data management, thermal control, and aerospace ground equipment are discussed in the following subsections.

Spaceframe. – The spaceframe will house, support, and protect all other elements of the spacecraft and payload during ascent, spacecraft separation, final orbit injection, and solo orbit. Design considerations are:

- Optimum configuration for allowable envelope volume
- Subsystem packaging to maximize radiation and meteoroid shielding of the payload
- Ease of installation into and separation from the Apollo vehicle
- Ease of access for installing the animals in the payload
- Optimum moments-of-inertia and mass distributions for attitude control

Guidance and control. – The Guidance and Control System will control on-orbit angular rates so that the animals experience no more than 1×10^{-3} g steady state and 1×10^{-2} g transient accelerations. The design will also take into consideration such factors as antenna pointing, solar array orientation, drag area, and thermal control.

Electrical Power. – The electrical power subsystem is required to supply power to all spacecraft and payload equipment during all phases of the mission. System responsibility will cease at the 28v (nominal) DC bus. It will interface with the pad ground source during on-pad operations, be self sustaining from lift off to launch vehicle separation, and as a self-sustaining system during solo-orbit operations. The planned orbits are low-inclination orbits out of ETR. Therefore, if a solar energy source system is used, it must be capable of operation on a continuous basis throughout an approximate 50-50 light-dark schedule.

Data management. – The data management subsystem will transmit and receive all spacecraft and payload data between the spacecraft and the ground stations. The data to be measured will include spacecraft diagnostic information and payload operating characteristics. The system will also include antennas, recording equipment for storing data which are gathered between ground station readouts, any timers or programmers for cyclic operation of equipment, and the necessary command equipment for enabling, initiating, or stopping spacecraft or payload functions or for overriding programmed functions. The system will be compatible with the Apollo ground station net (MFSN) with minimum impact on ground equipment existing between late 1969 and 1971.

Thermal control. – The thermal control subsystem will maintain the spacecraft, payload, and all on-board systems within required temperature limits during the mission sequence from prelaunch to earth landing. Consideration will be given to such factors as:

- Internal heat sources and absorbers (chemical, metabolic, mechanical, and electrical components)
- External heat sources and absorbers (launch environment, ascent heating, proximity to launch vehicle, solar, albedo, and earth shine)
- Equipment location and mounting (conduction and convection paths, heat barriers)
- Unequal thermal expansion or warping of critical structures (booms, antennae, solar arrays, etc.)

Aerospace ground equipment (AGE). – The approach for the fulfillment of all AGE requirements will be the maximum utilization of existing equipment or its modification, where necessary, and the identification of any additional equipment required.

AGE equipment requirements will be established by three major program phases:

- Vehicle systems test at the factory
- Systems tests and prelaunch operations at the launch site
- Postlaunch operations at the recovery site

Handling, checkout, and servicing equipment will be utilized to accomplish all of the above mentioned program phases. Their designs will eventually be based upon the following basic considerations:

- Multipurpose usage
- Compliance with the requirements for the handling and transportation loads imposed upon the spacecraft
- Maximum expected life of two years with reasonable servicing and replacement of parts
- Interchangeability of parts between all similar AGE and components
- The preference for semi-automatic equipment
- Human engineering principles
- Mobility (where applicable)

- Coupling to the spacecraft without degradation to spacecraft subsystems reliability or the establishment of unrealistic test conditions
- ETR prelaunch operations policies and procedures
- NASA/LMSC postflight operations requirements

The design of the spacecraft will not dictate the use of new AGE equipment or techniques without adequate tradeoff analysis data.

PRELIMINARY MISSION ANALYSIS AND AAP REVIEW

This section presents the preliminary mission analysis and review of candidate S/AAP configurations (Tasks 3 and 4) to support the evaluation and selection of system concepts. The principal consideration of the analysis was the AAP mission profile and its potential impact upon the OPE mission and spacecraft design. The mission analysis results provided the framework for accomplishing the following tasks:

- Establishment of spacecraft configuration requirements and preliminary subsystem interfaces (Task 4)
- Completion of spacecraft system tradeoff studies and definition of candidate spacecraft/payload systems (Task 5)
- Selection of preferred spacecraft and subsystem concepts (Task 6)

The preliminary mission analysis was performed in the following sequential steps:

- Evaluation of S/AAP mission profiles
- Definition of the optimum mission profiles for maximum experiment utilization
- Determination of best suited S/AAP flights capable of accommodating the spacecraft

Evaluation of S/AAP Missions

The evaluation of the S/AAP Missions is discussed in the following paragraphs.

Basis, scope, and AAP description. The 14 March 1967 status of the NASA-approved AAP planning schedule is the basis for the schedule information presented in this section. This planning schedule and the payload modules approved for flight are presented in Table 5.

The material presented in Refs. 1 and 3 is consistent with Table 5 and was used as the basis for this section. Reference 1 constructs the Cluster A design reference mission. It was prepared by Lockheed under contract NAS 8-21003 with NASA-MSFC as part of the AAP Payload Integration Final-Definition Phase (Phase C) results. Reference 2 comprises the MSFC Phase D proposal – Preparation Guidelines.

As indicated in Table 5, the near-earth orbital Apollo missions are designated the "200 Series" flights. They are grouped to comprise the Cluster A, B, C, and D missions. Each cluster mission is executed by assembling in orbit the combined payloads carried into orbit by multiple launches of the Saturn 1B launch vehicle. Cluster

TABLE 5

AAP PLANNING SCHEDULE
(10 March 1967)

	Flight	Launch Vehicle	Payload Module (Excluding SLA)	Launch Year/Quarter					
				68	69	70	71	72	73
Cluster "A"									
Basic Mission	A1	207	CSM, Rack/LM and SS	▮					
	A2	209	OWS-1, AM, MDA, Nose Cone						
	A3	211	XCSM	▮					
	A4	210	LM, Rack/ATM-A, Nose Cone						
Resupply Mission	A5	212	XCSM		▮				
	A6	213	XCSM		▮				
Alternate Mission	A7	214	LM-A, Rack/EMR		▮				
	A8	215	XCSM		▮				
Cluster "B"									
Basic Mission	B1	216	CSM			▮			
	B2	217	OWS-2, AM, MDA, Nose Cone						
	B3	218	XCSM			▮			
	B4	219	LM-A, Rack/ATM-B, Nose Cone						
Resupply Mission	B5	220	XCSM				▮		
	B6	221	XCSM or LCSM				▮		
Cluster "C"									
Basic Mission	C1	517	(Not identified)				▮		
	C2	222	XCSM						
Alternate Mission	C3	223	LM-A, Rack/EMR				▮		
Contingency Payloads	C4	224	(Not identified)					▮	
	C5	225	(Not identified)					▮	
Cluster "D"									
Basic Mission	D1	521	(Not identified)					▮	
	D2	226	XCSM					▮	
Alternate Mission	D3	227	LM-A, Rack/EMR					▮	
Contingency Payloads	D4	228	(Not identified)						▮
	D5	229	(Not identified)						▮
Sync. Orbit Missions									
Three Missions	–	513	XCSM, LM-A, ATM-C			▮			
	–	514	XCSM, OWS-3, AM, MDA				▮		
	–	522	XCSM, LM-A, ATM-D						▮
Lunar Orbit Missions									
Two Missions		510	(Lunar Mapping and Survey)			▮			
		518	(Lunar Mapping and Survey)					▮	
Lunar Orbit/Surface Missions									
Eight Missions		511	(Lunar Surface Exploration)			▮			
		512	(Lunar Surface Exploration)			▮			
		515	XCSM, LM Shelter				▮		
		516	XCSM, LM Taxi				▮		
		519	XCSM, LM Shelter					▮	
		520	XCSM, LM Taxi					▮	
		524	XCSM, LM Shelter						▮
		525	XCSM, LM Taxi						▮
Interplanetary Mission									
One Mission		523	(Interplanetary Flight Module)						▮

orbital lifetimes are extended by resupplying the cluster expendables and by crew rotation through later flights of the Saturn 1B. As shown in Table 5 , the time period covered by the Cluster C and D basic/resupply launches is 4th-quarter 1971 through 3rd-quarter 1973. These flights are excluded from consideration in this section because of the NASA objective to fly the OPE spacecraft before the long duration manned interplanetary missions are undertaken. Only the Cluster A and B missions according to the 14 March 1967 status (approved by MSFC) are evaluated in the following paragraphs.

Potential changes in basis of evaluation. The schedule presented in Table 5 has been studied by LMSC-AAP for improvements that would best achieve the MSFC integration requirements. Recommended changes to the MSFC schedule and payload-flight groupings have resulted, one of which is reported in Ref. 4. This material was presented to NASA-MSFC on 26 May 1967. These, as yet unapproved versions, differ importantly from their latest approved counterpart. The divergence could result in different OPE designs for optimum experiment utilization and integration. The unapproved versions are discussed in Appendix A.

New MSFC Phase D Proposal – Preparation Guidelines were obtained during the final period of preparing this report. These new guidelines are contained in Ref. 5. Mission descriptions, mission profiles, and associated data which differ importantly from the previous MSFC guidelines are included, and summarized in Appendix B for information. Four alternate mission profiles which respond to these new guidelines are analyzed in Ref. 6.

Due to the complexities of integrating many different experiments, the requirements for AAP flights are changing and probably will continue to change until the results from Phase D become available. The changing nature of the AAP plans are apparent in Table 5 where alternate flights and flights to carry contingency payloads are assigned.

As shown in Appendix B, some of these plans include methods of accomplishing OPE mission objectives by means of modularized packages incorporated as part of the overall cluster. This study recognizes that such alternate approaches are feasible. However, the analyses which follow are directed primarily to the design of an independent OPE spacecraft that depends on S/AAP flight vehicles only for orbit injection and experiment retrieval. If, for some reason, the modular approach should later be deemed more attractive to NASA, the preliminary designs of critical subsystems could be repackaged with little impact on program continuity.

Cluster A basic mission. The Cluster A basic mission consists of four uprated Saturn IB flights, carrying the following payload modules:

Flight	Vehicle	Payload Module (excluding SLA)
A1	207	CSM, Rack/LM and SS
A2	209	OWS-1, AM, MDA, Nose Cone
A3	211	XCSM --
A4	210	LM, Rack/ATM-A, Nose Cone

The OWS is the orbiting workshop, synonymous with the Modified S-IVB Stage. These four flights are planned for launch over a period of 3 to 7 months (depending upon final mission requirements) beginning in 2nd quarter 1968.

Integrated mission profile: The Cluster A integrated mission profile is presented in Fig. 14. It consists of two dual launches separated by a period of 3 to 6 months during which the OWS-1 is stored in orbit in an inactive mode. The initial altitude of 260 n. mi. was tentatively selected to ensure an adequate lifetime in orbit to accomplish the longer duration second portion of the mission. This altitude will ultimately be based upon the payload-weight/booster-capability compatibility. The first manned portion of the mission (A1/A2) has a 28-day duration during which the OWS-1 is made habitable and the experiments are activated. The second manned portion of the mission (A3/A4) has a duration of 56 days during which the OWS-1 is reactivated and the solar ATM experiments are performed. Both of these mission segments are referred to as dual launches since CSM rendezvous is required to complete the assembly in orbit. The sequence of events to be performed in each segment is presented in Tables 6 and 7. The completely assembled Cluster A configuration is illustrated in Fig. 15.

Experiments and mission weight status: Experiments and experimental tasks to be conducted aboard Cluster A are identified in Table 8. The grouping of experiments into four launch payloads is dependent upon the exact definition of their requirements and the cluster consumable requirements. For the experiment groupings and requirements used in Ref. 3, the weight status for each mission presented in Table 9 applies. All missions show positive performance margins although all flights are currently above the target weight. No mission is presently weight critical.

Return requirements and return capability: Due to the large amount of data to be collected, the Apollo CM will require modification to accommodate the return payload packages. The current mission return requirements are compared against the CM return capability in Table 10. Although the data from which the return requirements were derived (volume and weight of individual return packages) may not be firm, it is expected that as many requirements for the experiments will increase as will decrease, so that the entries in Table 10 can be considered good working numbers.

The CM capability is actually an area of some flexibility requiring iteration between the experiment needs and the making use of CM spares. Considering the possible CM capability indicated on the summary table, stowage volume appears to be a greater problem than weight. The actual dimensions of return packages cannot always make efficient use of the vacated spaces. Data return packages and techniques similar to those employed by the Discoverer Program are being considered.

SM propellant requirements and propellant capability: The estimated SM propellant requirements of the A1/A2 flights for each major phase of the 28-day mission are compared against the current SM capability in Table 11. Propellants required of the secondary propulsion systems (SPS) and reaction control system (RCS) are subtotaled, and a 10 percent contingency factor is applied to yield the total propellants required. Table 11 indicates that the available tank capacity of the SPS is sufficient

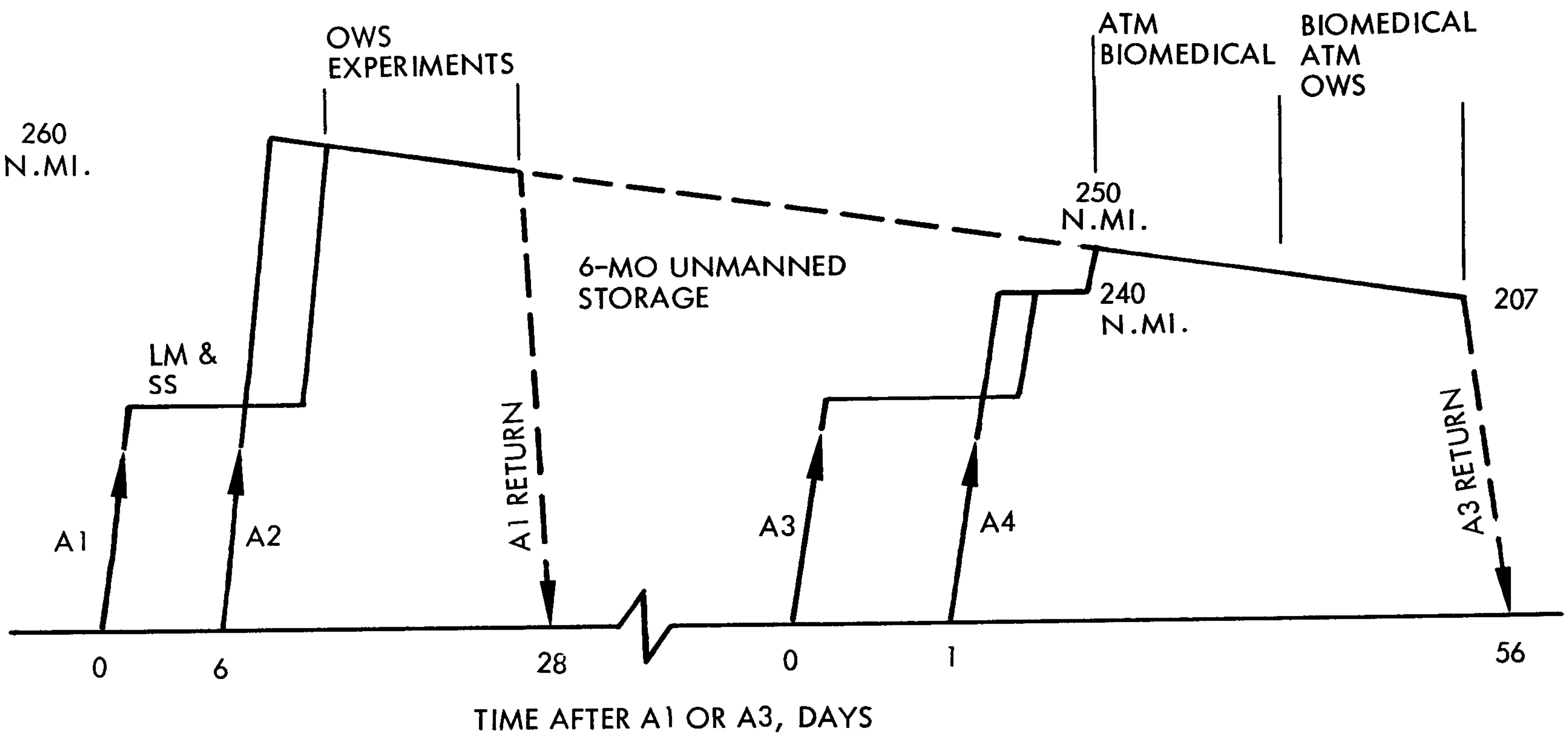


Fig. 14 Cluster A Integrated Mission Profile

TABLE 6
A1/A2 SEQUENCE OF EVENTS

Day	A1/A2 Events, Referenced From A1 Launch Day
1	Launch A1 from LC 34 at KSC. Injection orbit is 80 by 120 nautical miles. Transpose CSM and dock to LM and SS. Jettison S-IVB spent stage. Perform apogee service propulsion system (SPS) burn to circularize orbit at 120-nautical-mile altitude.
1–5	Activate and perform LM and SS and scientific experiments that are earth oriented.
6	Launch A2 from LC 37B at KSC. The A2 injection orbit is 260 by 260 nautical miles. Jettison nose cap and open SLA panels. Perform CSM terminal phase maneuvers in order to rendezvous with the A1 vehicle.
6–10	Perform maneuvers to dock the CSM and the LM and SS to the OWS. Complete the LM and SS/MDA radial docking and CSM/MDA axial docking activities and initiate the OWS passivation and habitation procedures. Deploy AM solar arrays, connect umbilicals, and pressurize the OWS. Complete LM and SS operations, deactivate experiment module, and retrieve film package via EVA.
11–25	Conduct OWS engineering experiments and biomedical experiments.
26–27	Retrieve all data packages and stow in CM for return. Transfer equipment to storage locations and deactivate OWS. Depressurize OWS and isolate pressure-sensitive equipment in AM. Orient OWS long axis toward center of earth to achieve gravity gradient stabilized mode.
28	Separate CSM from MDA. Conduct deboost, entry, and recovery operations.

TABLE 7

A3/A4 SEQUENCE OF EVENTS

Day	A3/A4 Events, Referenced for A3 Launch Day
1	Launch A3 from LC 34 at KSC. Injection orbit is 80 by 120 nautical miles. Jettison S-IVB spent stage. Perform apogee SPS burn to circularize parking orbit at 120-nautical-mile altitude in order to provide adequate lifetime while waiting for the launch of A4.
2	Launch A4 from LC 37B at KSC. Injection orbit is 240 by 240 nautical miles. Jettison nose cone and open SLA panels. Perform CSM maneuvers in order to rendezvous with the LM/ATM within the 7-hour IU lifetime of the A4 booster. Hard dock CSM and LM. Jettison S-IVB. Initiate phasing ellipse in order to adjust relative positions of OWS and the CSM/LM, as required.
3 – 8	Perform maneuvers to rendezvous the CSM/LM and OWS. Transfer two crewmen to the LM. Free-fly the LM and hard dock to a radial MDA port while the CSM stands by in a stationkeeping position. Perform CSM hard docking to the axial MDA port. Reactivate OWS, connect umbilicals, set up experiment equipment. Checkout and prepare LM for ATM operations.
9 – 22	Conduct experiment operations with ATM as first priority and biomedical as second priority.
23 – 50	Conduct experiment operations with reduced ATM emphasis and continuing biomedical observations. Conduct correlative scientific experiments primarily during this period.
51 – 53	Continue experiment operations with primary emphasis on biomedical data. During this period, perform OWS experiments to be repeated from A1/A2.
54 – 55	Stow all data packages to be returned in CM. Transfer equipment to unmanned storage locations and deactivate OWS. Orient OWS in gravity gradient attitude.
56	Separate CSM from MDA. Conduct deboost, entry, and recovery operations.

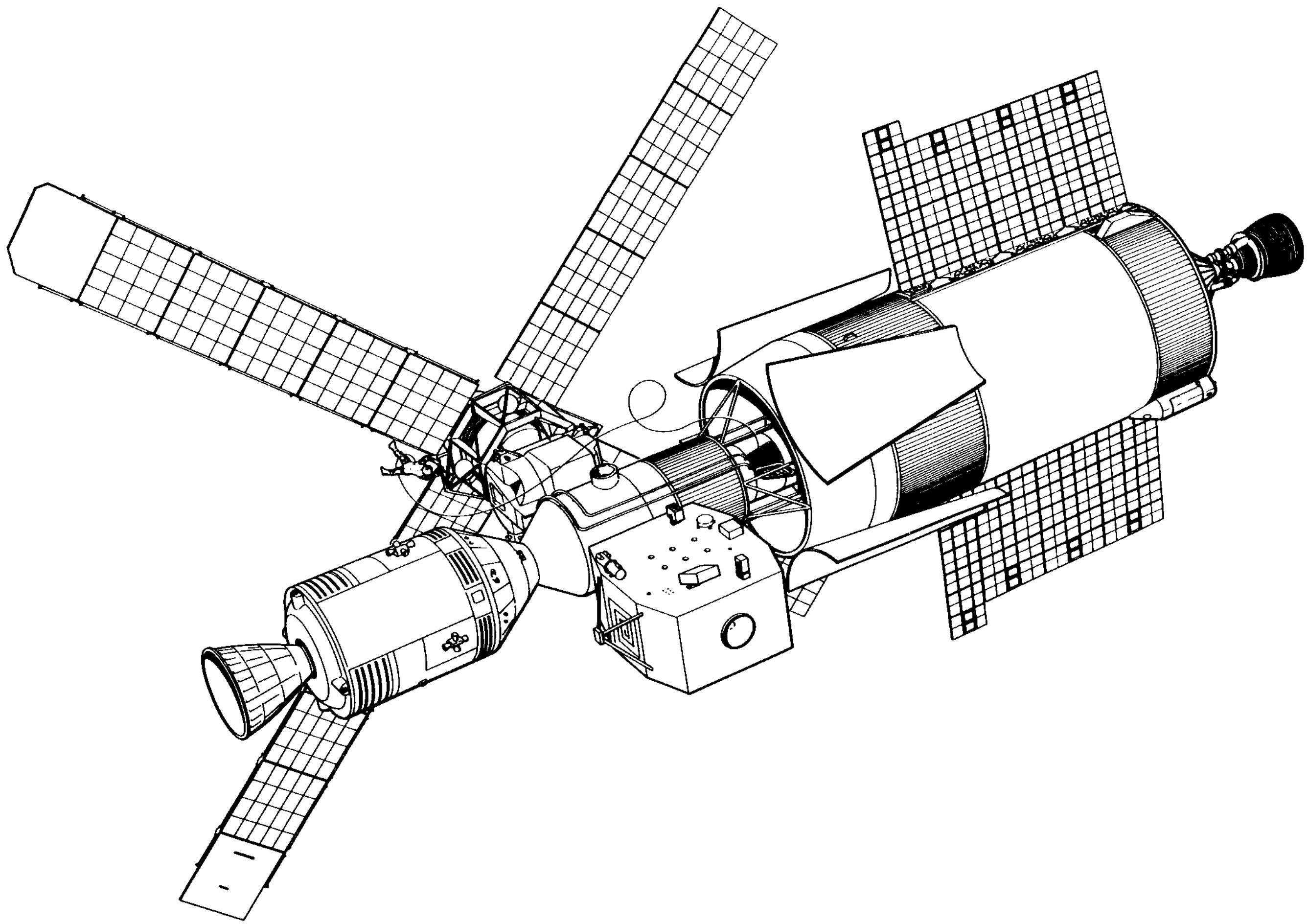


Fig. 15 Cluster A Configuration

TABLE 8

CLUSTER A PLANNED EXPERIMENTS

Exp. No.	Experiment Name
S052	White-Light Coronagraph High-Altitude Observatory
S053	UV Spectro-Heliograph; EUV Spectrograph; EUV Telescope/ Naval Research Laboratory
S054	X-Ray Spectro-Heliometer
S055	UV Spectro-Heliometer; UV Spectrometer; Hydrogen-Alpha Telescope/Harvard College Observatory
S056	X-Ray/EUV Telescope/Goddard Space Flight Center
----	Lunar Mapping and Surface Survey
D017	Carbon Dioxide Reduction
D018	Integrated Maintenance
D019	Suit Donning and Sleep Station Evaluation
D020	Alternate Refraints Evaluations
D021	Expandable Airlock
D022	Expandable Structure for Recovery (Operation)
M018	In-Flight Vectorcardiogram
M050	Metabolic Cost of In-Flight Tasks
M051	In-Flight Assessment of Cardiovascular Function
M052A	Bone and Muscle Changes
M052B	Bone and Muscle Changes
M053	Human Vestibular Function
M054	Neurological Study
M055	Time and Motion Study
M402	Spent Stage Habitability
M439	Star Horizon Automatic Tracking
M466	Spent Suit Evaluation

TABLE 8 (Cont.)

Exp. No.	Experiment Name
M469	Removal and Retrieval of the Stabilizer Platform ST-124 and Related Components
M479	Zero-g Fammability
M486	Astronaut EVA Hardware Evaluation
M487	Spent Stage Habitability Evaluation
M488	Zero-g High Pressure Gas Expulsion
M489	Zero-g Heat Exchanger Service, Water Wicking, and Boiling
M492	Joining Tubular Assemblies in a Space Environment
M493	Self-Contained Electron Beam Welder
S005	Synoptic Terrain Photography
S006	Synoptic Weather Photography
S009	Investigation of Cosmic Radiation Using Nuclear Emulsions
S015	Effects of Zero Gravity on Single Human Cells
S017	X-Ray Astronomy
S018	Micrometeorite Collection
S019	Ultra-Violet Stellar Astronomy
S020	Ultra-Violet X-Ray Solar Astronomy
S027	Galactic X-Ray Mapping
S061	Potato Respiration
S063	Ultra-Violet Airglow Horizon Photography
S065	Multiband Terrain Photography
T002	Manual Navigation Sightings
T004	Frog Otolith Function
T017	Meteoroid Impact and Erosion
T021	Meteoroid Penetration Flux-Velocity
T020	Jet Shoes for EVA Space Locomotion
T022	Heat Pipe Experiment
T023	Surface Absorbed Material Collection for Analysis

TABLE 9

CLUSTER A MISSION WEIGHT STATUS
(Basic Mission, Flights A1–A4)

Flight	Launch Vehicle	Injection Altitude (n. mi.)	Target Weight (lb)	Current Weight (lb)	Weight Capability (lb)	Weight Margin (lb)
A1	207	81 × 120	38,000	38,951	40,300	+1,349
A2	209	260 × 260	25,000	25,762	27,900	+2,138
A3	211	81 × 120	38,000	38,852	40,500	+1,648
A4	210	240 × 240	28,300	28,614	29,800	+1,126

TABLE 10

CLUSTER A MISSION RETURN REQUIREMENTS AND RETURN CAPABILITY

Mission	Return Requirements		Possible CM Return Capability*		Margin	
	Weight (lb)	Volume (ft^3)	Weight (lb)	Volume (ft^3)	Weight (lb)	Volume (ft^3)
A1	164.0	4.76				
A2	241.0	16.215				
Total	405.0	20.975	533.0	19.0	+128	-1.975
A3	111.0	3.953				
A4	402.8	18.967				
Total	513.8	22.920	533.0	19.0	+ 20	-3.920

*Basic CM return capability is 188 lb and 7.64 ft^3

TABLE 11

A1/A2 PROPELLANT REQUIREMENTS OF THE SM

Mission Event	SM SPS (lb)	SM RCS (lb)	
Initial Orbit Operations			132
Separation/Docking		62	
Orbit Adjustment		51	
Circularization	238	19	
Rendezvous and Docking			677
Preterminal Phase Initiation (TPI)	1,506	152	
TPI and Braking		318	
Stationkeeping/Docking		207	
Orbit Operations			348
Orbiting OWS (15 days)		213	
LM and SS (14 days)		135	
Deorbit	1,085		47
Subtotal	2,829		1,204
Contingencies (10 percent)	280		120
TOTAL REQUIREMENT	3,019		1,324
A1 CSM CAPABILITY	3,019		1,224
APPARENT MARGIN	+3%		-8%

(+ 3 percent margin), but that the current RCS design does not provide the required tank capacity (-8 percent margin). A reduced experiment activity level may be considered to eliminate the negative margin.

The estimated SM propellant requirements of the A3/A4 flights for each major phase of the 56-day mission are compared against the current SM capability in Table 12. The table indicates that the SPS capability is just adequate, but that the RCS capability is about one half the established requirement. Major modifications to the RCS storage and feed system will be necessary to eliminate the negative margin.

Data handling requirements and readout time available: The data handling requirements of the Cluster A experiments are compared against the readout time available in Fig. 16 . The minutes/day of data to be acquired was obtained by a summation of the individual data requirements in Ref. 3. The ground-station readout time available was previously reported (Ref. 5). As indicated in Fig. 16 , the data acquired in minutes of recording exceeds the readout time available during the A1/A2 and A3/A4 experiment phases. The A1/A2 data excess can be handled by using a read/record ratio greater than 1 and/or providing additional onboard storage to hold the data for later transmission. It appears that the A3/A4 excess can best be handled only by a reduction in the experiment data handling requirements.

Cluster A resupply and alternate missions. – The Cluster A resupply/ alternate missions, if actually flown, would consist of four uprated Saturn IB carrying the following payload modules:

Flight	Vehicle	Payload Module (excluding SLA)
A5	212	XCSM
A6	213	XCSM
A7	214	CM-A, Rack/EMR
A8	215	XCSM

The four flights are scheduled for planning purposes in 2nd and 3rd quarter 1969, as indicated in Table 5 . Utilizing the A5, and A6 periodic resupply vehicles and crew rotation, the Cluster A mission duration could be extended to approximately 1 year. Flights A7 and A8 are alternate flights to be used in contingency situations to achieve the Cluster A basic mission objectives. The degree of certainty that the resupply and alternate flights will be used is unknown. However, certainty is believed to be less than that which might be associated with the flights which comprise the basic mission (A1 through A4).

Experiments to be carried to the Cluster A orbit by the resupply and alternate vehicles have not been identified. The weight status of the individual vehicles and their experiment requirements for payload return, SM propellant, and data handling are therefore unknown at this time.

TABLE 12

A3/A4 PROPELLANT REQUIREMENTS OF THE SM

Mission Event	SM SPS (lb)	SM RCS (lb)	
Initial Orbit Operations			142
Separation	192	28	
Circularization		19	
Orbit Adjustments		24	
Stationkeeping		71	
First Rendezvous (A3 with A4)			545
Pre TPI	1,035	186	
TPI and Braking		257	
Stationkeeping/Docking		102	
Second Rendezvous (CSM/LM with OWS)			846
Pre TPI	715	280	
TPI and Braking		485	
Stationkeeping/Docking		81	
Orbit Operations			
Cluster			565
ATM		44	
Other Experiments		516	
Stationkeeping		5	
Independent LM	(Uses LM RCS)		
Deactivate/Deorbit			52
Reentry	1,017	19	
Separation		24	
Stationkeeping		9	
Subtotal	2,959		2,150
Contingencies (10 percent)	300		215
TOTAL REQUIREMENT	3,259		2,365
A3 XCSM CAPABILITY	3,259		1,224
APPARENT MARGIN	0%		-93%

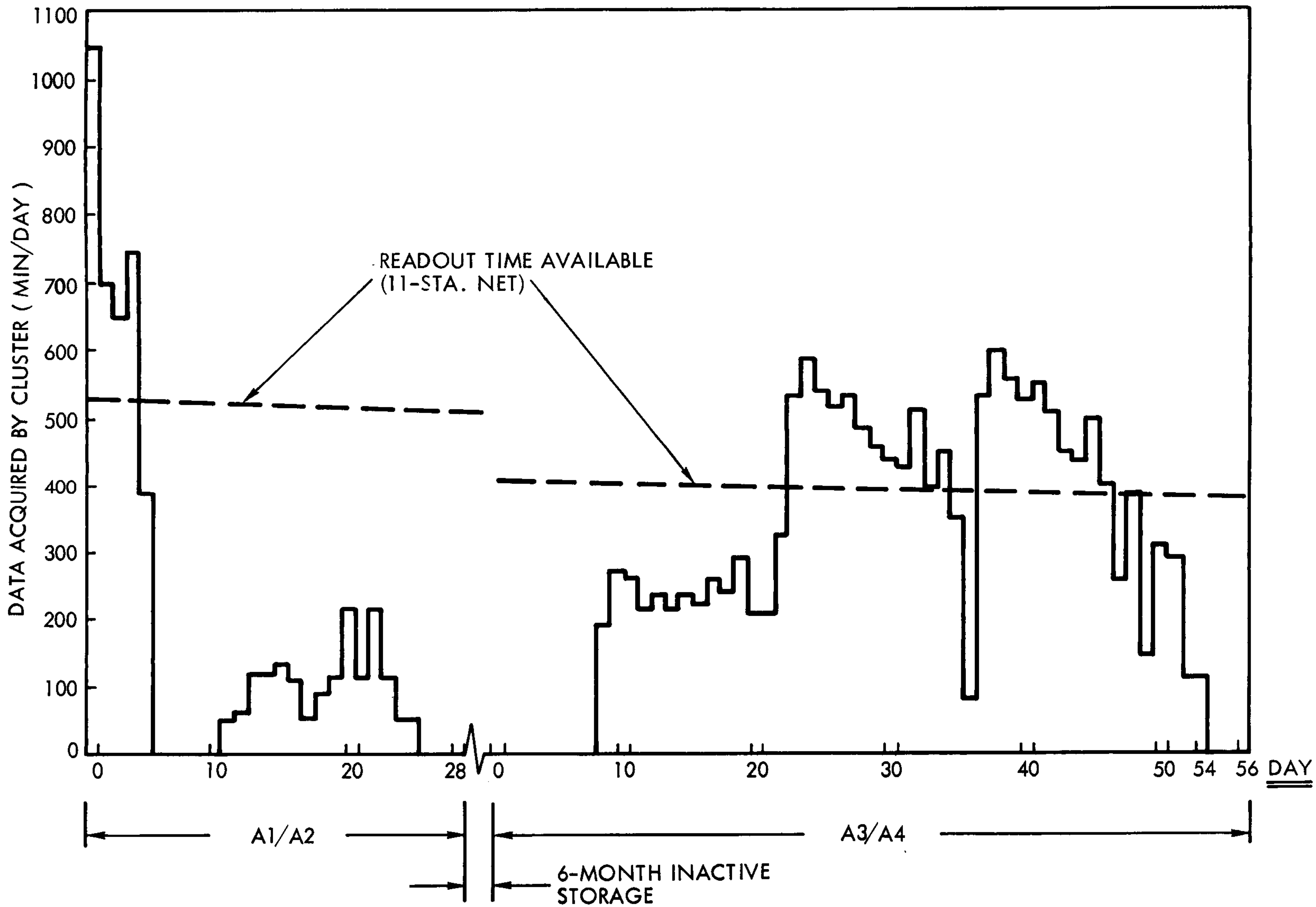


Fig. 16 Cluster A Experiment Data Handling

Cluster B basic and resupply missions. – The Cluster B basic and resupply missions consist of six uprated Saturn IB flights, each carrying the following modules:

Flight	Vehicle	Payload Module (excluding SLA)
B1	216	CSM
B2	217	OWS-2, AM, MDA, Nose Cone
B3	218	XCSM
B4	219	LM-A, Rack/ATM-B, Nose Cone
B5	220	XCSM
B6	221	XCSM or LCSM

These flights are scheduled for launch beginning in 4th quarter 1970. Their individual payload modules are essentially the same as carried by Flights A1 through A6, except for the deletion of the LM & SS. As indicated in the above tabulation, the specific payload aboard B6 has not been selected.

The Cluster B integrated mission profile has not been established. The experiments to be launched by the cluster mission are identified but have not been grouped into specific vehicles. The weight status of the individual vehicles and their experiment requirements for payload return, SM propellant, and data handling are therefore unknown.

Definition of Optimum Mission Profile

The essential parameters involved in the definition of an optimum mission profile for the OPE spacecraft are:

- Initial Orbit Altitude
- Orbit Inclination Angle

The assessment of these parameters, individually and in combination, is presented in the following paragraphs to yield:

- In-orbit lifetime of candidate spacecraft designs as a function of initial orbit altitude
- For the minimum lifetime allowed (six months), the final orbit altitude of candidate spacecraft designs as a function of initial orbit altitude
- The total velocity required of the CSM to rendezvous with the spacecraft at the end of the in-orbit phase
- Significance of orbit inclination angle with respect to ETR range safety constraints and orbit regression
- Ground-station contact times for a specific altitude/inclination combination
- Shift in ground traces as a function of altitude and inclination angle
- Effect of ground stations used on readout times available
- Maximum percentage time in sunlight as a function of altitude/inclination combinations

The ranges of the variables which are treated are:

- Orbit Altitude: 140 to 260 nautical miles
- Orbit Inclination Angle: 28.5 to 50 degrees
- MSFN Ground Stations: 11- and 4-station networks
- In-Orbit Lifetime: 6 to 12 months

All information except the selected mission profiles are presented parametrically.

In-orbit lifetime: OPE spacecraft orbit decay predictions were computed based upon the altitude-density model in Ref. 7, using the computer program of Ref. 8. The density model accounts for the diurnal bulge, semi-annual and annual fluctuations, the 11-year cycle of solar activity, and the correlation with the geomagnetic index. The ballistic parameter, $B=C_dA/2M$, was determined by the mass and external configurations of candidate spacecraft designs and their attitude history relative to the velocity vector. For the candidate spacecraft designs presented in Fig. 17 , the values of ballistic parameter tabulated in Table 13 were computed. All computations used free molecule flow theory to estimate drag coefficient (C_d). Spacecraft weights in the range from 4000 to 5300 lb were assumed.

Figure 18 shows the orbit altitude decay from an initial altitude of 260 n. mi., for four values of B (ballistic parameter). The range of B covers the range of all of the candidate spacecraft designs. As indicated on the figure, the initial conditions assumed includes an orbit inclination angle of 29 deg. The figure shows that no candidate spacecraft will decay to less than 220 n. mi. in six months. After 12 months, however, the orbit altitude reached is highly sensitive to values of B greater than roughly 1. Spacecraft configuration "ID" (illustrated in Fig. 17) , having a maximum B of 1.70, would have decayed to roughly 160 n. mi. after 12 months on orbit.

Crossplots of Fig. 18 are presented in Fig. 19 to show better the effect of B on the orbit decay from 260 n. mi. For the 12-months-of-decay curve, the sensitivity of orbit altitude with B increases abruptly as B is increased above roughly 1.1. As indicated in Table 13, high values of B are associated with spacecraft configurations having solar paddles deployed. Higher values of B are due to increased values of drag from the deployed paddles. Again, the value of orbit inclination angle is 29 deg.

The data of Fig. 18 are shown differently in Fig. 20 . This figure highlights the orbit altitude after six months of decay. As indicated on the figure, if the final orbit altitude is arbitrarily fixed at 200 n. mi., the initial orbit altitudes which would be required would vary between 226 and 244 n. mi. for six months on orbit.

Allowed range of orbit inclination angle: The range of orbit inclination angle is limited by Range Safety constraints against direct overflights of land masses and flights so close to land masses that they cannot be protected from land impacts by standard range safety equipment. The allowed range is illustrated in Fig. 21. As indicated on the figure, an orbit inclination angle of 28.5 deg corresponds to a

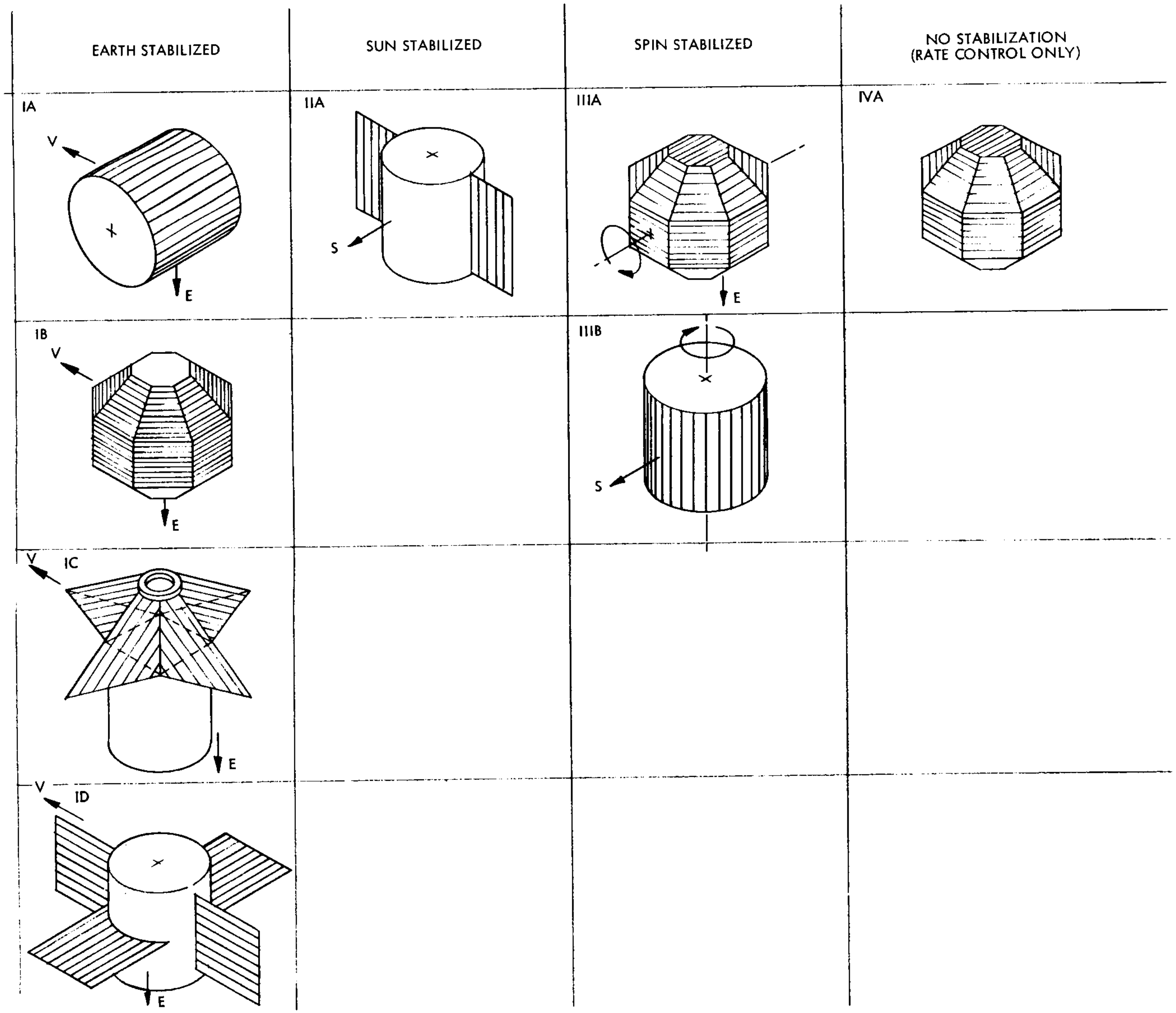


Fig. 17 Candidate OPE Spacecraft Configurations

TABLE 13

VALUES OF BALLISTIC PARAMETER FOR THE CANDIDATE OPE SPACECRAFT CONFIGURATIONS

Configuration Code No.	Ballistic Parameter, B, (ft^2/slug)		
	Maximum	Minimum	Nominal
IA	1.68	1.27	1.48
*IA'	0.77	0.46	0.62
IB	0.95	0.72	0.84
*IB'	0.77	0.46	0.62
ID	1.70	1.12	1.41
IIA	1.16	0.88	1.02
IIIA	0.66	0.5	0.58
IIIB	0.67	0.51	0.59

*Reduced size versions.

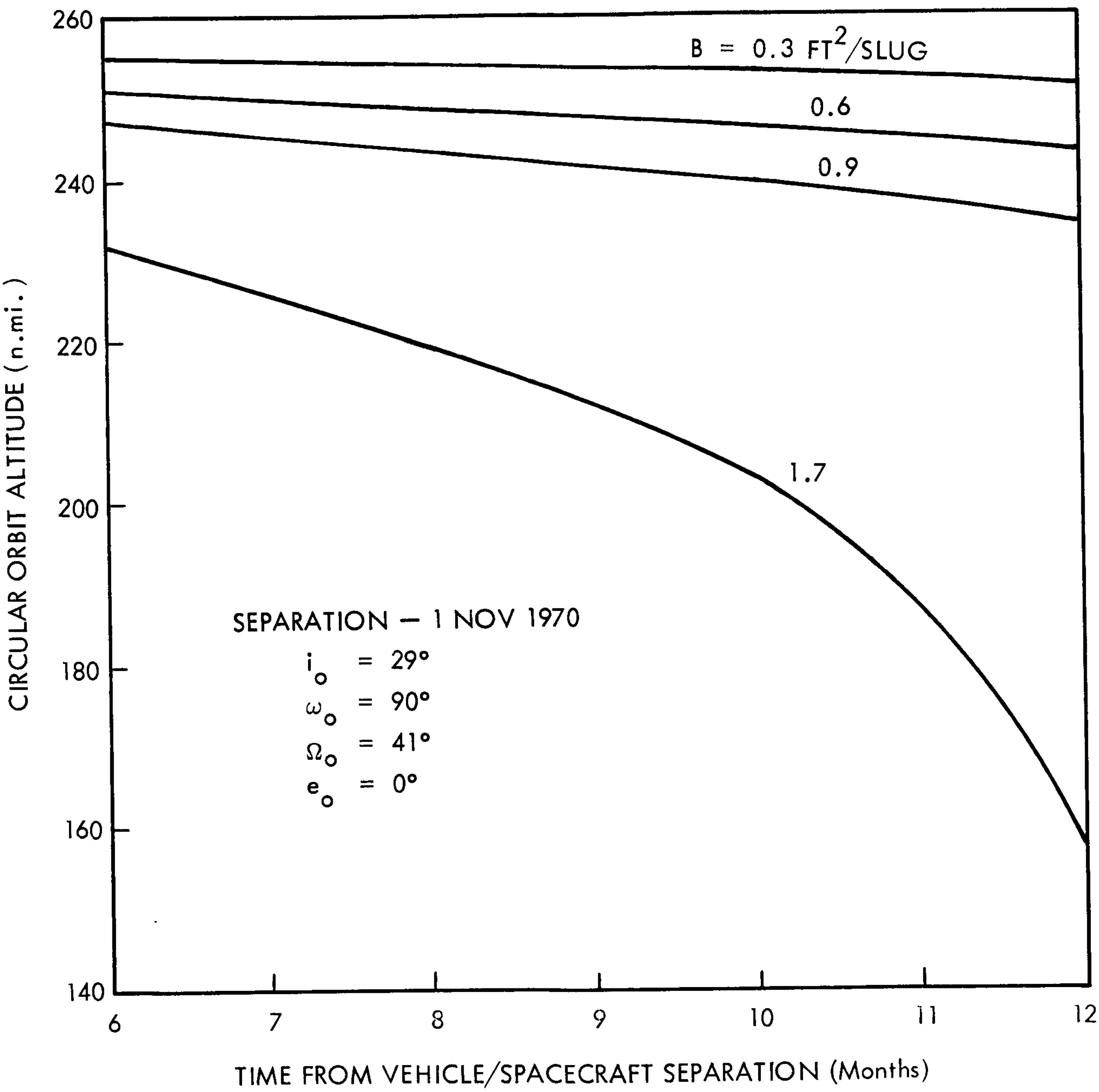


Fig. 18 Orbit Altitude Decay from 260 N. Mi.

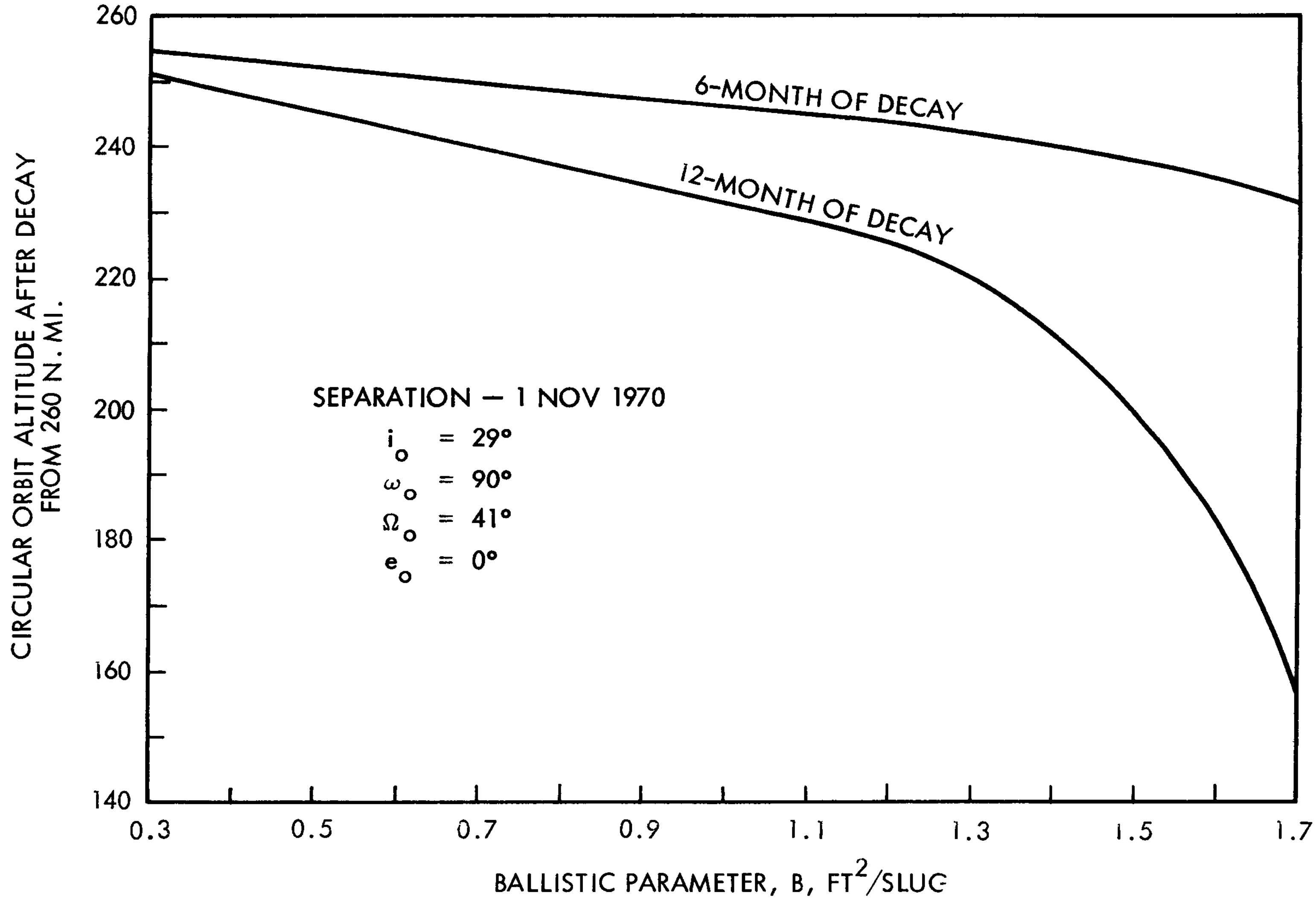


Fig. 19 Orbit Decay vs. Ballistic Parameter from 260 N. Mi.

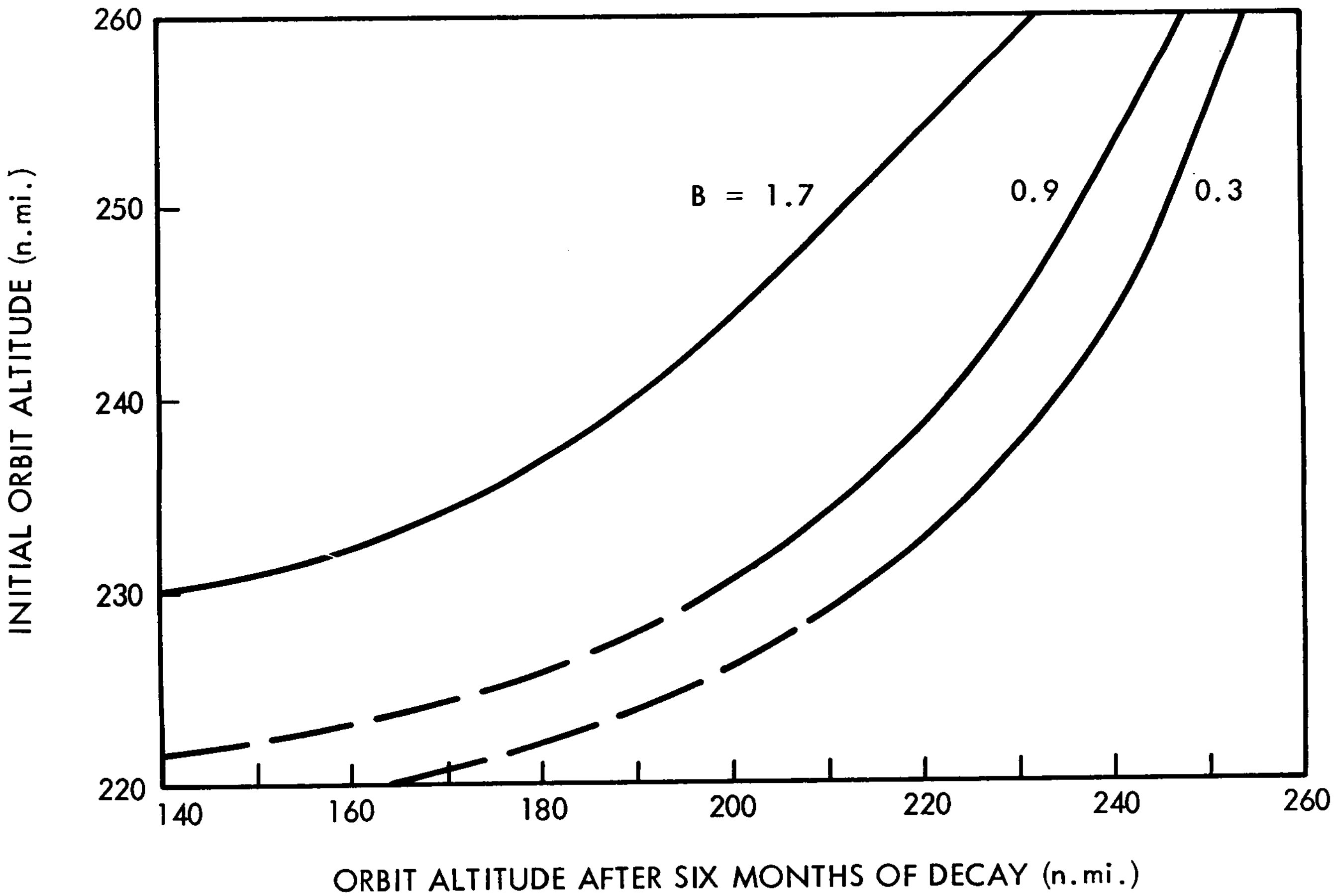


Fig. 20 Variation of Initial (With Final) Orbit Altitude

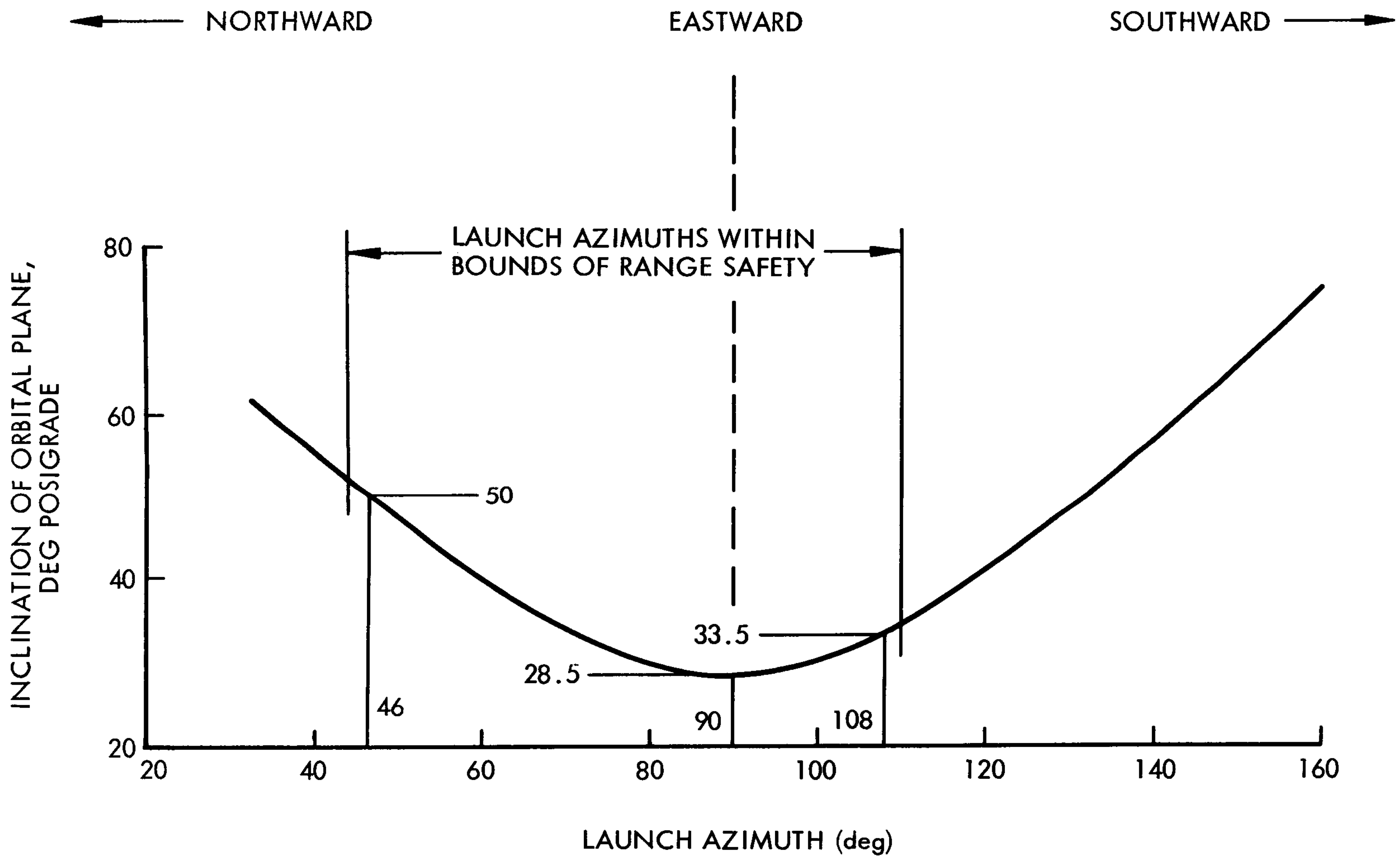


Fig. 21 Launch Azimuths Within Bounds of Range Safety

due-east launch (azimuth angle = 90 deg) from ETR. Northeasterly launches having orbital inclinations up to about 52 deg and southeasterly launches having orbital inclinations up to about 34 deg are permitted. The value of inclination angle to be used in the AAP Cluster A flights (29 deg) will take maximum advantage of the inertial velocity of earth's rotation.

The three points that are flagged on the curve in Fig. 21 are commonly used in mission planning. The 50-deg-inclination, 46-deg-azimuth point represents the maximum departure from a due-east launch which is commonly used for planning purposes. This combination of inclination/azimuth angles is currently (5 June 1967) being considered for the Cluster B 1969 flights. The 33.5/108-deg combination is planned for NASA-Ames flights of the Biosatellite.

Orbit regression and period: The variations of the nodal regression rate and orbital period with altitude are shown in Fig. 22 for the 29-deg-inclination case. The figure shows that the nodal regression rate of the orbit plane due to the earth's oblateness is 0.442 deg/rev westward for a 260 n. mi. orbit (with no drag) with a 29-deg-inclination. Having a period of 94.15 minutes (shown in figure), a spacecraft will make 15.3 orbits every 24 hours. Over a period of one year, the total nodal regression of the orbit plane is approximately 2,470 deg or 6.85 complete revolutions westward. Since the sun will have made one complete revolution eastward over this same time period, the orbit plane in the course of one year makes 7.85 complete revolutions relative to the sun. This final result does not account for orbit decay from 260 n. mi. due to drag.

Revolutions of orbit plane relative to the sun, OPE spacecraft and cluster: Figure 23 combines the orbit decay data from Fig. 19 with the orbit regression and period data from Fig. 22. For the initial conditions of a 260 n. mi. altitude and a 29-deg inclination, the following data may be drawn from Fig. 23 :

- After 12 months of decay, OPE spacecraft configuration IIA (see Fig. 17 for sketch) will reach an orbit altitude of 227 n. mi., at which time 7.95 revolutions of the orbit plane relative to the sun will have been completed.

- Cluster A, at 207 n. mi. after 12 months of decay (according to Fig. 23) will have completed 8.03 revolutions of its orbit plane relative to the sun.

- If it is assumed that the Cluster B ballistic parameter is approximately the same as that for Cluster A (1.43), OPE spacecraft IIA separated from Cluster B could be approximately 29 deg out of plane with the cluster at the end of its 12-month orbital mission.

For such large plane separations, the total energy required of the CSM to rendezvous with the spacecraft would be very high in comparison with the CSM's total propulsion capability. Configuration IIA was also used in this example because it was typical of the geometric approach to which design considerations were leading.

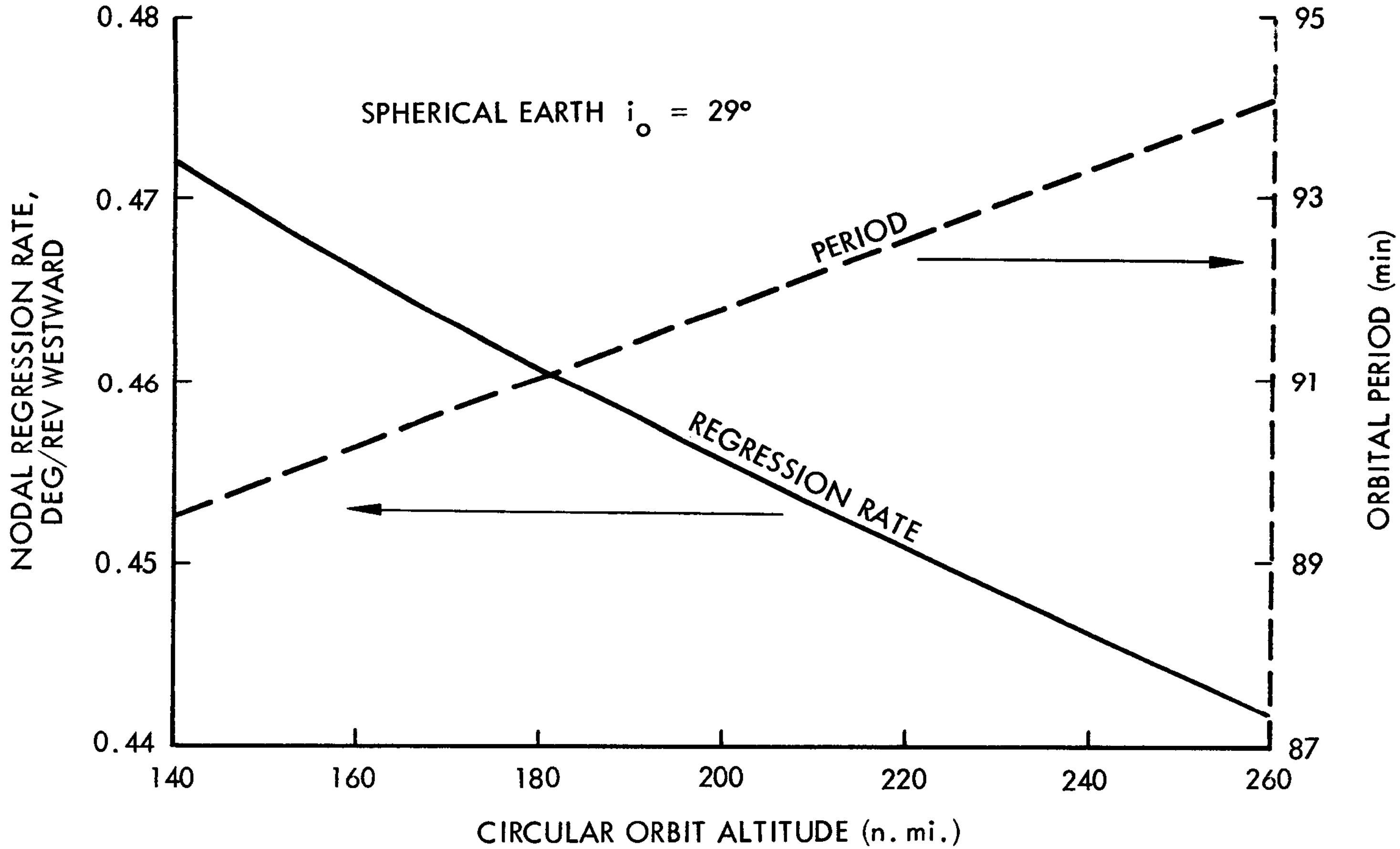


Fig. 22 Variation of Regression Rate and Period with Altitude

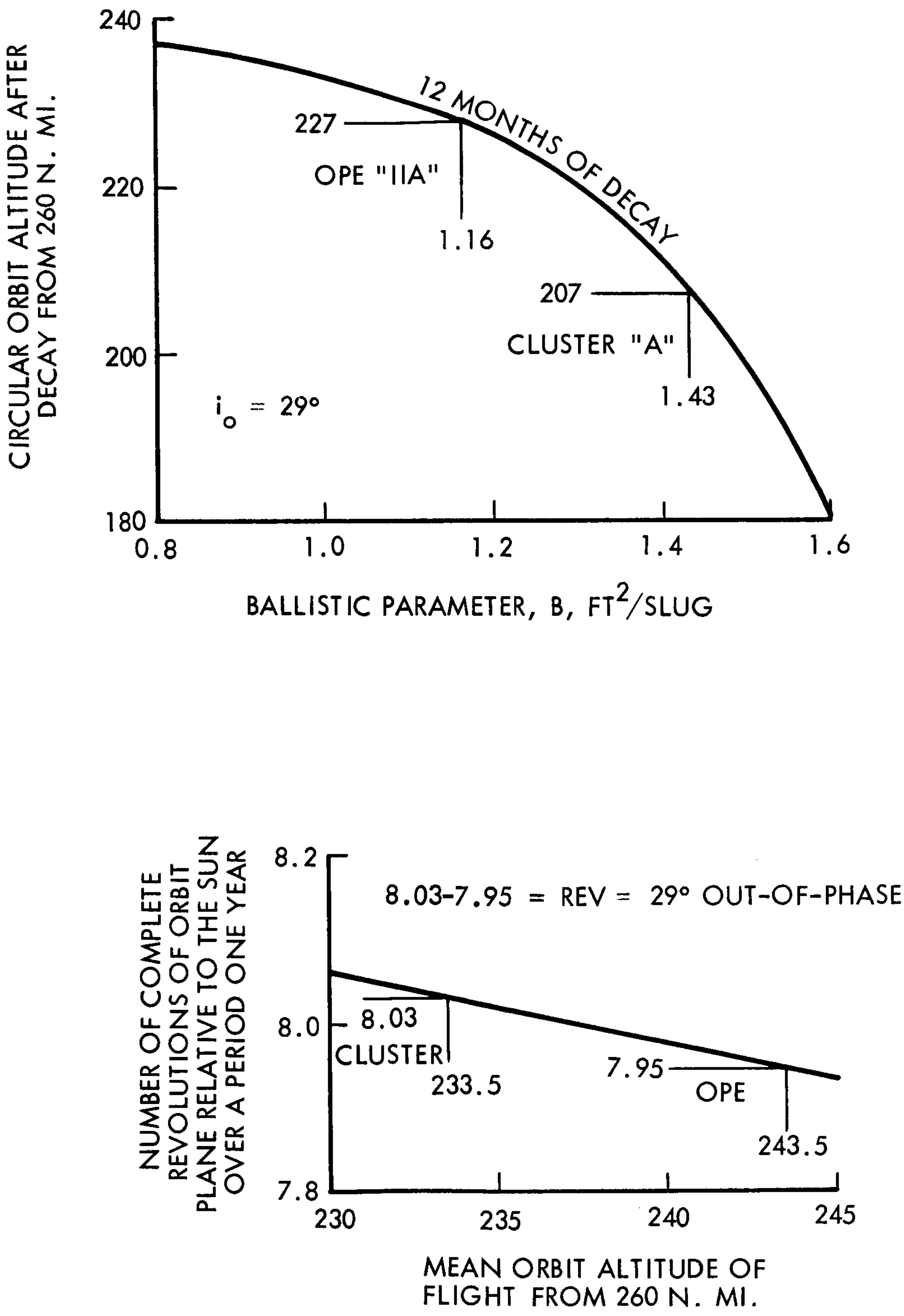


Fig. 23 Separation Between Spacecraft and Cluster Orbit Planes

Ground station acquisition: Radar acquisitions of the OPE spacecraft were computed on the basis of a 5-deg look angle (5 deg up from horizon) using the Manned Space Flight Network (MSFN) stations. All tracking times less than one minute were ignored.

The orbital ground trace plot for one day in orbit at 260 n. mi with a 29-deg inclination is presented in Fig. 24 . Superimposed are the radar acquisition ranges (circles, approximating actual shapes) of 11 of the 14 MSFN stations. The three Unified S-Band (USB) stations (Madrid, Canberra, and Goldstone) which use the 85-ft antennas, usually limited to deep space missions only, are assumed to be deactivated. Radar acquisitions by the 11 ground stations during the individual orbits which comprise the first 24-hr period are subtotaled in Table 14 . As indicated in this table, a grand total of 530.7 min of readout time is available. The station offering the greatest contact time is Hawaii (56.6 min).

Figure 25 illustrates the reduction in acquisition times available if the number of stations used were reduced from eleven to four. The activation of only four stations (Guaymas, Corpus Christi, Cape Kennedy, and Carnarvon) has been considered by the AAP for portions of the AAP mission in which the cluster is essentially in storage in an inactive mode. As indicated in Fig. 25 , activation of only the four stations would reduce the available readout time per orbit pass by about 50 percent. It may be inferred that during the initial months in orbit, track times of at least 40 min/day would be available.

Variation of acquisition time with altitude: Orbit decay will effect a reduction in the total readout time available, as shown in Fig. 26. An orbit decay from 260 to 200 n. mi. is associated with a 15 percent reduction in the min/day of readout time available. An OPE requirement for roughly 35 min/day of ground contact time, if actually required, would represent about 10 percent of the readout time available at 200 n. mi. For the four-station case, this OPE requirement would represent roughly 20 percent of the readout time available at 200 n. mi.

Longitudinal shift between ground traces: It is useful to know how various day-to-day ground traces are situated with respect to one another for a given mean orbital altitude. The longitudinal spacing of one ground trace K with respect to another K+N is shown in Fig. 27 for the altitude range of interest. The plot uses a 33.5-deg-inclination initial condition assumption, although it is representative of a wide range of (posigrade) values of inclination as will be seen later. The 33.5/108 inclination/azimuth-angle combination is allowed by range safety as indicated in Fig. 21.

Figure 27 is interpreted by entering at a selected altitude, then reading off the degrees of earth longitude that the K+N trace is located either east or west of the Kth trace. For instance, at an altitude of 260 n. mi., the Kth and K+ 15th trace are seen to superimpose at the same earth longitude. One orbit later (K+ 16th trace) the orbit is seen to shift approximately 24 deg of earth longitude to the west. One orbit earlier (K+ 14th trace) the shift is also 24-deg, but this time to the east. The orbit regression (degrees of earth longitude shift per orbit) is, therefore, 24 deg/orbit for the values of altitude and inclination angle selected. Such plots are useful in the determination of orbital altitudes that will result in optimized rendezvous and recovery opportunities.

Orbital Ground Traces

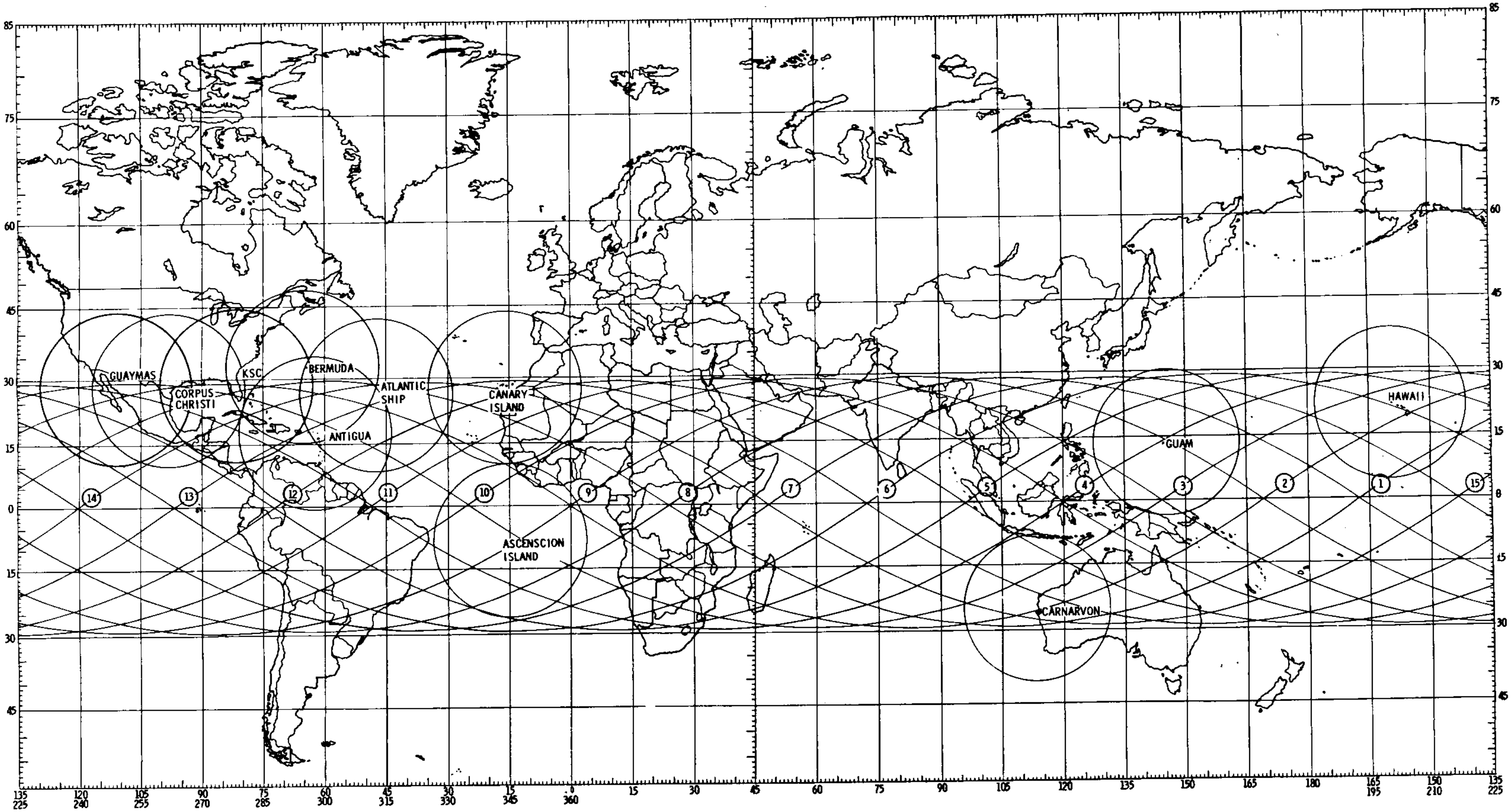


Fig. 24 Orbital Ground Traces, First Day in 260-N.Mi. Orbit at 29-Deg Inclination

TABLE 14

GROUND CONTACT SCHEDULE

Orbit No.	Hawaii	Guaymas	Corpus Christi	KSC	Bermuda	Antigua	Atlantic Ship	Canary Islands	Ascension Island	Carnarvon	Guam	Total
1	6.0	8.3	8.0	8.1	8.0	7.4	8.9	4.4	4.4	8.6		72.1
2	8.6	8.5	8.6	9.0	7.4	9.0	7.1		8.7	8.1	[1.0	76.0
3	8.8	9.0	8.6	7.4		8.1			8.6		5.5]	56.0
4	8.3	8.5	6.8								8.8	32.4
5	8.8	3.4									8.3	20.5
6	8.9										3.9	12.8
7	7.2								6.6		4.4	18.2
8									8.6		7.3	15.9
9									2.2		8.6	10.8
10									4.2		6.3	10.5
11								7.8				7.8
12						8.2	7.2	8.9		6.3		30.6
13				3.7	5.8	9.0	8.8	8.9		8.6		44.8
14			6.3	7.8	7.9	7.3	8.8	9.0		8.6		55.7
15		7.8	8.6	9.0	8.9	6.6	8.7	8.6		8.4		66.6
Total	56.6	45.5	46.9	45.0	38.0	55.6	49.5	47.6	43.3	48.6	54.1	530.7

Look Angle = 5 deg
Circular Altitude = 260 nm
Inclination = 29 deg
[ ] Indicates one pass
All entries in minutes

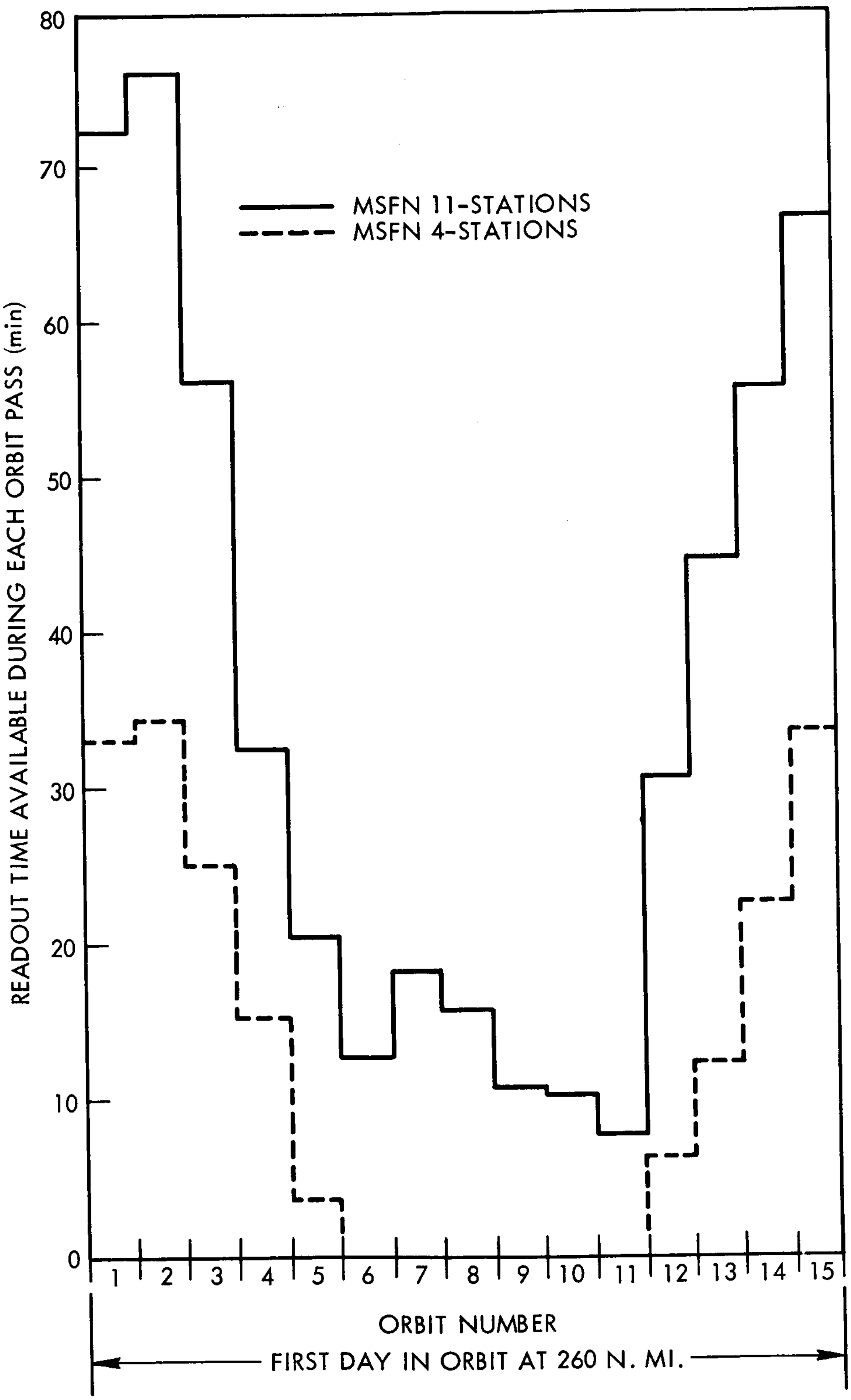


Fig. 25 Variation of Track Times with MSFN Stations Used

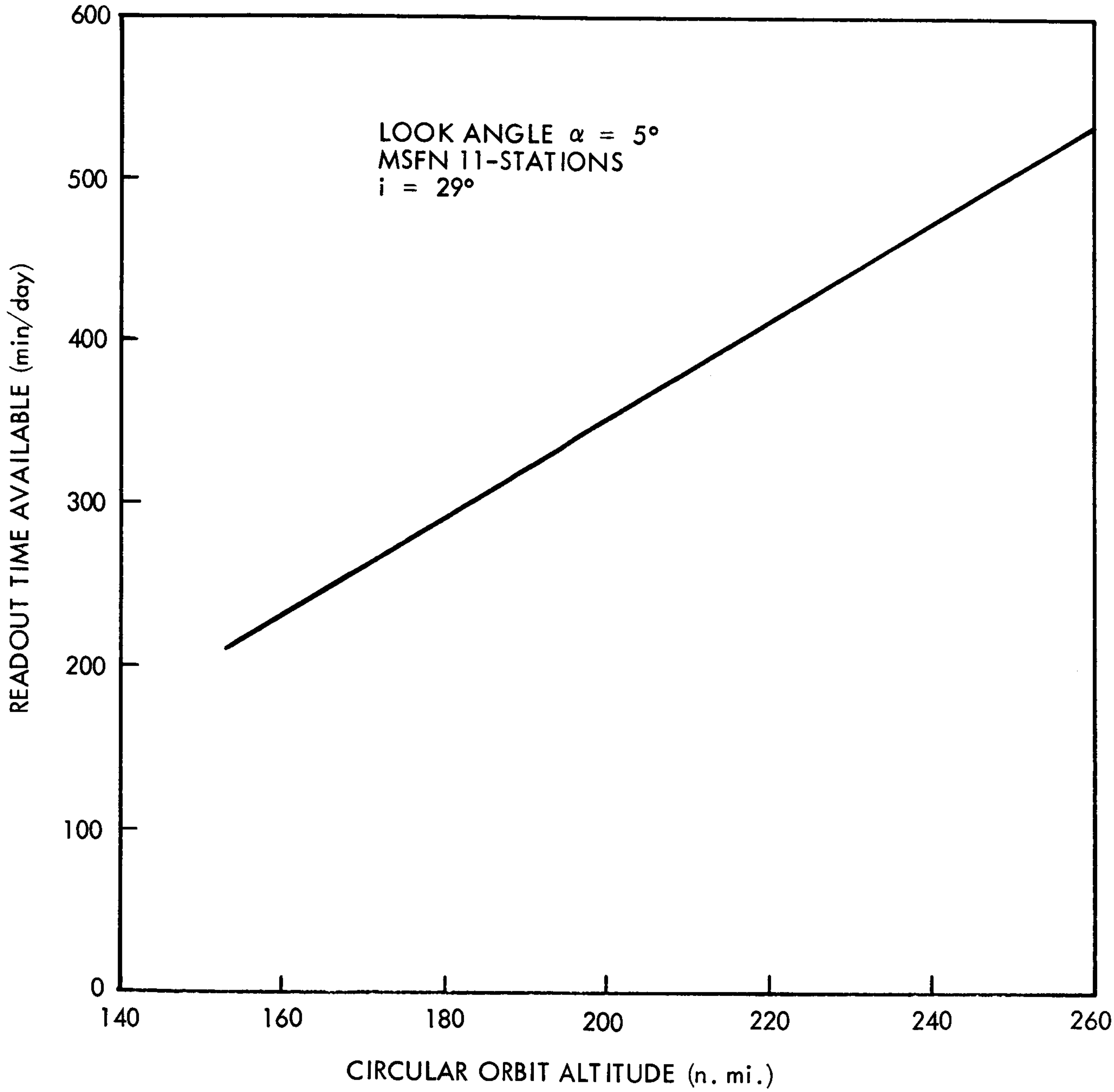


Fig. 26 Variation of Acquisition Time with Altitude

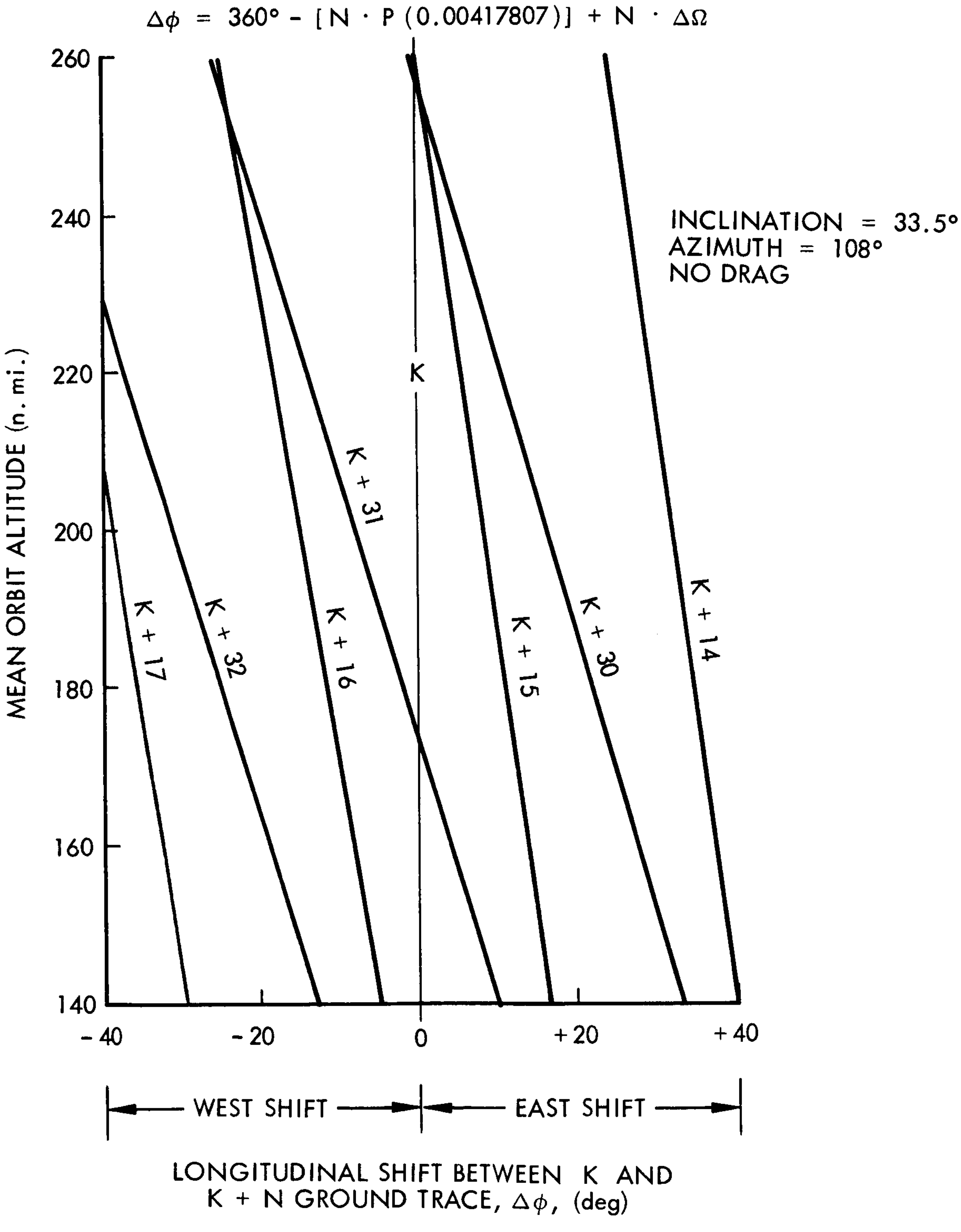


Fig. 27 Longitudinal Spacing of Ground Traces

Inclination angle effect on longitudinal shift: Figure 28 presents the variation of orbit regression (longitudinal shift) with orbital period (or altitude) for three values of inclination angle. The figure additionally shows an approximately constant effect of inclination angle upon the orbit regression (less than 0.2 deg/orbit for the full range of inclination plotted).

The longitudinal shift, or orbit regression, is better illustrated in Fig. 29. The upper plot shows the longitudinal shift of the orbital plane to the west in degrees per revolution (of the spacecraft) about the earth. It is seen that a change in altitude from 250 to 150 n.mi. causes an increase in regression of about 0.025 deg/rev at any inclination. The lower plot shows the accumulated effects of the longitudinal shift after 15 revolutions (1 day) and 31 revolutions (2 days). It is seen that after 31 revolutions, the orbital plane could rotate westward almost 15 deg, at 30-deg inclination and 200 n.mi. This means that on revolution 1, if the spacecraft ground trace passed through the equator at a local earth time of t_o, 31 revolutions later the spacecraft ground trace would pass through the equator at a local earth time of t_o minus 1 hr.

Coplanar transfer between circular orbits: CSM propulsion will be required at the end of the OPE in-orbit phase to execute CSM/spacecraft rendezvous to retrieve the primate canisters. The velocity which must be imparted to the CSM depends upon its orbital position relative to the spacecraft. A transfer in the common orbit plane (coplanar transfer) is required to reduce the altitude and phase differential between the two vehicles to zero.

The Hohmann transfer ellipse has long been recognized as the minimum energy two-pulse transfer between coplanar circular orbits. The total velocity which must be imparted to the CSM, if initially at 200 n.mi., to rendezvous with the spacecraft at altitudes ranging up to 260 n.mi., is presented in Fig. 30. It is seen that only about 65 ft/sec must be imparted for each altitude difference of 20 n.mi.

The total velocity required for a coplanar transfer between any two circular orbit altitudes is presented in Fig. 31. In this figure, r equals earth radius plus altitude, and the subscripts 1 and 2 indicate initial and final conditions, respectively. Earth radius is approximately 3,444 n.mi. Values of the initial circular orbit velocity (U_1) are presented in Fig. 32. These two figures comprise the basic data required to yield the total velocity requirement for the general case.

Noncoplanar transfer between circular orbits: The special case for a single-impulse transfer from 200 n.mi. to an orbit plane divergent with respect to the initial plane by β deg, is presented in Fig. 33 for a 29-deg inclination. The figure shows that for large longitudinal separations ($\Delta\lambda$), large increments of velocity must be imparted to the CSM.

Such large expenditures of velocity must be avoided. Fortunately, there are several feasible approaches for accomplishing this. Separation of the two-orbit planes is a function of the respective altitudes of the two vehicles. This, in turn, is a function

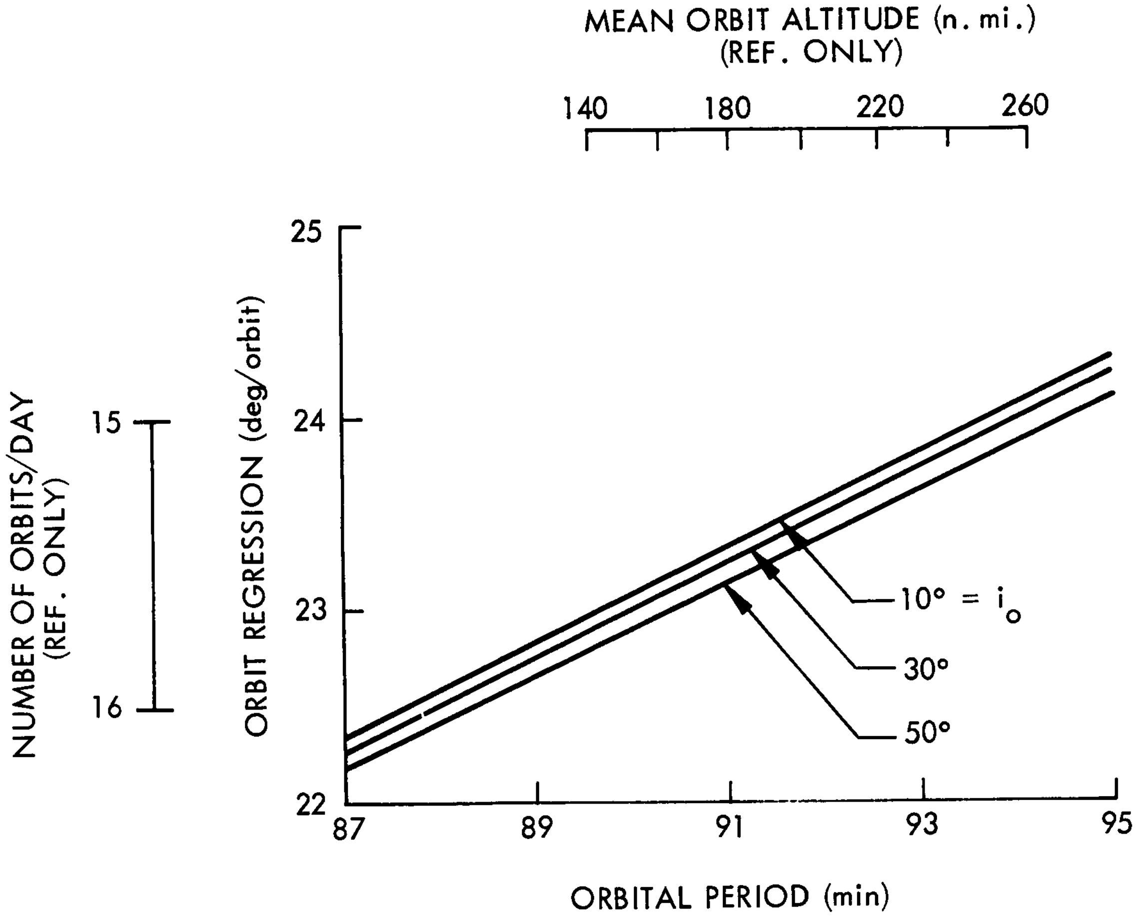


Fig. 28 Inclination Angle Effect on Longitudinal Spacing of Ground Traces

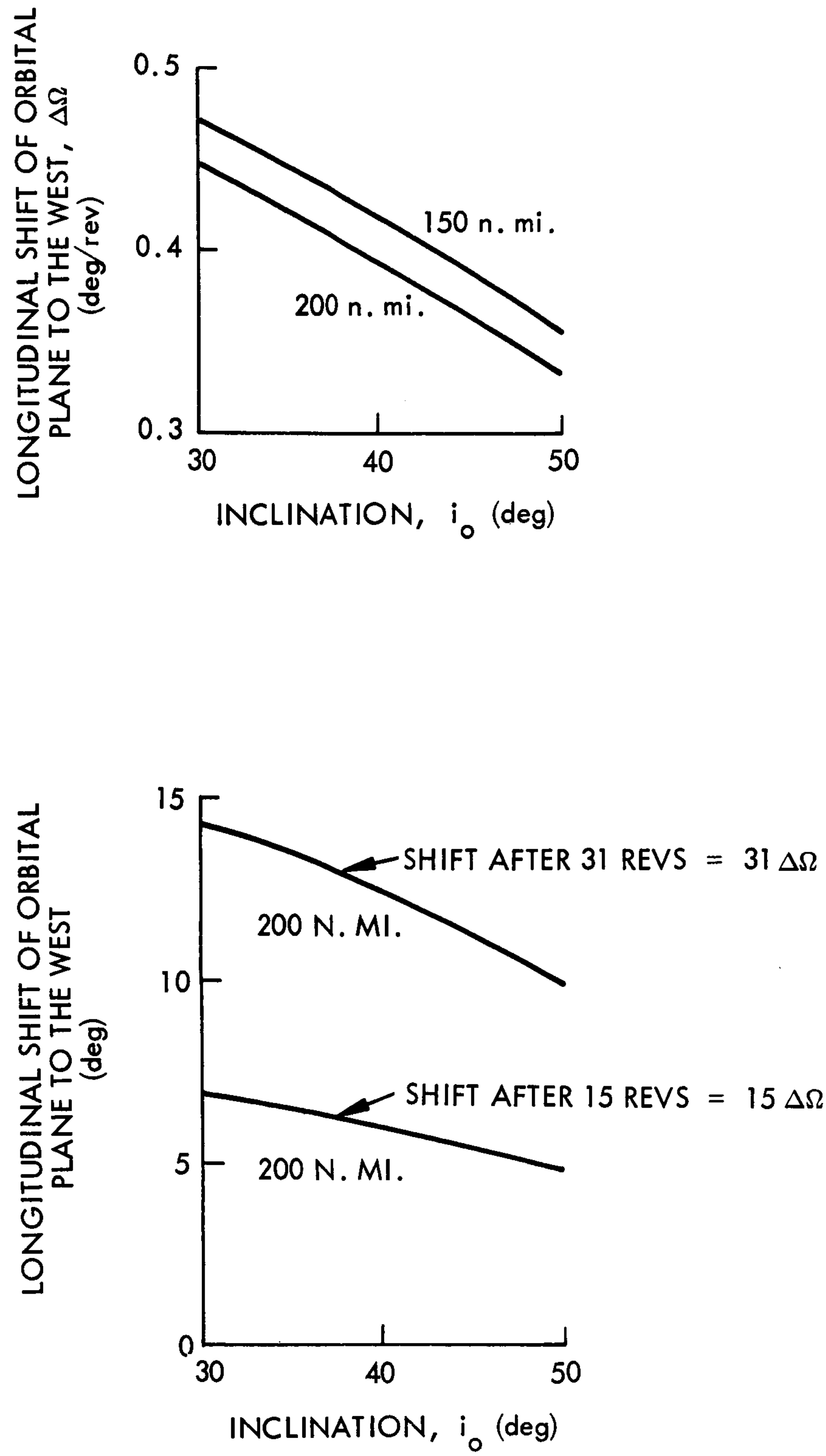


Fig. 29 Earth Oblateness Effect

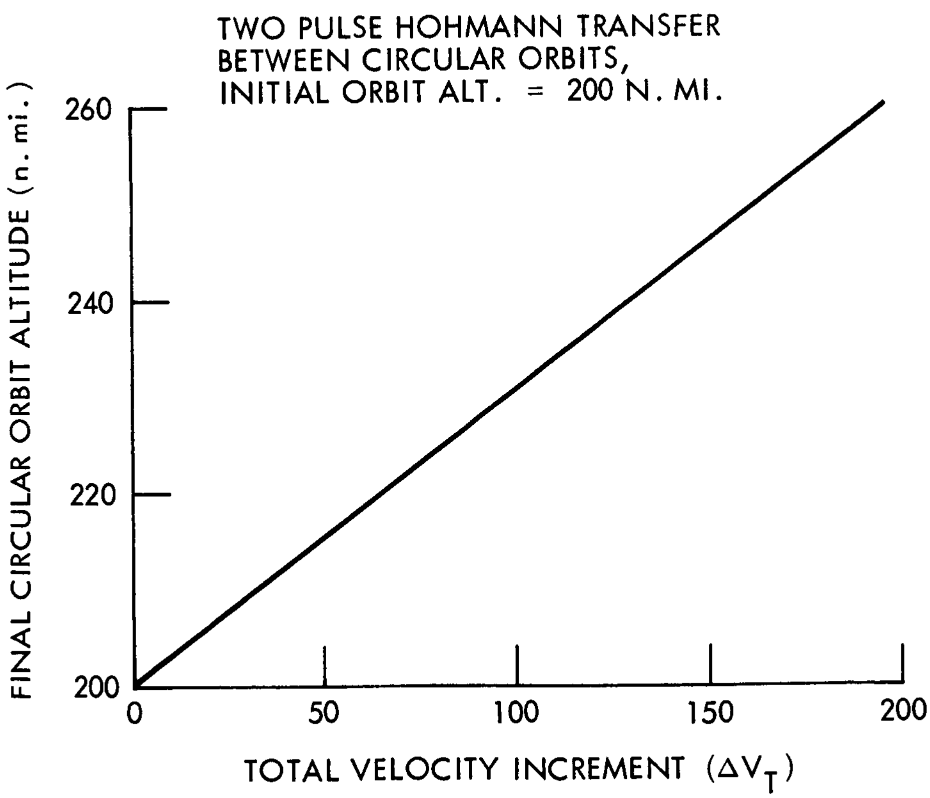


Fig. 30 Total Velocity Requirement for Coplanar Transfer from 200 N. Mi.

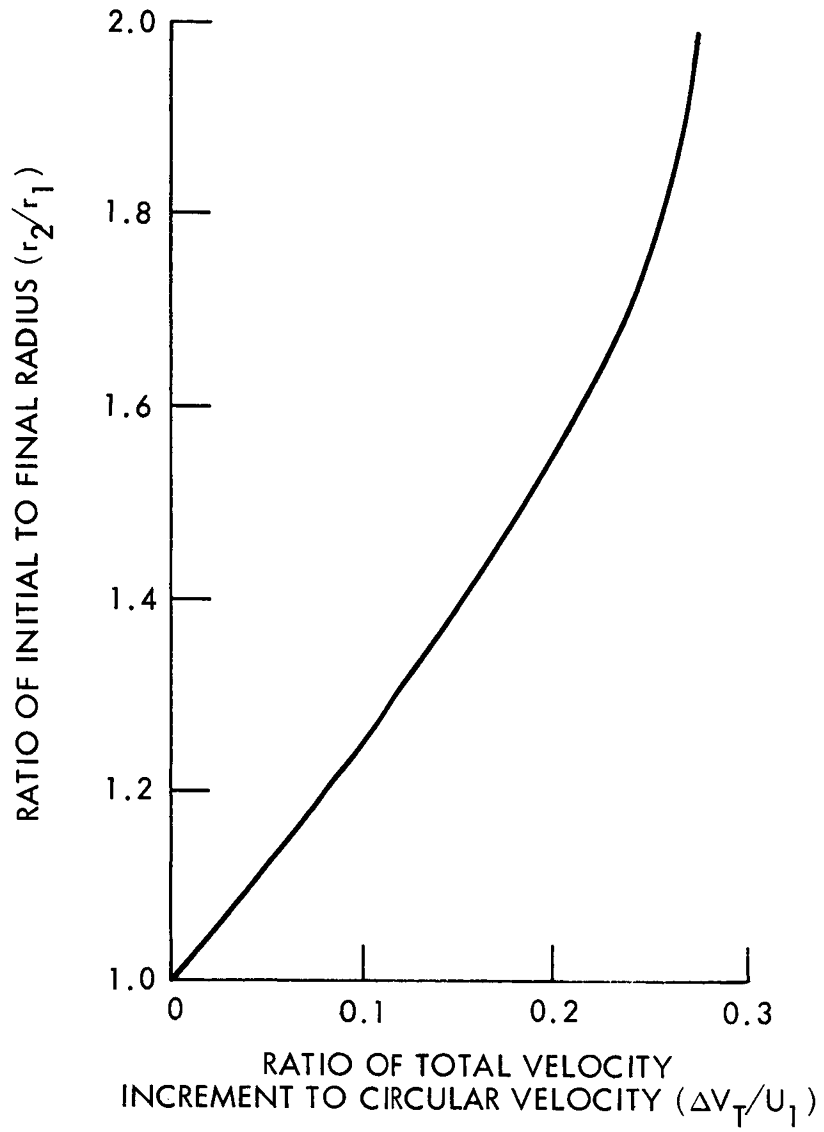


Fig. 31 Total Velocity Requirement for Coplanar Transfer, General Case

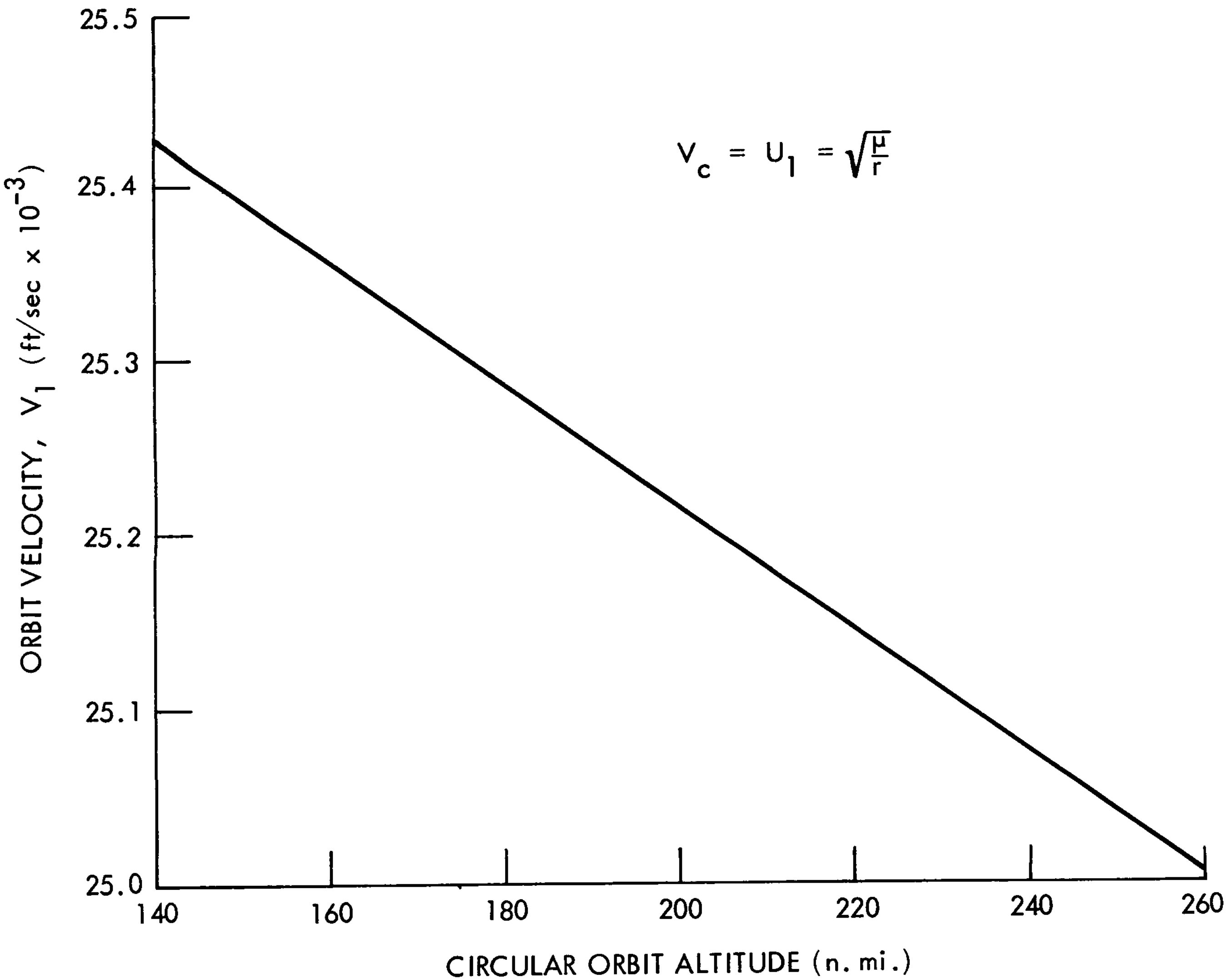


Fig. 32 Variation of Orbit Velocity With Circular Orbit Altitude

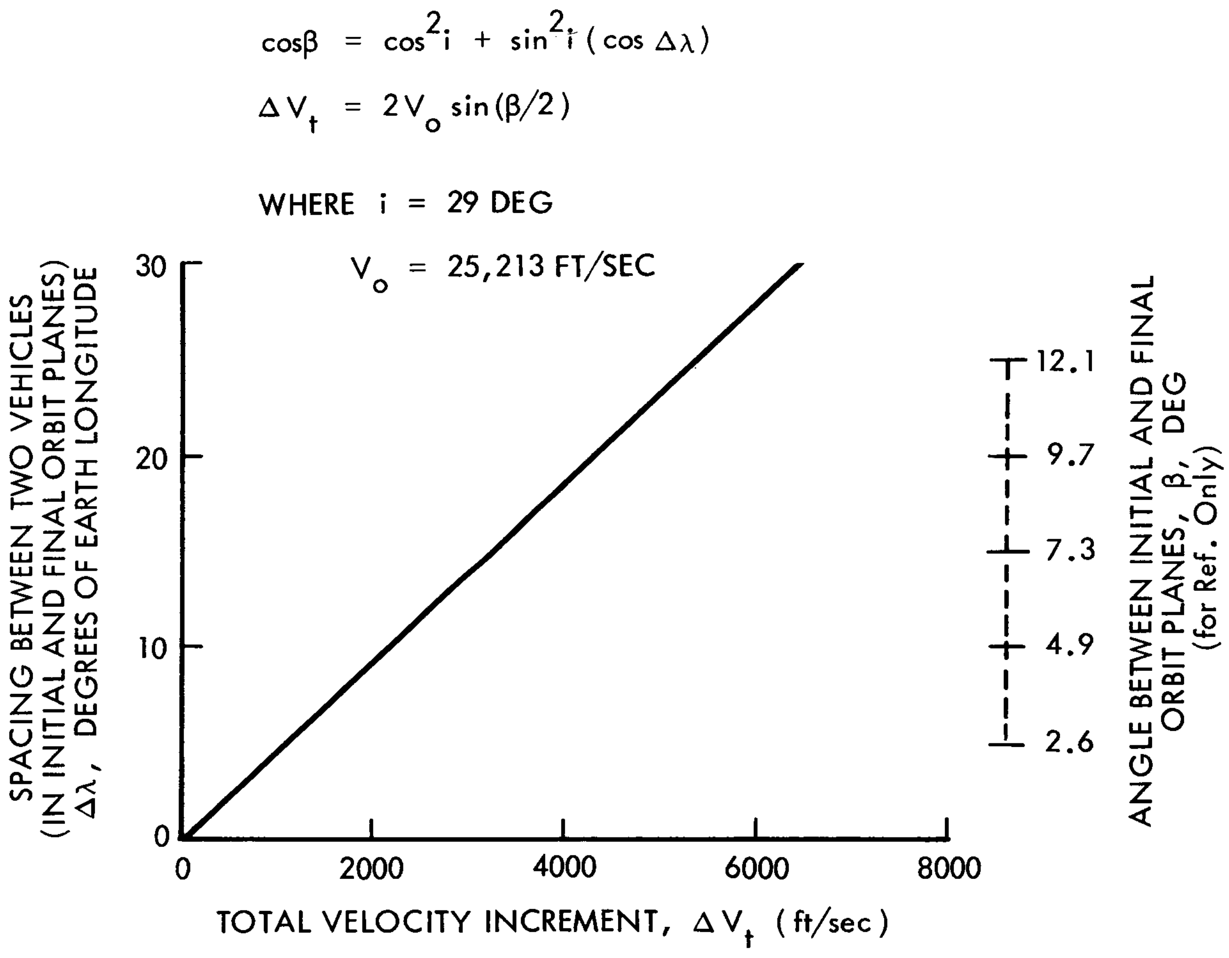


Fig. 33 Total Velocity Requirement for Single-Pulse Noncoplanar Transfer at 200 N.Mi.

of the ballistic parameters of the vehicles. Therefore, the plane separation effect can be reduced considerably by matching the ballistic parameter of the OPE spacecraft to that of the cluster. An alternate approach would be to inject the OPE spacecraft at an initial altitude different from that of the cluster so as to nullify any out-of-plane separation at the end of the 12-month mission duration.

Spacecraft illumination: The percentage of time that an OPE spacecraft in circular orbit spends in sunlight is required to size the spacecraft solar paddles. The geometry and basic definitions of the illumination problem are shown in Fig. 34. From a knowledge of the occultation angle for the planar case, and the angle between the normal to the orbit and the earth-sun line, the percentage time in sunlight (for a given fixed day) can be computed from spherical trigonometry. The basic parameters which were obtained and their variations with orbit altitude are plotted in Fig. 35. The figure is interpreted as in the following example:

- Given: Circular orbit altitude is 220 n. mi.
- From Fig. 35 : $(t_s)_{min}$ = 61 percent and i^* = 46.5 deg
- Conclusion: For 220 n. mi. altitude, all orbits with inclinations greater than 46.5 deg <u>may</u> have a period of continuous illumination, while the minimum time in sunlight cannot be smaller than 61 percent during any 24-hr period.

The maximum values of (t_s) are plotted in Fig. 36. It is seen that for 220 n. mi. and 46.5 deg, the maximum time in sunlight is greater than 90 percent, but not 100 percent. A range of percentage times in sunlight for other values of altitude may be obtained from Figs. 35 and 36.

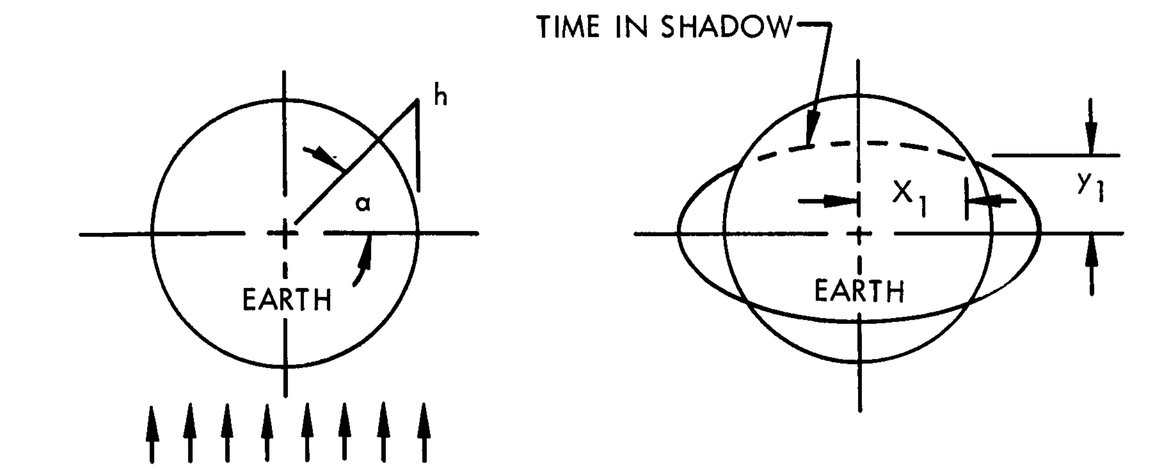
<u>Optimized mission profiles</u>. – Optimized mission profiles for the OPE spacecraft will meet the following mission requirements:

- The predicted orbital lifetimes must satisfactorily provide the required mission duration of 6 to 12 months.
- Spacecraft solar energy requirements must be achieved.
- The drag deceleration cannot exceed 0.001 g.
- The orbits chosen must in no way adversely affect the program requirements.
- The spacecraft must be interrogated at least once per day.

These mission requirements are individually analyzed in the following paragraphs.

Optimized altitude from lifetime: The numerical accuracy of the lifetime computations is ±15 percent. This is due to the state-of-the-art limitations for predicting the pertinent perturbation terms (diurnal variations in atmosphere density, etc.) and the cumulative error associated with long integration times.

Approximate analytical solutions were used to obtain an integrable drag function instead of closed form solutions to the nonlinear differential equations of motion. The variation of final altitude associated with the numerical accuracy of the lifetime calculations is shown in Fig. 37. A planned terminal condition could fall within the error



(a) PLANAR OCCULTATION ANGLE (a) (b) ORBIT PROJECTION (viewed from sun)

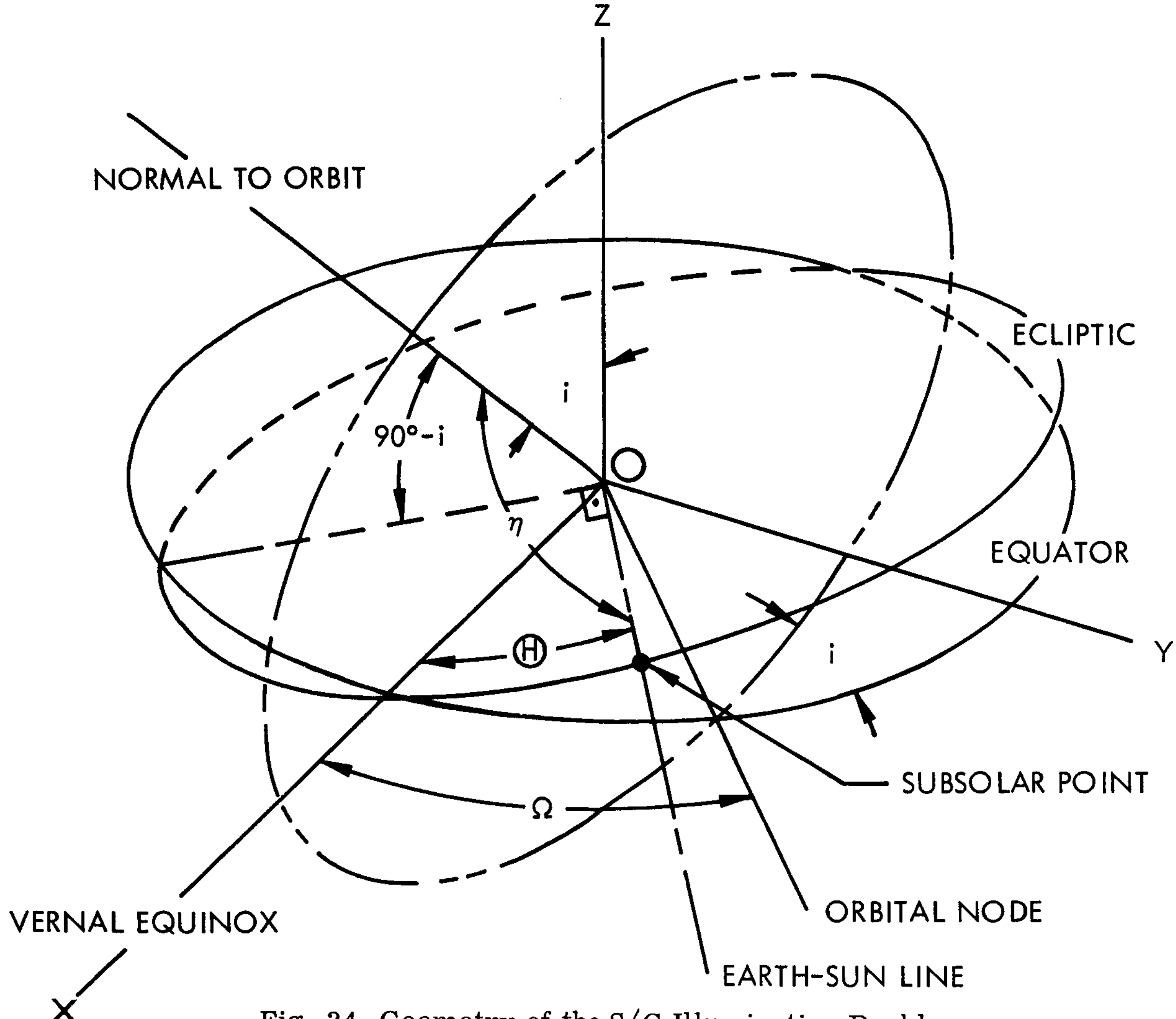


Fig. 34 Geometry of the S/C Illumination Problem

$$\sin \alpha = \sqrt{\frac{2R_e h + h^2}{R_e + h}}$$

$$i^* = 90° - \alpha - i\,.$$

$$(t_s)_{MIN} = \frac{90° + \alpha}{180°}$$

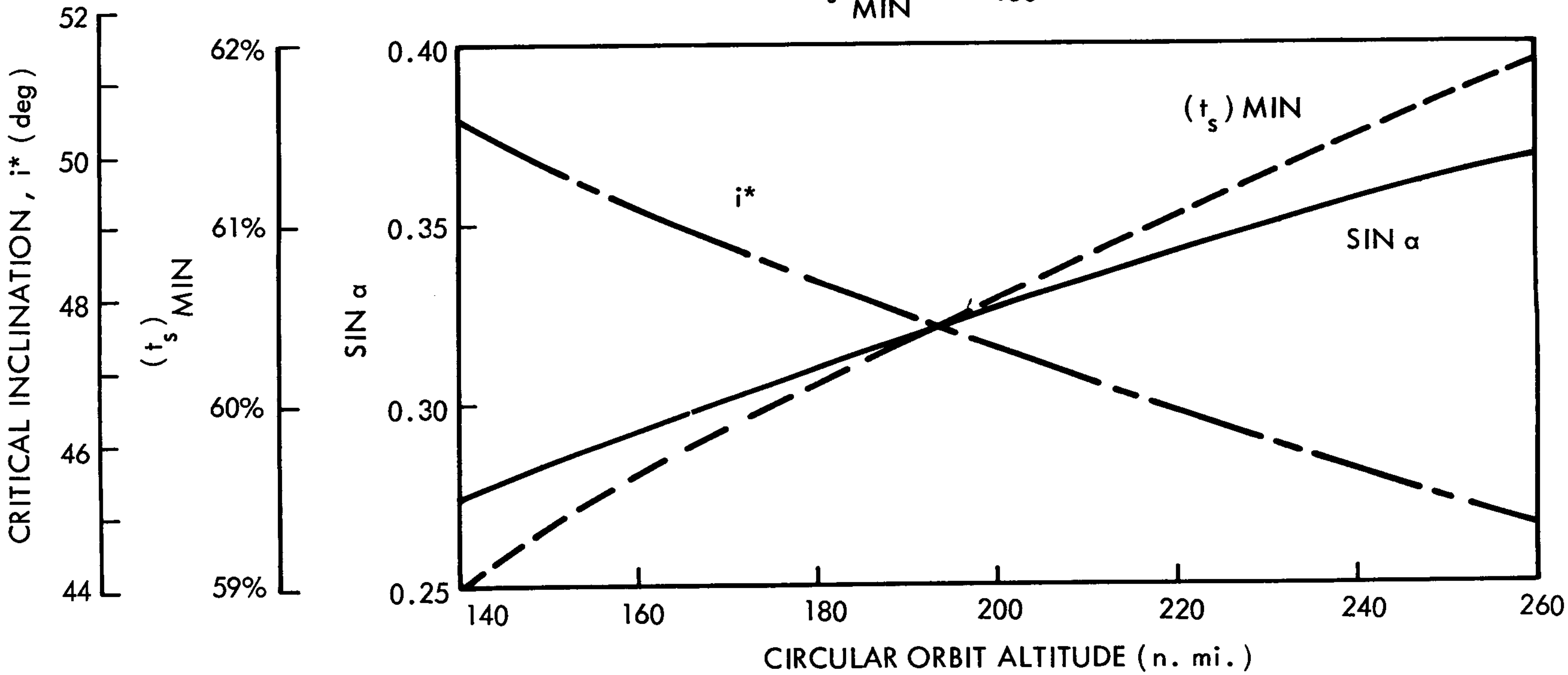


Fig. 35 Basic Parameters of the Illumination Problem

$$\frac{\sin \alpha}{\sin \eta} = 1$$

$$(t_s) = \frac{90° + \sin^{-1}\left|\frac{\sin \alpha}{\sin \eta}\right|}{180°} \times 100\%$$

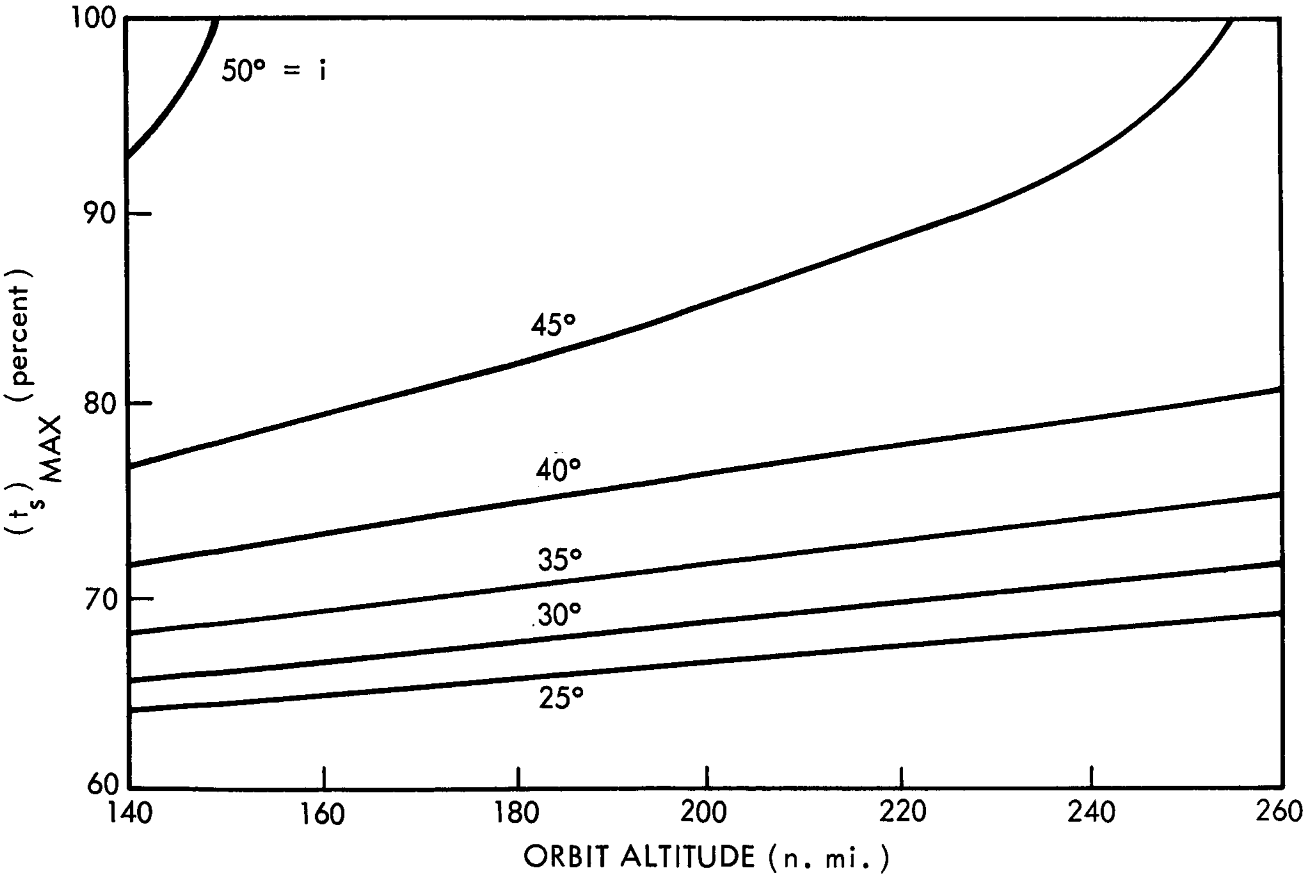


Fig. 36 Maximum Possible Percentage Time in Sunlight

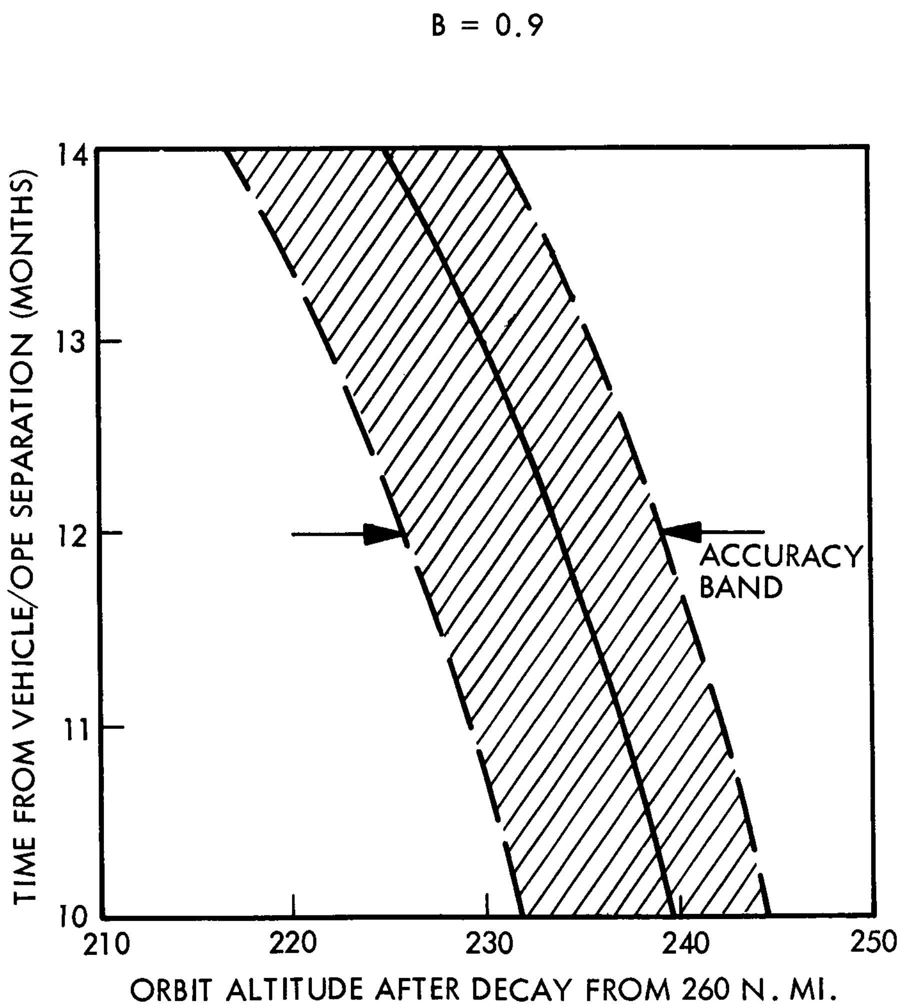


Fig. 37 Altitude Variation Associated with Atmosphere Unknowns

band, shown due mostly to unanticipated diurnal variations. It is seen that even the minimum altitude is adequate and that any altitude in the band is within the range of reasonable rendezvous maneuvers. In addition, satellite orbital data to be made available before the launch date will afford more precise lifetime predictions.

Lowest weight electrical power subsystem: It has been determined that the spacecraft will be powered by solar energy and that two solar paddles will be provided. Approximately 400 watts of electrical power must be available (on the average) to the spacecraft and to the payload. When the spacecraft is orbiting in the earth's shadow, power stored in secondary batteries will be used. Because the minimum percentage time that the spacecraft spends in sunlight is a function of altitude, the minimum electrical power subsystem weight is also a function of altitude.

The interdependencies among these parameters are shown in Fig. 38. It is seen that the subsystem weight reduction due to an increase in altitude is small. The weight of this subsystem cannot be significantly reduced by adjusting the altitude over the range of interest.

Optimized rendezvous situation: The energy required of the CSM to rendezous with the spacecraft is minimized when their orbital decay rates are approximately the same. Equal decay rates can be achieved by designing the OPE spacecraft such that its average value of ballistic parameter is equal to the cluster's. The estimated average ballistic parameter of the cluster is 1.43 $ft^2/slug$. A spacecraft average ballistic parameter of 1.43 can be designed into the spacecraft as illustrated in Fig. 39. The spacecraft ballistic parameter of 1.16 (current value) can be changed to 1.43 (desired value) by reducing the spacecraft weight by 11 percent and increasing its average area projected to the velocity vector by 10 percent. There are several ways to vary B, and so Fig. 39 cannot be considered complete or definite. The intention here is to show that the rendezvous propulsion requirements of the CSM can be greatly reduced by reasonable and practical spacecraft modifications, such as the addition of drag-inducing deflector plates to the solar paddles. The best solution will depend on a precise definition of the cluster's average ballistic parameter.

Optimized altitude from drag criteria: The mission objectives include a requirement for a maximum allowed steady-state force acting upon the spacecraft (specifically the primates) of 10^{-3} g. After orbital injection is achieved, steady-state forces can originate from drag deceleration and attitude-control-subsystem (ACS) operations. If it is assumed that ACS operations can contribute most of the allowed steady-state force, the drag deceleration must be several orders of magnitude less, say 10^{-5} g. Figure 40 shows the altitudes at which the drag force equals 10^{-5} g as a function of spacecraft W/C_dA according to two widely used atmosphere models. It is seen that the drag force acting on the spacecraft (current design and weight estimate) can reach 10^{-5} g at approximately 110 n. mi., a much lower altitude than is conceivable during the orbital phase of the mission. It is concluded that an optimized altitude cannot be selected from considerations of the limiting drag force.

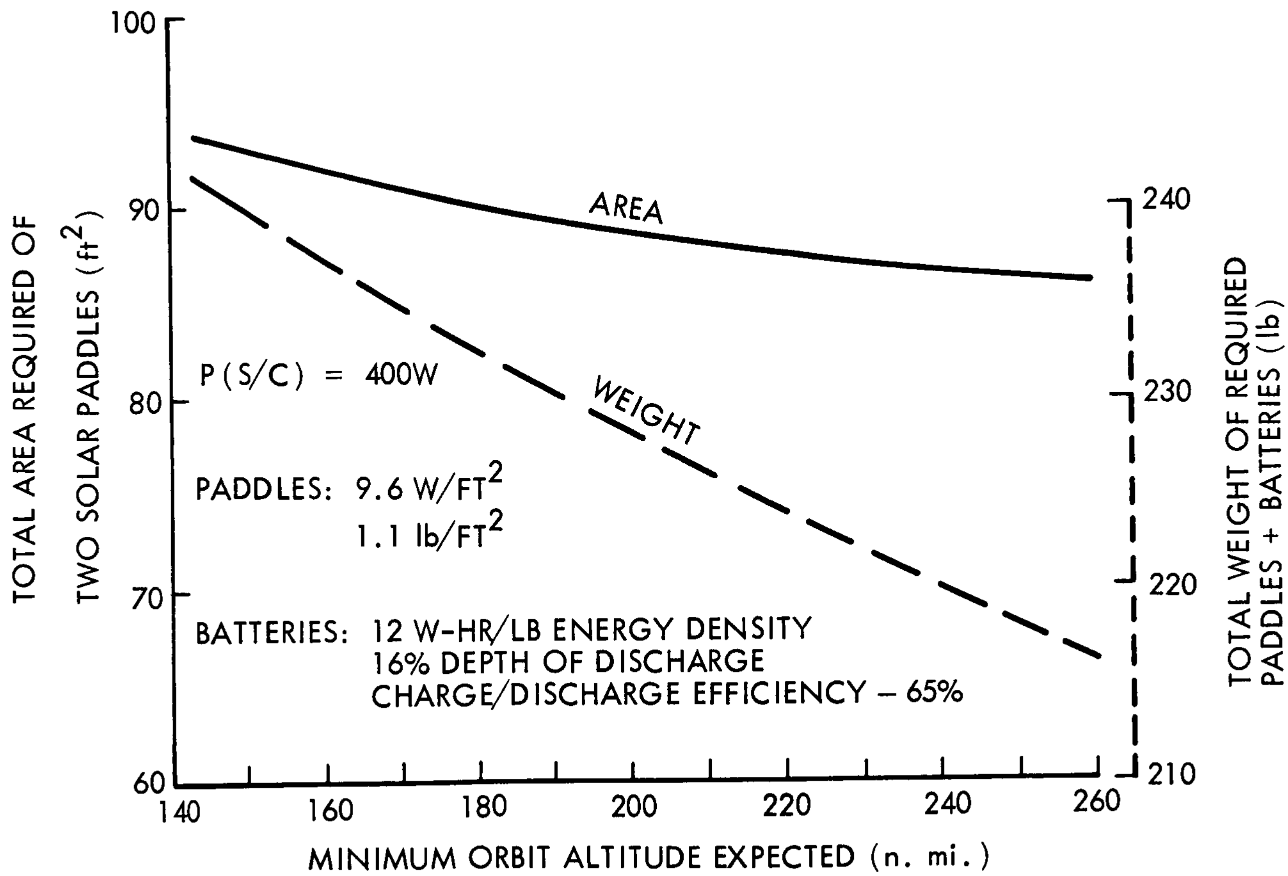


Fig. 38 Altitude Effect on Solar-Paddle Area Required and Total Weight of Electrical Power Subsystem

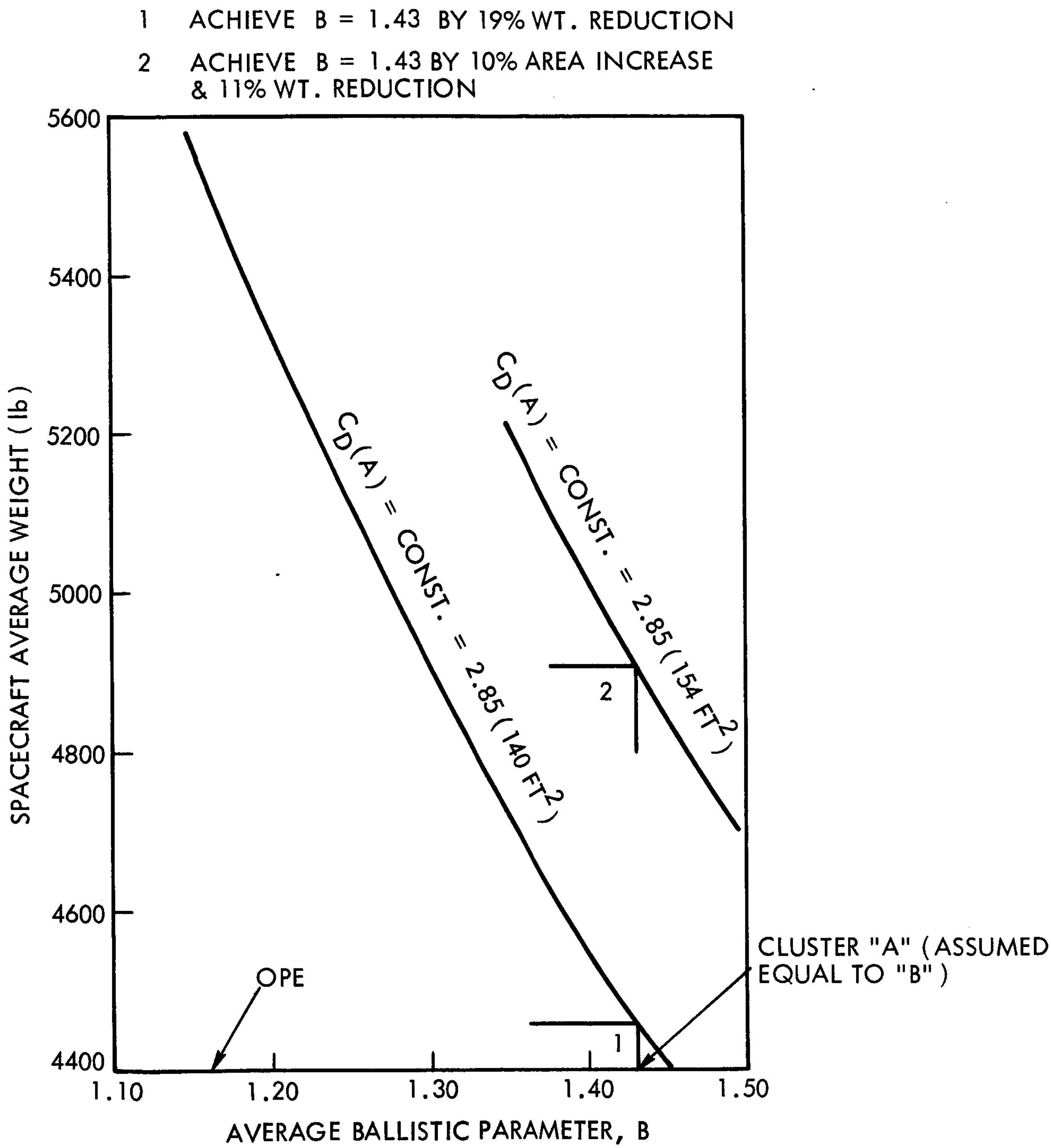


Fig. 39 Requirements for Equal OPE and Cluster Orbital Decay Rates

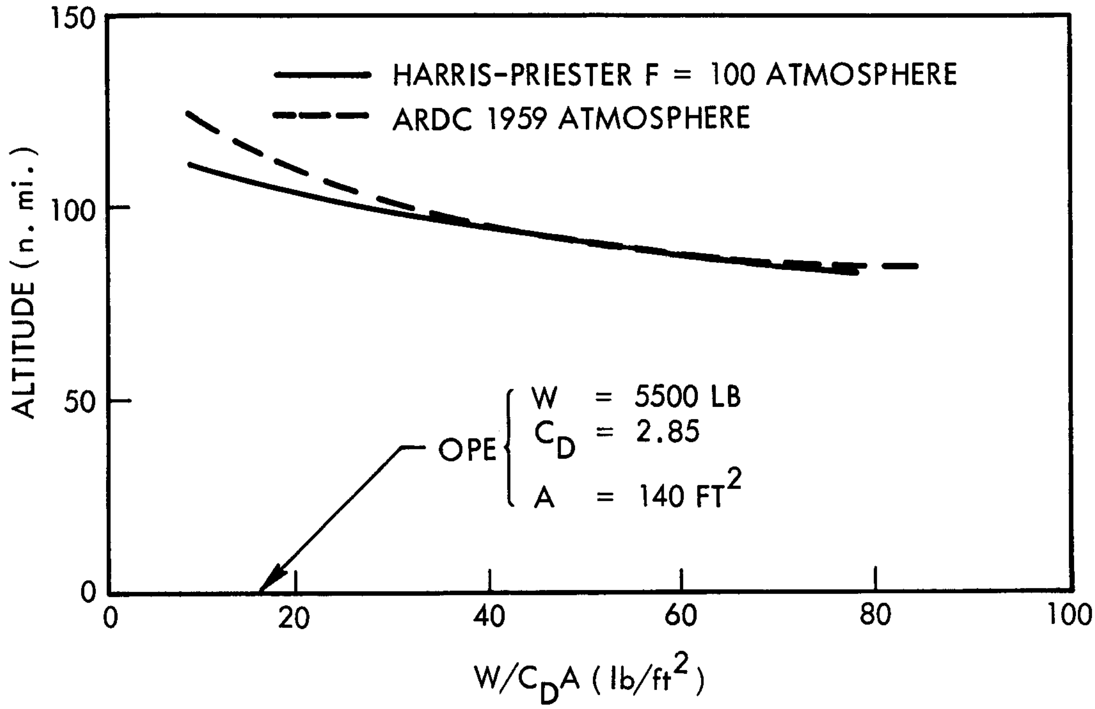


Fig. 40 Altitude Vs. W/C_DA for the Drag Force Equal to 10^{-5}g

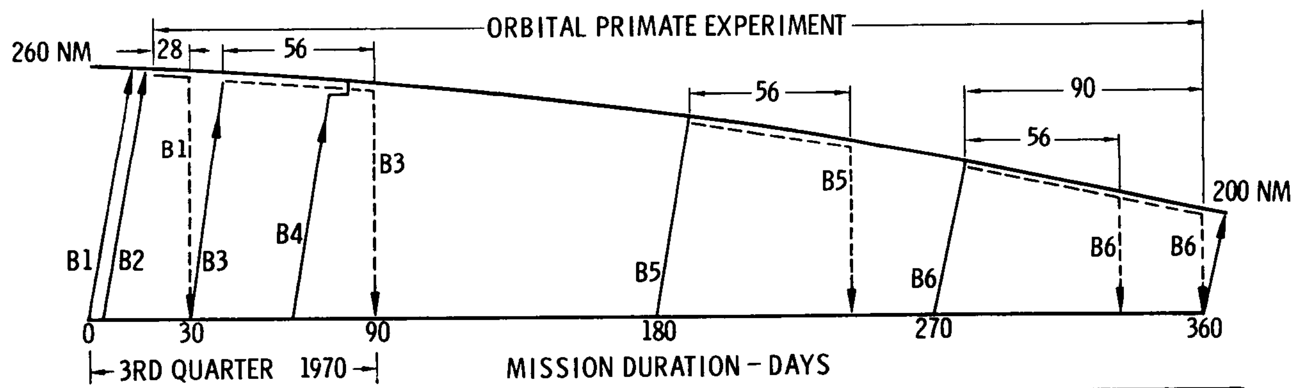
Other: Optimized altitudes and inclinations from other points of view would maximize ground station coverage and/or the spacecraft illumination. However, Fig. 25 through 28 show that more than adequate coverage is available for wide ranges of altitude and inclination. In addition, Fig. 35 shows that the minimum percentage time that the spacecraft spends in sunlight can be varied by only a few percentage points as a function of altitude. Therefore, either point of view does not afford a satisfactory basis for optimizing the spacecraft mission profile.

Summary: The AAP mission profiles are adequate. Changes to optimize them to best achieve the spacecraft/experiment objectives lead to small improvements offering an insignificant widening to the framework within which the spacecraft must be designed and integrated to the cluster. The prerequisites to a fruitful mission optimization analysis include more stringent requirements for spacecraft support during the mission, as well as candidate AAP mission profiles divergent from those currently being considered.

Determination of Best Suited S/AAP Flights

Figure 41 shows the basis for selecting the best suited AAP flight, and defines the assumed mission profile for this flight. Based upon NASA/MSFC guidelines dated 22 March 1967, the cluster will be injected into a 260 n.mi. circular orbit at an inclination of 29 deg. As shown in Fig. 41 , the cluster altitude after 12 months of decay will be approximately 200 n.mi. The estimated weight of the OPE spacecraft is 5,500 lb. It is seen in the figure that Flight B1 (using Vehicle 216) is the only flight which has a performance capability that exceeds the estimated spacecraft weight. Flight B1 is therefore the best suited S/AAP flight. Its initial orbit altitude of 260 n.mi. is sufficiently high to resist excessive orbit decay for a period of 1 yr, thereby eliminating the need for an orbit-adjust propulsion system on the OPE. The mission profile also shows that a CSM rendezvous with the OPE can be executed by the B5 or B6 CSM after the OPE has completed 6 to 12 months of orbital flight. The mission profile shown in Fig. 41 was selected as the reference mission on which the preliminary OPE designs and operations are based.

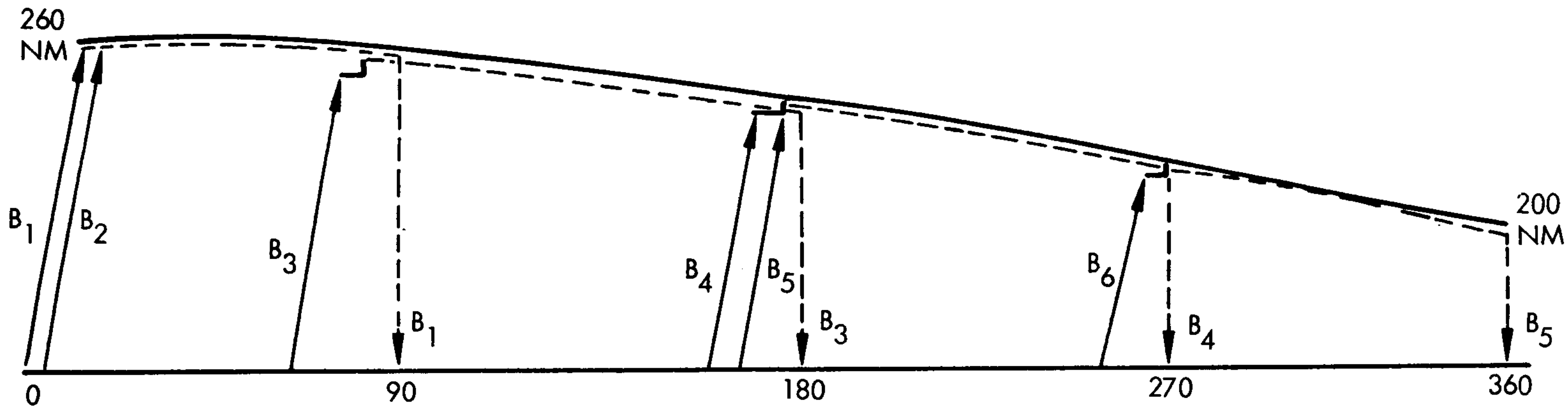
Figure 42 shows an alternate Cluster B mission profile based upon further NASA/MSFC studies dated 13 Jun 1967. These studies show that the orbit inclination may vary from 29 deg to 50 deg. It is seen in Fig. 42 that available payload capabilities for experiments are not large enough to accommodate the independent OPE. If required, a modular OPE which relies upon the cluster for attitude control, stabilization, communications, and electrical power, and weighs approximately 4,000 lb could be launched by Vehicle 214 (Flight B2) and maintained in orbit for one year. Modular-OPE/Cluster integration concepts are analyzed in Ref. 9.



FLIGHT VEHICLE NUMBER	B1 216	B2 217	B3 218	B4 219	B5 220	B6 221
CARRIERS	CSM	OWS-2 AM/MDA	XCSM	LM-A/ ATM-8	XCSM	XorL CSM
CSM	28,371	-	33,777	-	33,777	34,777
CARRIERS	-	12,300	-	22,446	-	-
SLA AND NOSE	3,950	5,167	3,950	5,017	3,950	3,950
MISCELLANEOUS	500	6,584	800	750	800	800
GROSS WEIGHT	32,821	24,051	38,527	28,213	38,527	39,527
CAPABILITY	40,300	27,900	40,300	29,800	40,300	40,300
MARGIN PLUS EXPERIMENTS	7,479	3,849	1,773	1,587	1,773	773

OPE ESTIMATED WEIGHT $\boxed{5,500\ LB}$

Fig. 41 Assumed S/AAP Mission Profile



FLIGHT VEHICLE NO.	B1 213	B2 214	B3 215	B4 216	B5 217	B6 218
CARRIERS	LCSM	OWS-2 AM/MDA	LCSM	LCSM	LM/RACK	LCSM
CSM	34,777	~	34,777	34,777	~	34,777
CARRIERS	~	12,300	~	~	22,446	~
SLA AND NOSE	3,950	5,167	3,950	3,950	5,017	3,950
MISCELLANEOUS	800	6,584	800	800	750	800
GROSS WEIGHT	39,527	24,051	39,527	39,527	28,213	39,527
* CAPABILITY: 108° LAUNCH AZIMUTH	39,500	33,300	39,500	39,500	33,300	39,500
47° LAUNCH AZIMUTH	44,300	36,100	44,300	44,300	36,100	44,300
MARGIN PLUS EXP: (MIN.)	-27	9,249	-27	-27	5,087	-27
(MAX.)	4,773	12,049	4,773	4,773	7,887	4,773
OPE TARGET WEIGHT (MODULAR TYPE)		4,000				

* WITH STRAP-ONS

Fig. 42 Alternate S/AAP Mission Profile

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TRADEOFF ANALYSES AND CONCEPT SELECTION

Tradeoff analyses were conducted to support the selection of optimum subsystems, leading to the selection of the preferred subsystem/space craft concepts and approaches. Studies were conducted in the following areas:

- Gas Supply
- O_2/N_2 partial pressure control
- CO_2 removal
- Temperature and humidity control
- Contaminant removal
- Food and water supply
- Waste management
- Cage and life cell configuration
- Animal retrieval
- Mass measurement
- Data management
- Electrical power
- Stabilization and attitude control
- Spacecraft configuration

Basis of Evaluation

All potentially desirable candidate concepts were considered in each of the above areas and evaluated on a consistent basis from considerations of reliability, cost, weight, power, volume, and development status. The basis for evaluation was arrived at as described in the following paragraphs.

The various considerations were first evaluated in terms of their relative importance in the Orbiting Primate Experiment program. As a result of this evaluation, the following relative weightings were assigned:

Weight (including power penalty)	20%
Volume	10%
Reliability	30%
Cost	20%
Development status	20%
Total	100%

Power penalties were combined with weight by converting the power requirements of a system into pounds and adding this number to the physical weight of the system. The conversion factor used was 2.06 lb/w, a value found by LMSC experience to be a close approximation for preliminary design of systems using solar cell power. For a particular system, the sum of physical weight and power in terms of weight was termed the Total Equivalent Weight (TEW) of the system.

A point-rating system was then established to arrive at a weighted value for each of the above considerations. Total Equivalent Weight was assessed on the basis of 50 lb/point with a theoretical zero-weight system receiving a perfect weighted value of 20 points. Weighted value for TEW was computed by the equation:

$$\text{Weighted Value} = -0.02\ \text{TEW} + 20$$

This equation is plotted in Fig.43. This method admits the possibility of negative weighted values to reflect the penalty incurred by excessively heavy candidate systems.

In the same way, volume (V) was assessed on the basis of 10 ft^3/point with a theoretical zero-volume system receiving a perfect weighted value of 10 points. Weighted value for volume was computed by the equation:

$$\text{Weighted Value} = -0.1V + 10$$

This equation is plotted in Fig.44and reflects the negative weighted values imposed by excessively high-volume candidate systems.

Ratings for reliability and cost were based on a 5-point relative scale in which the various candidate systems being evaluated were compared against one another and each assigned a number to indicate its standing on the scale. The highest reliability was rated at 5 and the lowest at 1; the highest cost was rated at 1 and the lowest at 5. Hardware costs were estimated on the basis of supplying one development, one qualification, and five production units.

Concept development status was also graded on a 1-to-5 scale. In this case, a discrete concept development status was defined for each point on the scale as follows:

Space operational	=	5
In qualification	=	4
In advanced development	=	3
In early development	=	2
Marginally available for 1970 flight	=	1

The ratings for reliability, cost, and development status were then multiplied by factors of 6, 4, and 4, respectively, to arrive at weighted values conforming to the relative weightings of 30 percent, and 20 percent, and 20 percent previously assigned to these considerations. All weighted values were finally summed to arrive at a total score and the selected concept chosen was the one receiving the highest total score.

Each set of candidate concepts was independently reviewed and rated by senior technical personnel experienced in the particular area of engineering being evaluated. Final selections were tested for reasonableness and sensitivity of assumptions by a senior project board. In some cases, differences of judgment occurred in matters of reliability, cost, or development status. However, in no case did such differences

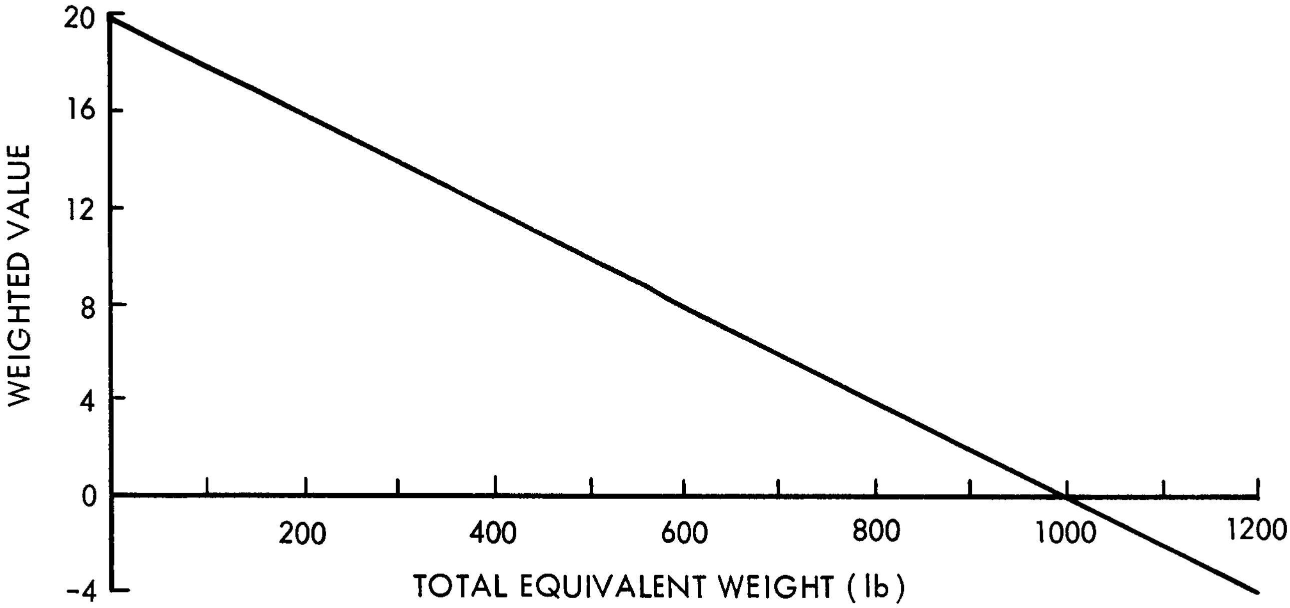


Fig. 43 Total Equivalent Weight Vs. Weighted Value

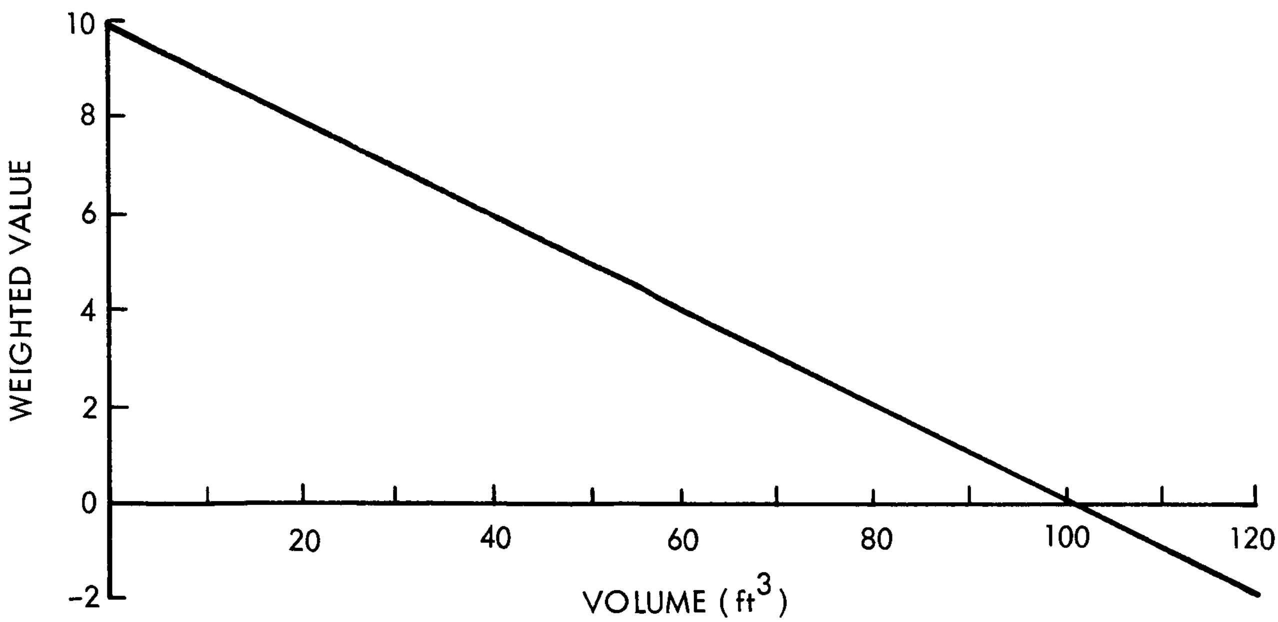


Fig. 44 Volume Vs. Weighted Value

result in a different outcome in regard to the selected candidates. The rating system was also tested for sensitivity to the hypotheses of (1) relative importance of weight, volume, reliability cost, and development status and (2) the point values assigned to weight and volume. Reasonable perturbations in these hypotheses did not result in changes regarding the selected candidates.

Tradeoff Analyses

The discussion of tradeoff analyses consist of the following:

- Thermal and atmosphere control subsystem
- Metabolic support subsystem
- Life cell
- Special equipment
- Data management subsystem
- Spacecraft subsystems

Thermal and atmosphere control subsystem. – This subsystem involves gas supply, O_2 and N_2 partial pressure control, CO_2 removal, temperature and humidity control, and contaminant removal.

Gas supply: Three candidate concepts were investigated for supplying breathing gases to the animals. These are shown schematically in Fig. 45.

In Concept 1, both oxygen and nitrogen are stored individually in high-pressure tanks at 3,000 psia. This pressure corresponds to the lowest vessel-to-fluid weight ratio for each gas although tank factor is relatively independent of pressure within the range of interest. The tank materials were selected as AM 350 stainless steel for oxygen and Ti-6Al-4V titanium for nitrogen. The tank factor in each case is three pounds total weight per pound of gas stored. A relief valve is provided to prevent overpressurization during fill and to allow for temperature variations. The gas quantities are determined from temperature and pressure sensing.

Concept 2 is a cryogenic system in which both oxygen and nitrogen are stored in individual tanks. The basic tank construction for each gas is the same and consists of:

- An aluminum inner shell
- Three vapor shields
- Fiberglass supports between shields
- Superinsulation between shields
- Nylon net outer wrapping

Differences between tank construction for the gases are in thickness of material and insulation which is greater for the nitrogen tanks. A review of the oxygen usage

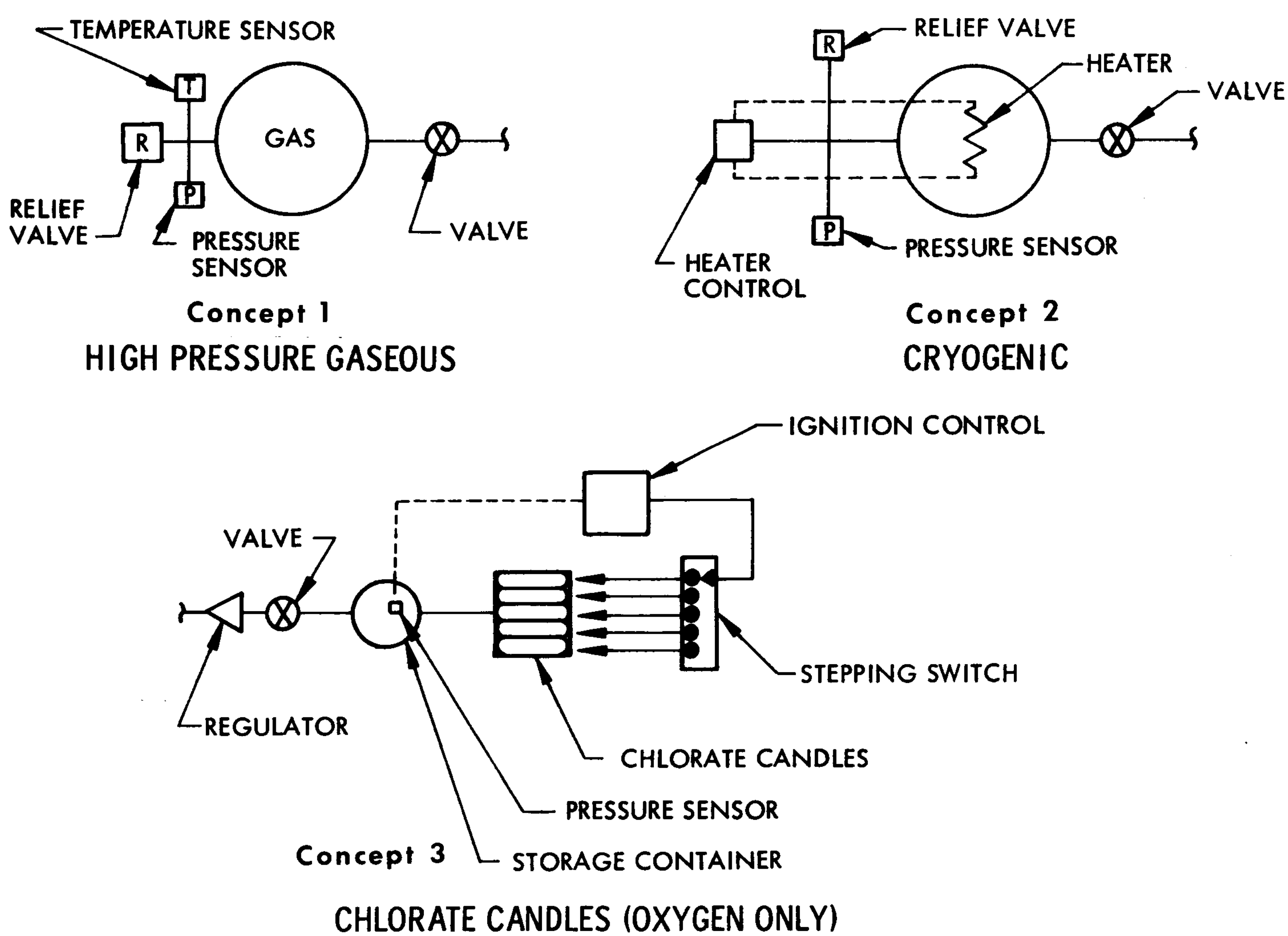


Fig. 45 Gas Supply Subsystem Candidates

rate indicated that it could be stored subcritically at a low weight penalty factor using a nominal storage pressure of 250 psia and temperature of 225° R.

The nitrogen usage rate is very low. In order to minimize gas venting, tank pressure must be allowed to increase. The nitrogen storage tank, therefore, would be operating supercritically at a nominal pressure of 925 psia and temperature of 250°R. Tank penalty factors are 1.6 and 3.1 pounds total/pound of fluid for oxygen and nitrogen, respectively. The cryogens are heated to operating pressures by heat leaking into the storage vessels and from auxiliary heaters when demand exceeds boil-off from heat leak. The heaters are conservatively sized at 5w for oxygen and 3w for nitrogen. Tank overpressurization is prevented by venting through a relief valve.

The advantages of the cryogenic storage systems are that the higher fluid density and lower operating pressure allow lower container weights and volume for a given capacity. The disadvantages are that system performance is strongly affected by the amount of heat leaking into the storage vessel and that the design of the vessel is greatly influenced by usage rate and storage time. For a condition of low usage and long storage, the allowable heat leak into the vessel must be held at a minimum to prevent excess losses from venting. This leads to more complex tank construction by increasing insulation thicknesses and heat shielding.

Concept 3 uses chlorate candles to generate gaseous oxygen. The candles are individually lighted by an electric match and the oxygen output is stored in a pressure container at a nominal 100 psia and regulated for life-cell usage. A low-pressure sensor in the storage container activates the ignition of the candles. Progressive ignition is maintained by a stepping-switch box. Each 3-lb candle (2.25 in. diam. and 11 in. long) can produce 1.2 lb of oxygen. The weight penalty factor is 2.5 lb total/ lb of usable oxygen. The evaluation of the candidate gas supply concepts is shown in Table 15.

Chlorate candles and gaseous oxygen storage methods are seen to be heavier than cyrogenic storage. Differences are caused by mechanisms and tank factors required for chlorate candles and gaseous tanks. Subcritical storage of a large quantity of oxygen with its usage rate (0.8 lb/ day) results in a lighter cryogenic storage system. The nitrogen supercritical storage method is heavier than gaseous storage due to the higher tank penalty factor associated with the more complex tank design which involves a pressure shell capable of containing the supercritical pressure, heat shielding and insulation blankets, and standoffs.

In reliability, the cryogenic method rated lower than gaseous storage because of the possibility of heater failure or insulation breakdown. Chlorate candles are rated lowest because of possibility of electrical failures of match mechanism control sensor, ignition control unit, or stepping switch mechanism.

The cost of development and qualification of flight hardware is highest for cryogenics due to requirements for equipment to withstand extreme thermal conditions.

TABLE 15

EVALUATION OF GAS SUPPLY SUBSYSTEMS

METHOD	Weight	Power		Total Equivalent Weight		Volume		Reliability		Cost		Conceptual Development Status		Score
	Lb	Watts	Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
OXYGEN														
• Cryogenic (Subcritical)	400	5	10.6	410.6	11.8	4.3	9.6	3	18	1	4	3	12	55.4
• Gaseous	750	-	-	750	5.0	16.2	8.4	5	30	5	20	5	20	83.4
• Chlorate Candles	780	-	-	780	4.2	5.9	9.4	2	12	3	12	2	8	45.6
NITROGEN														
• Cryogenic (Supercritical)	284	3	6.2	290.2	14.2	2.2	9.8	3	18	1	4	3	12	58.0
• Gaseous	275	-	-	275.0	14.5	7.8	9.2	5	30	5	20	5	20	93.7

Extensive development and qualification testing is required for subcritical storage and for the durations required by the mission. Chlorate candles require control, sensing, ignition, and switching mechanisms, but do not have to withstand the extreme cold conditions of cryogenic storage. Gaseous storage does not require the ability to withstand extreme cold conditions and also is less complex than the mechanisms for chlorate candles. Development and qualification testing for gaseous storage is minimal.

Chlorate candles, utilized in submarines and in some versions of extravehicular activity life support packs, have not been developed to provide primary oxygen support for long-term missions. Extensive development of this system for this mission would be required. Cryogenic systems have been used on past manned and unmanned space vehicles and the technology is highly advanced. For missions of this duration, subcritical storage would be required; this technology requires further development. Gaseous storage methods have been used extensively on past programs and require the least additional development and qualification.

On the basis of the above evaluation, gaseous storage of both oxygen and nitrogen was selected as the preferred approach.

Oxygen (O_2) and nitrogen (N_2) partial pressure control: The three candidate concepts for O_2 and N_2 partial pressure control are shown in Fig. 46.

Concept 1 is a system in which total pressure is maintained with nitrogen admitted through a total pressure regulator. The total pressure nominal value is 14.7 psia. Oxygen partial pressure is maintained by supplying oxygen through a flow-control valve controlled by an optical (ultraviolet attenuating) oxygen partial pressure sensor. The oxygen partial pressure is 3.5 psia nominal. This method has the ability to supply oxygen and nitrogen as demanded by leakage and metabolic requirements.

Concept 2 is a fixed bleed system in which oxygen and nitrogen are stored mixed in the same tank at a mixture ratio of 61 percent (N_2/total). The bleed rate of the life cell is established at a rate capable of limiting methane concentration and allowing a wide range of metabolic oxygen consumption (from 0.85 to 2.15 BMR/animal). The bleed rate is assumed constant throughout the mission.

Concept 3 is a bleed system with adjustable flow control. Cabin leakage is measured during initial portions of the mission utilizing an oxygen partial pressure sensor. During this period the nitrogen flow control is adjusted until the correct oxygen partial pressure is achieved and the setting remains fixed throughout the rest of the flight. Total pressure is maintained through a total pressure regulator. This method has the ability of supplying oxygen as demanded by leakage and metabolic requirements. Adjustments to the flow control may be made throughout the flight provided the oxygen partial pressure sensor remains operative.

This method relies on a constant rate of leakage from the cabin and once adjusted can tolerate only an 8 percent decrease or 9 percent increase in leak rate to maintain oxygen concentration between 18 and 30 percent. The evaluation of these concepts, shown in Table 16, is based on using high pressure gaseous storage tankage as the supply source.

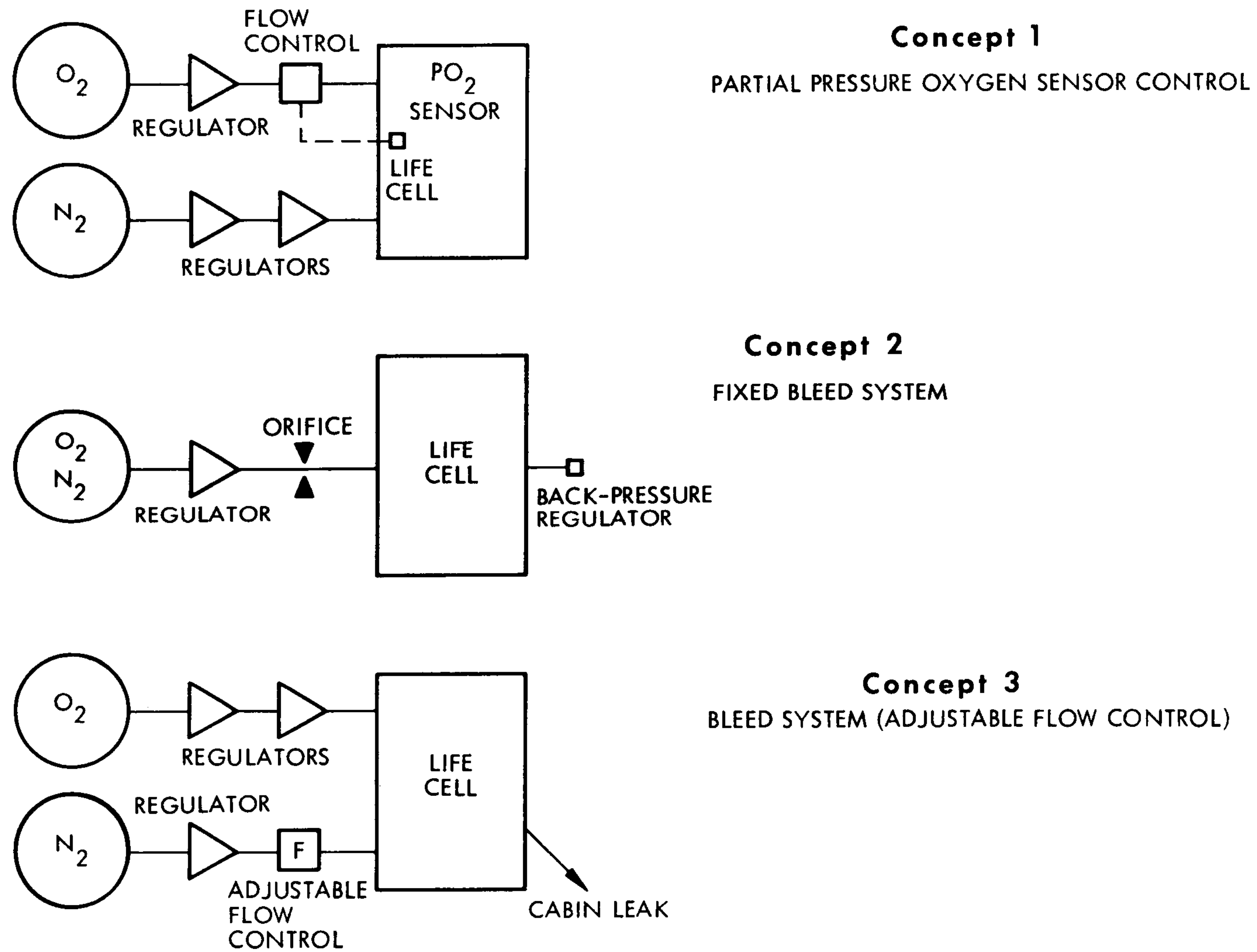


Fig. 46 Oxygen and Nitrogen Partial Pressure Control Candidates

TABLE 16

EVALUATION OF OXYGEN AND NITROGEN PARTIAL PRESSURE CONTROL SUBSYSTEMS

METHOD	Weight	Power		Total Equivalent Weight		Volume		Reliability		Cost		Conceptual Development Status		Score
	Lb	Watts	Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
Partial Pressure Oxygen Sensor Control	1,032	10	20.6	1,052.6	- 1.1	24.3	7.6	2	12	3	12	2	8	38.5
Fixed Bleed System (Mixed Oxygen-Nitrogen)	3,120	-	-	3,120	-42.5	79.5	2.0	4	24	5	20	4	16	20.5
Bleed System (Separate Oxygen-Nitrogen)	1,025	-	-	1,025	- 0.5	24.0	7.6	3	18	5	20	4	16	61.1

The fixed bleed system of Concept 2 results in a high total equivalent weight because of the high bleed rate to accommodate a wide metabolic range. The partial pressure oxygen sensor control is slightly heavier than the bleed system (Concept 3) because of additional sensor and controller requirements.

The large quantities of gas required for the fixed bleed system are reflected in a larger volume requirement and a subsequently lower rating. The ratings between the partial pressure oxygen sensor control and the bleed system of Concept 3 show negligible difference.

The fixed bleed system, rated the most reliable, has minimum moving parts and the least possibility of failures. The Concept 3 bleed system, rated less reliable, has additional regulators and a flow controller. Sensor and control requirements make the partial pressure oxygen sensor control system least reliable.

Cost for development, qualification, and flight hardware is low for Concepts 2 and 3. Additional controls, development, and qualification of the partial pressure oxygen sensor control system are more costly. Therefore, this system is rated lower.

Most of the components used in the bleed systems have reached flight qualification status, consequently these systems were given high ratings in development status. Long-life partial oxygen sensor/ control equipment is in the early development state. Considerable effort will be required to fully develop and qualify sensor/ control systems.

The bleed system of Concept 3 was selected as the preferred approach, but a combination of Concepts 1 and 3 appears to offer considerable advantages. Since the capability of switching between Concepts 1 and 3 involves only minimum equipment (valve, oxygen flow control, and nitrogen total pressure regulator), it appears feasible to provide this option. Since oxygen partial pressure sensors are required to measure cabin oxygen concentration, the selected system will use an identical sensor system for control and have the additional capability of switching to the bleed system of Concept 3.

Carbon dioxide removal: Five candidate concepts were investigated for CO_2 removal. These are shown schematically in Fig. 47.

Concept 1 uses potassium superoxide (KO_2) and lithium hydroxide (LiOH). The KO_2 absorbs carbon dioxide and water vapor to produce oxygen. With KO_2 alone, variations in metabolic or leakage rates can result in insufficient carbon dioxide absorption or oxygen overproduction. These problems are avoided by adding an LiOH bed in parallel with the KO_2 bed. This system is sized to provide sufficient oxygen for a cabin leakage of 0.3 lb/ day and an average metabolic rate up to 1.5 basal. Control is achieved through a bypass valve responding to the oxygen partial pressure sensor/ controller. The circulated gas is directed through the KO_2 bed until the oxygen level reaches 3.5 psia. The bypass control then directs a portion of the flow through the LiOH bed.

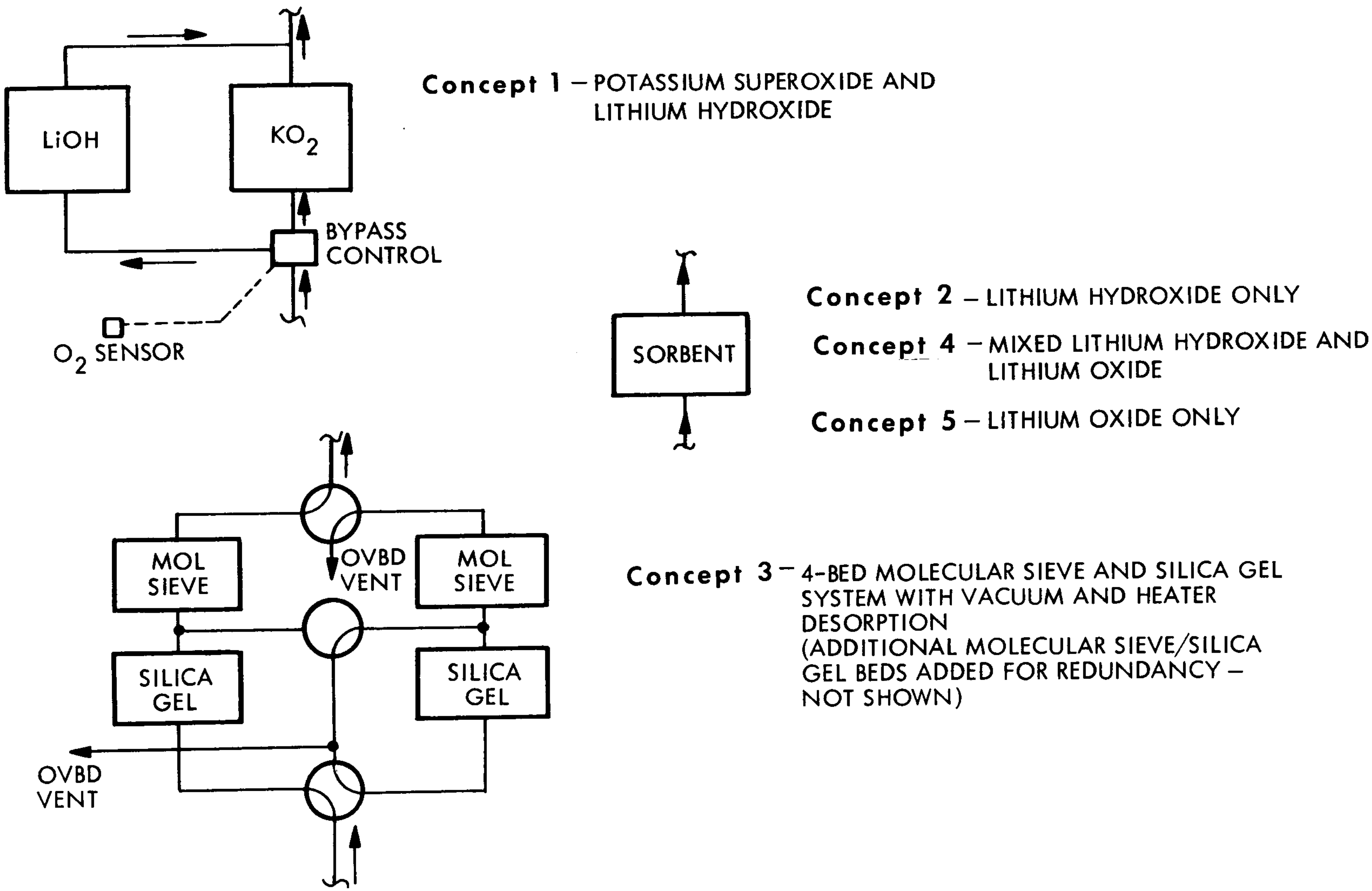


Fig. 47 Carbon Dioxide Removal Subsystem Candidates

Concept 2 uses lithium hydroxide only and provides for continuous flow through a packed bed for carbon dioxide removal by chemisorption.

Concept 3 uses silica gel beds in series with Molecular Sieve beds. The silica gel adsorbs the water vapor from the gas before entry to the Molecular Sieve bed. The Molecular Sieve material adsorbs the carbon dioxide. While one set of beds is adsorbing, the previously saturated set is desorbed to space. Electrical heaters are used to aid the desorption of water vapor from the silical gel beds.

Concept 4 is a mixed lithium hydroxide and lithium oxide system which utilizes continuous flow chemisorption for carbon dioxide removal. Lithium hydroxide produces 0.41 lb of water vapor/ lb of CO_2 absorbed. Lithium oxide absorbs water vapor and carbon dioxide. The system is sized with enough lithium oxide to absorb the water vapor produced by the lithium hydroxide.

Concept 5 uses lithium oxide only to adsorb carbon dioxide and water vapor. Excessive adsorption of water vapor could reduce the cabin relative humidity below an acceptable limit. Therefore, this concept may require additional controls to maintain relative humidity limits.

The evaluation matrix is shown in Table 17. The values on the matrix reflect that potassium superoxide, lithium hydroxide, and lithium oxide methods include requirements for chemicals, canisters, gaseous oxygen supply, and controls. The Molecular Sieve method includes requirements for oxygen supply, gas loss during desorption, desorption heater, dessicant, Molecular Sieve, canisters, and accessories. The ability of the Molecular Sieve method to desorb to space results in less sorbent requirements and, therefore, less weight. The power penalty for this method is moderate; as a result, the method has the lowest total equivalent weight. Weight differences between other methods are less significant. Due to its lower sorbent requirements, the Molecular Sieve unit also occupies less volume than the other methods. The volume differences between the other methods are less significant.

The lithium hydroxide/ lithium oxide methods (Concepts 2, 4, and 5) utilize continuous flow without elaborate control mechanisms. These three methods are rated equally high in reliability. The Molecular Sieve method employs adsorption/ desorption valve controls and timer mechanisms which result in a relatively complex system. The Molecular Sieve may be adversely affected by water carry-over from the silica gel. These considerations reflect in a relatively low reliability rating for this concept. The potassium superoxide/ lithium hydroxide method requires gas sensing and bypass control mechanisms, factors which contribute to its low reliability rating.

The lithium hydroxide method is least costly because of its simplicity and its operational development status. The lithium hydroxide/ lithium oxide methods are slightly more expensive because of additional development requirements. The potassium superoxide/ lithium hydroxide and the Molecular Sieve methods involve relatively expensive components and considerable development and qualification effort, resulting in lower ratings for these methods.

TABLE 17

EVALUATION OF CARBON DIOXIDE REMOVAL AND OXYGEN SUPPLY SUBSYSTEMS

METHOD	Weight	Power		Total Equivalent Weight		Volume		Reliability		Cost		Conceptual Development Status		Score
	Lb	Watts	Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
Potassium Superoxide and Lithium Hydroxide	1,115	-	-	1,115	-2.2	23.5	7.[illegible]	2	12	3	12	2	8	37.5
Lithium Hydroxide, Gaseous Oxygen	1,140	-	-	1,140	-2.8	26.9	7.3	4	24	5	20	5	20	68.5
Molecular Sieve, Gaseous Oxygen	799	28.5	59.8	859	2.8	17.2	8.3	2	12	2	8	3	12	43.1
Mixed Lithium Hydroxide, Lithium Oxide, Gaseous Oxygen	1,094	-	-	1,094	-1.9	22.9	7.7	4	24	4	16	3	12	57.8
Lithium Oxide, Gaseous Oxygen	1,079	-	-	1,079	-1.6	22.0	7.8	4	24	4	16	3	12	58.2

From the standpoint of concept development status, the lithium hydroxide method is operational and is therefore rated highest. The Molecular Sieve method is currently in an advanced state of development for the 45-day Apollo life-support system. Concepts 4 and 5 are not as highly developed as lithium hydroxide but would require relatively little effort for qualification. The potassium superoxide/ lithium hydroxide method is in a relatively early state of development, resulting in the lowest rating for this method.

As shown on the matrix, the lithium hydroxide and gaseous oxygen combination is the selected approach for CO_2 removal and oxygen supply.

Temperature and humidity control: Three candidate concepts were studied for temperature and humidity control. These are shown in Fig. 48.

In Concept 1 a dessicant bed is used for moisture removal. Air is passed either through or around the dessicant bed in response to a humidity sensor for humidity control. The primary method of heat rejection from the life cell is by radiation to the external skin. The system is designed so that there is excess heat rejection from the life cell. This is accomplished by surface absorptance and emittance selection. The life cell is heated to maintain temperature control through the use of a resistance heater controlled by a thermostat. Silica gel and lithium chloride were selected as the most attractive dessicants.

In Concept 2 a condensing cold plate is used for moisture removal and heat rejection. Conduction paths are provided from the cold plate to the external radiator. The system is designed so that there is always excess heat rejection from the cold plate. The cold plate, incorporated in the life cell wall, is heated to maintain the required dew point. Bypass air flows around the cold plate as controlled by a temperature sensor. The bypass flow is limited to ensure that the design maximum humidity is not exceeded under low heat loads. The conduction paths and the life cell wall are insulated, as shown in Fig. 48 to ensure that the conduction paths represent the major heat loss path from the life cell to the spacecraft skin.

Concept 3 is a system in which a condensing heat exchanger is used for moisture removal and heat rejection. A fluid heat transfer circuit is provided between the condensing heat exchanger and an external radiator. The life cell is thermally isolated from the spacecraft skin so that the circulating fluid system is the primary heat rejection path. Bypass fluid flows around and/ or through the external radiator to control the heat exchanger inlet temperature at the design point. Bypass air flows around and/ or through the condensing heat exchanger to provide temperature control in the life cell. The bypass air flow is limited to ensure that the design maximum humidity is not exceeded under low heat loads. The heat exchanger air flow is determined by temperature control requirements. The evaluation of the three concepts is shown in Table 18.

The semi-passive dessicant systems impose considerable weight penalties due principally to large quantities of required dessicants, but also due to the electrical

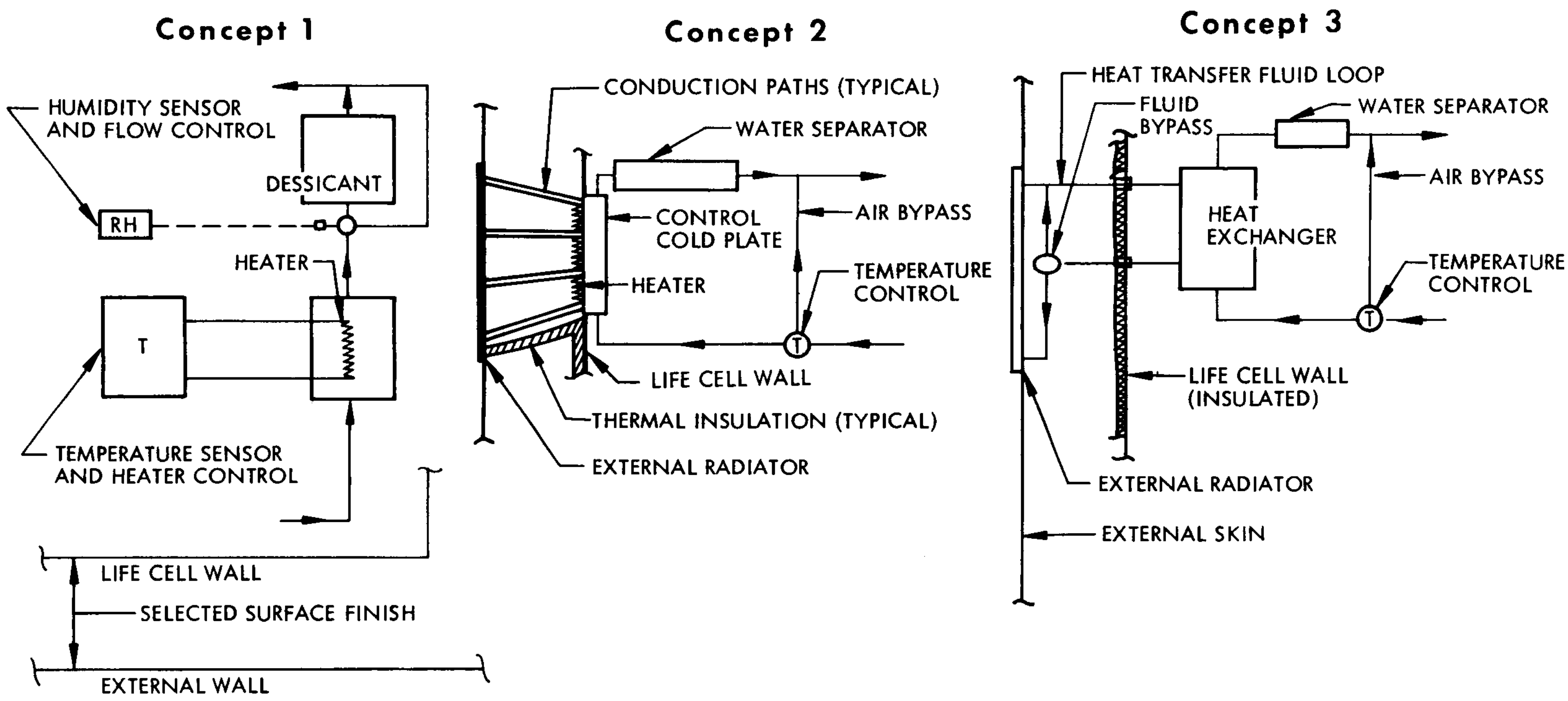


Fig. 48 Temperature and Humidity Control Subsystem Candidates

TABLE 18

EVALUATION OF TEMPERATURE AND HUMIDITY CONTROL SUBSYSTEMS

METHOD	Weight	Power		Total Equivalent Weight		Volume		Reliability		Cost		Concept Development Status		Score
	Lb	Watts	Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
Semi-Passive with Heater and Dessicant Water Storage														
1. Silica Gel	2,759	400	825	3,584	-51.7	69	3.1	4	24	4	16	3	12	3.4
2. Lithium Chloride	430	400	825	1,255	- 5.1	16.6	8.3	3	18	2	8	2	8	37.2
Semi-Passive with Heater and Water Separators	87	180	370	457	10.9	0.6	9.9	3	18	2	8	3	12	58.8
Active Circulating Fluid with Condensate Moisture Removal	79	15	30.9	109.9	17.8	2.1	9.8	2	12	3	12	5	20	71.6

power penalty imposed by heaters. The semi-passive system with condensation and water separation is the lightest of the semi-passive methods due to eliminating dessicant requirements and reducing heater requirements. The active (circulating fluid) method is the lightest of all the concepts due to lower power required and slightly lighter equipment weight.

The semi-passive dessicant systems have significantly greater volume requirements than the remaining methods due to the large amount of dessicant required. There is little volume difference between the semi-passive method with water separators and the active system.

The semi-passive silica gel method is rated the most reliable since there are minimum operational parts involved with this method. Lithium chloride is rated slightly less reliable because it becomes liquid and the containing method may fail, allowing the liquid to escape. The semi-passive method with water separation is rated about the same as the lithium chloride system. This rating is based upon the possibility that the small openings in the water separator could become clogged with particulate matter. The active system is rated lowest in reliability because of the larger number of moving parts in this system.

The silica gel dessicant semi-passive method is the least complex and requires minimum equipment and, therefore, is least costly. The lithium chloride and water separator systems require more development and, therefore, greater cost. The active method is rated slightly more expensive than the silica gel dessicant method because of the additional cost of hardware.

In terms of development status, the active method of temperature and humidity control is the most developed. This type of system has been flown on past spacecraft programs and will be flown on forthcoming Apollo flights. Passive humidity control systems have been utilized to a lesser extent on past programs and are not considered to be as developed as the active method. The lithium chloride method requires further development to achieve a satisfactory two-phase containment design together with humidity sensor and flow-control mechanisms.

The active circulating system of Concept 3 was selected as the preferred approach.

Contaminant removal: Two concepts, shown schematically in Fig. 49 were considered for toxic contaminant removal.

Concept 1 is a bleed-and-sorbent system where contaminants are eliminated by sorption on treated charcoal beds, or, for those contaminants which are not well sorbed (such as methane), eliminated by overboard leakage.

In Concept 2, contaminants are removed in sorbent beds and by catalytic oxidation. The catalytic oxidizer operates at approximately 750°F. This method eliminates the requirement for overboard leakage to control poorly sorbed contaminants.

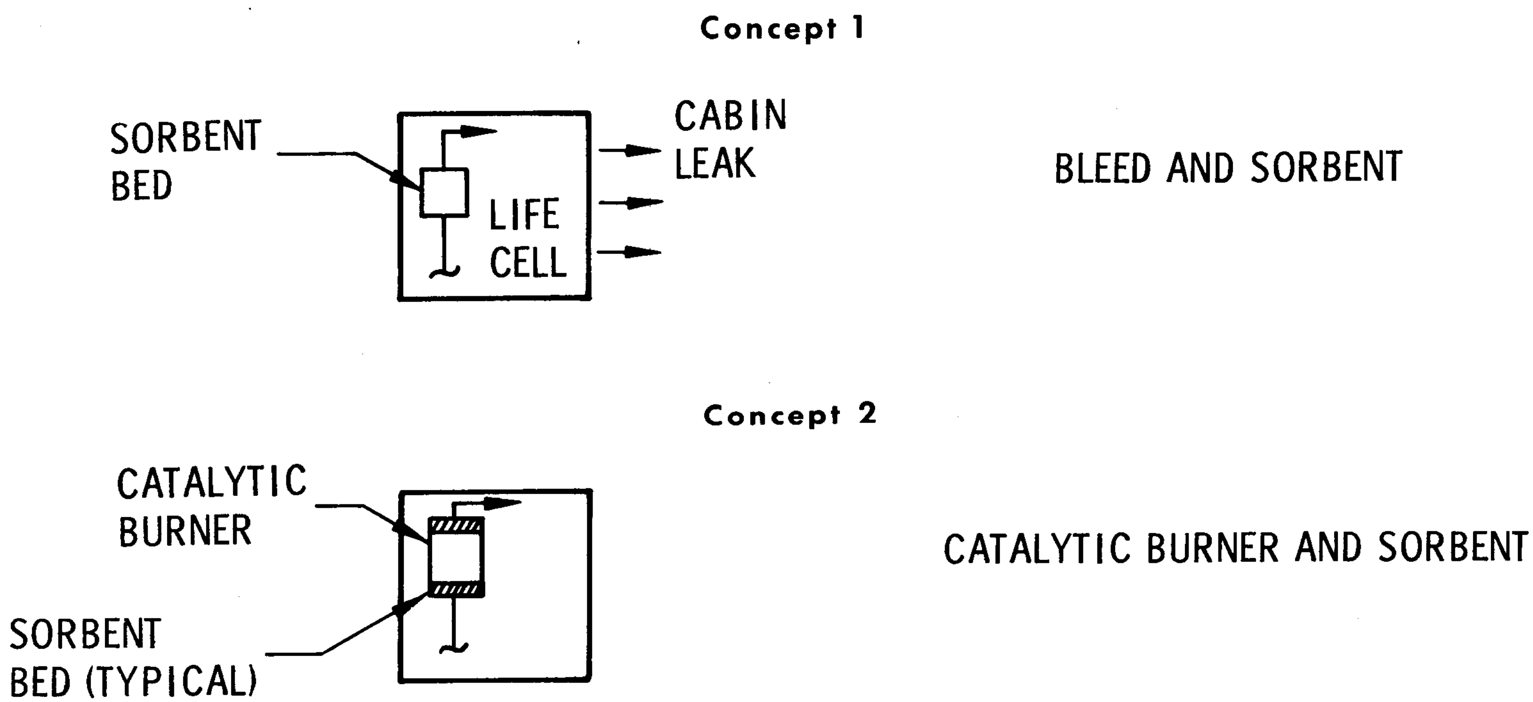


Fig. 49 Contaminant Removal Subsystem Candidates

The evaluation matrix is shown in Table 19. There is no significant difference in weighted values for total equivalent weight and no difference at all in volume. The catalytic burner could experience heater failure and is therefore rated lower in reliability. The bleed method involves less equipment and development and therefore costs less. The bleed and sorbent system has been relied on for past systems and is planned for future Apollo flights; it is rated as the most developed. On the basis of total score, the bleed and sorbent method was selected as the preferred approach.

Metabolic support subsystem. – The metabolic support subsystem involves the feeder and food supply, drinking water, waste management, and cage assembly.

Feeder and food supply: Based on a review of the state-of-the-art and an evaluation of the experience of LMSC and other industrial and government organizations associated with similar experiments, a number of food supply concepts were investigated during the early phase of the study:

- An all fluid diet
- Dry food pellets – prepackaged
- Dry food pellets – random stored

The all fluid approach is extremely attractive in meeting purely mechanical criteria such as space and reliability, and there are no extensive engineering development requirements. Before such a system can be regarded as operational, however, considerable research is needed into the physiological and psychological aspects of a total fluid diet and the long-term effects on the animals. Furthermore, no such all-encompassing liquid food is available at this time. Consequently, the all-fluid appoach was not pursued.

A number of systems have been developed in the past making use of various prepackaging techniques for dispensing dry food to the animals. The main problem in this case, however, is the number of pellets involved. One animal's food supply for the duration of the experiment consists of over 40,000 pellets, each weighing one gram. As a result, systems which have proved promising for short duration experiments become impractical in light of the large number of servings involved. Systems which were considered included a tubular magazine arrangement, similar to that employed to deliver 1,100 spherical pellets on the Air Force BRAVO Program. Also reviewed was the highly complex system developed for the Air Force BIOS Program in which 2400 flat tablets were attached to eight drums of tape and presented in sequence to the animal at eight dispensing units. The application of such principals to the longer duration experiment was ruled out on the grounds of space requirements for storing the pellets, and the extreme complexity and, hence, poor reliability of the conveyor and dispensing mechanisms.

The random storage approach, employing approximately spherical in shape pellets, has been used successfully in the past by LMSC. While such a system calls for some development work on the method of conveying the pellets to the dispenser and storing them in such a manner that they do not become locked together, the

TABLE 19

EVALUATION OF CONTAMINANT REMOVAL SUBSYSTEMS

METHOD	Weight	Power		Total Equivalent Weight		Volume		Reliability		Cost		Conceptual Development Status		Score
	Lb	Watts	Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
Bleed and Sorbent	46	-	-	46.0	19.1	1.49	9.8	4	24	5	20	5	20	92.9
Catalytic Burner and Sorbent	50	3	6.18	56.2	18.9	1.55	9.8	2	12	3	12	3	12	64.7

Remarks

0.07 lb/day leakage is required to maintain the cabin methane concentration less than 4.6%. Anticipated spacecraft leakage is in excess of this (0.3 lb/day).

mechanisms required are relatively simple and the design is not limited by the number of pellets to be stored. Consequently, effort was concentrated on various random storage arrangements.

In developing concepts for the food container and its allied systems, attention was directed towards two primary objectives:

- A reliable conveyor system to ensure that a pellet is always available in the dispenser when the animal gets the feeding signal.
- Some means of ensuring that the pellets are always in the area of the conveyor system and do not become locked together in the upper part of the container.

The two concepts investigated are shown in Fig. 50 . Concept 1 uses a conveyor belt system which forces the pellets down a tube and into the dispenser. A cam-operated leaf spring agitates the pellets locally and directs them into the conveyor. Several concepts for driving the pellets down the container were studied. These included an inflatable gas bag and a piston operated by gas or a spring. The arrangement shown is a piston pulled down the cylinder by a wire attached to a constant torque mechanical drive unit. This offers the advantage of light weight, system reliability, and a constant force for the duration of the experiment. Locking of the pellets in the upper part of the container is prevented by a jet of nitrogen which enters the lower part of the cylinder causing the piston to back off. After agitating the food pellets, the gas escapes through a relief valve in the piston and enters the life-support environment of the vehicle.

Concept 2 utilizes a principal which has proved successful in previous experiments. The bottom end of the cylinder is a rotating plate with 40 recepticles around the periphery in which the food pellets are collected. The plate rotates one-fortieth of a revolution each feeding cycle, and a lever delivers a pellet to the dispenser from each receptacle in turn. The center portion of the rotating plate is a cam surface and a similar surface is provided on the piston. The two surfaces are mechanically connected so that as the system rotates, a rising and falling rotary motion is imparted to the pellets, preventing a jam condition from developing. As in the previous concept, a constant torque device maintains the force on the piston as it directs the pellets down the cylinder towards the conveyor system.

The evaluation of the two feeder concepts is shown in Table 20. Preliminary study indicated a small weight and volume advantage in favor of the concept employing the rotating plate conveyor. The principal basis for comparison, however, was that of reliability. The receptacle plate progresses 40 times to make one complete revolution. Consequently, by the time a given receptacle reaches the dispenser, it has had 39 opportunities to collect a pellet. This offers a reliability advantage over the belt conveyor arrangement that allows only one opportunity per feeding cycle for a pellet to enter the system.

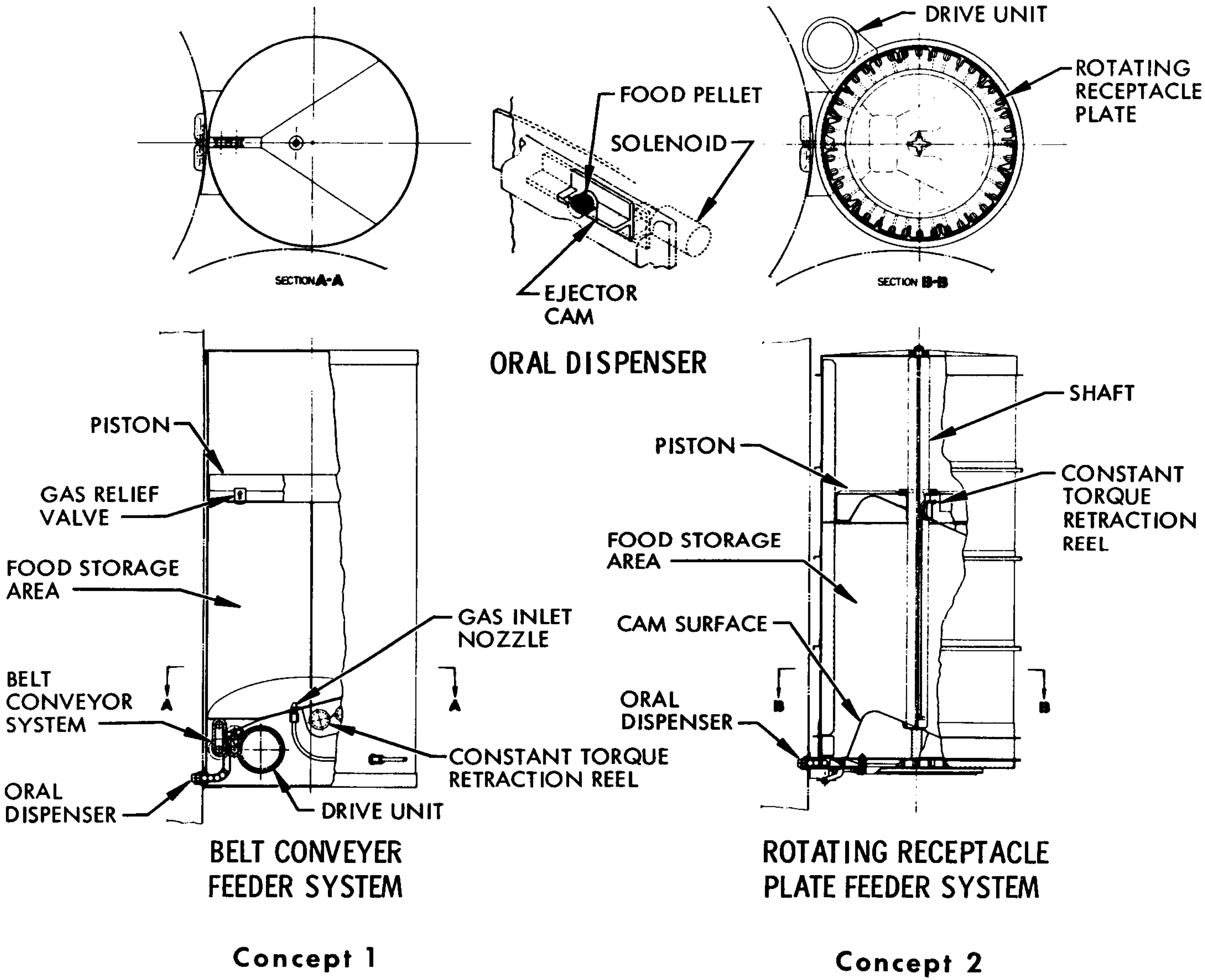


Fig. 50 Candidate Feeder Subsystems

TABLE 20

EVALUATION OF FEEDER SYSTEMS

Method	Weight	Power		Total Equivalent Weight		Volume		Reliability		Cost		Development Status		Score
	Lb	Watts	Weight Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
Belt Conveyor	51	.01	.02	51	19	6.5	9.3	3	18	3	12	2	8	66.3
Rotating Plate Conveyor	49	.03	.06	49	19	6.5	9.3	4	24	3	12	2	8	72.3

In either configuration, considerable investigation is needed to ensure that a problem is not created by the food pellets becoming locked in a solid mass that would prevent their moving down the cylinder and entering the conveyor system. The degree to which this is a problem in a container such as is being considered can be determined only by testing. It appears, however, that the mechanical agitation of the cam surfaces in the rotating plate conveyor concept will provide a more positive means of dealing with the problem than would the periodic nitrogen jet approach of the belt conveyor concept. The results of the studies conducted indicate the superiority of the rotating plate conveyor system.

In either case, further study is needed to determine the abrasive effect of agitating the food pellets; it may be necessary to apply a hard coating to the pellets to prevent their grinding away during the time span of the experiment. The mouth-operated dispenser described below is used in both concepts.

The food pellets selected for this program are approximately spherical in shape and nearly 0.5 in. in diameter. During the early phase of the study, two methods were considered for presenting the food to the animal:

- Manual – in which the animal removes the pellet from the dispenser with its fingers.
- Oral – in which the animal grasps the dispensing unit with its jaws and a pellet is automatically ejected directly into its open mouth.

On the recommendation of the Principal Investigator, effort was concentrated on the oral method since it offers less opportunity for the animal to waste the pellet after having extracted it from the dispenser. The dispenser configuration finally selected is shown in Fig. 50.

Drinking water supply and waste water storage: Four candidate concepts were considered for the drinking water supply and waste water storage subsystem. These concepts are shown schematically in Fig. 51a and 51b.

In Concept 1 the drinking water supply is contained in two hydro-pneumatic storage tanks. This type of tank is used since it provides ullage control under zero-g conditions and ensures positive feeding of water to the dispensing system. A separate pressurization system is included in this concept to provide the pressurant for expulsion of water from the tanks at a regulated pressure and flow. Drinking water is supplied to both dispensing systems (one in each animal cage) from one supply tank until it is emptied. At that time the solenoid valves at the outlet of each tank are actuated and the water is supplied from the full tank. Waste water from the environmental control system (ECS) water separator is allowed to flow into a single hydro-pneumatic storage tank by the actuation of a solenoid valve. Water flows into this tank by maintaining a pressure in the storage tank lower than cabin pressure. This pressure differential is maintained throughout the mission by an absolute pressure vent valve. Operation of this valve on the ground will not be required due to the small volume of waste water generated during this period as compared with the volume of

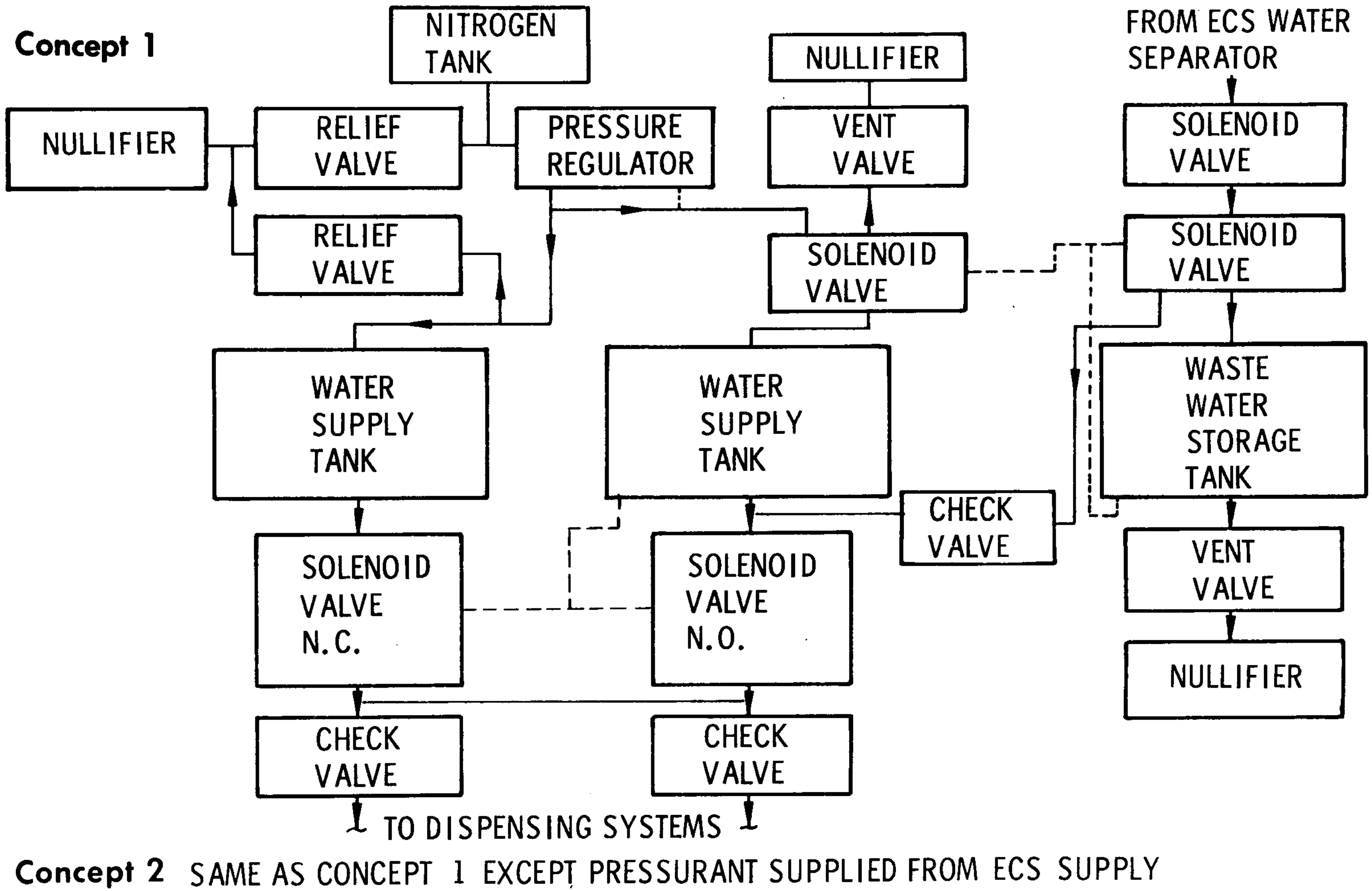
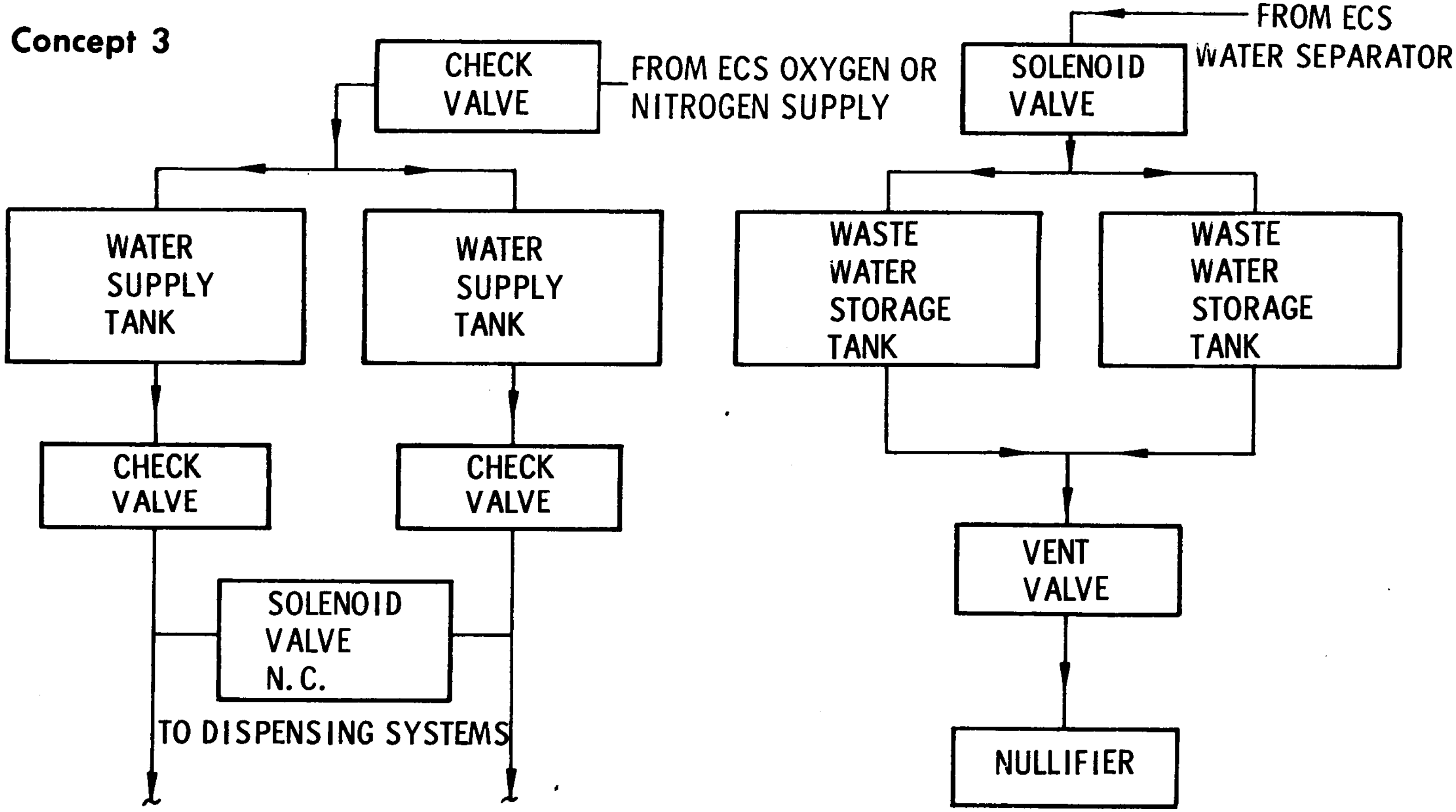


Fig. 51a Candidate Concepts for Drinking Water Supply and Waste Water Storage

Concept 3



Concept 4 SAME AS CONCEPT 3 EXCEPT FOR INTEGRAL GN_2 SUPPLY AS IN CONCEPT 1

Fig. 51b Candidate Concepts for Drinking Water Supply and Waste Water Storage (Cont.)

low pressure air contained on the opposite side of the bladder. When this tank is filled with waste water, two solenoid valves are actuated. One valve removes the pressurant supply from the empty water supply tank and connects this side of the tank to a vent valve which maintains a pressure lower than cabin pressure. The other valve directs the flow of waste water from the ECS to the empty drinking water storage tank.

Concept 2 is identical to Concept 1 except that the nitrogen pressurization system used for the expulsion of water is eliminated and the pressurant is supplied from the ECS. This allows the option of supplying either nitrogen or oxygen for pressurization.

Concept 3 employs separate drinking water supply and waste water storage hardware with hydro-pneumatic storage tanks used to provide ullage control as in the foregoing concepts. Pressurant for the positive feeding of water to the dispensing system is obtained from the ECS as in Concept 2. The waste water storage tanks function as described in Concept 1 but have adequate capacity to accept the total waste water generated during the mission without switch-over to a depleted water supply tank.

Concept 4 is identical to Concept 3 except that a separate nitrogen storage and pressure regulating system for drinking water expulsion is included in the same manner as in Concept 1.

The evaluation matrix for the four candidate concepts is shown in Table 21. All concepts have similar weights and a total volume spread of less than 27 percent. Electrical power required to operate the solenoid valves and the weight penalty associated with this power have not been included in the evaluation because of the very short time that power is used and since the valves operate once only during the mission. Latching solenoids which remove power once actuated were the only type considered for this application. The rather complex valving and controls required to permit dual usage of the drinking water storage tank as a waste water storage tank results in lower system reliability than a system not requiring this dual usage. The addition of a separate pressurization system to either an integrated or a separated drinking water supply and waste water storage system also results in a decreased overall reliability when an existing (ECS) pressurization source is available for this purpose, especially when the ECS can provide pressure from either its nitrogen or oxygen supply. The separated system of Concept 3 offers another reliability advantage to the drinking water supply system by permitting the installation of a normally closed solenoid valve between the outlets of the two supply tanks permitting feeding of both animals from one tank should the water in the other tank be unavailable for any reason. Considering all factors, Concept 3 was selected as the preferred approach.

Drinking water dispensing: Three candidate concepts were considered for the drinking water dispensing subsystem. These are shown schematically in Fig. 52

TABLE 21

EVALUATION OF DRINKING WATER SUPPLY AND WASTE WATER STORAGE SUBSYSTEMS

METHOD	WEIGHT	POWER		TOTAL EQUIVALENT WEIGHT		VOLUME		RELIABILITY		COST		DEVELOPMENT STATUS		SCORE
	Lb	Watts	Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
1	106.5	-	-	-	17.9	20.4	8.0	2	12	3	12	3	12	61.9
2	97.4	-	-	-	18.0	20.2	8.0	3	18	3	12	3	12	68
3	98.7	-	-	-	18.0	27.4	7.4	4	24	4	16	3	12	77.4
4	107.8	-	-	-	17.8	27.6	7.3	3	18	3	12	3	12	67.1

TABLE 22

EVALUATION OF DRINKING WATER DISPENSING SUBSYSTEMS

1	6.6	-	-	-	19.9	0.021	10	2	12	3	12	3	12	65.9
2	9.6	-	-	-	19.8	0.03	10	3	18	3	12	3	12	71.8
3	8.7	-	-	-	19.8	0.026	10	4	24	4	16	3	12	81.8

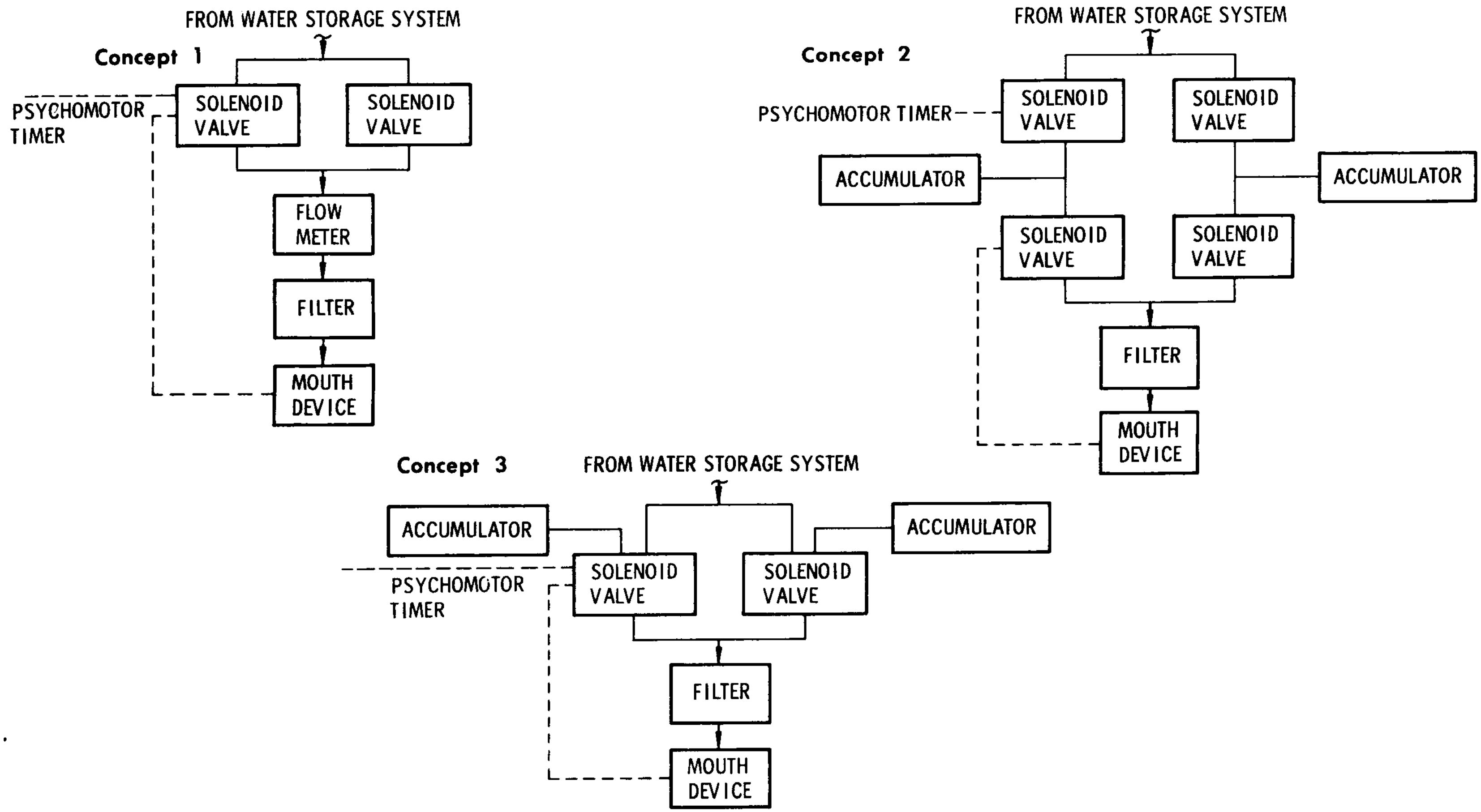


Fig. 52 Candidate Concepts for Drinking Water Dispensing

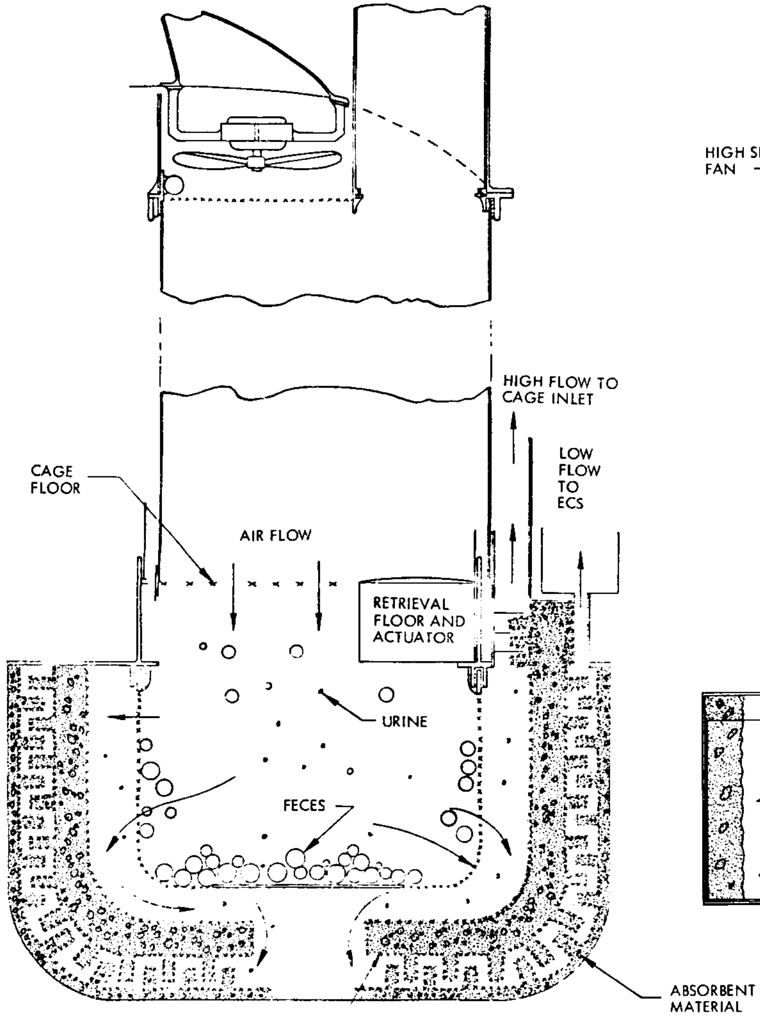
Concept 1 is the simplest candidate and consists of a normally-closed spring-loaded solenoid valve, a flow meter, a filter, and a mouth device. The solenoid valve is enabled by the successful completion of a psychomotor task. Closure of the mouth-device switch results in delivery of drinking water to the animal. The quantity is determined by a timer in the enabling circuit. The use of a flow meter to accurately measure the flow of up to 2 cc of water event for a period of one year is also required. A filter to prevent contamination of the water supply from the mouth device end is also incorporated.

Concept 2 incorporates an additional normally-closed solenoid valve and a spring-loaded bellows aliquot accumulator. In operation, the upstream solenoid valve is energized by the psychomotor programmer and permits charging of the aliquot accumulator, and, when full, the valve is deenergized. Upon successful completion of the psychomotor task, the downstream valve is enabled and upon closing of the mouth-device switch, the water in the aliquot accumulator is delivered to the animal. The flow sensor, using the aliquot water as a conductor monitors and verifies the water delivered to the animal.

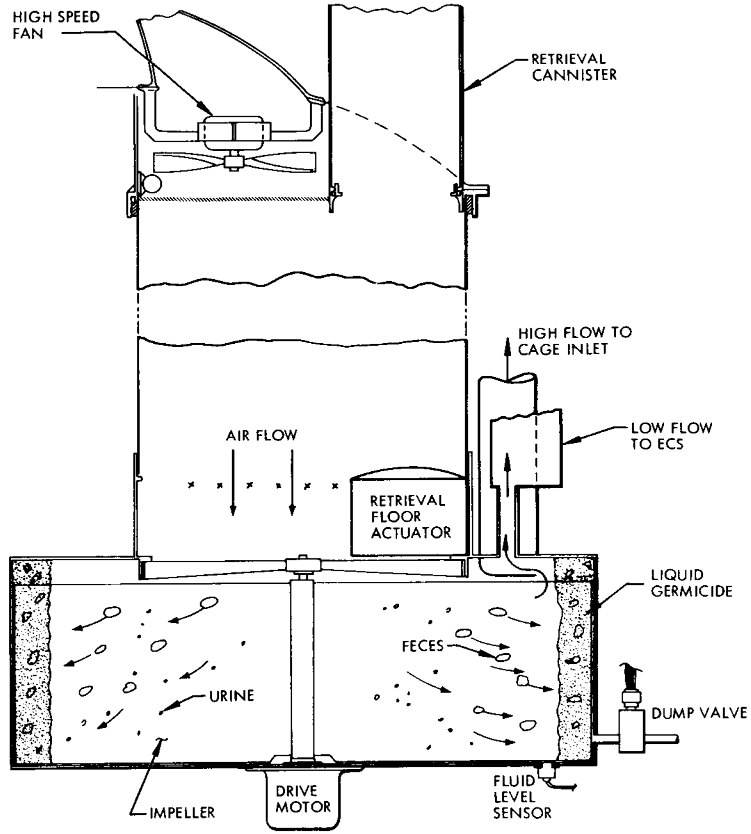
Concept 3 uses an aliquot accumulator and a flow sensor on the mouth device, but a 2-way 3-position normally closed solenoid valve is used to both charge the accumulator and, when actuated, deliver the aliquot to the animal. This system is quite similar to Concept 2 except that it is not possible to have a continuous flow of water from the supply to the animal.

The evaluation of the candidate water dispensing systems is shown in Table 22 Concept 1 received a low reliability rating even though it is quite simple. This is because it is possible to dump the water from a supply tank into the lifecell in the event the solenoid valve fails in the open position. Concept 2 is perhaps the most complex of the three candidates and, although more reliable than the first system, requires the operation of two valves for the delivery of one aliquot of water. It is also possible, in the event of failure of both valves in the open position, to dump the water supply into the lifecell. Concept 3 is only slightly more complex than Concept 1; however, it delivers a known quantity of water at a controlled pressure each time the solenoid valve is actuated and, regardless of failure of either the solenoid valve or the aliquot accumulator or both, the supply water cannot be dumped into the lifecell. Therefore, primarily on the basis of reliability, Concept 3 was selected as the preferred approach.

Waste management: Waste management is considered to be one of the most critical design areas because its operation requirements are both unique and formidable, and also because its proper functioning directly affects the health and well-being of the animal. The selected system must work equally well and with a high degree of reliability on the launch pad as well as in the weightless condition. Four candidate concepts were investigated which meet these requirements with varying degrees of confidence (Figs 53a and 53b).

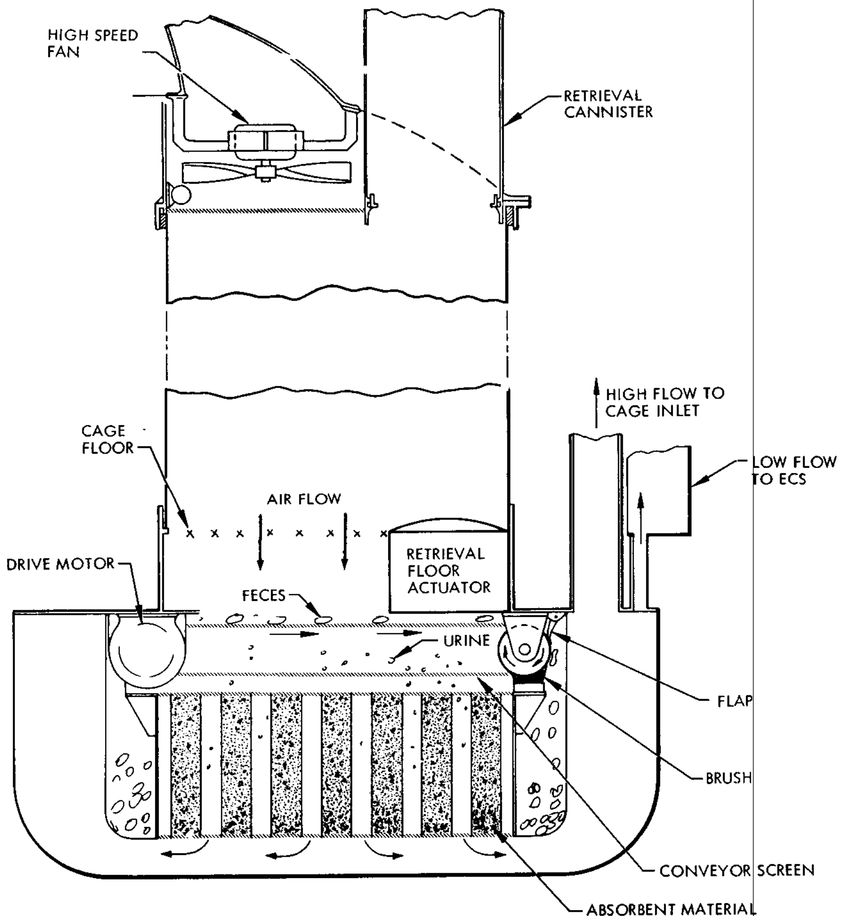


Concept I

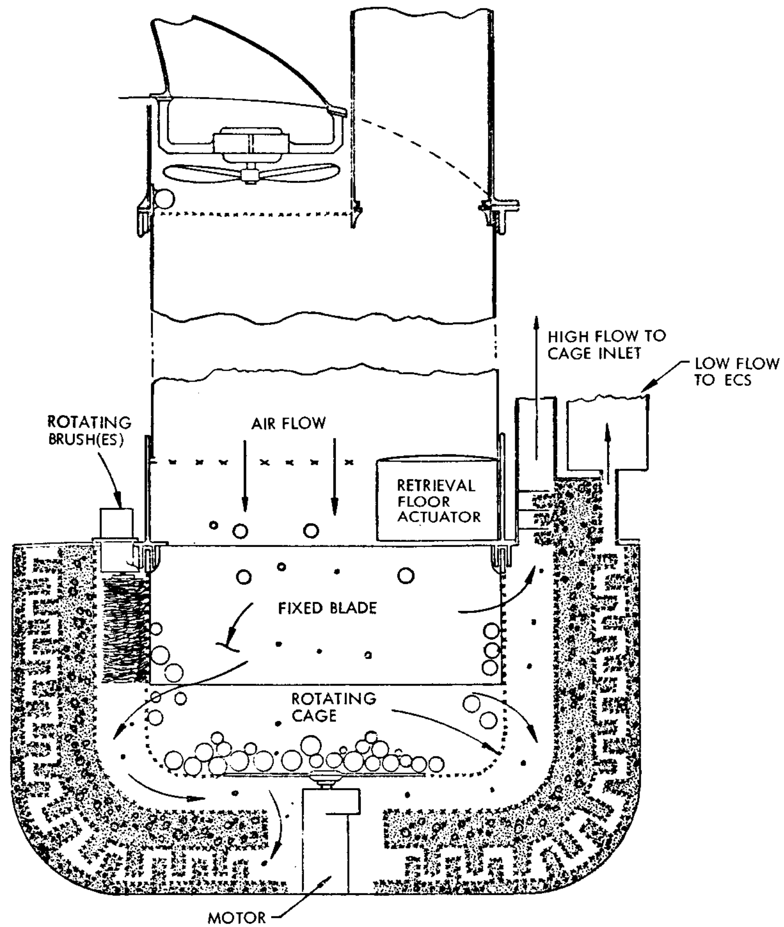


Concept II

Fig. 53a Waste Management Subsystem Candidates



Concept 3



Concept 4

Fig. 53b Waste Management Subsystem Candidates (Cont.)

In all concepts, air flow is used to direct waste material into a receptacle below the cage in the absence of gravity. Periodically, a high-flow fan located at the air inlet above the cage will be turned on, directing a high volume of air (~2000 cfm) through the cage to force waste material entrapped in the cage area to the waste management area below the cage. In Concepts 1, 2, and the first two modular developments of Concept 3, a germicide will be introduced into the waste management area directly above the collected fecal material at the time the high-speed fan is turned on. This will control harmful bacterial growth.

Concept 1 is semi-passive in that high-and low-air flow is used to direct waste into the receptacle below the cage floor. The fecal material is collected in a mesh basket. The urine-laden air is passed through the mesh basket where the air flow is split in two directions. One direction is relatively high-volume air flow where the air is directed across the surfaces of an absorbent material and then directly exposed to the urine. As the air flows across the urine in the absorbent material, the urine is evaporated. The absorbent material is pretreated with a germicidal chemical to control bacterial growth. The low-flow air is then directed through the lithium hydroxide and charcoal beds, a water separator, and then to the inlet area at the top of the cage. This system provides maximum waste disposition with a minimum of mechanics while on the launch pad as well as in orbit.

Concept 2 is based on centrifugal separation of the wastes. Several inches of liquid germicide are placed in a receptacle below the cage. In the receptacle are rotating paddles operated by a motor beneath the receptacle. While on the launch pad, the paddles are stationary and the waste material drops through the grid in the cage floor and into the liquid. During the launch mode just prior to initial orbit insertion, the paddles will be set in motion, forcing the liquid up and out against the sides of the receptacle. The paddles will operate continuously during the mission at a low rpm. The paddles will be counterrotating to prevent perturbations to the stability of the spacecraft. Air flow will direct waste material into the receptacle. The paddles will be designed with a certain amount of helix to aid in directing the waste material into the germicidal liquid. When the waste material builds up to a predetermined level, a fluid level sensor will open a solenoid valve, and a portion of the contaminated liquid will be dumped into a storage tank. The discarded liquid will then be replaced with an equal amount of germicidal chemical.

Concept 3 has a wire mesh conveyor belt that covers the entire area below the cage floor. Below the mesh conveyor is an array of absorbent material pads containing a germicide. Air flow directs the waste material through the cage floor where the fecal material is deposited on the mesh conveyor. Urine is passed through the mesh and directed across the absorbent pads. The urine, picked up by the absorbent material, is evaporated into the air stream and removed in the humidity control water separator. An electric motor periodically moves the conveyor, depositing the fecal material in receptacles. The mesh conveyor is rotated past a flap and brush to clear the mesh and prevent its becoming clogged so as to prevent passage of urine-laden air. The direction of the conveyor is alternately reversed to obtain a reasonable equal distribution of fecal material.

Concept 4 is unique in that it lends itself to a predetermined sequence of modifications as deemed necessary by findings resulting from the developmental phase of the program. Utilizing the basic system as defined in Concept 1, the fixed fecal collecting basket is replaced with a rotatable one. Inside the basket is a fixed scraper. The basket is periodically rotated to allow the scraper to clear the upper portion of the basket to ensure adequate air flow. The second evolutionary step was considered as being a potential requirement because of the possibility of the scraper causing the fecal material to be embedded in the mesh basket. A solution to this problem was to add a rotating brush outside the basket which would maintain a clear passage through the mesh. A third modification was the addition of a warm water and germicidal flush. Using waste water mixed with a germicide, the cage would periodically be washed down. As one of the behavioral tasks, the animal could be trained to go up inside the retrieval capsule where he would actuate a switch on command. This switch would start the high-speed fan and open a valve to allow flow of the water and germicide which would enter the cage area through a header directly below the high-speed fan. The flush system would be automatically turned off after a predetermined time interval. This would also signal the animal to come out of the retrieval canister. Flush water would be absorbed in the absorbent material and later evaporated, condensed, separated, and delivered to the waste water storage system. The design approach to this evolutionary process is to start with the simple basic Concept 1 and to add the above modular refinements only as they are found by development testing to be required.

The evaluation of the four candidate concepts is shown in Table 23. Weight, volume, and electrical power requirements calculated for the candidate waste management systems indicated a wide variation in these values. The values generally increase with a correlating increase in complexity, particularly in case of the power requirements. The motor driven schemes are further burdened by the high-speed fan that is common to all concepts. Concept 1 is the simplest and lightest because it is essentially passive and, therefore, more reliable than Concepts 3 and 4. Concept 2 is considered less reliable than Concept 1 because of the rotating paddles, level control, and interface with waste water storage required with this concept. The conveyor system used in Concept 3 will allow the fecal material to dry by evaporation before it is stored in the containers at the ends of the conveyors, and the bacterial growth problem will be reduced if not eliminated. However, the power required to drive the conveyor belts is quite high, and the reliability of such a system is rather low. Concept 4 also suffers from the complexity of the waste collection basket and brush drive motors and the associated lower reliability rating.

Because of its low weight and volume, high reliability, and relatively high development status, the simple passive waste management system described as Concept 1 has been selected for further design during the preliminary design phase of the study. The inherent development potentials, outlined in the description of Concept 4, constitute the other major factor in the selection of this system. If necessary, during the hardware development phase of the program, the system may be modified in the progressive steps outlined. Such modification will not require extensive redesign and will not affect the cage or lifecell basic configuration.

TABLE 23

EVALUATION OF WASTE MANAGEMENT SUBSYSTEMS

METHOD	WEIGHT	POWER		TOTAL EQUIVALENT WEIGHT		VOLUME		RELIABILITY		COST		CONCEPT DEVELOPMENT STATUS		SCORE
	Lb	Watts	Weight Lb	Lbs	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
I – Passive	100.0	4	8.24	108.3	17.8	5.55	9.4	3	18	3	12	3	12	69.2
II – Liquid Gemicide	324.5	10	20.6	345.1	13.1	4.46	9.5	2	12	3	12	2	8	54.6
III – Conveyor	158.0	20	41.2	199.2	15.1	8.70	9.2	2	12	2	8	2	8	52.3
IV – Modified Passive	125.0	12	24.3	149.3	17.0	6.4	9.4	2	12	2	8	2	8	54.4

Cage assembly: Several geometric shapes were explored to arrive at a suitable cage configuration. However, very early in the analysis, the advantages of a cylindrical configuration became apparent from the standpoints of equipment placement, structural integrity, and fewer corners in which waste matter could accumulate. The cylindrical shape was, therefore, selected as the preferred approach. This choice was later confirmed by its compatibility with the retrieval method and waste management system.

The selected concept for a cage configuration is shown in Fig. 54. The basic cylinder is 34 in. in diameter by 48 in. high to give a volume of 25.2 ft^3. The structure consists of a 0.005-in. thick corrosion resistant steel liner supported by an external frame structure of aluminum rings and forgings. The smooth walls of the liner will help to minimize waste deposition. The selected concept provides sufficient volume for animal activity and is sturdy enough to resist damage by the animal. It also forms a part of the animal retrieval system.

Lifecell. – The lifecell subsystem forms the basic structure of the spacecraft and the pressure vessel which contains the metabolic support system. Consideration was given to the size, shape, and general arrangement of the lifecell to ensure a workable interface with the many subsystems which are required to support the primates. The design approach taken in the study of lifecell candidates was to work from the inside out; that is, the basic needs of the primates were analyzed and the equipment defined. This provided a envelope which must be contained within a given size and shape of pressure vessel.

With the basic payload envelope defined, various shapes of lifecell candidates were studied with emphasis placed on structural integrity, seal technique, retrieval-canister interface, ground handling, easy installation/removal of the metabolic support system, interface with the spacecraft, weight, and production costs. Early in the evaluation it became apparent that the design and placement of the cage system had a strong influence on the shape of the lifecell. Another strong factor in the cage arrangement was its interface with the retrieval system. Three configurations were considered as shown in Fig. 55.

The first design explored was based on an elliptical lifecell with elliptical dome ends. The elliptical shape evolved as a result of placing two cylindrical cages tangent to each other at the lifecell center. While this shape lends itself to good utilization of space within a given area, it does not provide a good structural shape as a pressure vessel. Preliminary stress analysis revealed that circumferential rings of considerable weight would be required to carry the pressure loads. The elliptical shape is not easy to fabricate and tooling cost would be high.

Keeping the same cage arrangement, the next configuration studied employed a cylindrical section with dome ends. This shape is more desirable as a pressure vessel and provides good volume-to-weight ratios. Cage placement can be relatively close to the outside diameter of the lifecell which provides a large area at the center of the lifecell for metabolic support equipment installation. Circumferential rings at

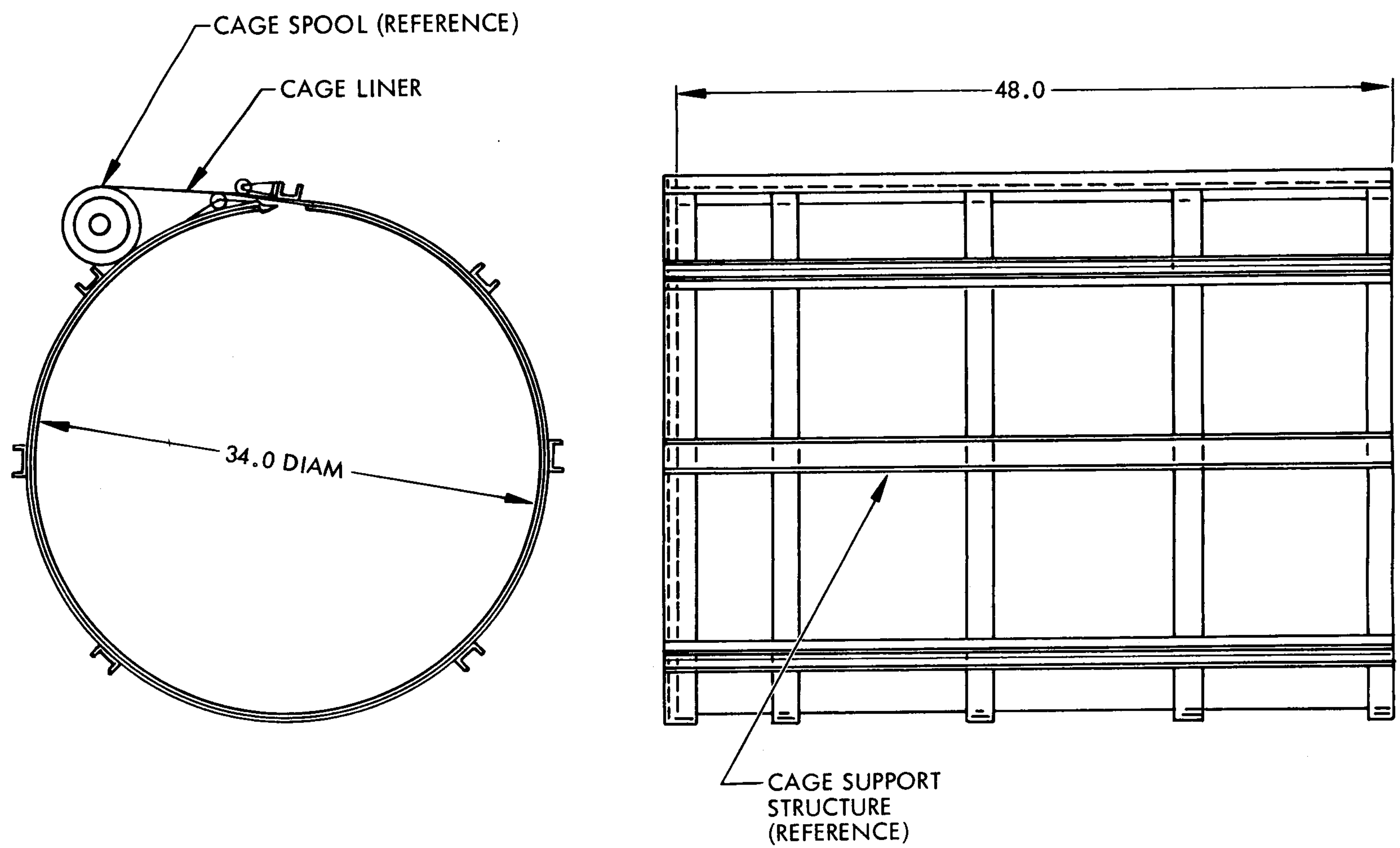


Fig. 54 Selected Cage Configuration

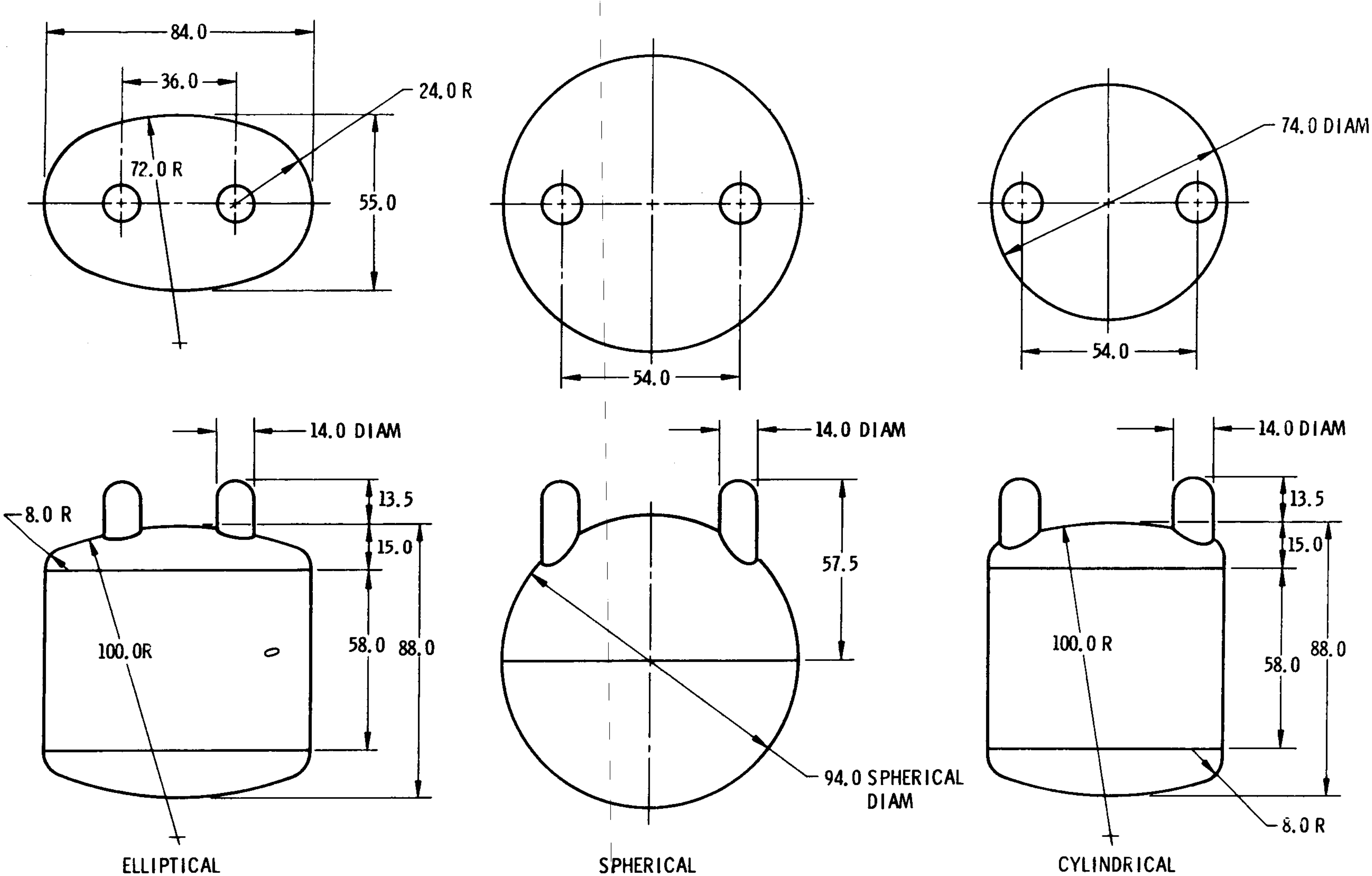


Fig. 55 Candidate Lifecell Configurations

the dome interface with the cylindrical section to provide attach points for the spacecraft structure. The bottom end dome is removable for installation of the complete metabolic support system. Common to all lifecell configurations, one dome end is held in place by the use of a Marman V-band clamp using an O-ring seal. Lockheed experience with this type of clamp/seal arrangement has proven most satisfactory on two previous lifecell designs, both of which were subjected to a hard vacuum, one for 30 days duration.

A sphere was selected for the third candidate system. The sphere is the best structural shape for a pressure vessel and would seem to be a good candidate for a lifecell. However, studies indicated that to house the cage arrangement and metabolic support system, a 92-in. diameter sphere would be required. While this shape provides the most volume, not all of it is useful volume. That is, parts and components must now be designed to fit within the spherical shape to best utilize the volume. Rings and intercostals are required to mount equipment and carry the load to the structural shell. Additional rings are required to provide mounting interfaces for the spacecraft. While the spherical shell is structurally efficient, considerable structure is required both internally and externally to distribute the loads into the shell.

The evaluation of the candidate lifecell configurations is shown in Table 24 . The weight figures for all candidates represent total lifecell subsystem including structural shell, V-band clamp, internal-support structure, external rings, and spacecraft interface fittings. Reliability was equally high for each of the lifecell candidates. Sound structural engineering and test would produce a reliable unit for each concept. The main tradeoff to consider is the weight of structure to ensure the reliability.

Cost of development and fabrication would be different for each concept. The elliptical design would require considerable tooling to form the dome ends and the elliptical center section; rings would be costly to machine. Due to these factors, the fabrication cost is considered high for the elliptical lifecell.

Tooling and production costs for the cylindrical shaped lifecell would be nominal. Tooling is required to spin the dome ends; however, since the dome ends are symmetrical, only one die will be required. The rings being round, rather than elliptical, are much less expensive to machine.

The spherical lifecell is rated between the elliptical and cylindrical designs. Cost in tooling will be higher due to the size of the dies required to spin each hemisphere. Interior structure, such as intercostals, are more difficult to form due to the compound contours of the spherical surface.

Concept development status of both the cylindrical and spherical designs are rated the same. Both shapes are common to pressure vessel design and present no development problems. The elliptical pressure vessel is not as common a design and would require some development testing.

TABLE 24

EVALUATION OF LIFECELL CONFIGURATIONS

METHOD	WEIGHT	POWER		TOTAL EQUIVALENT WEIGHT		VOLUME		RELIABILITY		COST		CONCEPT DEVELOPMENT STATUS		SCORE
	Lb	Watts	Weight lb.	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
ELLIPTICAL/ DOME END	225	0	0	225	15.5	171	-7.1	5	30	1	4	3	12	54.4
CYLINDER/ DOME ENDS	150	0	0	150	17.0	219	-11.9	5	30	3	12	5	20	67.1
SPHERE	200	0	0	200	16.0	236	-13.6	5	30	2	8	5	20	60.4

The matrix total score indicates the best candidate lifecell configuration to be cylindrical design. The design offers a good compromise between the optimum pressure vessel structural shape and one which allows maximum useful volume for installation of the ECS and metabolic support equipment without special packaging requirements.

Special equipment. – Special equipment involves the mass measurement system, behavioral panel, animal retrieval, and retrieval canister.

Mass measurement system: Among the concepts investigated for mass measurement were techniques using an oscillating spring mass, torsional pendulum, centrifuge, impulse momentum, conservation of momentum, linear acceleration, and inertia beam balance. All these concepts require complexity in mechanisms and positive animal restraint. In itself, the degree of animal restraint required to perform mass measurement four times a day is considered an overriding factor in eliminating any of the foregoing methods. Therefore, a nearly passive approach was adopted in which animal handling is reduced to a minimum. The concept is shown in Fig. 56.

The selected concept uses "soft" X-ray radiation. In this system, the animal is positioned between a radiation source and a radiation detector. To take a measurement, a collimating aperture in the shield of the source is opened, and the radiation transmitted through the animal's body is measured by the counter connected to the detector. Knowing the source characteristics and the composition of the animal, the mass is determined by radiation attenuation and geometry.

The concept uses cobalt-57 as the radiation source and a cesium iodide scintillation counter as the detector. The source material is commercially available and several millicuries should be sufficient to allow the accumulation of 100,000 counts. Scintillation counters of the type planned have been successfully flown as experiments on Pioneer VI and VII flights and are being planned for future use on long-duration cosmic ray space experiments. The technique itself has been used successfully to measure propellant quantities, aircraft oil inventory, and lettuce head mass. Radiation exposure to the animal is minimal, being on the order of one millirad delivered to the skin per measurement.

Behavioral panel: The behavioral panel concept was largely determined by the detailed requirements specified by the Principal Investigator. Therefore, no formal tradeoff analysis was made in this area. The concept, based on the specified requirements, is covered in the System Description section of this report.

Animal retrieval: Recovery of the animals from the orbiting spacecraft will require the use of retrieval canisters installed in the lifecell and a means of transferring the animals into these canisters. The canister itself is discussed in following sections of this report. The method of transferring a primate into the canister is complicated by the fact that the system must be capable of functioning properly whether the animal is alive or dead, tractable or recalcitrant. Four candidate concepts were explored and evaluated.

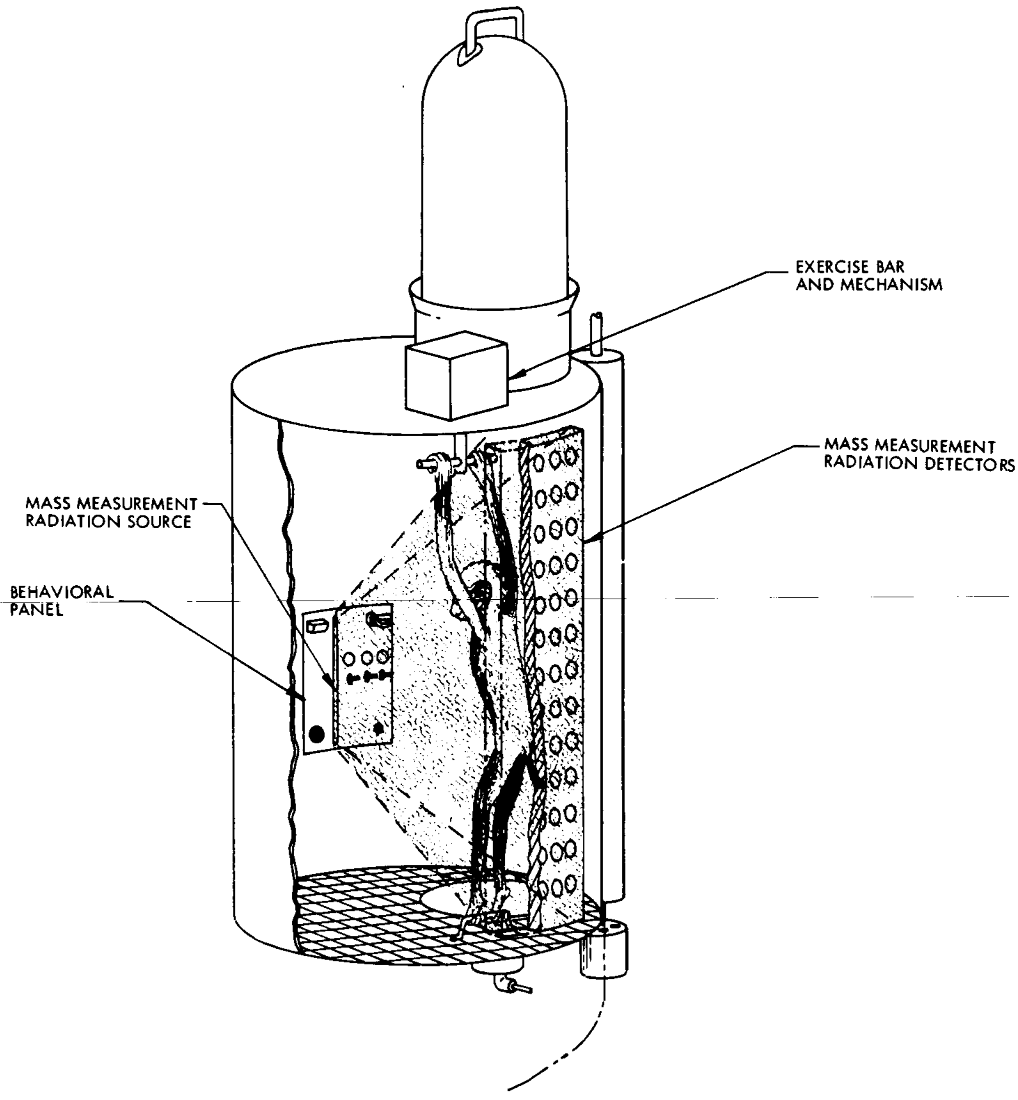


Fig. 56 Mass Measurement – Radiation Technique

Retrieval Concept 1 is shown in Fig. 57 . This concept utilizes the cage wall structure to confine the animal within a small known volume that is cylindrical in shape with the same diameter as the retrieval canister. This is accomplished by rolling the thin-steel cage wall on a motor-driven roller to form a tube leading into the retrieval canister. A domed floor plate is then forced upward through the tube by a telescoping pneumatic actuator located beneath the cage floor. As the domed plate moves upward, the animal is forced into the retrieval canister. At the end of the actuator travel, the domed plate is forced into a seal ring located in the canister and becomes the end closure. The dome will separate from the telescoping actuator on removal of the retrieval canister by the astronaut. The canister is secured to the lifecell at its mating joint by a V-band clamp with a quick release toggle to permit easy removal of the canister by the retrieving astronaut.

Concept 2 is shown in Fig. 58 . This concept uses the same retrieval canister as described in the first concept. However, this method utilizes a pneumatic tubular bladder to raise the cage floor to the top of the cage by admitting air pressure between the floor peripheral sliding seal and the cylindrical cage wall. This air causes the floor to rise due to the area differential between the minimum bladder tube diameter and the floor sliding seal or cage wall diameter. A domed plate forms the center of the floor, and as the floor rises into the conical section joining the top of the cage and the bottom opening of the retrieval canister, the animal is forced into the retrieval canister. As pressure is applied to the ring stiffened bladder tube, the domed floor plate is forced into the bottom opening of the retrieval canister and forms the canister closure in the same manner as described in Concept 1.

Concept 3 is shown in Fig. 59 . This retrieval system utilizes an elevating floor to locate the animal in the upper end of the lifecell where it may enter the canister of its own accord or be forced to enter by a separate action of the retrieval canister. The floor is raised to the upper end of the lifecell by a telescoping pneumatic actuator located under the cage floor. The retrieval canister is the same as that described for Concepts 1 and 2 except that it has an inner piston to which radial L-shaped rods are attached by hinges on its lower skirt. These hinged rods are secured to form the upper end of the cage by a latching ring which, upon actuation, releases the spring-loaded, hinged, L-shaped rods that spring downward through the radial floor rods. Simultaneously, with release of the hinged rods, the canister piston is raised by venting space the air above the piston in the canister outer shell causing the piston to be drawn upward into the outer shell of the canister. This action forces the animal to the center of the floor and draws it upward with the piston into the retrieval canister. When the piston has reached its upper position, a spring-loaded hinged door is released and swings into the closed position and seals the retrieval canister. Fluid and air connections to the inner piston are made automatically when the piston has reached its maximum upward travel. The canister is mounted to the lifecell in the same manner and may be removed by the retrieving astronaut as described under Concept 1.

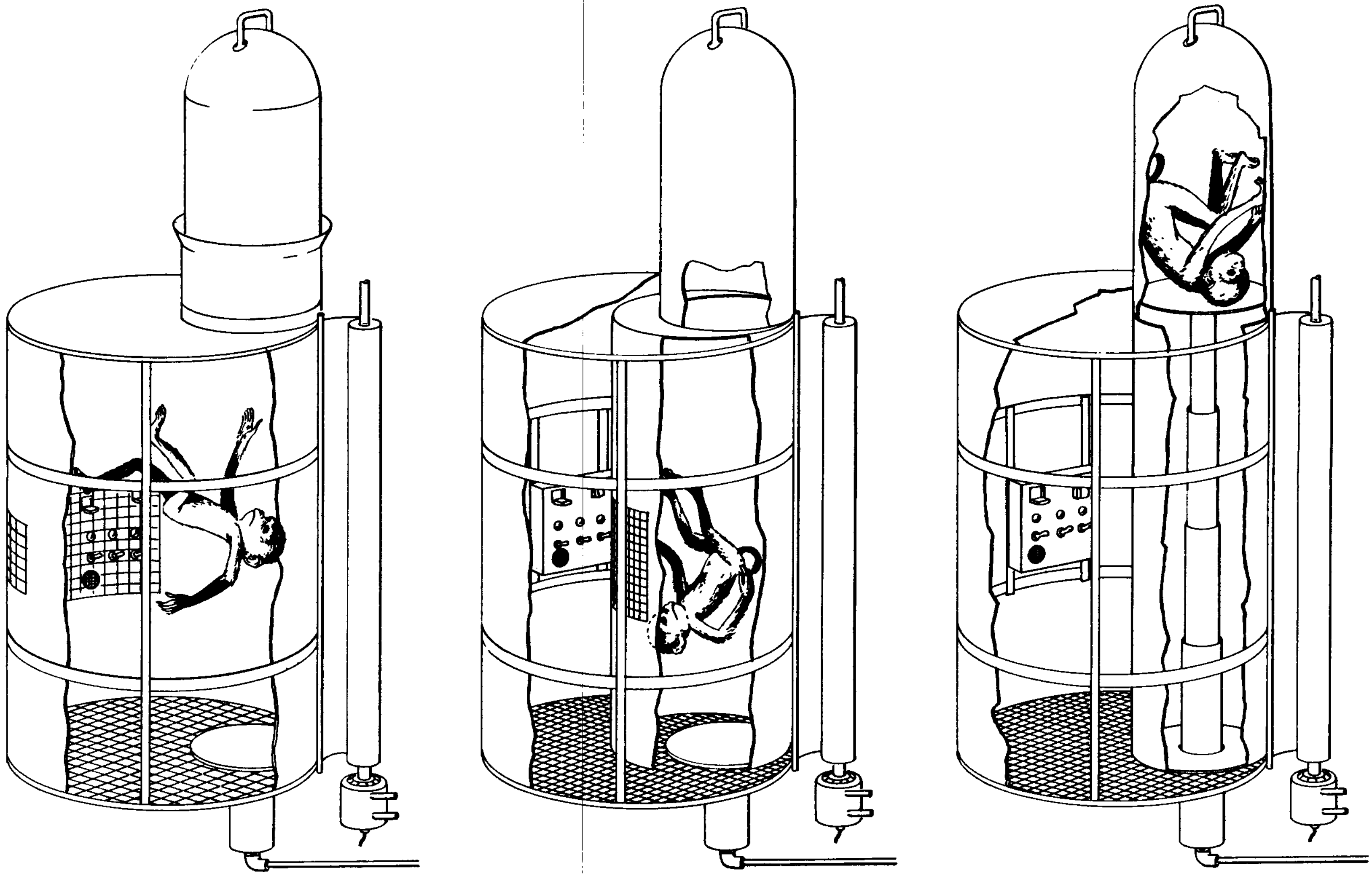


Fig. 57 Animal Retrieval – Concept 1

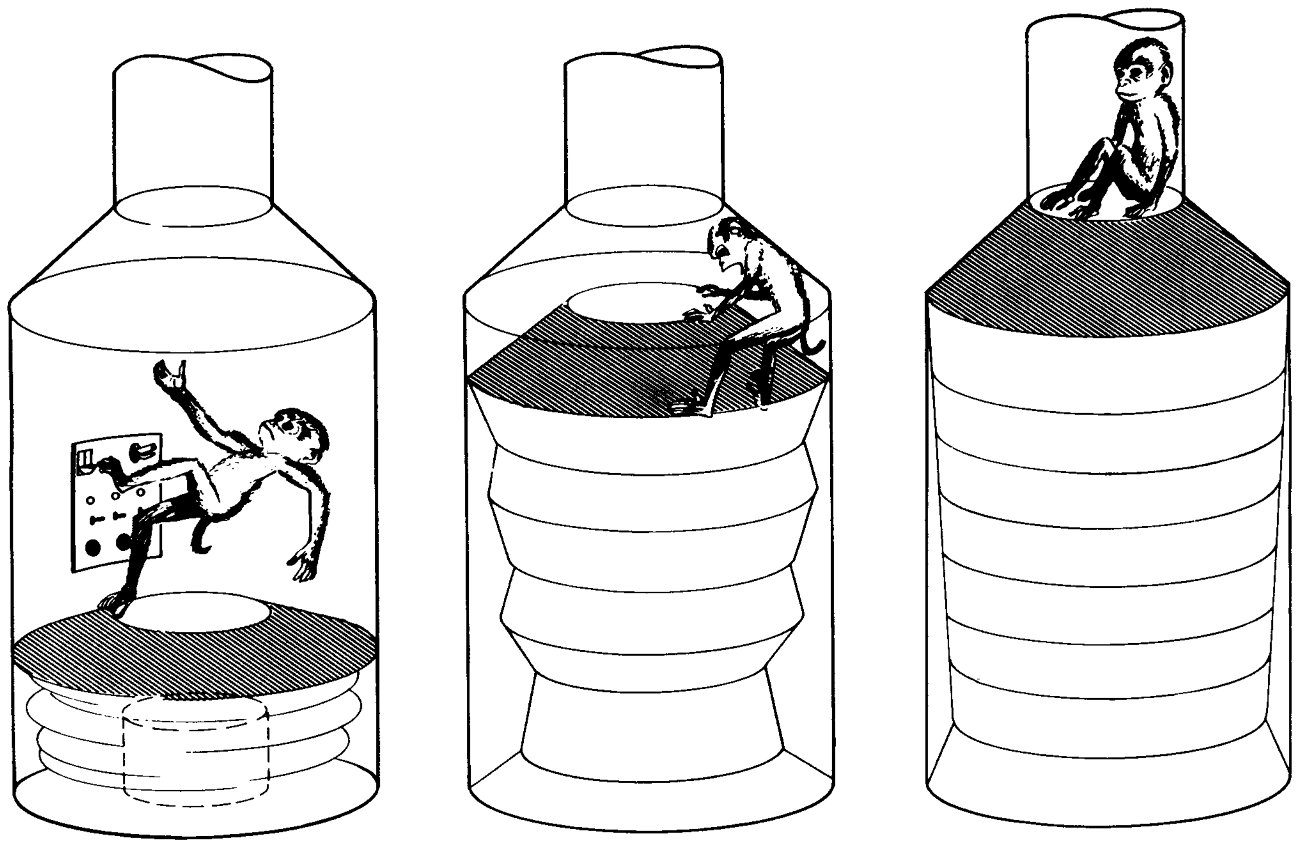


Fig. 58 Animal Retrieval – Concept 2

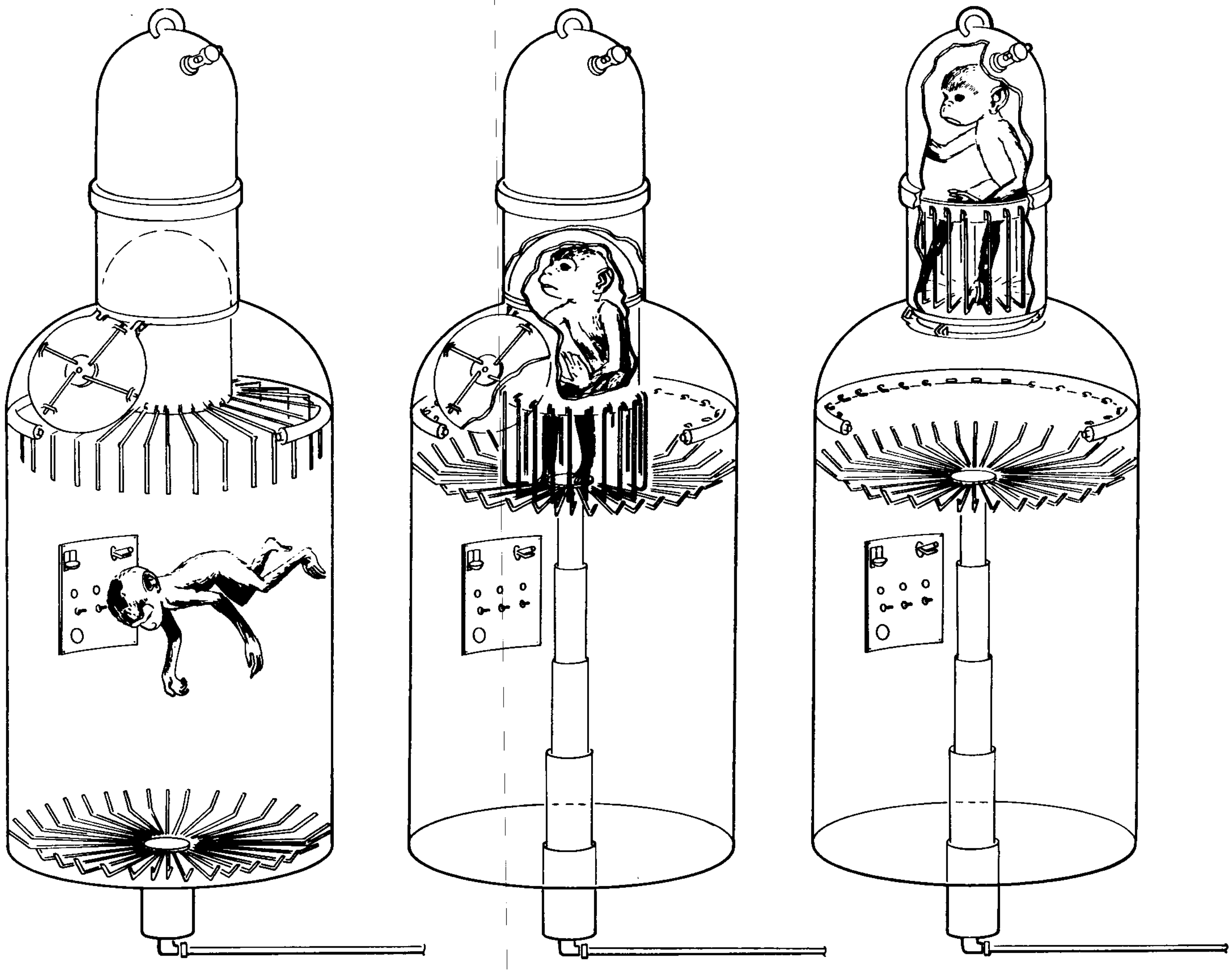


Fig. 59 Animal Retrieval – Concept 3

Concept 4 (not shown) accomplishes insertion of the animal in the retrieval canister by the application of a small force "downward" on the entire spacecraft. This force can be produced by a small solid rocket that is ignited by ground command and results in a small acceleration of approximately 1/4g causing the animal to rise to the top of the cage and into the retrieval canister. A conical section joins the top of the cage and the bottom opening of the retrieval canister and causes the animal to be guided into its bottom opening. The upward motion of the animal is further aided by a flow of air from the lifecell through the canister and to space through a vent valve located in the upper end of the canister. This valve is opened after the rocket thrust has been initiated and is closed by the animal's body striking a trip lever in the top of the retrieval canister. Simultaneously, with the opening of the canister vent valve, a telescoping pneumatic actuator located beneath the floor is actuated and raises a domed floor plate into the bottom opening of the canister to form a sealed closure in the same manner as described for Concept 1. The retrieval canister mounting and removal provisions as well as the fluid preservative system are the same as described for the previous configurations.

The evaluation matrix for the candidate retrieval concepts is shown in Table 25. The weights and volumes listed on the matrix do not include the weight and volume of the retrieval canister since these values are included in the data presented in the retrieval canister fluid and life support system evaluation. The weights and volumes listed here pertain only to those features in the cage and canister peculiar to a specific concept.

While the weight and volume of the concepts presented are significant factors, the ability of a retrieval system to insert an uncooperative, ill, or deceased animal into the retrieval canister and seal it positively is considered the most significant factor in this evaluation. Concepts 1 and 3 are the most positive in this respect, with Concept 3 receiving a lower reliability rating because of the more complex closure door and rod release mechanisms as well as the possibility of a hand or a foot of an ill or deceased animal protruding through the retrieval canister rods and preventing complete closure of the canister door. The ability of an uncooperative animal to resist the action of the pneumatic tubular bladder rising floor and closure of the canister in Concept 2 has resulted in the low reliability rating of this system. These same comments also apply to Concept 4, in addition to the lack of a positive cage volume reduction scheme, resulting in the very low reliability rating given this system. The significantly lower weight of Concept 1, as compared to the other concepts, is due to the very thin steel used for the cage wall as well as a tubular guide during insertion of the animal into the canister. As shown on the evaluation matrix, Concept 1 achieved the highest score and was selected as the preferred approach.

Retrieval canister: Three concepts were evaluated for the retrieval canister. These concepts took into account the requirements for life support of a live animal as well as a preservation method for a dead animal.

Concept 1 is shown in Fig. 60a. This concept utilizes a high-pressure storage tank and associated valving mounted on the retrieval canister for the supply of

TABLE 25

EVALUATION OF ANIMAL RETRIEVAL SUBSYSTEMS

METHOD	WEIGHT	POWER		TOTAL EQUIVALENT WEIGHT		VOLUME		RELIABILITY		COST		CONCEPT DEVELOPMENT STATUS		SCORE
	Lb	Watts	Weight lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
I – Reel	59.4	0.10	0.2	59.6	18.8	1.10	9.9	3	18	3	12	2	8	66.7
II – Bag	169.2	-	-	169.2	16.6	2.12	9.8	2	12	3	12	2	8	58.4
III – Rods	135.4	-		135.4	17.3	0.69	9.9	2	12	3	12	2	8	59.2
IV – Thrust	125.5	-		125.5	17.5	0.83	9.9	1	6	2	8	3	12	53.4

Concept 1

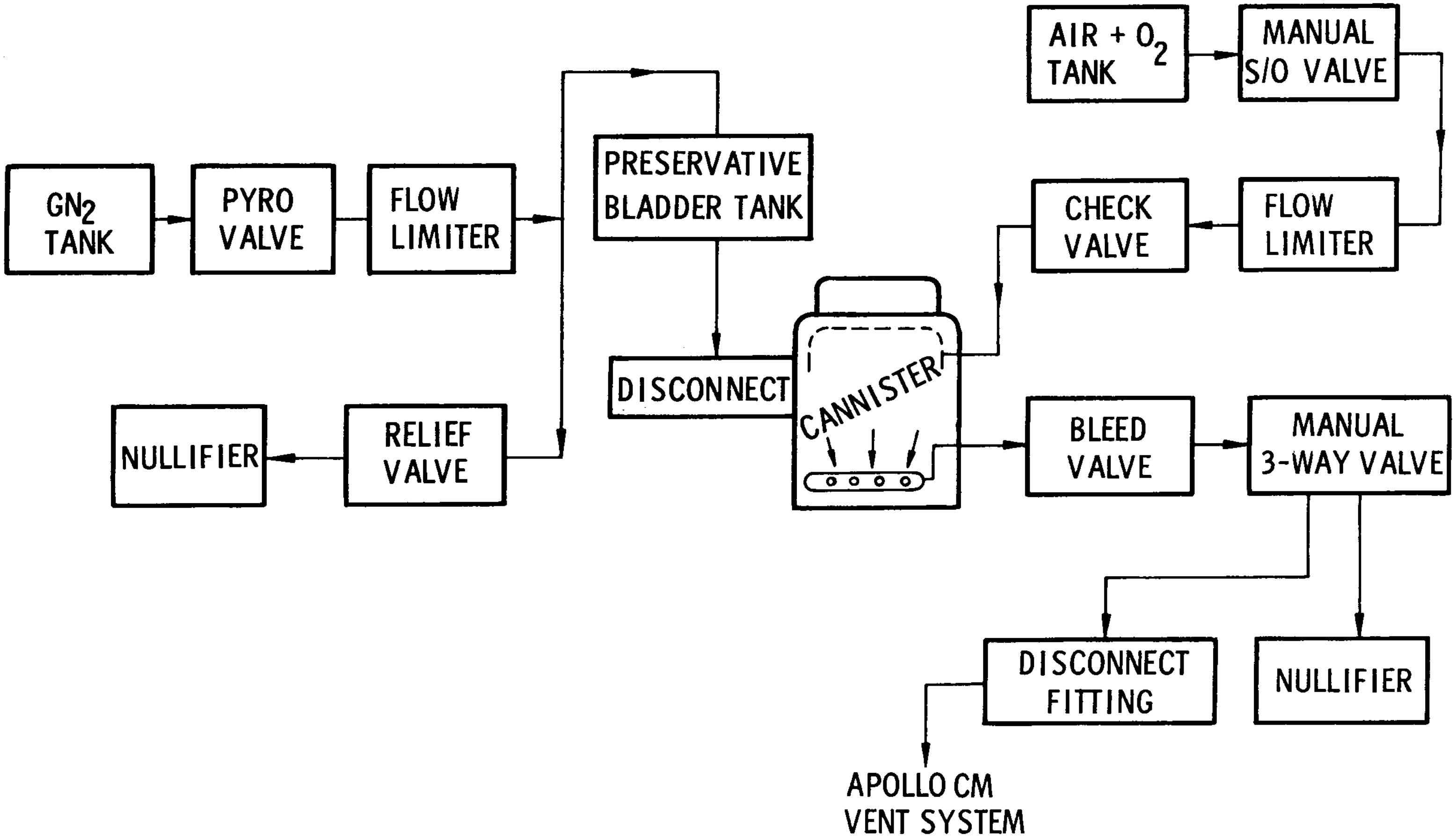


Fig. 60a Candidate Concepts for Retrieval Canister Fluid and Life Support Subsystems

oxygen-enriched air to the animal for 24 hours when it is contained within the retrieval canister. Actuation of the supply is accomplished by the retrieving astronaut by opening the manual shutoff valve located on the outlet of the air supply tank. This enriched air is admitted to the top of the canister through a diffuser plate at a rate in excess of that required by the animal. Excess air, CO_2, and other gasses are vented overboard through a collector ring located near the bottom of the tank, a bleed valve, a manual 3-way valve, and a thrust nullifier, thus assuring the animal a constantly changing air supply. The manual valves permit temporary shutdown of this system during entry and installation in the Apollo Command Module to avoid contamination of the Command Module cabin atmosphere. A vent disconnect fitting is provided to interface with the ACM vent system. When the retrieval canister/command module interface is complete, the manual valves are opened and air flow through the canister is resumed. Fluid preservative for the preservation of a deceased animal is contained in a remotely located hydropneumatic storage tank that is plumbed to the canister through a disconnect fitting which may be removed by the retrieving astronaut. A separate pressurant supply tank, pyrotechnic valve, and flow limiter are provided for the expulsion of the preservative from the hydropneumatic tank into the retrieval canister. Air being displaced by the entering preservative fluid is vented overboard through the previously described vent system. Actuation of this system will be accomplished by ground command.

Concept 2 is shown in Fig. 60b. This system is the same as Concept 1 system except the preservative expulsion pressurant is provided by the lifecell metabolic system nitrogen supply. This permits a total weight savings of approximately 8.4 pounds.

Concept 3 is also shown in Fig. 60b. A passive metabolic support system is employed in this concept which uses KO_2 and LiOH to provide the breathing oxygen required and to absorb the CO_2 produced by the animal. These compounds are contained within the retrieval canister and become activated automatically upon installation of the canister door. The preservative fluid system is the same as that used in Concept 2. However, the bleed valve has been replaced with a low-pressure relief valve to allow venting of the air through a vent nullifier because it is not required during occupancy by the animal. This eliminates the requirement for any interface with Command Module functional systems.

The evaluation of these three concepts is shown in Table 26. Concepts 1 and 2 are very similar with respect to weight, volume, cost, and development status. However, the reliability of Concept 1 is reduced due to the added complexity of a separate preservative expulsion pressurization system. Concept 3 has considerable weight and volume advantage over Concepts 1 and 2 due to the elimination of the high-pressure breathing air supply tank and associated valves, flow limiter, and plumbing. In addition, the elimination of the bleed vent system and the necessity for a functional interface with the ACM vent system has resulted in a higher reliability and lower cost value for this concept. Therefore, Concept 3 was selected as the preferred approach.

Data management subsystem. – Various methods of accomplishing the required objectives for the data management subsystem were investigated. However, very early in the tradeoff analysis phase, it became apparent that the overriding consideration

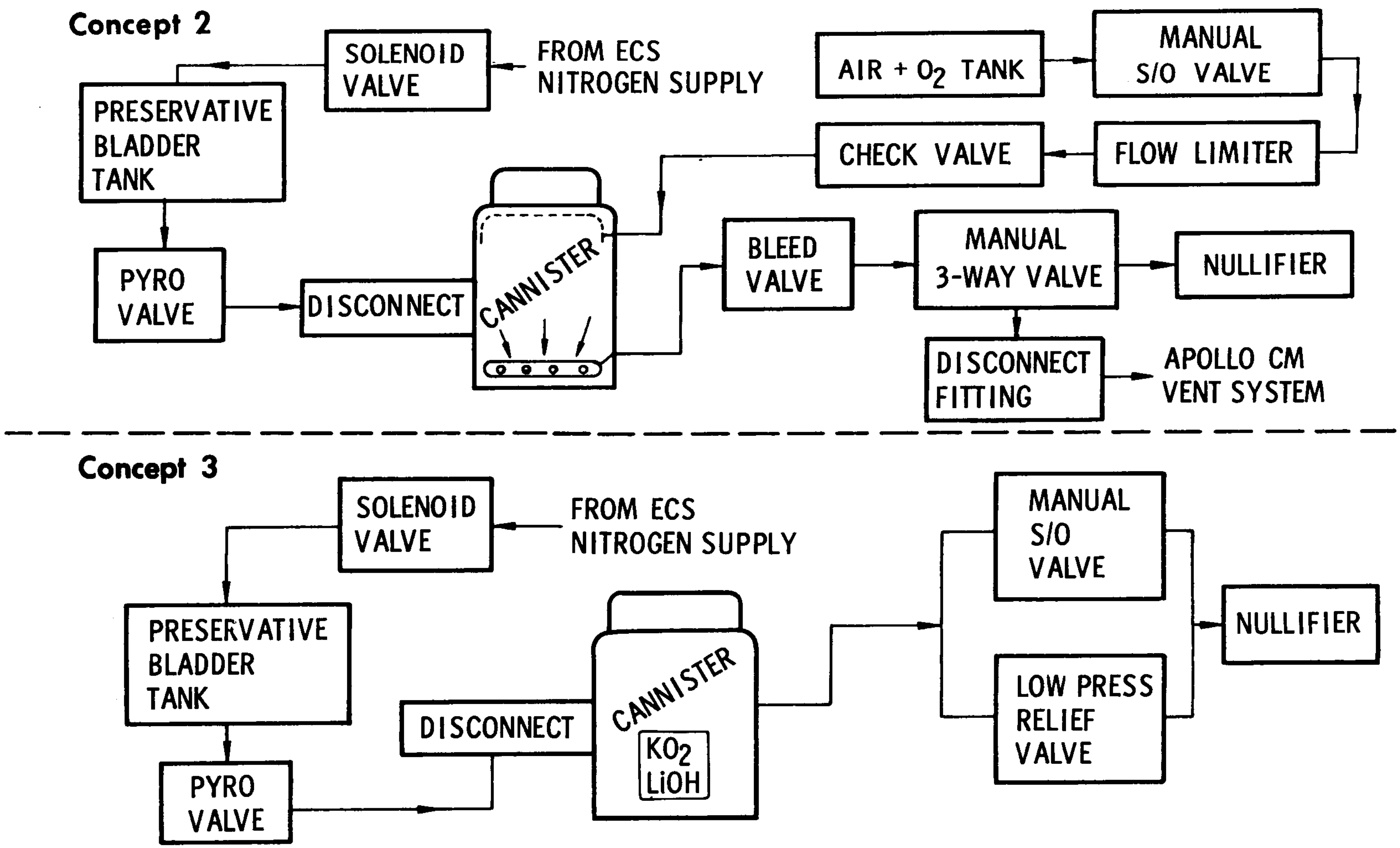


Fig. 60b Candidate Concepts for Retrieval Canister Fluid and Life Support Subsystems (Cont.)

TABLE 26

EVALUATION OF RETRIEVAL CANISTER FLUID AND LIFE SUPPORT SUBSYSTEMS

METHOD	WEIGHT	POWER		TOTAL EQUIVALENT WEIGHT		VOLUME		RELIABILITY		COST		CONCEPT DEVELOPMENT STATUS		SCORE
	Lb	Watts	Lb	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
1	264.4	-	-	-	14.7	11.6	8.8	2	12	3	12	3	12	59.5
2	256.0	-	-	-	14.9	11.3	8.9	3	18	3	12	3	12	65.8
3	175.4	-	-	-	16.5	9.1	9.1	4	24	4	16	4	12	77.6

was the Statement of Work requirement for compatibility of the data management subsystem with the configuration and capabilities of the Apollo ground network. Since at this stage of development, AAP plans are still in the process of change, it is conceivable that the configuration and capabilities of the Apollo ground network will change. However, it is certain that Apollo flight hardware must keep pace with any changes in ground station equipment. In addition, this hardware will be developed and qualified for flight as part of the Apollo Program. For these reasons, the fundamental tradeoff decision was the qualitative one to use the Apollo Unified S-Band data management equipment, modified only as necessary to meet OPE data requirements, as the core of the OPE data management subsystem.

Qualitative tradeoff analyses were also made in the areas of critical biodata and environmental sensors. These are summarized below.

Activity counter: The following candidate concepts were considered:

- Transmitter Signal Strength Variation
- Implanted Magnet
- Implanted Coil
- Proximity Detector
- Ultrasonic Motion Detector

The transmitter signal strength variation method makes use of variations in signal strength of the already implanted temperature transmitter. It requires an external antenna detector, digitizer, and binary counter. Its primary advantage is that it requires no additional implant. However, in this method the loss of the temperature channel results in the loss of activity measurement.

In the implanted magnet method, high impedance coils detect animal motion through induced coil-voltage outputs. It requires a Teflon-coated bar magnet plus external detection coils, digitizer, and binary counter. Its advantage lies in its sensitivity and low power consumption. It has the disadvantage of offering little experience in actual application for this purpose.

In the implanted coil method, a tuned coil causes signal strength variations in the transmitter/receiving antenna due to animal movement. It requires an implanted tuned coil, external carrier transmitter, receiver, antenna, detector, digitizer, and binary counter. Its disadvantages include relatively high power consumption, complexity of instrumentation, and little experience in actual use for this purpose.

In the proximity detector method, the animal's body works as a variable capacitive load for an oscillator. It requires an oscillator/receiver, detector, digitizer, and binary counter. No additional implant is required. Among its disadvantages are low triggering sensitivity for minor movements and relatively high power requirements. In addition, animal wastes on the antenna will change the basic oscillator frequency.

The ultrasonic motion detector method uses a principle in which animal movements disturb the distributed sound pattern generated by an ultrasonic generator inside the cage. It requires a generator, detector, digitizer, and binary counter. It has a very high sensitivity and requires no additional implants. Its disadvantages include high power requirements and the fact that positional changes of the TV camera will register as false animal movement. In addition, the animal may suffer adverse effects from continuous exposure to ultrasonic energy.

The recommended approach is to use the transmitter signal strength variation method as the primary means of measuring activity and to include the implanted magnet technique as a backup method.

Carbon dioxide detector: The following candidate methods for CO_2 detection were investigated:

- Electrochemical
- Ionization
- Infrared
- Mass Spectrometer

The electrochemical method uses a glass electrode sensitive to pH, a reference electrode, and electrolyte gel in which pH changes provide a variation in cell potential. The potential change is proportional to the log of CO_2 concentration. It is small, simple, rugged and has low power consumption. On the other hand, electrolyte evaporation limits sensor life, the method has poor stability, is very sensitive to circuit impedance, requires large amplification, and has low response.

In the ionization sensor, CO_2 is removed from a reference stream and both reference and measurement streams are passed through ionization chambers. The ionization current difference between the reference and measurement chambers is proportional to the CO_2 concentration in the measurement stream. It offers the advantages of being simple and qualified for spaceflight, and of having very low power consumption and low cost. Its disadvantages are that a sorbent is required to remove CO_2 and it is less accurate than other techniques considered.

The infrared sensor detects infrared attentuation of a sampled atmosphere at two narrow wavelength intervals – one at the absorption band unique to CO_2 ($\sim$ 4.27 μ), the other at a band where no normal atmospheric gases absorb ($\sim$ 4 μ). The ratio of attenuation values provides an output proportional to CO_2 concentration. Its advantages include the facts that it is a classic laboratory and industrial technique, uses simple optical and electronic systems, has low power consumption, and equipment is already under development for spaceflight applications. No significant disadvantages are known.

In the mass spectrometer, sample gas is admitted to an ionization chamber through a viscous flow pressure divider and molecular leak. Ions are accelerated through a focusing system to an exit aperture. The magnetic field bends the ion stream to a circular path depending upon the mass-to-charge ratio. The ion collectors are positioned to receive current at specific positions to continuously indicate CO_2 concentration. Its advantages include rapid response, high accuracy, an output signal directly proportional to CO_2 concentration, and the fact that equipment is already under development for spaceflight application. It can also be used to measure other gases of interest. Its disadvantages are high weight and power, critical filament life, and is currently the least developed of the candidates considered.

On the basis of the above factors, the infrared sensor was selected as the preferred concept for CO_2 detection.

Oxygen detector: Three candidate methods were studied for O_2 detection. These were:

- Electrochemical
- Ultraviolet
- Mass Spectrometer

The electrochemical method uses a small enclosure containing a gold cathode, a silver anode, and a cellulose-base KCl gel covered by a Teflon membrane. Oxygen passing through the membrane diffuses to the cathode. A potential of 0.8v applied across the two electrodes results in a current flow directly proportional to the oxygen partial pressure. It is small, simple, rugged, has rapid response, low power consumption, and is qualified for space use. Its disadvantages are that electrolyte evaporation limits sensor life, the method has relatively poor stability, and requires large amplification.

The ultraviolet sensor detects ultraviolet attenuation of the sampled atmosphere at 1,470 A through the use of a xenon source, sample path, and detector. The source limits the spectral bandwidth; the fixed path length of the sample cell allows the determination of O_2 concentration from observed transmittance; and the detector converts the transmitted energy into an electrical signal proportional to O_2 concentration. It offers the advantage of using simple optical and electronic systems under development for spaceflight application. Its disadvantage is that it requires the highest power of all techniques compared. One version of this sensor provides a water vapor measurement in conjunction with the oxygen measurement.

The mass spectrometer operates in a manner similar to that described for the CO_2 mass spectrometer.

Due to the many advantages of the ultraviolet sensor, this method was selected as the preferred approach for O_2 sensing; it also provides a water vapor measurement.

Radiation detectors: Four types of radiation dosimeters were considered:

- Plastic scintillator – chemical composition $(CH_{1.1})_n$
- Tissue equivalent ionization chamber
- Silicon solid state detector
- Passive

The plastic scintillator uses a detector which is chemically similar to tissue and thus shows proportionality to absorbed energy. While not perfectly tissue-equivalent, the energy absorption characteristics are much closer to tissue than in any other practical system. The light output from the scintillator is proportional to the total energy deposited for either heavy ionizing particles or electrons over a wide energy range. High sensitivity is obtainable by use of a photomultiplier tube-high density detection medium. The ionization chamber uses the Bragg-Gray principle: the ionization level measured by the detector is proportional to the total energy deposited in the large volume of material surrounding the detector element. The most severe limitation with this detector is its relatively low sensitivity.

NASA has developed a Tissue-Equivalent Ionization Chamber (TEIC) personal radiation dosimeter which was flown on the Gemini program and is scheduled for Apollo. The Nuclear Particle Detection System, also scheduled for Apollo, utilizes a solid state detector, but presently is not sensitive to electrons and only provides four channels of spectral data. The scintillator or solid-state detector is suitable for pulse height analysis in digital format to obtain the energy spectrum of radiation.

Passive dosimeters are either of the film type or are made of silver-activated phosphate glass. Films are available in at least three response ranges from 10^{-3} to 10^{3} rad, while a typical glass rod dosimeter covers a range from 10 to 10^{6} rad. Both types have good stability under widely varying environmental conditions and must be recovered for data analysis.

The selected approach is to use, as a primary system, a Plastic Scintillation Dosimeter plus two Tissue-Equivalent Ionization Chambers for telemetry (in near real time) of the various types and energy levels of radiation encountered during flight. These are located in the cage. The preferred approach also includes, as a back-up system, a combination of films plus silver-activated phosphate glass to obtain a post-flight picture of the total integrated dose received during the entire flight. These film packs will be installed in the retrieval capsules.

Spacecraft subsystems. – The spacecraft subsystems are described in the following paragraphs.

Interrelationship of subsystems: The spacecraft subsystems, i.e., thermal control, data management, electric power, and attitude control are interrelated to such a degree that they must be rated as a combined package rather than isolated subsystems. The optimum spacecraft does not necessarily contain all optimum subsystems since some subsystem requirements are diametrically opposed to others. Perhaps the best way to illustrate this fact – is to list a set of optimum spacecraft subsystems.

- Thermal Control – The optimum thermal control subsystem would be a spinning cylindrical spacecraft with low orbital inclination and its spin axis normal to the ecliptic. Spinning would reduce the circumferential skin temperature gradients, tend to provide a more uniform temperature, and reduce orbital temperature swings. The mean orbital temperature bandwidth as a result of seasonal variations would be minimized.

- Data Management – The optimum data management subsystem would require precise antenna pointing either by controlling the entire spacecraft or having an articulated antenna which could be pointed accurately regardless of spacecraft orientation. As a further refinement, the antenna should be able to acquire and track ground stations either automatically or upon command.

- Electrical Power – The optimum electrical power subsystem is a sun-oriented extended solar array. Radioisotope thermoelectric generators are non-optimum from a cost and handling standpoint.

- Attitude Control – Gravity gradient earth stabilization with control moment gyro damping or cold gas earth orientation represent two equally optimum attitude control subsystems.

Obviously, no one spacecraft can incorporate all these requirements since it would have earth orientation for attitude control and data management, sun orientation for the solar arrays, and be spinning for temperature control. The selected spacecraft concept must, therefore, be the best compromise among such alternatives. It is also noted that the orbital inclination and altitude are subject to variation as AAP plans change. Therefore, an important goal is to select systems which are as insensitive as possible to these potential variations.

Before resorting to postulation of an all-encompassing spacecraft concept matrix which embodies all the subsystem possibilities and then evaluating each concept, it is instructive to further examine the electrical power and attitude control subsystems separately.

Electrical power: Two generic types of power sources have been considered for the OPE: radioisotope thermoelectric generators (RTG's) and solar cells (see Fig. 61). Three RTG's have been evaluated in light of the OPE power requirements. The two alpha emitters evaluated were SNAP-27 (a unit that is currently being qualified for NASA) and SNAP-29 (a higher powered RTG that has recently entered the development phase.) A beta-emitting fuel was also evaluated. This system was based upon the 250 w RTG study recently completed by Lockheed for the Atomic Energy Commission.

The thermoelectric generator provides continuous heat to what is essentially the hot junction of a thermocouple. Heat is rejected from the cold junction by space radiation. Thus, the generated current supplies the load and charges a battery.

The purpose of the battery is to accommodate peak demands and reduce the fuel requirements to more nearly match the average power consumption. The obvious advantage of an RTG is its reliability and insensitivity to spacecraft configuration, orientation, orbit inclination, and shadowing. Cost and radiation hazard represent the more serious drawbacks to these systems.

Six solar cell array configurations were also evaluated. Various solar array configurations were postulated for three spacecraft orientation concepts.

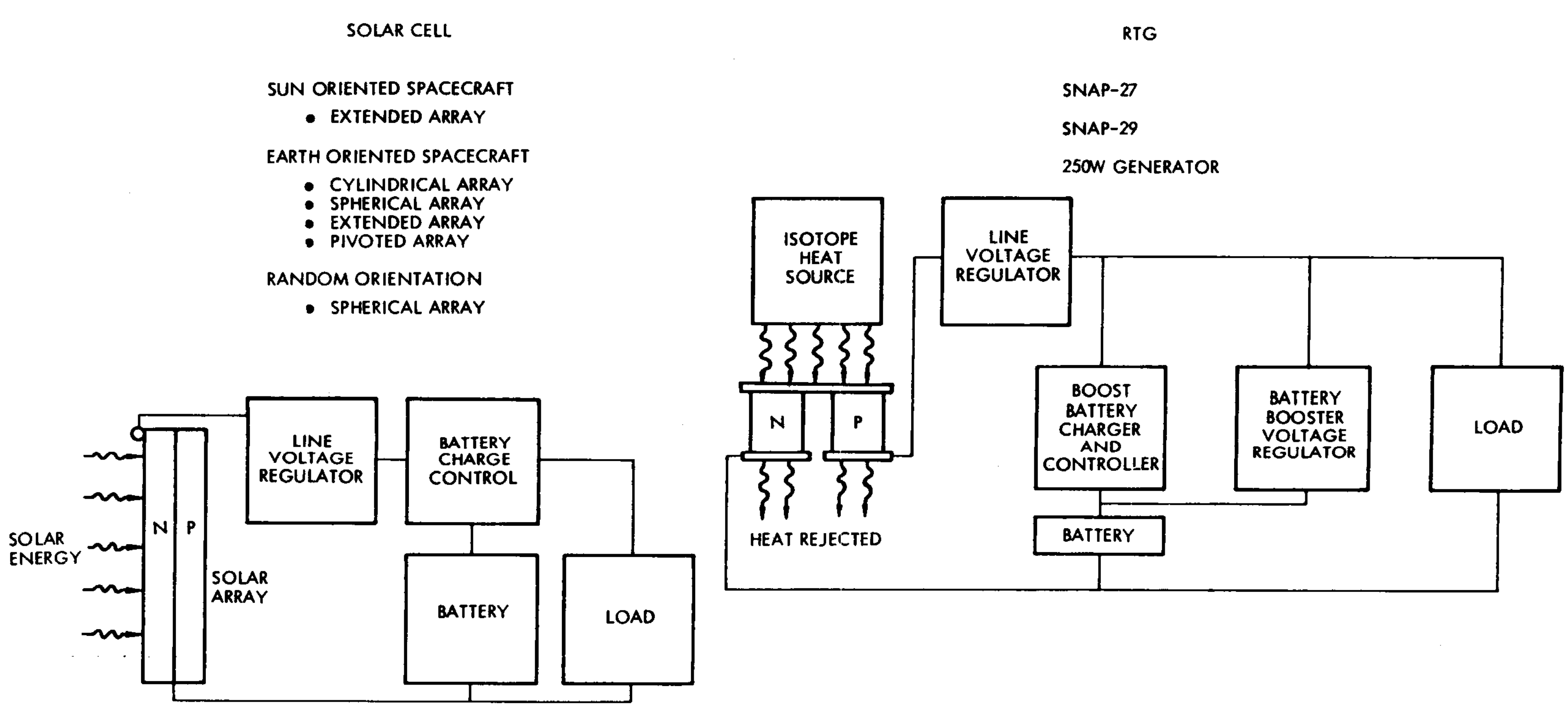


Fig. 61 Electrical Power Subsystem Candidates

Solar cells derive their energy from the photo-voltaic effect of incident sunlight on silicon semiconductors. They are sensitive to incident sunlight angle, shadowing, temperature, and configuration. The array must be sized at approximately twice the average spacecraft power consumption level so that batteries may be charged to supply power when the spacecraft is in the earth's shadow. Lower cost and more operational experience are the two major advantages of solar cell arrays over RTG's.

The evaluation of RTG power subsystems is shown in Table 27. The performance of the SNAP-27 generator has been based upon recent data published by the General Electric Company, the prime contractor for SNAP-27. Estimates of weight and volume for SNAP-29 were based upon preliminary data furnished by Martin Nuclear Division. The weights and volumes quoted are estimates that include the RTG power source, batteries, and power conditioning and control equipment. The weights and volumes for the beta-emitting fuel were based on the 250 w RTG study performed by Lockheed and also include the weights of the RTG, batteries, power conditioning, and required shielding.

The high reliability of the RTG is due to the fact that it depends less upon batteries and has more attractive failure mode characteristics. For purposes of comparison, the costs of the RTG's were based on the thermal fuel requirements at a cost of \$500/thermal watt for the alpha emitter (Pu^{238}) and \$50/thermal watt for the beta emitter (Sr^{90}). SNAP-27 is currently undergoing qualification tests; SNAP-29 is in the preliminary stages of development and could possibly be available by 1970. It is doubtful that the 250 w Sr^{90} fueled generator could be available by 1970.

Due to the order of magnitude cost differences between RTG power sources and solar cell power sources which cannot be effectively contrasted by the rating scheme employed, no further consideration was given to RTG's for this program.

The evaluation of solar cell power subsystems is shown in Table 28. Calculation of weight and volume for the six solar array configurations have been based upon technology and hardware currently available at Lockheed. The numbers shown include the weights and volumes of the solar array, secondary batteries, and power regulation and conditioning equipment.

In regard to reliability, the systems were ranked according to the following rationale:

1. Systems with orientation and deployment mechanisms are less reliable than those without orientation or deployment systems.
2. Solar array configurations with nearly constant output power are more reliable than those with large seasonal variations.
3. Systems with both large seasonal or orbital variations in output power and deployment and tracking mechanisms are least reliable.

Solar cell system costs were determined at the rate of \$800/ft^2 of solar array. These figures reflect only the costs of the solar cells and are only a comparative index of solar power system costs. All solar cell systems considered are operational concepts and are given the highest rating for development status.

TABLE 27

EVALUATION OF RTG POWER SUBSYSTEMS

Candidate	Weight		Volume		Reliability		Cost		Development Status		Total
	Lb	Weighted Value	Ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
SNAP-27	398	12	58	4	4	24	1	4	4	16	60
SNAP-29	546	9	65	3	4	24	1	4	2	8	48
250W RTG (Sr^{90} Fuel)	1,628	-12	65	3	4	24	2	8	1	4	27

TABLE 28

EVALUATION OF SOLAR CELL POWER SUBSYSTEMS

METHOD	WEIGHT		VOLUME		RELIABILITY		COST		CONCEPT DEVELOPMENT STATUS		TOTAL
	lb	Weighted Value	ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
SUN ORIENTED S/C											
Extended Array	229	16	4.5	9	2	12	5	20	5	20	77
EARTH ORIENTED S/C											
Cylinder Array	945	1	30.5	7	2	12	4	16	5	20	56
Spherical Array	775	5	24.2	7	3	18	4	16	5	20	66
Extended Array	531	9	13.2	9	1	6	4	16	5	20	60
Pivoted Array	291	14	6.5	9	1	6	4	16	5	20	65
RANDOM ORIENTATION											
Spherical	775	5	24.5	8	3	18	4	16	5	20	67

On the basis of this evaluation, a sun-oriented spacecraft with extended array is the selected subsystem. However, the question of spacecraft orientation remains to be resolved, therefore, this subsystem cannot be chosen at this point.

Attitude control subsystem: Eight attitude control subsystems were postulated and are shown in four functional groups in Fig. 62 . The concept of spin stabilization of the spacecraft was not considered due to its inherent induced gravity effect on the experiment.

Concepts 1 and 2 are gravity gradient systems. The gravity gradient technique offers a simple and reliable means of attitude control. The natural linearized control torques for small angle displacements due to mass attraction and gyroscopic effects of an earth-oriented body are:

$$M_x = -4\,\omega_o^2\,(I_y - I_z)\,\phi \quad , \quad M_y = -3\,\omega_o^2\,(I_x - I_z)\,\theta$$

$$M_z = -\,\omega_o^2\,(I_y - I_x)\,\psi$$

where x , y , z and ϕ , θ , ψ are associated with the roll, pitch, and yaw axes, respectively. The inertias are sized with booms and mass tips to provide sufficient control stiffness for overcoming all disturbances. Position control in pitch and roll is obtained with a dumbbell shape with yaw rate control provided through roll-yaw coupling. Yaw position control can be obtained with auxiliary booms as illustrated in Fig. 62.

Damping must be provided to remove oscillatory energy. One technique considered was the coupling of an auxiliary body to the vehicle via a dissipative joint. A magnetic ball damper enclosed in a viscous fluid or an auxiliary boom with a dissipative (friction) coupling are examples. Inherent in these examples is a relatively low damping ratio, i.e., 0.03 typical. The damping ratio is defined as the ratio of actual damping coefficient to the critical damping coefficient of the vibratory system. It is a measure of the amount of energy which is absorbed. A small value for the damping ratio indicates the system will continue oscillation for a long time after excitation. A large value (≥ 1) indicates the system will come to rest without oscillation.

Another technique considered was the use of control moment gyros (CMG). These are used in a passive mode and provide damping ratios as high as 0.3. They enable the use of smaller inertias since many of the disturbances encountered are resonance types which cannot build up with a high damping ratio. Also, they can provide additional yaw stiffness and position control by appropriate alignment of their angular momentum vectors.

Concepts 3, 4, and 8 are cold gas systems. Cold gas offers another reliable and simple technique for attitude control. In this application, a position sensor (sun or

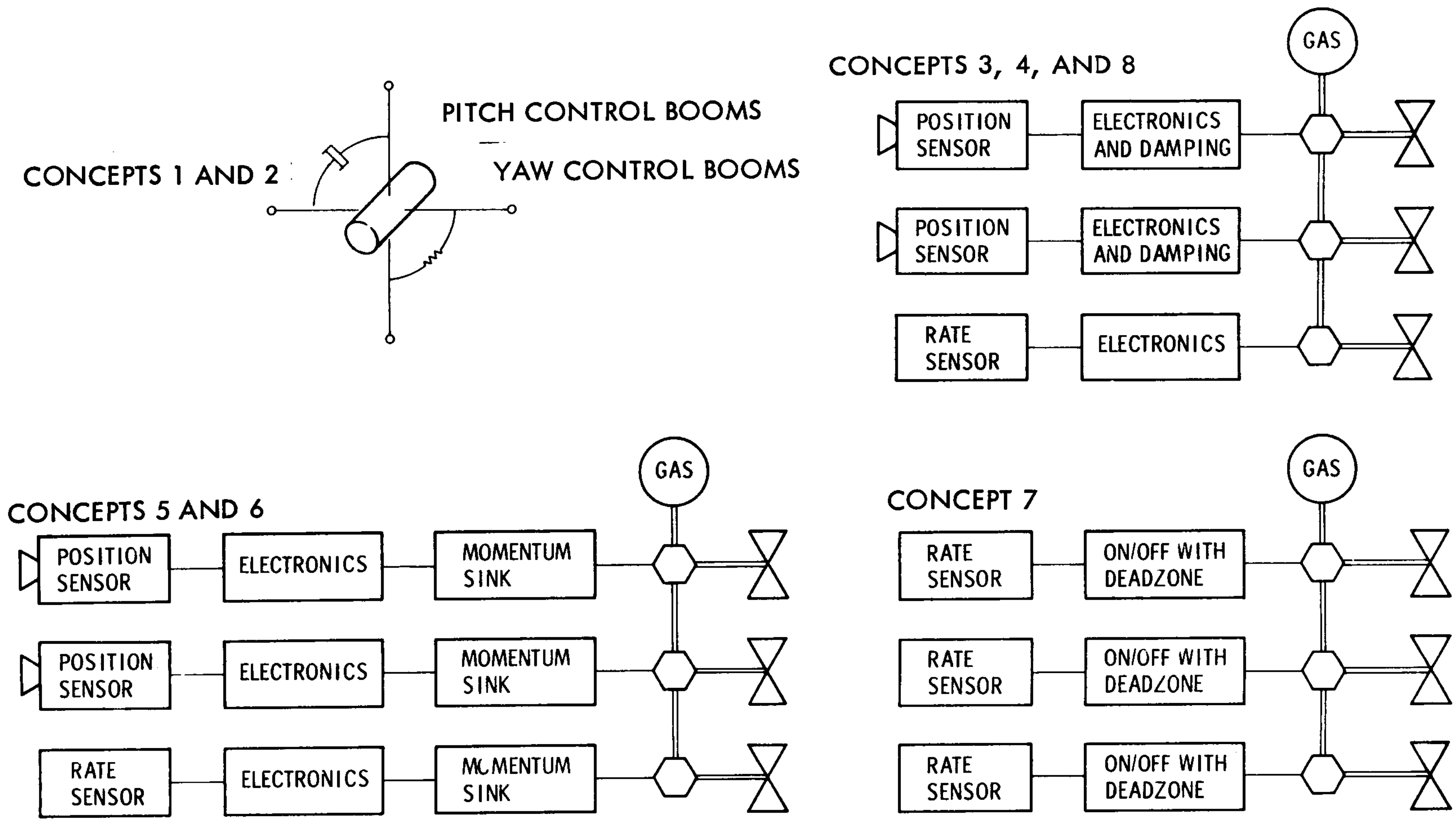


Fig. 62 Attitude Control Subsystem Candidates

horizon) detects attitude errors about two axes. This is used to drive an on/off amplifier with deadzone which in turn opens and closes the appropriate gas valves for that axis. Derived rate is used for damping. By sizing the deadzone as large as possible and the thrust level as low as possible, a minimum of control gas is required. In the case of a sun sensor, a gyroscopic or other type of memory is also required. The third axis is rate controlled with a rate sensor. Position control is possible but appears unnecessary for the application. Two earth-oriented cold gas systems were considered. One allowed a waste water dump of two lb/day and the other did not allow any waste water dump.

Concepts 5 and 6 employ a momentum sink. The momentum sink technique essentially removes disturbances from the vehicle by internal momentum storage. Its primary advantage is the ability to handle cyclic disturbances without mass expulsion. The noncyclic disturbances, however, eventually cause saturation. To prevent this, an auxiliary cold gas system is required for periodic momentum dumping. The equipment required is identical to that for the cold gas system except for the additional momentum sinks for all three axes. The weight of the momentum sink equipment tends to be offset by the reduction in gas requirements. Two types of momentum sinks were considered: reaction wheels and control moment gyros.

Concept 7 provides rate control only. This system was considered because there is no absolute position requirement for this spacecraft. Rate sensors are used on all three axes to drive a cold gas system which utilizes a deadzone on rate to conserve gas. A great majority of the disturbances vanish for this system since there is no attempt to hold position.

The evaluation of attitude control subsystems is shown in Table 29. Of the eight systems considered, cold gas rate control was the lightest because its gas requirements were low. The gravity gradient system followed closely. Cold gas position control with a water dump was the heaviest, the penalty being about 260 lb due to dumping waste water even though a thrust nullifier is used. All systems rated high in regard to volume since none exceeded 5 ft^3.

The cold gas earth-oriented system ranked best in cost because it has inexpensive components which can be purchased virtually off-the-shelf. The sun-oriented version requires a more expensive memory and thus costs more. The gravity gradient techniques were more expensive because of the need for damper development in the first system and cost of off-the-shelf control moment gryos for the second system. The momentum sink systems costs were high because of the development required and the inherent expensive nature of their components. The cold gas rate control system cost more than the earth-oriented cold gas system because of differences in development required.

Earth-oriented cold gas and gravity gradient with CMG damping ranked best in terms of development. These cold gas systems have been proven extensively in many applications. Lockheed has flown its Agena vehicle several times using gravity gradient control and CMG damping. Gravity gradient with passive damping ranked somewhat lower because of the development uncertainty of an adequate damper.

TABLE 29

EVALUATION OF ATTITUDE CONTROL SUBSYSTEMS

METHOD	Weight lb	Power		Total Equivalent Weight		Volume		Reliability		Cost		Development Status		Total
		Watt	Weight lb	lb	Weighted Value	ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
1. Gravity Gradient With Passive Damping	86	0	0	86	18	1.0	10	4	24	3	12	3	12	76
2. Gravity Gradient With Control Moment Gyro Damping	67	18	36	103	18	1.0	10	4	24	3	12	5	20	84
3. Cold Gas With Water Dump-Earth Oriented	460	10	20	480	10	4.8	10	3	18	5	20	5	20	78
4. Cold Gas Without Water Dump-Earth Oriented	200	10	20	220	16	2.0	10	3	18	5	2	5	20	84
5. Reaction Wheels With Auxiliary Cold Gas	193	35	70	263	15	1.5	10	2	12	2	8	3	12	57
6. Control Moment Gyros With Auxiliary Cold Gas	156	85	170	326	13	2.5	10	2	12	1	4	3	12	51
7. Cold Gas Rate Control	24	15	30	54	19	0.5	10	3	18	4	16	3	12	75
8. Cold Gas Without Water Dump Sun Oriented	200	10	20	220	16	2.0	10	3	18	4	16	3	12	72

The gravity gradient systems were best in terms of reliability because of their passive nature. For CMG damping, the CMG lifetime is over one year and even a failure in one CMG merely means the system operates in a somewhat degraded mode. The cold gas systems ranked somewhat lower in reliability because of their electronic components and possibility of valve failure. All other systems were even lower because they all contain a complete auxiliary cold gas system as well as other components.

On the basis of this evaluation, two systems (Concepts 2 and 4) are rated the highest. Neither of these concepts is applicable to sun orientation. Once again the actual selection must be deferred until the interraction of this subsystem with other selected subsystems can be evaluated.

Spacecraft configurations: Six spacecraft configurations (Fig. 63) were postulated as a result of combining the elements of electric power generation (by solar cells), attitude control, and thermodynamic control. Each spacecraft configuration was then analyzed to determine the quantitative effects of system variations, i.e., body mounted solar cells in one case and extended arrays in another, or an omnidirectional data link in one case and a unidirectional data link in another.

Spacecraft configuration 1 is a cylinder which is earth-oriented and has body mounted solar cells on all but the 45-deg sector facing earth and the flat ends. The cylindrical axis is normal to both the velocity vector and the local earth-vertical. Attitude control is maintained by either gravity gradient booms with control moment gyro damping, or by cold gas thrusters without water dump. This configuration has the advantages of a unidirectional data link and a possibly passive attitude control system.

Spacecraft configuration 2 is essentially a sphere which has been faceted for body-mounted solar panel installation on all but the earth-oriented facet. Attitude control is maintained by either gravity gradient booms with control moment gyro damping or cold gas thrusters without water dump. This configuration has the advantages of spacecraft configuration 1 plus a smaller seasonal variation in solar array output.

Spacecraft configuration 3 is a cylinder with four solar array paddles extended outward in mutually perpendicular pairs. Solar cells are applied to both sides of each paddle. The cylindrical axis is earth-stabilized by either gravity gradient booms with control moment gyro damping, or by cold gas thrusters without water dump. This configuration has the advantages of spacecraft configuration 1 plus a more efficient electrical power subsystem due to cooler extended solar arrays.

Spacecraft configuration 4 is a cylinder with two articulated solar array paddles. The cylindrical axis is earth-oriented. The paddles are kept oriented to the sun by means of a tracking/actuating subsystem. This configuration has all the advantages of spacecraft configuration 1 plus a more efficient electrical power subsystem due to cooler extended solar arrays and a limited ability to control solar incidence.

Spacecraft configuration 5 is a cylinder with two extended solar cell paddles which are oriented to the sun by means of cold gas thrusters without water dump. A

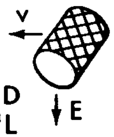
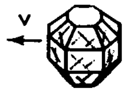
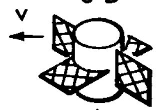
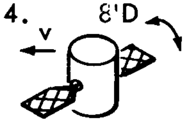
CONFIGURATION / SYSTEM	EARTH ORIENTED				SUN ORIENTED	RANDOM ORIENTATION
	1.  14.1'D 14.1'L	2. 13.8'D 	3. 8'D  8' x 8'	4. 8'D  8.5' × 8.5'	5. 8'D  7' x 7'	6.  13.8'D
THERMAL CONTROL	ACTIVE ECS SPACE RADIATOR, PASSIVE SKIN AND EQUIPMENT CONTROL, ISOLATION OF LIFE CELL FROM SKIN BY INSULATING BLANKET					
DATA MANAGEMENT	UNIDIRECTIONAL DATA LINK				OMNIDIRECTIONAL DATA LINK	
ELECTRICAL POWER	BODY MOUNTED SOLAR CELLS		PADDLE MOUNTED SOLAR CELLS			BODY MOUNTED SOLAR CELLS
	SECONDARY BATTERY					
ATTITUDE CONTROL	GRAVITY GRADIENT WITH CMG DAMPING				COLD GAS WITHOUT WATER DUMP	COLD GAS RATE CONTROL
	COLD GAS WITHOUT WATER DUMP					
REMARKS	SENSITIVE TO ORBIT INCLINATION		SOLAR PANEL DEPLOYMENT REQUIRED			THERMO-DYNAMIC ANALYSIS DIFFICULT
	TERMINAL ALTITUDE VARIATION		SENSITIVE TO ORBIT INCLINATION	GOOD POWER SYSTEM GROWTH		
			TERMINAL ALTITUDE VARIATION			

Fig. 63 Candidate Spacecraft Configurations

gyro or other memory provides inertial reference on the night side of the orbit. Additional antenna and transmitter power is required over that of an earth-oriented spacecraft. This spacecraft configuration has the advantage of an efficient electrical power subsystem which has absolute control of solar incidence.

Spacecraft configuration 6 is a faceted sphere completely covered with body mounted solar cells. Spin rates only are controlled by means of cold gas thrusters. Additional antenna and transmitter power is required as with configuration 5. The advantage of this configuration is that it can tumble at will. Therefore, the attitude control fuel requirements will be minimal.

The evaluation of spacecraft configurations is shown in Table 30. The scheme for rating spacecraft configurations is a minor variation from that used in the other rating systems. First, weight, power, and volume differences are used instead of absolute values. Second, actual power-to-weight conversions for each spacecraft are used to compute equivalent weight. Cost, reliability, and development status reflect a summation of component systems which compose the spacecraft.

Spacecraft 4 weighed the least, both in actual weight and in equivalent weight. Spacecraft 5 has the least volume, whereas spacecraft 1 has the most volume. The reliability ratings assigned to the spacecraft configurations were derived primarily from the reliability ratings of the electrical and attitude control systems. The reliability rating of the electrical system was given twice the weight of the attitude control system in the configurations where attitude control is not crucial to power generation.

Cost is determined by summing the various system costs. Spacecraft 5 has low cost mainly because of its low solar cell cost and low attitude control system cost. Spacecraft development status ratings are equivalent to the development status rating of the least developed spacecraft system.

The selected concept is spacecraft configuration 5. It has low weight, cost, and volume. Reliability and development ratings for this concept do not reflect any significant problems in reaching the required mission reliability or in meeting the 1970 flight date.

On the basis of this evaluation, the attitude control and electrical power subsystems selected are (1) cold gas without water dump (sun-oriented) for attitude control and (2) extended solar array (sun-oriented) for electrical power.

TABLE 30

EVALUATION OF SPACECRAFT CONFIGURATIONS

METHOD	Δ Weight lb	Δ Power		Δ Total Equivalent Weight		Δ Volume		Reliability		Cost		Concept Development Status		Score
		Watt	Weight lb	lb	Weighted Value	ft^3	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	Rating	Weighted Value	
1. Earth Oriented Cylinder	654	18	42	696	6	1,531	-143	3	18	1	4	5	20	-95
2. Earth Oriented Sphere	484	18	35	519	10	693	-59	4	24	2	8	5	20	3
3. Earth Oriented Extended Array	240	18	24	264	15	18	8	2	12	2	8	5	20	63
4. Earth Oriented With Pivoting Array	0	18	13	49	19	4	10	1	6	2	8	5	20	63
5. Sun Oriented Extended Array	74	25	14	88	18	0	10	2	12	5	20	3	12	72
6. Tumbling Sphere	444	30	58	502	10	693	-59	3	18	2	8	3	12	-10

PAYLOAD SYSTEM DESCRIPTION

This section presents the preliminary design description of all payload subsystems. In order to conform to NASA Publications Manual, SP7013, all drawings have been reduced to report-size pages. For all payload equipment, full-size drawings have been submitted separately as part of the Data Package required by the contract. Where design detail is lacking due to the reduced size of the drawings in this report, the Data Package should be consulted. The major portion of the payload subsystems are shown in the inboard profile of Fig. 64. The following sections describe the thermal and atmosphere control, metabolic support, life cell, special equipment and data management subsystems.

Thermal and Atmosphere Control

The integrated thermal and atmosphere control system is capable of providing suitable environmental conditions for the primates and for on-board equipment. It provides this support under the varying conditions imposed by the mission environment and duration. Environmental conditions maintained are summarized in the Requirements section of this report.

The subsystem is shown schematically in Fig. 65, with components identified in Table 31. It is designed to perform the functions of oxygen and nitrogen makeup, carbon dioxide and trace contaminant removal, humidity, temperature and pressure control, and atmosphere circulation. The subsystem consists of: (1) gas supply system, including storage and partial pressure control; (2) atmosphere recirculation and revitalization system including fans, condensing heat exchanger, carbon dioxide and trace contaminant removal systems; and, (3) thermal control system, including pumps, accumulator, cold plates, and radiator system.

A review was made of Apollo and Gemini existing hardware with a view to using as much of the existing hardware as possible. It was determined that for dynamic components such as compressors, pumps and fans, the existing hardware is generally over-capacity for the requirements of this system. Lesser components such as valves, transducers, regulators, disconnects, could be used either directly or with modification.

<u>Gas supply system.</u> – The life cell atmosphere is maintained at 14.7 psia total pressure with an oxygen partial pressure of 18 to 30 percent of 14.7 psia and the balance made up of nitrogen, carbon dioxide, water vapor and trace gases. The system has the capability for three modes of regulation and delivery: (1) an oxygen partial pressure sensor control, (2) an adjustable flow fixed-bleed control, and (3) a 5.0 psi pure oxygen mode for emergency operation. Each gas is regulated from its storage pressure (oxygen 4500 psia, nitrogen 3900 psia) to a lower working pressure of nominally 100 psia. The subsequent flow control devices and regulators further reduce this pressure to that

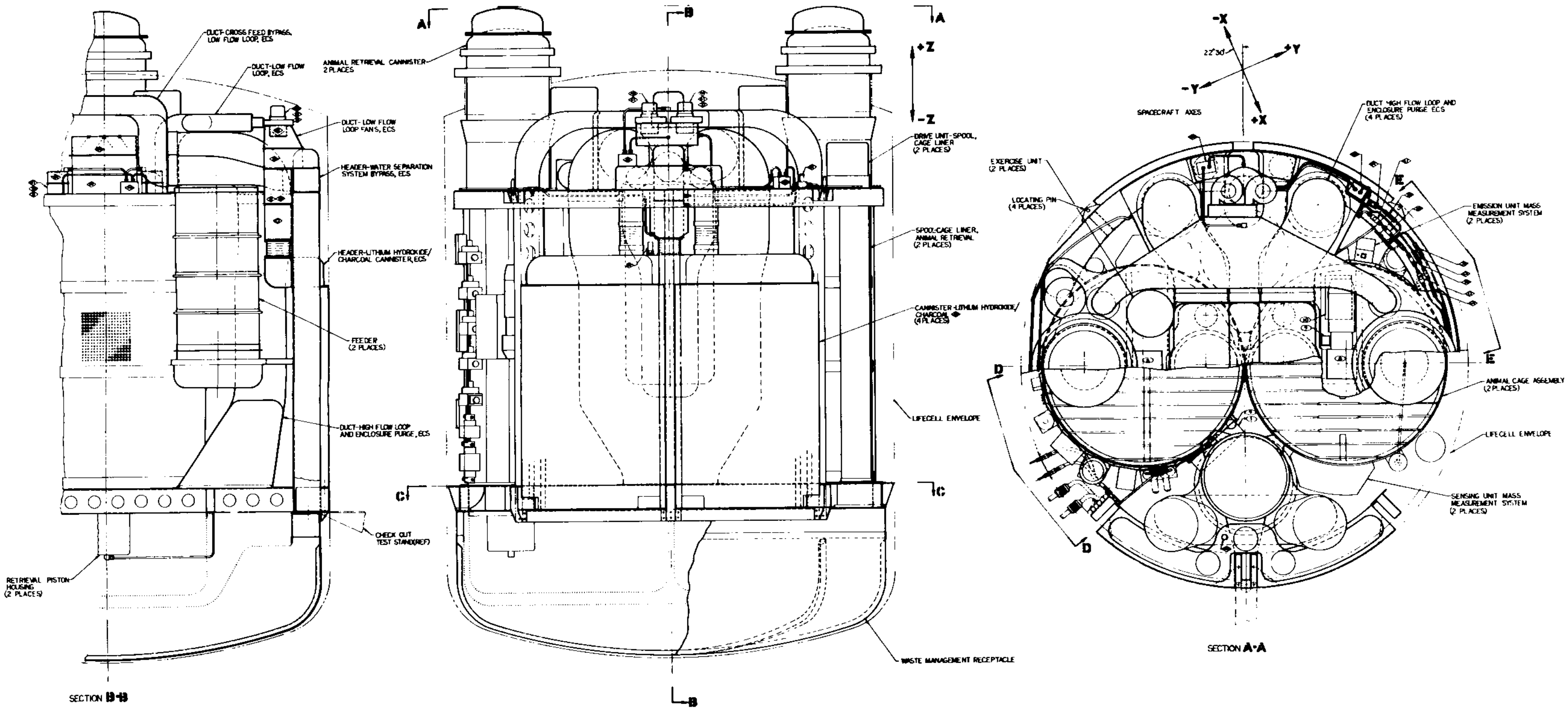


Fig. 64a Inboard Profile

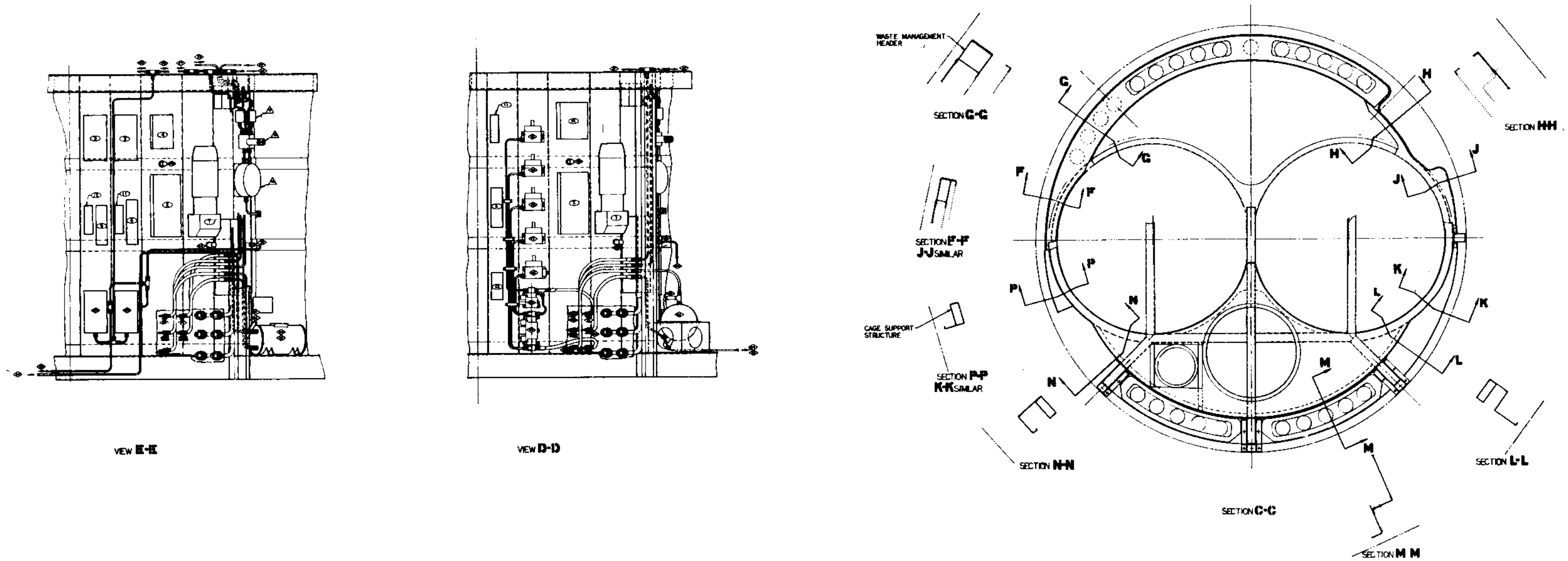


Fig. 64b Inboard Profile (Cont.)

LIFECELL EQUIPMENT INDEX

System	Ident No.	Component	No. of Units
Environmental Control	16	Regulator – 5 psi, O_2	1
System	17	Regulator – Total Pressure, O_2	1
Components Denoted by	18	Assembly – Flow Control, O_2	1
XX	19	Regulator – Total Pressure, N_2	1
	20	Assembly – Flow Control, Adjustable, N_2	1
	21	Flow Meter – Coolant	2
	22	Valve – Solenoid, O_2	1
	23	Valve – Control, O_2, N_2	1
	24	Module – Coolant Pump Package	1
	25	Sensor – Pressure, Pump Outlet	1
	26	Sensor – Temperature, Coolant Fluid	1
	27	Condensate Heat Exchanger	2
	28	Assembly – Orifice, Flow Balancing	2
	34	Accumulator – Coolant	1
	35	Filter – Coolant	1
	36	Canister – LiOH, Charcoal and Catalyst	4
	37	Water Separator	2
	38	Check Valve – Water	2
	39	Assembly – Bypass Temperature Control	2
	40	Fan – Low Flow Loop	4
	41	Check Valve – Fan, Low Flow Loop	4
	42	Check Valve – Fan, High Flow Loop	4
	43	Sensor – Temperature, Cabin Inlet Atmosphere	2
	45	Sensor – Differential Pressure, Low Flow Fan	2
	46	Sensor – Differential Pressure, High Flow Fan	2
	47	Fan – High Flow Loop	4
	48	Sensor – Differential Pressure, Water Separator	2
	55	Sensor – Low Pressure, N_2	1
	56	Sensor – Temperature, Heat Exchanger Outlet	2
	57	Sensor – Temperature, Low Flow Fan	4
	58	Sensor – Temperature, High Flow Fan	4
	59	Fan – Enclosure Purge	2
	60	Sensor – Temperature, LiOH Bed Outlet	2
	61	Control Valve – Water Separator	2
	62	Sensor – Temperature, Animal Enclosure	2
	63	Check Valve – Integrated Sensor	2
	64	Integrated Sensor – O_2, CO_2, H_2O, N_2	2
	66	Check Valve – Purge Fan	2
	70	Sensor – RPM, Low Flow Fan	4
	71	Sensor – RPM, High Flow Fan	4
	72	Port – Cage Atmosphere Sensor	2
Electronic and Electrical	1	Behavioral Panel – Lever and Light Assembly	2
Components – Denoted by	2	Control Unit – Behavioral Panel	2
XX	3	Near – Field Receivers, Biodata	2
	4	Dosimeter – Ratiation	1
	5	Dosimeter – Tissue Equivalent	2
	6	Amplifier – Audio System	1
	7	Television Camera – Fixed Optics	2
	8	Television Camera – Scanning Optics	2
	9	Pickup Coil – High Impedence	4
	10	Antenna – Near Field Receivers	2
	11	Activity Counter – Implanted Magnet	2
	12	Register – 16-Bit Storage	1
	13	Temperature Bridges	1
	14	Power Distribution Panel	1
Drinking Water Supply	9	Aliquot Accumulator	4
System. Components	10	Solenoid Valve	4
Denoted by XX	11	Bacteria Filter	2

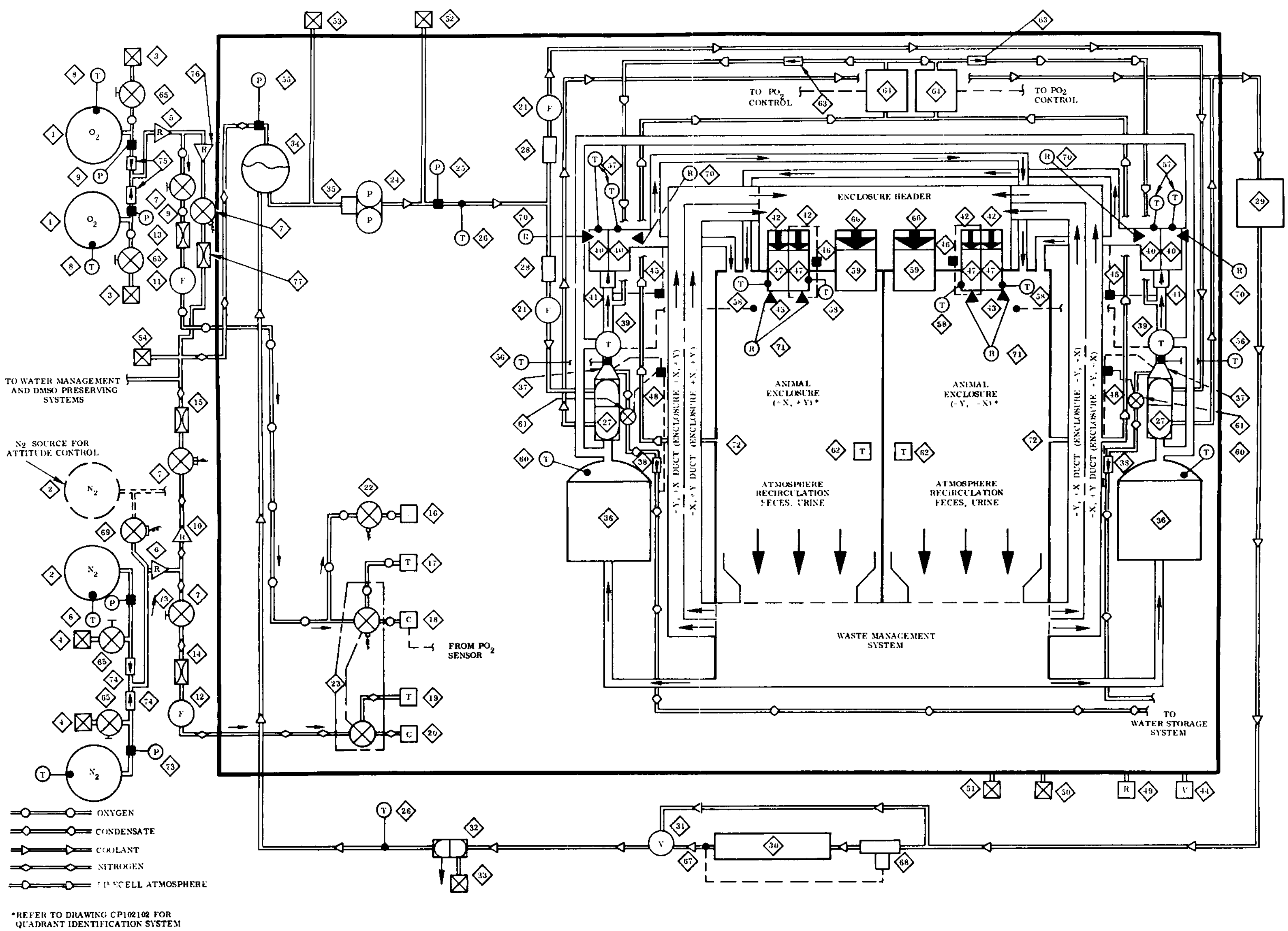


Fig. 65 Thermal and Atmosphere Control Subsystem Schematic

TABLE 31

THERMAL AND ATMOSPHERE CONTROL SUBSYSTEM COMPONENTS

Item No.	Description
1	Tank, Gaseous Oxygen
2	Tank, Gaseous Nitrogen
3	Coupling, Q_2 Disconnect, Filler
4	Coupling, N_2 Disconnect, Filler
5	Assembly, Regulator, and Relief Valve, O_2
6	Assembly, Regulator, and Relief Valve, N_2
7	Assembly, Valve, Manual Remote Command Shutoff, Low Pressure
8	Sensor, Temperature, Gas Supply
9	Sensor, Pressure, Gas Supply, O_2
10	Assembly, Regulator, Low Pressure, N_2
11	Assembly, Flowmeter, O_2
12	Assembly, Flowmeter N_2
13	Assembly, Flow Limiter, O_2
14	Assembly, Flow Limiter, N_2
15	Assembly, Flow Limiter, N_2
16	Assembly, Regulator, 5 psi, O_2
17	Assembly, Regulator, Total Pressure, O_2
18	Assembly, Flow Control, O_2
19	Assembly, Flow Control, Adjustable, N_2
20	Assembly Flow Control, Adjustable, N_2
21	Assembly, Flowmeter Coolant
22	Assembly, Valve, Solenoid, O_2
23	Assembly, Valve, Control, O2 N_2
24	Module, Coolant Pump Package
25	Sensor, Pressure, Pump Outlet
26	Sensor, Temperature, Coolant, Fluid
27	Assembly, Compensate Heat Exchanger
28	Assembly, Orifice, Flow Balancing
29	Goldplate, Battery
30	Assembly, Radiator
31	Assembly, Vernatherm Bypass
32	Assembly, Heat Exchanger Ground Cooling
33	Coupling, Freon Disconnect
34	Accumulator, Coolant
35	Assembly, Filter, Coolant
36	Assembly, Canister, LiOh, Charcoal and Catalyst
37	Assembly, Water Separator
38	Valve, Water, Check
39	Assembly, Bypass Temperature Control

TABLE 31 (Cont.)

Item No.	Description
40	Assembly, Fan, Low Flow
41	Valve, Check, Low Flow Fan
42	Valve, Check, High Flow Fan
43	Sensor, Temperature, Cadin Inlet Atmosphere
44	Assembly, Valve, Vent, Outflow, Manual/Remote Command
45	Sensor, Differential Pressure, Low Flow Fan
46	Sensor, Differential Pressure, High Flow Fan
47	Assembly, Fan, High Flow
48	Sensor, Differential Pressure, Water Separator
49	Assembly, Relief Valve, Cabin Pressure
50	Fitting, Cell Purge, Inlet
51	Fitting, Cell Purge, Outlet
52	Coupling, Coolant Disconnect, Servicing, Inlet
53	Coupling, Coolant Disconnect, Servicing, Outlet
54	Coupling N_2 Disconnect, Low Pressure
55	Sensor, Low Pressure, N_2
56	Sensor, Temperature, Outlet, Heat Exchanger
57	Sensor, Temperature, Fan, Low Flow
58	Sensor, Temperature, Fan, High Flow
59	Assembly, Fan, Enclosure Purge
60	Sensor, Temperature, Bed Outlet, Low Flow
61	Assembly, Valve, Solenoid, Water Separator
62	Sensor, Temperature, Life Cell
63	Valve, Check, Integrated Sensor
64	Sensor, Integrated, PO_2, $PCO_2 PN_2$
65	Assembly, Valve, Manual Shutoff, High Pressure
66	Valve, Check, Cell, Purge Fan
67	Sensor Temperature, Heater, Radiator Control
68	Heater, Radiator Control
69	Assembly, Valve, Isolation, N_2
70	Sensor, RPM, Low Flow Fan
71	Sensor, RPM, High Flow Fan
72	Port, Sensor, Gas Sampling
73	Sensor, Pressure Gas Supply, N_2
74	Valve, Check, High Pressure, N_2
75	Valve, Check, High Pressure, O_2
76	Assembly, Regulator, Low Pressure, Oxygen
77	Assembly, Flow Limiter, O_2

required for cell and other system control. The nitrogen system also supplies the pressure required for the coolant accumulator, water supply and preservative systems. This source is taken downstream of the 100 psi regulator and is further reduced to a nominal 35 psia. A similar regulated supply is available from the oxygen system. Shutoff valves are incorporated to allow servicing of the gas systems without actuation of cabin regulators. Flowmeters are incorporated in the supply lines to detect gas flow rates to the cabin. The systems also have flow limiters installed to restrict flows in the event of a failure which might tend to rapidly deplete the gas supplies. High pressure gas disconnects allow servicing of the tanks.

A spacecraft maximum allowable leakage rate of 0.3 lb/day is assumed for the purpose of sizing oxygen and nitrogen quantities. It is believed that this value is achievable and is consistent with leakage rates experienced on other spacecraft.

The nitrogen supply tanks are cross-connected to the supply for the attitude control system. An isolation valve (normally closed) is installed in the crossover line. Periodically throughout the mission the valve is opened to allow pressures between the systems to equalize, thereby eliminating excursion of the c.g. which could be caused by differences in tank gas quantities between the systems. The valve is controlled by remote command. In the event of a gross failure in either system, the valve will be closed, thereby isolating the failure.

A pressure relief valve is installed to prevent over-pressurization of the spacecraft. A vent valve is incorporated to provide depressurization of the lifecell either during prelaunch operations or prior to orbital removal of the retrieval canisters. The valve has manual control and remote command capability.

Oxygen partial pressure sensor control: This is the primary mode of operation and utilizes nitrogen to maintain a 14.7 psia total cell pressure. Oxygen partial pressure is sensed and oxygen supplied through a solenoid valve in response to the control signal caused by reduction in oxygen partial pressure to a value of 3.25 psia. This system responds to variations resulting from either leakage or changes in metabolic rate and maintains 3.5 ±0.85 psia oxygen partial pressure. The oxygen sensor used will be an ultraviolet spectrophotometer similar to the unit currently under development for NASA Langley Research Center by Perkin-Elmer.

Fixed bleed with adjustable flow control: In this mode of operation, oxygen is used to maintain the 14.7 psia total cell pressure. Nitrogen is supplied through the adjustable flow control valve. During the initial period of the flight the partial oxygen sensor detects cell oxygen partial pressure. The nitrogen supply rate can be adjusted, through ground command, until the desired 3.5 psia nominal oxygen partial pressure is achieved. This adjustment can be made periodically if the oxygen partial pressure sensor remains operable. The bleed system responds directly to variations in oxygen consumption caused by changes in metabolic rates, and will maintain the desired oxygen concentration as long as cabin leakage remains constant. Figure 66 shows how the oxygen concentration varies with leak rate. This method will be used as the primary back up for the oxygen partial pressure sensor control method. It is maintained within adjustment by periodic operation and a check of the resulting oxygen partial pressure.

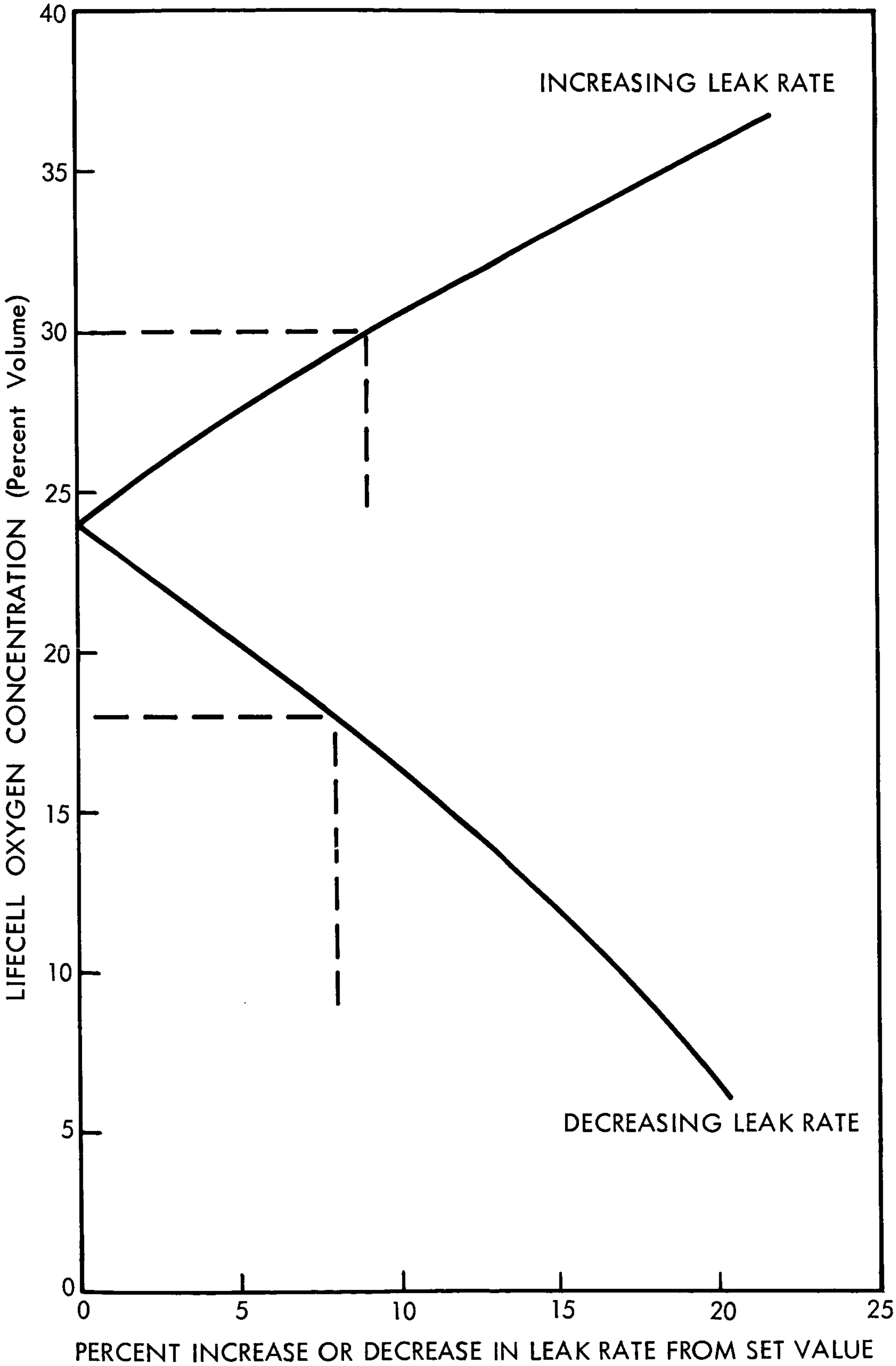


Fig. 66 Bleed Gas Oxygen Concentration Control System (Separate O_2 and N_2 Tanks – Fixed N_2 Bleed)

Low pressure oxygen control (5.0 psi): The capability is also provided to revert to a 5.0 psia pure oxygen system through ground command in the event of failure of the two-gas system. In this mode, oxygen is supplied to maintain cell pressure through the 5.0 psi total pressure regulator. Cabin oxygen enrichment will be accomplished prior to depressurization to 5 psia to prevent animal anoxia.

Atmosphere recirculation and revitalization. – Atmosphere recirculation and revitalization maintains cell nominal conditions of 75°F temperature and 50 percent relative humidity. The system is designed to maintain a volumetric flow of 424 cfm through both cages which provides a velocity within each cage of 30 ft/min to carry waste materials into the waste management system. Exiting from the waste management system, the flow divides into two main branches. These branches are further divided into high and low volumetric flow paths.

The gas in the high flow loop is not processed but is circulated to maintain waste removal. Circulation is effected by the high flow fans located in the header covering the animal cages. One high flow fan over each enclosure is running continuously and each fan is capable of producing a volumetric flow of 200 cfm at 0.1 in. $H_2O\Delta P$. The 200 cfm flow for both cages (400 cfm total) is circulated from the waste management system through the four ducts, one located on each side of each animal enclosure, into the enclosure header, through the fans, down the life cell and back to the waste management system. Reverse flow due to fan failure is prevented by incorporation of low pressure drop check valves. An additional fan over each cage is provided for redundancy.

One purge fan is installed over each enclosure to aid in the transport of waste material to the waste management system. These fans are capable of producing 2000 cfm. Each fan is operated for a period not exceeding 5 min., twice a day. The purge fans are not operated simultaneously. These fans are provided with a counter rotating mass to nullify angular momentum changes.

The 200 and 2000 cfm flows utilize the same ducting. The ducting was sized to provide minimum pressure drop for the 2000 cfm flow. Calculations indicate duct, entrance, and exit losses to be 0.150 in. of water for this flow rate exclusive of the check valve. Design of the check valve to afford a pressure drop equal to or less than 0.05 in. of water will maintain pressure drop losses less than 0.2 in. of water. An alternate approach would be to locate the check valve on the outlet side of the fan thereby reducing area losses while also providing some diffusing action into the cage. The duct pressure losses for the 200 cfm fans are calculated at 0.0015 in. of water.

The atmosphere in the low flow loop carries moisture from animal respiration, perspiration and from vaporization of moisture from urine and fecal material in the waste management system. The flow passes through the trace contaminant and carbon dioxide removal canisters, through the condensing heat exchanger and water separator assembly, temperature control valve, fan assembly and into a header which supplies flow to each animal enclosure. The low-flow gas mixes with the high flow gas near the inlet to the cage, producing the desired cage temperature. The cell mixed inlet temperature is sensed and used to actuate the temperature bypass control that controls the flow around the condensing heat exchanger. The total process flow through the cage is provided by the fans in each low flow path. A second fan is provided in each path for redundancy. In the event one low flow path fails entirely, maximum flow will be directed through the active heat exchanger maintaining both thermal and humidity control for both cages within the design limits. The crossover ducting between both lithium hydroxide canisters permits gas to be drawn through both canisters when only one low flow path is active. There is no flow in this duct under normal conditions when fans in both paths are active. The low flow fans run continuously and each fan is capable of producing a volumetric flow of 12 cfm at 3.5 in. $H_2O \Delta P$.

Pressure loss calculations for the flow rates and duct sizes of the system indicate a total drop in the low flow loop of 2.17 in. of water. The losses are estimated as listed below:

Section	ΔP (in. H_2O)
Waste Management (12 cfm)	0.009
Lithium Hydroxide Canisters (12 cfm)	1.223
Heat Exchanger (7.8 cfm)	0.89
Water Separator (7.8 cfm)	0.042
Ducting Losses	0.01
Total	2.17

The two low flow and two high flow fans which are operating continuously also contribute to gyroscopic stiffness as required by the attitude control system. All fans are installed with axes parallel and rotate in the same direction. Reverse flow due to fan failure is prevented by incorporation of low-pressure-drop check valves.

Gas composition is sampled from a location approximately midway down the cell enclosures, and analyzed by integrated instrumentation sensors. The low flow is drawn through the sensors by the low flow fans.

Contact was made with Rotron, Joy and Torrington fan manufacturers to determine the feasibility of fan selections and capability of meeting the mission requirements. LMSC was advised that the 12 cfm and 200 cfm fans similar to those selected for this mission (Radial Blower Model R Type 3501 and Tubeaxial Propimax 3) are built to military specifications and have exceeded 10,000 hours of operating life in other applications.

Carbon dioxide removal: Lithium hydroxide (LiOH)* is provided to maintain the life cell carbon dioxide partial pressure below the 1 percent maximum limit. Each animal is estimated to produce 0.335 lb of carbon dioxide per day at an average metabolic rate of 1.5 basal.

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*Lithium hydroxide – Anhydrous, environmental grade
6 x 8 granules
Type II
Foote Mineral Company
Route 100, Exton, Pennsylvania

The weight of lithium hydroxide required to remove the carbon dioxide produced is found as follows:

$$W(LiOH) = \frac{W(CO_2)}{K}$$

where

$W(LiOH)$ = Weight of lithium hydroxide (lb)

$W(CO_2)$ = Weight of carbon dioxide produced (lb)

K = Lithium hydroxide adsorption capacity (lb CO_2/lb LiOH) = 0.892 (97 percent of K theoretical = 0.92)

$$W(LiOH) = \frac{(0.335)(379)}{0.892} = 142 \text{ lb/animal}$$

The total weight of lithium hydroxide required to remove carbon dioxide from the atmosphere for the total mission duration and including a 10 percent margin is 312 lb. The quantity is divided between each of the two low flow paths. Suitable filter media is provided to prevent LiOH dust from entering the life cell atmosphere. Figure 67 presents the LiOH requirements. The adequacy of the flow rate through the bed is next determined from

$$\dot{m} = F \eta_r C_i \quad \text{(Ref. 10)}$$

or

$$C_i = \frac{\dot{m}}{F \eta_r}$$

where

$\dot{m}$ = Contaminant production rate (0. 67 lb/day for two animals)

F = Volumetric flow rate (24 ft^3/min or 34,600 ft^3/day)

η_r = Removal efficiency (assumed as 0.5 as an easily achievable design)

C_i = Contaminant concentration (lb/ft^3)

The maximum allowable concentration is 1 percent of 14.7 psi = 0.147 psia or $9(10^{-4})$ lb/ft^3. Solving for the contaminant concentration (C_i) from the above equation

$C_i = 0.39(10^{-4})$ lb/ft^3 which is much less than the allowable concentration.

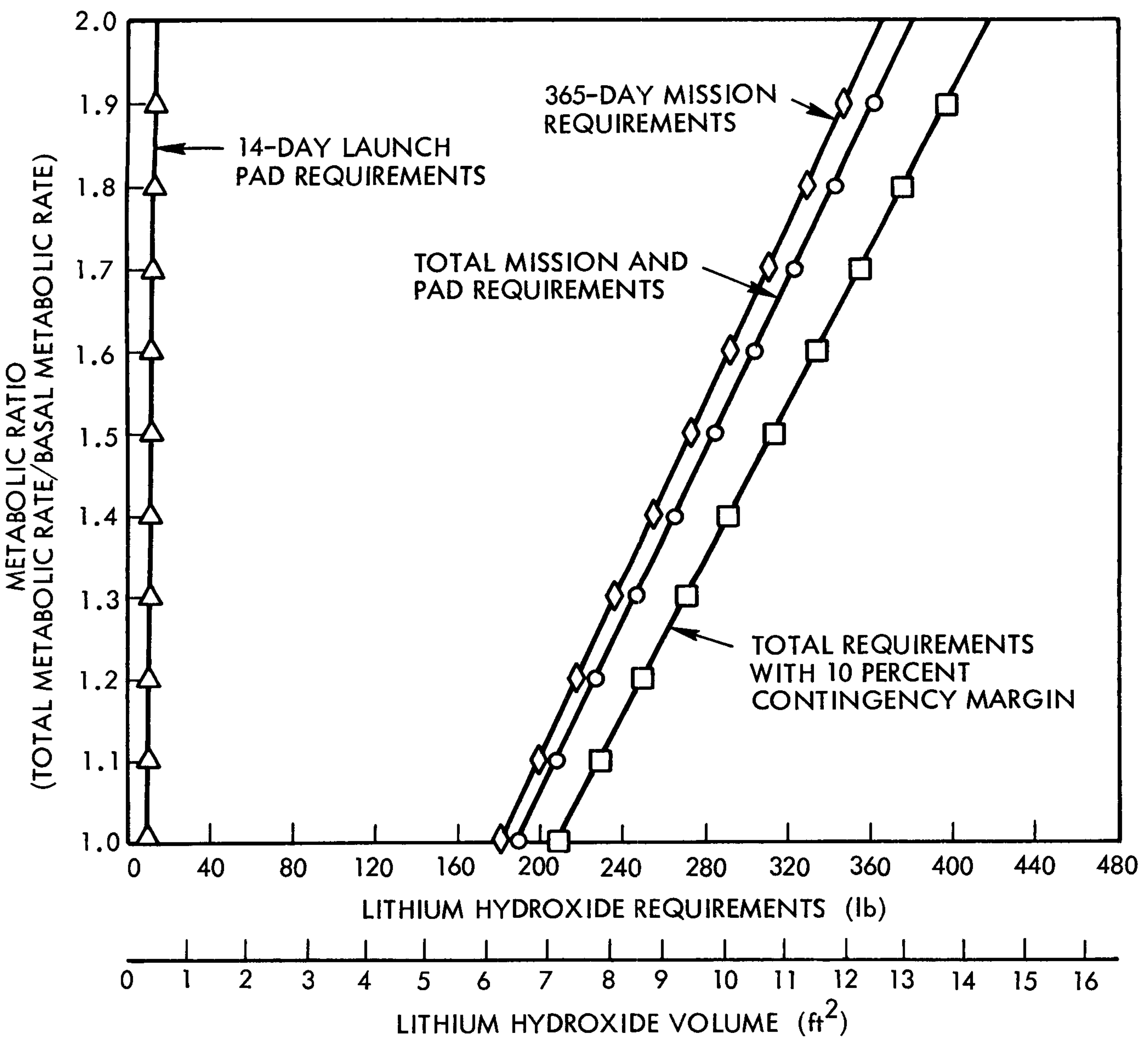

Fig. 67 Lithium Hydroxide Requirements for Two 6.0 kg Rhesus Primates (Basal Metabolic Rate 48.2 Btu/hr)

Trace contaminant removal: The principal trace contaminants of concern are methane, ammonia and carbon monoxide. The hazard of combustion from methane buildup in the lifecell is eliminated by normal overboard leakage. This minimum leakage value is determined as follows:

Allowable methane equilibrium concentration = 4.6 percent (Ref. 11).

Methane production of the Rhesus is based upon a conservative estimate of fecal output equal to 50 percent of food intake. At 1.5 BMR the fecal output will be 112 gm/day total. Data available from chimpanzee tests (Ref. 12) show a methane production of 221×10^{-5} lb CH_4/day for a daily fecal output of 140 gm/day. The Rhesus methane production rate is therefore

$$112/140 \times 221 \times 10^{-5} \text{ or } 177 \times 10^{-5} \text{ lb } CH_4/\text{day}.$$

The flow rate required to limit methane concentration is determined by rearranging the same equation previously defined for carbon dioxide removal. In this case, however, since the quantity of methane being leaked overboard is the total concentration in the mixture, the removal efficiency (η_r) will be 1.0.

$$F = \frac{\dot{m}}{\eta_r C_i}$$

or

$$F = \frac{\dot{m}}{C_i}$$

The allowable contaminant concentration is:

$$C_i = \frac{P}{RT} = \frac{(0.046)(14.7)(144)}{(96.4)(535)} = 0.00187 \text{ lb/ft}^3$$

$$F = \frac{1.77\ (10^{-3})}{1.887\ (10^{-3})} = 0.948 \text{ ft}^3/\text{day}$$

For a mixture density of 0.0732 lb/ft^3 the mass flow rate ($\dot{m}$) is:

$$\dot{m} = (0.948)\ (0.0732) = 0.069 \text{ lb/day}$$

Ammonia (NH_3) and a variety of other contaminants will be eliminated by chemically treated charcoal and overboard leakage. The charcoal,* acting as a substrate for the impregnation of phosphoric acid (H_3PO_4) for ammonia removal, will also remove a substantial quantity of other trace contaminants. It is assumed that the quantity needed for ammonia will suffice for the other contaminants.

It is estimated that the two animals will produce 0.5 gm/day of ammonia (Ref. 13) and that urea breakdown in the urine wicks will contribute another 1.5 gm/day. For a mission of 365 days and a launch pad operation of 14 days, a total of 758 grams of ammonia will be generated. The theoretical capacity of the treated charcoal for ammonia removal is 0.025 gm NH_3/gm charcoal (Ref. 10).

$$\text{Charcoal required} = \frac{758}{0.025} = 30{,}300 \text{ gm or } 66.8 \text{ lb}$$

Using a factor of 0.8 for maximum allowable absorption, the quantity of charcoal required is $\frac{66.8}{0.8} = 83.4$ lb.

A 10 percent contingency margin brings the value to 92 lb required for the mission. This quantity is divided between each of the low flow paths.

The adequacy of the flow rate through the bed is next determined from

$$C_i = \frac{\dot{m}}{F\,\eta_r}$$

where

$\dot{m}$ = 2 gm/day or 4.42 (10^{-3}) lb/day

F = 24 ft^3/min or 34,600 ft^3/day

η_r = 0.60 (assumed as an achievable design value)

The TLV allowable concentration is 3.5 Mg/M^3 or 2.185 (10^{-7}) lb/ft^3. Solving for the contaminant concentration (C_i) from the above equation;

C_i = 2.13 (10^{-7}) lb/ft^3 which is much less than the allowable concentration.

Carbon monoxide will be removed by passing a portion of the process flow through a low temperature catalytic oxidizer (as contrasted with the high-temperature catalytic oxidizer eliminated in the trade-off studies) which uses a 0.5 percent Pd on alumina catalyst which is commercially available. The flowrate through the oxidizer must produce a space velocity low enough to permit sufficient residence time for the chemical

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*Barneby-Cheney Charcoal (activated) — 4 × 10 mesh — Type BD-1
1534 North Knolles, Los Angeles 63, California — Impregnated with phosphoric aicd per LMSC-R-66888

reaction to take place. Approximately 2000 hr^{-1} is desirable for this reaction. Space velocity (φ) and flow rate (F) are related as follows:

$$\varphi = \frac{F}{V\,e}\ hr^{-1} \quad \text{(Ref. 10)}$$

where

V = volume (ft^3)

e = void fraction (0.5)

$$\varphi = \frac{F}{0.5V} = 2000\ hr^{-1}$$

$$\frac{F}{V} = 1000\ hr^{-1}$$

The carbon monoxide production rate ($\dot{m}$) is estimated at 0.40 cc/hr or 10.1 (10^{-7}) lb/hr (Ref. 13). The TLV allowable concentration is 29 Mg/M^3 or 1.81 (10^{-6}) lb/ft^3.

The removal efficiency (ηr) is assumed to be 0.08. The flow rate through the oxidizer is then found

$$F = \frac{\dot{m}}{\eta r\ Ci} \quad \frac{10.1\ (10^{-7})}{0.08\ (1.81)\ (10^{-6})} = 7\ ft^3/hr$$

or 0.1165 ft^3/min

The required oxidizer volume is now determined

$$V = \frac{F}{1000}\ hr^{-1} = \frac{7}{1000} = 0.007\ ft^3$$

or 12.1 $in.^3$ total.

It has been found, however, on past tests (Ref. 14) that for a volume of 9.4 $in.^3$ at 0.25 cfm the oxidizer becomes poisoned by contaminants in approximately 30 days. The volume required must therefore be increased by a factor of $\frac{379}{30}$ = 12.6. The catalytic oxidizer volume is 153 $in.^3$ and is divided evenly in the LiOH canisters (4). The flow rate is also divided to maintain the proportionality and is therefore 0.025 ft^3/min through each oxidizer.

Humidity control: The relative humidity within the lifecell is maintained at a nominal value of 50 percent by condensing water vapor from the gas flowing through the condensing heat exchanger. The coolant temperature circulating through the heat exchanger provides a dew point temperature of 51°F.

The condensate water extracted from the airstream by the heat exchanger is collected by the water separator and transported to external water storage tanks by a 5.0 psi differential pressure between the gas side of the separator and the storage system.

The water separator is designed passively using hydrophobic and hydrophilic materials to separate the water from the passing gas. The collected liquid is passed through a microporous filter by the differential pressure. This item is currently being developed by LMSC for NASA, Langley Research Center.

Thermal control. — Thermal conditions are maintained by an active coolant system utilizing a 62-38 percent glycol-water solution circulated by a pump. The fluid is circulated through the condensing counterflow heat exchanger where it condenses water vapor from the circulating gas and accomplishes necessary cooling. The heat exchanger is of plate fin construction and most likely constructed of steel. The heated coolant is then routed to the battery cold plate where during normal operation, it receives additional heat. The fluid is then directed to the radiator where the excess heat is rejected to space. A thermally-actuated bypass valve controls the radiator outlet temperature to maintain a constant 45°F. The coolant is then circulated through a pressurized accumulator which: (1) acts as a "cold sink" during launch and early flight phases, (2) provides for thermal expansion, (3) acts as a reservoir for leakage makeup, (4) maintains minimum pump inlet pressure to prevent pump cavitation, and, (5) damps out pump surges. The coolant lines entering the lifecell from the radiator and proceeding to the accumulator, pump package, and heat exchanger are below lifecell dew point temperature and are insulated to prevent condensation. All equipment between these points including heat exchangers, water separator assemblies, flow meters, pump package and accumulator, are also insulated. In the event the radiator outlet temperature approaches the minimum allowable design temperature for the coolant (-45°F), the outlet temperature sensor will actuate the electrical heater on the inlet side of the radiator. Coolant servicing disconnects are located in a place readily accessible for ground servicing. Table 32a presents the method of temperature control for each phase of the mission. A summary of lifecell thermal loads are also listed in Table 32b.

The system heat and moisture balance for various conditions are shown in Figs. 68 through 73 . The calculated average loss by cell heat leak is 15 Btu/hr. A condition of minimum load, a heat leak 10 times average is shown in Fig. 72 .

Expendable requirements. — A tabulation of expendable requirements is given in the following table. A 10 percent contingency margin is included. Figure 67 shows the lithium hydroxide requirement. Other expendables (food, water and metabolic oxygen) were discussed in Design Criteria and System Requirements section. The gas quantities listed below include metabolic oxygen requirements and an overboard leakage rate of 0.3 lb/day. Expendable quantites for two 13-lb Rhesus monkeys are:*

Oxygen (lb):	246	Lithium hydroxide (lb):	312
Nitrogen (lb):	95	Charcoal (lb):	92

CO catalyst (in^3): 153

*379-day normal mission, plus 10 percent contingency

TABLE 32a TEMPERATURE CONTROL METHOD VS. MISSION PHASE

Mission Phase	Temperature Control Method	
	Cooling	Heating
Prelaunch	Evaporative Heat Exchanger Utilizing Ground Freon Supply	Internal Equipment and Animal Metabolic Input
Initial Transient	Cold Sink Fluid Within Accumulator	Internal Equipment and Animal Metabolic Input
Orbit Condition	Active Circulation of Water Glycol Solution with Radiant Heat Rejection	Internal Equipment and Animal Metabolic Input
Recovery	Animals in Retrieval Canisters, Passively Cooled by Command Module Atmosphere	Metabolic Input

TABLE 32b LIFECELL THERMAL LOADS SUMMARY

Heat Loads	Average (Btu/hr)		Minimum (Btu/hr)	
Sensible				
Animals	120		70	
Lights	48		-	
Psychomotor	75		55	
		243		125
Fans	157		157	
LiOH Bed	20		12	
		177		169
		420		294
Latent				
Animals	24		14	
LiOH Bed	10		6	
Urine	62		62	
		96		82
Total Removed by Condensate Heat Exchanger		516		376
Pump		51		51
Battery Coldplate		218		41
*Radiator Heat Rejection		785 Btu/hr		468 Btu/hr

* Based on adiabatic wall conditions. Anticipated heat leak from the lifecell will reduce radiator rejection requirements. See thermal balance diagrams.

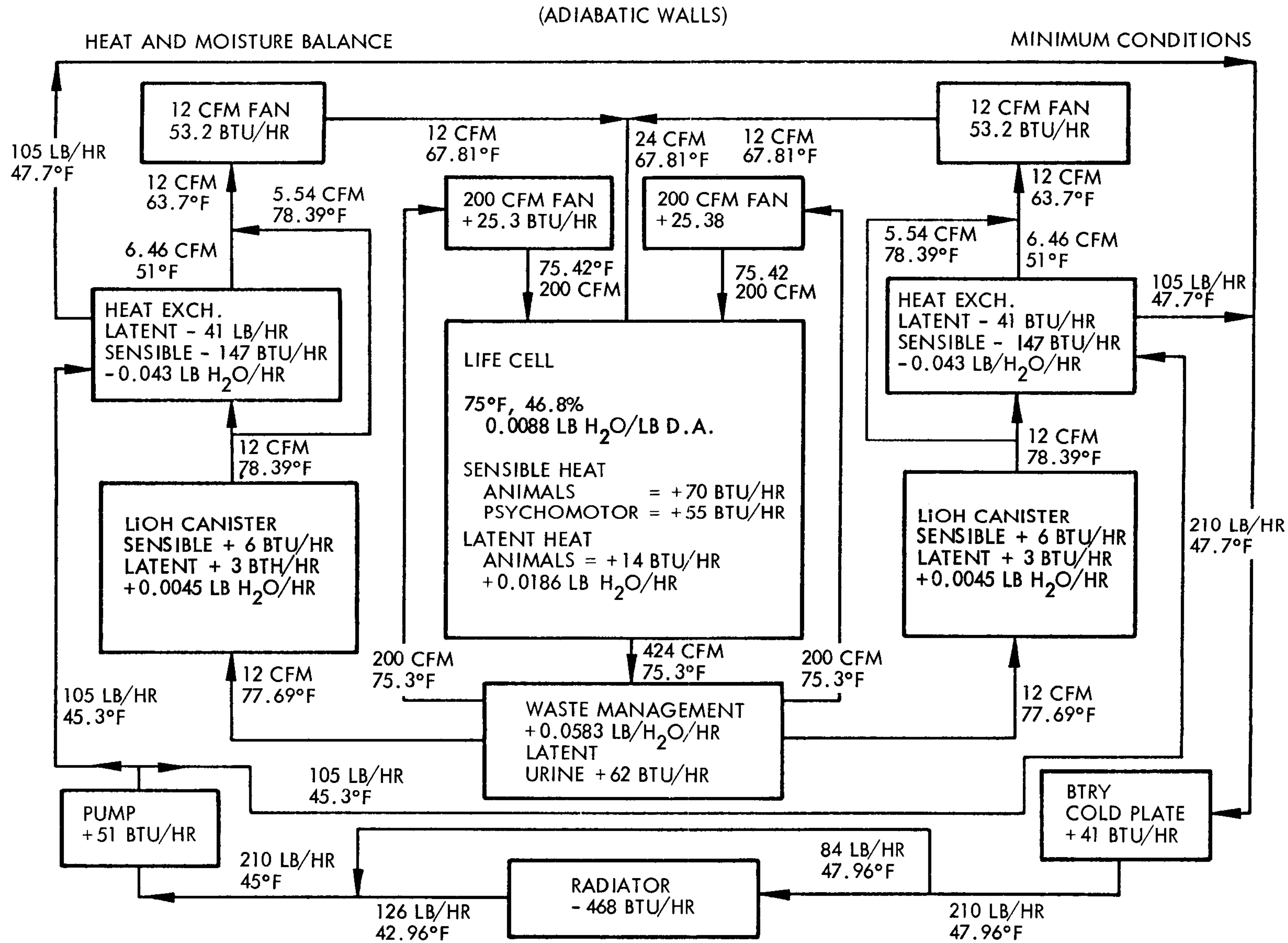


Fig. 68 Heat and Moisture Balance – Minimum

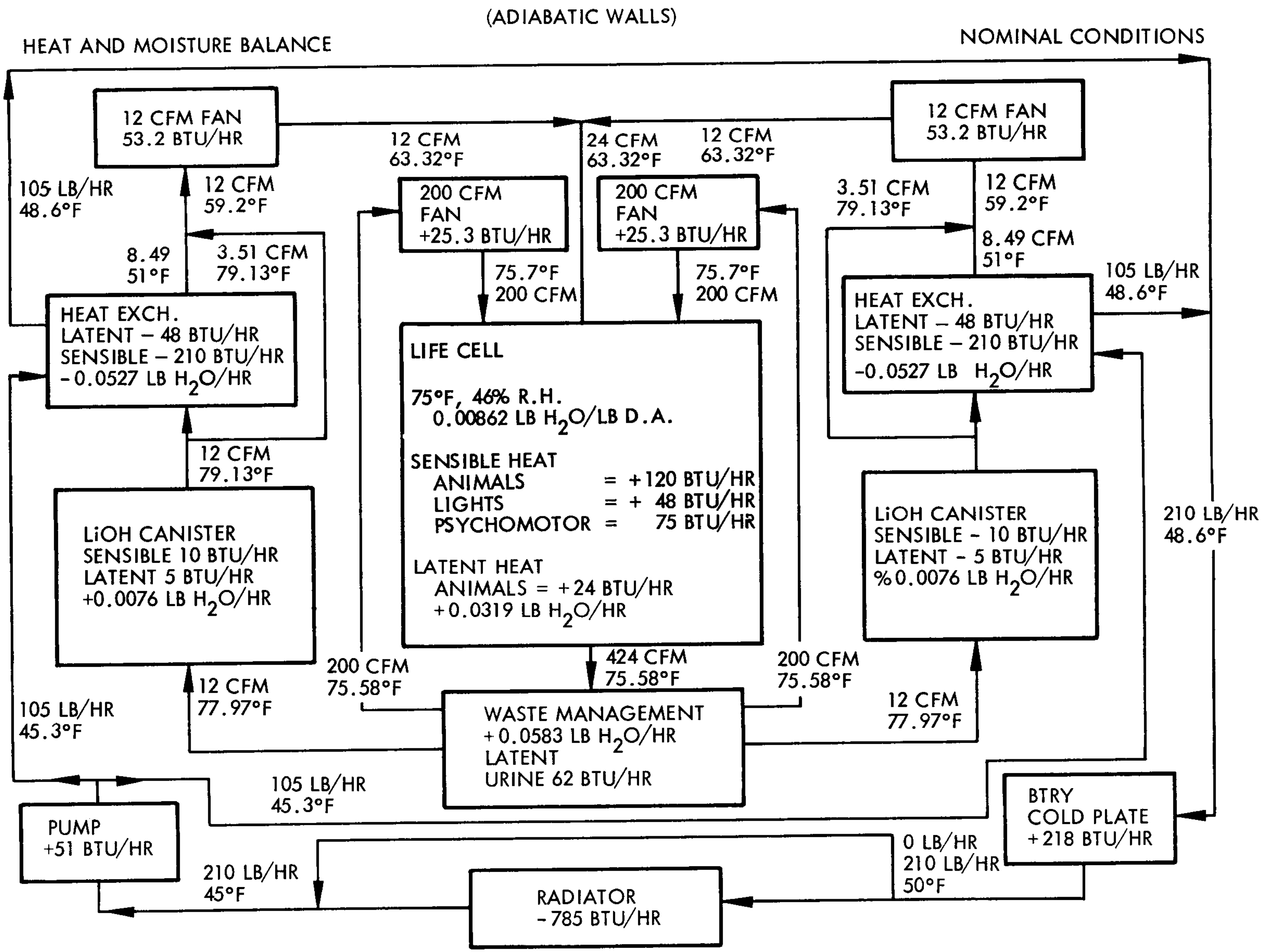


Fig. 69 Heat and Moisture Balance – Nominal

(ADIABATIC WALLS)

HEAT AND MOISTURE BALANCE ONE LOW FLOW LOOP INOPERATIVE

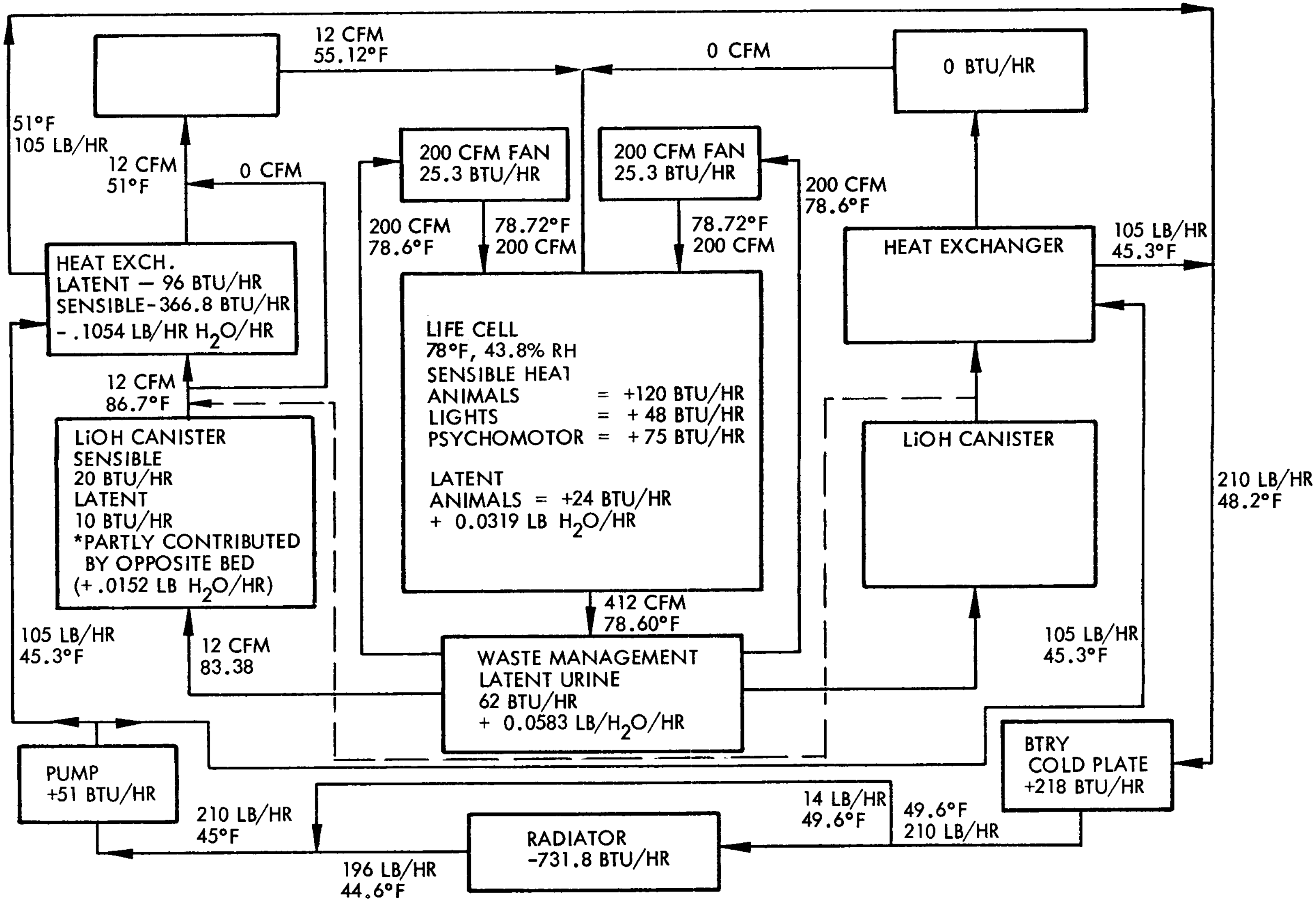


Fig. 70 Heat and Moisture Balance – One Low Flow Loop Inoperative

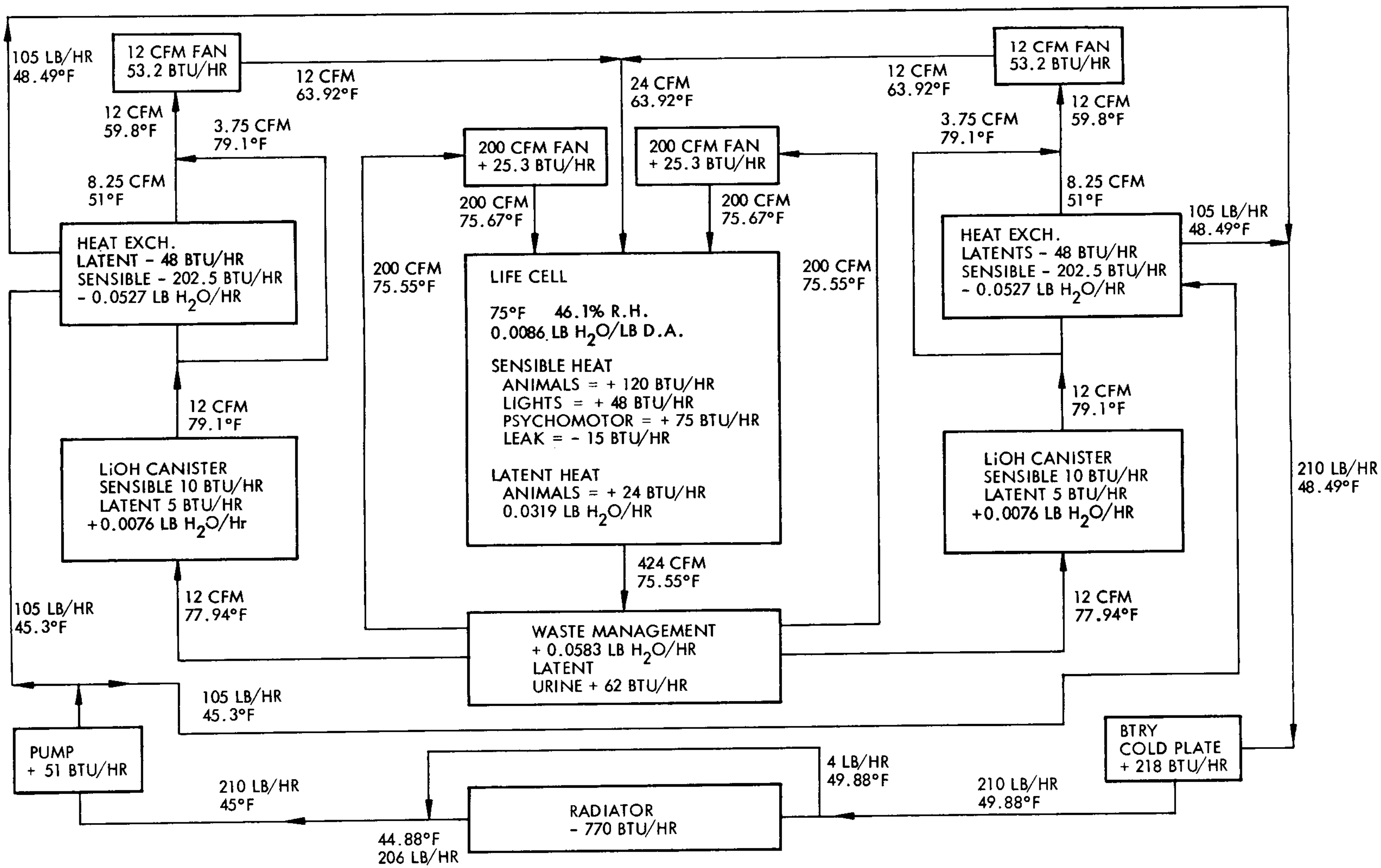


Fig. 71 Heat and Moisture Balance, 15 Btu/hr, Nominal

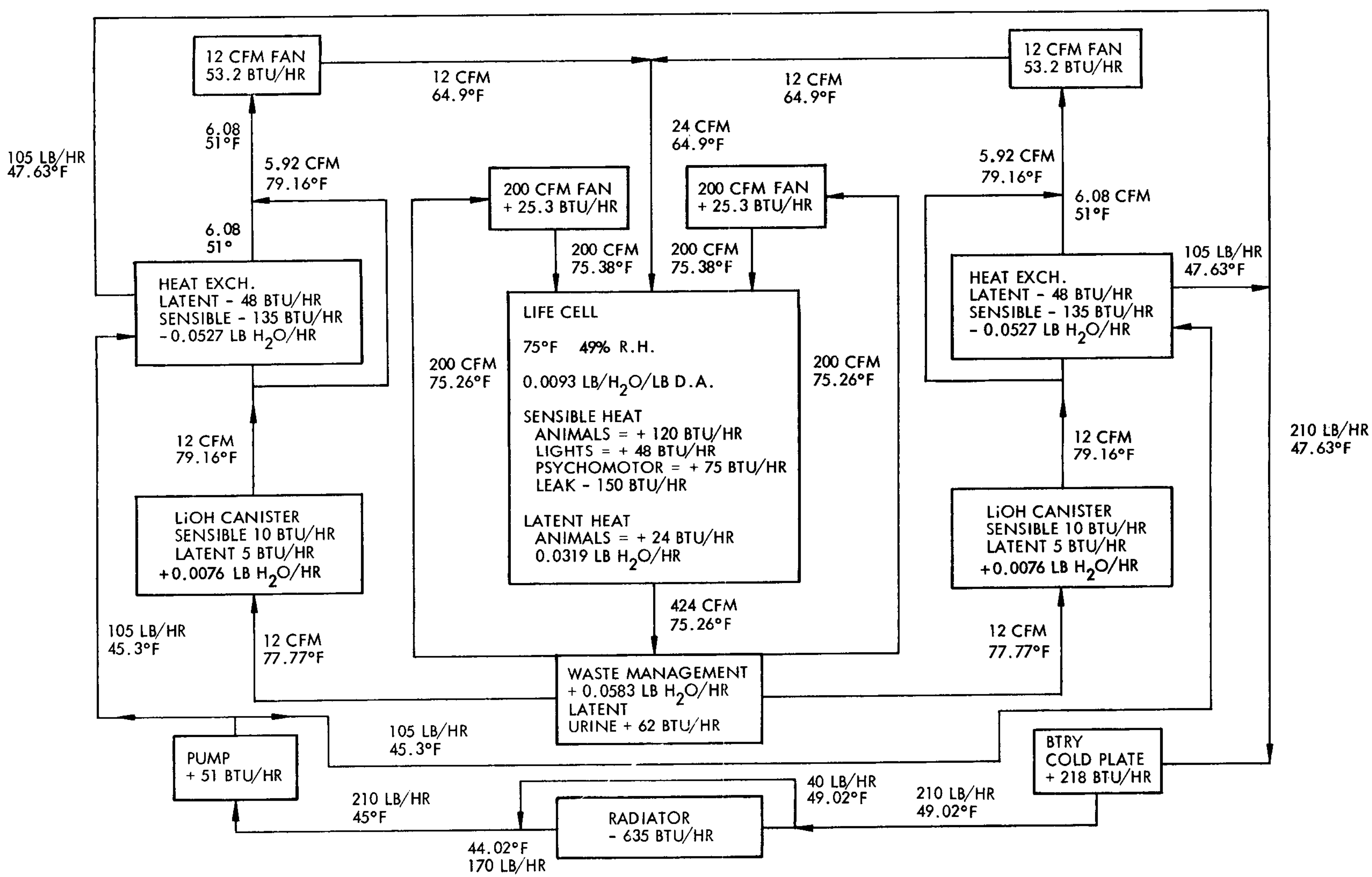


Fig. 72 Heat and Moisture Balance, 150 Btu/hr, Nominal

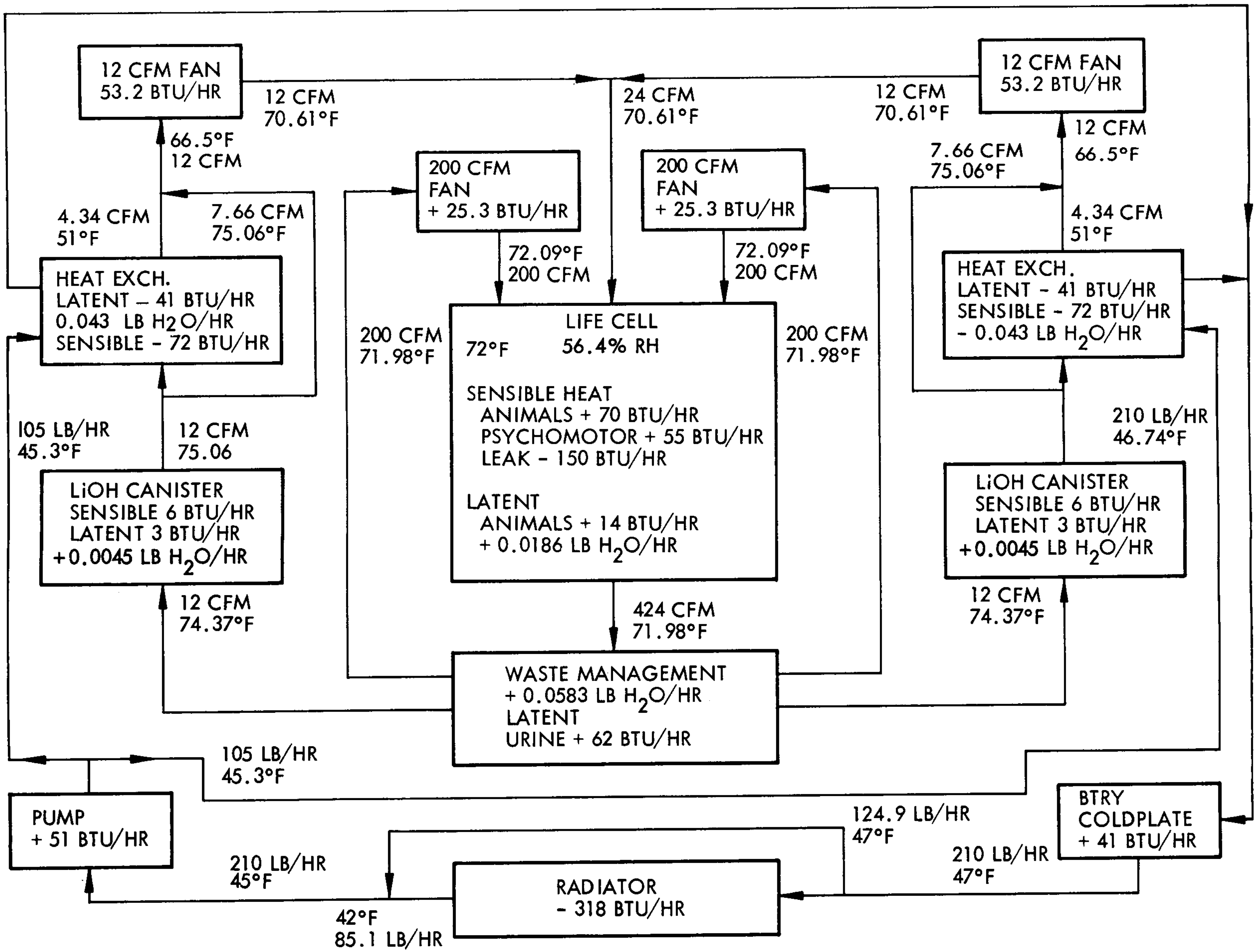


Fig. 73 Heat and Moisture Balance, 150 Btu/Hr, Minimum

Metabolic Support

The metabolic support subsystem comprises food, water, waste management, and living enclosure or cage. Design details in these areas are presented in the following paragraphs.

Feeder and food supply subsystem. – As described in the tradeoff analysis discussion of the feeder, solid food requirements for each animal throughout the experiment will be met by a supply of pellets which will be dispensed on a task reward basis or by an over-riding ground command.

The food supply system is divided into two separate units, one serving each animal. Each unit consists of a storage container and an actuator system, located within the lifecell as shown in Fig. 64. An oral dispensing device protrudes inside the wall of the animal enclosure and delivers pellets directly into the animal's mouth. Pellets are approximately spherical in shape and about 0.5 inch in diameter. The arrangement of the storage container, delivery system and oral dispenser is shown in Fig. 74. This configuration has been developed from a principle employed successfully in earlier experiments.

The possibility of the pellets becoming jammed together in the cylinder has been recognized as a potential problem area. Therefore, in order to circumvent such a problem, a feature is incorporated whereby a slow undulating rotary motion is imparted to the stored pellets.

The storage container is cylindrical in shape, 14 inches diameter by 30 inches long. It is made up of welded 6061 aluminum alloy sheet material with end structure machined from plate. The lower end of the cylinder is closed by a rotating plate with 40 receptacles equally spaced around the periphery. Each receptacle is sized to receive one pellet so that as the plate rotates, 9 deg each feeding cycle, pellets are collected from the container and carried around to the area of the dispensing unit. This sequence, which in effect gives each receptacle 39 chances to collect a pellet before it reaches the oral dispenser, greatly enhances the reliability of the system.

A piston in the upper part of the cylinder, acting under the force of a constant tension spring device, directs the pellets down toward the receptacle plate. The piston is connected to the receptacle plate by a cruciform section shaft upon which it slides, so that the two rotate together. Matching cam surfaces on both the piston and the receptacle plate impart vertical motion to the pellets as the system rotates. An annular ring of rubber "paddles" around the lower part of the cylinder overcomes any tendency for minor pellet jams to develop locally in the area of the receptacle plate.

Having reached the appropriate position, the pellet is transferred from the receptacle plate to the oral dispenser by means of a lever connected to the main actuator unit which is located axially below the food cylinder. The dispensing unit is so arranged that when the animal grasps the mouthpiece with its jaws, a microswitch is activated, causing a solenoid operated cam to slide across, ejecting the pellet directly into the animal's mouth. With such an arrangement, the possibility of the animal rejecting or otherwise wasting the pellet is minimized.

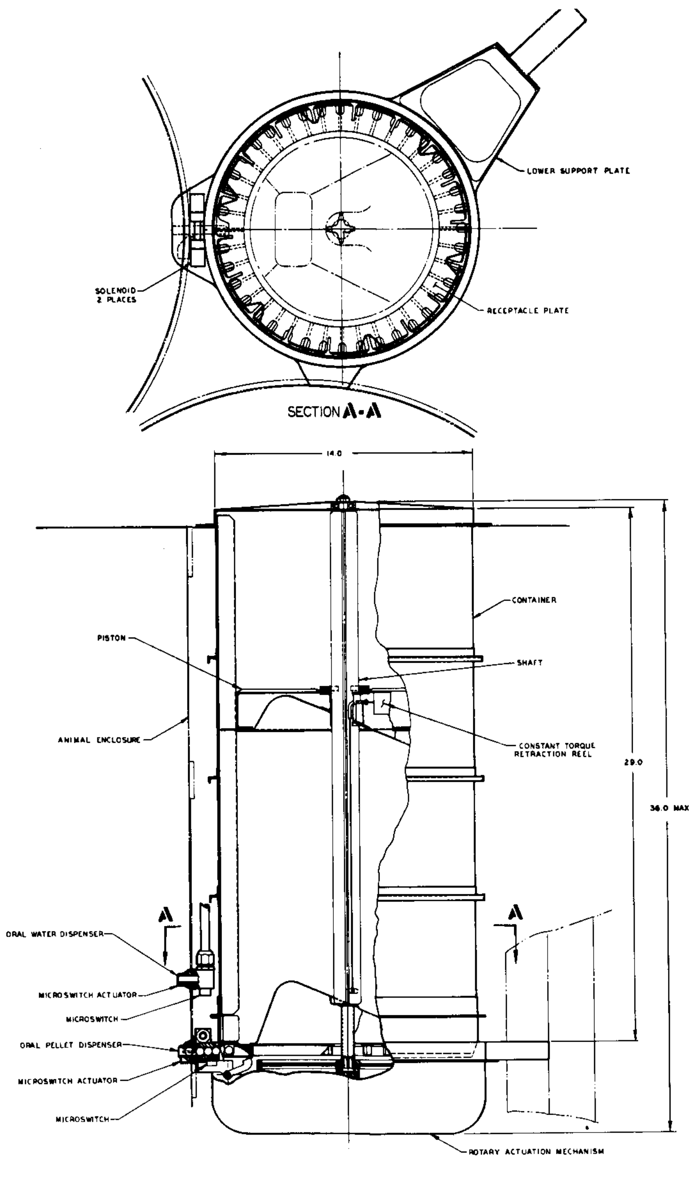


Fig. 74 Water Dispenser and Feeder Layout

In view of the large number of feeding cycles involved over the duration of the experiment, special attention has been paid to ensuring the maximum reliability for this unit. The number of electrical components has been held to a minimum. In the oral dispenser two activating microswitches are provided, either of which will complete the actuator circuit, and the ejector cam is provided with a redundant solenoid system. The main actuator system which rotates the receptacle plate, and also operates the lever to transfer the pellet to the oral dispenser, is driven by a single electric motor, with a second motor providing the necessary degree of redundancy. This system is based on a simple ratchet type escapement with sufficient space available to permit comparatively rugged components, thus avoiding the degradation of reliability which frequently accompanies delicate mechanisms.

In addition to these features, it is possible that the reliability of the overall feeder system could be enhanced further by incorporating a second oral dispenser into the design of each feeder unit. In this way either feeder would have the capability of serving both cages for a reduced period of time. Additional animal training would probably be required, however, since the secondary dispenser, which would only be used in an emergency, would be located away from the food/water/behavioral panel cluster to which the animal would have become accustomed. If it can be ascertained that such training is feasible, then this added redundancy feature could readily be incorporated into the existing design.

Also affecting the overall reliability of the system are two unknown factors associated with the pellets.

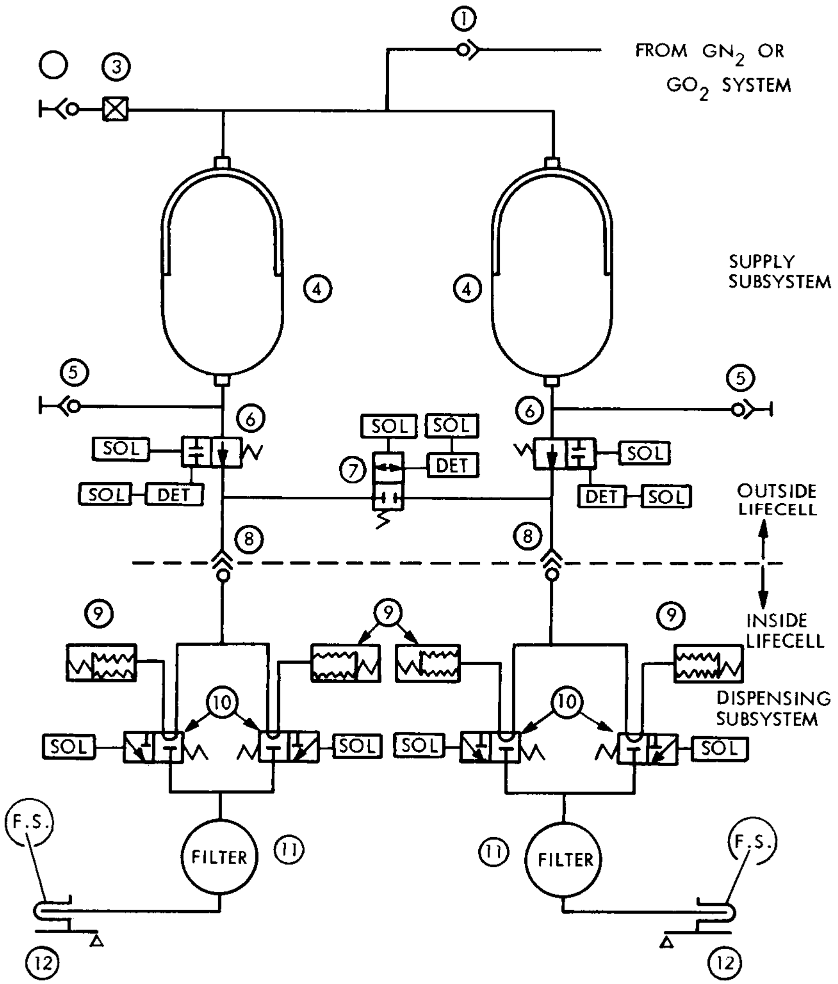
In the current pellet configuration there is a cylindrical ridge around the basically spherical pellet. This may well aggravate any tendency which the pellets may have to jam together when stored over an extended period. The other potential problem area is the surface of the pellet which may be prone to abrasion as a result of the agitation of the pellets in the storage canister. It is also possible that some surface deterioration may take place due to contact with the environment of the lifecell over an extended period. Consequently, it is anticipated that the food pellets will have to be manufactured without the cylindrical ridge and be provided with an edible protective coating to resist abrasion and/or oxidation throughout the duration of the mission.

The proposed method of supplying food in the form of low moisture content pellets does not present any undue microbiological problems. The moisture content of the pellets is below the levels required for cell growth. Therefore, if the conditions for pellet storage are such that this low moisture level is maintained, any microbial contamination of the food would be unable to develop growth. If a problem of moisture control should develop, then it appears feasible, based upon current food technology, to incorporate in the pellets preservatives and/or hydrophobic coatings which would prevent microbial growth.

Drinking water subsystem. – The drinking water supply is contained in two hydropneumatic accumulator tanks with a hydraulic or water reservoir capacity of 386 lb of water in each tank, giving a total supply quantity of 772 lb of drinking water. This quantity was determined on the basis of the total requirements for two 13-lb Rhesus monkeys. The water reservoir side of each tank is separated from the pneumatic side by a flexible, impermeable water-gas separation member. Water expulsion pressurant is either oxygen or nitrogen gas at a regulated pressure of 35 psia and is supplied from the integrated thermal and atmosphere control system, as shown in Fig. 75. The drinking water is transferred to the dispensing part of the subsystem located inside the life cell through a 2-way 2-position normally-open latching and unlatching solenoid valve and a self-sealing through-bulkhead fitting located in the outlet lines from each tank. A self-sealing disconnect fitting is plumbed into the outlet line of each tank to provide a means of filling the tanks with water during the checkout sequence. The drinking water storage tanks are mounted on the outside of the lifecell, one diametrically opposite the other to account for spacecraft center of gravity considerations. One tank is mounted with the water reservoir outlet at the top of the tank, and the other tank with the outlet at the bottom. This arrangement results in no change in the c.g. on the vertical axis as the water is used from each tank. A 2-way 2-position normally-closed solenoid valve with a latch and solenoid unlatch is located in a line that connects the outlet lines of the accumulator tanks downstream from the previously noted solenoid valves. This cross-feed line and valve arrangement permits either tank to supply water to either water dispensing system. These valves are actuated by command only as required in the event of unequal usage from one tank or the other (as indicated from measured water delivery) to prevent or correct any shift in the spacecraft center of gravity. Drinking water dispensers are provided for each animal as shown in the system schematic. Each dispenser consists of an aliquot accumulator, a solenoid valve, a filter, a mouth device with redundant lip switches, and a flow sensor. Redundant aliquot accumulators and solenoid valves are provided in each dispenser due to the high actuation rate (over 100 actuations every 24 hours) necessary to provide the animals' daily water requirement.

All of the components used in the dispenser are located inside the lifecell and outside the animal cages as shown in Figs. 64a and 64b. The mouth device protrudes through the cage wall above the psychomotor task panel and is accessible to the animal from the inside of the cage as shown in Fig. 74.

The drinking water supply is plumbed from the self-sealing through-bulkhead fittings to the inlet port of the 3-way 2-position solenoid valves. The aliquot accumulators are plumbed to the return port of these same valves thus permitting the accumulators to be charged from the supply tanks at all times other than when the solenoid valve is actuated. The aliquot accumulator is a spring-loaded positive displacement device with an adjustable capacity of 1.5 to 2.5 cc and has sufficient force when charged to deliver the entire aliquot through a 0.45 μ filter with a minimum area of 3.0 cm^2 at a flow rate of 2.0 cc/sec. The aliquot accumulator incorporates a linear transducer that permits monitoring of the quantity of water discharged during delivery to the animal since the water is discharged from the aliquot accumulator only when the solenoid valve is energized by the mouth device lip switch. The outlet of the 3-way 2-position



LEGEND

1. GN_2/GO_2 CHECK VALVE
2. SELF-SEALING DISCONNECT COUPLING
3. MANUAL SHUTOFF VALVE
4. HYDRO-PNEUMATIC ACCUMULATOR TANK
5. SELF-SEALING DISCONNECT COUPLING
6. SOLENOID VALVE, 2-WAY 2-POSITION N.O. (LOCKING AND SOLENOID UNLOCKING)
7. SOLENOID VALVE, 2-WAY 2-POSITION N.C. (LOCKING AND SOLENOID UNLOCKING)
8. SELF-SEALING THROUGH-BULKHEAD DISCONNECT COUPLING
9. ALIQUOT ACCUMULATOR
10. SOLENOID VALVE, 3-WAY 2-POSITION
11. BACTERIA FILTER
12. MOUTH DEVICE WITH LIP SWITCH AND FLOW SENSOR

Fig. 75 Drinking Water Supply and Dispensing System and Legend solenoid permits the water to be directed through the filter to the mouth device when the valve is actuated. The flow sensor, located in the mouth device, permits monitoring and gives verification of delivery of water to the animal from the aliquot accumulator. The 0.45 μ filter is provided to prevent contamination of the delivery and supply subsystem due to the animal blowing or otherwise forcing food particles, etc. upstream rather than drinking the water.

The actuation of the solenoid valve, resulting in delivery of drinking water to the primate, is dependent on the behavioral programmer enabling the electrical circuit so that closing of the lip switch on the mouth device will deliver current to the valve solenoid. Normally the delivery of water to the primate depends upon successful completion of tasks at the behavioral panel; however, an overriding ground command may enable the delivery circuit at any time. Selection of the redundant aliquot accumulator and solenoid delivery valve is made by ground command when it is determined that a failure has occurred.

The water supply and dispensing system components and plumbing will be manufactured from stainless steel to prevent corrosion. The entire drinking water subsystem will be aseptically handled throughout its manufacture, assembly and test. It will be finally cleaned and sterilized with a dilute solution of hydrochloric acid and flushed with sterile water prior to loading with drinking water. The drinking water stored aboard the spacecraft will be sterile and free of material that can be readily utilized as primary nutrients to support the growth of microorganisms. Even though a bacteriological filter is used immediately upstream of the mouth device, there is a possibility that microorganisms may "grow" through the filter medium. The probability of this occurring is not known. However, the probability can be made to approach or become zero through the use of silver ion containing filters and/or maintaining an antimicrobial chlorine residual in the drinking water supply. The latter approach is incorporated in the present design.

Waste management subsystem. – As a result of the tradeoff studies, a passive waste management system was selected which collects and separates solid and liquid wastes, including a total of 95 lb of feces and 540 lb of urine. Quantities of wastes other than urine and feces, e.g., emesis, hair, nails, food particles and skin, will be small and can be adequately handled by the present design. Following subsections describe microbiological considerations, general design features, wicking material, purge fan, modular growth possibilities and waste water storage aspects of this subsystem.

Microbiological considerations: The extent to which microorganisms might be generated or their growth supported by the monkey's fecal and urinary waste is unknown. In fact, quantitative microbiological information that could be used for engineering design guidelines was not available in the literature. Information was available from LMSC's independent research project on waste processing and utilization. Representative data on the number of microorganisms in monkey feces is shown in Table 33. It is estimated that the total number of cells per gram of feces may vary from 10^9 to 10^{15}. It can be seen that the counts are very similar for fecal specimens from each of the two monkeys even though they were fed different diets. From this data, it is reasonable to assume that the microbial loading of the waste system from feces would be of the magnitude of 5×10^{12} cells/monkey/day.

TABLE 33

MICROBIOLOGY OF MONKEY FECES

Diet	Monkey No. 1 Experimental Pellets	Monkey No. 2 Standard Monkey Chow
Average Stool wt/Day	63 gm	53 gm
Approximate Microbial Cells/gm		
Total Aerobic	10^{10}	10^{9}
Total Anaerobic	10^{10}	10^{9}
Coliform Count	10^{5}	10^{7}
Streptococci	10^{10}	10^{9}
Estimated Total Cells/gm*	10^{10} to 10^{15}	10^{9} to 10^{15}
Estimated Number of Cells	10^{10} to 10^{15}	10^{9} to 10^{15}
Added to Waste/Day	63×10^{10} to 63×10^{15}	53×10^{9} to 53×10^{15}

*Total is not additive from individual group counts since a large number of aerobes are facultative anaerobes.

The fate of these organisms is not known at this time. In an untreated fecal stool maintained under moisture and temperature conditions that allow metabolic processes to continue, certain organisms will die because of toxic products or lack of required nutrients while others will grow and reproduce until their nutritional source is depleted. Eventually, it is expected that all activity within the feces will cease. However, the point in time when activity ceases is undetermined. An LMSC experiment on storage of human feces indicates that activity continues for several months. In August, 1965, a 115 gm untreated fecal sample was sealed in an 8 oz can fitted with a pressure gauge. Within a few weeks the pressure build-up within the can exceeded 15 psig of the gauge, and the sample was then placed in a 4°C refrigerator. The pressure dropped to approximately 2 psig at that temperature but has been slowly increasing over the two-year storage period until it is now 9.5 psig at 4°C. Thus, for this sample, activity has continued over a two-year period although the rate has been slow as judged by increased pressure at constant temperature.

If drying of the feces occurs, certain groups of organisms apparently die and the others are inactivated by lack of sufficient moisture to carry on metabolic processes. However, there is a significant survival of organisms even though for practical considerations their activity is arrested. This is illustrated in Fig. 76, which shows the survival of bacteria in a fecal sample that has been dried in vacuum (27-28 inches Hg) over potassium pentoxide (P_2O_5). The survival of coliform organisms is quite evident during the first few days of drying whereas the total count of aerobic and anaerobic bacteria decreases approximately one order of magnitude in the first few days and then levels to only a slight die-off. This same pattern has been observed in other samples of dried human feces. A similar pattern would be expected in air-dried feces with the exception that metabolic activity would extend over a longer time period and consequently any gaseous metabolic by-products would be released to the atmosphere.

The re-wetting or maintaining moisture levels of fecal material will encourage the metabolic activities of the microbial population. Growth conditions will be especially enhanced if urine is the wetting agent since in addition to moisturizing, urine also contains nutrient materials for the organisms. Urine is an excellent nutrient medium for a large number of microorganisms. A microscopic count of bacteria in a sample from a 12-hour collection of urine from monkeys revealed an excess of 10^8 cells per milliliter. The presence of feces, urine, food particles, and material shed from the monkey's body establish the potential conditions for luxuriant microbial growth. It is probable that such growth will occur unless suitable antimicrobial processes are incorporated in the system.

Methods for the control of microbial growth can generally be classified as physical or chemical. The following physical methods were considered for their potential use in the OPE:

(1) Drying of waste to less than 20% moisture content
(2) Heating waste to pasteurization temperatures, 80° – 90°C
(3) Reduce temperature of waste to (0° C or less) freezing
(4) Radiation with isotopes or ultraviolet (UV)
(5) Remove major microbial source (waste) from the life cell
(6) Increased salt concentrations due to evaporation

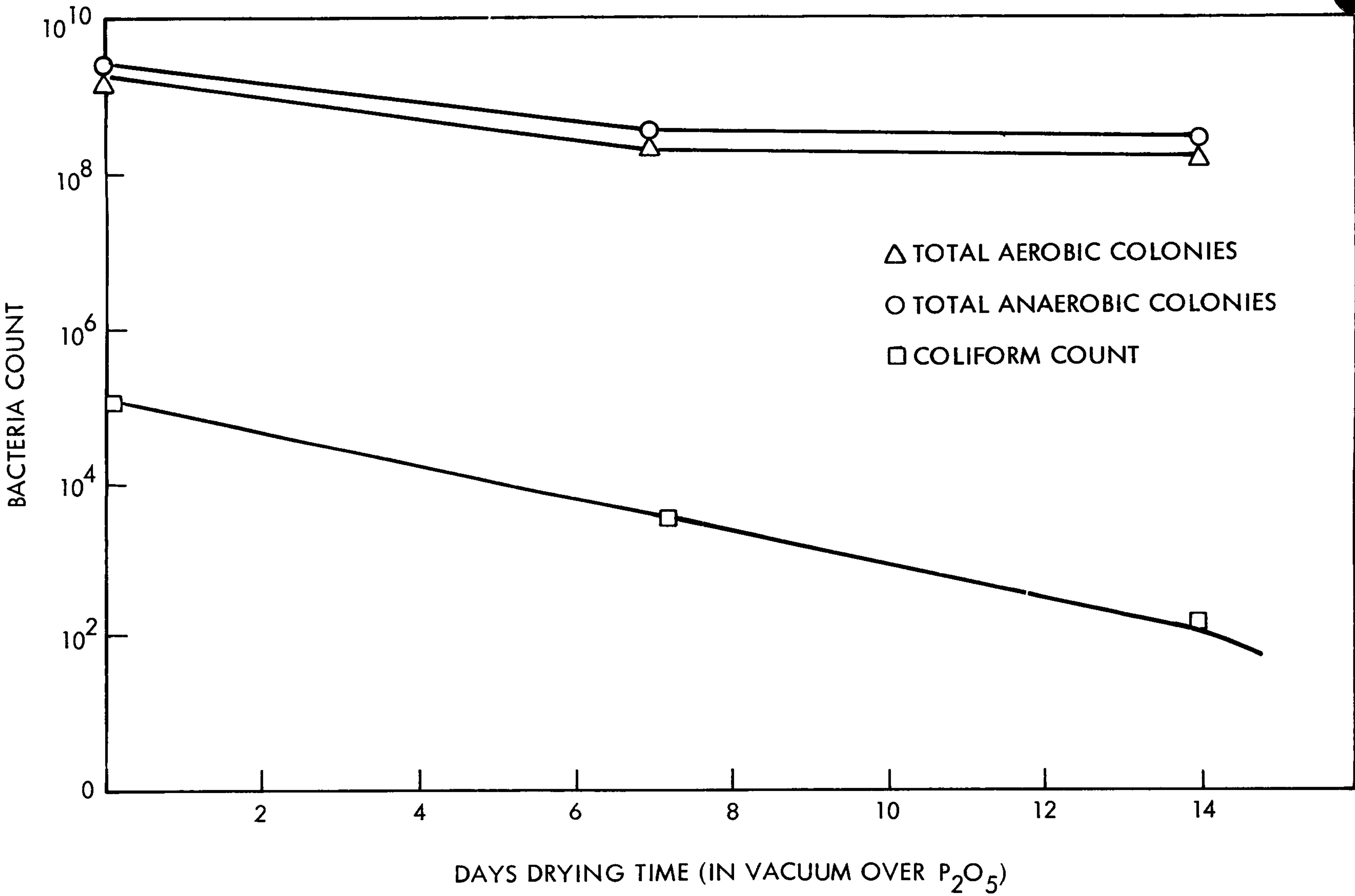


Fig. 76 Survival of Bacteria in Dried Feces

In general, the use of physical methods of microbial control introduce engineering constraints that increase the complexity of the space vehicle and consequently reduce the functional reliability. Items five and six, and the application of ultraviolet radiation may be exceptions to this generality. The removal of waste from the life cell is necessary if the mission is to have any chance of success. Waste removal is required for physical reasons as well as microbiological. Item six, the increase in salts concentration, is a natural consequence of the evaporative process and thus might be considered a bonus derived from urine evaporation. The resulting increase in the concentration of salts due to the evaporation of water from urine and other waste can be inhibitory to many microbial cells. The extent of this inhibition is difficult to estimate but it would be expected to increase with time since there will also be an increasing salt concentration at the evaporative surfaces. It is probable that this effect would not be noticeable in the earlier stages of operation since it would take time for inhibitory concentrations to build up.

Although the use of UV irradiation does not appear to present any undue engineering constraints, its effectiveness to control microbial activity in this application is extremely doubtful. Ultraviolet radiation of 2537Å wave length has been demonstrated to be a lethal agent for microorganisms. However, there are several factors which greatly affect the efficient use of this agent. For microbial inactivation to occur, the microorganism must be "directly hit" by the lethal wavelengths. Also, the penetrability of these wave lengths is very small. Thus, microorganisms which are in the shadows or beyond the depth of UV penetrability are not affected. Thus, microbial cells may be protected in the atmosphere by particulate matter, on surfaces by overlayering with deposited material, or in liquids by suspended matter or by being beyond the depth of UV penetration. The effectiveness of UV in liquids is inversely proportional to the mineral and organic content. Dissolved iron salts have been shown to markedly reduce the germicidal activity of UV. Deposits of material on the UV lamp or reflectors also reduces the emission and reflectance of the lethal wave lengths. In standard practice, the lamps are cleaned at regular periodic intervals to remove deposits of particulates and oily residues. The lamp output of germicidal wave lengths decreases with time, producing the net effect in the OPE application of decreased protection as the potential for greater microbiological activity increases. The direct use of ultraviolet radiation in the life cell is inadvisable since UV is capable of damaging sensitive animal tissues when used in sufficient intensity to have germicidal efficiency. Although the germicidal wave length of 2537Å does not produce the maximum erythemal effect, prolonged exposure to this wave length may lead to undesirable physiological effects. It appears that the several disadvantages in the use of UV and the rather precise control required for its use prohibit its application in the OPE.

With the exception of the removal of waste from the life cell, it is concluded that the physical methods that have been considered for microbial control in the OPE are not applicable to a passive system approach.

Chemical germicides are commonly used for the control of microbial contamination and activity and certain types of these agents may be applicable to the microbiological problems of the OPE. For use in the life cell, a chemical germicide should meet the following requirements:

(1) Have a broad antimicrobial spectrum
(2) Be non-toxic at the use dilution levels
(3) Be non-corrosive to space vehicle materials
(4) Be active in high dilutions
(5) Maintain activity in presence of organic materials
(6) Be active in water-based solutions or suspensions
(7) Remain stable in storage for at least 12 months
(8) Have good diffusion or penetrability characteristics.

A number of chemical germicides were readily eliminated from consideration because of known incompatibilities with the above requirements in addition to others such as gaseous agents that would create engineering problems. The general groups of agents which were scrutinized more closely for their potential use are:

(1) Iodophors
(2) Phenolic derivatives
(3) Chlorine derivatives
(4) Oligodynamic metals
(5) Quaternary ammonium compounds

The iodophors are chemical formulations in which there is a combination of iodine and a solubilizing agent or carrier. This combination or complex either contains or slowly liberates free iodine when diluted with water. The active antimicrobial agent is iodine which is a potent germicide. There are many commercial preparations which utilize surfactants as the carrier or solubilizing agent for iodine and the available iodine in these formulations generally ranges from 1 to 2 percent. They are used primarily as disinfecting and sanitizing agents for eating and drinking utensils and in dairies. It is claimed that the toxicity of these compounds is greatly reduced from that of free iodine in aqueous solutions. An example of the toxicity factor is the acute oral toxicity test on white rats for an iodophor (Mikroklene) manufactured by Economics Laboratory, Inc., of St. Paul, Minnesota. With the concentrated form of the preparation, the LD_{50} is 8.9 gm per kilogram of body weight and the LD_0 is 6.1 gm.

In general, the iodophors are excellent antimicrobial agents. However, their use in the OPE is questionable because (1) the active iodine reacts with organic matter and thus reduces the germicidal potential; (2) their compatibility with materials is problematical; and (3) the effect on the animals of long-term exposure to low-level concentrations of iodine is not known.

The phenolic compounds that are used as antimicrobial agents include a vast number of chemical derivatives and formulations which range in activity from inhibition to killing of microbial cells. Considering these compounds as a group, one is usually aware of such problems as toxicity, corrosiveness, irritating odors, water insolubilities, and inactivation by protein materials that are characteristic of many of the phenolics. Such characteristics would readily eliminate the compounds from use in the OPE. However, derivatives have been developed which reduce or eliminate many of the disadvantages and also increase germicidal activity. It, therefore, appears possible that certain of these derivatives may be applicable to microbial control in the OPE.

Active chlorine is an excellent antimicrobial agent when used under the proper conditions. Chlorination is a common practice for potable water treatment and for sanitizing various food handling and dairy equipment. The active chlorine is usually derived from solutions of hypochlorites or chloramines. A number of other compounds have been derived as carriers of active chlorine. This group of compounds does not appear to be compatible with the requirements for the OPE because of problems associated with corrosiveness, compatibility with materials, reaction with organic matter, and dissipation of germicidal activity, instability in solution, and potential toxic reactions. However, there may be specific areas of application such as carrying a chlorine residual in the drinking water supply as a protective mechanism against possible microbial contamination.

Certain metals, especially the heavy metals, or ions of these metals, exhibit antimicrobial activity. This activity is referred to as oligodynamic, meaning effective in small quantities, and the metals possessing this activity are referred to as oligodynamic metals. Antimicrobial activity has been commonly observed with silver, mercury, copper, and zinc. The oligodynamic action of silver and silver compounds has been studied quite extensively and a variety of processes, including silver-containing filters, have been developed to utilize the antimicrobial action for water purification. It may be possible to use silver ion filters to remove microorganisms from the water system.

The quaternary ammonium compounds, commonly referred to as "Quats" are a special group of chemical germicides characterized as amines in which the nitrogen radical has a covalence of 5 such as found in ammonium chloride (NH_4Cl). The Quats might also be considered as derivatives of this compound in which alkyl and phenyl groups are substituted for the hydrogen atoms. The familiar "Zephiran" or "Benzalkonium Chloride" is an example of this type of germicide. The Quats are widely recognized as effective germicides and the many derivatives that have been developed have been extensively investigated. The advantages of the quaternary ammonium compounds as germicides are as follows:

(1) Broad spectrum antimicrobial activity
(2) Low toxicity levels
(3) Active at high dilutions
(4) Active in presence of organic matter
(5) Good stability

(6) Possess detergent activity
(7) Little or no odor
(8) Low corrosion factor

In consideration of these general characteristics, this group of germicides appears to have the greatest potential for use in the control of microbiological contamination problems in the OPE Waste Management System.

Waste management subsystem design: Utilizing the results of the microbiological study, and adhering to the concept of simplicity as a key to extended mission reliability, the waste management subsystem shown in Fig. 77 was evolved.

The subsystem consists of a wire mesh basket under each animal cage to contain solid waste material. The basket is of 0.25 in. mesh stainless steel to allow liquid waste materials to pass into an area common to both animal cages in which wicking material, which has been impregnated with a germicide or other agent(s) to inhibit bacterial grown and odor production, absorbs liquids passing through the waste collection baskets. The fecal material is separated from the urine in order to effect a reasonable amount of fecal drying to inhibit microbiological growth.

An air flow through each animal cage of approximately 212 cfm at a velocity of approximately 30 feet per minute, in the direction required to move the waste materials from the cage into the waste collection area, is provided by the thermal and atmosphere control system.

Wicking material: The wicking material installation is configured in such a manner that approximately 12 cfm of the air flowing through each animal cage is diverted through the wicking material while the remaining 200 cfm is allowed to return directly to the animal cages.

The 200 cfm air flow from each cage is directed through a 180 deg change in direction, after passing through the solid waste container basket, with sufficient velocity to deposit entrained urine on one side of the wicking material.

The 12 cfm air flow is directed through the wicking material in such a manner that the water content of the urine is evaporated. The wicks are assembled in a convoluted form to provide maximum surface area within a reasonable envelope volume.

The wicking material assumed in the preliminary design is 4 lb/ft^3 heat-felted fiberglass. This material was selected on the basis of its liquid absorption capacity, light weight and inert character. Metal wicks which are also inert were not used because of the weight penalty involved.

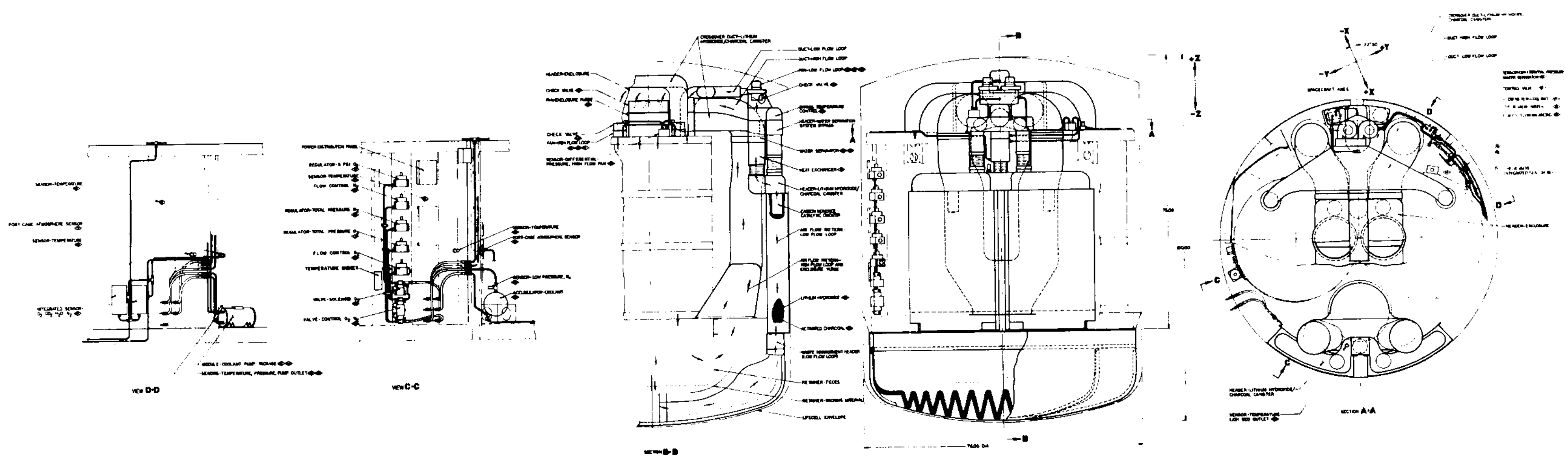


Fig. 77 Thermal and Atmosphere Control Subsystem (Showing Waste Management Subsystem)

Polyurethane and cotton felt wicking materials were also considered and are currently under performance test at LMSC for comparison with the fiberglass. These tests involve daily loading of various wick samples at a urine rate/wick volume ratio of 1.136×10^{-4} lb/in^3/day which approximates the expected operational loading for the design presented. Urine from a female Rhesus monkey is used and it is contaminated with fecal material prior to application to the wicks. Objectives of these tests* include observations of wick clogging and microbiological growth as a function of time for conditions where (1) no germicide is used, (2) iodophor and quaternary ammonium solutions are applied to the wicks concurrently with the urine, and (3) the wick material was initially impregnated with a germicide.

A basic design approach has been to provide a reasonable degree of overdesign regarding the wick/urine ratio. The design presented has a wick/urine weight ratio of approximately 5 percent. A common design value for air evaporation water reclamation systems is 1.5 percent. Tests such as those described above, however, will have to be completed to provide design verification.

The required urine water evaporation rate per cage is 0.7 lb H_2O/day. With a flow of 12 ft^3/min, an increase in specific humidity of 3.8 grains/lb dry air is required. This would raise inlet gas at 75°F and 50 percent relative humidity to approximately 55 percent RH. Wick surface area available to accomplish this mass transfer is 83 ft^2 per cage which allows a considerable margin for uneven urine distribution through the wick.

The pressure drop through the wicking material, arranged as shown in Fig. 78, is not more than 0.5 in. H_2O. Water and other volatiles evaporated from the urine are carried through the lithium hydroxide and charcoal beds for contaminant removal and to the condensing heat exchanger/water separator where water is condensed, separated from the air stream and delivered to the waste water storage system.

Purge fan: A high capacity fan (2,000 cfm) in each animal cage is capable of creating an air velocity of approximately 300 feet per minute through the animal cage. This capability will be utilized for two different purposes: (1) to dislodge fecal or other waste material adhering to the cage walls or floor structure and force the dislodged material into the waste collection basket, and (2), to serve as a noxious stimulus in the avoidance task where the animal responds by climbing into the retrieval canister. By utilizing the behavioral task programmer for the timing mechanism, the animal cages are purged for a duration of one minute with a high velocity air flow four times during a 24-hr period and can, through ground command, be purged more or less frequently.

*Advanced Life Support System Development Program - Waste Management Study (In preparation)

Modular growth capability: One of the salient features of this system is its growth capability. If tests during the hardware development phase of the program indicate the need, the following modifications can be accomplished with minimum design impact:

(1) The fixed waste collection basket can be replaced with a rotatable one with a fixed helical scraper in near contact with the inside wall of the basket. This would mechanically force waste materials adhering to the sides of the basket down into the bottom of the basket, thus removing gross obstructions from the upper portion of the basket.

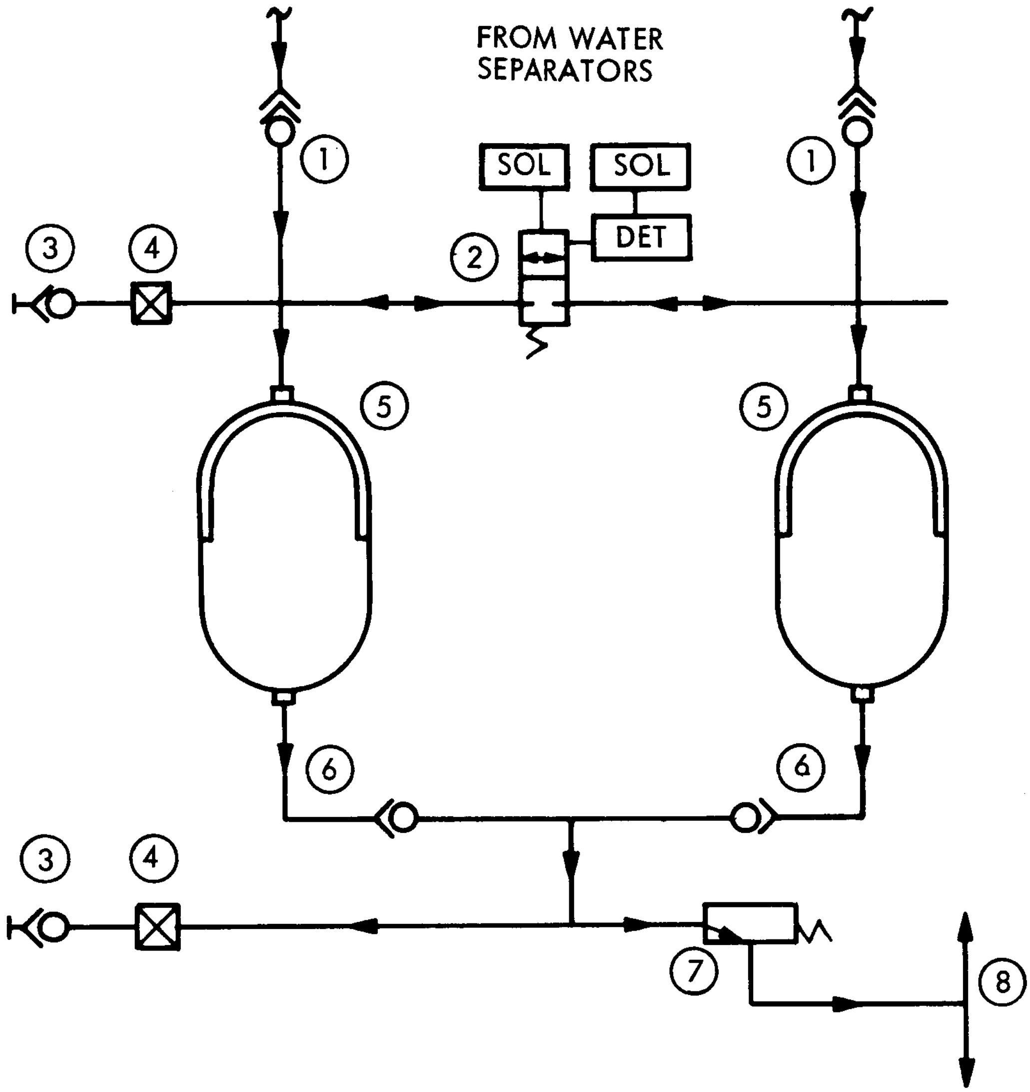
(2) To further assure adequate air passage through the basket walls, one or more rotating brushes on the outside of the baskets, to force smaller and/or hardened waste into the basket, can be added to the system in conjunction with, or independent of, the scraper modification.

(3) A water and germicide flushing system can be added utilizing the accumulated waste water and a suitable germicide to wash down the cage walls, floor and waste management components. This system would be actuated in conjunction with the psychomotor avoidance task, for which the animal is already trained to retire into the retrieval capsule, and with the activation of the 2,000 cfm fan. Ground command through the behavioral task system provides selection capability to the flight director.

(4) The capability to supply a germicidal solution directly into the wicking material can be readily incorporated into the overall system if such a requirement is revealed during development testing.

Waste water storage: A waste water storage subsystem is provided to eliminate the large quantity of gas for the spacecraft attitude control system that would be required if the waste water were vented overboard even through a thrust nullifier. A secondary advantage to providing a waste water storage subsystem aboard the spacecraft is that with a balanced arrangement of storage tanks, there will be little, if any change in spacecraft roll moment of inertia, and center of gravity.

The waste water subsystem designed for use on the OPE will transfer the water from the thermal and atmosphere control subsystem water separator located inside the lifecell to two accumulator tanks located on opposite sides of the outer surface of the lifecell, as shown on Fig. 78.



1. Self-sealing through-bulkhead disconnect coupling
2. 2-way 2-position solenoid valve (locking and solenoid unlocking)
3. Through-bulkhead self-sealing disconnect coupling
4. Manual shutoff vlave
5. Hydro-pneumatic accumulator tank
6. Check valve
7. Vent valve
8. Thrust nullifier

Fig. 78 Waste Water Storage Subsystem

The waste water storage subsystem is shown schematically in Fig. 78 and consists of two hydropneumatic accumulator tanks, each with a capacity of 440 lb of water, which are plumbed to the water separators inside the lifecell through self-sealing through-bulkhead disconnect fittings. Separate plumbing lines are used from each water separator to an individual accumulator tank to prevent imbalance of the spacecraft during the 14 days of operation prior to launch. This imbalance would occur since one tank is installed with the water reservoir at the lower end and the other with it at the upper end to minimize c.g. shift on the vertical axis of the spacecraft during space flight. However, the head difference during the 14 day prelaunch operation would result in all of the water (35 lb maximum) being transferred to the lower tank. A pressure vent system is provided on the pneumatic side of the accumulator tanks and maintains the pneumatic pressure approximately 5.0 psi below that of the lifecell water separators. This pressure differential across the accumulator tank diaphragm provides the pressure for transfer of water from the separators to the tanks with a minimum of functional components. This pressure differential is maintained by a vent valve which is set to crack at 10.5 psia and reseat at 10.0 psia. This valve vents the pneumatic side of the tanks through a thrust nullifier to minimize the generation of upsetting torques. Check valves are installed at the pneumatic outlet of each accumulator tank to prevent equalization of air pressure in the tanks in the event of unequal quantities of water being transferred and resulting differences in displacement and pressure. The normally-closed 2-way 2-position latching and unlatching solenoid valve, installed between the inlets of the hydraulic side of the accumulator tanks, is opened upon orbit injection and permits the transfer of water from one tank to the other or the transfer of water from either separator to either tank.

Self-sealing disconnect couplings and manual shutoff valves (to ensure zero leakage) are installed to permit ground preparation and checkout of the subsystem. This subsystem is placed in operation by removing any air from the hydraulic reservoir side of the accumulator tanks through the disconnect coupling provided. The manual shutoff valve is then closed and the pressure on the pneumatic side of the tanks reduced to 9.0 psia by means of a vacuum pump. The waste water storage subsystem will now transfer water from the separators when the separator dump valves are opened. The pressure differential now existing across the accumulator tanks diaphragms will effect the transfer of all the waste water generated by the primates during their 14-day prelaunch occupancy of the lifecell without further attention to the subsystem. After the spacecraft has been placed in orbit, the vent valve will maintain the required suction pressure during continued subsystem operation.

Cage and retrieval mechanism. – Two individual cages are located inside the lifecell and form part of the internal equipment package. As shown in Fig. 79, each cage provides a cylindrical animal enclosure 34 inches in diameter and 48 inches long. The animal remains in the cage, unrestrained, throughout the experiment, until it enters the retrieval canister for return to earth upon completion of the mission. The lower end of the cage is closed by a grill, through which fecal material passes to the waste management system. A steady stream of air passes through the cage, with periodic purge cycles, entering through ducts and fans located in the top of the cage.

Integral with the cage is a system for inserting the animal, whether dead or alive, into the retrieval canister at the end of the experiment. Access to the retrieval canister is gained through a 14-inch diameter opening in the upper end of the cage. The remainder of the upper end of the cage is closed by a grill.

Of the various approaches studied for accomplishing this, the arrangement finally selected is shown in Fig. 80. The retrieval canister is offset to one side of the cage and a thin liner, which covers the cylindrical walls of the cage, rolls up on a spool located on the side of the cage structure. Short skirts at the top and bottom of the cage, concentric with the retrieval canister, ensure that as the liner retracts, the animal is entrapped in a cylinder, 14 inches in diameter, leading directly into the retrieval canister. A piston housed in the lower skirt is operated by a telescoping pneumatic actuator and lifts the animal up the cylinder formed by the liner and into the canister. The piston is activated by an explosive valve which releases compressed air from a small bottle mounted on the outside of the cage adjacent to the actuator. This approach minimizes gas-supply interface problems.

The basic cage is constructed of aluminum alloy with annular ring frames and longitudinal members. Annular structure consists of machined rings top and bottom and two intermediate rings formed from plate stock. The upper ring includes a number of secondary members which support the lower end of the retrieval canister and other items of equipment located on the top of the cage. A sheetmetal diaphragm covers the top of this ring. The lower ring supports the piston assembly for the retrieval system and is closed by a grill of 1/4-inch diameter stainless steel bars spaced 2 inches apart and welded to a light stainless steel ring. The grill assembly attaches to the aluminum structural ring with mechanical fasteners. Longitudinal structure consists of three formed channel section longerons and a series of sheetmetal panels which enclose about 90 deg of the cage to provide support for the various items of equipment mounted on the outside of the cage structure.

The cage liner is made of stainless steel, 0.005 inch thick. One end of the liner is attached to the cage at a point adjacent to the retrieval canister, while the other end passes through a slot in the cage support structure and rolls onto a 5-inch diameter spool located outside of the structure. The spool is supported top and bottom by brackets integral with the end rings of the cage structure. The upper bracket also supports the electric motor and gear box which rotate the spool to retract the liner prior to retrieval of the animal.

In the area where the two cages are tangential with each other, the liner is pierced by a pattern of closely spaced holes arranged to cover an 8-inch square area thus permitting a limited degree of social contact between the animals during the mission.

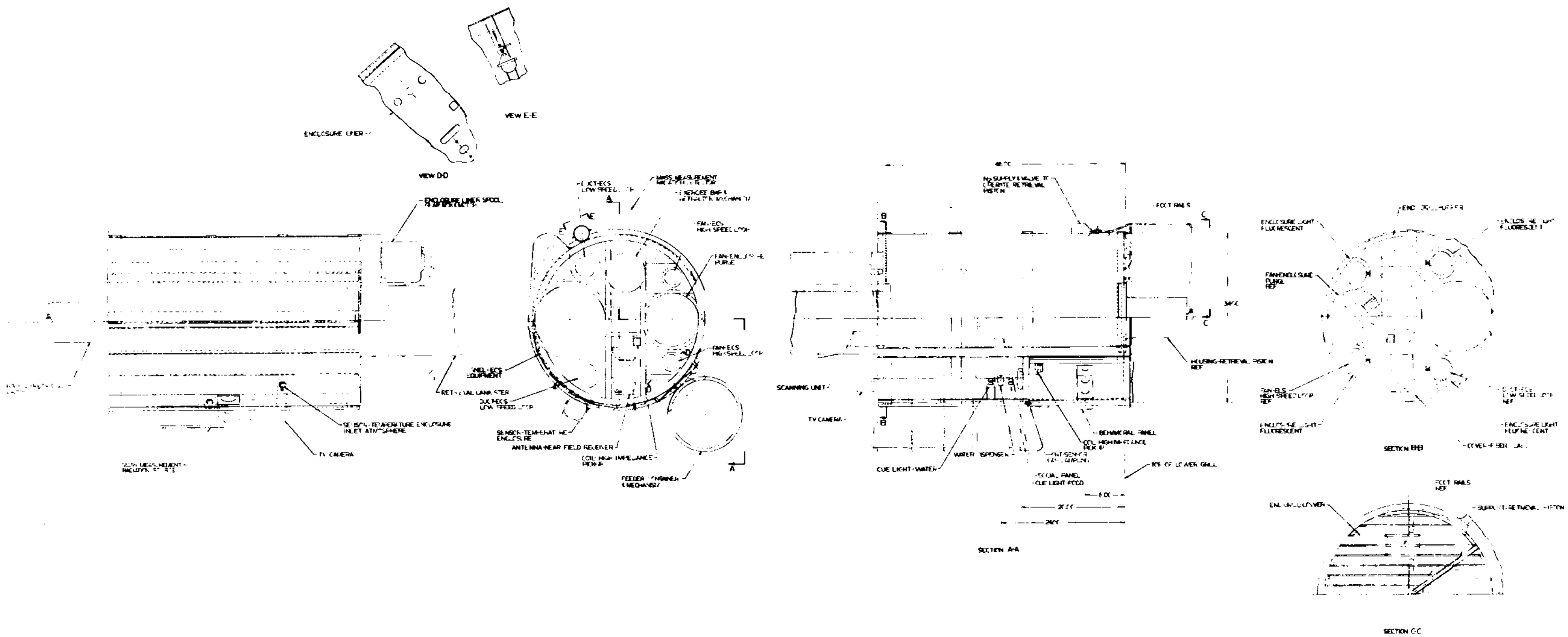


Fig. 79 Cage Layout

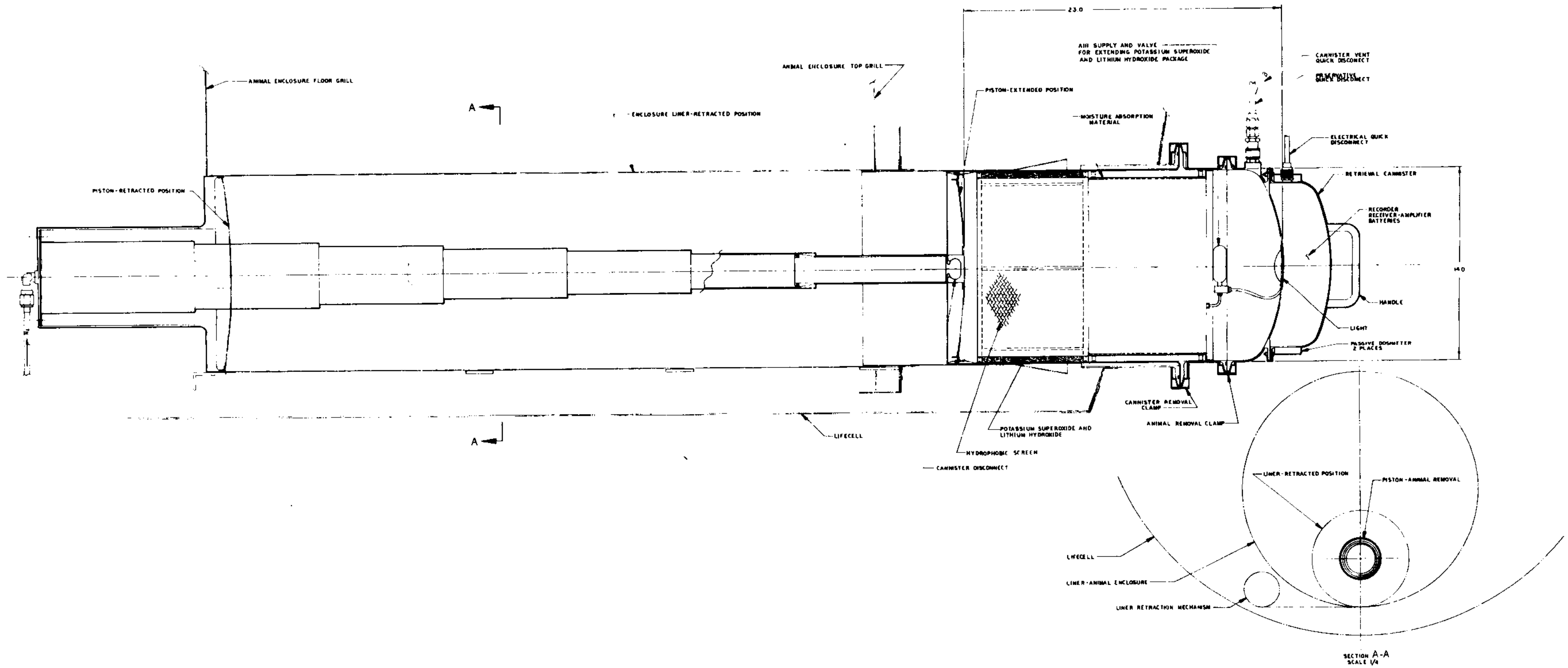


Fig. 80 Animal Retrieval Subsystem Layout

The holes are large enough to permit tactile contact but small enough to prevent finger protrusion from one cage to the other.

The liner is pierced in a number of other places to provide access to the cage for the various functional and life support systems. Three levers on the behavioral panel protrude through the liner, as do the oral dispensers for the pellet feeder and water supply systems. These protrustions are arranged so that there will be no interference as the liner retracts. Other holes are provided for various environmental sensors, an antenna, and the side viewing television camera which is mounted on the outside of the cage structure. Other items mounted on the cage side wall structure includes the various pieces of electronic equipment located inside the lifecell and the mechanical components associated with the environment control system. Also mounted on the outside of the cage are the emission and sensing units of the mass measurement system.

On top of the cage, in addition to the support for the animal retrieval canister, there are mounted two high-flow fans and one large cage purge fan, all enclosed in a fiberglass header which is common to both cages. Between this header and the retrieval canister is located the exercise unit. The exercise unit provides a tee handle inside the cage and is attached to a mechanical/hydraulic system by which various isotonic exercises can be programmed. The exercise unit is so located relative to the mass measurement system that the animal is in the appropriate position for mass measurement during the exercising cycle. The tee handle is retracted up and out of the cage prior to initiation of the retrieval cycle. Also located above the cage is the high-resolution overhead television camera. This unit is equipped with gimbaling optics, which permit scanning of the cage interior. The second television camera, mounted on the side of the cage is set up for high frame-rate visual study of the animal while it is operating the exercise unit. Below the diaphragm which closes the top of the cage are located four flourescent lamps which provide illumination on a programmed basis, and the microphone and speaker for audio monitoring and stimulation. In addition to these components, other antennas are located in this area for monitoring animal activity and receiving biological data from implanted sensors. These antennas are protected from the animal by a fiberglass panel; the remainder of the equipment is protected by a screen of aluminum wire mesh welded to the lower edge of the structural ring.

Microbiological problems of the cage center primarily on the problem of waste handling. The microbiology of the animals' waste or excretory products has been discussed previously. Any accumulation of these products within the cage affords a direct microbiological hazard to the animal. The approach to relieving this hazard is cleanliness, which is promoted by (1) the cylindrical design of the cage so that the chance for waste to collect and accumulate is minimized, (2) the use of nonadhering surface coatings, (3) the periodic purge flow and (4) the potential for periodic germicidal treatment.

In the weightless condition, it is highly probable that waste products at some time, will contact nearly all exposed surfaces within the lifecell. To assist in the control of possible microbial buildup, nontoxic germicidal surface coatings will be used along with possible periodic treatment with a germicide which would also tend to remove any accumulation.

The dessemination or entrainment of microorganisms in the recirculated atmosphere of the lifecell is a potential microbiological hazard to the animals, if the source of these organisms is not properly controlled. Thus, the emphasis is placed upon the control of the origin of the microorganisms and not upon the consequence of the non-controlled source.

Lifecell

The lifecell is located inside the spacecraft where it is supported at its lower end by the internal truss structure of the spacecraft as shown later in Fig. 122. As the principal structural member of the spacecraft, it provides a shell which encloses both animal cages, the food supply, waste management, thermal and atmosphere control and other animal support equipment. Atmosphere, water and other tankage are mounted on the outside of the shell structure. Provisions are also made for the two animal retrieval canisters and the mounting structure for the docking collar.

The lifecell is a pressure vessel which, in addition to accommodating the above equipment, provides structural support for the internal equipment which is integrated into a single chassis-mounted unit. The readily detachable lower dome of the lifecell permits removal of this complete internal package as a single unit (Fig. 81). The lifecell internal equipment will be checked out, except for pressure check, outside of the lifecell, utilizing a ground support fixture provided with all fluid and electrical interface connections. Following checkout, internal equipment is physically mated with the lifecell and all fluid and electrical connections transferred from the ground support fixture to the lifecell. This transfer is facilitated by the use of zero-leak quick-disconnects on all fluid and gas lines.

After extensive study into various possible shapes, the configuration selected for the lifecell consists of a cylinder with domed ends as shown in Fig. 82. The basic pressure vessel is 76 inches in diameter and 100 inches long overall. End closures are domed to 100 inch radius, the upper dome intersecting the cylinder wall at a structural corner joint and the lower dome fairing into the cylinder wall with a 10-inch radius.

Construction generally is of welded 6061 aluminum alloy sheet material with extruded and machined stiffening members on the outside of the shell.

The lower 26 inches of the pressure vessel are removable and attach to the upper structure by means of a vee-band coupling. This lower portion consists of a spun dome welded to a cylindrical skirt and stiffened by an annular ring. The edge member is machined and incorporates the lower lobe of the vee-band coupling.

The upper portion of the pressure vessel, which is the principal component of the lifecell, is made up of a cylinder supported by longitudinal and annular members and a closing dome which is stiffened by radial members, and attaches to the upper ring of the cylinder.

The cylinder is aluminum alloy sheet material, chem-milled to the optimum thickness dictated by pressure loads except around welds and in areas of stress concentration.

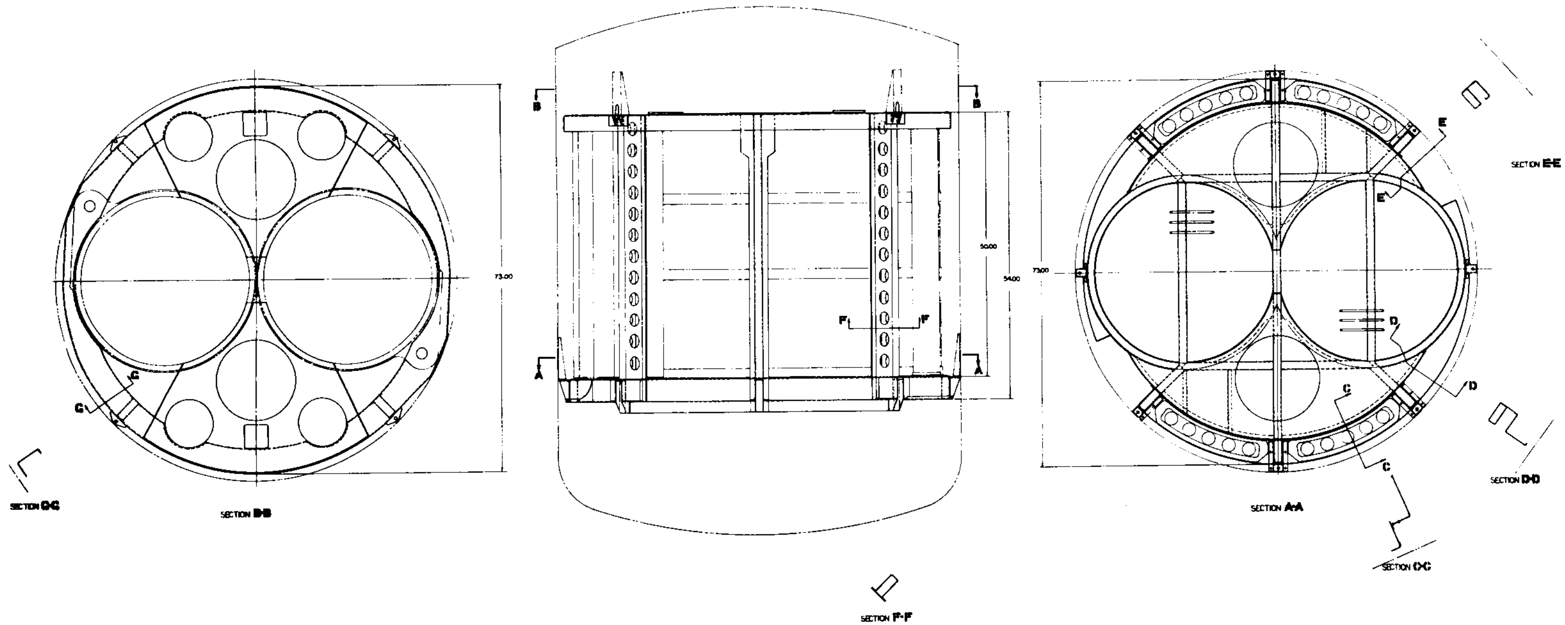


Fig. 81 Lifecell Support Structure

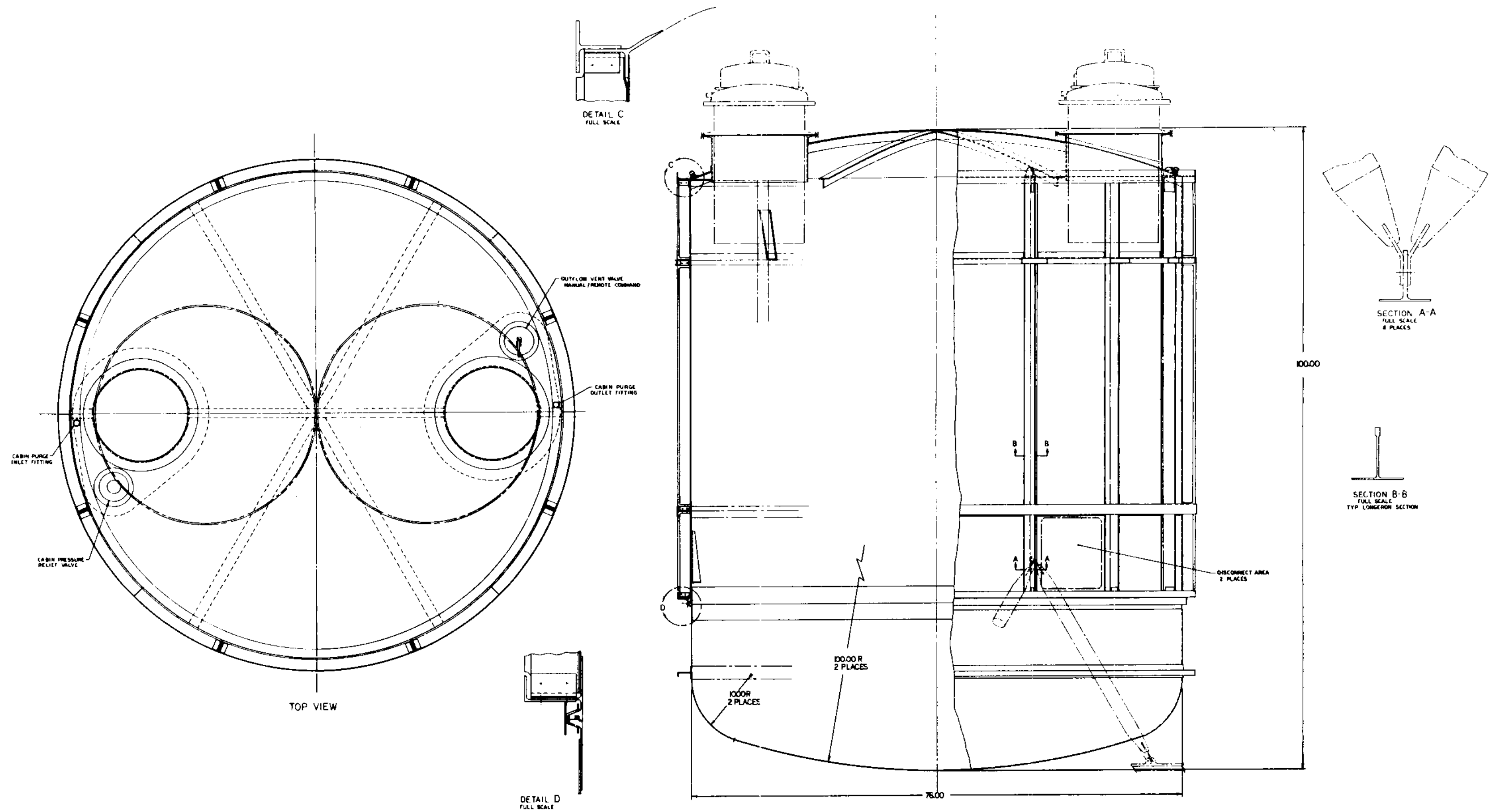


Fig. 82 Lifecell Layout

Longitudinal structure consists of 16 longerons equally spaced around the periphery of the cylinder. These members share with skin in transferring end loads due to internal pressure down to the vee-band coupling. Eight of the longerons, spaced alternately, attach at their lower ends to the internal truss structure of the spacecraft. These joints carry the full vertical load of the lifecell and its equipment during launch. At their upper ends, the longerons react the transient compressive load from the docking collar supports during docking operations. These loads, however, are more than offset by the tension due to the internal pressure. The longerons also serve the secondary function of supporting the side-mounted external storage bottles. The longerons are T-section extrusions welded to the skin and, except in the area where the truss attaches, the outstanding webs are chem-milled to optimum thickness. Annular structure consists of four ring members. The lower ring is a machined T-section of aluminum alloy which includes the upper lobe of the vee-band coupling. The longitudinal members are clipped to the outstanding leg of the T which is stabilized by an aluminum alloy angle attached with mechanical fasteners. Above this is an intermediate ring built up with formed aluminum alloy intercostals welded to the skin between the longerons, and a continuous cap member attached with mechanical fasteners. These two rings react the horizontal component of the load imparted by the support truss and also the couple resulting from the arm between the attachment and the shear face. A similar intermediate ring near the top of the cylinder stabilizes the skin locally and reacts side loads transferred to the pressure vessel by the internal equipment package. The upper ring is a machined Y-section member of aluminum alloy and forms the intersection joint between the cylinder wall and the closing dome. The upper ends of the longerons are clipped to the outstanding leg which is backed up with a rolled steel T-section ring attached with mechanical fasteners. These two rings combine to react the horizontal component of the radial tension load in the dome due to the internal pressure. Eight pie-shaped fittings on the upper surface of the rings at the longeron locations provide attachment for the struts supporting the docking collar.

The upper dome is spun of aluminum alloy sheet material. Six radial stiffeners provide local support in the way of the externally mounted preservative bottles. Two openings in the upper dome, about 14.5 inches in diameter, accept the animal retrieval canisters. Skirts are welded to each opening and incorporate the lower lobe of the vee-band coupling, which seals the canister into the lifecell.

Interfacing with the internal equipment, eight machined fittings inside the pressure vessel near the bottom of the main section of the shell, transfer vertical loads from the equipment support chassis into the longitudinal members of the lifecell. Four fittings adjacent to the upper intermediate ring of the lifecell accept bayonet-type pins on the package to locate the unit laterally and react side loads.

Two reinforced areas of the shell just above the vee-band coupling are provided for the location of 12 electrical disconnects and 11 pneumatic/hydraulic disconnects. Lines to these components are disconnected from the inside after the lower dome of the lifecell has been removed, permitting the main section of the lifecell to be lifted off of the internal equipment package.

Special Equipment

The OPE special equipment consists of the mass measurement, behavioral panel, and retrieval canister subsystems. These three subsystems are described in the following sections.

Mass measurement. – For monitoring the possible changes in mass of the monkey, a noncontact type device has been designed and pertinent parameters have been estimated. The mass measurement device is based on the change in nuclear radiation transmission as the mass of the monkey changes. The device (which has been given the name of the Nuclear Scale) consists of a radiation source and shutter positioned near the behavioral panel and a radiation detector on the opposite side of the animal cage. Both source and detector are mounted to the outside wall of the animal cage. To reduce the weight of the Nuclear Scale, the radiation detector is restricted in area coverage. This requires that the monkey be in some semi-fixed location inside the cell so as to intercept the radiation emitted by the source. It is convenient to plan on obtaining this position when the monkey is using the exercise bar equipment.

Scintillation detectors of the general type planned have been built by LMSC and successfully used on the Gemini program and to study space radiation, as well as on the Pioneer VI and VII spacecraft. Uncorrected electronics drift in the latter spacecraft amounted to less than five percent.

The technique of radiation attenuation mass measurement itself has been used successfully on numerous materials including propellant quantity gaging, aircraft oil inventory monitoring, and determining the mass of lettuce heads before harvesting.

Description of subsystem: The subsystem concept is shown in Fig. 79. Preliminary design is as follows: The radiation source is cobalt-57. This isotope emits 122 kev gamma rays, has a half life of 267 days (in one year the output declines by a factor of 2.6), and is commercially available at a price of about $25 per millicurie. The source is 1/4 in. wide by 8 in. long and is shielded with a tungsten shield of cylindrical cross-section with an outside diameter of 0.37 in. The shutter on the source is spring-loaded and solenoid-actuated to form a fail-safe shutter mechanism.

The radiation detector consists of an array of thin cesium iodide scintillation crystals each connected by means of a light pipe to a photomultiplier tube and associated electronics. The detector covers an area 13 by 39 in. by means of an array of 27 crystals each of which covers a 4-1/3-in. square. Estimated parameters for the Nuclear Scale are total weight of about 40 lb, volume of about 2800 $in.^3$, and power consumption of about 7 w for 1 hour of measurement.

Accuracy and radiation dose: To determine anticipated sensitivity of the Nuclear Scale, it was necessary to determine areal density (weight per unit area) of the various parts of the monkey's body. The reference OPE monkey weighs 12.75 lb and has a total projected frontal area of 208 $in.^2$. Weight distribution data for man were adopted by making the assumption that the relative areal density for man applies to the monkey. Man's weight is distributed as follows: head, 6.9; trunk, 46.1; arms, 12.5; and, legs, 34.5. Man's projected frontal area was assumed to be: head, 8.9; trunk, 29.2; arms, 23.2; and, legs, 38.6. For the reference monkey, the projected frontal area distribution is: head, 11.1; trunk, 37.9; arms, 27.1; and, legs, 23.9. Manipulating these three sets of data gives the following weight distribution for the monkey: head, 8.6; trunk. 56.3; arms, 14.6; and, legs, 20.3.

Calculations for a 10 millicurie source show that the central detector will accumulate 23,600 counts per second with the reference weight monkey in place. A ten-second counting period allows the accumulation of 236,000 total counts, for which the standard deviation due to the random nature of radioactivity decay is 0.2 percent. For a 1 percent decrease in weight, the anticipated count during the same period is increased by 0.63 percent. At the level of 3 percent weight decrease, the signal changes by 2 percent. For purposes of preliminary estimates, the accuracy appears to be some place between 1 and 3 percent. As the counting period is short, three or four repetitions could be obtained within a minute, and the results averaged to give more confidence in numerical values.

Detector electronics drifts have been compensated for using a built-in calibration source check count immediately before or after a mass measurement. By using the array of individual detectors shown, failure of a few detectors is not crucial. Because the detectors occupy a surface area that overlaps the "shadow" cast by the monkey during measurement, some position changes during the measurement can be tolerated.

The source is shielded so that the dose rate to the monkey is about 10^{-3} millirad/hr with the shutter closed. For a 12-second measurement, monkey skin dose per measurement is about 0.005 millirad. In the highly unlikely event that the shutter fails in the open position, the average dose rate to the animal would be only about 1.7 millirad per hour. These radiation doses are insignificant. Because of the use of a shielded metallic source outside of the animal cell there is no credible method for radioactive contamination of the experimental environment. The permissible concentration for cobalt-57 is 100,000 times higher than the permissible level for plutonium-238. No major flight safety problems are anticipated.

<u>Behavioral task subsystem</u>: – The behavioral task subsystem is designed to obtain behavioral data on the two primate subjects, under orbital flight conditions, by analyzing their responses to a psychomotor program. A block diagram for this subsystem is shown in Fig. 83. The work panel, together with switch, lever and cue light details, is shown in Fig. 84 (CP102074). The location of the work panel and food and water dispensers within the cage is shown in Fig. 86.

The general procedure for the behavioral regimen is called multiple schedule (MS). In the MS different stimuli signal the occasion of different behaviors. This particular MS has five task components and a rest period. Four components with visual stimuli control food or water reward responses; one of these tasks incorporates an auditory discriminative stimulus, and one task requires avoidance (AVD) and noxious stimulus. The food or water reward tasks; timing (TIM), vigilance (VIG),

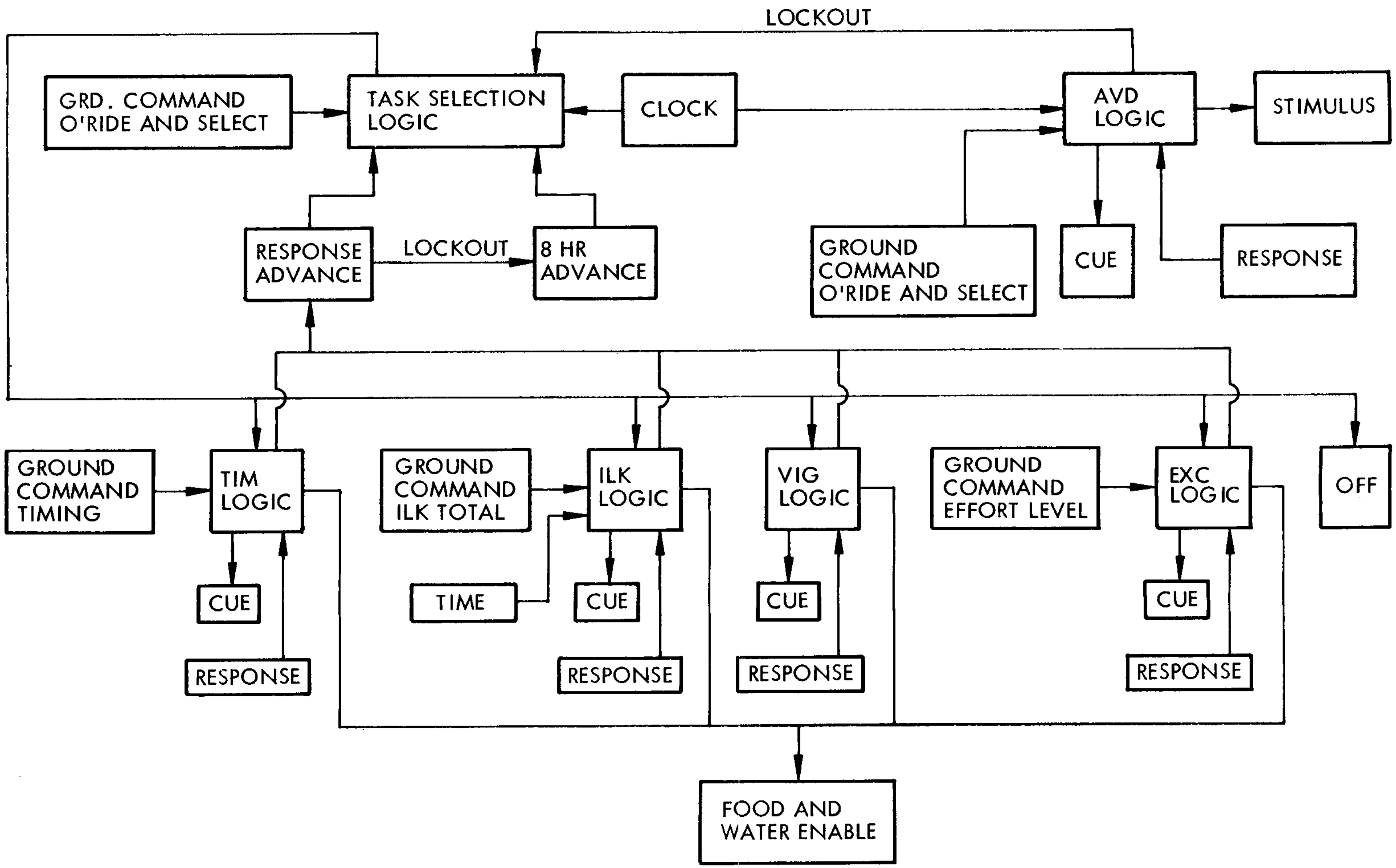


Fig. 83 Behavioral Subsystem Block Diagram

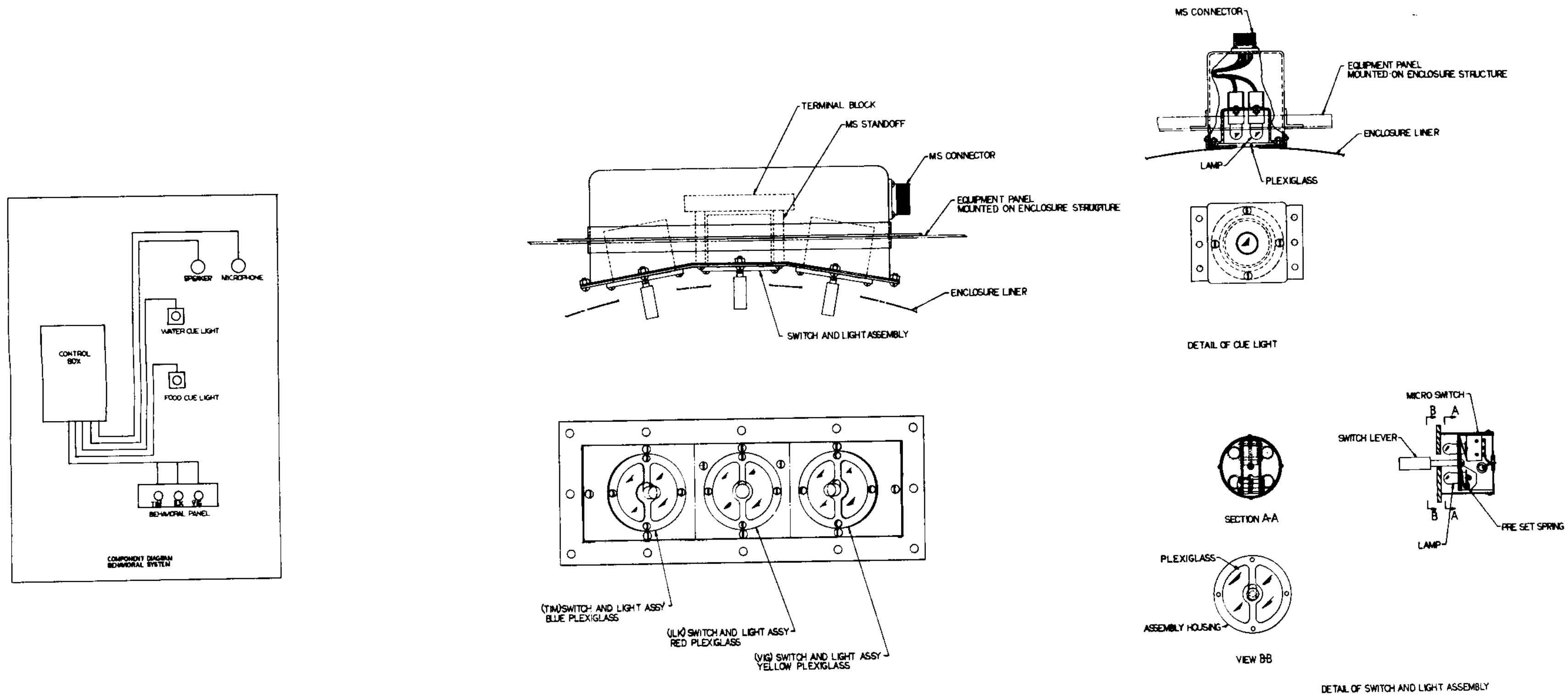


Fig. 84 Behavorial Panel Layout

interlock (ILK), and exercise (EXC), alternate with each other in a scrambled sequence with an occasional off-period. The AVD task is programmed to occur only four times a day, as near as possible to 15 minutes, 5, 9, and 13 hours after the start of the 14-hour lights-on period. A correct response will cause the program to advance to the next task.

The TIM task is initiated by the illumination of a blue light around the TIM lever on the work panel. The correct behavior is to space responses at least a specific minimum number of seconds apart. The first response after the TIM stimulus (cue) light comes on starts the timing interval. The second response, if it exceeds the time interval, enables the food and water dispensers, illuminates their cue lights, and terminates the TIM light.

If the second response does not exceed the minimum time, the timing task is reset and restarts. A response is defined as an upward actuation of the lever of at least 10 milliseconds duration. The handle must be released for a second response to occur. The time interval between responses is adjustable upon ground command to 5, 15, 30, 35, or 60 seconds.

The VIG task is initiated by the illumination of a yellow light around the VIG lever on the work panel. The correct behavior is to respond when the frequency of an intermittant auditory click increases slightly. The reference click rate is approximately six per second and the test click rate is approximately eight per second.

The reference click is on for 2 seconds and off for 2 seconds. The test click remains on for 2 seconds. The test click rate begins a variable period of time after the VIG task begins. The following sequence of time periods is contemplated: 28, 4, 16, 12, 40, 8, 24, 8, 8, 4, 20, 24, 12, 8, 20, 4, 44, 12, 8 and repeat. If a response is made during the reference click, it has no effect. Similarly, if a response is not made during the test click, this has no effect. The reference click rate returns again after another period averaging 16 seconds. This task continues until a correct response terminates it or after approximately 40 minutes has elapsed.

The ILK task is initiated by the illumination of a red light around the ILK lever on the work panel. In this task the number of responses (lever actuations) and elapsed time are interlocked so that the primate may respond rapidly a relatively large number of times and receive the reward sooner or he may respond slowly a relatively few number of times and receive the reward later. For instance, any combination of responses and seconds totaling 50 may enable the reward, such as 40 responses and

10 seconds, or 5 responses and 45 seconds, except that a final lever actuation (response) is required to illuminate the food and water cue lights and enable the lip switches. The combined number of responses and seconds required to enable the food and water reward and terminate the ILK task are adjustable by ground command from 1 to 100.

The EXC task is initiated by the illumination of the green cue light adjacent to the exercise bar. This task consists of the primate alternately pushing up and pulling down on the exercise bar through an excursion of approximately 16 inches; this figure is established by the physical dimensions of the life cell in its present configuration and can be increased at some cost in structural and/or mechanical complexity. A system consisting of a hydraulic cylinder actuated by the primate, forcing the hydraulic fluid through a variable orifice will be used to provide the variable force component required as shown in Fig. 85 (Animal Exercise Unit). The animal cage floor is designed to provide toe holds for the animal, thus providing the necessary reaction element in the EXC task. Correct response to this task consists of actuating the bar a given number of times against a given resistance. The number of bar actuations will be adjustable upon ground command to the following numbers: 1, 2, 4 and 8. The resistive force will be adjustable, upon ground command to the following levels: 1/8, 1/4, 1/2, 3/4, 1, 1-1/2, 2, 3, 4 and 5 lb. Prior to the performance of the EXC task the radiation mass measurement system will be energized to "weigh" the primate.

The AVD task is initiated by the sounding of a 3000 Hz tone for a period of 10 seconds, which serves as a warning cue that a noxious stimulus is going to occur. The noxious stimulus is the high air velocity occurring when the 2000 cfm fan is energized. The primate can avoid this stimulus by retiring into the retrieval capsule during the fan operation.

The rest period is one minute in duration and, in addition to occurring in the random program as previously described, also occurs coincident with the AVD task.

The behavioral task subsystem is designed so that it may be started and stopped by its own timing system or by ground command. If the primate completes a task but fails to take a reward, the reward is made available for 10 minutes. Tasks are locked out during this period. If the reward is not taken in 10 minutes, it is lost and the program is resumed. The reward is either food or water but not both for a single correct response; although both are offered, the acceptance of either one deactivates the other until another correct response starts the cycle over again.

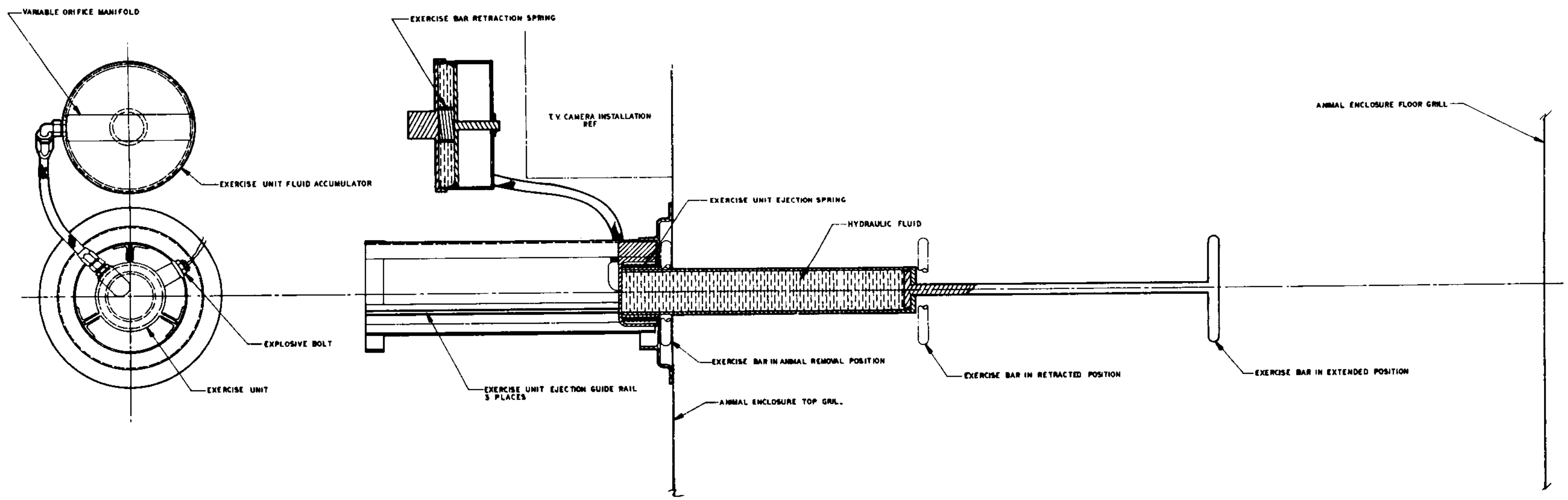


Fig. 85 Excercise Unit Layout

Retrieval Canister. – The retrieval subsystem as shown diagramatically in Fig. 86 has been designed for use on the OPE to provide (1) a container in which a 6.0 kg Rhesus monkey may be transferred from the OPE spacecraft to the Command Module by a retrieving astronaut for subsequent return to earth, (2) passive metabolic support for the primate for a period of up to 48 hours, and (3) a means of receiving and recording metabolic data from the live primate's implanted transmitters during retrieval and reentry.

While recovery of both primates alive is the major objective of the OPE program, the possibility of one or both animals expiring does exist. In this unfortunate event, it is of crucial importance to the program that the animal(s) be recovered in a condition suitable for histopathological analysis following return to earth. In order that this goal may be met, a means of preserving the animal has been included as a part of the retrieval canister subsystem design.

The retrieval canister is a formed and machined aluminum structure, cylindrical in shape. It is 14 inches in outside diameter, except for the clamping ring flange which is approximately 17.5 inches in diameter, and has an overall length of 30 inches. An electronic compartment is provided at the upper end of the canister and is formed by an inverted cylindrical pan which is bolted to a flange located on the animal compartment domed end closure. This electronic compartment has a volume of 407 $in.^3$ and contains a timer, radio receiver, amplifier, recorder, and batteries to power this equipment to receive and record metabolic data when the animal is in the canister. The timer will be started when the live animal is inserted in the canister prior to recovery and will energize the radio receiver, amplifier, and recorder for a period of one minute. The timer will cycle this equipment off for a period of 11 minutes and on for 1 minute, giving 5 minutes of recorded data per hour. A timer override switch is provided on the top of the electronic compartment that will be actuated by the astronauts in the command module at the beginning of reentry for continuous recording during this phase.

The animal compartment is cylindrical in shape with an inside diameter of 13 inches and a length of approximately 23 inches as shown on Fig. 81 (CP102071). The upper end is closed by a removable dome bulkhead which also forms the lower side of the electronic compartment. The lower end is open to the top of the animal cage and provides the entry to the retrieval canister. A machined, flanged ring with an O-ring gasket gland is located on the outside of the animal compartment and forms one-half of a V-band joint. The other half of this joint is located on the upper end closure of the lifecell over each animal cage and provides the means by which the retrieval canisters are located and mounted to the spacecraft. A V-band clamp ring with a toggle release mechanism is used to secure each canister to the lifecell and forms a leak-proof joint that may be easily and quickly opened by an astronaut when the canisters are removed for retrieval.

The upper domed end closure of the animal compartment is joined to the cylindrical body of the canister by a V-band O-ring joint and clamp ring and provides the

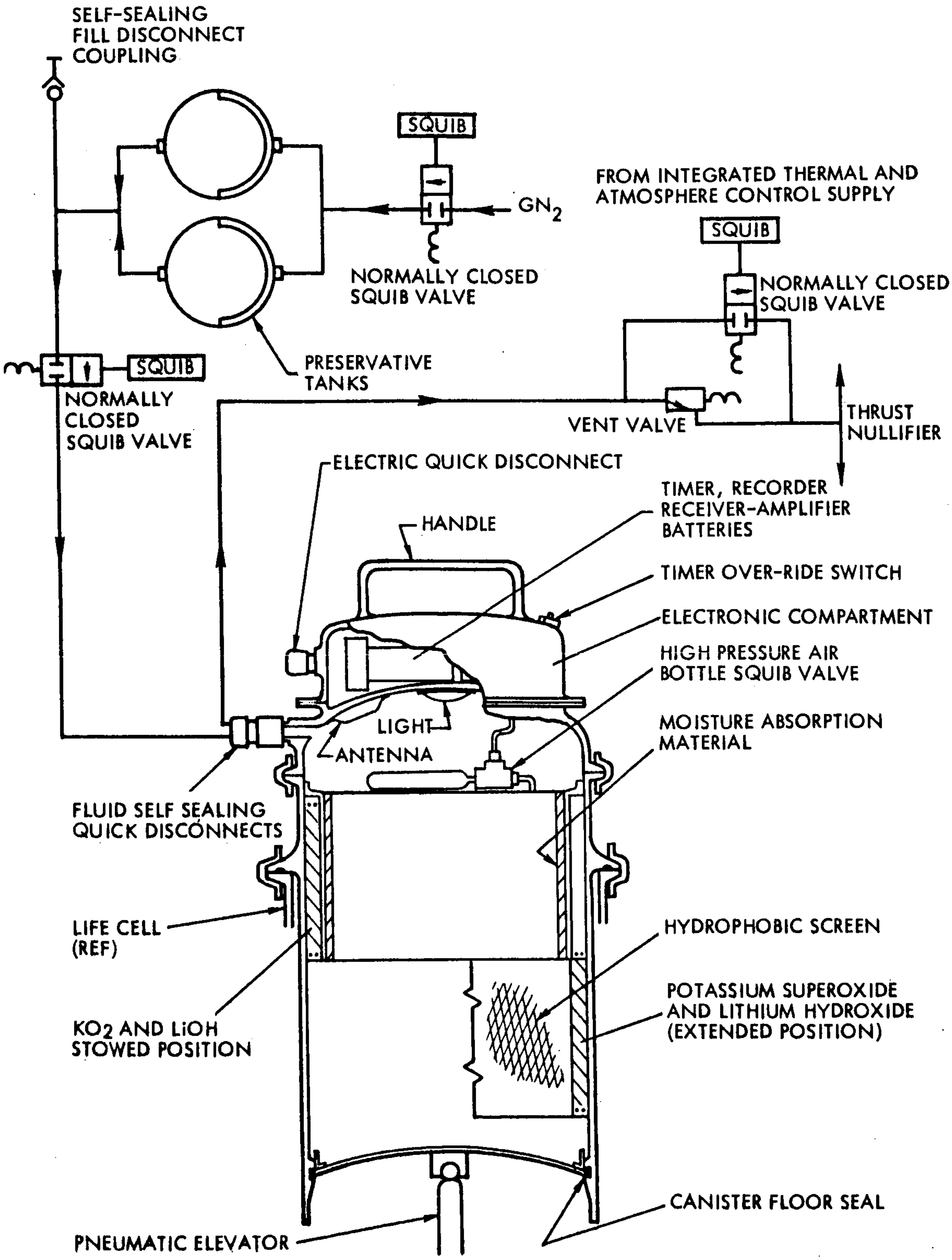


Fig. 86 Retrieval Canister, Life Support and Preservation Subsystem

means by which the animals are inserted into the lifecell prior to launch and removed from the canister after retrieval.

Located within the animal compartment are a light, an antenna to pick up the radiated signals from the animal implants, metabolic support chemicals, moisture absorbant, a compressed air bottle and a squib valve. The light is provided to signal the animal, in conjunction with an audio cue, to enter the canister prior to activation of the purge fan used to clean the animal cage. It will also provide illumination for the animal during the retrieval period. A mixture of potassium superoxide (KO_2) and lithium hydroxide (LiOH) are used for atmosphere control during the retrieval period. These chemicals are contained within a cylindrical piston-type holder that is sealed from the lifecell environment. The seal is achieved by the use of O-ring gaskets on the inner and outer diameter of the piston at both the upper and lower ends as shown on Fig. 80 (CP102071).

Upon initiation of the retrieval operation, the squib valve is fired which releases air from the high pressure bottle which in turn forces the chemical holder piston downward and exposes the KO_2 and LiOH to the canister atmosphere. A hydrophobic screen is used to cover the chemicals and prevent the direct contact of urine droplets with the KO_2 which reacts rapidly with water. A moisture absorbant material is installed on the inner surface of the piston holder and will entrap any free urine droplets that contact the material.

The floor of the retrieval canister is located on the floor of the animal cage and serves as a part of the cage floor until the retrieval operation is initiated. As soon as the cage wall is in the retrieval position as shown in Fig. 79 (CP102070), a pneumatic telescoping elevator, located under the center of the domed canister floor, is actuated. This elevator raises the floor upward and forces the animal as shown in Fig. 80 (CP102071), into the canister. The floor is inserted into the canister to form the closure at the bottom of the animal compartment and at the upper limit travel of the pneumatic elevator. The edge of the dome shaped floor is forced into an annular seal located in the lower skirt of the canister cylindrical body. This seal is covered by protector ring during the flight to prevent the accumulation of any substance that might impair or prevent a leak-proof seal being formed with the edge of the domed floor. This protecting ring is forced upward by the floor as it is raised to its final position. This protecting ring also covers a spring loaded segmented latching ring which decreases its effective diameter after the floor has passed upward through it and serves to lock the floor in place. The floor is secured to the pneumatic elevator by a spring loaded ball detent which allows the floor to separate from the elevator upon removal of the retrieval canister from the lifecell by the retrieving astronaut.

Atmosphere control is maintained as required by the animal in the sealed retrieval canister by a mixture of potassium superoxide (KO_2) and lithium hydroxide (LiOH) which exposed to the canister internal atmosphere as described earlier. As the animal breathes, the carbon dioxide (CO_2) produced is partially removed by the LiOH, the remainder of the CO_2 is absorbed by the KO_2 with a release of oxygen (O_2) as a product of this reaction. This mixture contains 1.40 lb of KO_2 and 0.36 lb of

LiOH and is sufficient to remove all the CO_2 produced, and sufficient O_2 to sustain the animals life but not overproduce O_2 to the extent of causing a condition of O_2 toxicity. The quantities of KO_2 and LiOH noted above are sufficient to sustain a 6 kg Rhesus monkey for a period of 48 hours.

Preservation of the deceased animal(s) will be accomplished by the use of a preserving fluid which is a mixture of dimelhylsulfoxide and formaldehyde which will be transferred into the canister animal compartment so that the cadaver is completely immersed. The principal of using this solution for preserving intact animals has been investigated at LMSC. The results of these experiments to date indicate a satisfactory degree of preservation.

The fluid preservation part of the retrieval canister subsystem is shown on Fig. 86 and consists of two hydropneumatic accumulator tanks, squib valves, and a vent valve which are connected to the canister by two self-sealing quick disconnect couplings. The preserving fluid is contained in the 2 hydropneumatic accumulator tanks, each having a fluid capacity of 1.0 ft^3 thus providing a total of 2 ft^3 which is slightly less than the volume of the canister animal compartment. When it has been determined that an animal has died, the animal is inserted into the canister as described earlier; however, the KO_2 and LiOH chemical holder piston is not actuated and remains in the sealed position. This is necessary since the preserving fluid reacts with these chemicals. With the deceased animal sealed in the canister, the hydropneumatic tank gas inlet and fluid outlet squib valves are actuated to the open position by ground command and the 35 psi gaseous nitrogen from the lifecell integrated thermal and atmosphere control system supply is admitted to the pneumatic side of the accumulator tanks. The fluid preservative is forced out of the tanks and into the canister and the air contained in the canister is vented overboard, through a vent valve and a thrust nullifier, as it is displaced by the incoming preservative. The vent valve will crack at 20.0 psia and reseat at 19.0 psia after the air and any excess fluid are vented overboard. A normally closed squib valve is placed in parallel with the vent valve and will be actuated by ground command some time prior to retrieval operations. This will allow the excess preserving fluid to vaporize and vent to space, thus reducing the weight of the canister and minimize its effect on the command module during reentry. Additional experimentation is required to determine proper vent rate and time required for this phase of the operation. The retrieving astronaut will disconnect the fluid, vent, and electrical lines and the canister will be completely sealed automatically during the remainder of the retrieval operation.

Data Management

The data management subsystem was established with the requirement that it be compatible with the Apollo Ground System. To attain this objective, the Apollo unified S-band system has been selected where possible. Modification to the S-band system will be required, since all of its capabilities are not required for OPE.

Figure 87 shows a block diagram of the airborne data management subsystem. This system includes many components, some of which have been designed, others that have been designed but require changes, and some that are completely a new design. In the succeeding paragraphs, this block diagram is discussed in general and then each subsystem is discussed in detail.

General description. – The behavioral work panel data are obtained in digital form and recorded on an 8-channel digital recorder. This recording is continuous and will be read out over a station on command. The work panel data are multiplexed in real-time so that data can be obtained real-time when the stored data are being transmitted to ground to prevent loss of any data. A continual time word is sent along with either stored and transmitted data so that time of occurance can be determined.

The behavioral work panel data (both stored and real-time) are connected to the parallel digital inputs of the PCM and timing equipment assembly, and is read out upon command at the predetermined sampling rate.

The signal conditioner assembly receives the analog signals from sensors of different types and processes them. The analog signals into the multiplexer must be 0 to 5 volts. This signal conditioner package will consist of amplifier and bridge circuits to obtain the 0 to 5-volt analog signal.

The signal processor controls all signals that go to and from the transmitter. Its function as a data processing and distribution center provides the necessary interface between RF data and onboard equipment.

The signal processing assembly consists of many relays that are controlled by ground station commands. These commands direct these data into the S-band transceiver where they are PCM upon one of the subcarriers of the S-band system, and are transmitted through the power amplifier to the Diplexer. A ground command selects which antenna will be used for transmission on the basis of received ground station signal strength.

The biodata are collected from each animal by separate implanted trnasmitters. There is a transmitter for each of the following signals: ECG, temperature and respiration. These transmitters are on continuously and use very little power. They can transmit the signal 3 to 5 ft and are detected by an antenna at the top of each cage. The antenna at the top of each cage is cannected to three tuned receivers. These receivers (Fig. 87) amplify the signal, demodulate it and then it is connected to the analog multiplexer inputs.

Biodata will also be commutated so that they may be transmitted over the PCM, or PM subcarrier in the place of audio.

The desired frequency response of the electrocardiogram is 200 Hz. The fastest sampling rate of the multiplexer is 200 samples/sec. To obtain the desired frequency response, each ECG signal will be connected to three separate channels in parallel.

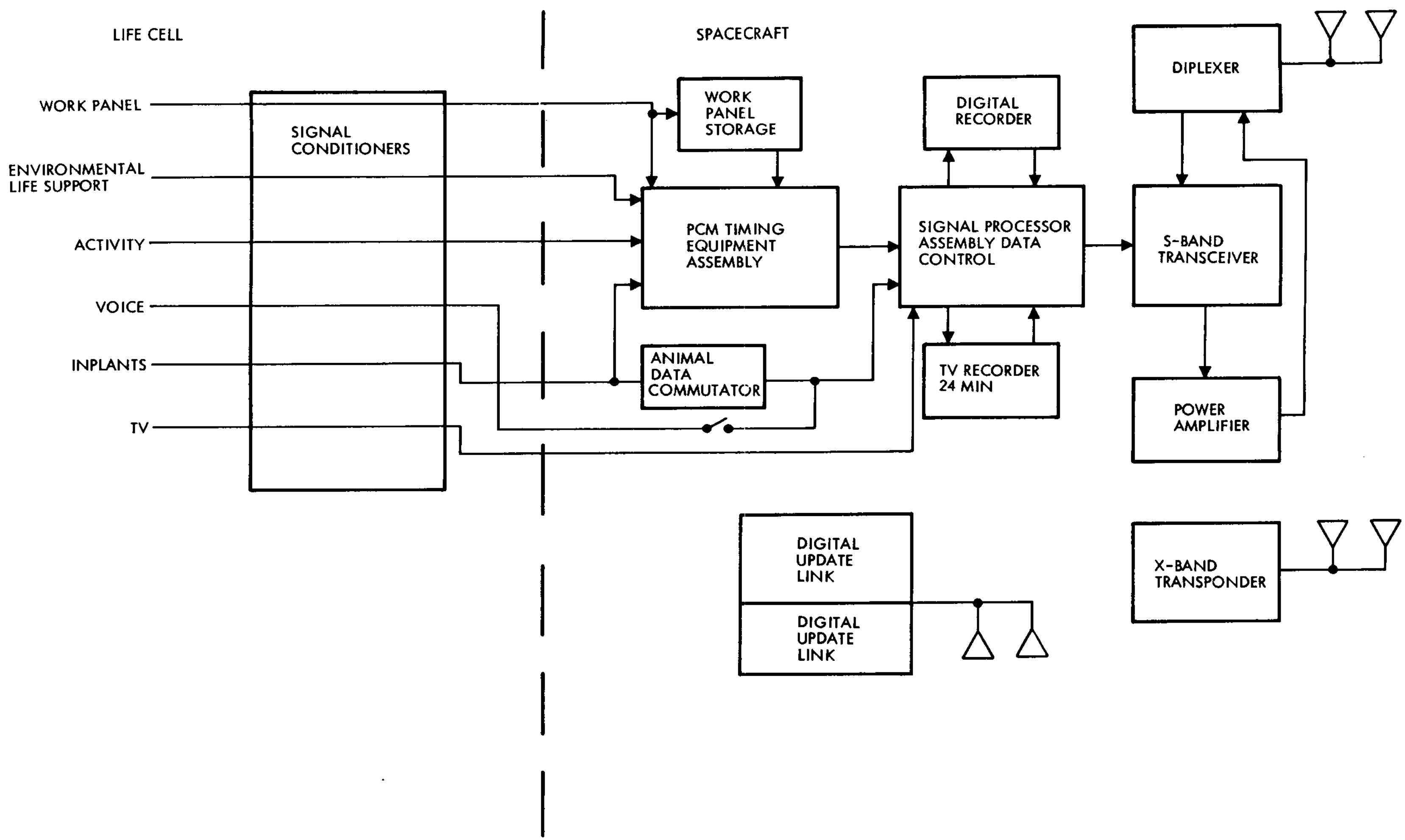


Fig. 87 Airborne Data Management Subsystem

To obtain a frequency response of 200 Hz the sampling rate should be at least 2.5 times the desired frequency response. Three channels will give 600 samples/sec or a frequency response of 240 Hz.

The desired frequency response for the temperature, respiration and animal activity signals for these animals are 5 Hz. These signals can be sampled at a slower rate and will also be connected to the analog inputs of the multiplexer.

The biomedical signals from the demodulator (Fig. 88) are also connected to the animal commutator. A command from the ground can connect these signals to the subcarrier used normally for audio. This command is used to transmit all extended periods of continuous biodata on the PM subcarrier while the stored data are being transmitted on the PCM subcarrier. The signal processor is used to direct biodata by commands to either store the biodata in serial-digital form, or connect it to the S-band system for transmission.

The environmental, life support, and spacecraft data will all be handled in the same form. The data will be detected by sensors and connected into signal conditioners to obtain a 0 to 5-volt signal that can be compatible with the rest of the system.

The audio subsystem is made up of microphones installed in the work panel, and a preamplifier and amplifier that can supply a 0 to 5 volt analog signal to the signal processor and then on to the PM subcarrier. The audio signal is not recorded on the spacecraft. It is read in real-time whenever the PCMTEA and S-band transmitter are operating, with one exception. As stated before, the subcarrier and PM mode used for audio will be used for commutated biodata upon command from the ground. This command will be given when long continuous (5 minutes) biodata are to be transmitted.

The TV subsystem contains four cameras, two for each cage. Each cage has one high-resolution, low-frame-rate camera that is set near the work panel. The other is located at the top of the cage with a movable prism network so that the cage may be scanned if desired. The TV camera at the top of the cage will have a lower resolution than the fixed camera, but will have a faster frame rate so animal movement may be observed. Though there are four cameras for the two cages, only one camera can be operated at a time. Commands from the ground can select the camera for use. Ground commands will also be used to operate the prism so that the camera may scan the cage. The camera receives the image and processes it so that it can be either stored or transmitted in real-time to ground. There will be ground commands and a stored command to record the TV data on the on-board tape recorder. The stored or real-time TV data will be connected to the signal processor assembly for selection as to its being stored or real-time transmitted. The TV data is frequency modulated on the carrier frequency of the S-band system, and is transmitted to the ground station.

Three types of radiation dosimeters are to be used. One is a photomultiplier scintillator counter that uses a digital storage register to count the number of rads/min. The other is one developed by NASA for a back-up system. The third is a system of passive film packs mounted in each of the retrieval capsules which will be recovered with the animal.

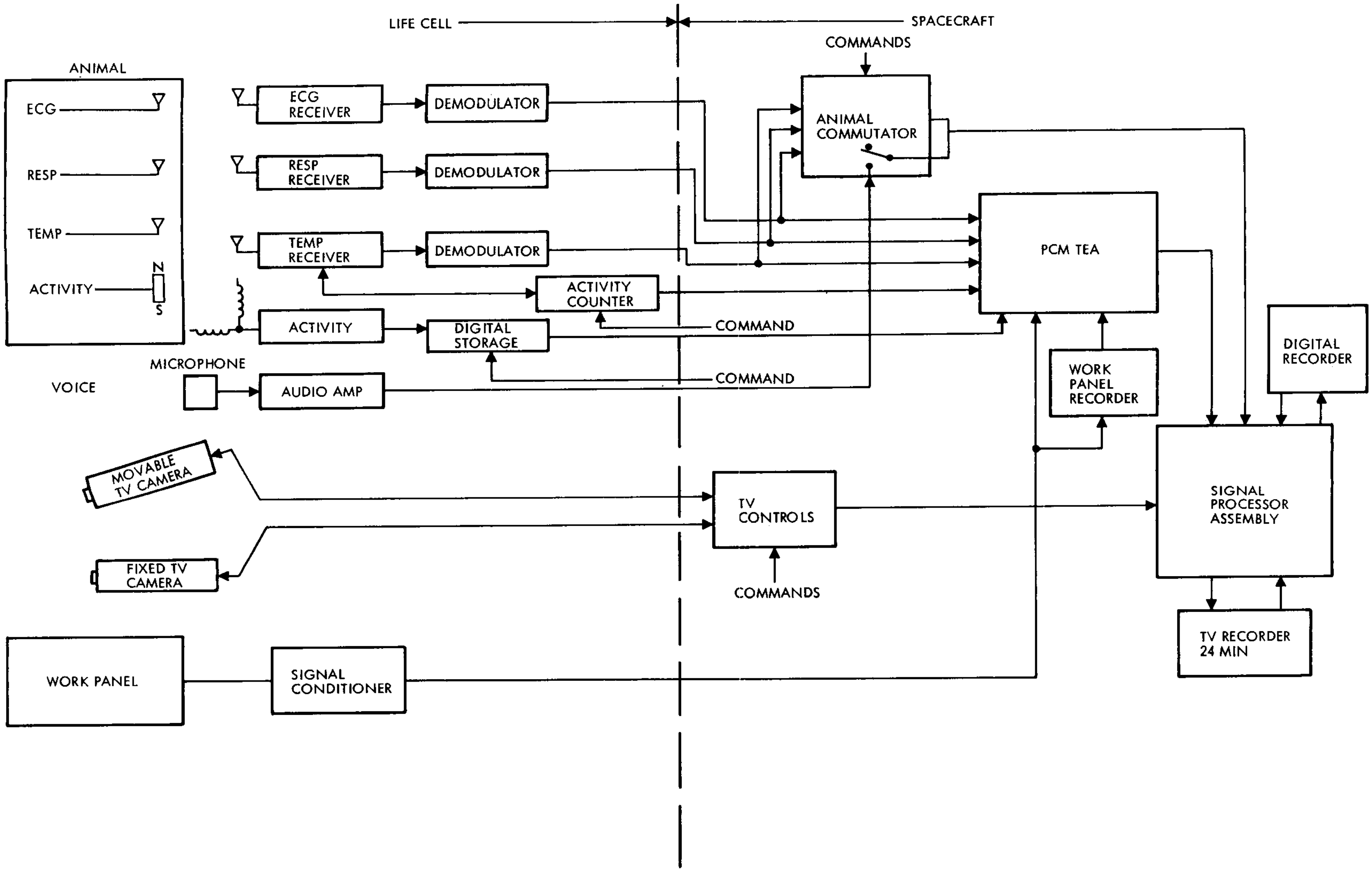


Fig. 88 Animal Data Flow Diagram

Two types of animal activity counters are provided. The primary system is designed to use the signal received from the transmitter indicating the animal's temperature. Changes in received signal strength are detected and these variations counted and stored for a period of 12 minutes. They are then dumped, reset, and made ready to count the activity for the next 12 minutes. The other system operates similarly but receives its signal from a different source. The animal has an implanted magnet which works in conjunction with pickup coils at the top of each cage and at the side at a 90 deg angle from each other. These two pickup coils (per cage) detect any animal movement, store it for 12 minutes and then dump the information to the recorder and restore.

The function of the update link is to receive, verify, and distribute digital updating information sent to the spacecraft by the MSFN at various times throughout the mission, to update or change the status of operational systems. The update link consists of detecting and decoding circuitry, a buffer storage unit output relay driver, and a power supply. This system utilizes a two-tone format in which each tone frequency represents either a binary "1" or "0."

The unified S-band equipment (USBE) consists of a receiver, transmitter, and power supply. This equipment is used with a power amplifier to obtain the desired transmitting power at the antennas.

The USBE system can transmit data on three different modes: FM, PM, and PCM. This system can also be used for tracking the spacecraft.

The pulse code modulation timing equipment assembly (PCMTEA) converts all data inputs from the various sources throughout the spacecraft into one serial digital output signal. Input signals to the PCM telemetry equipment are high-level analog (0 – 5 volt) and parallel digital words. There are 277 high-level analog multiplexed signals and 26 parallel digital inputs that may be used.

The timing equipment provides precision square wave timing pulses of several frequencies to time correlate all spacecraft time-sensitive functions. It also generates and stores the real-time date, hour, minute, and second time from launch in binary coded decimal format for transmission and display to MSFN.

The X-band transponder will be used only during rendezvous between the spacecraft and the Apollo craft. The transponder in the S-band system can be used to track the spacecraft from the ground but more accurate tracking is required for rendezvous operations.

The antenna will be flush-mounted with the spacecraft skin. There will be a center-fed cavity providing a coverage large enough so that only two S-band and two X-band antennas will assure ground contract for any position of the spacecraft.

Work Behavioral Panel. – The behavioral panel data are all in digital form. The timing clock signal originates in the timing equipment assembly so that all data will be in synchronization with the rest of the spacecraft. The work panel signals are stored continuously on a tape recorder and then read out over the desired station. During readout of stored data, real-time work panel data are also transmitted along with the stored data. This eliminates any loss of data.

There are eight digital words of eight bits each to define all psychomotor functions for both animals. These data are recorded and then played back at 30 times the recorded rate. With a playback limit of 200 words per sec , a recording rate of five words per second has been selected. Since there are eight words per group, the maximum allowable recording rate is 0.625 word groups per second on an eight-track digital recorder.

An analysis of the maximum function rate of the two combined psychomotor panels shows the possibility of a maximum of three word groups during one scan period of 1.6 sec. Therefore, there are three intermediate holding registers to defer word groups occurring within the reading period. They will be read out in sequence if more than one register is full. Since each word group is time-tagged there can be no ambiguity of data.

The behavior panel is further described in a preceding section of this report.

Biodata. – The implanted biodata subsystem consists of sensors, signal conditioners, transmitter and power supplies. Each animal has three separate subsystems to detect, condition and transmit the following signals: electrocardiogram, respiration, and temperature.

The signals transmitted from the three implanted transmitters within each animal are detected by a single antenna at the top of each cage. Connected to this antenna are three tuned receivers. Each receiver is tuned to a frequency of one of the transmitters within the animal. This signal is then demodulated and amplified to the desired 0 – 5 volt analog signal. The total biodata subsystem includes six different frequency transmitters with separate power suppliers, two receiver antennas, and six separately tuned receivers and demodulator.

Environmental life support and spacecraft data. – The environmental life support and spacecraft data take the form of pressure and temperature measurements as shown in Fig. 89. Three types of pressure are measured: differential, gage, and absolute. Standard practices are used for these measurements as flight qualified equipment can easily be obtained.

Two-gas atmosphere sensor: The two-gas sensor is a small, low power instrument for monitoring the principal contaminants and atmosphere gases. This unit utilizes the optical absorption technique to accurately measure the concentrations of the principal gases. It measures the partial pressures of O_2, CO_2, and H_2O, using

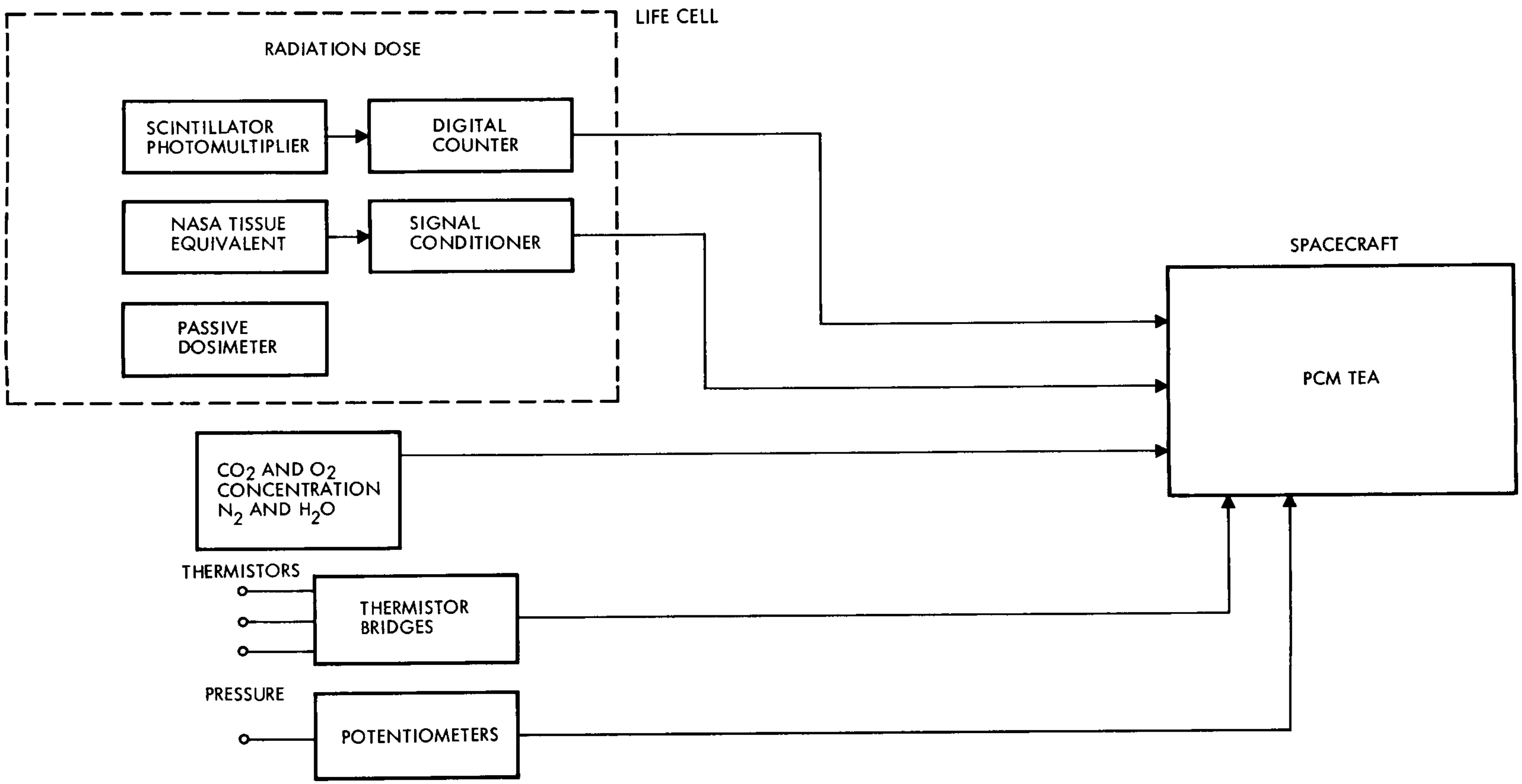


Fig. 89 Lifecell Environmental Data Flow Diagram

ultraviolet and infrared channels. With these data with the total pressure measurement, the partial pressure of N_2 can be obtained.

Two of the above systems are used. One system is used for life support systems control and the other for measurement. To provide a higher reliability, additional circuitry is included so that if either of the systems fail, a command from the ground can be executed to make the other perform the functions of measuring and controlling the atmosphere.

Temperature: The temperature requirements for this program can be carried out by the use of thermistors. A thermistor is a metallic oxide resistor that has a negative temperature coefficient. The resistance changes greatly with small temperature changes. Thermistors may be used to measure temperatures from -450°F to 1,200°F, making them suitable for almost any temperature measurement. The thermistor is normally used in a bridge circuit so that a 0 to 5-volt signal can be obtained over the temperature range of concern. Temperature bridges can be designed which require no amplification.

Audio data. — Audio signals from the primate are obtained by the use of a microphone placed within each of the work panels. The output of these microphones is summed, amplified, and connected to the signal processor. Upon commands from the ground, this signal is connected to one of the subcarriers and is pulse modulated (PM) and transmitted to the MSFN.

Provisions have not been made to record any audio data on-board the spacecraft. Audio data will be transmitted to MSFN when other data are being transmitted with one exception (Fig. 90). One of the requirements is for 5 min of biodata six times/day. This requirement would load the USBE to capacity, and the stored digital data could not be read out over the station. To eliminate this problem, the biodata are commutated with a separate small commutator and are transmitted over the PM subcarrier by a command from ground. This makes it possible to transmit TV, stored data, and real-time biodata for a full station contact. The use of a small commutator will save power and will make it possible to collect all the required data.

TV data. — The video system will utilize two cameras per lifecell. One high resolution unit is mounted horizontally in the top section of the cell looking at a gimballed first surface mirror which is positioned to view down into the cell through a glass or plastic port. This port is necessary to protect the camera-mirror system from waste material and is provided with a cleaning system to keep the face plate free of waste deposits. This camera can be used to observe the animal and the cage itself from the top of the cage. This camera will have a viewing angle of 36 deg and will be in focus for 0.5 to 2 ft.

The other camera is a low resolution system positioned in or near the psychomotor panel to observe the animal while it is in this area. This camera will look directly through another port which will also be kept clean and free of waste deposits. This camera will have a view angle of 18 deg and be fixed.

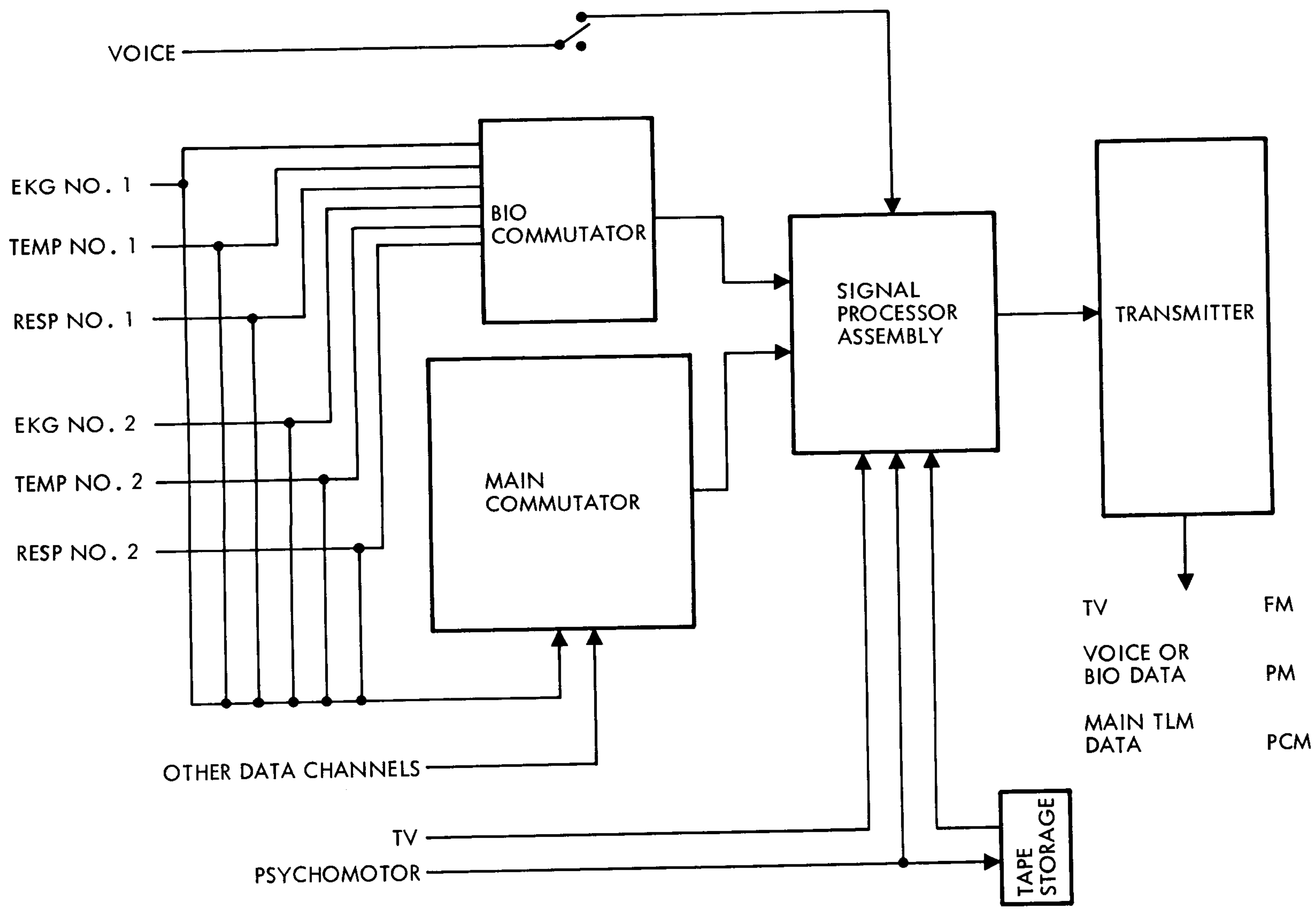


Fig. 90 Data Commutation Diagram

The controls for the TV cameras are located outside the lifecell. They must be designed to operate in a space environment. This requires a new design as flight control equipment could not be found to supply the required functions. Ground commands are used to select the desired camera. If either of the high resolution cameras located at the top of each cage is selected for viewing, a pulse command will be used to rotate the optics for selection of the desired viewing portion of the cage. Only one camera can be used at a time. The signal from the selected camera can be either stored on a magnetic tape recorder or transmitted in real-time over the station. This mode of operation will be selected by ground or stored commands. The T.V. data wil be transmitted by frequency modulation of the carrier frequency. The carrier frequency can be frequency modulated from 0.400 to 1.5 MHz.

The high resolution cameras will operate at a 0.625 frame rate per sec and the low resolution system will operate at 10 frames per sec. These frame rates have been established to be compatible with the ground station requirements.

High resolution TV will be sampled for the first minute of each daylight hour, one minute just after the start of the 10-hour dark period, one minute midway in the dark period, and one minute just before the start of the next light period. Low resolution data will consist of four five-minute samples a day and may be either recorded or played out in real-time. Available excess recording capability of the TV recorder may be used as required by the Principal Investigator for either high- or low-resolution data. Any additional time available over the tracking stations may be used for real-time readout.

Radiation data. — The main unit for measurement of radiation dosage is a plastic scintillator counter. The least count is set at one millirad. When 1 millirad of dosage is encountered, it dumps a count into a digital storage register consisting of 16 binary stages of nondestructive readout. This system stores a continuous accumulation of dosage and reads out the total. The two 16 bit words are connected into the PCMTEA digital inputs for storage or transmission. This unit will be calibrated in the lab and will not require recalibration in flight.

A back-up system is the NASA equivalent tissue system. This system works on the same principle as the above system. The passive units are film pack dosimeters placed inside the retrieval capsule. They cannot be read out but will be retrieved with the animal.

Animal activity data. — Animal activity data are discussed in the following paragraphs.

Implanted magnet: A convenient means of detecting motion of the animal is the detecting of changes in a magnetic field that is emitted from the animal. This can be accomplished by implanting a small permanent magnet within the animal. Teflon-coated magnents suitable for implanting are readily available, and the ability to use such an implant has been well demonstrated. Detection of this magnetic field can be accomplished by means of several high impedance coils located about the cage (Fig. 91). Any motion of the animal will induce a voltage in the coil that can be detected by simple circuitry.

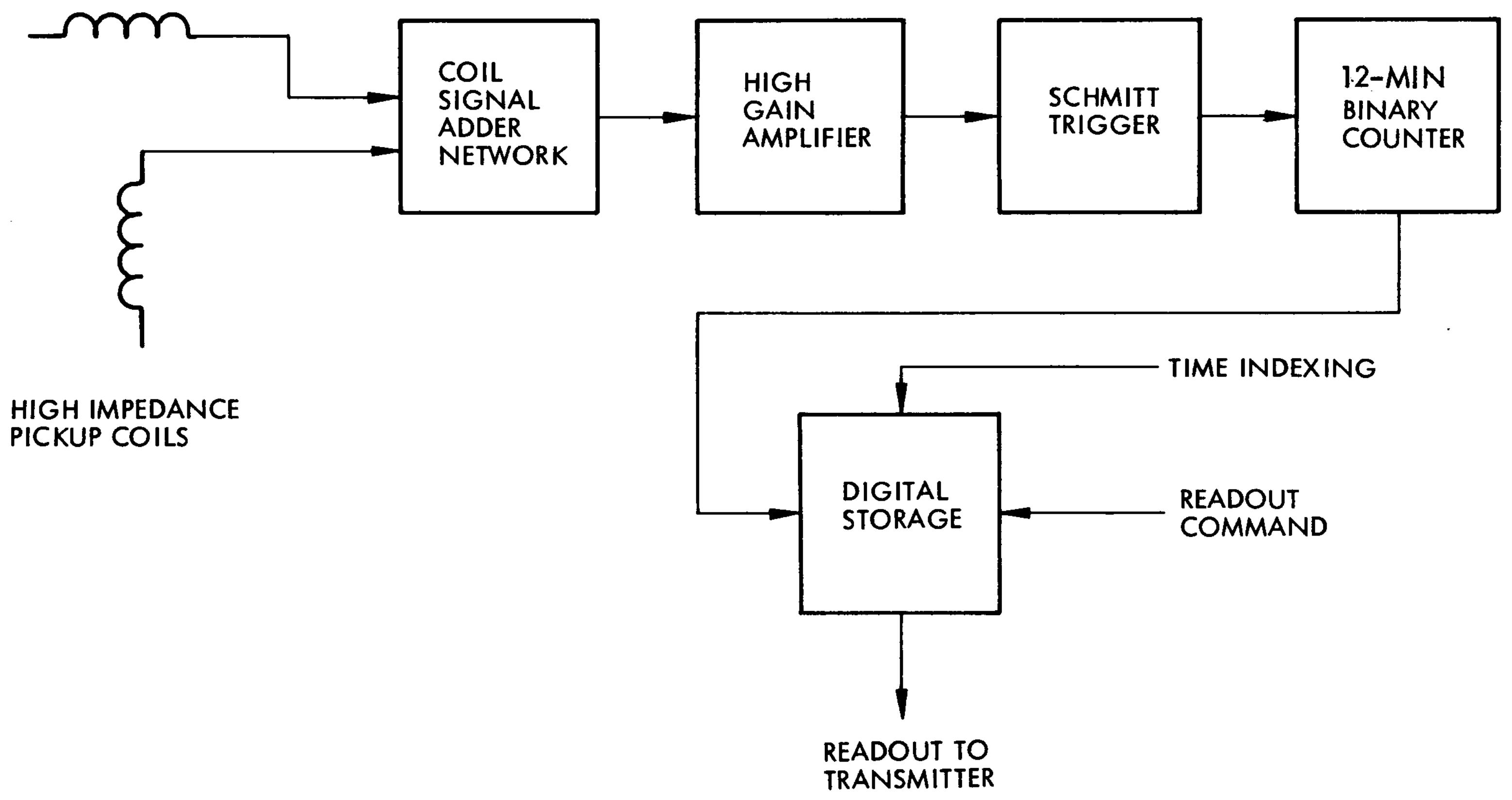


Fig. 91 Backup Implanted Magnet Activity Sensor

The primary advantage of such a system is the high sensitivity and very low power consumption. The basic output is an analog signal roughly proportional to the velocity of the animals' motion and the distance from a pickup coil. The analog signal can be conveniently converted into digital information for counting and data storage purposes. Two high impedance pickup coils are located in each lifecell: one at the top of the cage and one 90 deg from it on the side of the cage. These pickup coils have access to the cage through an open port. The animals' activity is accumulated and stored for a 12-minute period, dumped into the recorder and the next 12-minute period will be accumulated. Twelve minute periods of animal activity will be obtained and a running table will not be kept.

Indwelling telemetry signal strength: If the indwelling instrumentation is operated continuously, it is possible to determine animal motion by observing the field strength of the received signal. Any motion of the animal will change the level of the signal at the receiver. By observing the AGC level of the receiver, it is possible to determine the received signal strength. The primary advantage of this system is that it requires no additional implants or external equipment other than an output amplifier from the receiver AGC. The basic output is an analog signal roughly proportional to the animal's movement. This analog signal can be conveniently converted into digital information for counting and later storage purposes (Fig. 92). This counter operates on the same principle as the above one for 12-minute periods, then dumps, resets, and starts counting again.

Data management equipment. – The equipment discussed below will be located outside the lifecell and exposed to the space environment.

PCMTEA. – The pulse code modulation and timing equipment assembly consists of a rugged lower-power, precision-pulse-code-modulated (PCM) telemeter and precision timing equipment (TE) combination. Both the PCM and TEA are packaged in a single unit (Fig. 93). This unit has been designed for the Apollo program. Minor design changes will be made to this system for the OPE program. Not all of the parallel digital inputs are required, nor are the serial start and stop words required.

PCM telemetry equipment: The function of the pulse code modulation (PCM) telemetry equipment is to convert all of the data inputs from various sources in the spacecraft into one serial digital output signal. This single output signal is routed to the premodulation processor for transmission to the MSFN or to the data storage equipment for storage. Input signals to the PCM telemetry equipment are of two general types – high level analog and digital. All data inputs are routed through the data distribution panel and then to the PCM telemetry equipment.

During mission phases when transmission to earth is possible, the PCM telemetry output is supplied to the premodulation processor, where it is prepared for transmission via the USBE. When transmission to earth is not possible, the signal is stored in the digital storage equipment for subsequent transmission.

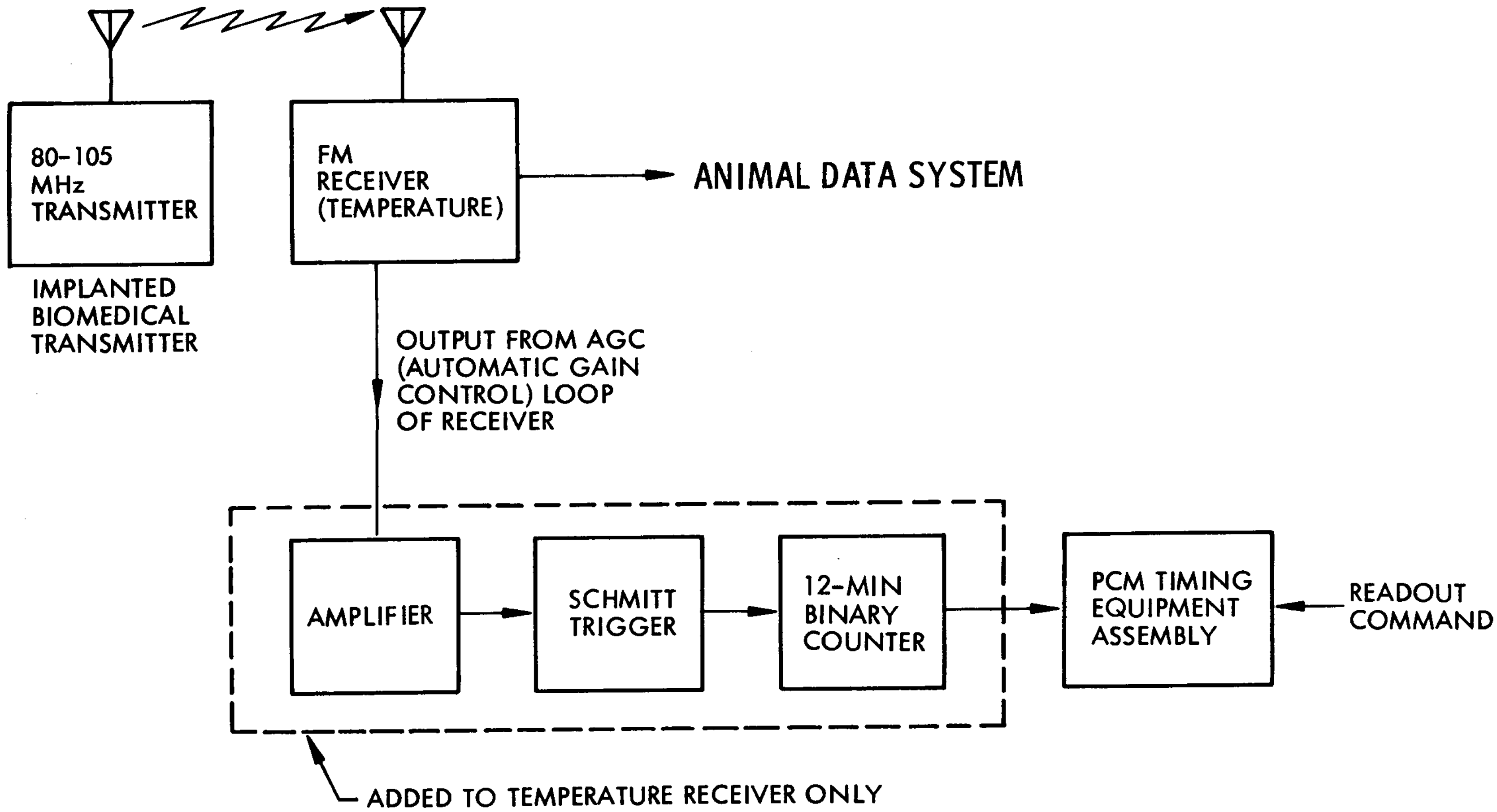


Fig. 92 Near Field Receiver Signal Strength Activity Counter

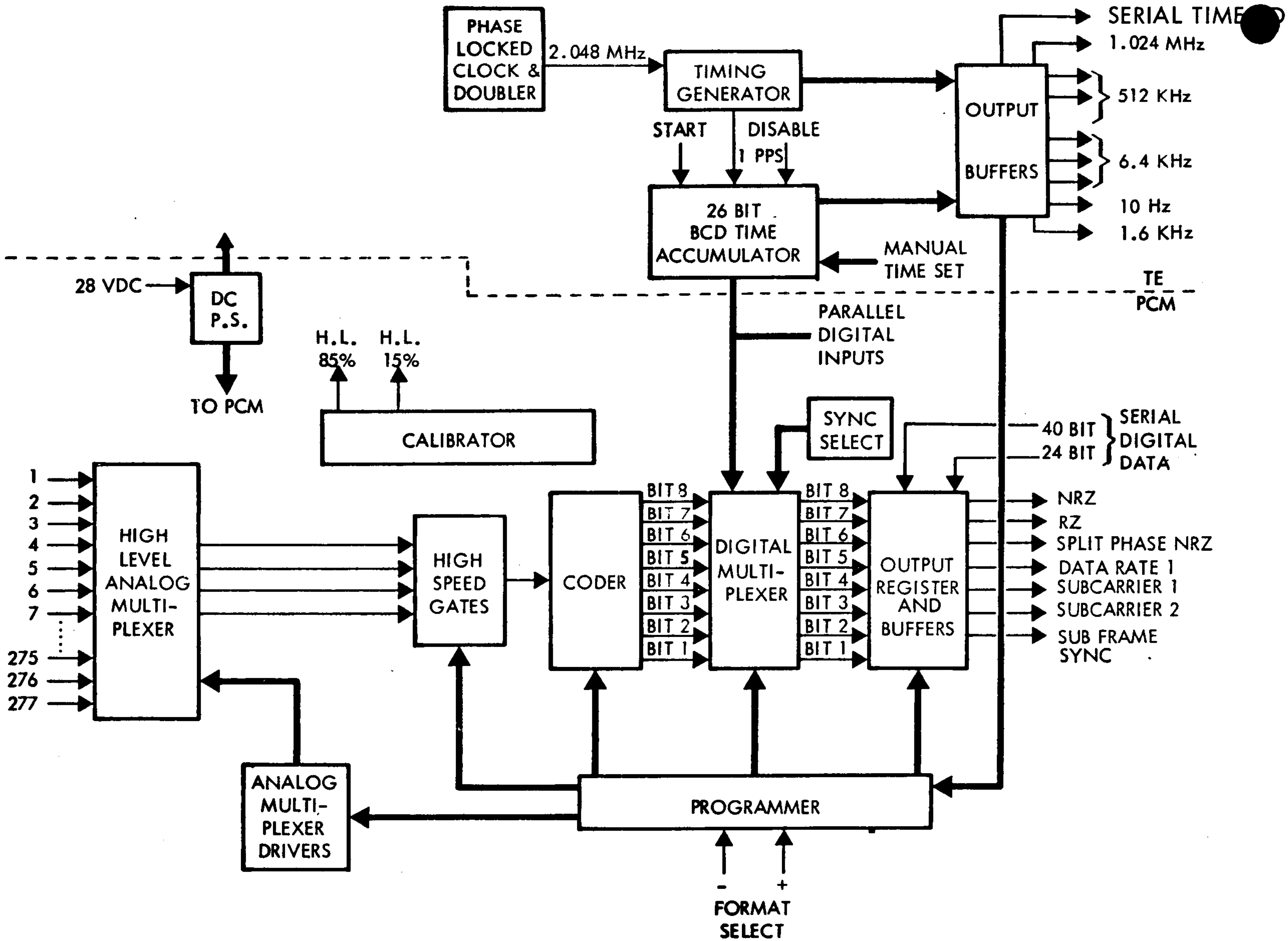


Fig. 93 PCMTEA Block Diagram

There are 277 channels of 0 to 5 volts of analog data as follows:

Channels	Sampled (cps)
7	200
22	100
8	50
45	10
195	1

There are 528 channels of parallel digital inputs available for external use as follows:

Channels	Sampled (cps)
16	200
32	100
40	50
8	10
432	1

TEA equipment: The timing equipment generates an internal highly stable 1.024 MHz signal for operation of the PCMTEA. It furnishes 1.024 MHz, 512 kHz, 6.4 kHz, 1.6 kHz, and 0.10 Hz output timing signals that are used throughout the spacecraft. The TEA also furnishes inputs for synchronization inputs as well as the time of day and the day of the month of the PCM portion of the equipment.

Signal processor. – The signal processor is a new equipment design. It consists of relays, controls, and logic circuits to direct data to the desired locations. Within this equipment, data are directed to the appropriate storage equipment or to the desired mode of modulation within the S-band equipment. All commands from the digital up-date link will be directed through this equipment. There will also be commands originated within this unit to operate the fixed time interval commands that are required on the spacecraft. These commands are required to store data when the spacecraft is not in contact with the ground station. It will use the square wave signal from the timing equipment assembly and then count down and select the desired time interval for the required commands.

Unified S-band equipment (USBE). – The USBE (Fig. 94) consists of a receiver, transmitter and power supply contained in a single electronics package which can be used during earth orbital missions to provide a means for transmitting real-time or recorded analog TLM data, biomedical data, and TV data to the MSFN when the spacecraft is within range of a station equipped for S-band operations. It can be used for PCM TLM data, FM TV data, PM voice and animal data, tracking and ranging, and reception of up-date commands. This unit was designed for the Apollo system and can be used for OPE with minor changes.

Spacecraft tracking can be accomplished at MSFN stations equipped with S-band equipment by measuring the change in frequency of the carrier received from the

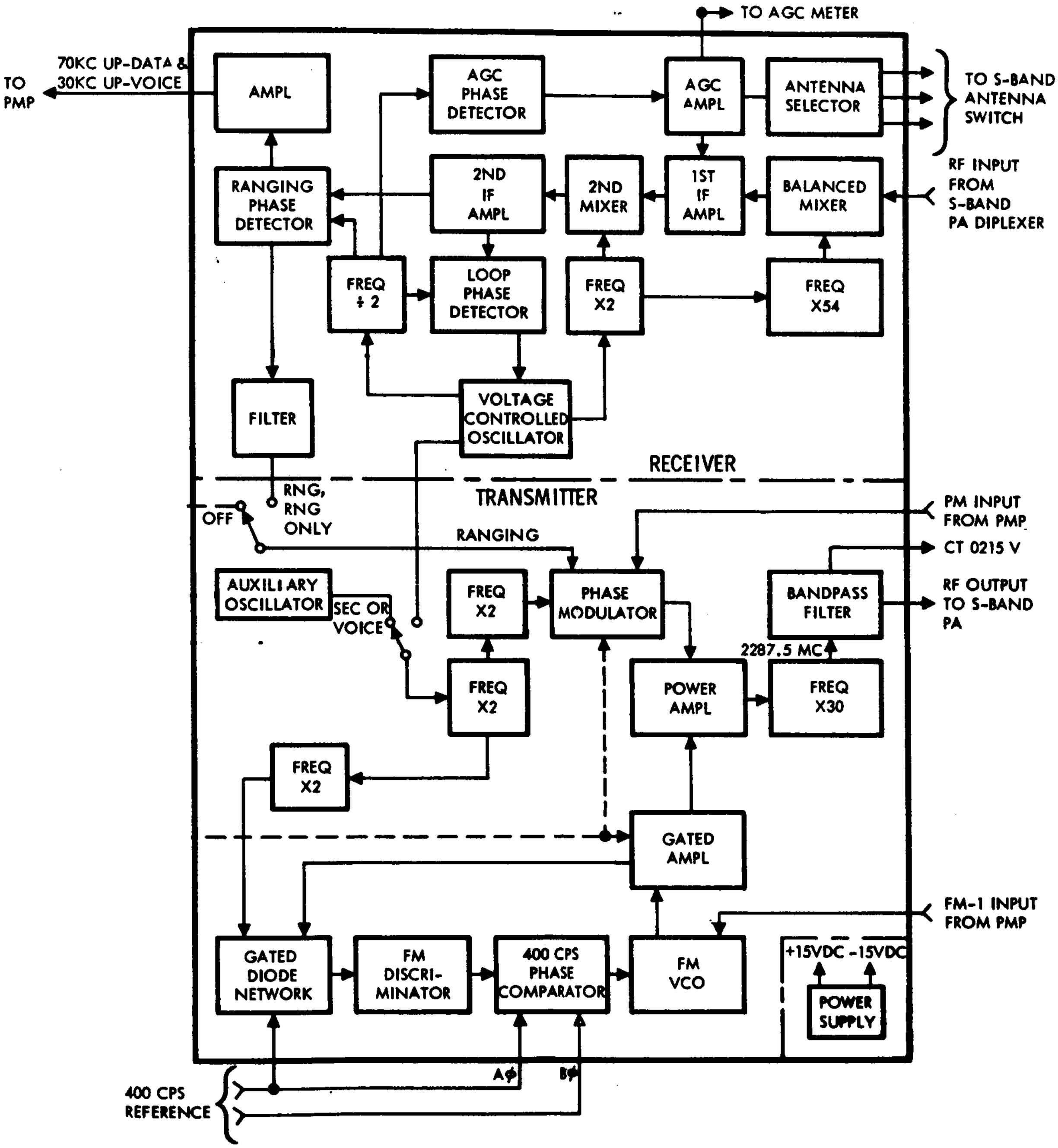


Fig. 94 Unified S-Band Equipment

spacecraft. This is called single or one-way doppler technique. This method is not as accurate as is desirable, however, because of the relative instability of the local oscillator in the USBE. The primary technique which is employed is the two-way or double-doppler method. In this case, a stable carrier of known frequency is transmitted to the spacecraft where it is received by the phase-locked receiver, multiplied by a known ratio, and then retransmitted to the MSFN for comparison. Because of this capability, the USBE is also referred to as the S-band transponder.

For determining spacecraft ranges, the MSFN phase-modulates the transmitted carrier with a pseudorandom noise (PRN) binary ranging code. This code is detected by the USBE receiver and used to phase modulate the carrier transmitted to the MSFN. The MSFN receives the carrier and measures the amount of time delay between transmission of the code and reception of the same code, thereby obtaining an accurate measurement of range. Once established, this range can be continually updated by the double-doppler measurement. The MSFN can also transmit update commands and voice signals to the S/C USBE by means of two subcarriers, 70 kHz for update, and 30 kHz for upvoice.

The USBE receiver is a narrow band, double-conversion superheterodyne, automatic, phase-tracking receiver that accepts a 2106.4 MHz, phase-modulated RF signal transmitted by the MSFN. This signal contains the update subcarriers and the pseudorandom noise (PRN) code for turn-around ranging and is supplied to the receiver via the diplexer in the S-band power amplifier equipment. After passing through two mixers and two IF amplifiers, the resulting signal is presented to two separate circuits: the loop phase detector, and the ranging phase detector. In the ranging phase detector, the 9.531 MHz IF is detected and the 70 kHz updata and 30 kHz upvoice subcarriers are extracted, amplified, and routed to the update discriminators in the PMP equipment. The ranging signal is also derived, filtered, and routed to the USBE transmitter as a modulating signal input to the phase modulator.

S-band power amplifier equipment. – The S-band power amplifier (PA) equipment (Fig 95) is used to amplify the RF output from the USBE transmitter when additional signal strength is required for adequate reception for the S-band signal by the MSFN. It consists of a diplexer, a traveling wave tube for amplification, power supplies, and the necessary switching relays and control circuitry. The S-band power amplifier is contained in a single electronic package which was designed for the Apollo system. This unit can be used with the USBE without any changes.

All received and transmitted S-band signals pass through the S-band PA diplexer. The 2106.4 MHz S-band carrier received by the S/C enters the S-band PA diplexer from the S-band antenna equipment. The diplexer passes the signal straight through to the USBE receiver. The 2287.5 MHz output signal from the USBE transmitter enters the S-band PA where it is either bypassed directly to the diplexer and out to the S-band antenna equipment, or amplified first and then fed to the diplexer. There are three power amplifier modes of operation: bypass, low power, and high power.

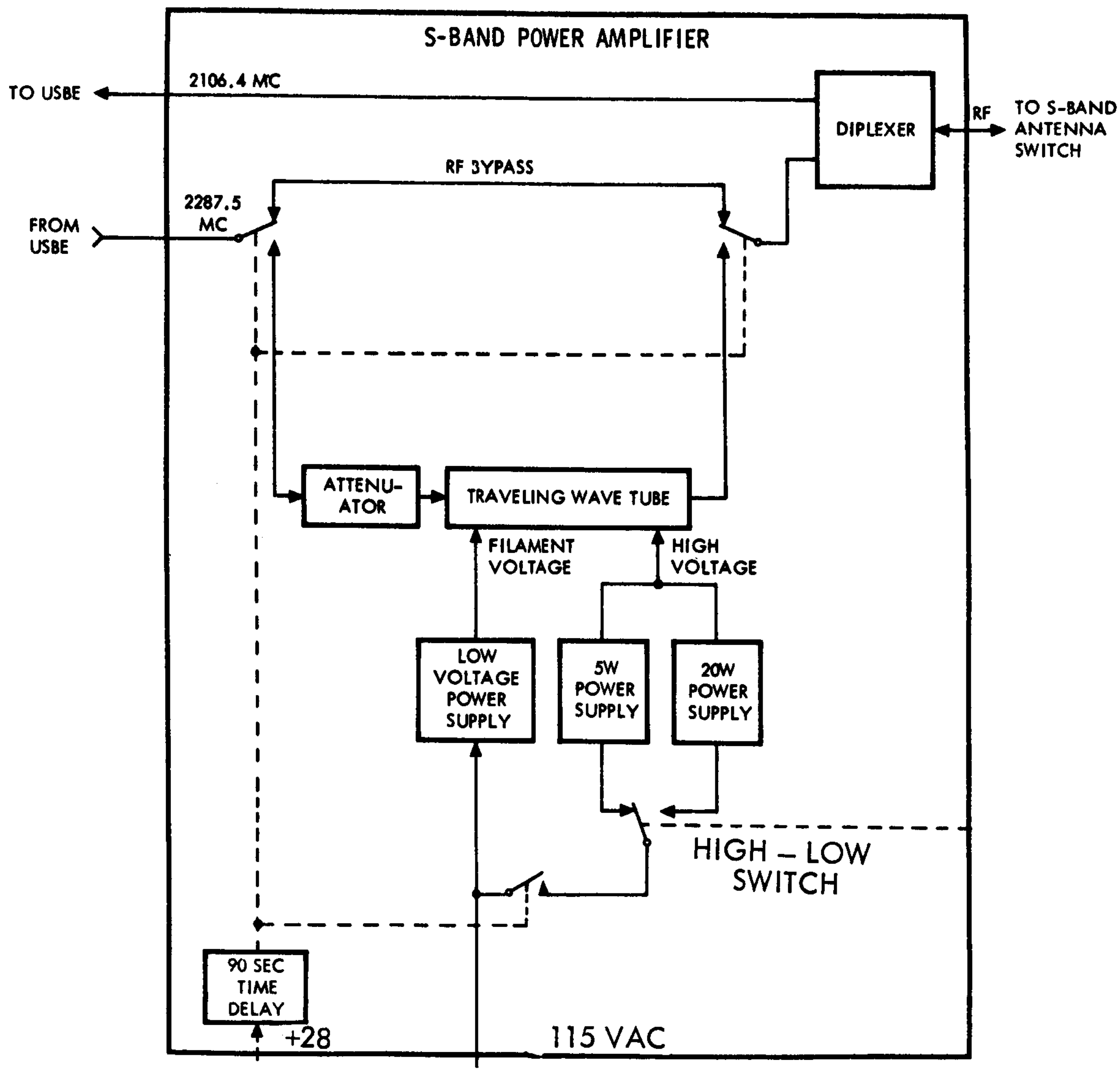


Fig. 95 S-Band Power Amplifier Equipment

S-band antenna equipment. – The antenna for the USBE (Fig. 96) is a round cavity with a center electrode feed that may be flush mounted to the side of the spacecraft. The spacecraft shall be sun-stabilized which sets the antenna requirements at two antennas for the S-band system. Antenna switching and complex signal strength detection will not be required for antenna selection for transmission. The S-band power amplifier can supply enough power that a magic tee may be used to equally divide the power between the two antennas. This system will ensure reception of the signal even if the S/C goes out of control.

An approximate pattern for the S-band antenna is shown in Fig. 97. This pattern may change slightly with exact frequency design. With the use of two antennas, there shall be three strips across the earth that cannot receive the signal. The width of these strips will depend on the final antenna design. Calculations show that 16 watts of transmitted power are required but the designed system will supply 20 watts at the selected antenna. The additional power will make the no-signal strips narrower.

The antenna used for S-band transmission shall be selected by ground commands depending on the received signal strength at the ground station. This command will be completed through the VHF update link receiver. Twenty watts of power are required because of the bandwidth of 1 MHz required for TV transmissions. In the final design, the location of the antennas on the spacecraft will also consider where these no-signal strips shall fall upon the earth with respect to receiving stations.

Updata link equipment (UHF). – The function of the updata link equipment (UDL) is to receive, verify, and distribute digital updating information sent to the spacecraft by the MSFN at various times throughout the mission to update or change the status of operational systems. The UDL (Fig. 98) consists of a UHF FM receiver, a transistor mode switch, detecting and decoding circuitry, a buffer storage unit, output pulses, output relay and a power supply.

The S-band update mode can be selected when the USBE is in operation. In this mode, the UHF receiver in the UDL is deactivated and its function is replaced by the USBE receiver and the PMP. Update information can be transmitted within the S-band signals. When this signal is received by the USBE receiver, the 70 kHz subcarrier containing the updata information is extracted and sent to the updata discriminator in the PMP. The resulting composite audio frequency signal is routed to the sub-bit detector in the UDL.

The UHF receiver may be used when the USBE is not in operation. Digital data is sometimes required for guidance and timing equipment may require correction due to power failure or drift in on-board equipment. The real-time commands may be used to perform a multitude of on-off controls and alarm functions.

There are 64 real-time commands any one of which may be selected at a time. The UDL supplies a pulse, 30 millisecond wide and this pulse can be used to drive relays. On-board, there is a relay box containing 32 relays. This gives 32 relay commands and 32 pulse commands that could also be made into relay commands if

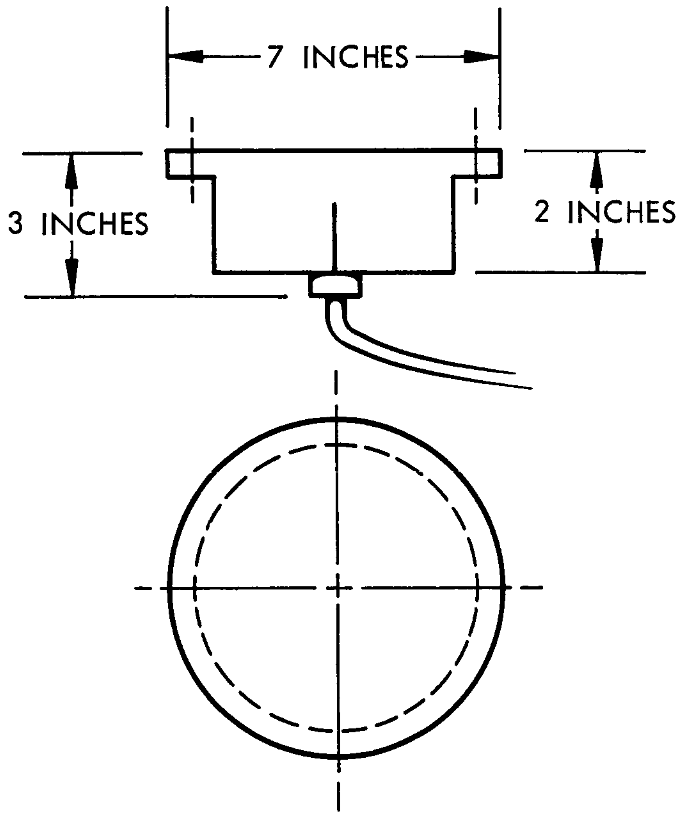


Fig. 96 S-Band Antenna

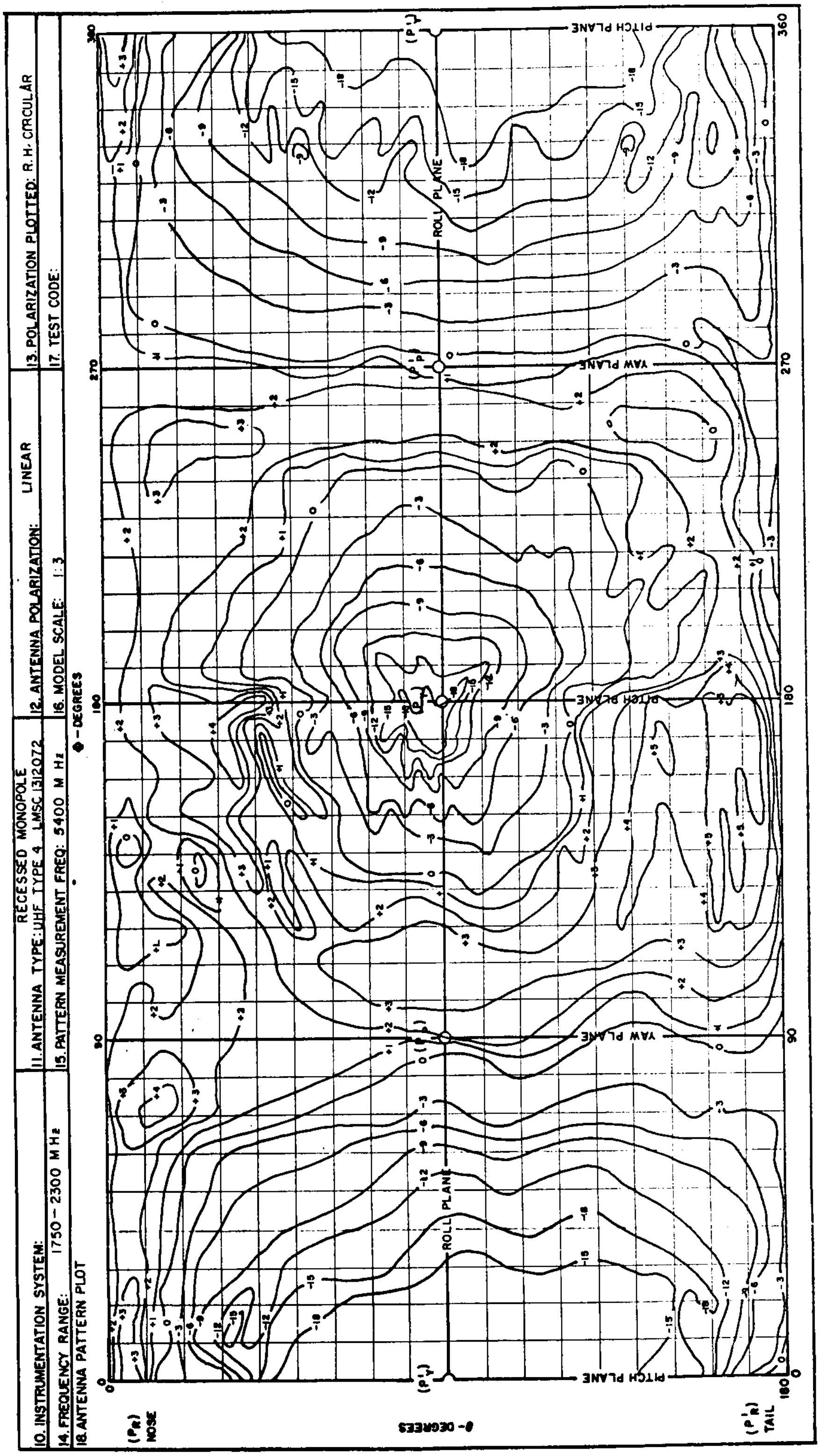


Fig. 97 Antenna Pattern

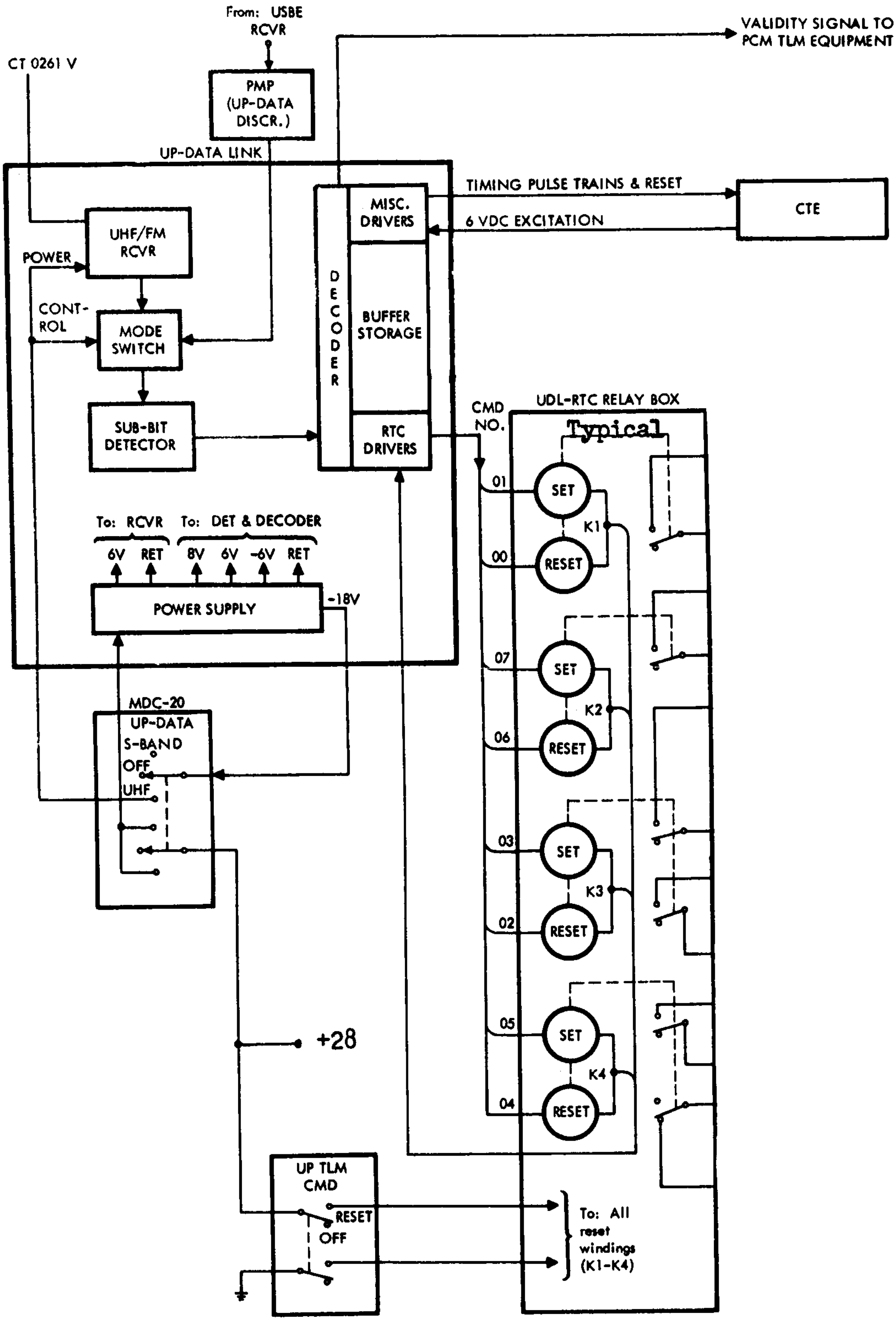


Fig. 98 Up-Data Link Equipment

necessary. This unit has been designed for the Apollo program and can be used for the OPE program with small changes.

The above system will be supplied in duplicate. Two of these systems will be used on the OPE program. One will be used as the primary system with a backup system. Most commands will have to be redundant using the backup system. Each system shall have a common UHF antenna with a ground command to select the receiver and command system. This selection can be accomplished by either using the USBE or the UHF receiver that was last used.

Data storage equipment. – The data storage equipment shall include three separate tape recorders to save power and to increase reliability. One large recorder would require excessive power and larger storage time than smaller ones. Also, the psychomotor recorder will operate continuously either in the record or reproduce mode. Though this recorder does operate continuously, it has a low bit/sec rate and can be a small, slow speed, low power consumption unit, whereas the Video recorder requires a high-frequency, fast speed and high power consumption recorder.

The PCM TLM data are obtained from the PCM TLM data equipment as a single serial pulse train at a bit rate of 51.2 kbps. These data will be recorded on a tape recorder that is capable of 16,384 bits/sec. This will be accomplished by using four data tracks simultaneously and using control circuits to direct one fourth of the data on each channel. A Lockheed Type IX tape recorder has been selected. It is a PCM recorder that operates at 12.75 ips on recorder, and can record continuously for 20 minutes. The playback time is 5 min. This unit has a start time in the record mode of 2 sec and in the read mode it is 4 sec. Its volume is 321 in.^{3} and weighs 11 lb. It requires 15 watts of max. power.

The command which turns on the PCM TLM unit will also turn on this recorder, and a timer will turn it off in 10 sec. The data requirement of 120 samples/day for 10 sec is the same as 10 sec sample every 12 min. These data could be stored on 3 min of recording tape for each orbit. This would allow 5 sec for the tape to come up to speed, 10 sec of data, and 5 sec to shut it off. These data can then be read out over one station each orbit. Additional storage is required if the rate sensors turn on the system when the animal biomedical parameters fall out of limits.

The tape recorder selected for the recording of psychomotor data is a Lockheed Model 546. It has already had six months of continuous orbit life. It can record eight channels on 254 feet of 1/4-in. tape, and has a total recording time of 113 min. This time will cover any one orbit but must be readout at least once on each orbit. This unit will run continuously and will read out on command. It will record and reproduce data from the psychomotor panel in parallel and form eight-bit words. While the tape recorder is supplying data to the signal processor assembly, the new data are not lost because the current data are also being commutated by the PCM timing assembly and being transmitted to the MSFN.

The video tape recorder will have a 1 MHz bandwidth necessary to record both the high and low resolution video data. This recorder will be designed and built by Lockheed using a 1.5 MHz carrier. This unit will have 10 tracks that can record 2.4 minutes/track. The recorder will operate at 100 ips and will use 1/2-in. tape. The required recording time is 24 minutes and this is accomplished by recording on the first track in one direction for 2.4 minutes then reversing the direction of the tape while changing to the second track. This can be done until the 10 tracks are filled. A small amount of data (about 6 sec) will be lost in the changing of tracks.

The total amount of recording time available will vary according to the number of times the recorder must be started and stopped as a 3 second run-up and 3 second coast-down time is required each time the unit is used. It will operate with a signal to noise ratio of better than 23 db, consume approximately 20 watts of power at 28 volts, and physically measure about 8 × 10 × 6 inches maximum.

X-band transponder. – The X-band transponder (Fig. 99) is used only for rendezvous and when the animals are to be retrieved. The S-band system can be used for tracking the spacecraft during the flight within 30 ft but closer tracking is required when rendezvous is required. The X-band system is a typical radar system that can obtain very close tracking or location. It shall be turned on by a command from the ground when closer location is required. The X-band antennas will be located near the S-band antennas, and will have the same basic design but will be slightly smaller.

Equipment provisions. – Provisions have been made in a number of fields to detect troubles before, or as soon as, they arise. If troubles are detected soon enough, corrective measures may be taken to solve the problem before it can cause additional difficulties.

The provisional equipment for the biomedical monitoring will include a system to detect biomedical limits of the physiological signals. When these pre-selected limits are exceeded, a command will be generated to operate the required equipment to obtain real-time, or stored data, of the critical data during the time that the physiological signals are in the out of limits periods. The signal from this system shall turn on certain equipment, but provisions shall be made that a ground command can over-ride this on-board generated command.

The life support system must be instrumented and controlled for O_2 and CO_2 partial pressure. This is a very important measurement, and to increase its reliability, a separate system has been incorporated to instrument, and another to control, these two gasses. Provisional equipment has been designed into the system so that if either of these two systems fails, the other units will take over the responsibility of measuring and control of the life support environment.

Rate detector. – A system shall be incorporated to detect heart rate, respiration rate and temperature. These three analog signals shall have limits attached to each one. If either the upper limits or lower limits of these signals are exceeded, a command shall be generated to turn on the tape recorder and record these critical data (Fig. 100).

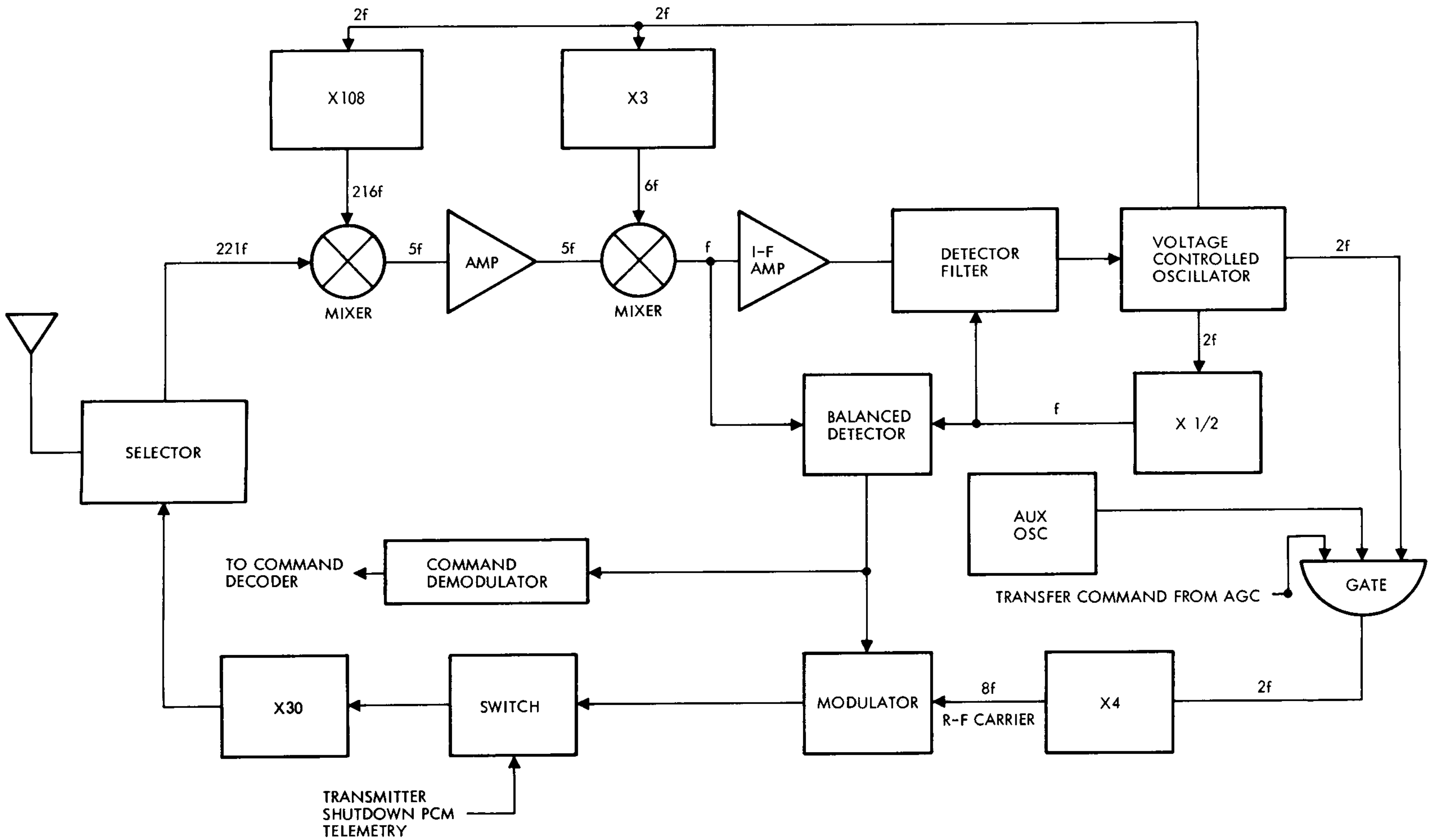


Fig. 99 Block Diagram X-Band Transponder

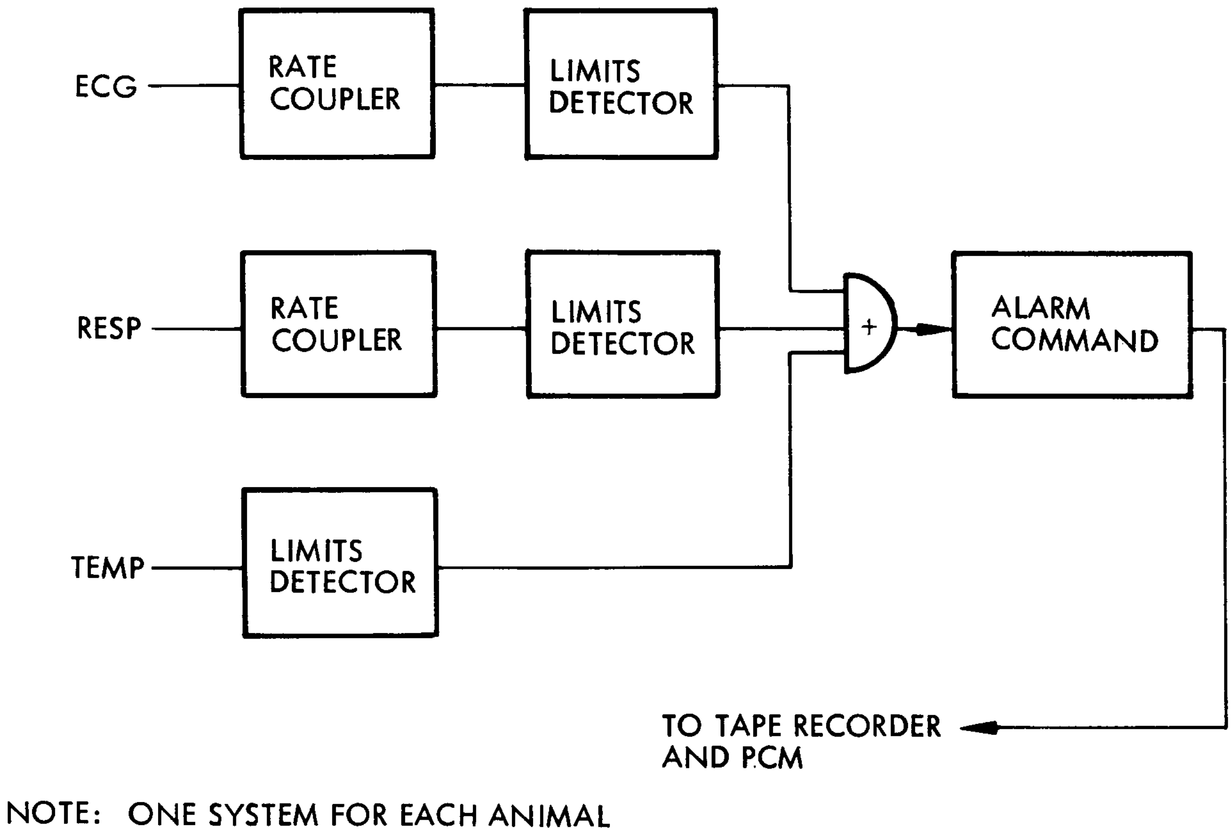


Fig. 100 Rate Detector and Limits Block Diagram

The temperature signal is already in analog form, but ECG and respiration signals must be changed to a rate signal. This is accomplished by amplifying the biosignal, then connecting it to a Schmitt trigger. The Schmitt trigger then triggers a one-shot multivibrator, and the biosignal is integrated into an analog signal. Minimum and maximum limits are detected which are used to control the command to the tape recorder.

In-flight calibration. – The normal approach for in-flight calibration is to supply fixed voltage values to certain commutated channels of the multiplexer each time data are transmitted, so that they serve as reference voltages. This calibration method gives only a calibration check of the equipment, including the multiplexer, and the rest of the equipment following it. On the OPE, it is desired to have a calibration of the signal conditioners prior to the multiplexer and, therefore, the following approach has been taken (Fig. 101).

The in-flight calibration for the OPE takes place at the sensor input and gives a calibration and check of the equipment from the sensor to the ground station. A composite signal is gated into the signal conditioner at the sensor input. This composite signal consists of a pulse train consisting of square waves, ramp signals, and square waves of half-amplitude. This signal has a frequency of one cps. When these three signals are received at the ground, much information can be learned about each system that it has been processed through. It will give information as to frequency response, phase shift, and nonlinear amplification.

The composite signal source and gates consist of integrated circuits. They are reliable and use very little power. The composite signal generator is turned on and off with the multiplexer. One channel of the multiplexer has this composite signal on it continuously. This channel can then be compared with the same signal that has been inserted at the inputs of the signal conditioners.

Station contact. – A typical station contact lasts approximately 7 minutes. During this time, data can be transmitted or stored, commands may be given, or tracking may be carried out. Three types of modulation are used to transmit data to the MSFN.

FM frequency modulation: The carried frequency may be modulated with a deviation from 10 Hz to 1.5 MHz. This modulation is used for the transmission of video data. During a station contact, real-time video data or stored video data may be transmitted to the MSFN. The type of video data transmitted is selected by command from the MSFN. During the transmission of video data, commands must be transmitted over the UHF link, and tracking cannot be carried out on the S-band system.

TLM PCM: The TLM data from the main commutator or TLM stored data are transmitted on a subcarrier which is 1.024 MHz away from the center frequency of the FM carrier. During a station contact, a ground command selects either real-time or stored TLM data to be PCM on the subcarrier for transmission to the MSFN.

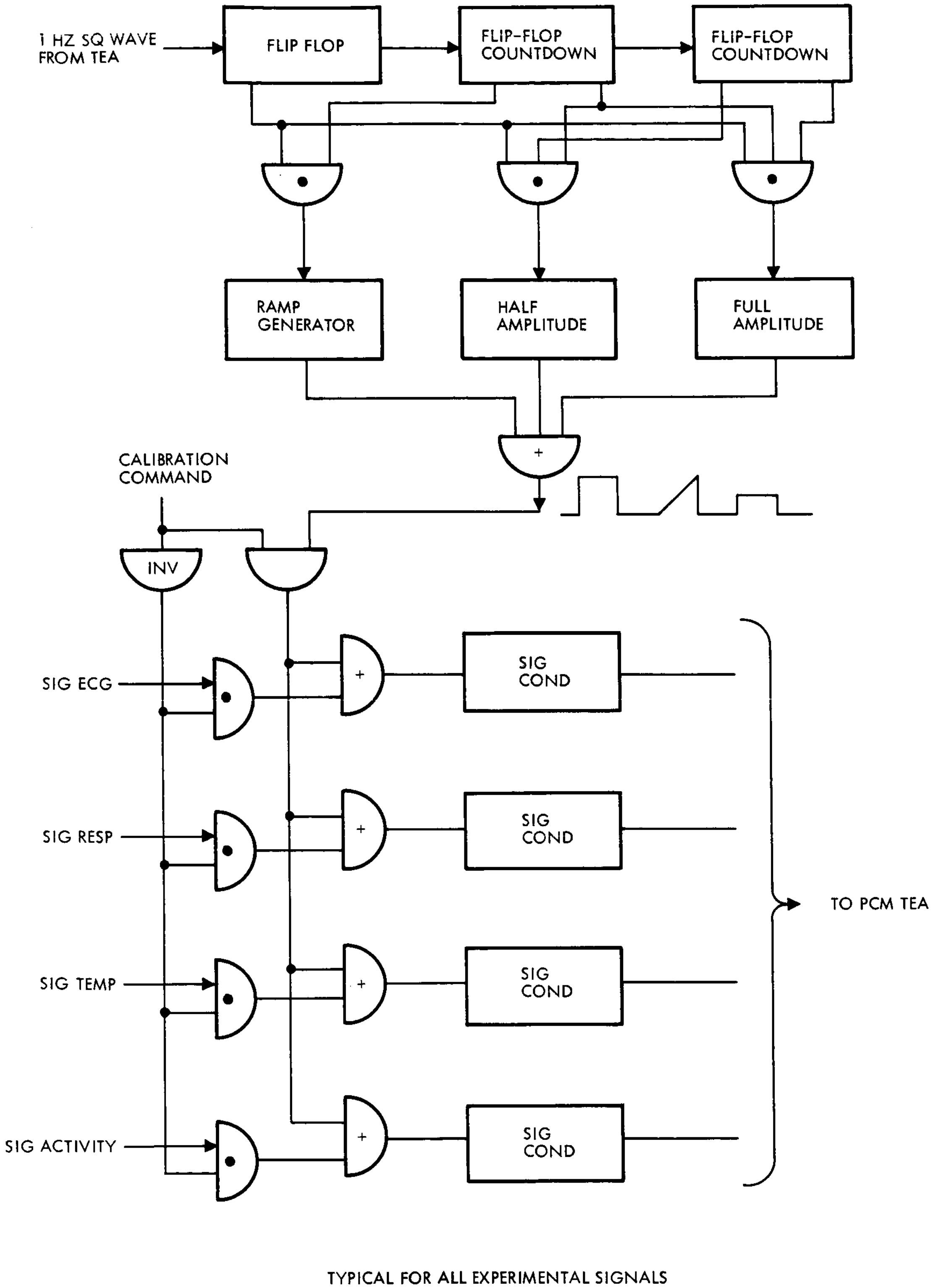


Fig. 101 In-Flight Calibration for Experiment Data

On-orbit data, which consist of 10 seconds of data every 12 minutes, can be stored on three minutes of tape.

The work panel data are stored on a separate tape recorder and are also read out each orbit over the station. They are connected into the parallel digital inputs of the PCM TEA unit and are read out with the other PCM data. The stored work panel data for one orbit can be read out in 4 minutes. Tracking using the USBE can be carried out during the transmission of PCM data.

Audio and biodata PM: Audio and biodata can be transmitted during the same time as video (FM) or TLM (PCM). Audio or biodata are transmitted on a subcarrier that is 1.25 MHz from the carrier's center frequency. During a station contact, real-time audio or biodata may be transmitted over the subcarrier in the PM mode. This subcarrier is also used for spacecraft tracking, using the USBE. If the USBE is used for commands or tracking, data cannot be transmitted on this subcarrier.

SPACECRAFT DESCRIPTION

This section presents the preliminary design description of the spacecraft subsystems. Also included are a weight summary of all payload and spacecraft equipment and a discussion of the considerations involved in modifying the OPE vehicle to include a simulated gravity environment.

Spacecraft

The spacecraft consists of the thermal control, guidance and attitude control, and electrical power subsystems. These subsystems are described in the following paragraphs.

Thermal control subsystem. – The basic thermal design task was to formulate a spacecraft design which thermally integrated the OPE spacecraft subsystems such that appropriate temperature control is achieved for operational as well as nonoperational conditions. The specific thermal requirements are listed below.

Temperature requirements: Table 34 lists the thermal requirements for the critical OPE spacecraft subsystem components for operational as well as nonoperational periods from launch to orbital phases of flight.

The ECS Radiator must be capable of rejecting a 785 Btu/hr heat load while receiving the maximum incident environment energy input consistent with orbital conditions. Under such conditions, the radiator must provide an outlet fluid temperature of 45°F with a 5° gradient across the radiator. The radiator must not freeze under a low thermal input load of 468 Btu/hr while receiving the minimum incident environmental energy input consistent with orbital conditions.

Thermal control concept: Thermal considerations regarding temperature control extend from prelaunch through ascent, parking orbit, and orbit phases of flight. Orbital phases of flight were given initial consideration.

1. On-orbit thermal control – The OPE spacecraft will fly at an orbit altitude of 260 nautical miles in a circular earth orbit through all solar incidence angles (β) (see Fig. 102) between ±53.5 deg. The spacecraft will be sun-oriented during the sunlit portion of the orbit with the spacecraft +Y axis pointing at the sun and with the solar arrays normal to the sun. The +Z axis, which is the pitch axis as well as the angular momentum axis, can be pointed in any direction within the X-Z plane. While in the earth's shadow, the spacecraft loses the sun-oriented attitude and is free to rotate about the pitch axis where rate control restricts the spin rate, if any, to a maximum limit. Concurrently, the vehicle can precess as much as 4.3 deg in roll or yaw from the sun-oriented position. As the spacecraft leaves the earth's shadow and enters the sunlit portion of the orbit, it recaptures its sun-oriented position within 20 seconds from the time it leaves the earth's shadow.

Table 34

SPACECRAFT THERMAL REQUIREMENTS

Critical Components	Temperature Limits (°F)	
	Operational	Nonoperational
Lifecell atmosphere	(see para. on atmosphere control)	—
Tanks and contents	50 to 100	50 to 100
Battery	40 to 70	—
Data management	-30 to +158	-30 to +158
Solar cells	-200 to + 150	-250 to + 250
ECS radiator	-45 to + 35	NA

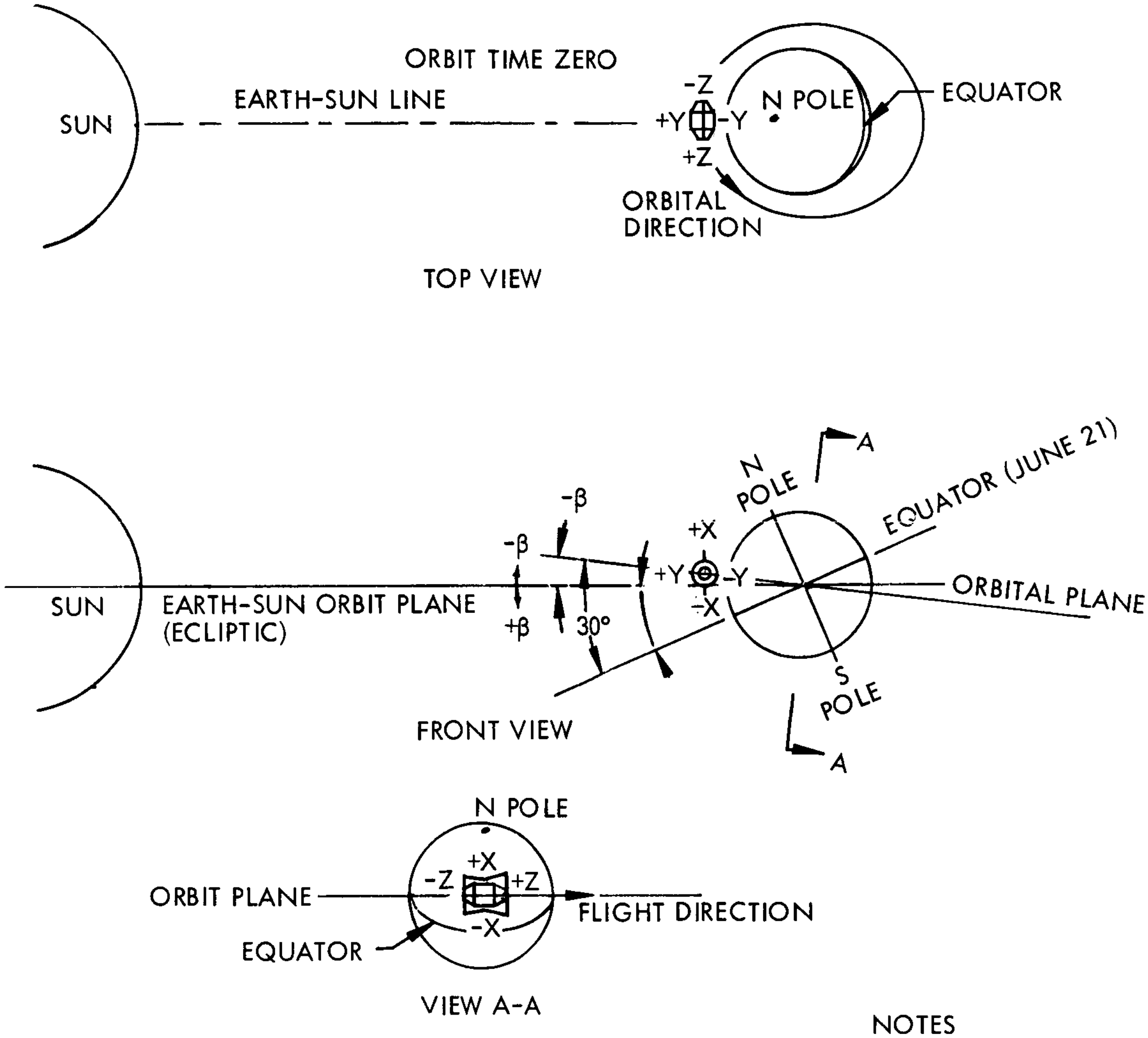
It became apparent that a comprehensive spacecraft thermal analysis where all possible combinations of attitude and orbit solar incidence angle (β) are considered is beyond the scope of this study.

This preliminary design analysis was restricted to projected critical combinations of orbit and spacecraft attitudes. The analysis was performed primarily to establish design criteria for equipment selection, basic thermal design characteristics and thermal control concept feasibility.

- Lifecell and radiator. As a result of initial analyses, a thermal design philosophy for integrating the various components was formulated. Since the lifecell has its own active environmental control system, it was logical to thermally isolate the cell from the rest of the spacecraft such that this system could function independently and be thermally insensitive (as much as possible) to spacecraft environment. It was further decided that the ECS heat rejection radiator (external to the lifecell) should be insulated from all other spacecraft components in order to confine the heat exchange between the lifecell and radiator fluid circulation circuit. This circuit utilizes glycol water to transport the rejected heat from the cell to the radiator. In this manner, maximum atmosphere control effectiveness can be achieved. In order to minimize radiator area and external loads from solar and albedo heat inputs, a low solar absorptance was used and the radiator placed in a region of the spacecraft such that the environmental inputs were minimal.

- Data management subsystem. The data management subsystem package can dissipate from 53.3 to 158 watts of power. Accordingly, this package must be placed near the spacecraft skin from which the generated heat can be rejected to space by application of appropriate thermal finishes to the skin surfaces. The heat dissipation is also used to control package temperatures.

- Battery. Since the power variation of the battery precluded reliable passive temperature control, it was decided to use the coolant circuit of the active control system to control battery temperatures. The heat generated by the battery is rejected to the coolant fluid passing through the cold plate



NOTES

1. β (orbit solar incidence angle) is defined as the angle between the plane of the orbit and the earth-sun line.
2. Line zero is defined as the time when the spacecraft is nearest the sun.

Fig. 102 Orbit Geometry

on which the battery is mounted and transported to the ECS radiator. This battery cold-plate assembly is insulated from the spacecraft skin to minimize external environmental effects. Furthermore, the battery heat is used as an aid in maintaining the radiator above the coolant freezing temperature.

- X-band transceiver. The X-band transceiver operates only during the 12-hr rendezvous period when the primates are being retrieved. Appropriately, this unit is placed in a region where temperatures are compatible with the nonoperational temperature requirements during dormant orbital periods. When the transceiver is activated, two methods were considered for rejecting the generated heat:(1) heat rejected to the spacecraft skin or (2) heat rejected to an evaporant coolant within the package. The evaporative coolant approach was selected.

- Lifecell and associated equipment. Since the lifecell is to be maintained near 75°F, and also because the water and atmosphere tanks are thermally inert (i.e., non-heat generators), it was appropriate to thermally couple the atmosphere and water tanks to the lifecell so that the active thermal control system can be used to control the temperature of the atmosphere supply, water, and preservative tanks.

2. Thermal control during prelaunch, ascent, and parking orbit – During prelaunch activities, ground cooling will be provided to remove the spacecraft electronic heat load and to maintain desired temperatures. For ascent, a passive technique employing thermal shielding will be used, where necessary, to protect the OPE spacecraft from excessive temperatures caused by aerodynamic heating of the SLA. The ECS coolant surge tank has sufficient thermal capacity for approximately 15 minutes of operation. Beyond this, a small freon supply could provide cooling through the ground cooling heat exchanger. During the parking orbit phase, passive techniques will be used for temperature control and, if necessary, will be supplemented with heaters, temporary auxiliary active control systems, or spacecraft attitude restraints.

Thermal design for on-orbit temperature control: The desired thermal characteristics were implemented into the design of the OPE spacecraft for on-orbit temperature control. The general design goal was to employ passive techniques to the greatest extent possible, augmented with heaters only where absolutely necessary. Accordingly, optical properties of selective thermal finishes and superinsulation thermal shields were used for the radiation design in conjunction with a judicious arrangement of heat conduction paths and equipment locations within the spacecraft. Thermal lag arising from heat capacity and thermal resistance is used to attenuate orbital temperature variations of all components.

1. Environmental control system radiator – The ECS radiator was investigated first. The disc end on the -Z side of the pitch axis was selected as a potentially desirable location for the radiator. The incident albedo and earthshine heat rates to this location were obtained using an LMSC developed computer program. Solar energy is not directly incident on this end, except for a small amount which impinges

on this surface during the 20-sec sun-acquisition period when the spacecraft leaves the earth's shadow. During this 20-sec period, the spacecraft may be tilted such that the angle between the plane of the disc surface and the spacecraft-sun line could be as much as 15 deg. However, since a low solar absorptance surface will be used for the radiator and since the 20-sec sun acquisition time is very short, the amount of solar energy absorbed during this period is negligible. Heat rates were obtained for both $\beta = 0$ deg and $\beta = +53.5$ deg and for the six Z-axis (pitch axis) positions, which are described in the following subsection.

It is anticipated that the above combinations of orbits and spacecraft attitude would encompass the maximum and minimum heat input conditions on the radiator for all possible combinations of orbits and attitudes expected during the orbital mission lifetime. This heat rate information was used to establish the radiator thermal design.

The radiator size necessary to dissipate the heat required to maintain desired lifecell thermal and humidity conditions under varying orbital heat rate input is determined from a review of the variables involved. This review shows that the maximum energy input occurs at solar incidence angle $\beta = 0$, orbit time = 1412.5 sec (based on a total orbit time of 5650 sec) with reference to the orbit position nearest to the sun and for the spacecraft attitude where the +Z-axis is in the orbit plane. In an attempt to minimize radiator size, a low solar absorbtance material, i.e., Optical Solar Reflector (OSR), was chosen as a radiator finish. For this condition, the solar energy input is zero, and albedo energy input is 0.00111 Btu/sec-ft^2 and the earthshine input is 0.01794 Btu/sec-ft^2. The heat to be rejected is 785 Btu/hr (0.218 Btu/sec). Thermal finish properties of the radiator (OSR) are an absorptance (α_s) = 0.05 and an emissivity (ϵ_t) = 0.8. The radiator temperature is 35°F (495°R) and radiator heat losses to (or gains from) the spacecraft are negligible. The radiation area is determined as follows:

$$A = \frac{(Qr)\ \text{net}}{\epsilon_t \sigma T^4 - \alpha_s (I_s + I_a) - \epsilon_t (I_{es})}$$

where,

Qr = heat rejected (Btu/sec)
σ = Stefan-Boltzmann Constant, $0.476\ (10^{-12})$ Btu/sec-ft^2°R^4
ϵ_t = radiator emissivity
α_s = solar absorptivity
T = radiator temperature, °R
I_s = incident solar energy input (Btu/sec-ft^2)
I_a = incident reflected solar energy input (albedo) (Btu/sec-ft^2)
I_{es} = incident infrared radiation from earth (Btu/sec-ft^2)

$$A = \frac{0.218}{(0.476)(10^{-12})(0.8)(495)^4 - 0.05\,(0.00111) - 0.8\,(0.01794)}$$

$$A = 27.4\ \text{ft}^2$$

In the event the radiator outlet temperature approaches the minimum allowable design temperature for the coolant (-45°F), the outlet sensor will actuate the electrical heater on the inlet side of the radiator, thereby maintaining coolant temperature above freezing.

2. Spacecraft surfaces – An initial finish pattern was obtained for the spacecraft based on a representative orbit ($\beta = 0°$) and a representative spacecraft attitude, where the Z axis (pitch axis) is in the orbit plane and the +Y axis (yaw axis) points toward the sun during the entire orbit (no rotating about the Z axis while in the earth's shadow). This finish pattern was then verified for the $\beta = +53.5$-deg orbit.

Environmental heat rates were obtained for this orbit condition and a thermal finish pattern was defined for the spacecraft as shown in Table 35 . Optical properties of the thermal finishes are specified in Table 36 . The degraded solar absorptance properties of the thermal finish were used since degradation will occur within the one-year mission lifetime as a result of exposure to the space environment. Solar absorptance is the only optical property that will degrade. The thermal finishes considered could provide, when combined in the form of mosaics, a wide range of solar absorptance (α_S) and infrared emittance (ϵ) values and, accordingly, this offers a considerable latitude in obtaining the appropriate properties of α_S and ϵ and in making final adjustment to ascertain an optimum design.

In prescribing the thermal finish design on the external surfaces of the spacecraft skin, the goal was to obtain orbital average skin temperatures as near as possible to the 75°F lifecell temperature. This was desired in order to minimize the need for excessive thicknesses of superinsulation and low conductive insulation throughout the lifecell support structure.

3. Lifecell and radiator insulation – Multilayer superinsulation in the form of a laminated blanket was used between the lifecell (including the attached atmosphere, water, and preservative tanks) and the skin, on the ECS radiator surface internal to the spacecraft, and on the titanium support truss extending between the cell pressure vessel and docking collar as well as between the pressure vessel and the skin. This was necessary to provide the required high thermal isolation for the lifecell and radiator. A 2-in. thick superinsulation thermal blanket was selected based on a superinsulation conductance value of 1.0×10^{-4} Btu/hr-°F-ft, which is about an order of magnitude greater than indicated in tests on laboratory-type samples. The thermal blanket will consist of 120 alternate layers of 1/4 mil aluminized mylar and tissue glass. (Each layer consists of a sheet of mylar and a sheet of tissue glass.) The conductance value used is considered conservative and is based on experience regarding installations for other projects at Lockheed. Additionally, conduction insulation through the support trusses was achieved by using titanium Ti (13V-11Cr-3Al), which has a thermal conductivity of 3.9 Btu/hr-°F-ft, and by making the cross sectional area normal to the heat flow as small as possible, but still large enough to provide mechanical integrity.

4. Data management package – The data management package consists of several subsystem modules which are mounted directly to a six-sided aluminum enclosure. The heat generated within the modules is transfered to the enclosure. This heat is then

TABLE 35

THERMAL FINISH DESCRIPTION

Component	Description
Exterior of Spacecraft	
Spacecraft Skin:	
• Skin area on conical sections. (following angular distances are in the X-Y plane where the +X axis is taken as the reference point and the positive angular distances are toward the +Y axis; angles 0° to 180° are on the sun side)	
0° to 36°	Black thermatrol
36° to 73°	38% black thermatrol, 62% white thermatrol
72° to 108°	19% black thermatrol, 81% white thermatrol
108° to 144°	38% black thermatrol, 62% white thermatrol
144° to 216	Black thermatrol
216° to 324°	Chemically polished aluminum
324° to 360°	Black thermatrol
• Skin area on cylindrical surface, except region adjacent to data management system.	
0° to 36°	Black thermatrol
36° to 72°	25% black thermatrol, 75% white thermatrol
72° to 108°	10% black thermatrol, 90% white thermatrol
108° to 144°	25% black thermatrol, 75% white thermatrol
144° to 216°	Black thermatrol
216° to 324°	Chemically polished aluminum
324° to 360°	Black thermatrol
13.1 ft^2 cylindrical skin area adjacent to data management package.	25% OSR (Optical Solar Reflectors) with silver as the second surface mirror, 75% white thermatrol
• Disc end on docking collar side	Chemically polished aluminum

TABLE 35 (Cont.)

Component	Description
<u>Exterior of Spacecraft</u> (Cont.)	
Docking Collar:	Chemically polished aluminum
Solar Panels (shaded side):	White thermatrol
ECS Radiator	OSR with silver as the second surface mirror
Separation Ring (exterior surface):	
on sun side	Same as skin
on shaded side	Same as skin (use Mystik Tape to get equivalence of chemically polished aluminum).
<u>Interior of Spaceraft</u>	
Internal surface area of skin between docking collar and the point of skin from which the superinsulation thermal shields (thermal blanket) breaks away and covers the lifecell end nearest the docking collar side.	Black thermatrol
Surface of skin and radiation panels (near D. M. S.) adjacent to superinsulation thermal shields.	Chemically polished aluminum
Exterior surface of lifecell and tanks, except for following cell and tank regions:	Chemically polished aluminum surfaces; mystic tape applied to tanks
1/3 of cylindrical surface of tank which is adjacent to the lifecell pressure vessel and the region of the pressure vessel defined by the projection of the 1/3 region of the tank on to the vessel surface.	Black thermatrol
Interior surface of ECS Radiator:	Lockspray gold

TABLE 35 (Cont.)

Component	Description
Interior of Spacecraft (Cont.)	
Data Management System (D. M. S.):	
• Exterior surface of Data Management System (D. M. S.) enclosure facing skin and radiation panels (5 sides)	Black thermatrol
• Remaining one side adjacent to superinsulation thermal shield	Chemically polished aluminum
Separation ring attached to conical skin	Lockspray gold
Docking Collar	Black thermatrol
Radiation Panels:	
• Surface of radiation panel facing skin and D. M. S enclosure	Black thermatrol
• Surface of radiation panel adjacent to superinsulation thermal shields	Chemically polished aluminum
Internal skin surface facing radiation panel and D. M. S. enclosure	Black thermatrol

TABLE 36

OPTICAL PROPERTIES
(Nominal Values)

Thermal Finish or Material	Solar Absorptance (α_S)		Infrared Emittance (ϵ)
	Undegraded	Degraded	
Black thermatrol paint	0.92	0.92	0.95
White thermatrol paint	0.18	0.32	0.95
Chemically polished aluminum	0.16	0.16	0.06
Lockspray gold	—	—	0.06
OSR, silver	0.05	0.05	0.80
Mystic tape	0.12	0.12	0.04
Solar cells	0.76	0.76	0.79

rejected to the skin of the spacecraft. Good heat transfer between the module and the enclosure is accomplished by providing large and smooth mating surfaces. To further enhance the heat flow between the modules and enclosure, all of the internally exposed surfaces of the modules and enclosure are painted with a high emittance thermal finish such as black thermatrol paint.

The thermal finish on the skin surface adjacent to this package was prescribed specially to reject the heat generated by the package. The design was based on (1) a maximum continuous power dissipation of 158 watts, (2) average orbit heat rates to the skin for the $\beta = 0$ orbit, (3) a package temperature near the maximum permissible value, and (4) degraded white thermatrol paint on the skin. The surfaces on the exterior of the package which face the skin and the radiation panels, as well as the surfaces of the skin which face the package, and the surfaces of the radiation panels were painted with black thermatrol.

It must be noted that assumption of a 100-percent duty cycle for the power dissipation was very conservative. In all probability, a considerably lower duty cycle with a longer cycle period will occur. This will significantly relax the thermal problem and the temperature range will be reduced. If future detailed investigations reveal that in some orbits the lower permissible temperature is exceeded when a combination of minimum power dissipation and environmental heat input occurs, heaters can be employed to maintain temperatures above the minimum limit.

5. X-band transceiver – The X-band transceiver is used only during the rendezvous period when the primates are being retrieved by the astronauts. During the orbital phases of flight this unit is inactive. Upon activation, the unit dissipates 70 watts of power. It would be desirable to locate this unit in a region of the spacecraft where the temperatures of the skin are compatible with the unit's thermal requirements for the

inactive orbital periods. Then, upon activation, the unit would reject the heat to the skin which radiates this energy to space. However, no such location was found, although additional analysis may disclose a favorable site. The docking collar region has potential in this regard. Therefore, the transceiver was placed between the superinsulation blanket and the lifecell pressure vessel where its dormant temperature is the same as the lifecell pressure vessel. Upon activation, the heat generated in the transceiver is rejected to a coolant contained within the package. The latent heat of vaporization of the coolant is used to absorb the rejected heat and the vapor is then expelled overboard. The pressure of the coolant is regulated such that the coolant will boil at the appropriate temperature.

6. Batteries – The batteries are mounted on a cold plate and are placed between the superinsulation blanket and the lifecell. Low-emittance surfaces are also used on the battery enclosure to minimize the heat exchange between the battery and the pressure vessel. The battery generated heat is rejected to the coolant which is used for the lifecell environmental control system. The coolant flows through passages in the cold plate. In this manner, the battery temperatures are controlled within limits.

7. S- and X-band antennas – For the temperature control of the S- and X-band antennas, thermal finishes are applied to the external surface of the antennas on the sun side of the spacecraft. No thermal finishes are applied to the antennas on the shaded side because of possible interference with the RF performance of the antennas The solar absorptance to infrared emittance ratio needed is exhibited by such finishes as polished aluminum. For the analysis, it was assumed that the nonconductive filler exhibits a high absorptance and emittance properties ($\alpha_s = 0.92$, $\epsilon = 0.95$).

Analysis for on-orbit thermal control: For integration design evaluation, an analytical thermal model of the OPE spacecraft was formulated. The Lockheed "Thermal Analyzer," which is a high-speed digital computer program, was used to perform the appropriate heat balance for obtaining temperatures.

In the analysis, the lifecell was assumed to be a constant 75°F since the environmental control system (ECS) was the predominate control system and since the lifecell was highly insulated from the spacecraft and associated electronic equipment.

In the determination of the absorbed orbital environmental heat rates and the thermal radiation coefficients for heat transfer, the spacecraft skin surface and the solar panels were subdivided into several elements and the thermal finish pattern for these elements was prescribed. An existing computer program was used to facilitate calculations of the orbital heat rates because of the complex geometrical relationship between the sun, earth, and spacecraft. The components of absorbed heat rates consist of the direct environmental energy absorbed and the reflected environmental energy absorbed from other reflective spacecraft surfaces. Shading of surfaces was included in determining the heat rates. The radiation coefficients for heat transfer between external surfaces of the skin and solar array elements as well as between these elements and space were determined using an existing computer program. The multiple interreflections that occur between these surfaces were also included in the determination of these coefficients.

The geometrical variables considered in the determination of the heat rates were: (1) the angle between the orbit plane and the earth-sun line, (2) the orbit altitude, and (3) the spacecraft attitude and configuration. Specifically, for the spacecraft integration analysis, the heat rates were obtained for β = 0 deg and + 53.5 deg, an altitude of 260 nautical miles for the +Z axis (pitch axis) in the orbit plane, the + Y axis pointing at the sun, all three axes being space oriented (no rotation in the earth's shadow), and for a circular orbit path. The heat rates to the ECS radiator were evaluated for β = 0 deg and β = + 53.5 deg and for the following six spacecraft positions:

(1) + Z axis is in the orbit plane.
(2) + Z axis yaws 45 deg in the X-Z plane from position (1) toward the - X axis.
(3) + Z axis yaws 90 deg in the X-Z plane from position (1) toward the - X axis.
(4) + Z axis yaws 90 deg in the X-Z plane from position (1) toward the + X axis.
(5) + Z axis yaws 135 deg in the X-Z plane from position (1) toward the + X axis (45 deg beyond the + X axis position toward the original - Z axis position).
(6) + Z axis yaws 180 deg in the X-Z plane from position (1) toward the + X axis.

Performance evaluation of orbital thermal control system: The OPE spacecraft thermal design for integrating the subsystem components was evaluated on an orbital average basis for both the β = 0 deg and the β = +53.5 deg orbits. The results are shown in Table 37 and Figs. 103 and 104. Both the table and these figures illustrate the nominal on-orbit average temperature for the case where the exterior spacecraft thermal finish has degraded. Component temperatures are shown in the table and the spacecraft skin and solar array temperatures are presented in the figures. The temperatures shown in the figures represent an average of the temperatures within the indicated sections of the skin and solar array. For the components, the results represent the temperatures for the case where the component is assumed isothermal. The tabulated results for β = 0 and β = +53.5 deg are also applicable to other orbits. See Table 38.

The data management system attains a maximum temperature of 129°F when the power dissipation is a maximum 158 watts and a 100-percent duty cycle is used. The estimate for the minimum temperature of 10°F was made for 53.3 watts of power dissipation and 100-percent duty cycle operation. Thus, the on-orbit average temperatures ranged between 10°F and 129°F which is well within the allowable limits of -30 to + 158°F. This thermal performance provides a total temperature margin of 69°F which is considered sufficient to accommodate orbital temperature variations and temperature uncertainties. A further refinement would be to slightly modify the thermal finish pattern such that the performance is centered within the allowable limits. It must be noted, however, that for reasons indicated in the previous subsection, these results are considered conservative.

TABLE 37

ON-ORBIT AVERAGE TEMPERATURES (° F)

Component	Temperature (° F)		Allowable Temp. (° F)	
	For $\beta = 0°$	For $\beta = +53.5°$	Min.	Max.
Data Management System				
• For 158 watts of continuous power dissipation and degraded thermal finish on exterior of skin	121	129	-30	158
• For 53.3 watts of continuous power dissipation and undegraded thermal finish on exterior of skin	10	>10	-30	158
X-Band Transceiver (dormant)	75	75	-30	158
Docking Collar	48	53	—	—
Tanks and Content	75	75	50	100
Solar Cells	93 to 104	100 to 115	-250	150
ECS Radiator	(See Note 1)	(See Note 1)	-45	35

Note 1: Radiator is maintained between -45° F and +35° F at all times with the aid of a heater, when needed, for preventing temperatures lower than -45° F.

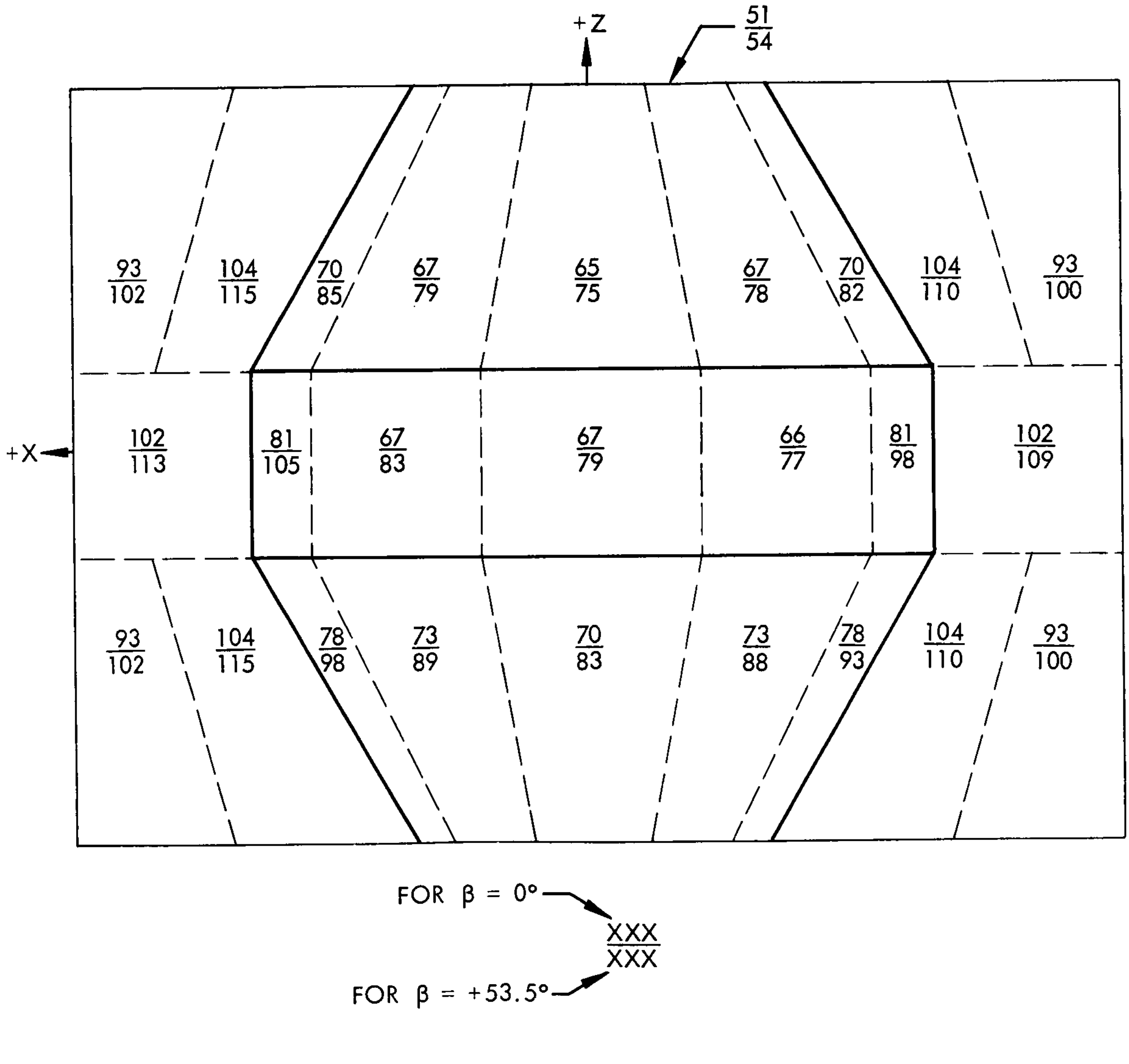


Fig. 103 On-Orbit Average Temperatures (° F) for Sunlit Side of OPE Spacecraft

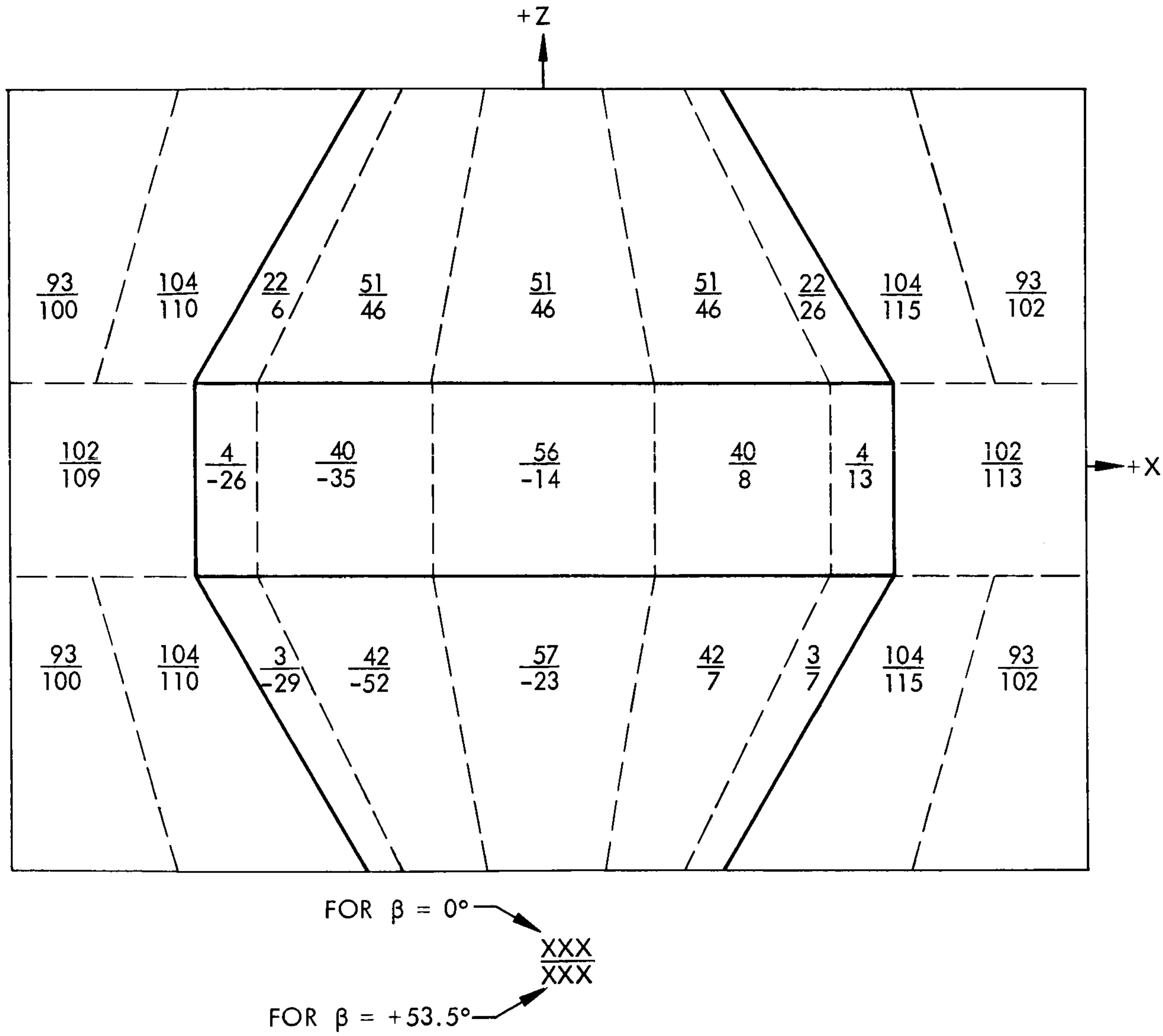


Fig. 104 On-Orbit Average Temperatures (° F) for Subsolar Side of OPE Spacecraft

TABLE 38

OTHER APPLICABLE ORBITS

Spacecraft Attitude (t = 0)	Flight Direction	Orbit Solar Incidence Angle		
		β = 0 deg	β = -53.5 deg	β = +53.5 deg
+Z leading	Normal	X		X
	Reversed	X	X	
-Z leading	Normal	X	X	
	Reversed	X		X

In all probability, peak orbital temperatures of the solar cells will be greater than the 150°F maximum limit because of orbital average temperatures being as high as 115°F. For the design, it was assumed that the entire solar illuminated surface of the array exhibited solar absorptance and infrared emittance properties which are consistent with solar cells. Peak temperatures can be easily lowered to be compatible with the 150°F limit by applying a low solar absorptance and high infrared emittance thermal finish, such as OSR or white thermatrol, to a certain percentage of the array on the sun side. This combination of solar cells and applied thermal finish will form a mosaic which would exhibit a lower effective absorptance. However, prior to using the mosaic, a refinement of the thermal finish design of the spacecraft skin near the array would be investigated. Presently, the skin surfaces near the array are painted with 100 percent black thermatrol which exhibits a high solar absorptance and infrared emittance. The purpose of applying this paint to these regions was to minimize the amount of incident environmental energy which impinges on the skin and then reflects onto the array. Because of the high emittance, however, the skin impedes the array heat rejection to space. An alternative would be to have these skin surfaces exhibit high solar and infrared reflectance characteristics, such as provided by chemically polished aluminum or Mystic tape. The reflective surfaces would appreciably reduce the resistance to the array heat rejection to space and, thereby, tend to lower array temperatures. The reflective skin surfaces will also increase the amount of environmental energy which is received by the array as a result of reflections from the skins and, thereby, tend to increase array temperatures. However, the net effect may result in lower temperatures.

Of particular concern is the amount of heat loss (or gain) from the lifecell through the superinsulation and support structure. A design objective was to limit this loss to 50 Btu/hr. The calculated net orbital average loss was estimated to be less than 15 Btu/hr for β = 0 deg and +53.5 deg. The effect of a 15 Btu/hr heat loss on the lifecell environmental control system is negligible since it constitues a very small percentage of the minimum lifecell heat rejection of 468 Btu/hr. Similarly, the spacecraft heat leak to the ECS radiator was also found to be very small and, even if combined with the lifecell loss, the total effect is still not appreciable.

In actual practice, heat leaks through multilayer insulations are more a function of joint design and blanket fabrication techniques than conventional heat flow through the laminae. Therefore, the final solution to the thermal control problem will have to be achieved by testing.

Thus, with respect to the β = 0, -53.5, and +53.5 deg orbits, the analytical results indicate that the proposed thermal design, using primarily state-of-the-art passive techniques, is feasible in providing the required spacecraft temperature control.

Guidance and attitude control subsystem.— This section covers the requirements, analysis, and definition of the spacecraft attitude control subsystem. The first subsection evaluates the external disturbances with which the control system must contend. This is followed by a discussion of internal disturbances and their effect on the control system performance. The next subsection summarizes the spacecraft guidance and attitude control requirements. The last two subsections cover the control subsystem concept, analysis and design.

External disturbance torque evaluation: The gravity gradient torques are given by (neglecting cross-products of inertia and orbit eccentricity):

$$\begin{bmatrix} M_{1_{GG}} \\ M_{2_{GG}} \\ M_{3_{GG}} \end{bmatrix} = 3\,\omega_o^2 \begin{bmatrix} A_{23}\,A_{33}\,(I_3 - I_2) \\ A_{13}\,A_{33}\,(I_1 - I_3) \\ A_{13}\,A_{23}\,(I_2 - I_1) \end{bmatrix}$$

where ω_o is the satellite geocentric rate and the A_{i3} represent the direction cosines between the i^{th} body axis and the local vertical and are time varying functions of the orbit orientation and the vehicle true anomaly since the vehicle will be sun oriented. The maximum value of the torque $M_{j_{GG}}$ will occur when $A_{k3}\,A_{\ell 3} = 1/2$ and when this occurs then $M_{k_{GG}} = M_{\ell_{GG}} = 0$. For the values

$$I_1 = 2{,}176 \text{ slug ft}^2$$

$$I_2 = 2{,}395 \text{ slug ft}^2$$

$$I_3 = 2{,}325 \text{ slug ft}^2$$

and

$$\omega_o^2 = 1.32 \times 10^{-6} \text{ rad/sec}^2$$

$$\left(M_{1_{GG}}\right)_{Max} = 1.4 \times 10^{-4} \text{ ft-lb}$$

$$\left(M_{2_{GG}}\right)_{Max} = 3.0 \times 10^{-4} \text{ ft-lb}$$

$$\left(M_{3_{GG}}\right)_{Max} = 4.4 \times 10^{-4} \text{ ft-lb}$$

For the purposes of fuel consumption computations, these values will be divided by $\sqrt{2}$ to obtain an effective value while the vehicle is sun oriented.

The maximum solar torque for this vehicle is 2.4×10^{-5} ft-lb and is practically negligible with respect to the gravity gradient and aerodynamic torques.

The magnetic torques can become prohibitive if the assembled vehicle is not checked and, if necessary, compensated to obtain magnetic cleanliness. Therefore, the degree of magnetic cleanliness is specified as an equivalent 0.1 amp-turn about a loop 7 ft in radius. This will produce a maximum torque of 6×10^{-5} ft-lb at the altitudes of interest.

The aerodynamic torques were evaluated with the aid of a computer program which requires that the vehicle be described by flat plates and cylinders; thus, the approximation indicated in Fig. 105 was selected to satisfy the input limits of the program and to yield conservative force and moment coefficients. Referring to the numbering scheme shown in Fig. 105, Segments 1 to 20 and 31 and 32 were flat plates with one side exposed to the flow while Segments 21 to 30 were flat plates with both sides exposed. Finally, Segment 33 was input as is, a cylindrical segment. The program input was checked by selecting several convenient vehicle orientations and hand calculating the force and moment coefficients using functions tabulated in conjunction with Ref. 15; these results were then compared with the computer results and found to agree.

The computer results were obtained as a function of vehicle orientation. The velocity vector is referenced to the +1 axis by two rotations. The first rotation is an angle β (measured positive about the -3 axis) which defines an intermediate axis +1'. The angle Q is measured from the +1' axis with +Q in the direction of the -3 axis. By this definition Q is the orbit true anomaly.

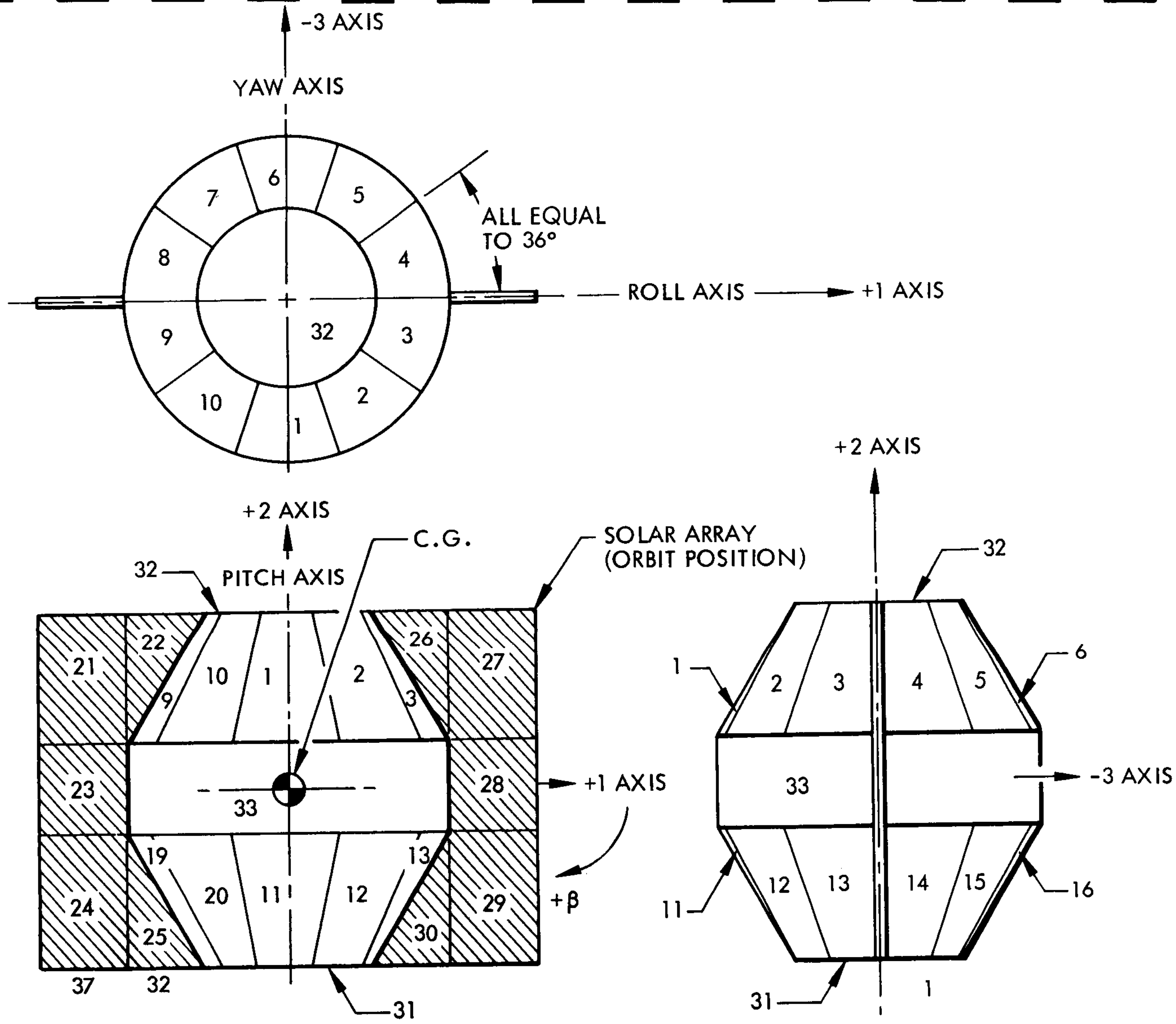


Fig. 105 Spacecraft Aerodynamic Model

The computer results did not account for the shading of one vehicle portion by another. After determining the shading effects, the computer results were adjusted accordingly. The resultant moment coefficients were then studied as a function of centroid-c.g. separation (denoted δ). The moment coefficients for a known δ are determined from the computer results using the following relation:

$$C'_m = C_m + \frac{\delta a}{L_{ref}} C_{fb}$$

where the subscripts a and b indicate body-axis directions which are mutually orthogonal to the axis about which the moment of interest is taken. To obtain actual torques from the coefficients, the following equation is used:

$$M = C_m \left(\frac{1}{2} \rho V^2 \right) A_{ref} L_{ref}$$

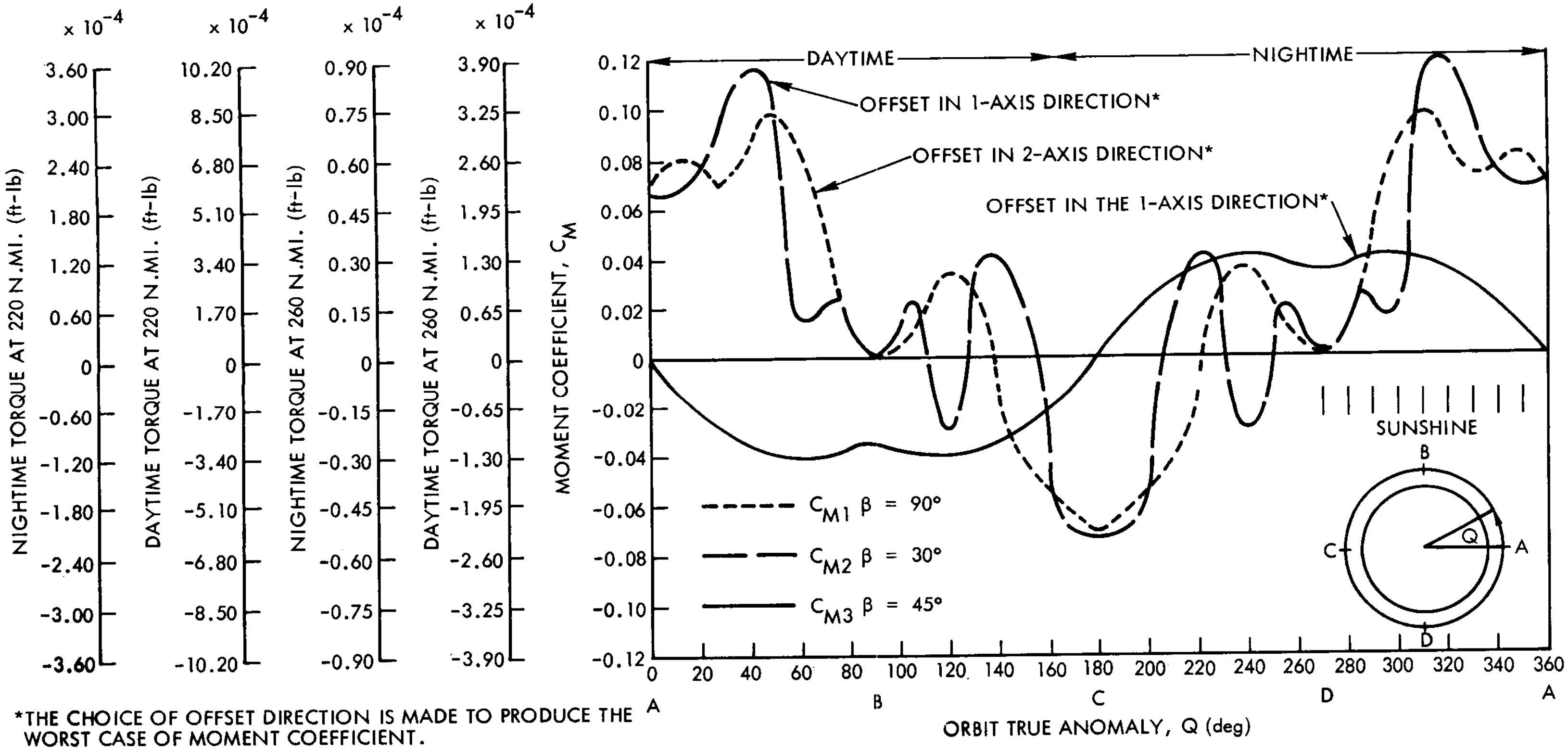
where ρ is density (which is different for "day and night"), V is velocity, and A_{ref} and L_{ref} are the vehicle reference area and length, respectively. These reference values were taken to be:

$$A_{ref} = 98.0 \text{ ft}^2$$

$$L_{ref} = 11.2 \text{ ft}$$

Figure 106 is a plot of the largest aerodynamic moment coefficients (and torques) that were calculated about the body axes for a c.g. offset of two inches (along a body axis) for revolutions of constant β. The "Worst Condition" in the title implies that the moments about each axis for an offset of 2 inches will not result in a pulse larger than that indicated in Fig. 106. Thus, it is assured that for a 2-in. offset in the 1, 2, or 3 direction, the pulse about the 1 axis will be no greater than that indicated for $\beta = 90$ deg with the offset in the 2 direction. (It should be pointed out that, because of the cyclic nature of the torques, it is immaterial whether an offset is in the positive or negative axis direction.)

The resultant torques for zero c.g. offset lead to smaller pulses, so every effort should be made to limit the amount of c.g. offset. It is expected that any offset of less than two inches will result in smaller pulses than those for $\delta = 2$. Also, since the aerodynamic torques are larger at lower altitudes, it might be advantageous to arrange for any time varying c.g. shift to be apparent early in the flight and to decrease with time.



*THE CHOICE OF OFFSET DIRECTION IS MADE TO PRODUCE THE WORST CASE OF MOMENT COEFFICIENT.

Fig. 106 Worst Condition Torques for CG Offset of 2 Inches

A summary of the total torques is presented in Table 39 where the gravity-gradient and aero* torques have been root sum squared to give realistic values.

TABLE 39

DISTURBANCE TORQUE VALUE SUMMARY

Torque	3 Axis Yaw ($\times 10^{-4}$ ft-lb)	1 Axis Roll ($\times 10^{-4}$ ft-lb)	2 Axis Pitch ($\times 10^{-4}$ ft-lb)
RMS G.G.	3.08	0.99	2.12
Solar	0.24	0.24	0.24
Magnetic	0.60	0.60	0.60
RMS Aero (Day)*	1.75	3.50	3.50
RMS Aero (Night)*	0.53	1.05	1.05
Day Torque Sum	5.67	5.33	6.46
Night Torque Sum	4.21	2.64	3.77

Internal disturbance torque evaluation: The animals provide a mechanism whereby momentum can be exchanged with the spacecraft. They can be considered to be momentum storage devices with a total combined storage capability of about 12 ft-lb-sec. Actually their ability to store momentum islimited only by their capacity to withstand high spin rates and their reaction time.

Any animal movement transverse to the 25 ft-lb-sec momentum wheel (described later) will result in nutational motion. More serious, however, is animal motion about the wheel momentum vector (the pitch axis).

Fuel expenditure due to animal movement could occur each orbit at sunrise resulting in a fuel loss equal to the animal stored momentum divided by the reaction jet lever arm or roughly 2 lb-sec, assuming the animals have stored 12 ft-lb-sec of momentum.

*Computed from worst value of moment coefficient (C_m) RMS over one orbit and RMS value of ρV^2 for orbit decay from 260 n.mi. to 220 n.mi. for day and night density values, respectively.

If the animals returned their momentum to the system following completion of the acquisition maneuver, an additional 2 lb-sec would be required for reacquisition yielding a total expenditure of 4 lb-sec. The possibility of this occurring enough at sunrise to become a serious fuel drain (100 or so out of the possible 5,840 orbits) is very remote.

During the daytime portion of the orbit, the animals could cause considerably more fuel expenditure if they could synchronize their momentum exchange cycles with the limit cycle. The resultant limit cycle they would induce is as shown in the phase plots of Fig. 107.

In Fig. 107(a), the motion begins at Point 1 with the vehicle drifting across the deadband at limit cycle rate with about 42 minutes ($P\ell/2$) required to traverse from the minus deadband to the plus deadband. At Point 2, the animals spin themselves up (the momentum exchange is assumed to occur in a very short length of time compared to the periods of interest) resulting in a step rate change of the vehicle to Point 3. The reaction jets commence pulsing before the animals return their stored momentum to the system at Point 4. The motion proceeds to Point 6 where the reaction jets turn on and return the motion to Point 7. This same motion and animal maneuver is repeated in Fig. 107(b), where the switching line slopes are steeper and a gas pulse is not triggered before the animals return their momentum to the system. The time between Points 3 and 4 is about 8 minutes.

A more reasonable length of time for the animals to retain this maximum momentum is probably about 2 minutes. Consequently, the animals could build up and return this maximum momentum any time during about 40 of the 42 minutes required to traverse the entire deadband, if the switching lines are nearly vertical, with zero increase in the fuel consumption for one cycle. A method of ensuring vertical switching lines during limit cycle and lines of finite slope during acquisition can be mechanized with solid state switches and a delay circuit. Figures 107(c) and 107(d) illustrate the effects of synchronized motion of the animals with their induced limit cycle. Clearly the vertical switching lines would greatly improve these situations.

An analog simulation is recommended to design the optimum switching line logic and specify the fuel loading contingency for animal movement.

Rotating equipment will only cause fuel expenditure if the equipment has a momentum which is appreciable with respect to the vehicle limit cycle momentum and if the equipment momentum is being varied. Any variable momentum storage device should be limited to changes of less than 0.02 ft-lb-sec. The high flow ECS fans are counterweighted to produce a zero net momentum.

Any translating on-board equipment should either be translated on a line through the mass center or be limited to contribute less than 0.02 ft-lb-sec of angular momentum.

Water dumping should only be accomplished in conjunction with the attitude control system by expelling the water through reaction jets. It is recommended that the

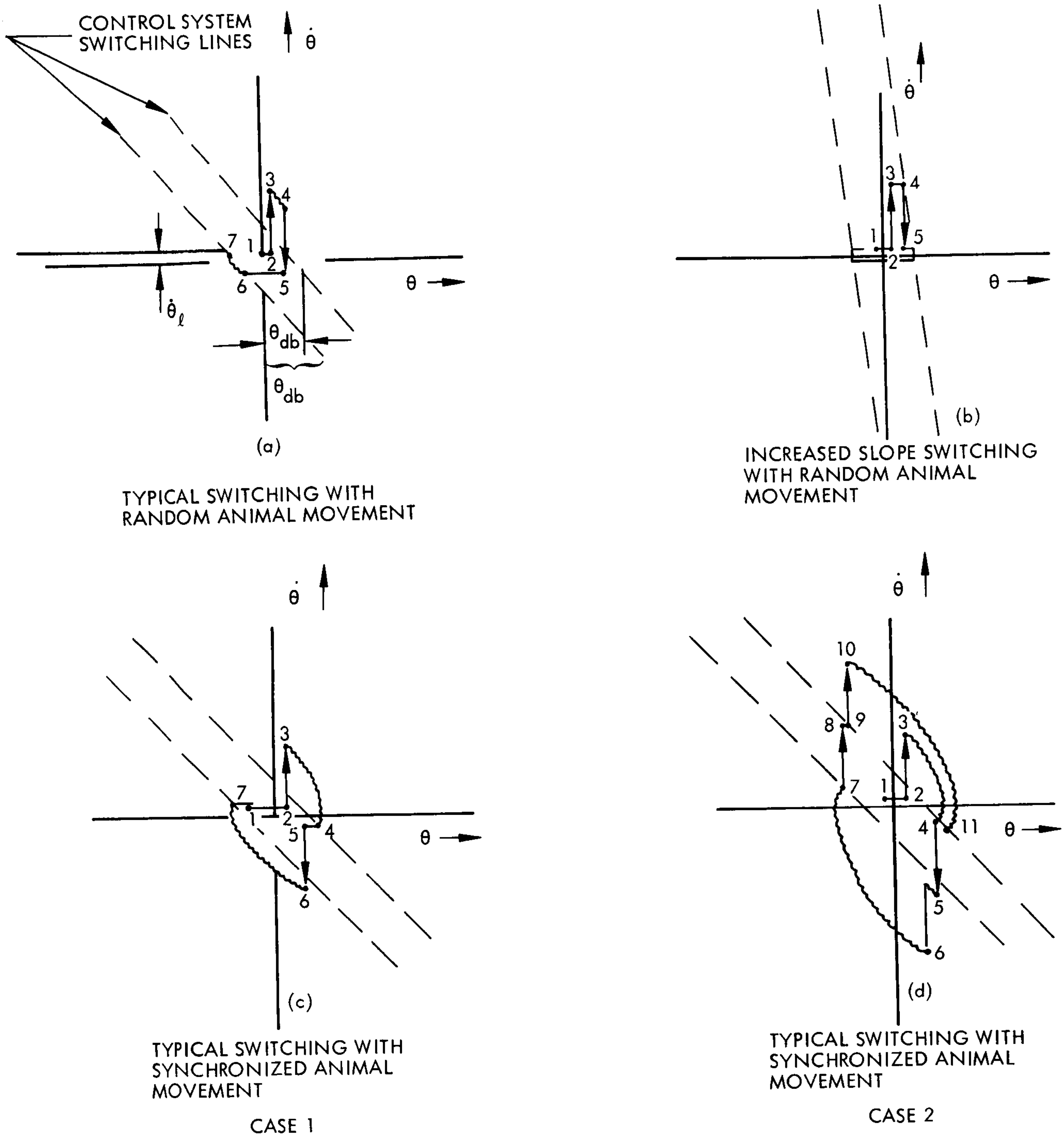


Fig. 107 Animal Momentum Effects on Vehicle Motion

water be stored on-board in a pressurized vessel and serve as a contingency or back-up attitude control fuel supply to the primary nitrogen cold gas system.

The leakage rate of the cabin environment gases and other expendables is estimated to be 0.3 lb/day. It is important that the leakage path or "lever-arm" of this mass expulsion be controlled. It is recommended that a symmetrical array of leakage ports, whose plane contains the vehicle mass center, be provided to expel this matter (110 lb over the years length of the mission) with no more than a 3-inch lever arm. This will result in a maximum control gas expenditure of less than 5 lb to counter the leakage torques.

Guidance and attitude control requirements: There are no requirements for on-orbit guidance of the spacecraft. Since an omnidirectional antenna will be used for communication purposes, the only attitude control requirements are (1) the solar array must be sun-oriented during the orbit daytime and, (2) the vehicle must not spin at a rate which will induce any appreciable on-board "gravity field."

Attitude control concept:

1. Attitude information – The requirements of the attitude control system can be summarized as follows: (1) the solar array must be oriented to within ±15 deg of the sun line during the daylight portion of the orbit, (2) the vehicle rates must be considerably less than 0.1 rad/sec about all axes, and (3) the system must operate for one year. The first requirement can be satisfied by maintaining the vehicle 3 axis (Fig. 105) within ±15 deg of the sun line. Four solar cells can be used to obtain the direction cosines between the vehicle 1 (roll) and 2 (pitch) axes and the sun line designated as the 3_s axis. These direction cosines are defined as S_{13} and S_{23} respectively. Controlling the vehicle such that S_{13} and S_{23} and their rates remain small will satisfy the attitude control pointing requirements in pitch and roll. It will be shown in a later section that the yaw rate requirement can be satisfied by coupling the roll and yaw motion with a momentum wheel and then torquing yaw with a component of the roll axis control torque. The momentum wheel also provides passive control during each satellite night when the sun sensors receive no signal. Hence, the attitude control requirements can be satisfied without employing a sensor for the third axis.

2. Equations of motion – The control system implementation will be shown following an explanation of the vehicle-momentum wheel dynamics. The Euler moment equations for the vehicle-wheel system are given by (neglecting cross-products of inertia):

$$M_{1_d} = \dot{\omega}_1 I_1 + \omega_2 \omega_3 (I_3 - I_2) + h\omega_3 - M_{1_c}$$

$$M_{2_d} = \dot{\omega}_2 I_2 + \omega_1 \omega_3 (I_1 - I_3) - M_{2_c}$$

$$M_{3_d} = \dot{\omega}_3 I_3 + \omega_1 \omega_2 (I_2 - I_1) - h\omega_1 - M_{3_c}$$

where M_{i_d} is the component of external disturbance torques along the vehicle i-axis, I_i the principal moments of inertia, ω_i the components of vehicle angular velocity with respect to an inertial reference frame, h is the wheel angular momentum along the -2 axis, and M_{i_c} is the component of the control torque about the vehicle i-axis. Expressing the ω_i in terms of Euler angles and their rates for a ψ (yaw), θ (pitch), and ϕ (roll) rotational sequence, the sun sensor output can be expressed by

$$S_{13} = -\sin\theta$$

$$S_{23} = \sin\phi\cos\theta$$

and for small values of θ and ϕ, the previous equations linearize to

$$M_{1_d} \cong \ddot{\phi} I_1 + h\dot{\psi} - M_{1_c}$$

$$M_{2_d} \cong \ddot{\theta} I_2 - M_{2_c}$$

$$M_{3_d} \cong \ddot{\psi} I_3 - h\dot{\phi} - M_{3_c}$$

and

$$S_{13} \cong -\theta$$

$$S_{23} \cong \phi$$

(The gravity gradient torques have been neglected in this discussion but will be considered as a disturbance torque.)

Requiring that

$$M_{1_c} = M_r \cos\alpha$$

$$M_{3_c} = M_r \sin\alpha$$

(where α = cant angle of roll-yaw jets and M_r = torque delivered by roll-yaw jets)

This yields the pitch and roll-yaw block diagrams of Figs. 108 and 109, where S represents the Laplace transform variable. The pitch motion is decoupled for small angles and its compensation with a lead-lag network is straight forward. Since on-off

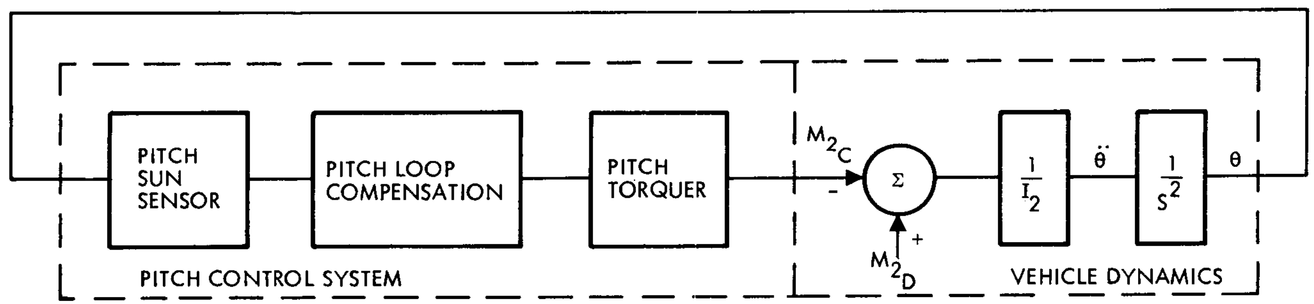


Fig. 108 Pitch Control System and Dynamics

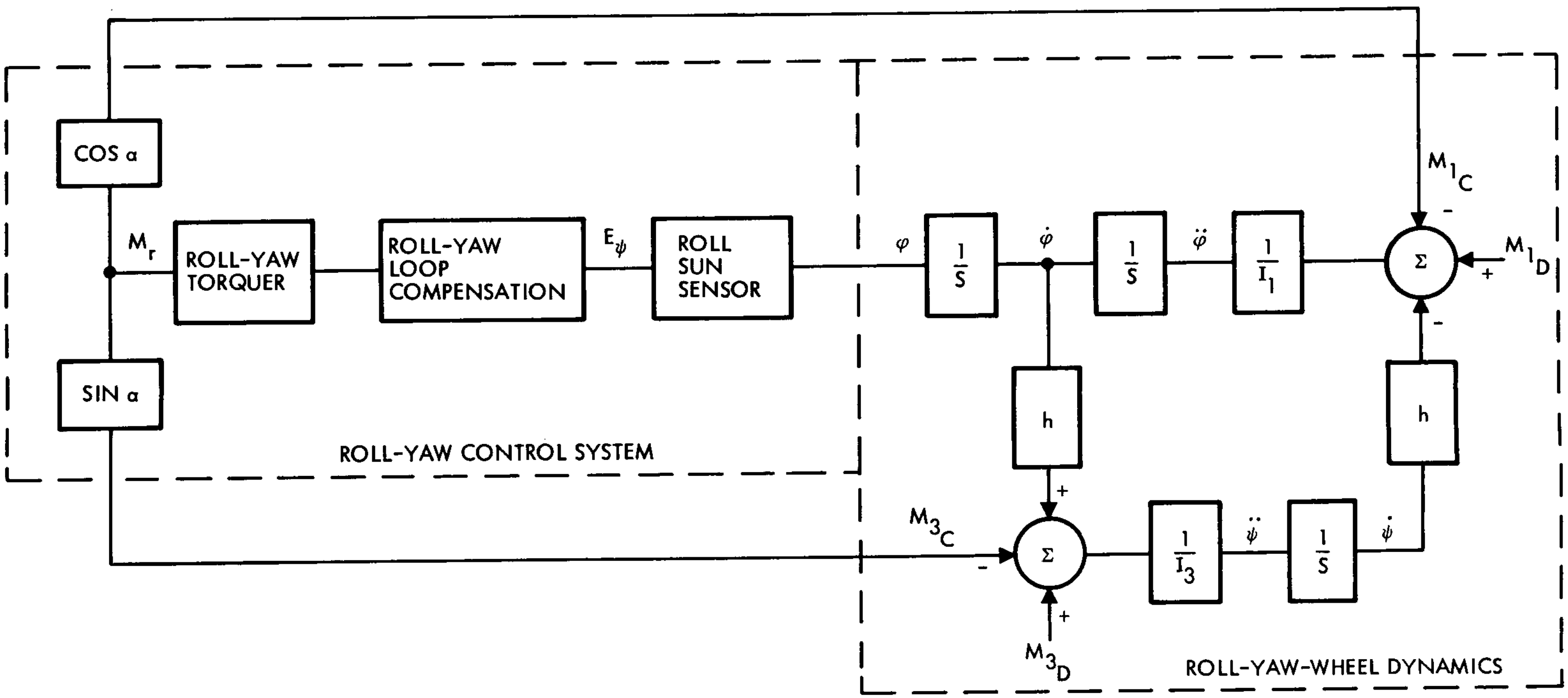


Fig. 109 Roll-Yaw Control System and Dynamics

reaction jets are to be used, an equivalent nonlinear compensation network (the pseudo rate network) is recommended. The resulting pitch control system is shown in Fig. 110.

While the pitch sun sensor is shown as one block, it will actually consist of four solar cells which are arranged to give a 2π steradian field of view and shade two of the sensors from the earth's albedo. The valves labeled 1a, 2a, 3a, and 4a in series with 1, 2, 3, and 4 respectively can be used to increase the system reliability and eliminate the possibility of losing all fuel in the event one valve fails open. The auxiliary valves can be continuously operated or held open to be closed if the main valve fails in the open position.

The roll-yaw control system will be identical to the pitch system except for gain changes. That such a control loop with offset roll-yaw reaction jets will control both roll attitude and yaw rate is illustrated in the next section.

3. Linearized roll-yaw stability – The transfer function between M_r and ϕ (Fig. 109) is readily shown to be

$$\frac{\phi(s)}{M_r(s)} = \frac{\cos\alpha\,(s - h\tan\alpha/I_3)}{I_1 s\left(s^2 + h^2/I_1 I_3\right)}$$

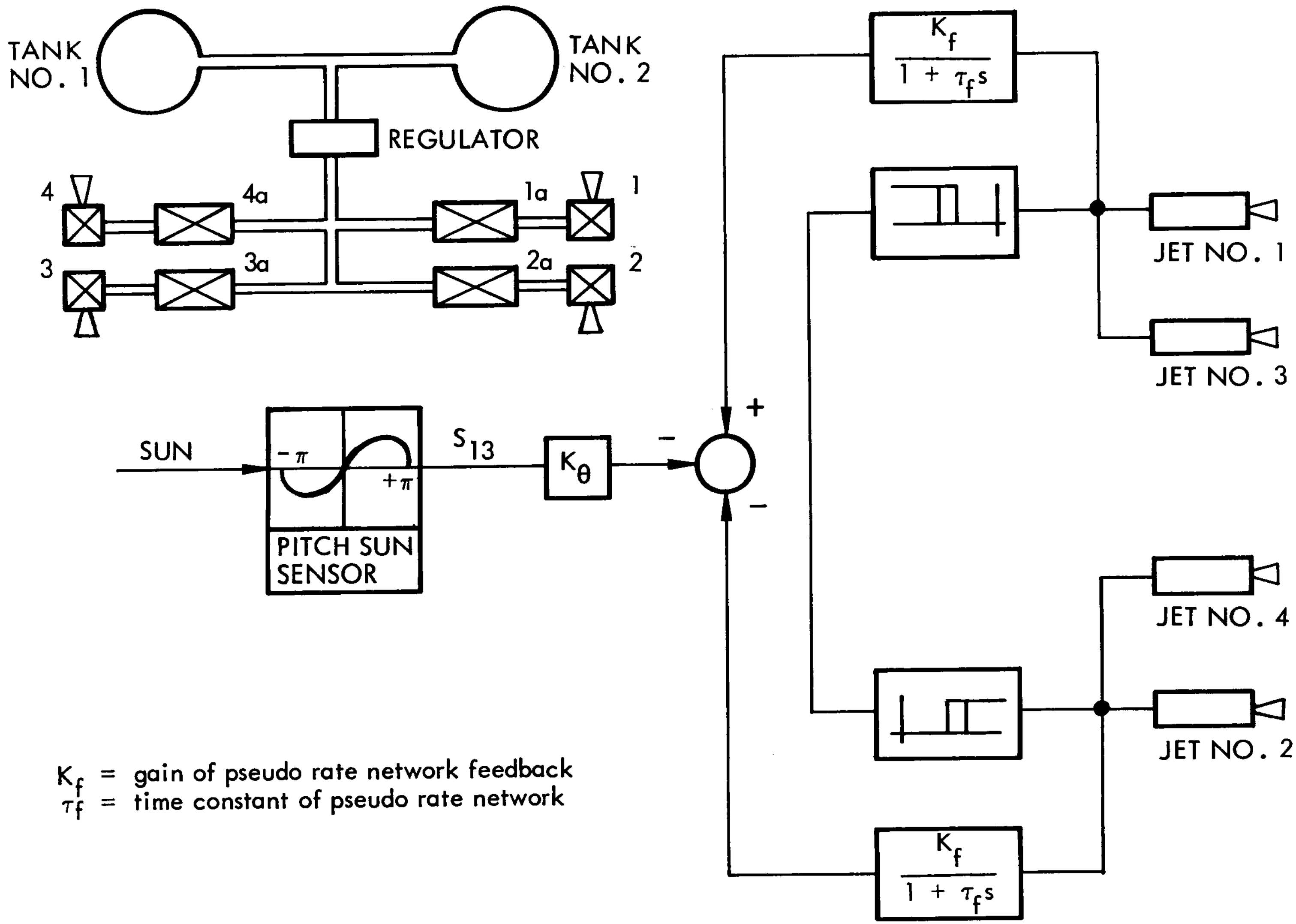
A plot of the poles and zero for the above equation is shown in Fig. 111.

For stability it is clear that a negative value for α (greater than - 90 deg) is required and that a lead-lag network (indicated by dashed zero and pole) of the form

$$\frac{M_r(s)}{E_\phi(s)} = -K_\phi \frac{(1 + \tau_1 s)}{(1 + \tau_2 s)}$$

where K_ϕ is the static control torque per unit sun sensor error, will stabilize the coupled roll-yaw system.

Neglecting the sensor lag, $E_\phi(s)$ may be equated to $\phi(s)$. Since the system has been shown to be stable (for the proper selection of time constants τ_1, τ_2, cant angle α and gain K_ϕ), the steady state values of roll error, ϕ_{ss}, and yaw rate, $\dot\psi_{ss}$, for step disturbance torques of amplitude M_{1d} and M_{3d} can be shown to be



K_f = gain of pseudo rate network feedback
τ_f = time constant of pseudo rate network

Fig. 110 Pitch Control System

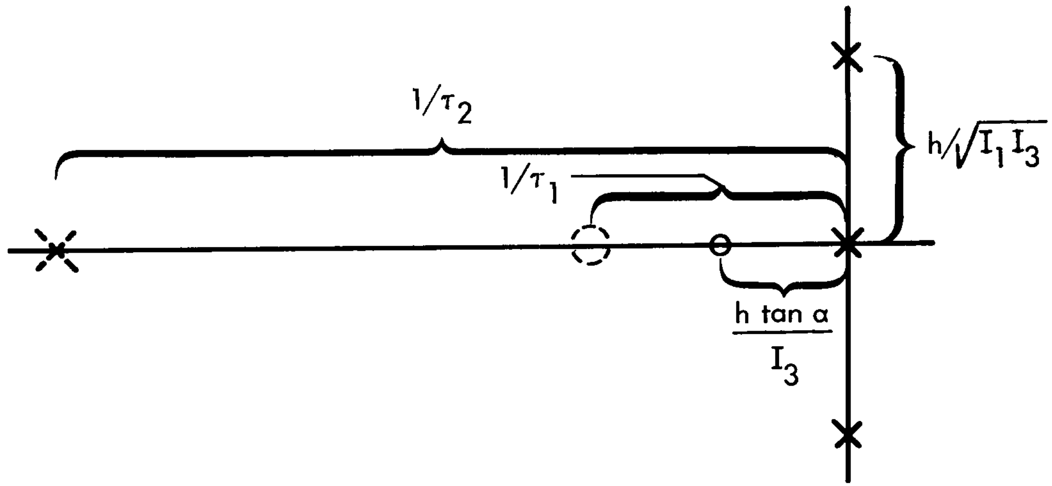


Fig. 111 Pole-Zero Relationship of Open-Loop System

$$\phi_{ss} = - M_{3_d} \Big/ K_\phi \sin \alpha$$

$$\dot{\psi}_{ss} = \left(M_{3_d} \cot \alpha - M_{1_d} \right) \Big/ h$$

The values of α , K_ϕ , and h must be selected to meet the stability requirements (damping ratio), the above equation, and large angle acquisition. While the exact specification of α and K_ϕ must be a subject for a detailed analog simulation, the following values are tentatively selected:

$$\alpha = -10 \text{ deg}$$

$$h = 25 \text{ ft-lb-sec}$$

$$K_\theta = K_\phi = 10 \text{ ft-lb/rad}$$

4. Passive control during satellite night – Each satellite night the sun sensor will lose signal and the reaction jet system will be dormant. The roll and yaw motion will be restrained by the momentum wheel such that the maximum roll and yaw rates under the influence of a constant disturbance of 7×10^{-4} ft-lb (the sum of the maximum expected roll and yaw torques, see Table 38) for a 25 ft-lb-sec wheel momentum would be

$$\left(\dot{\psi}_{max}\right)_n = \left(\dot{\phi}_{max}\right)_n = 0.28 \times 10^{-4} \text{ rad/sec}$$

If this rate were somehow to remain constant for the night time portion of the orbit (2,700 sec) the vehicle would precess only 4.3 deg. Adding to this a 10 deg roll deadband, the maximum acquisition angle in roll will be about 15 deg.

The maximum pitch acquisition initial rate each satellite morning is given by the sum of the limit cycle rate plus the rate imparted to the vehicle by an assumed maximum (constant) disturbance of 3.8×10^{-4} ft-lb (Table 38) or

$$\dot{\theta}_{acq} \cong 0.53 \times 10^{-3} \text{ rad/sec}$$

During the time this rate is being imparted, the vehicle can rotate as much as 32 deg. However, pitch reacquisition each satellite morning can be accomplished in less than 20 seconds with one pound thrusters except in those instances when the animals may have induced large spin rates to the vehicle by spinning themselves up in which case acquisition could take as long as several minutes.

In short, there is no reason to provide active attitude control during the satellite night as long as reacquisition can be readily accomplished. Future "large angle" studies may indicate a need to sequence the recovery maneuver each morning. This would allow the relatively large pitch error to be corrected before initiating roll-yaw control and limit the effects of large angle cross-coupling.

Performance: The minimum on-time of the valves must be large enough to assure a reasonable duty cycle in the presence of disturbance torques and yet small enough to give reasonable fuel expenditure in the absence of disturbances. Expressions for the maximum expected number of cycles by each valve, the disturbance free limit cycle fuel consumption, the disturbance free limit cycle period and rate, and the fuel consumption in the presence of a constant disturbance torque are given by

$$N = \frac{P_m M_d}{F\ell\Delta t}$$

$$W_\ell = \frac{2\,(\Delta t)^2\, F^2 \ell\, P_m}{8\theta_{db} I}$$

$$P_\ell = \frac{4\theta_{db}}{\dot{\theta}_\ell} = \frac{8\theta_{db} I}{F\ell\Delta t}$$

$$\dot{\theta}_\ell = \frac{F\ell\Delta t}{2I}$$

$$W_d = \frac{M_d P_m}{\ell}$$

where

N = number of cycles for one valve (limit cycle) during period of mission (cycles)
P_m = period of mission when reaction jets are used (sec)
M_d = constant disturbance torque (ft-lb)
F = thrust level of reaction jet (lb)
ℓ = reaction jet lever arm (ft)
Δt = minimum valve on time if one jet is being used to produce the control moment $\left(\frac{\Delta t}{2}\right.$ = the minimum on time if two jets are being used to produce a control moment couple$\left.\right)$ (sec)
W_ℓ = fuel consumption in absence of disturbances (lb)
θ_{db} = one-half the total deadband (rad)
I = moment of inertia (slug ft^2)
P_ℓ = undisturbed limit cycle period (sec)
$\dot{\theta}_\ell$ = undisturbed limit cycle rate (rad/sec)
W_d* = fuel consumption in presence of constant disturbance (lb)

*This value could triple if W_d is not at least 2.25 times as much as W_ℓ.

The performance indices are summarized in Table 40 for the limit cycle operation. Fuel consumption and valve duty cycle for acquisition each morning are summarized in Table 41. The total system performance and design indices are summarized in Table 42.

The value of Δt given in Table 40 considers only one jet being used to produce the pitch (or roll) control moment. If two are to be used this time must be cut in half to prevent a factor of four increase in W_ℓ (the zero torque fuel consumption). This reduction in the minimum on-time however will double the maximum valve duty cycle. Consequently, it is proposed that only one jet be used to generate a plus pitch moment and one be used to generate minus pitch moment (valves 1 and 2 in Fig. 110). If either of these fail, valve 3 or 4 (as the case may be) would be switched in and the failed valve switched out by an auxiliary valve (1a or 2a).

TABLE 40

LIMIT CYCLE PERFORMANCE INDICES FOR DAYLIGHT PORTION OF ORBIT

	Pitch	Roll-Yaw
N	15,300 cy	20,000 cy
Δt	0.080 sec	0.14 sec
ℓ	8.34 ft	6.16 ft
θ_{db}	10 deg	10 deg
$\dot{\theta}_\ell$	28.7 deg/hr	41 deg/hr
P_ℓ	1.4 hr	0.96 hr
W_ℓ	500 lb-sec	1,280 lb-sec*
W_d	1,220 lb-sec	2,810 lb-sec

*This is based on the conservative assumption that the roll and yaw disturbances (Table 39) are additive.

TABLE 41

ACQUISITION PERFORMANCE FOLLOWING PASSIVE NIGHTTIME CONTROL

	Pitch	Roll-Yaw
W_d	780 lb-sec	1,930 lb-sec
N	8,900 cy	12,500 cy
Acquisition Time	<20 sec	<50 sec

TABLE 42

CONTROL SYSTEM SUMMARY DATA

Control System Power	5 w
Cold Gas Tank Volume	8,400 $in.^3$
Volume of Electronics, Valve, and Sensors	50 $in.^3$
Maximum Cycles Per Valve	32,500 cy
Maximum Fuel Consumption	6,740 lb-sec
Total System Weight* (Exclusive of Tanks)	110 lb

*Not including the 25 ft-lb-sec momentum wheel (cabin fan).

The maximum fuel consumption figure (Table 42) includes a 10 percent loss in the control system fuel expenditure efficiency during acquisition. Assuming a specific impulse of 68 sec for nitrogen, the fuel will weigh 100 lb, which is the recommended fuel loading. It should be noted that this provides a zero fuel load contingency which is more than justified in light of the conservative assumptions made in the fuel consumption calculations namely (1) the nonphysical assumption that maximum RMS gravity gradient torques can continuously act about all three axes, (2) the maximum RMS aero torques can continuously exist about all three axes, (3) the gravity gradient, aero, solar, and magnetic torques continuously add in a worst-on-worst manner, (4) the coupled roll and yaw disturbance torques are additive when computing the roll-yaw axis fuel consumption, and (5) the magnetic field vector is continuously in the plane of each onboard current loop. Hence, in reality, a fuel contingency of roughly 30 percent exists to account for added fuel expenditure due to animal movement and leakage. The exact contingency can only be determined by a very detailed analog or digital simulation.

Electric Power subsystem. – This section covers the requirements, analysis and definition of the spacecraft power subsystem. The first subsection evaluates the detailed requirements to be satisfied by the power subsystem. The second subsection deals with the analysis and design of the subsystem itself.

Power subsystem requirements: Two primary factors govern the power subsystem requirements: (1) power load and duty cycle and (2) orbital characteristics.

1. Power load and duty cycle – Table 43 is a list of the installed electrical loads and a description of the duty cycle associated with each piece of equipment. A typical "normal" power profile is shown in Fig. 112. This power profile can be expected during most of the mission when the spacecraft and primates are functioning normally. A typical worst case power profile is shown in Fig. 113. This profile

TABLE 43

INSTALLED ELECTRICAL LOADS

Major Subsystems	Continuous Power (Watts)	Peak Power Watts	Peak Power Duty Cycle	Peak Power AC/DC
Data Management:				
Near-field receiver and demodulator	1			28V DC
Activity counters	8			28V DC
Signal conditioner	25			28V DC
PCM timing equipment assembly	15			28V DC
Vocalization	1			28V DC
Signal processor assembly	0	12	20 sec every 12 min	28V DC
Data storage (tape recorders)	2.2	24.7	7 min record and 7 min playback every 20 min	28V DC
Update link	10			28V DC
TV camera	0	65	17 min/day and 25 min/day	28V DC
Transmitter/receiver	26	0		28V DC
S-band amplifier/diplexer	0	65	7 min every 20 min	28V DC
X-band transceiver	0	70	Last 12 hr of the mission	28V DC
Thermal and Atmosphere Control:				
High-flow fans (2)	20			400 Hz 28 VDC
Controls	24			
Low-flow fans (2)	42			400 Hz

TABLE 43 (Cont.)

Major Subsystems	Continuous Power (Watts)	Peak Power		
		Watts	Duty Cycle	AC/DC
Thermal and Atmosphere Control (Cont.)				
Glycol pump	18			400 Hz
Purge fan		200	20 min/day	400 Hz
A-C losses	25			
Animal Support Systems:				
Feeder solenoid valves (2)	0	16	< second	28 VDC
Solenoid valves (2)	0	16	< second	28 VDC
Psycho panels (2)	6			28 VDC
Psycho panel lights (2)		18	Intermittent, 5 min	400 Hz
Mass Measurement				
Ambient lights (Fluorescent)		14	14 hr/day	400 Hz
Attitude Control:	10			28 VDC
Miscellaneous:	19	24		
Solar cell degradation allowance	23			
TOTAL	275.2	524.7 Peak	799.9 Installed	

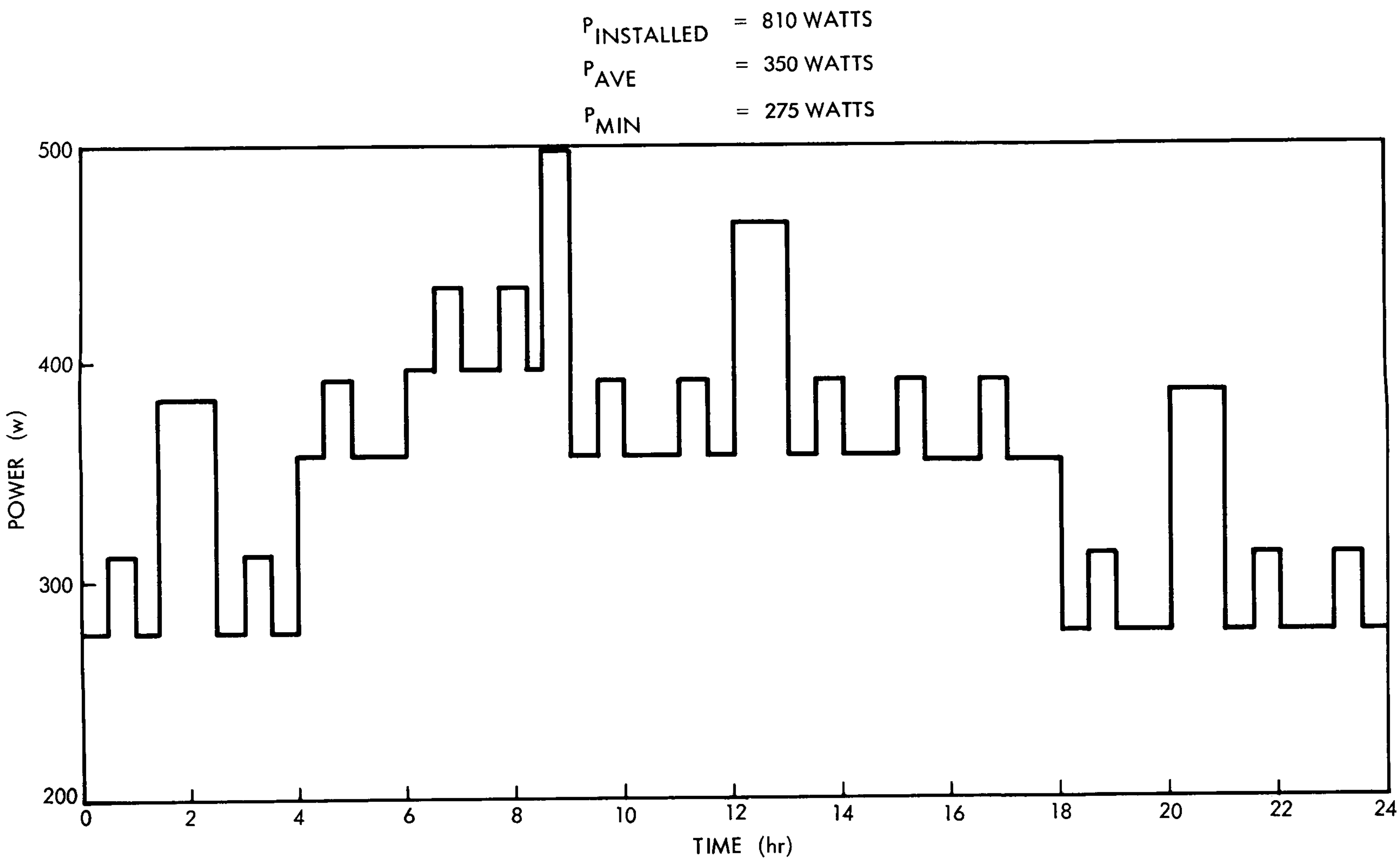


Fig. 112 Typical "Normal" Power Profile

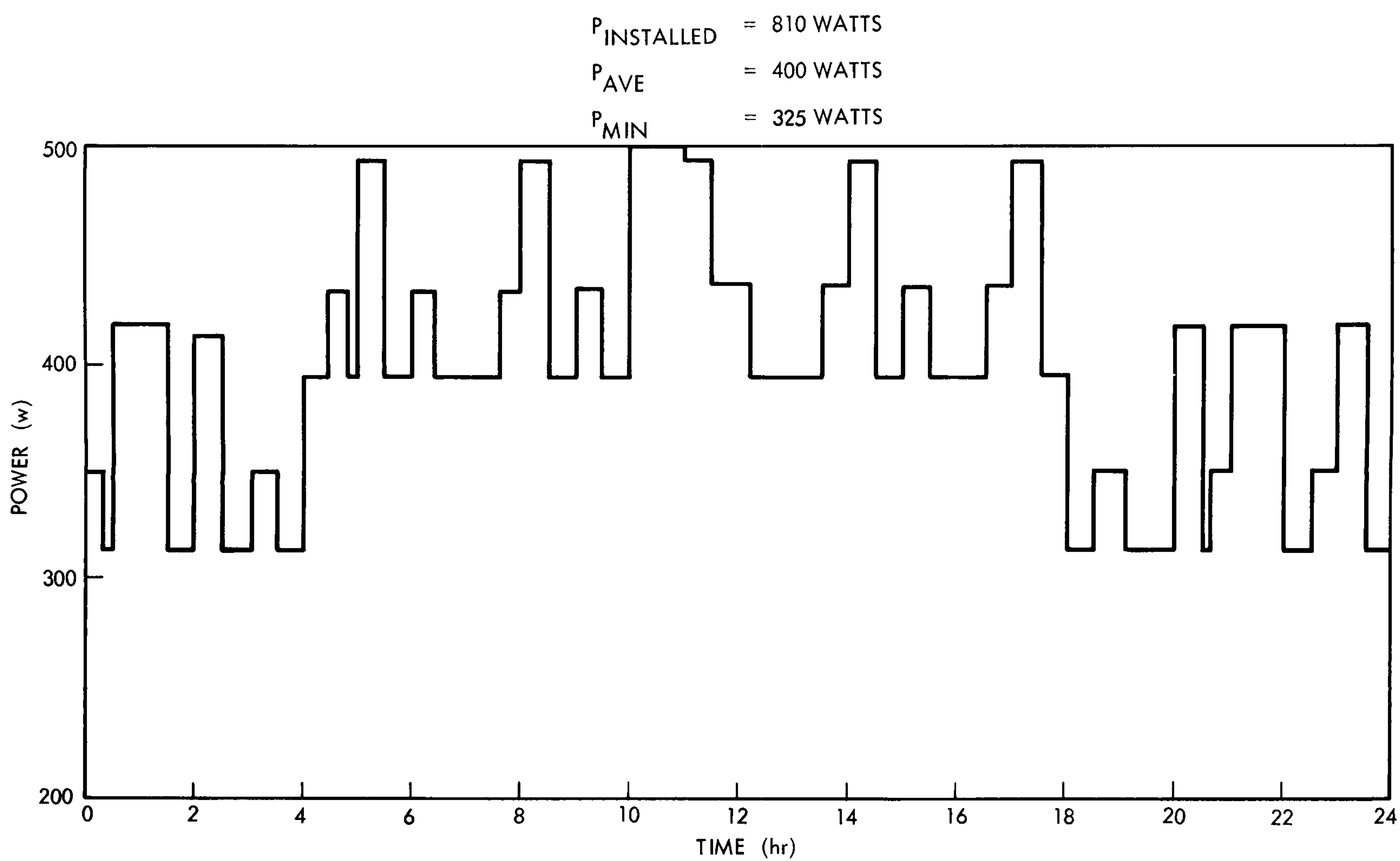


Fig. 113 Typical Worst Case Power Profile

reflects that portion of the mission when either of the primates is functioning abnormally and continual recording and playback of data is required, (i.e., the tape recorder is on continuously). This 400 watt average power was used as the design point for the selected power system.

2. Orbital characteristics – The spacecraft will be in an initial orbit of 260 n. mi. and a 29 deg inclination. If the orbit decays to about 150 n. mi. at the end of a year, then a 38-minute eclipse time results. This worst case was used in the solar power system sizing. The solar array was sized to charge the battery and meet the average electrical load as derived in Table 44.

Power subsystem design: The design of the solar power subsystem is divided into three major categories: solar panel design; secondary battery design; and, power regulation.

1. Solar panel design – This subsection describes the OPE solar panel design including information on the selected solar cell; the array structure; panel output voltage vs. current characteristics; and, temperature, ionizing and UV radiation and micrometeoroid effects.

2. Solar cells – The silicon solar cell is the only photovoltaic device that has been practically used as an electrical power source on board spacecraft. The solar cell used is an N/P 10 ohm-centimeter silicon solar cell with a 2 × 2 cm dimension and 12 mil thickness.

3. Solar panel structure – The aluminum honeycomb structural substrate, used on various spacecraft programs at LMSC and on the Surveyor and Nimbus spacecraft, has proven to be highly successful. Solar panel specific weights using this structure are approximately 0.9 lb/ft^2. The corrugated sheetmetal structural substrate has been used in Mariner and Ranger. The Mariner system weighed about 1.5 lb/ft^2. Die cast magnesium panels weighing about 1.5 lb/ft^2 have developed and flown by LMSC on various spacecraft programs. The LMSC, honeycomb solar panel will be used on the structural substrate for the OPE solar cell system. A breakdown of the solar panel weight is given in Table 45.

4. Solar panel electrical output – Cell packaging density on the LMSC honeycomb solar panel is 208 cells per square foot or a cell packing factor of 0.90. With nominally 10 percent efficient (at 1 sun illumination and 77°F) silicon solar cells, panels are fabricated having performance shown in Fig. 114. The current-voltage (IV) curve shown in Fig. 114 reflects packing, diode, wiring and solar cell cover transmittance losses. The solar panel output of 11.2 watts per square foot at air mass zero (AMO) and 77°F has been used throughout the preliminary design effort. Solar panel output as a function of its maximum operating temperature is shown in Fig. 115.

In sizing the total array, 208 cells per square foot of assembled array area are used. The average power requirement of 400 watts is translated into a solar array that must be capable of providing 892 watts of power to the spacecraft. Maximum solar panel operating temperature has been calculated to be 150°F for a thermal

TABLE 44

SOLAR ARRAY AND BATTERY POWER REQUIREMENTS

Max Average Spacecraft Load	400 watts
Losses – Wiring and Regulator	16 watts
Losses – Battery Circuit and Controller	16 watts
Battery Charging	460 watts
TOTAL	892 watts

TABLE 45

WEIGHT BREAKDOWN OF TYPICAL SOLAR PANEL

	Weight (lb/ft^2)
6-mil cover glass	0.0685
Cover glass adhesive (2 mil)	0.0115
Solar cell (12 mil)	0.2250
Electrode	0.0247
Solder	0.0248
Epoxy insulation	0.0219
3-mil adhesive	0.0219
1/2-in. thick honeycomb structural substrate	0.4160
Structure deployment	0.0800
Total for deployed array	0.8943

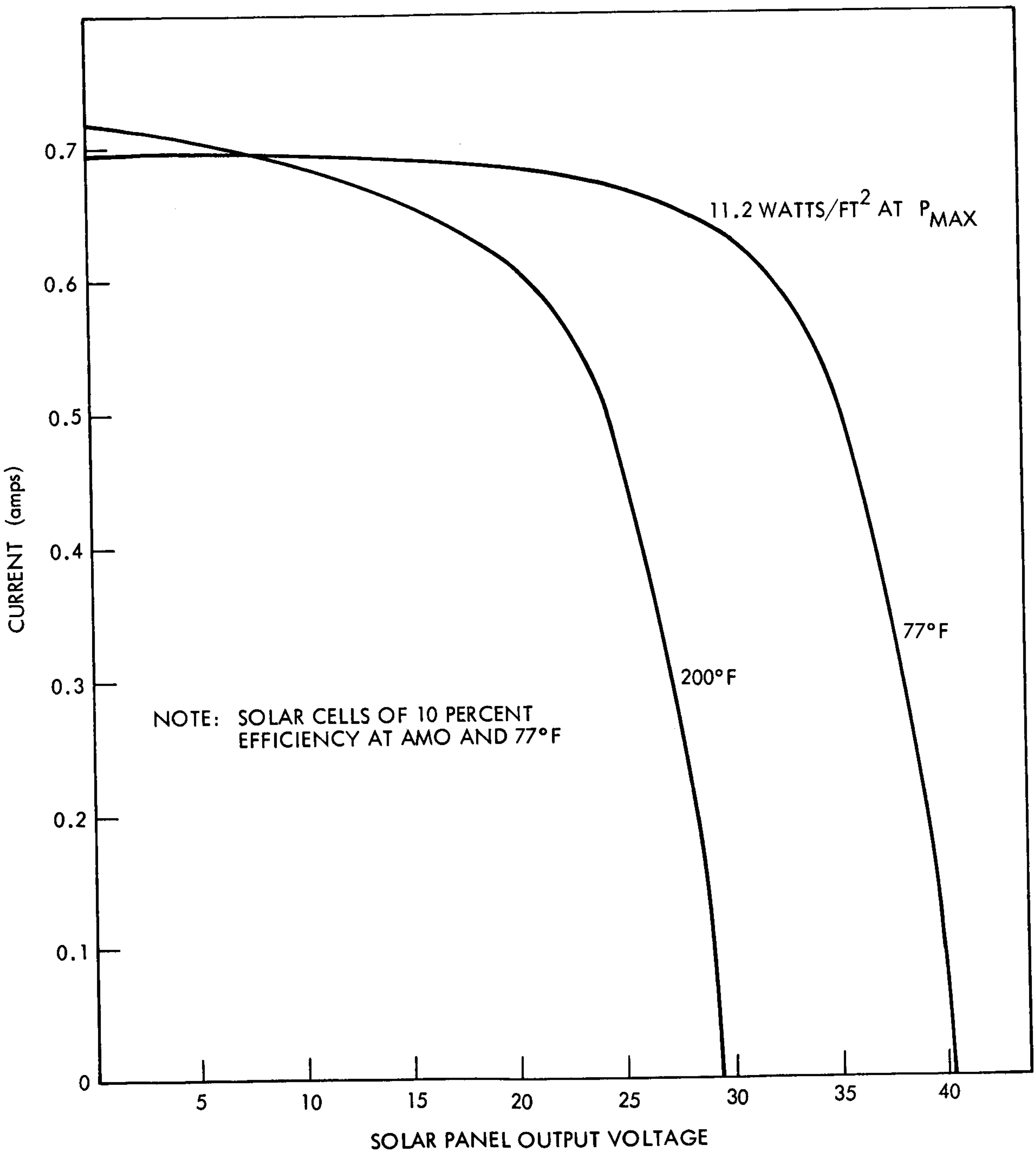


Fig. 114 Performance of Typical Solar Panel

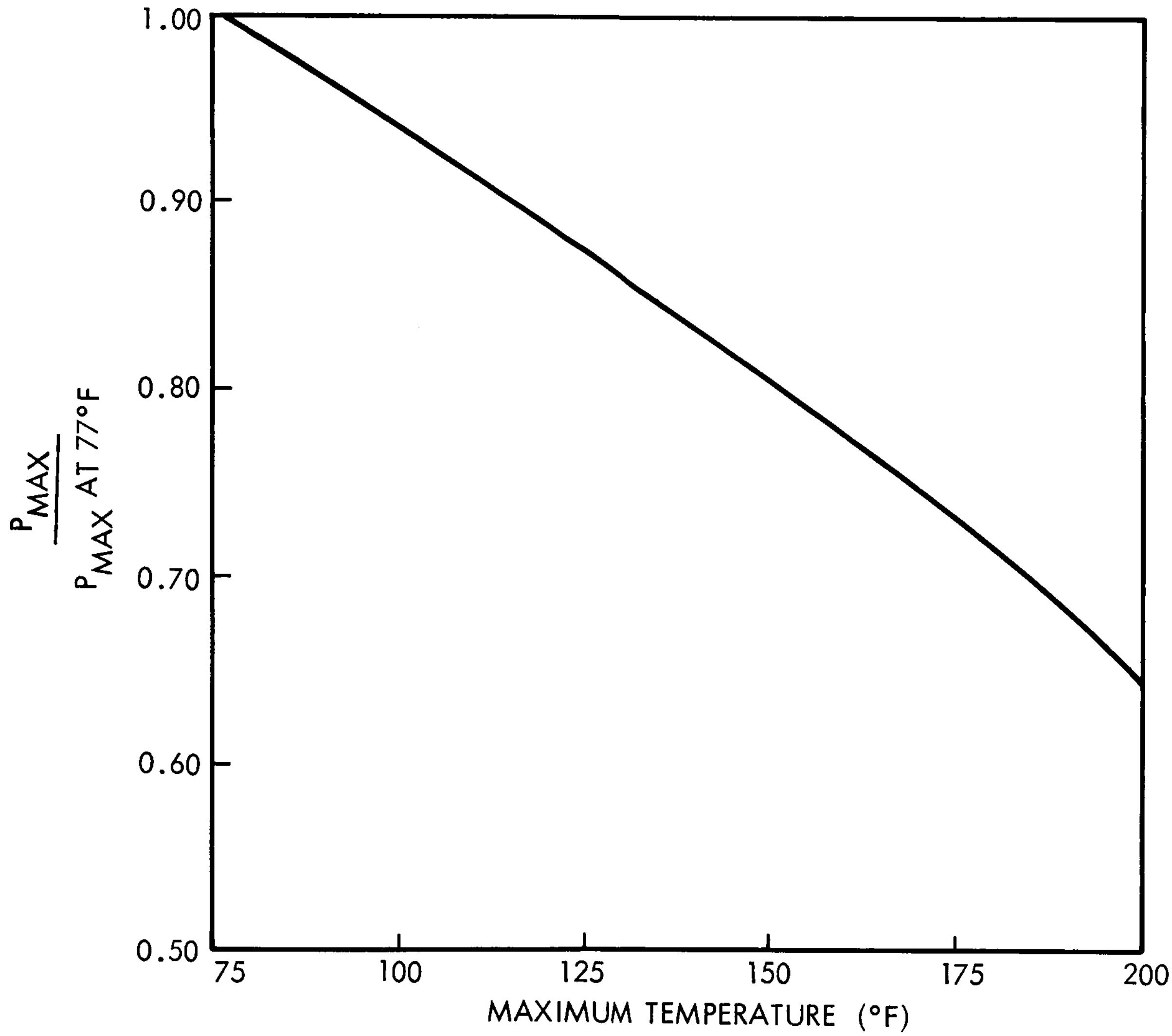


Fig. 115 Solar Panel Output Vs. Maximum Temperature

absorptivity-emmissivity ratio (α/ϵ) of solar cell coverglass of 1.125. The solar panel will be capable of providing 9.0 watts/ft^2 at this operating temperature. Total solar array area requirement is then 100 ft^2. Approximately 21,000 2 $\times$ 2 cm solar cells will therefore be required.

The cells will be assembled in submodules containing 45 cells each, nine connected in series and five in parallel. Nominal output of each series-parallel connected submodule will be 0.630 amps at 3.0 v. Thus, 10 submodules must be connected in series to obtain the cutoff voltage of 30 v and 47 connected in parallel to obtain the 892 watt power requirement. For the solar array, 470 submodules will be required.

5. Solar panel degradation – Ionizing radiation damage to the solar cells is expected to be negligible for this low earth orbit. For the one-year mission period, fluxes of Van Allen protons and electrons are:

Protons	4 Mev	>	3×10^8 P/cm^2
Electrons	0.5 Mev	>	3×10^{11} e/cm^2

The critical flux of N/P silicon solar cells is approximately 10^{13} 1.0 Mev electrons per square centimeter. Hence, radiation damage to the solar panel can be neglected.

Ultraviolet degradation of solar cell cover adhesive will also be slight if a blue filter is incorporated on the cover glass. For a one year period, no more than 2 to 3 percent degradation will occur due to ultraviolet coloring of the adhesive material.

Expected micrometeoroid fluxes of 10^{-5} - 10^{-4} particles/cm^2-sec (3×10^2 - 3×10^3 particles/cm^2-yr) in the 10^{-8} - 10^{-9} gram range are expected in this low earth orbit due to the micrometeoroid belts surrounding the earth and to interplanetary sources. Experimental data of the abrasive effect of these micrometeoroid impacts on solar cell cover slide material is lacking at this time. However, assuming that the density of these micrometeoroids is 0.44 gm/cc, and that the pitted area produced by the micrometeoroid impact is twice the projected area of the micrometeoroid, then the area abraded by the micrometeoroids in a one year period is predicted to be: 3×10^3 particles/cm^2/yr $\times$ (2) (1.8×10^{-6} cm^2/particle) = 0.011. Thus about 1 percent of the active solar area can be expected to erode away by the micrometeoroid impacts. This number although indicative of the magnitude of micrometeoroid degradation to be expected, is however, only as good an estimate as the assumed micrometeoroid distribution, and could differ by as much as an order of magnitude. This analysis, however, does indicate that the micrometeoroid degradation to the solar cells will probably be small.

Table 46 is a tabulation of expected micrometeoroid penetration depth and calculated solar cell degradation. It can be seen in Fig. 116 that 6 mil thick cover glass will easily prevent penetration of the micrometeoroids in the 10^{-9} gram range. Allowance for a solar cell output degradation of 5 percent (45 w) was provided by increasing the basic load by 23 watts.

TABLE 46

MICROMETEOROID EFFECTS ON SOLAR CELLS

M Particle Mass (gm)	Particle Radius (R) and Area (A) at 0.44 gm/cc (Assumes Spherical Shape)		N, Number of Particles of Mass M or Greater (Particles/cm^2/yr)	2 (A × N) (Percent Degradation Per Year)	d, Particle Penetration Depth in Fused Silica Mils (in. $\times 10^{-3}$)
	Radius (mm)	Projected Area (cm^2)			
1.25	8.8	2.3	1.5×10^{-11}	7.0×10^{-9}	1560
0.198	4.8	7.1×10^{-1}	1.8×10^{-10}	2.6×10^{-8}	850
0.012	1.9	1.1×10^{-1}	7.5×10^{-9}	17.0×10^{-8}	330
1.2×10^{-4}	0.40	5.0×10^{-2}	3.7×10^{-6}	3.6×10^{-6}	74
1.2×10^{-6}	0.09	2.5×10^{-4}	1.7×10^{-3}	8×10^{-5}	16
8×10^{-8}	0.036	4×10^{-5}	3×10^{-3}	2.4×10^{-5}	5.7
8×10^{-9}	0.0167	8×10^{-6}	3×10	4.8×10^{-4}	1.6
8×10^{-10}	0.0077	1.8×10^{-6}	3×10^{3}	10.8×10^{-1}	0.74

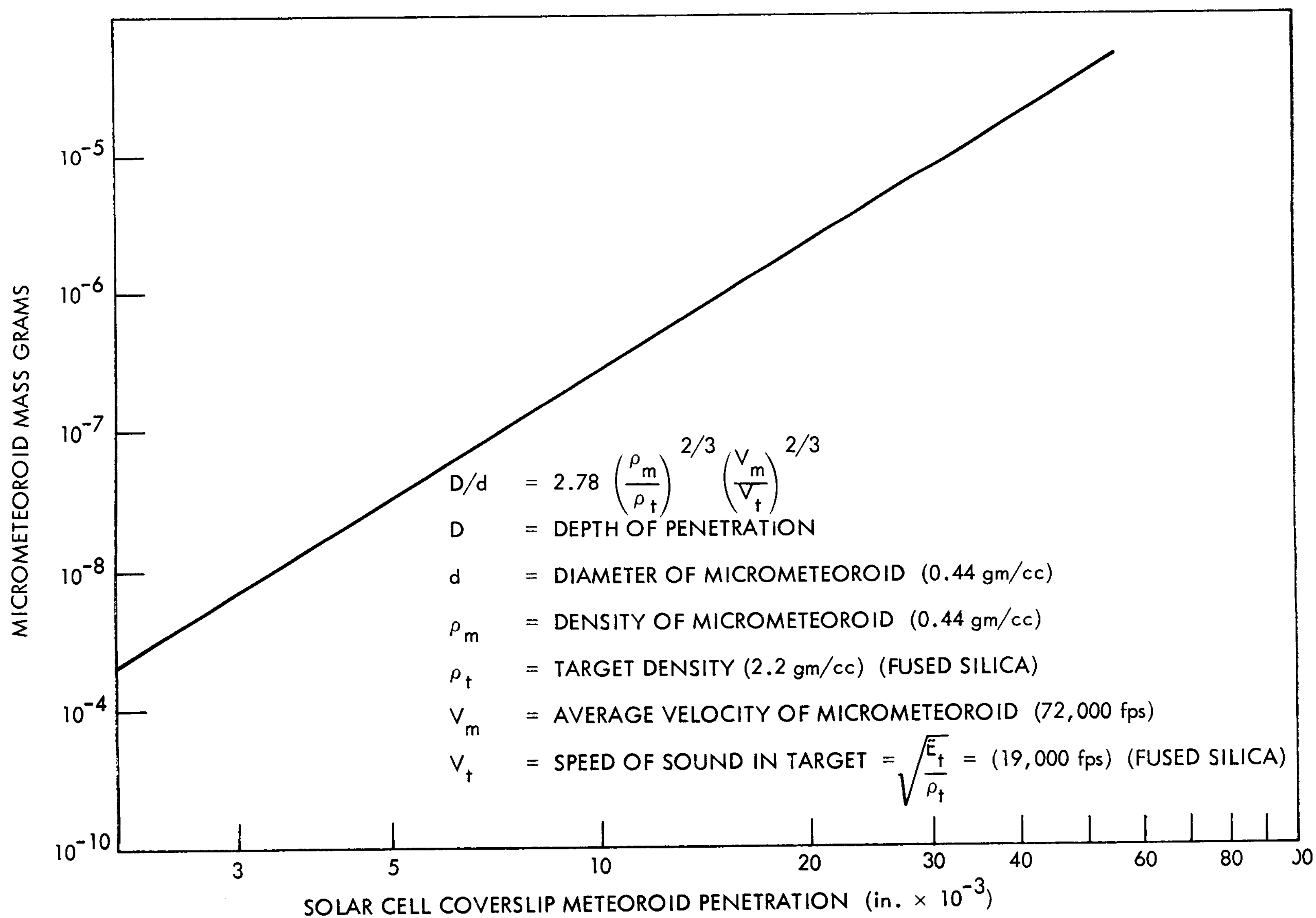


Fig. 116 Micrometeoroid Mass Vs. Depth of Penetration

Secondary battery design: As indicated in Table 44, the secondary energy storage system for the solar array must be capable of supplying 250 w-hr per orbit in order to provide the electrical power requirements during the time when the satellite is occulted from the sun by the earth.

1. Candidate batteries – The most advanced types of secondary batteries available for space applications are the nickel cadmium, silver cadmium and silver zinc batteries. Table 47 summarizes the selection criteria and characteristics of these three candidate secondary batteries. Temperatures as well as other parameters influence the characteristics of these batteries significantly, however, the table is representative of the relative advantages and disadvantages of these types of batteries.

The most important criteria in the battery selection for the OPE, are cycle life and wet stand life. The secondary battery will be required to withstand at least 6000 charge-discharge cycles during the one year mission. Secondly, operating temperature sensitivity and overcharge capability are other factors that will receive prime consideration so that high reliability will be realized.

2. Battery selection – The cycle life requirement precludes the use of silver zinc batteries. Either nickel cadmium or silver cadmium cells can meet the cycle life and wet stand requirements. The nickel cadmium battery is least sensitive to temperature effects and overcharge rates, thereby simplifying the design requirements of the battery charge controller. The nickel cadmium battery is also the more mature system from the standpoint of operational experience. LMSC has used this battery successfully in a large number of spacecraft. Due to the high confidence associated with the nickel cadmium battery, it is selected over the lighter silver cadmium battery.

3. Battery sizing – 250-watt-hr of energy will be withdrawn from the battery during the 0.63 hr shadow time. In order to obtain the 6000 cycle life required, the depth of discharge during this period must be limited to not more than 25 percent, and battery operating temperatures maintained between 40° and 70°F as indicated in Fig. 117.

This stringent battery thermal control requirement, and the availability of an active thermal control loop, led to integrating the battery into the heat rejection loop. An additional advantage of this approach is the stabilizing effect that the battery thermal loads have upon the thermal control radiator.

At 25 percent depth of discharge, the nickel cadmium battery must have a capacity of 1000 w-hr. Using two 900-watt-hr batteries, which permits operating in standby redundance, results in a 14 percent depth of discharge for each battery. The use of redundant batteries improves the reliability of the overall power system. The failure of one battery will not result in the failure of the entire power system, and operation on only one battery will result in an average depth of discharge of 28 percent on the remaining battery.

TABLE 47

SECONDARY BATTERY SELECTION CRITERIA AND CHARACTERISTICS

	Nickel Cadmium	Silver Cadmium	Silver Zinc
Cycle Life	>5000 cycles	>5000 cycles	<5000 cycles
Voltage Regulation	32%	50%	45%
Charge-Discharge Efficiency	65-80%	70-80%	70-85%
Sensitivity to Operating Temperature	Low sensitivity	Moderately sensitive	Very sensitive
Watt hr/lb	12	24	40
Overcharge Capability	Continuous up to 10 hr rate	Limited	Limited
Wet Stand Life	10 yr	2 yr	1 yr

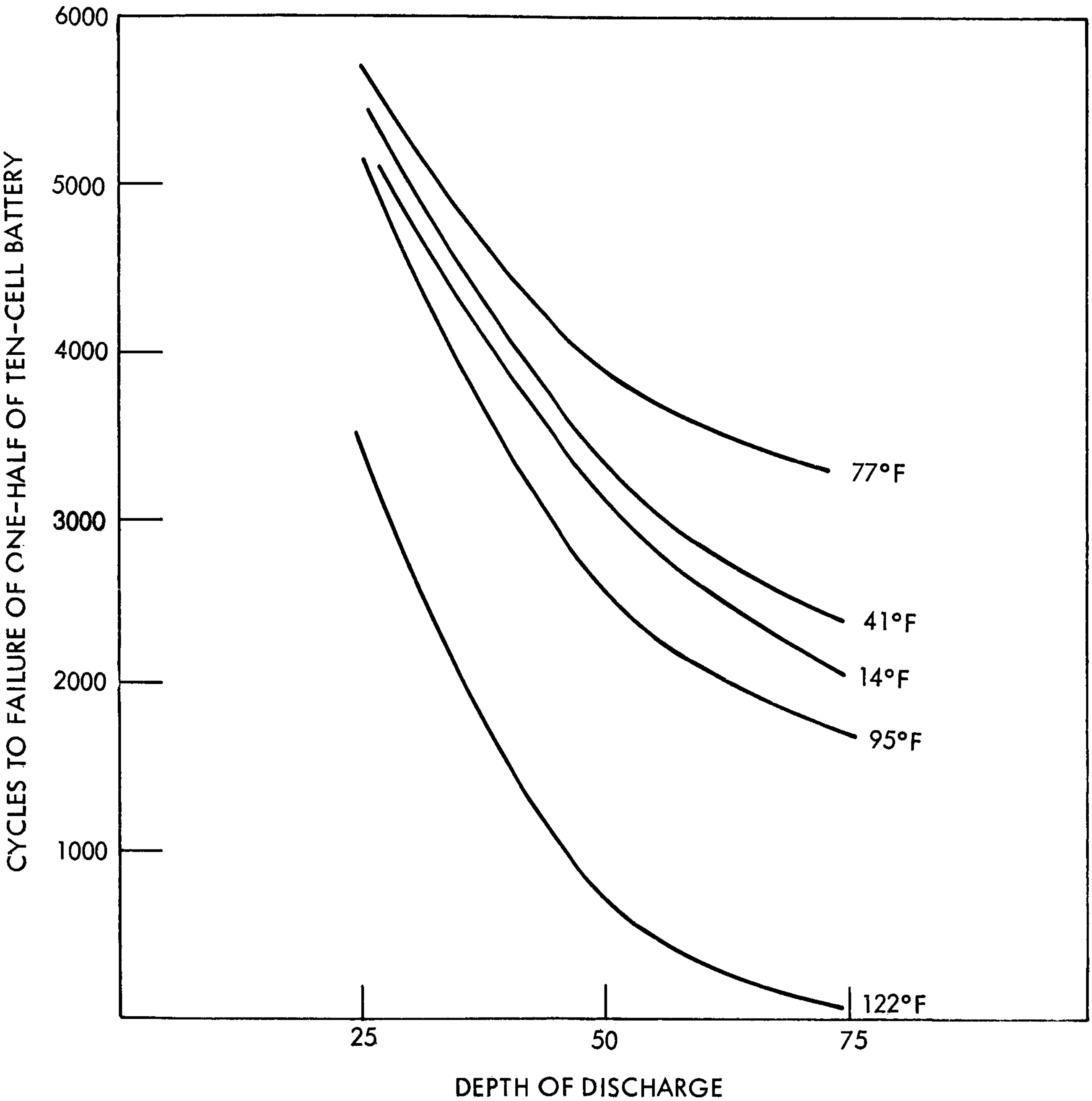


Fig. 117 Nickel Cadmium Battery Operating Life Factors

The following performance can be expected from each of the two required nickel cadmium batteries:

Nominal Capacity (amp-hr)	36
Voltage Range (v)	22–30
Weight (lb)	72
Dimensions (in.)	7 × 7-1/2 × 17
Nominal Charge Rate	~C/3
Nominal Discharge Rate	~C/5

4. Battery charge control – The overcharge condition of the nickel cadmium battery will be monitored by sensing its overcharge voltage. The nickel cadmium battery is characterized by a sharp rise in overcharge voltage at low battery temperature as shown in Fig. 118. At higher battery temperatures, the overcharge voltage is not a good indication of state of overcharge. Therefore, a temperature override has been incorporated that switches to a timer which limits the charging period at higher battery temperatures.

Power regulation: The central power regulation system is required to maintain the main bus within the voltage range of 22 to 30 volts. The low voltage limit is determined by the battery output voltage when completely discharged. Normally, the battery is not discharged below 86 percent of capacity and the bus voltage ranges between 26 to 30 volts. Upper voltage limit of 30 volts is maintained by voltage clipping.

1. Central shunt regulators – The simplest form of regulation is the shunt regulator utilizing the constant voltage characteristics of the Zener diode. Many variations of the Zener diode shunt regulator may be postulated, but in general series-parallel groupings must be used to provide the desired regulation voltages and power handling capability. A schematic of the approach is shown in Fig. 119 . The major drawback of the Zener diode approach is the variation of the "break over" or regulating voltage with variations in Zener diode operating temperature. The inherent reliability of this approach is very high because of the simplicity and passive operation of the system. Attractive failure modes increase the probability of mission success. For example, loss of a Zener by short circuiting will not cause loss of power to the load, the only effect will be to cause the system to regulate at a lower voltage. Likewise, an open circuit failure of a diode will not cause loss of power; only loss of voltage regulation capability will result. However, the power handling capability of this approach is limited, owing to the relatively low temperature at which the parasitic thermal loads must be rejected.

More complex shunt regulators utilizing feedback control and higher temperature heat dissipating resistors are classified either as proportional or pulse width modulation systems as shown in Fig. 120. In the Class A regulator, a Zener diode is used as a reference voltage and compared with the bus voltage. The error signal is used to drive the transistor which proportionately controls the amount of current dissipated by the parasitic resistor. In the pulse width modulated shunt regulator, the transistor is driven in a more efficient switching mode by this error signal making possible a greater power rating for a given component size. A filter must be incorporated in the output of this system to reduce the ripple to an acceptable value.

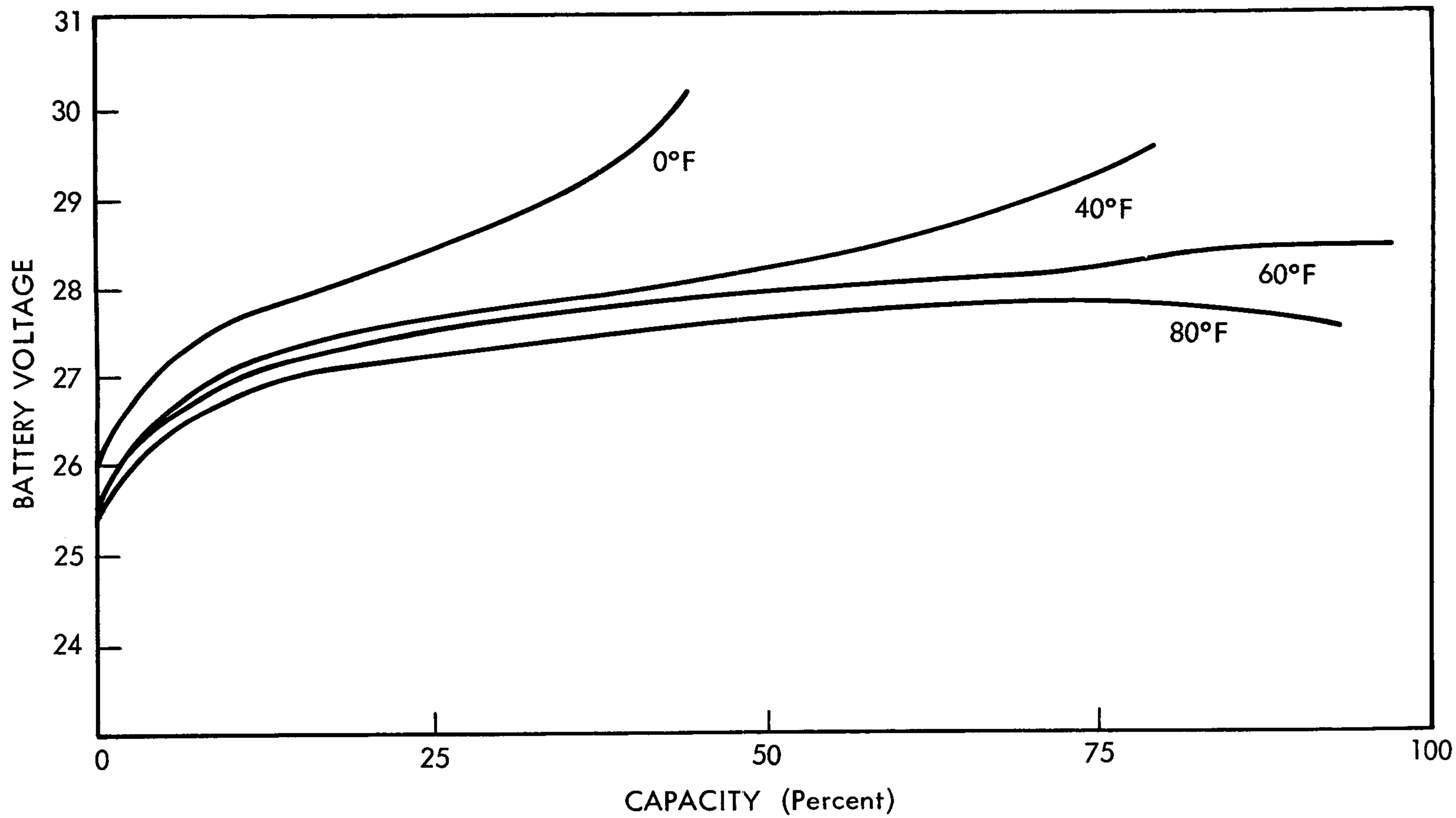


Fig. 118 Nickel Cadmium Battery Charge Characteristics

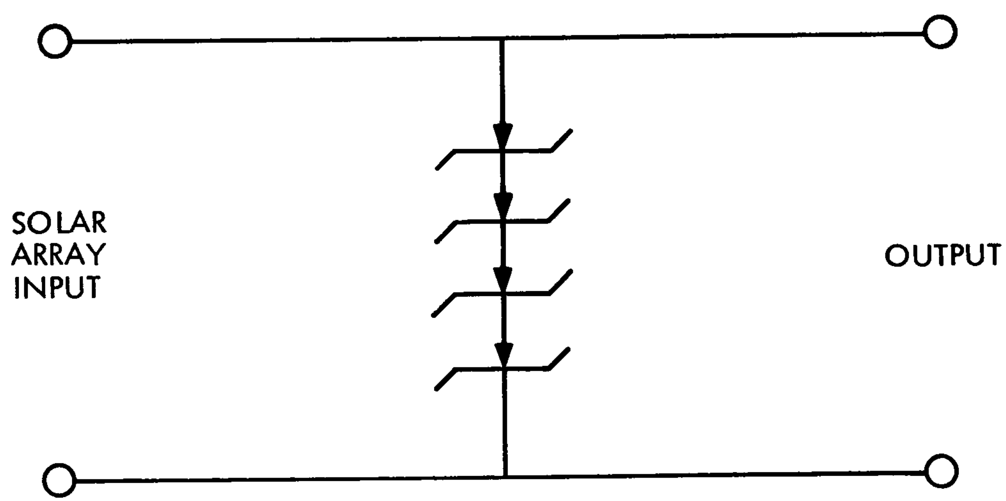
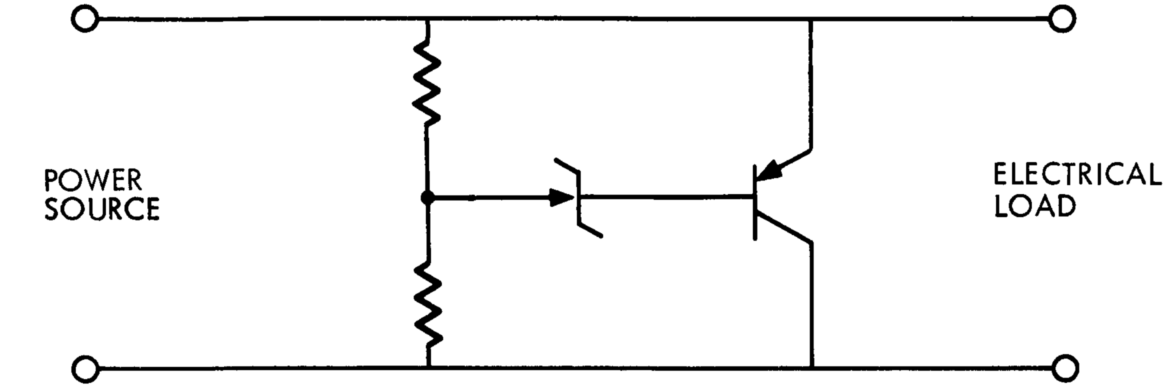
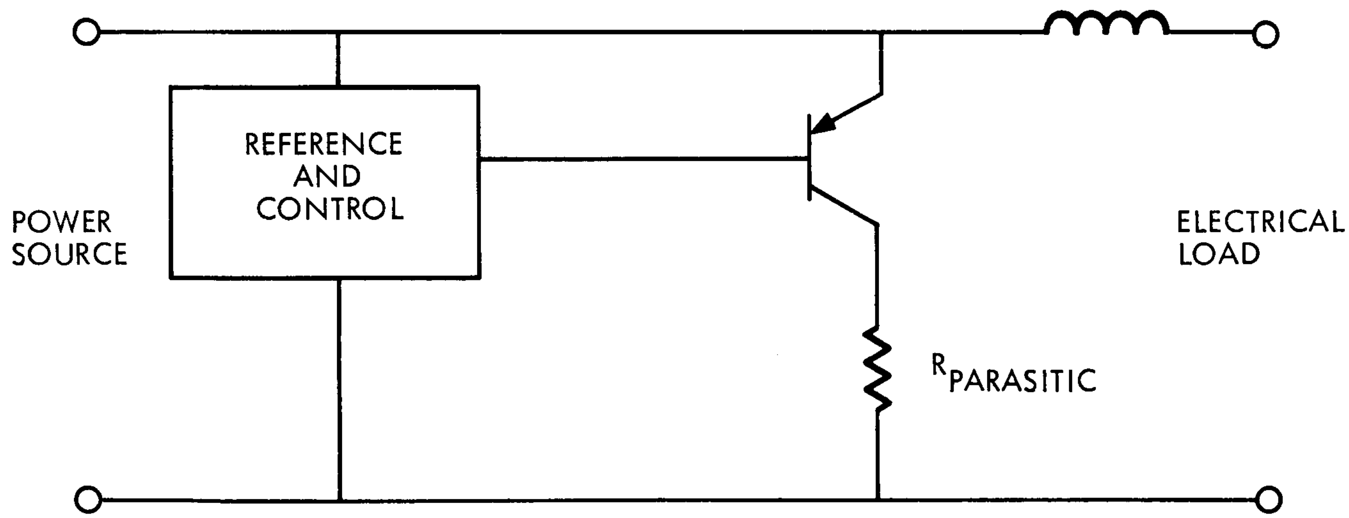


Fig. 119 Zener Diode Shunt Regulator



TYPICAL PROPORTIONAL SHUNT REGULATOR



TYPICAL PULSE WIDTH MODULATED SHUNT REGULATOR

Fig. 120 Central Shunt Regulator

These shunt type regulators require resistive parasitic load banks to dissipate the power not utilized by the load. With the active thermal control system used in the OPE, the shunt regulator becomes an attractive scheme because it can be used to stabilize the thermal loads to the radiator. When the electrical power load demand is low, heat rejected from the life cell is also correspondingly low. In this situation, however, the shunt regulator dissipation is high. On the other hand, when electrical power demands are high, heat rejected from the life cell regulator thermal load into the active life cell heat rejection loop, the radiator load will be, to a certain extent, stabilized.

2. Series regulators – Series regulators are also classified as Class A or pulse width modulated units and basic examples of these two approaches are shown in Fig. 120.

A schematic of the Class A arrangement is shown in Fig. 121. A breakdown of the Zener will tend to turn the transistor off. The primary drawback to this approach is that prohibitively large amounts of power must be dissipated by the transistor. This is particularly true when there are large variations of input power and/or power required. The OPE power profile indicates that power demand will vary from 270 watts to 500 watts. Solar panel temperature variations are also encountered, compounding the power dissipation problems of this approach.

The most desirable approach to series regulation is the pulse width modulation scheme. The on-off time of the series transistor is varied so that the train of pulses, after filtering, will result in the desired voltage.

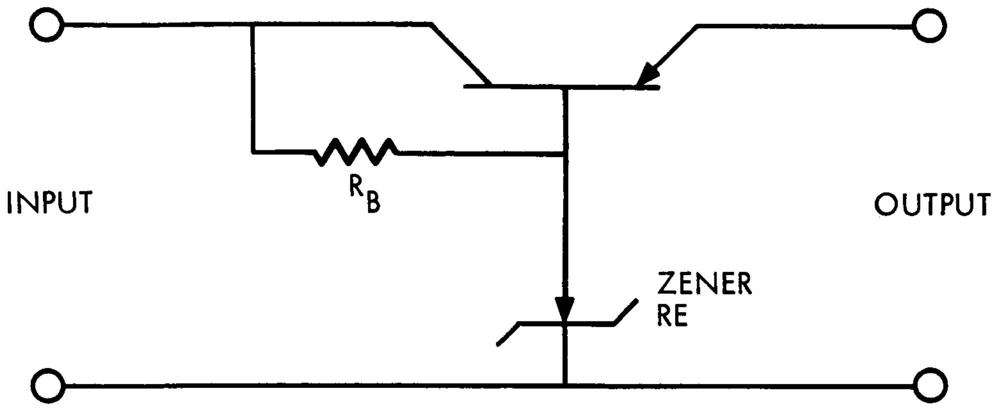
This approach minimizes the power dissipated by the switching element. The power dissipation capability of this approach is however, still somewhat limited. Application of this approach to solar arrays with large temperature variations, hence power input, is not advisable. This approach is best applied to solar cell power systems with relatively stable temperatures and electrical loads.

3. Regulator selection – Although reliability of the series regulator is theoretically the same as that of the shunt regulator, the failure modes of the shunt regulator are more attractive than those of the series regulator. For example, a short occurring in the load could easily burn out the series regulation elements, whereas the shunt regulator would not be affected by either short or open circuits in the loads, and power could be returned to the loads as soon as the fault is cleared.

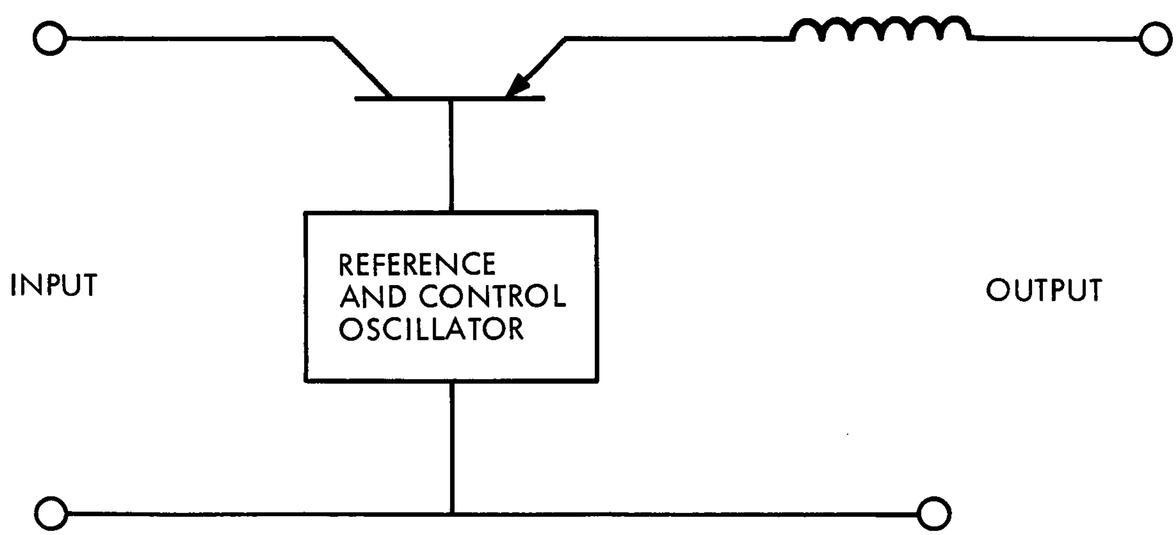
Table 48 summarizes the advantages and disadvantages of the power regulation concepts considered.

The central shunt regulation system is chosen primarily because of its advanced state of development, higher reliability through more attractive failure modes, and because of the advantages it offers for stabilizing the thermal inputs to the active thermal control system.

A schematic of the complete electrical power conditioning and distribution system is shown in Fig. 122.



CLASS A SERIES REGULATOR



PULSE WIDTH MODULATED REGULATOR

Fig. 121 Series Type Voltage Regulators

TABLE 48

REGULATOR STUDY SUMMARY

Regulation Scheme	Weight (lb)	Thermal Dissippation	System Compatibility	Comments
Zener Diode Shunt	8	Zener Diode	Good	• Limited power dissipation capability • Sensitivity to temperature variations
Proportional Shunt	6	Parasitic Load Resistor	Excellent	• Easily integrated with ECS radiator to stabilize thermal loads
PWM Shunt	6	Parasitic Load Resistor	Good	• Heavy output filter required • Possible EMI problems
Multiple Shunt	4	Parasitic Load Resistor	Good	• Developmental item only • No flight experience
Series Proportional	6	Solar Array and Transistor	Poor	• Severely limited power dissipating capability
PWM Series	6	Solar Array and Transistor	Average	• Limited power dissipating capability

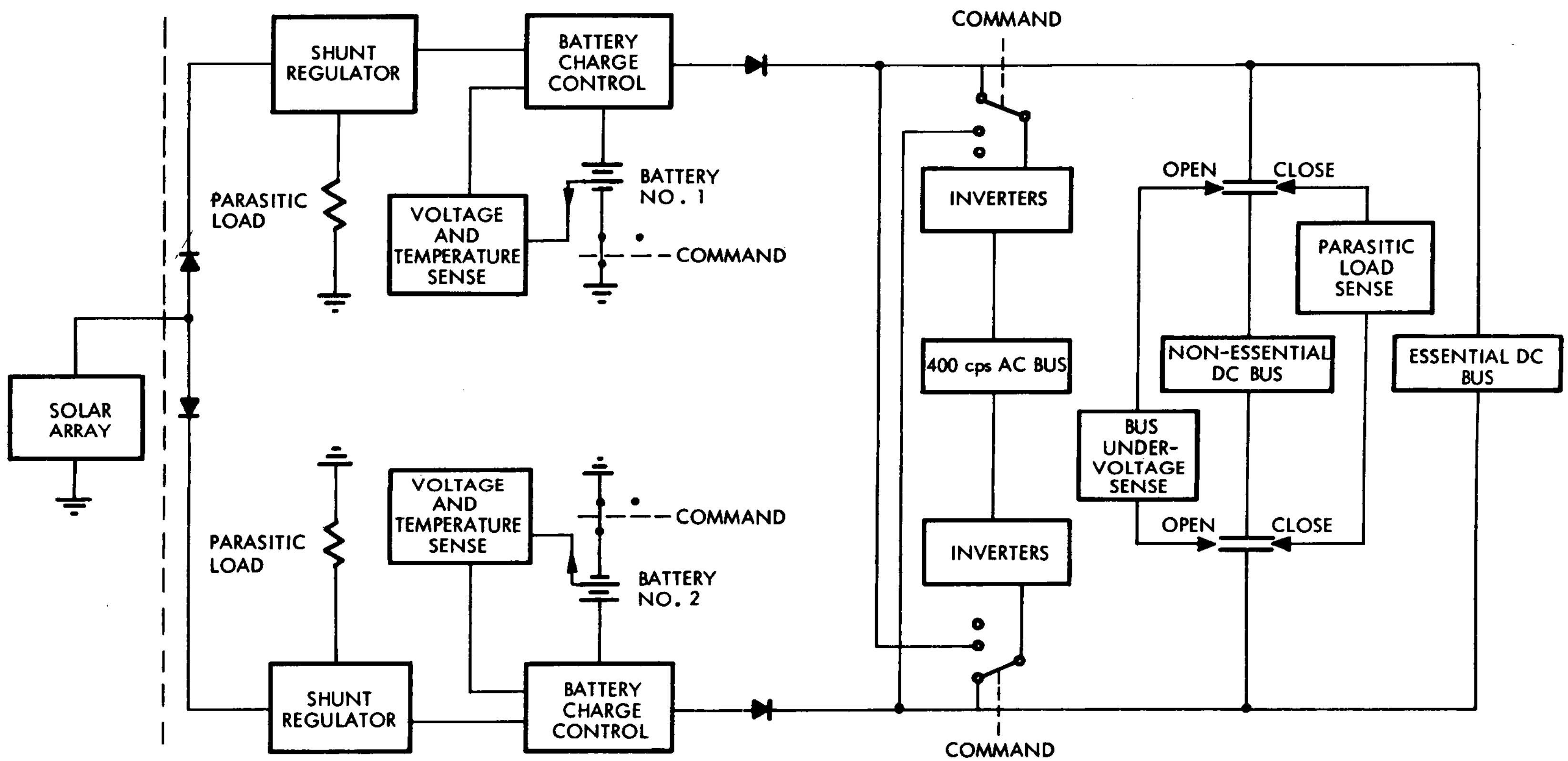


Fig. 122 Power Subsystem Schematic

Spacecraft General Arrangement

The following paragraphs describe the OPE spacecraft general arrangement. The lifecell and its contents are covered in another subsection. Included herein are discussions on the configuration, construction, and servicing features. The provisions for Special Operations such as primate insertion on the pad, remote deployments, and retrieval are also included. A summary of all pyrotechnics is made.

Configuration. – The configuration geometry and mass properties are discussed below.

Geometry: The independent spacecraft is shown in the launch condition in Fig. 123. The center body consists of a cylinder combined with two truncated cones.

The overall dimensions are 148 in. long by 128 in. diameter. The truncated cones, which have a 30-deg half-angle, are 74 in. in diameter on the small ends and are 46.8 in. in height. The cylindrical length is 54.4 in. Protruding from opposite sides of the spacecraft are two extended solar arrays which are 148 in. long and have a maximum span of 223.8 in.

Mass Properties: The center of gravity of the spacecraft is located on the longitudinal (Z) axis 74 in. from either end (Z = 0.0). The maximum predicted center of gravity excursion is 2.0 in.

Control of this excursion is based on subdividing expendables into units which are either balanced about the spacecraft center of gravity (in the case of expendables which are expelled from the spacecraft) or balanced about the position of ultimate residence (in the case of expendables which are not expelled from the spacecraft). The usage of expendables from these subdivided units is then controlled to maintain balance. Drinking water is stored in two equally sized tanks on opposite sides of the spacecraft and balanced about the c.g. Usage is monitored and can be switched from one or both tanks to the other in case of unequal usage rates. Furthermore, one tank is inverted because of the lengthy cylindrical portion.

Each retrieval capsule has two preservative tanks which service it. These tanks are balanced about the retrieval capsule. If the preservative is used, even for only one monkey, it proceeds to the retrieval capsule and causes no c.g. shift to the spacecraft. The expulsion of preservative just prior to retrieval of a dead monkey(s) would produce a c.g. shift, but since it would only be for a short period of time, i.e., 2 days, the effect on attitude control fuel consumption would be negligible. The important thing is to prevent long-term c.g. shifts.

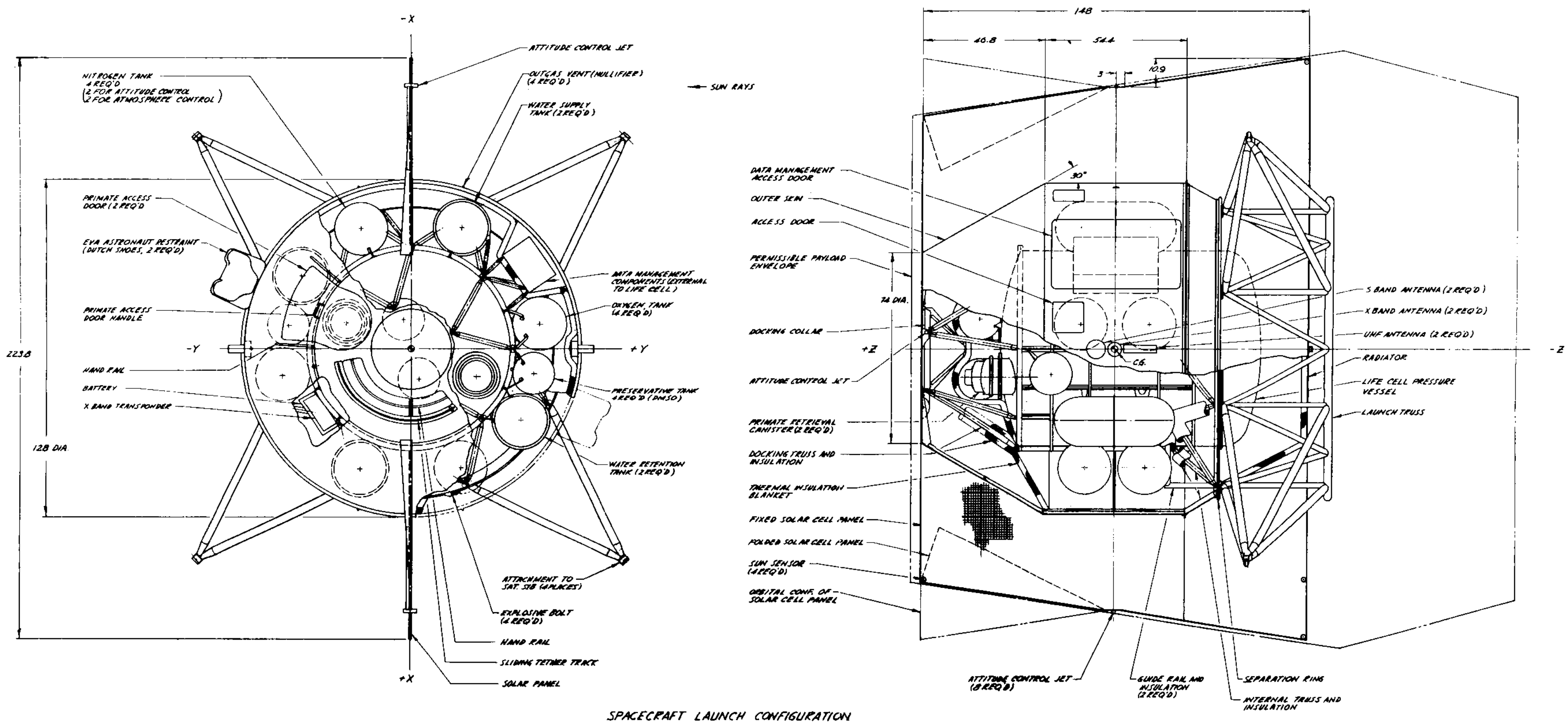


Fig. 123 Spacecraft General Arrangement

There are certain mass changes over which little control can be exercised. These are located in the lifecell (Fig. 124) and are itemized as follows:

- Food: A year's supply of pelletized food is contained in two cylindrical hoppers balanced about the spacecraft Z axis.

$$W = 198 \text{ lb} \quad , \quad Z = +9.5 \quad , \quad R^* = 19.0 \text{ in.}$$

- Feces, Urine, and Germicide Residue: Food is converted to feces at the rate of 0.33 to 0.50 lb feces per lb food. Urine and germicide residue are collected in the waste management subsystem at the rate of 2 to 5 percent total urine. These wastes are assumed to be evenly distributed below the cage floors.

$$W = 76.0 \text{ to } 124.6 \text{ lb} \quad , \quad Z = -40.5 \quad , \quad R = 17.2 \text{ in.}$$

- Lithium Hydroxide (LiOH) Bed: The LiOH beds experience a weight gain based on absorbing 0.33 lb CO_2/day/monkey. At the same time 0.2 to 0.4 lb water/lb CO_2 absorbed is released. Thus, the net gain per LiOH bed is

$$[0.33 \, (365 + 14)] \, (1 - 0.2) = 100 \text{ lb}$$

$$\Sigma W = 200 \text{ lb} \quad , \quad Z = -11.5 \quad , \quad R = 34.8 \text{ in.}$$

- Animals: Two 13-lb monkeys can range anywhere in the cage and retrieval canisters.

$$\Sigma W = 26 \text{ lb} \quad , \quad Z = -25.5 \text{ to } +22.5 \quad , \quad R = 17.2 \pm 8.5 \text{ in.}$$

  In the retrieval canister

$$Z = +39.5 \quad , \quad R = 27 \text{ in.}$$

The worst condition for vertical c.g. travel would be as follows (see Table 49):

- Monkeys at bottom of cages
- Food containers empty
- Waste containers full at the worst rates of waste buildup
- LiOH canisters saturated.

If one animal were to die and be encapsulated in the retrieval canister for one year then the imbalance would be less than half of -9,845 in. lb due to the repositioning of one animal. The resultant imbalance would be

$$\left(\frac{-9,845}{2}\right) + 312 + (13)(39.5) = -4,098 \text{ in. lb}$$

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*Radius from Z axis

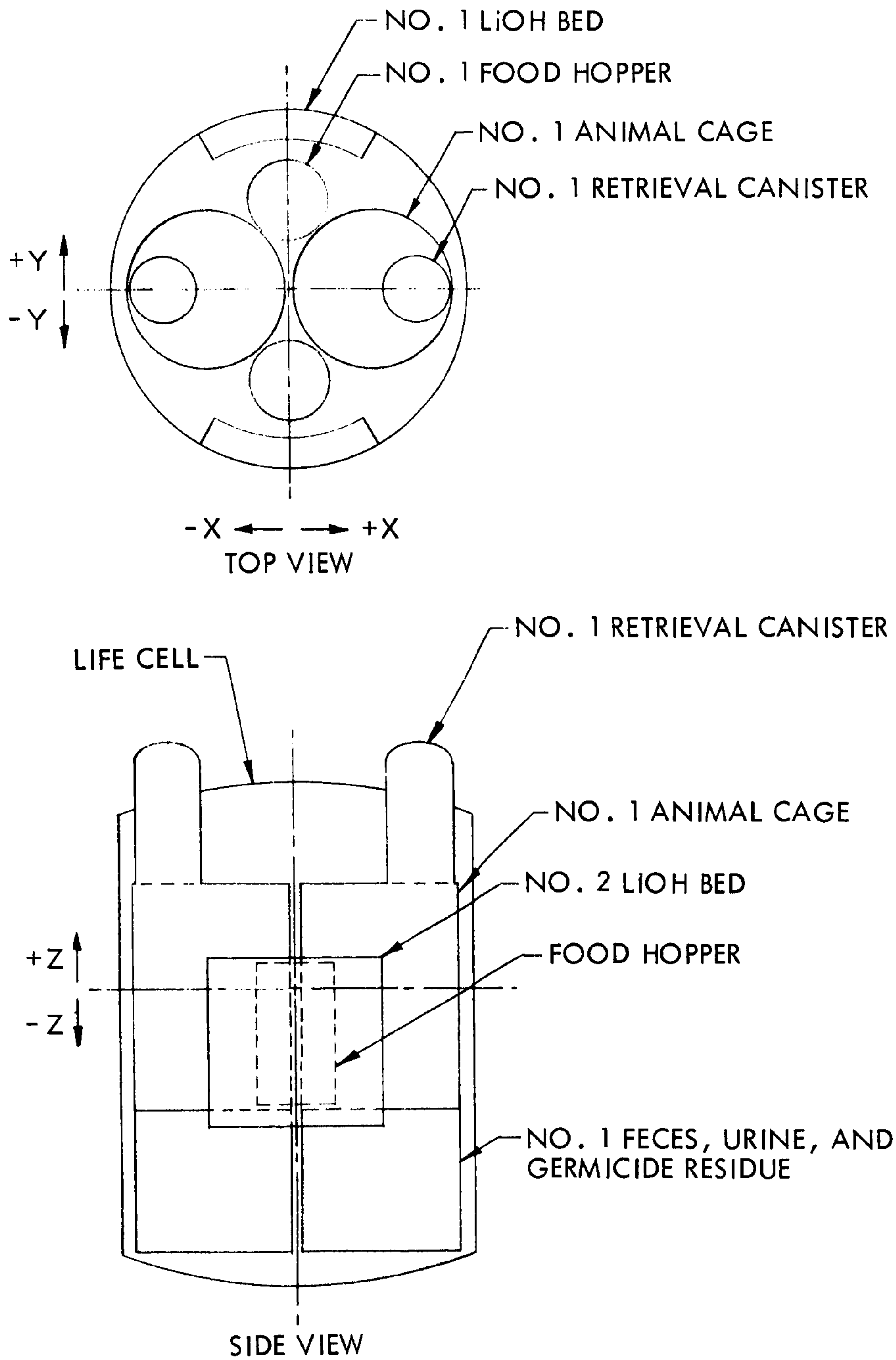


Fig. 124 Lifecell Variable Weights

TABLE 49

VERTICAL MASS IMBALANCE (± Z)

Item	W (lb)		Z (in.)		WZ (in. lb)		
	Initial	Final	Initial	Final	Initial	Final	Net
Food No. 1	98	0	+9.5	+9.5	+940.5	0	-940.5
Food No. 2	98	0	+9.5	+9.5	+940.5	0	-940.5
Waste No. 1	0	62.3	-40.5	-40.5	0	-2,520	-2,520
Waste No. 2	0	62.3	-40.5	-40.5	0	-2,520	-2,520
LiOH Bed No. 1	0	100	-11.5	-11.5	0	-1,150	-1,150
LiOH Bed No. 2	0	100	-11.5	-11.5	0	-1,150	-1,150
Animal No. 1	13	13	-1.5	-25.5	-19.5	-331.5	-312
Animal No. 2	13	13	-1.5	-25.5	-19.5	-331.5	-312
						Total	-9,845

The worst condition for lateral (±Y) c.g. travel would be as follows (see Table 50):

- One monkey dead near start of year and encapsulated in retrieval capsule
- One food container full and the other empty
- One waste container full and the other empty
- One LiOH canister saturated and the other unsaturated; however, due to atmosphere sharing by the LiOH beds, equal length plumbing, and symmetry, this is unlikely to happen.

TABLE 50

LATERAL MASS IMBALANCE (±Y)

Item	W (lb)		Y (in.)		WY (in. lb)		
	Initial	Final	Initial	Final	Initial	Final	Net
Food No. 1	98	0	+19.0	+19.0	+1,863	0	-1,863
Food No. 2	98	98	-19.0	-19.0	-1,863	-1,863	0
Waste No. 1	0	62.3	0.0	0.0	0	0	0
Waste No. 2	0	0	0.0	0.0	0	0	0
LiOH Bed No. 1	0	100	+34.8	+34.8	0	+3,400	+3,400
LiOH Bed No. 2	0	0	-34.8	-34.8	0	0	0
Animal No. 1	13	13	0	0±8.5	0	0	0
Animal No. 2	13	13	0	0	0	0	0
						Net Total	+1,537

If the LiOH beds were to saturate evenly, then the net imbalance would change to -1,963 in. lb. The worst value of lateral (±X) c.g. travel would occur at the same time as above for (±Y). (See Table 51.)

TABLE 51

LATERAL MASS IMBALANCE (±X)

Item	W (lb)		Y (in.)		WX (in. lb)		
	Initial	Final	Initial	Final	Initial	Final	Net
Food No. 1	98	0	0	0	0	0	0
Food No. 2	98	98	0	0	0	0	0
Waste No. 1	0	62.3	0	0	0	0	0
Waste No. 2	0	0	0	0	0	0	0
LiOH Bed No. 1	0	100	0	0	0	0	0
LiOH Bed No. 2	0	0	0	0	0	0	0
Animal No. 1	13	13	+17.2	-17.2 ± 8.5	+224	+224±111	±111
Animal No. 2	13	13	-17.2	-27	-224	-350	-126
					MAX NET TOTAL		-237

The spacecraft weight at initial orbital injection is estimated at 5,300 lb. The weight at the end of the year is estimated at 4,959 lb for a successful mission and 5,030 lb if one animal dies near the start of orbital flight.

Thus, the maximum c.g. shift expected is:

- Case A Successful mission (both animals survive for one year)

$$\Delta Mx = 0$$

$$\Delta My = 0$$

$$\Delta Mz = -9845 \text{ in. lb}$$

final $$\Delta z = \frac{-9845}{4959} = -1.97 \text{ in.}$$

$$\Delta R \text{ (the radial shift)} = 1.97 \text{ in.}$$

- Case B Loss of one animal near beginning of mission

$$\Delta Mx = -237; \text{ final } \quad \Delta x = \frac{-237}{5030} = -0.05 \text{ in.}$$

$$\Delta My = -1863; \text{ final } \quad \Delta y = \frac{-1863}{5030} = -0.37 \text{ in.}$$

$$\Delta Mz = -4098; \text{ final } \quad \Delta z = \frac{-4098}{5030} = 0.82 \text{ in.}$$

$$\Delta R \text{ (the radial shift is) } = \sqrt{(0.05)^2 + (0.37)^2 + (0.82)^2} = \sqrt{0.82} = 0.91 \text{ in.}$$

These c.g. shifts take place linearly with time during normal operation. Catastrophic loss of expendables could cause extensive c.g. shifts. For instance, the loss of one water tank at the first of the year would produce $386 \times 51 = 19,700$ in. lb of imbalance or a c.g. shift of $\frac{19,700}{5300-286} = 4.0$ in.

Any c.g. shifts impose a larger work load on the attitude control system primarily due to increased aerodynamic torque. For any nonspherical body, the center of pressure experiences excursions dependent on the angle of attack. The ultimate control of c.g. would have to be compatible with this center of pressure movement. However, the loss of one-half of the expendables would result in a foreshortened mission. It is therefore likely that the resulting higher attitude control fuel consumption rate would be offset by a shorter mission duration so that the total fuel supply would be adequate. A more detailed coverage of this subject is found in the foregoing sections that describe the attitude control subsystem.

The moments of inertia are I_{xx} 2176 slug-ft^2, I_{yy} 2325 slug-ft^2, and I_{zz} 2395 slug-ft^2.

Construction. – The spacecraft shown in Figs. 123 and 125 is designed to fit in the Saturn S-IB launch vehicle as a Lunar Module (LM) substitute. The launch truss replaces the LM descent stage and the spacecraft replaces the LM ascent stage. As with LM, a docking collar is provided for orbital injection by the Command/Service Module (CSM) propulsion system. A variation from LM operations is made by leaving the launch truss in the launch vehicle. The main elements of the structure are discussed below.

Launch truss: The launch truss is a tubular-welded-titanium space frame which supports the spacecraft within the Spacecraft/Lunar-Module Adapter (SLA) before, during, and after powered boost until spacecraft separation takes place. The four outriggers are fastened to the SLA at the normal LM hard points and utilize the same hold downs and release mechanisms as does the LM. The eight lower members of the outriggers are joined to a continuous tubular ring. The eight upper members of the outriggers each terminate in a V-block fitting that also provides for the junction of three additional truss members.

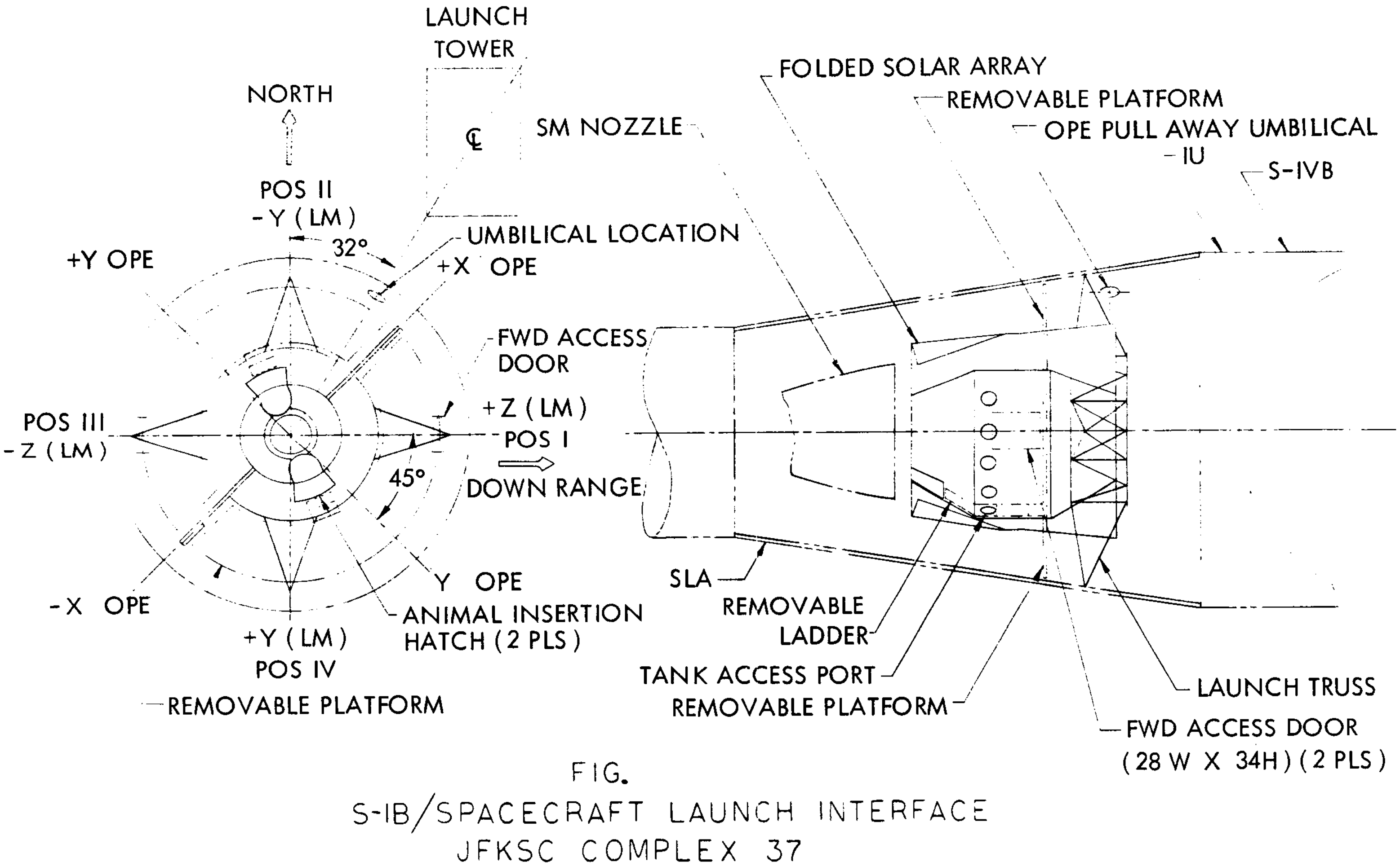


Fig. 125 S-IB/Spacecraft Launch Interface

Two of these three truss members are connected diagonally to the continuous tubular ring, and the third is connected vertically to the same ring. This arrangement (when coupled to the separation ring) permits load transfer to occur under any combination of linear or moment loading.

Internal truss: The internal truss connects and transmits all the spacecraft inertial and docking reactions to the launch truss.

Sixteen titanium lugs are fastened by bolts to the lifecell pressure vessel near its lower end (Z = -22.25). These lugs are thermally isolated from the lifecell by phenolic bushings and washers (Fig. 126). Welded to each lug is a diagonal tubular titanium member which is subsequently bolted to a continuous titanium separation ring. This separation ring (Fig. 127) has a Y cross section, one leg of which is beveled to form a mating V with the V-block fittings of the launch truss. The other legs of the Y, besides providing torsional stiffness, are attachment points for the outer skin. The separation ring mates with and provides radially compressive support for the eight V blocks of the launch truss by means of a circumferentially oriented key. Circumferential restraint of the launch truss is provided by tapered pins in the separation ring which engage tapered sockets in the eight V-block fittings.

Lifecell pressure vessel interface: The lifecell pressure vessel, although not considered a part of the spacecraft does provide the vital function of consolidating and transmitting most of the inertial loads of the spacecraft. This vessel is an aluminum shell with external rings and longerons which provide foundation for miscellaneous bracketry. The internal components of the lifecell are packaged on a common chassis that is bolted at eight inside points directly opposite the points where the sixteen internal truss lugs are fastened. This arrangement permits almost all the inertial load of the internal equipment to pass through the pressure vessel wall without loading the pressure vessel.

Equipment supports: All the major components located externally to the lifecell are supported by structure mounted to the exoskeleton of the lifecell pressure vessel. The mounts for the water supply tank and nitrogen sphere consist of web members welded to the tanks with outstanding legs that are fastened directly to the lifecell pressure vessel longerons. The addition of sway braces from the adjacent longerons complete the installations. Waste water storage tanks are supported by two longitudinal webs from the lifecell pressure vessel longerons. Oxygen and two of the four preservative spheres are supported by tie rods. The remaining two preservative spheres are bracketed directly off the upper dome of the pressure vessel. The data management electronics assembly, which is passively thermally controlled, is supported by a tubular extension frame in order to position it in close proximity to the outer skin. The battery and X-band transceiver are mounted directly on the pressure vessel longerons. The solar arrays are cantilever-mounted to the spacecraft skin with a field joint at Z = -27.0.

Tanks: All gases are stored in high-pressure vessels having a maximum inner diameter of 20 inches. This restriction was imposed in an effort to limit the overall

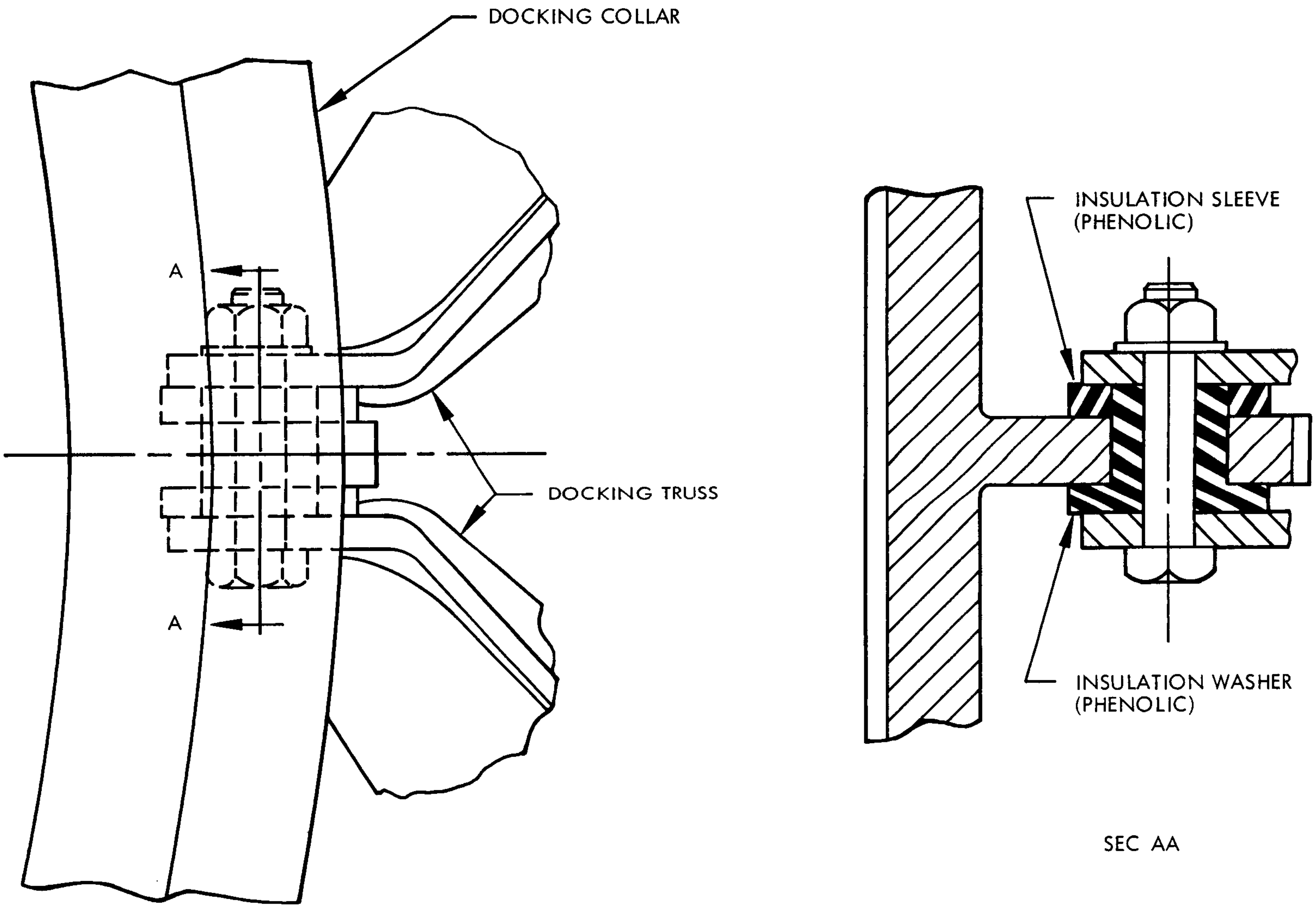


Fig. 126 Typical Thermal Insulation Joint

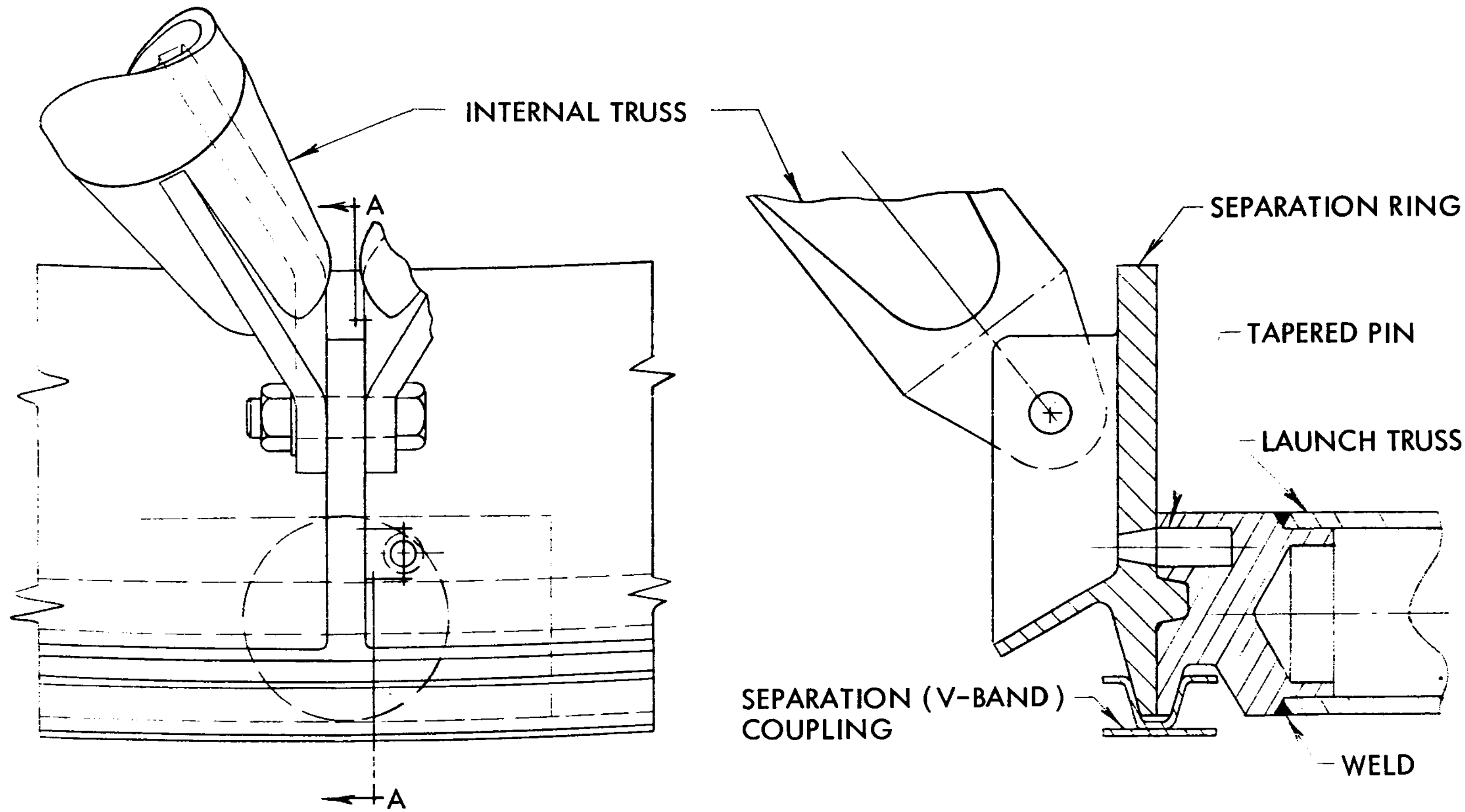


Fig. 127 Typical Ring Separation Joint

spacecraft diameter. Tank factors (ratio of tank weight to content weight) for gas storage are relatively independent of the number of tanks and the storage pressure up to about 6,000 psia for oxygen as long as they remain spherical. Therefore, the gases were subdivided into 20-in. spheres to comply with the spacecraft overall diameter goal and, also, to achieve a reasonable working pressure. If consistent storage pressures were desired, the tank diameters could be reduced, and/or their number increased.

Oxygen is stored in four 20-in. diameter spheres at a pressure of 4,550 psia. The tank material selected is 17-4 PH stainless steel that is heat treated to condition H 900. This material was selected on the basis of its strength/weight ratio, weldability, machinability, formability, toughness, oxygen compatibility, and corrosion resistance.

Nitrogen for both the attitude control fuel and the atmosphere makeup is stored in four 20-in. diameter spheres at a pressure of 3,900 psia. The tank material selected here is titanium (13V-11Cr-3A1) in the annealed condition. This selection was made on the basis of its strength/weight ratio, (which is superior to 17-4PH), corrosion resistance, and weldability.

The water supply tanks are of the bladder expulsion type. Two cylindrical tanks with hemispherical ends are used to contain the one-year supply of drinking water. The expulsion pressurant is at 35 psia. The tank material selected is 17-4PH stainless steel for the same reasons as stated for the oxygen tanks. The water retention tanks are identical to the water supply tanks.

Docking collar and truss: A female docking collar is mounted on the Z axis flush with the upper disc (Z = +74.00). The perimeter of the docking collar is supported at each of four points by a pair of diagonal tubular titanium struts which branch out and connect to eight points on the upper ring of the lifecell pressure vessel. At the time of docking, a one atmosphere pressure differential exists across the lifecell pressure vessel wall that helps carry the loads of docking and subsequent thrusting.

A conductive heat barrier is installed at each of the eight pinned joints on the top of the lifecell pressure vessel by installation of phenolic bushings and washers in the same manner as discussed in the foregoing description of the internal truss.

Outer skin: The outer skin consists of a single thickness of 0.025 aluminum sheet which is sized to defeat, in conjunction with the lifecell pressure vessel, meteoroid penetrations. The probability of no meteoroid penetration is in excess of 0.950.

The outer skin is arranged in three sections that are supported from the perimeter of the docking collar and the separation ring. Field joints are located circumferentially at Z = +74.00, Z = -27.2, and Z = -41.0. The top section of the outer skin can be removed by unfastening the fasteners at Z = +74.00 and Z = -27.2, and lifting upward. The bottom section is removed by unfastening the joint at Z = -41.0 and lowering. The middle section is emplaced during manufacture and is not removable in the field. All the skins are stiffened with fiberglass rings to prevent flutter and also to provide support for the insulation blanket.

Insulation blanket: The insulation blanket (Fig. 128) consists of alternate layers of aluminized mylar reflectors and tissue glass insulators spaced at 60 layers per inch for a total thickness of 2 inches. The blanket is supported by nylon threads spaced every 8 inches which penetrate all the layers and are tied to teflon buttons. One button is bonded to the inner surface of the outer skin and the other button retains the blanket. The blanket is subdivided into sections which nest in compartments formed by the fiberglass rings mentioned above. As a final measure, to prevent sagging, a dacron net is suspended over the blanket from ring to ring.

The insulation blanket covers the entire inner surface of the outer skin except at the top (Z = +44) of the spacecraft where the blanket crosses over and covers the top of the lifecell pressure vessel. Pockets cover the retrieval canisters and preservative tanks.

Radiator: The radiator is located on the bottom disc (Z = -74.0) and occupies the entire surface (Fig. 129). The heat transfer fluid, ethylene glycol, is carried in 0.25-in. diameter hollow-bulb "T" extrusions which are intimately joined to the radiator surface sheet by brazing. The radiator surface sheet and stand-off fluid line arrangement was designed to minimize meteoroid puncture problems. A silicone bracket thermally separates the radiator from the rest of the spacecraft outer skin.

Deployments. – Deployments of the spacecraft separation and solar panel are discussed in the following paragraphs.

Spacecraft separation: The separation of the spacecraft from the launch vehicle is accomplished by release of two taut V-band/V-block systems (See Fig. 127). The separation ring is machined with a 20 deg wedge surface on its perimeter. A complementary 20-deg wedge surface is machined on the eight V-block fittings of the launch truss. The eight V-block fittings and the separation ring form double wedge surfaces which are held together by the force of taut circumferential bands pressing female V-clamps against these double wedge surfaces. Wedge-shaped strips are fastened periodically to the lower surface of the separation ring between the eight V-block fittings of the launch truss in order to prevent the taut bands from rolling. These strips are slightly recessed radially from the eight V-block fittings to insure good clamping action at the fittings. The separation ring and the eight V-block fittings are keyed so that no relative radial displacement can occur, and pinned so that no relative circumferential displacement can occur between them.

The two semicircular taut V-bands are anchored near the points where the separation ring passes through the solar array paddles. Two redundant detonating-type turnbuckles are installed as tensioning devices in each band making a total of four such devices. These turnbuckles are of the dual cartridge type, and are identified as Standard Pressed Steel Company P/N SD 720-100-59259. The detonating cartridges are identified as McCormick-Selph Associates P/N 804437.

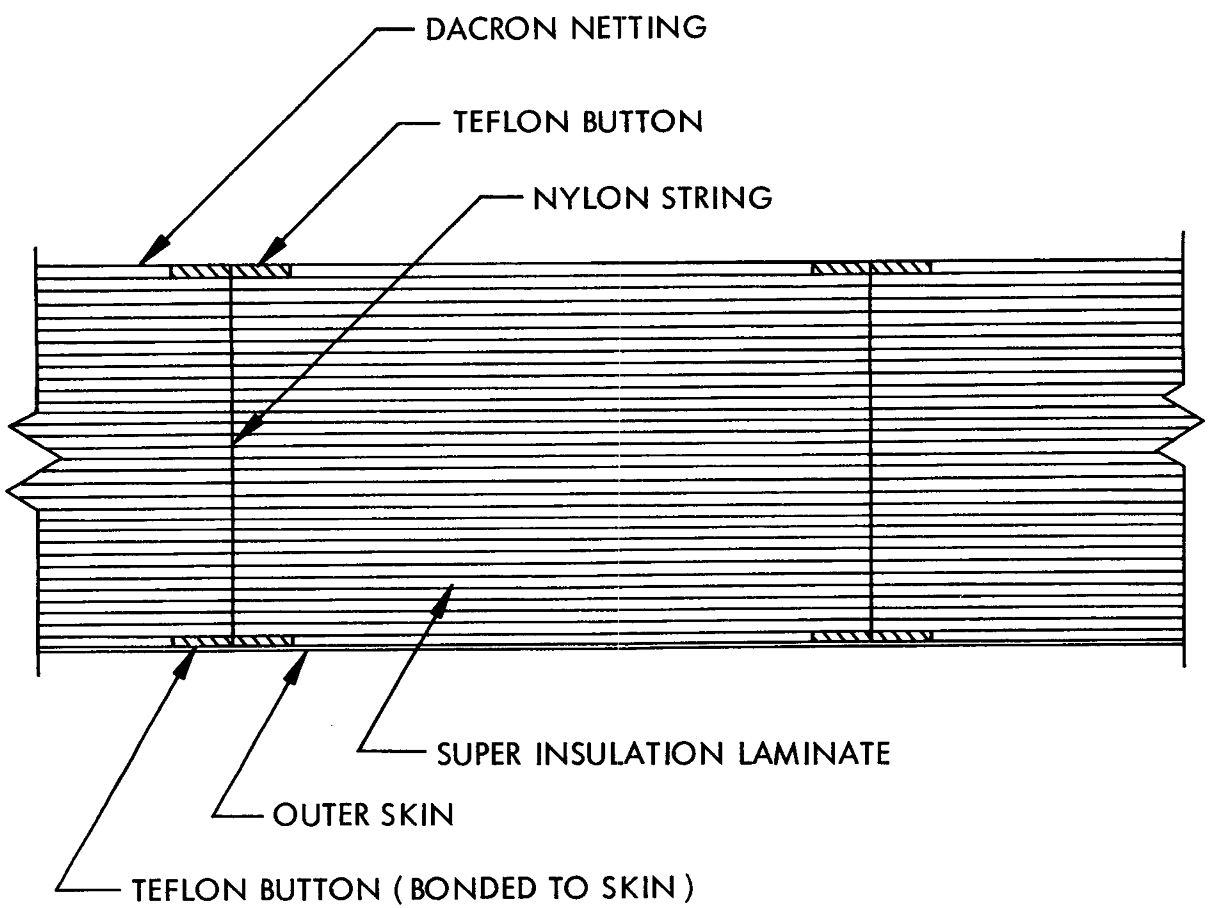


Fig. 128 Insulation Detail

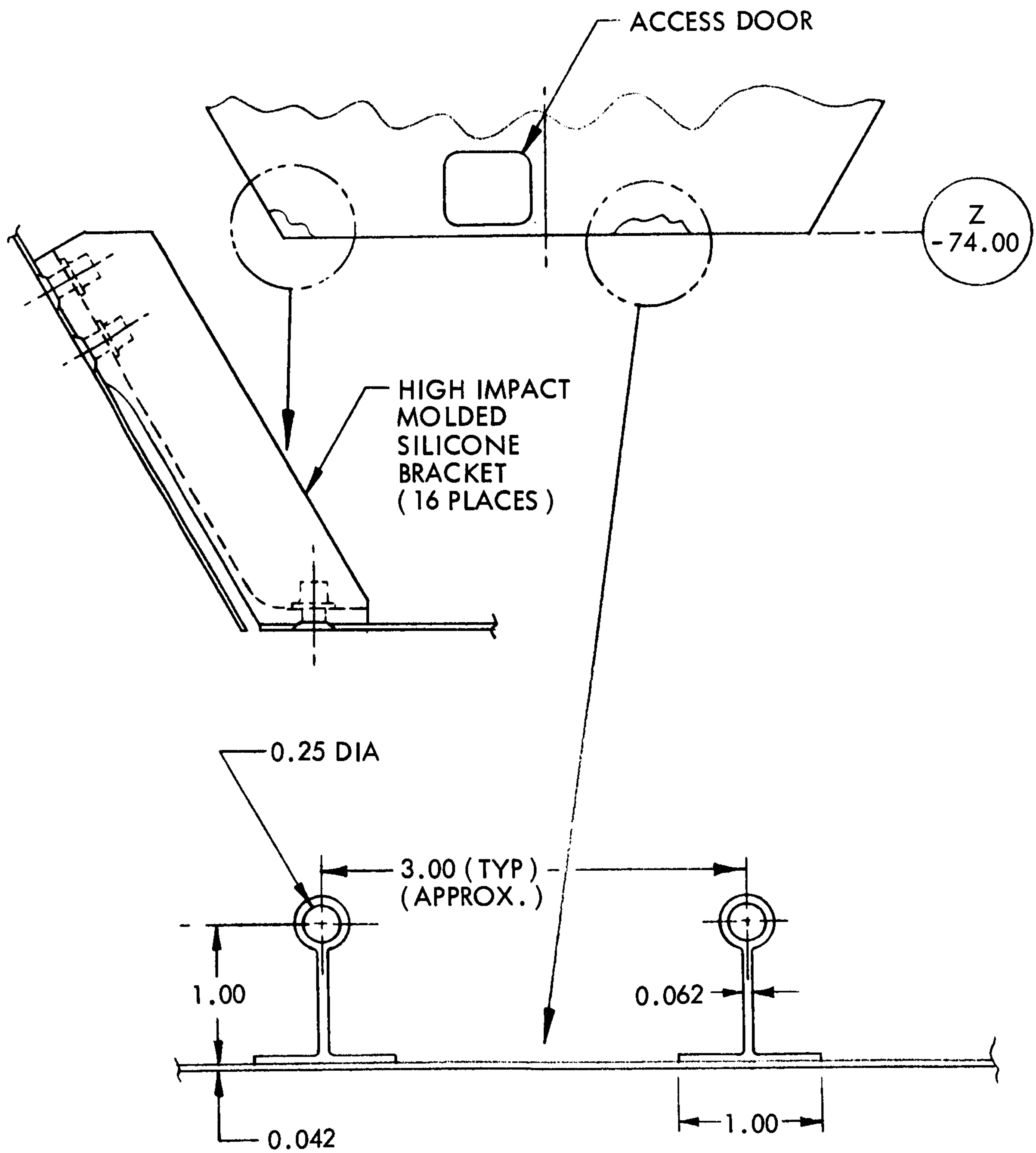


Fig. 129 Radiator Detail

Appropriate tethers and springs are attached to the taut V-bands and wedges to constrain them from damaging or entangling with the spacecraft during the separation sequence. The lower portion of the spacecraft and its solar arrays, due to their overlap with the SLA and launch truss, require guidance during the separation event. Separation guides located at 2 points maintain alignment between the spacecraft and the launch vehicle (preventing damage to the solar arrays) while the spacecraft/CSM is backing out of the SLA and launch truss.

Initiation of the separation is accomplished by command from the command module. In the event of failure of the primary separation device, the LM release mechanisms can be functioned. As a last resort, astronaut EVA could be used to initiate separation.

Solar panel: The upper outboard corners of the solar arrays must be folded to conform within the SLA launch envelope. The fold is made such that the solar cells are exposed at all times. A self-latching device with a lanyard release holds the panel folded against the torque of a spring. When the lanyard is pulled, the latch is released, and the solar panel unfolds to its orbital configuration. A dashpot softens the opening shock and a second self-latching device locks the panel in the open position. The lanyards are pulled by the deployment of the SLA petals. Other events could trigger the deployment such as docking or astronaut EVA.

Pyrotechnics. – All pyrotechnics aboard the spacecraft and payload are summarized in Table 52.

Servicing. – The servicing of lifecell subsystems, fluid and gas tanks, battery and data management components, umbilicals, and various connectors is discussed in the following paragraphs.

Removal of lifecell subsystems: Accessability of the lifecell subsystems has been a primary requirement of the spacecraft design. The lifecell pressure vessel has been designed so that all internal components are mounted on one chassis. Removal of this assembly from the pressure vessel permits direct access to a majority of the components.

The steps in removal of the lifecell subsystems are:

1. Install lifting eye in docking fitting and elevate spacecraft to convenient work position.
2. Remove radiator service panel and disconnect coolant fluid lines.
   (a) Remove fasteners and disconnect electrical leads joining the lower third of the solar arrays to the upper two thirds.
3. Remove lower skin attachment fasteners.
4. Remove lower skin and insulation blanket.
5. Remove lifecell pressure vessel lower dome V-band clamp.
6. Remove lifecell pressure vessel lower dome.

TABLE 52

PYROTECHNICS SUMMARY

No. Units	Location	Function	Type	Vendor Part No.
4	Z = -40.0	S/C separation	Detonating explosive bolt	Bolt Standard Pressed Steel Co. SD 720-100-59259 Detonating Cartridge (2/bolt) McCormick-Selph Assoc. 804437
4	Z = -55.0	Backup S/C Separation	---	G. F. E. (Normal LM release)
8	Z = +30.0	Preservative expulsion and KO_2 release	Gas generator piston valve	Similar to Pyrodyne Inc. 2780-5
1 1	Z = -42.0 Z = -50.0	Cut umbilical lines	Gas piston type guillotine	
2	Z = -24.0	Retrieval piston gas	Gas generaator piston valve	Similar to LMSC 1508074 Similar to Pyrodyne Inc. 2780-5

7. Disconnect inner electrical and plumbing fittings from feed-through panels.
8. Place support under subsystems.
9. Remove subsystem fasteners.
10. Lift spacecraft off of subsystems.

These steps may not be taken after the SLA has been mated to the instrument unit (IU) of the launch vehicle.

Charging tanks: Service disconnects and manual shutoff valves for all fluids and gases are accessible through individual service panels located in the outer skin at Z = +20.0 (See Fig. 125). The DMSO preservative fill and shutoffs are accessible through the removable retrieval fairings. The charging of all fluid and gas tanks may be accomplished at any time up until about 10 hr from launch when the SLA must be vacated. However, cleanliness requirements may require charging in other than launch-pad conditions.

Tank repairs: Repairs to any of the tanks require removal of the upper skin and insulation blanket. Steps in removal are:

1. Support spacecraft from below.
2. Remove all tank charging panels and fairings.
3. Unfasten the insulation blanket joint (Z = +44.0) between lifecell and outer skin.
   (a) Unfasten upper and side reflector panels surrounding the data management subsystem module.
4. Unfasten upper skin fasteners at the docking collar and separation ring.
5. Attach lifting sling to upper skin.
6. Lift skin.
7. Tanks are now accessible for repair or replacement.

These steps may not be taken after the spacecraft has been mated to the SLA.

Battery and data management subsystem components renewal: The battery and the data management sybsystem components external to the lifecell pressure vessel are accessible for removal or repair through removable panels in the outer skin (Z = 0.0).

Umbilicals: Fly-away ground services required are:

- 15 amperes d.c. @ +28V
- Electric power return
- Coolant in
- Coolant out
- S-band R/F hardline out
- X-band R/F hardline out
- Airconditioning to passively cooled components

The pull-away connectors are located in the SLA wall at Z = -55.0. The leads pass through two guillotines which are functioned at vehicle first motion. The purpose of the guillotines is to permit subsequent spacecraft separation at either the primary or secondary separation points.

Human factors. – Human factors relating to retrieval and primate insertion are discussed in the following paragraphs.

Retrieval: Access to the retrieval canisters is achieved by removing two sectors of the outer skin (See Fig. 130). A handle is linked to six self-locking latches on each sector which open simultaneously when the handle is squeezed. The sector is then free to be jettisoned.

A "dutch shoe" fitting and pair of waist-level restraint eyes are located at each work station. The insulation blanket, which covers the retrieval canister, is removable with a fast-acting one-hand operable separation device.

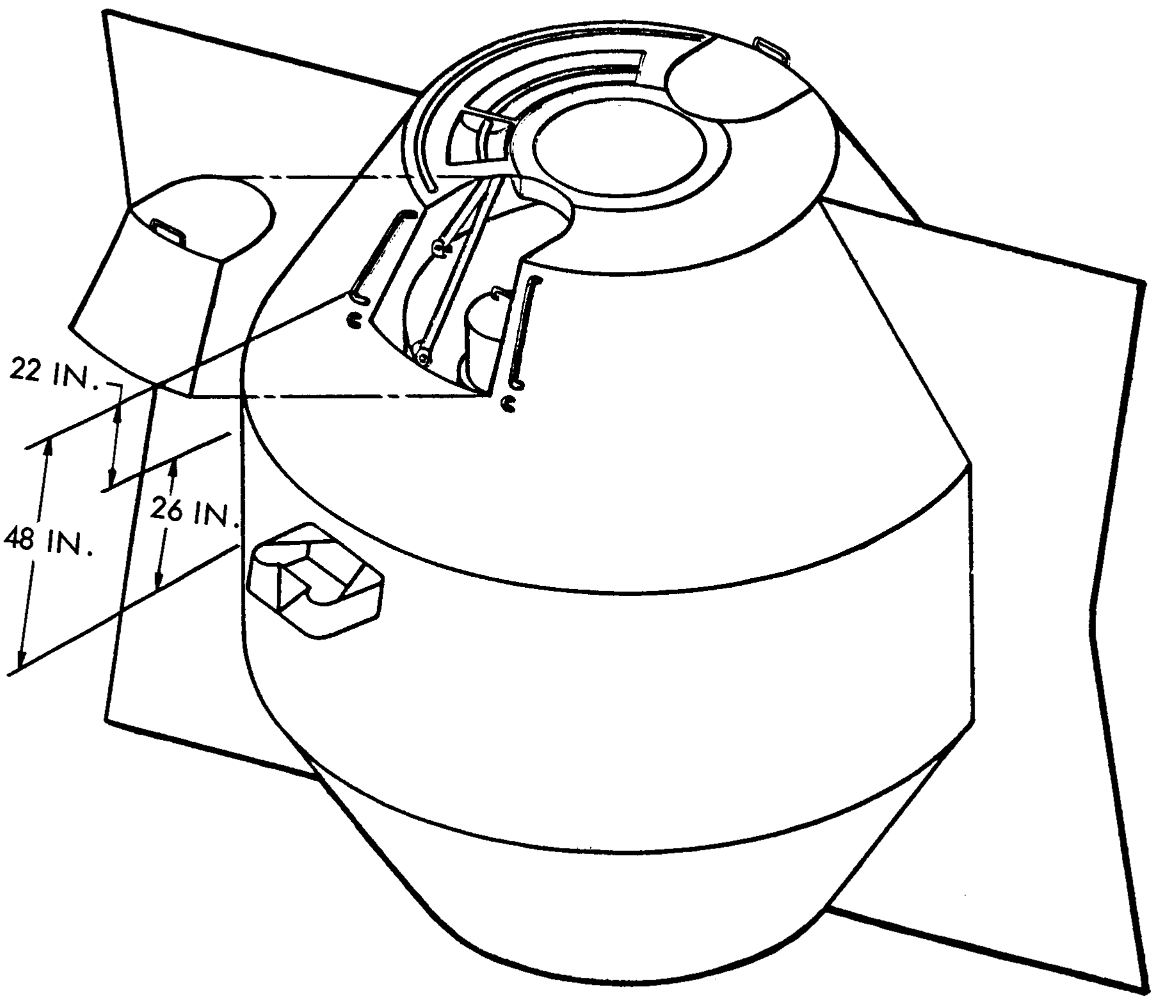


Fig. 130 Astronaut Access to Retrieval Canisters

Two service disconnects per canister are then exposed. An over-center latch releases the taut V-band which frees the canister for removal. Each canister has an extraction handle. To traverse from one retrieval work station to the other, a depressed hand railing is provided.

Primate insertion: Insertion of the primates on the launch pad is accomplished by removing a taut V-band above the retrieval V-band. This releases the upper cap of the retrieval canister and permits the animal to be transferred from his carrying container directly into the lifecell. Access is provided by removing the sectors described previously and removing the layer of insulating blanket covering the retrieval canister.

Weight Summary

The following section summarizes the weight of the OPE spacecraft in its launch condition. The tabulation is organized in two subsections, 1) the life cell pressure vessel and its contents (payload), and 2) the spacecraft. The total OPE weight at launch is 5500.0 lb. At the time of orbital injection, the total weight is 5257.9. The weights are summarized in Tables 53 through 56.

TABLE 53

PAYLOAD (LIFECELL) WEIGHT STATEMENT

Environmental Control Subsystem		552.0
High Flow Fans – 4 at 3.75	15.0	
Low Flow Fans – 4 at 6.0	24.0	
Cabin Purge Fans – 2 at 10.0	20.0	
Lithium Hydroxide – 2 at 156.0	312.0	
Lithium Hydroxide/Charcoal Canister – 2 at 39.0	78.0	
Charcoal – 2 at 46.0	92.0	
Condensate Heat Exchanger – 2 at 2.0	4.0	
Temperature Control Valves	2.0	
Coolant Pumps – 2 at 3.5	7.0	
Accumulator	12.0	
Check Valves	1.0	
Vent Valve Assembly	2.0	
Relief Valve Assembly	2.0	
Coolant Plumbing	3.0	
Waste Water Plumbing	3.0	
Atmosphere Supply Plumbing	1.0	
Ducting	10.0	
Coolant	10.0	

TABLE 53 (Cont.)

Waste Management Subsystem			134.0
Waste Container Baskets — 2 at 1.5		3.0	
Wicking Material (25,000 in.^{3}) at 5 lb/ft^3		72.5	
Psychomotor Subsystem			34.0
Programmers — 2 at 4.0		8.0	
Work Panels Assys. — 2 at 3.0		6.0	
Exercise Bar Assys. (Including Hydraulic Fluid) — 2 at 10.0		20.0	
Feeding Subsystem			255.0
Canisters			
Pistons Rotating Plates Dispensing Assys. Plate Drive Motors Constant Torque Spring Assys.	2 at 24.5	49.0	
Food (46,643 gm/canister)		206.0	
Retrieval Subsystem (2)			145.0
(less Preservative & Tanks)			
Drinking Water Subsystem (2)			29.6
(less Tanks & Water)			
Life Cell Pressure Vessel			260.0
(Breakdown on Separate Sheet)			
Metabolic Support Structure			65.0
Cage Complex — 2 at 42.0			84.0
(Does Include Elevator & Motor)			
Data Management Subsystem (See Breakdown in Table 55)			103.2
(Inside Life Cell)			
Mass Measurement Subsystem — 2 at 40.0			80.0
Contingency			12.5
Animal – 2 at 13.0			26.0
TOTAL PAYLOAD WEIGHT			1771.8

TABLE 54

SPACECRAFT WEIGHT STATEMENT

Data Management Subsystem (outside Life Cell)		199.7
Box	16.8	
Components (incl. A/C Electronics) (See breakdown in Table 55)	169.4	
Support Structure	12.0	
Reflector	1.5	
Water Tank Subsystem		826.8
Tank – 2 at 9.9	19.8	
H_2O	772.0	
Support Structure	35.0	
Nitrogen Tank Subsystem		499.0
Tank – 4 at 67	268.0	
N_2	184.0	
Support Structure	37.0	
Regulators	10.0	
Waste Water Tank Subsystem		22.3
Tank – 2 at 9.9	19.8	
Support Structure	2.5	
Preservative Tank Subsystem		279.0
Tank – 4 at 3.0	12.0	
Preservative	250.0	
Support Structure	17.0	
Oxygen Tank Subsystem		665.0
Tank – 4 at 93.5	374.0	
O_2	246.0	
Support Structure	40.0	
Regulator	5.0	
Battery Subsystem		177.0
Cold Plate	10.0	
Battery – 2 at 72	144.0	
Support Structure	15.0	
Charger & Voltage Regulator	8.0	
X-Band Transceiver Subsystem		28.8
X-Band Transceiver	20.8	
Water Boiler (including 3 lb H_2O)	5.0	
Support Structure	3.0	

TABLE 54 (Cont.)

EVA Provisions Subsystem		13.0
Hand Rail Provisions	2.0	
Dutch Shoes	5.0	
Tracks & Tethers	1.0	
Access Door Provisions	5.0	
Antenna Subsystem		12.6
S-Band Antenna – 2 at 1.3	2.6	
X-Band Antenna – 2 at 1.0	2.0	
UHF Antenna	4.0	
Mounting Provisions	4.0	
Solar Array Subsystem		107.4
Solar Panels – 100 ft^2 at 0.894 lb/ft^2	89.4	
Support Structure	15.0	
Deployment Provisions	3.0	
Docking Truss Subsystem		92.4
Docking Collar	50.0	
Support Tubes – 8 at 2.29	18.4	
Tube Fittings – 12 at 2.0	24.0	
Internal Truss Subsystem		110.8
Fittings – 16 at 2.0	32.0	
Tubes – 16 at 0.86	13.8	
Separation Ring	65.0	
Launch Truss Subsystem		203.3
LM Fittings & Hold Downs – 4 at 4.0	16.0	
Upper Outrigger Tubes – 8 at 5.26	42.1	
Lower Outrigger Tubes – 8 at 5.80	46.4	
Tubular Ring	37.6	
V-Block Fittings – 8 at 4.0	32.0	
Vertical Tubes – 8 at 1.11	8.9	
Diagonal Tubes – 16 at 1.27	20.3	
Separation Subsystem		38.8
V-Band	29.8	
Turnbuckles & Squibs – 4 at .50	2.0	
Anchors – 4 at 1.0	4.0	
Tethers & Springs	2.0	
Guillotine – 2 at 0.50	1.0	
Attitude Control Subsystem		11.0
Jets – Valves & Plumbing	10.2	
Sun Sensors – 4 sets at .2	0.8	

TABLE 54 (Cont.)

Outer Skin Subsystem		190.0
Skin (less radiator)	150.0	
Stiffeners & Attachments	25.0	
Access Provisions (doublers, etc.)	15.0	
Thermal Control Subsystem		227.0
Insulation Blanket	143.0	
Paint & Other Finishes	54.0	
Tube Insulation	20.0	
Access Provisions	10.0	
Radiator Subsystem		28.3
Skin & Tubes	26.3	
Vernatherm Valve	1.0	
Ground Cooling Heat Exchanger	1.0	
TOTAL SPACECRAFT		3,732.2

TABLE 55

DMS (INSIDE LIFECELL)

Implants		0.9
EKG – 2 at 0.13	0.3	
Temp. – 2 at 0.13	0.3	
Respiration – 2 at 0.13	0.3	
Near-Field Receiver Assy.		1.4
Receiver Assys. – 2 at 0.44	0.9	
Antenna – 2 at 0.25	0.5	
Activity Counter		2.7
Additional Electronics – 2 at 0.38 (Ampl., Schmitt Trigger, Stor. Counter)	0.8	
Magnets – 2 at 0.19	0.4	
Pickup Coils – 4 at 0.38	1.5	
Vocalization		1.5
Microphone – 2 at 0.5	1.0	
Amplifier – 1 at 0.5	0.5	

TABLE 55 (Cont.)

Environmental Measurements		38.7
Temp. Thermistor – 20 at 0.19	3.8	
Temp. Thermistor – 33 at 0.03	1.0	
Differential Pressure – 8 at 1.0	8.0	
Gage Pressure – 6 at 0.38	2.4	
Absolute Pressure – 1 at 0.44	0.5	
Air Flow – 2 at 0.31	0.6	
Water Flow – 1 at 0.19	0.2	
Atmosphere Composition – 2 at 5.0	10.0	
Dosimeter – 1 at 8.0, 2 at 0.5, 4 at 0.06	9.3	
1-Axis Rate Gyro – 3 at 0.28	0.9	
Accelerometer – 6 at 0.19	1.2	
Fan Speed Sensor – 8 at 0.1	0.8	
TV Subsystem		48.0
Camera Head – 4 at 3.5	14.0	
Controller – 1 at 25.0	25.0	
Lens & Cleaner – 4 at 1.5	6.0	
Tilt Mechanism – 2 at 1.5	3.0	
Misc. – wire, plugs, brackets, etc.		10.0
TOTAL WEIGHT DMS INSIDE LIFE CELL		103.2

TABLE 56

DMS BOX COMPONENTS (OUTSIDE LIFECELL)

S-Band Transceiver	20.2	
Amplifier/Diplexer	16.8	
Digital Update Link	22.4	
PCM & Timing Equipment Assembly	35.0	
TV Recorder	20.0	
Work Panel Recorder	8.0	
Data Storage Recorder	11.0	
Signal Processor Assembly	11.5	
Animal Data Commutator	1.5	
Temperature Bridges	1.5	
Rate Sensor (Medical)	1.5	
Rate Sensor (Attitude)	1.0	
Attitude Control Electronics	4.0	
Miscellaneous (wire, plugs, brackets)	15.0	
TOTAL DMS OUTSIDE LIFE CELL		169.4

Artificial Gravity Considerations

The creation of artificial gravity effects while in orbit requires some means of altering either the direction or magnitude of the animal's velocity vector. Altering the velocity of the entire spacecraft is too expensive in terms of fuel consumption, especially over long time intervals. A simpler method is to maintain the overall spacecraft velocity and alter only the animal's velocity. Two methods of doing this are: (1) rotate the animals in a centrifuge within the spacecraft, or, (2) rotate the entire spacecraft.

Assuming that the independently orbiting spacecraft design which has been discussed throughout this report is to be modified to produce artificial gravity, the centrifuge concept is ruled out. The centrifuge concept requires complicated mechanization, sweeps large volumes, requires large amounts of power, produces inferior "gravity" gradients, and interferes with the roll-yaw coupling of the onboard angular momentum.

A more favorable method is to rotate the entire spacecraft. Not only does it require less mechanization, power, and payload disruption, but it produces a more stable spacecraft and actually mitigates the temperature swings produced by the space environment.

The rules which govern artificial gravity are (see Fig. 131):

$$m_2 \ell_2 = m_1 \ell_1$$
$$a_1 = \ell_1 \omega^2$$

where m_1 and m_2 are the masses of the two rotating bodies, ℓ_1 and ℓ_2 their distances from the spin axis, ω the rate of rotation, and a the acceleration.

Design objectives are:

$\ell_1 + \ell_2$ be a minimum

m_2 be a minimum

induced gravitational field be as uniform as possible $(\Delta a = 0)$

Practical limits are:

$\ell_1 + \ell_2 \leqq 300$ ft

$m_2 \leqq 100$ lb

± 20 percent gravity variation, $\Delta a = 0.4\, a$

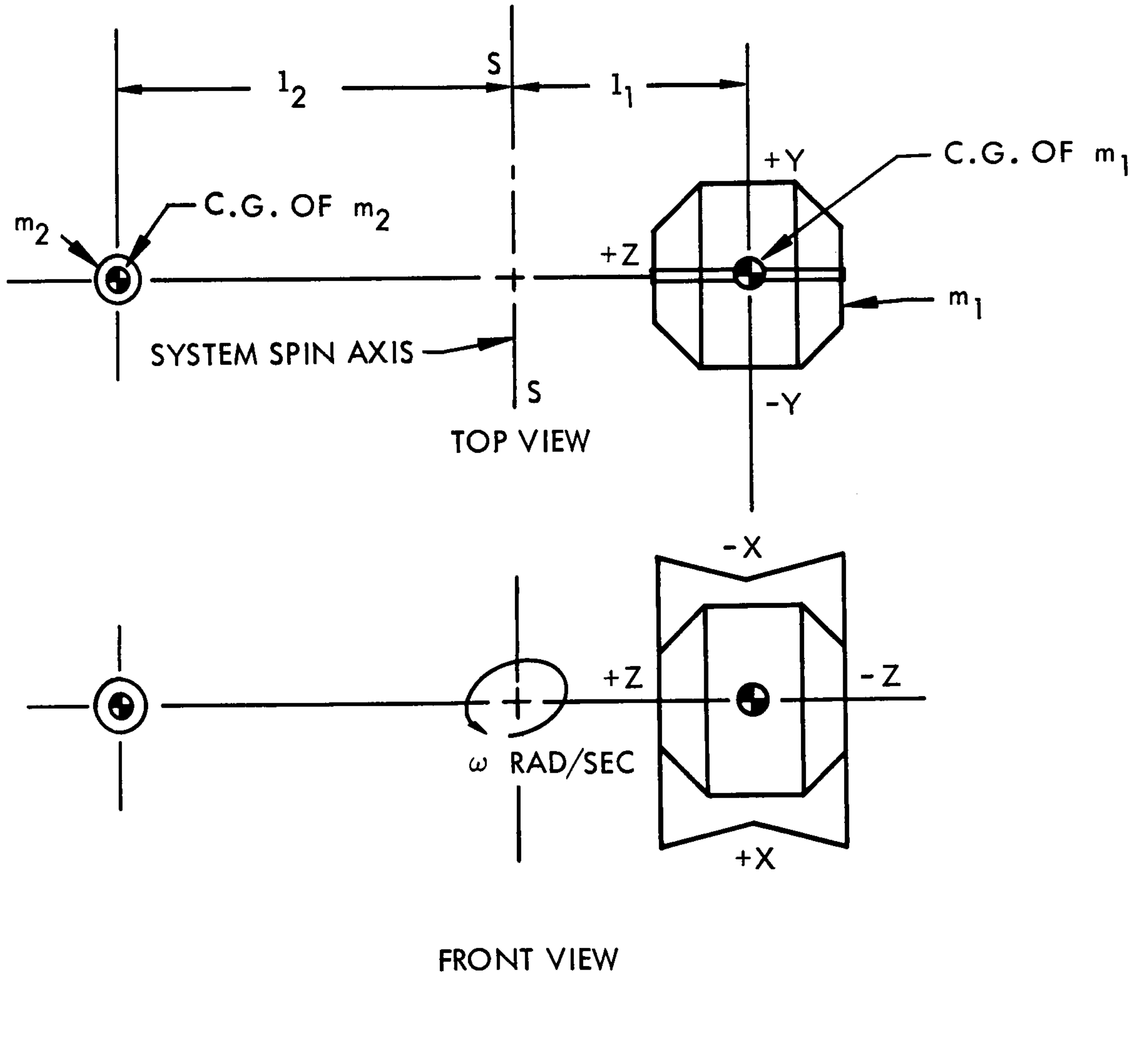


Fig. 131 Artificial Gravity Arrangement

Existing values are:

$m_1 = 5300/g$ slug

ℓ_1 variation $= \pm 2$ ft

$a_1 = 32.2$ ft/sec^2 Case 1 for an induced acceleration of 1 g

$a_2 = 32.2/6$ ft/sec^2 Case 2 for an induced acceleration of 1/6 g

For purposes of design, certain tradeoffs must be made:

$$\ell_1 = \frac{\Delta\ell_1 a_1}{\Delta a_1} = \frac{m_2\ell_2}{m_1} \quad \text{or,}$$

$$m_2\ell_2 = \frac{m_1\Delta\ell_1 a_1}{\Delta a_1}$$

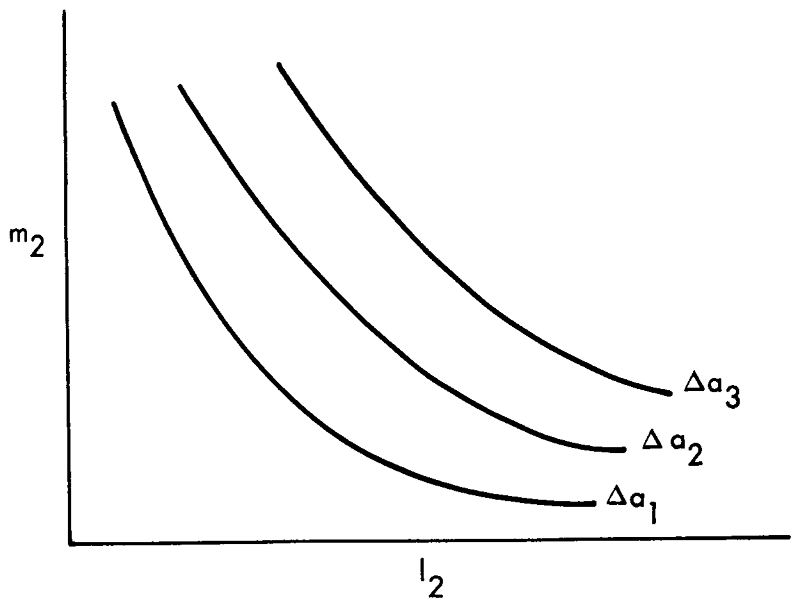
Thus the system weight (m_2) can be expressed figuratively by Fig. 132. For Case 1, using the practical limits above, $m_2 = 182.9/g$ slug, and $\omega = 1.8$ rad/sec. For Case 2, $m_2 = 182.9/g$ slug and $\omega = 0.74$ rad/sec.

The mechanical steps taken to convert the OPE spacecraft from its present configuration (zero-g) to an induced gravity condition are:

1. Reposition attitude control jets to position shown on Fig. 133 . The No. 1 jet(s) in the xz plane is used to maintain spin rate (ω). The No. 2 jet in the yz plane is used to precess the spin axis to keep it pointed at the sun.
2. Add a spin rate sensor for control purposes. This sensor could take the form of a linear accelerometer on a long lever arm with its output filtered to remove the effects of transient linear accelerations.
3. Reduce the magnitude, or reorient the direction of the angular momentum vector of the onboard rotating equipment. A simple fix would be to operate pairs of blowers in opposite directions.
4. Install an extensible mass system in the + Z end of the spacecraft.

Operational steps would be as follows:

1. Proceed with normal operations up to and including orbital injection and separation. For best results, separation should take place with + Y axis pointing as close as possible to the sun.
2. Command the attitude control system to spin spacecraft about Y axis. Solid rocket motors could assist the jets for this purpose.
3. Permit spacecraft to "lock on" the sun.
4. Deploy the extensible mass. For best results, this deployment should take place over a long time span, say 2 or 3 orbits.



$\Delta a_1 < \Delta a_2 < \Delta a_3$

Fig. 132 Relationship of Spin Mass and Weight

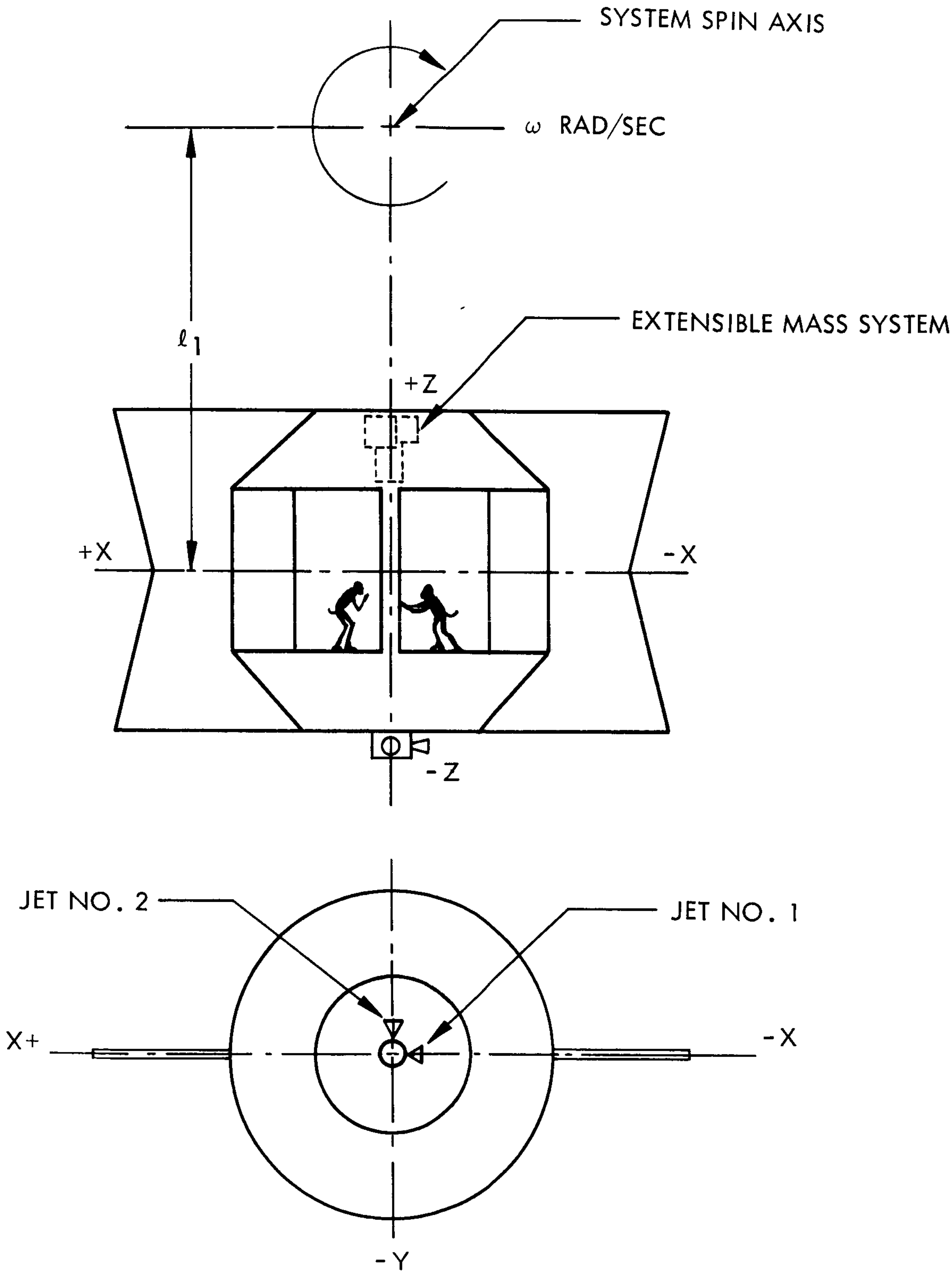


Fig. 133 Attitude Control Jet Location for Simulated Gravity

For retrieval operations, the steps to be taken are:

1. Jettison the extensible mass by operation of a pyrotechnic device.
2. Command despin of the spacecraft. Solid rocket motors or yo-yo's could be used as a supplement to the jets.
3. Dock and retrieve animals in normal manner.

The jettisoning of the extensible mass will result in a slight change of orbit of the spacecraft. The angular momentum of the system about the system spin axis remains the same after jettisoning but due to the release of the centrifugal force, each mass proceeds tangentially increasing its radius and decreasing its angular rate about this system spin axis. The tangential velocity either adds or subtracts from the orbital velocity and the spacecraft, obeying Kepler's laws, adjusts to a new orbit. The maximum $\Delta V = \omega \ell_1$. The rotation of the spacecraft about its Y axis remains the same as before jettisoning.

RELIABILITY CONSIDERATIONS

To obtain a preliminary assessment of the reliability of the payload and the degree of redundancy required to achieve a high reliability, a "Failure Modes and Effects Analysis" (see Appendix D) was conducted. The object of this study was to identify critical components of the systems and the hazard level associated with failure of these components. The objectives of the experiment govern the hazard level definitions which are listed below:

1. Circumstances which would: –
   – cause death to the primates not attributable to extended weightlessness
   – prevent preservation and/or retrieval of the primates.
2. Failures which would reduce the quality of data below an acceptable level.
3. Failures which would reduce the mission duration to six months or less but with adequate data and animal retrieval.
4. Failures which would reduce the mission duration to six months or more with good quality data and animal retrieval.

The combination of hazard level and the probability of component failure was used to determine redundancy requirements.

Only the highest quality of space qualified components were considered, and the individual component reliability figures were obtained from Refs. 16, 17, and 18 and from data drawn from Manned Orbiting Laboratory, AAP, and Gemini experience.

The overall reliability values for each subsystem are shown below. The values shown represent the probabilities that each subsystem performed its required function for a one-year space mission when all redundancies are taken into account. The reliability values given for each component are at 60-percent confidence level (general for space qualified hardware). No better confidence level than 50 percent can be assumed for the predicted subsystem values, as in the absence of performance data, there is an equal chance of them being in error as being correct. Mathematical models for the subsystem are given in the Failure Modes Effects and Analysis in Appendix D.

Subsystem	Reliability
Integrated Thermal and Atmosphere Control	0.9993
Drinking Water Supply and Dispensing	0.99905
Food Supply and Dispensing (Note: No Reliability figure for the food pellets was assigned)	0.9994

Subsystem	Reliability
Animal Retrieving w/o Preservation	0.9986
Animal Retrieving with Preservation	0.9972
Electrical Power Supply	0.9988
Attitude Control	0.9987
Airborne Data Management	0.9497
Waste Water Management	0.9994
Overall Payload	0.942

FINAL MISSION ANALYSIS

This section contains the results of Task 10, Final Mission Analysis. The findings are presented in the form of an operations analysis which traces the OPE mission from prelaunch activities through recovery of the primates. Description of the mission profile and orbit characteristics are covered under the section entitled Preliminary Mission Analysis and Review of S/AAP Configurations.

Positive assurance that the spacecraft and payload designs are consistent with established range policies and procedures and acceptable operational activities is the principal objective of this analysis. Refinements to the day-by-day operations are left to future program phases when the study results can be more profitably implemented.

Prelaunch Operations

Prelaunch operations consist of all of the steps required to ready the spacecraft for flight from the time it arrives at KSC until it is launched. The major objectives of prelaunch checkout are:

- Verify OPE performance as an integrated system, to assure accomplishment of mission objectives.
- Establish compatibility of installed systems and subsystems.
- Minimize the risk of launching an OPE which has degraded performance or delaying the launch of a flight ready vehicle.

The prelaunch checkout of the OPE is performed to achieve maximum confidence of mission success and crew safety (during launch and rendezvous periods) by operating the spacecraft during a series of prelaunch tests extending from arrival of hardware until liftoff.

End-to-end testing is employed as the basic approach to checkout. Input stimuli are applied to identifiable functional flow paths within the subsystem or system under test, and the system response is measured at the end of the flow path or paths. The input is applied at as high an assembly level as possible to permit fault isolation to the lowest replaceable unit.

Whenever technically feasible and practicable, the inputs applied for checkout will be of such a nature that all operational and redundant modes of the system under test will be excited and verified. Electrical, mechanical, and fluid "making and breaking" between connections of flight equipment will be minimized.

Common checkout logic and methods are provided in order to use test data efficiently for the detection of impending malfunctions. Checkout logic and methods used for tests at the factory are ideally the same as those used for tests at the launch site and the postflight examination laboratory. The use of common GSE at all three locations is a GSE-design and program objective.

Assurance that the launch window will not be missed because of OPE delays can be obtained through "dry runs" at the launch site, as well as through planning which takes into account the vehicle activities to be performed. Figure 134 illustrates the time phasing of the major vehicle prelaunch milestones. It is seen that the CM, SM, and SLA will be received at the Kennedy Space Center on the T minus 95 day, and that mating to the vehicle is nominally performed on the T minus 56 day. The day-by-day vehicle activities prerequisite to meeting these milestones are presented in Fig. 135 for the case of CSM/LM/SLA payload. OPE prelaunch operations will be keyed to Fig. 135.

Checkout is accomplished in a building block approach using GSE as the basic tool to monitor and control the tests. The general sequence of the tests is:

- Pre-installation and installation checkout
- Individual subsystem verification
- Integrated OPE checkout
- Launch complex operations

Figure 136 lists the subsystems requiring a performance verification, and illustrates their role in the assembly of an integrated OPE spacecraft. Based upon an assessment of the vehicle activities in Fig. 135, the following OPE test milestones are tentatively established:

Milestone	Work Days Before Launch
• OPE equipments received at KSC	T minus 95
• Installation checkout completed	T minus 91
• Individual subsystems verified	T minus 68
• Integrated OPE (minus primates) checkout completed	T minus 62
• OPE flight readiness demonstrated	T minus 14
• Final combined readiness test completed	T minus 10

The last two of these milestones are met at the launch complex, with the primates aboard the spacecraft. Of course, spacecraft and payload instrumentation checks will be performed via telemetry and hardlines on an intermittent basis up to liftoff.

Government Furnished Equipment (GFE). - The GFE required will include the fixed facilities, instrumentation, and services normally provided by ETR agencies to Range users, in this case the OPE spacecraft contractor. Key items in these three groupings of GFE are presented in Tables 57 to 59. It is seen in Table 57 that the key facilities required are a Spacecraft Assembly Building (SAB), Ordnance Test Building (OTB), and facilities at the pad. The SAB is an existing hangar designated for

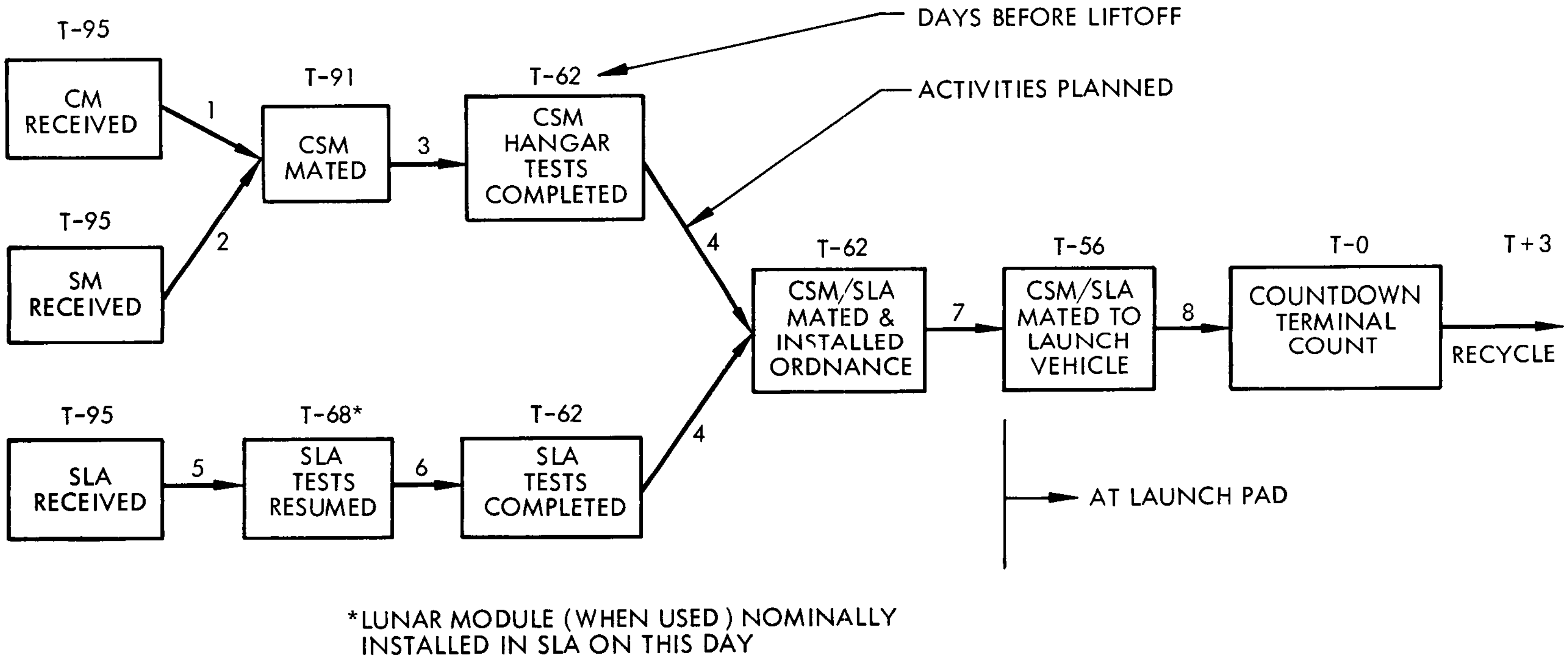


Fig. 134 Vehicle Sequence of Events (Baseline)

Activity Grouping*	Activity
1	Unload CM. Move to MSOB. Inspect. Move to Altitude chamber.
2	Unload SM. Move to MSOB. Inspect. Move to altitude chamber.
3	Mate CSM. Perform cabin leak check. Connect GSE. Install boot and align gimbal. Service ECS. Power-up. Perform cryogenic system verification test. Check ECS functions. Perform simulation altitude run and docking electrical test. Prepare for unmanned altitude run and perform run. Service ECS water and cryo system. Prepare for manned altitude run with fuel cells and perform run. Prepare for manned altitude run after fuel cell cooldown and perform run. Drain cryo, remove stowage, move to - 134 stand, install NE and HGA and test. Perform docking electrical interface test. Move to East Stokes.
4	Mate CSM/SLA. Install quads, install ordnance, prepare for move to vertical Assembly Building.
5	After extended storage period, perform SLA electrical and mechanical work.
6	Mate payloads (assigned to SLA) to lower SLA sections. Perform payload/SLA alignment checks. Mate upper and lower SLA. Install SLA platforms. Perform docking umbilical checks. Install SLA ordnance. Move to East Stokes.
7	Move CSM/SLA to Vertical Assembly Building and mate to launch vehicle.
8	Install dummy tower and set-up complex. Perform simulated integration test. Determine RCS leak and verify functions. Perform MCC-H interface test. Demate dummy tower and mate flight LES. Electrically mate spacecraft (CSM/SLA) to vehicle and perform EDS test. Prepare for, perform and evaluate OAT #1 with plugs in. Prepare for, perform and evaluate OAT #2 with plugs out and prepare for swing arm. Perform swing arm test and flight electrical mate. Reconnect umbilical and evaluate. Prepare for and perform simulated flight test. Prepare for move to pad. Move ML to pad and mate. Move MSS to pad and connect. Connect GSE. Perform MSS/ML GSE test. Perform spacecraft systems verification, and cryo servicing and calibration. Perform vehicle sequence malfunction test. Prepare for and perform FRT. Reconfigure and evaluate. Perform vehicle pressure test. Prepare for and load hypergolic and RP-1. Perform MCC-H interface test. Precount countdown. Complete countdown terminal count.

*Refer to Fig. 134.

Fig. 135 Vehicle Activities Planned

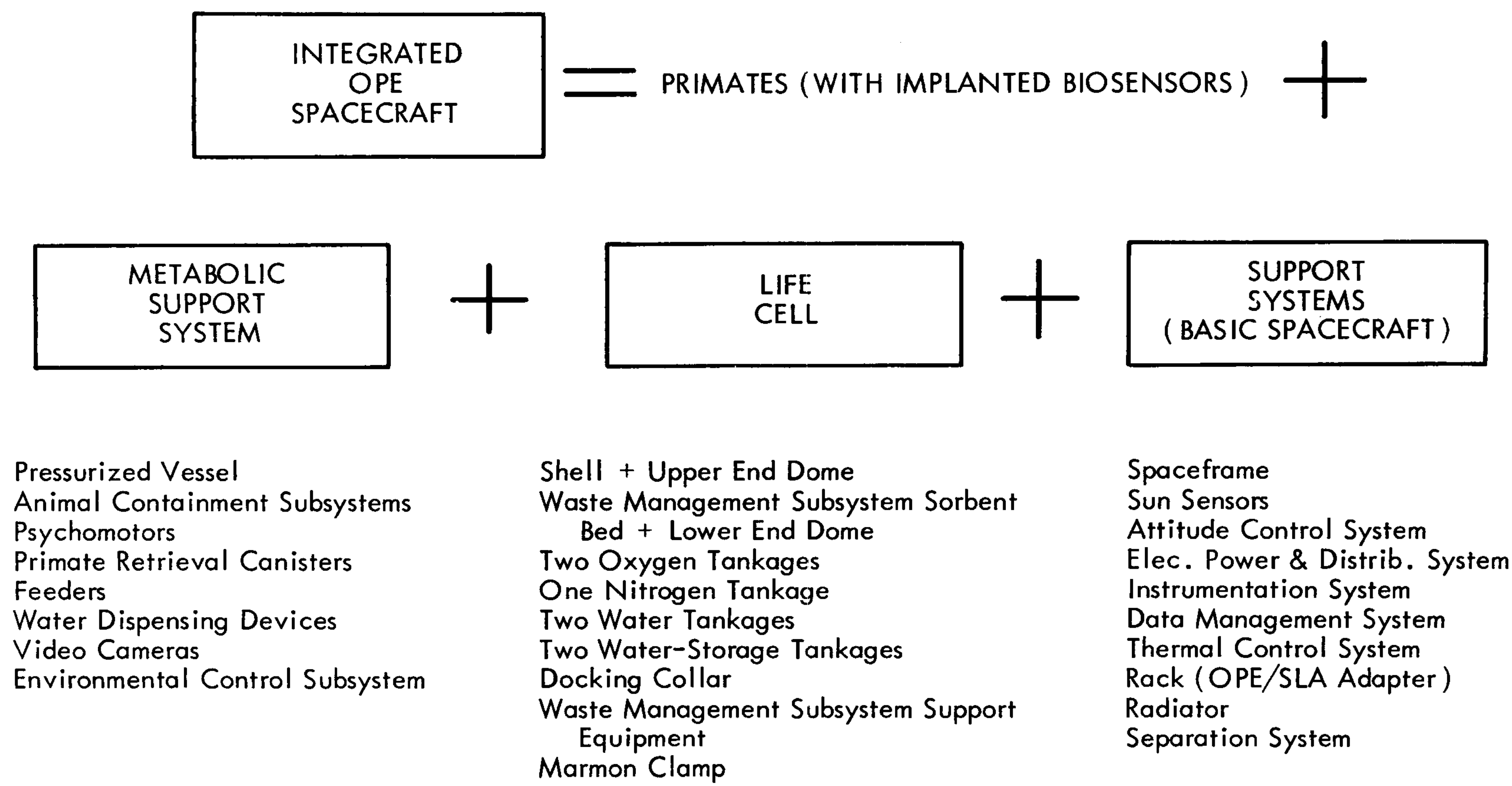


Fig. 136 OPE-Spacecraft Systems Breakdown

TABLE 57

FIXED FACILITIES REQUIRED

• Spacecraft Assembly Building (SAB)
Systems test laboratory Clean room Office area Battery laboratory Shops Gas storage area Pressure test area Bonded stores area Receiving and inspection area Electronics repair laboratory Ground station area
• Ordnance Test Building
Controlled environment room Ordnance storage area Hardwire data links to SAB Antennas Oxygen venting Sensing alarms
• Pad Area
Umbilical fluid lines Umbilical quick disconnect couplings Umbilical vacuum lines Terminal room Service tower power, grounding Communication facilities on service tower Service tower gas and vacuum lines Blockhouse room Hardline data link, pad to SAB

TABLE 58

INSTRUMENTATION REQUIRED

- Electrical and Electronic

Sweep generator	Insulation tester
Noise figure meter	Ammeter
AC voltmeter	Igniter tester
Signal generator	RMS voltmeter
Sweep generator	Battery charger
Oscillator	Differential voltmeter
Square wave generator	pH meter
Function generator	Conductivity meter
RF wattmeter	Digital voltmeter
Impedance bridge	VTVM
Deviation meter	Power supply
Camera	Audio oscillator
Oscilloscope	Variable attenuator
Dual beam oscilloscope	Multimeter
Plug-in units	Timer
Counter	Decade resistor box
Frequency converter	Variac
Time interval unit	Bond meter

- Dimensional and Physical

Pressure gauge
Ultrasonic cleaner
Balance scales
Torque wrenches
Micrometers

TABLE 59

ETR SERVICES REQUIRED

- SAB command antenna
- RF frequency protection
- Magnetic tapes from all ETR downrange stations receiving OPE data
- Recordings of SAB-ground station communications from final combined readiness test until spacecraft LOS
- Terminal room wideband data link
- Inter-facilities intercom
- MOPS services
- Telephone services
- Teletype services
- Range timing signals
- Visual countdown and status indicators
- Reproduction services
- Standards lab services
- Maintenance services
- Fire protection and medical ambulance
- GSE and monkey trailers air conditioning
- X-ray coverage
- Cleaning service, general supplies, and hardware
- Office equipment and equipment maintenance
- Emergency power to monkey trailers
- Vehicle services for hauling and lifting
- Film processing
- Fluids, gases, and chemicals
- Chemical and physical analysis
- Test instrument maintenance and calibration
- Documentatry photography
- Data processing and disposition

receiving, inspection, checkout, and assembly of the spacecraft, and is also used as general offices, a storage, and repair area. The OTB is a remote existing building designated for testing of spacecraft tankages and for installation of pyrotechnic devices.

At ETR, program range support requirements are levied through the use of National Range Documents (NRDs). These are as follows:

- Operational Program Estimate (OPE): introduces the Range to the program; serves as the authority for the Range to program the necessary resources.

- Program Requirements Document (PRD): A combination of about 90 standard forms common to all ranges which delineate in considerable detail all services and equipments required, and other specific information about the program.

- Program Support Plan (PSP): ETR's answer to the requests contained in the PRD.

- Operations Requirements (OR): describes the specific requirements necessary and updates the PRD.

- Operations Directive (OD): Specifies how ETR will meet the operational requirements, and mobilizes and commits the resources of ETR.

These five NRD documents are essential to the prelaunch planning required for the OPE spacecraft. The items listed in Tables 57 through 59 are formally requested in the PRD, usually 18 to 24 months prior to the first scheduled test at ETR. Changes in these requests and consequently changes in ETR's responses are documented via monthly publications of the PSP and OR. The OD is the final, negotiated document available during the week prior to launch.

Ground support equipment (GSE). - The GSE equipment requirements will be established by three major program phases:

- Subsystems/systems tests at the factory
- Subsystems/systems tests and prelaunch operations at the launch site
- Postflight operations at the recovery site

To afford a multipurpose usage of GSE and for other reasons (i.e., budgetary reasons), the monitoring tasks to be accomplished at all three locations are accounted for in the design of each piece of GSE. The GSE can then be divided into the following major functional groups:

- Handling equipment
- Checkout equipment
- Servicing Equipment

Key items included in the individual equipment groupings are identified in Tables 60 through 62.

TABLE 60

HANDLING EQUIPMENT REQUIRED

• Transport and test dollies
• Hoisting fixtures and slings
• Moment of inertia fixtures
• Weight and center of gravity fixtures
• Alignment fixtures
• Recovery handling equipment
• Lifecell work stand
• Shipping and storage containers
• Protective covers

TABLE 61

CHECKOUT EQUIPMENT REQUIRED

- Animal simulators
- Lifecell test set
- Thermal control test set
- Atmosphere control test set
- Metabolic control test set
- Rate gyro holding fixture
- Electrical power-supply test set
- Attitude control test set
- Ground power unit
- Telemetry calibration and sensor power unit
- Recorder console
- Prelaunch monitor unit
- Waste management test set
- Mass spectrometer
- Telemetry ground station
- Lifecell television receiving and monitoring console
- Lifecell ground air conditioner

TABLE 62

SERVICING EQUIPMENT REQUIRED

• Astronaut belt ties (to tether canisters during EVA)
• Mechanical GSE/vehicle interface connectors
• Electrical GSE/vehicle interface connectors
• Fluid GSE/vehicle interface connectors
• Umbilical connections for environmental control
• Pad to blockhouse cabling
• Hardline for telemetry video
• Blockhouse payload control and monitor console
• Spacecraft/S-IVB separation pyrotechnics analyzer
• Postflight primate-retrieval-canisters environmental control system (tentative)

Spacecraft disassembly. - Shipment of an OPE spacecraft (minus primates and expendables) to the Kennedy Space Center will be executed such that the potential for damage en route is minimized. Since transportation loads are recognized through experience as sometimes approaching flight loads, shipment as a completely assembled spacecraft is assumed the selected method. With this assumption, the spacecraft would be packaged utilizing the spacecraft rack/SLA hardpoints also used during flight. Prelaunch operations would therefore begin with spacecraft disassembly, followed by inspection and checkout of subsystems.

The first level of disassembly of the spacecraft (minus primates) results in the following pieces of major hardware:

- Radiator and radiator fluid lines
- Lower end dome of lifecell
- Metabolic support system
- Basic spacecraft + OPE/SLA adapter ("rack")

The radiator and radiator fluid lines located outside of the lifecell wall are first removed. The marman clamp which attaches the lower end dome of the lifecell to its shell is detached, and the dome lowered out from the spacecraft through the rack. This end dome of the lifecell also provides the housing for the sorbent bed of the waste management subsystem, and is a completely passive unit free of cabling and line connections with the rest of the spacecraft. With the end dome removed, simple access to all electrical and fluid connectors (approximately 24) between the lifecell and spacecraft is provided and these connectors ("snap-lock" types) are disconnected. The metabolic support system is then lowered out from the spacecraft through the rack.

The equipments which comprise the basic spacecraft and rack, including the docking collar and retrieval canisters, remain intact during this disassembly process. During disassembly, the spacecraft is supported "off the floor" by an overhead hoist. The hoist supports a "simulated docking probe" mated to the spacecraft docking collar, in the same fashion as would be the case during CSM/OPE docking in flight. An adjustable platform located beneath the spacecraft supports the metabolic support system when lowered onto it. This adjustable platform also serves as a work stand.

The primate retrieval canisters are disassembled from the spacecraft after two hatches (located along the sides of the docking collar structure) are removed. The canister flange bolts and lower pressure seal are removed, and the canisters withdrawn.

During the early "all systems checks" at the pad, electronic simulators are used in place of the two primates. The primates with implanted biosensors are installed late in the launch countdown. Primate installation is accomplished by removing the two hatches and the upper dome of the primate retrieval canisters. The hatches and dome are moved laterally through the SLA crew access door for temporary storage outside. The electronic simulators are removed and the primates are inserted into the retrieval canisters, and the reverse of the disassembly process is completed. Canister pressure leak tests are then performed and followed by spacecraft final Confidence Readiness Tests (CRTs).

Descriptions of prelaunch tests. - Fulfillment of the checkout objectives is the purpose of the prelaunch tests. Concepts and criteria of these tests delineate the modes of testing. Schedule limitations give information relative to when the tests are to be completed, and schedule milestones are met to assure compliance with the schedule limitations. GFE and GSE are used to perform the tests. The tests to be performed are further described in succeeding subsections.

Pre-installation and installation checkout: Pre-installation and installation checkout is performed during the final assembly and acceptance flow as shown in the typical schedule in Fig. 137. As indicated in the figure, the spacecraft arrives at KSC as a completely assembled configuration. There, modification to the configuration is performed in conjunction with OPE experiment checkout.

As part of the pre-installation checkout, the metabolic support system and basic spacecraft will undergo a functional verification. KSC, working in conjunction with ETR agencies, is responsible for meeting the facility and equipment requirements established in the PRD and OR documentation. The spacecraft contractor should supply all interconnecting cabling required to conduct the electrically mated checkout shown in Fig. 137. Compatible GSE is used in functional verification testing both on the bench and when installed in the vehicle.

After installation, the metabolic support system is checked to ascertain proper mechanical installation on the lifecell hardpoints. Connector mating and visual checks are performed. Electrical power is applied and voltage checks made as a cursory indication of proper electrical installation.

Individual subsystem verification: Individual subsystem and spacecraft verification are accomplished during the initial portions of the typical KSC facility flow shown in Fig. 138. The performance of individual subsystems and their respective interfaces are demonstrated predominantly by controlling the monitoring stimuli and measurements through GSE. As indicated in the figure, some tests are performed in sequence at different facilities for schedule efficiency.

Integrated OPE checkout: The integrated OPE checkout is conducted in the SAB (Spacecraft Assembly Building) in four parts:

- Integrated systems checkout to demonstrate system compatibility by operating the electronics subsystems through all modes of operation.
- Electromagnetic interference (EMI) checks in a controlled EMI environment to verify payload-spacecraft-vehicle electromagnetic compatibility.
- Mission-oriented checkout to verify particular mission modes and operations, possibly a complete "walk-through" or "dry run" of the planned prelaunch activities with the complete spacecraft.
- Thermal-vacuum tests to demonstrate environmental control inside the lifecell under simulated space thermal.

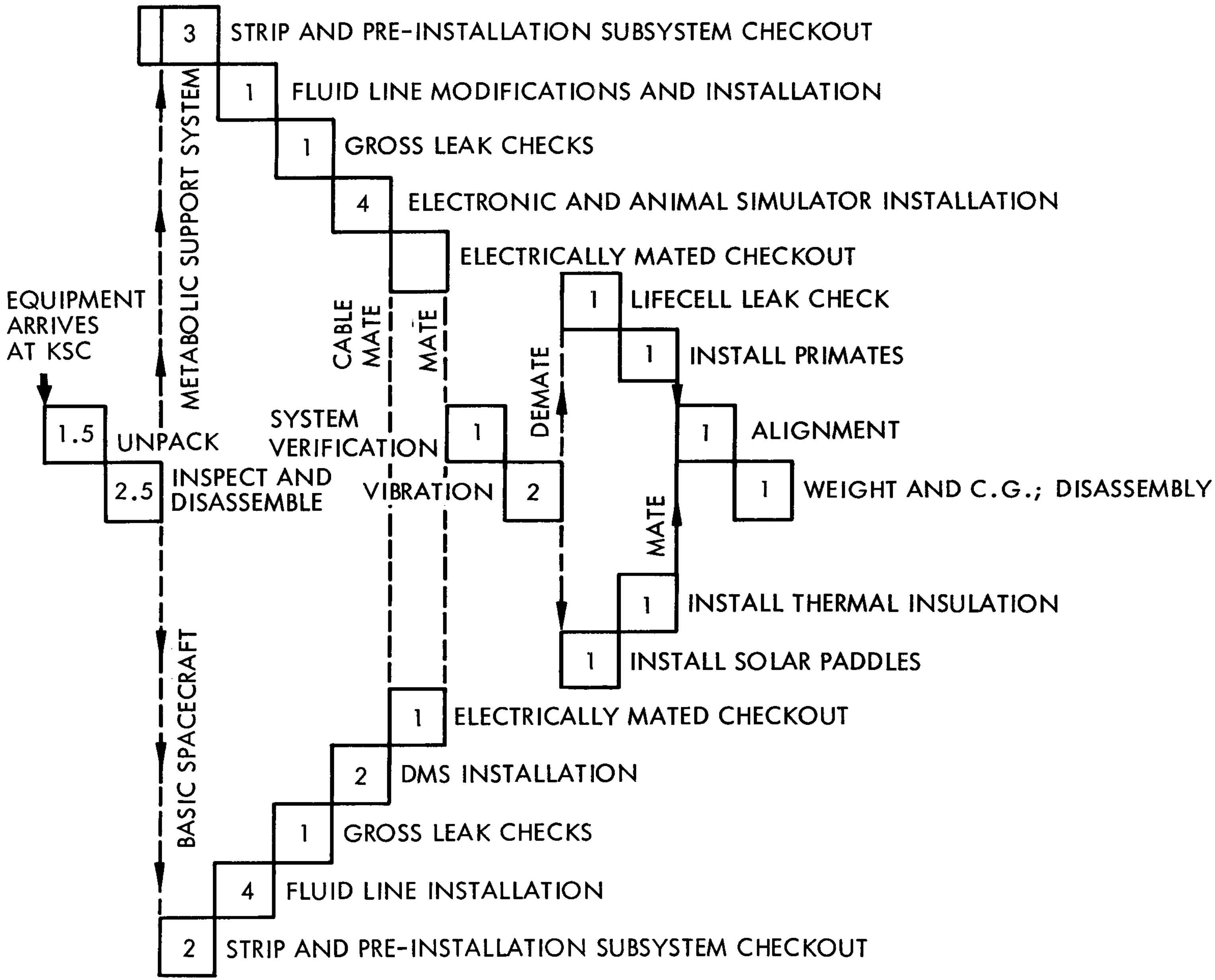


Fig. 137 Prelaunch Final Assembly and Acceptance Flow

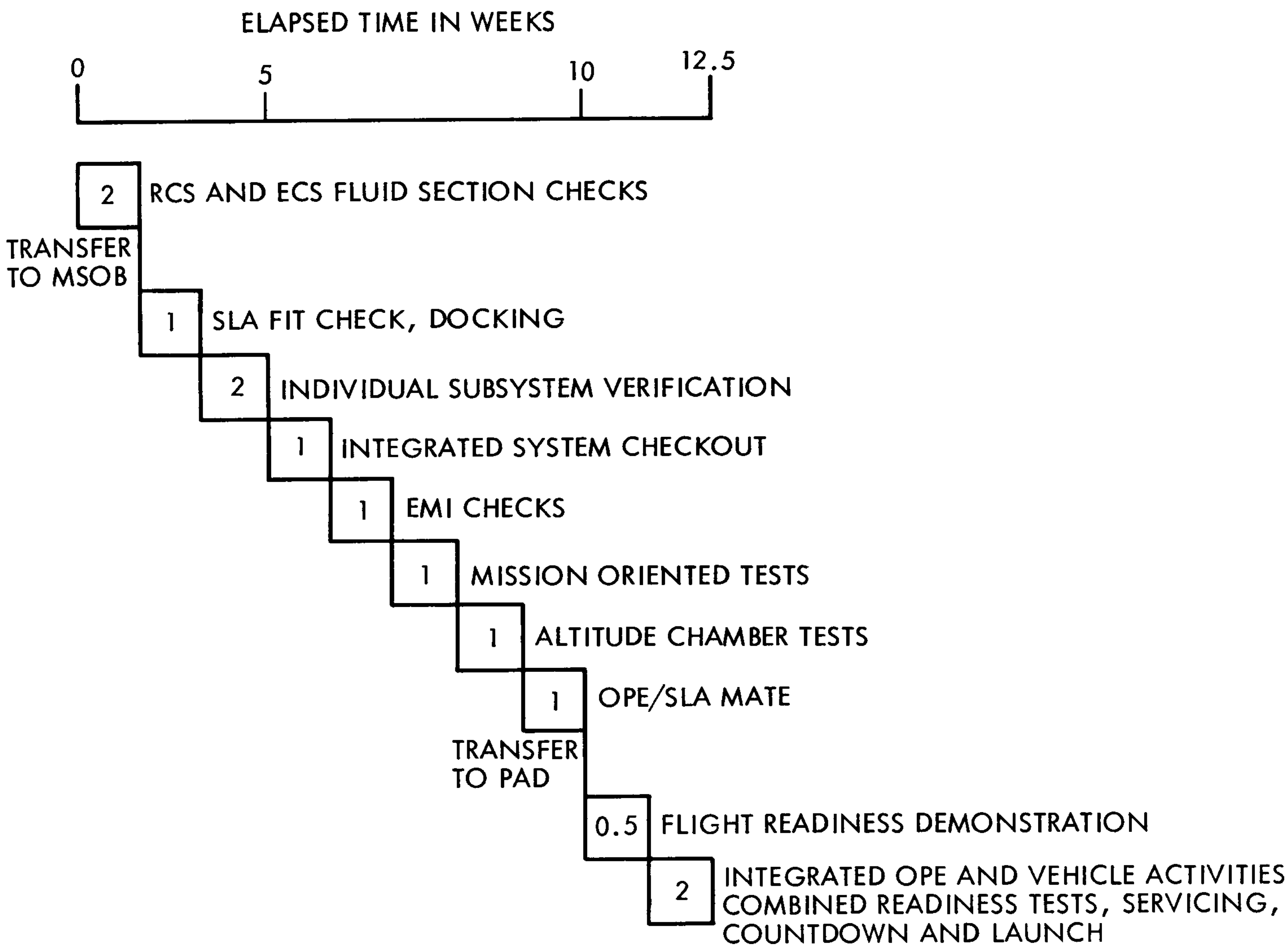


Fig. 138 KSC Facility Flow

Launch complex operations: When the OPE is installed below the CSM inside the SLA on the vehicle, final integrated checks, servicing, and countdown are performed. The primates, with implanted biosensors, are installed and their "electronic simulators" withdrawn during the countdown at a point within two weeks before launch.

Launch Operations

Launch operations comprise all events which take place from the time of SIB launch until the OPE is placed on orbit. The following paragraphs discuss the sequence of these events and the data requirements during this phase of the mission.

Sequence of launch events. - Table 63 is an abbreviated version of the sequence of events which will occur during the period from liftoff to final injection of the OPE into the 260-n. mi. circular orbit. No changes in requirements of the vehicle trajectory program are necessary. The general requirements for OPE injection into a 260-n. mi. circular orbit, using the procedures preliminarily planned by the AAP for Lunar Module flights are applicable. For preliminary design purposes, the AAP Cluster A reference mission profile and the nominal Saturn 1B environmental spectra have been used. Table 62 indicates that the events required are the same as those to be required for the Lunar Module flights.

TABLE 63

LAUNCH SEQUENCE OF EVENTS

(Flight B1)

Time (sec)	Event
0	Liftoff
15	Complete vertical rise; begin pitch program
140	Complete pitch program; booster engines cutoff
146	Ignite S-IVB second stage
(*)	Jettison S-IB first stage
626	S-IVB cutoff
---	Transpose CSM and dock to OPE
---	Jettison SLA/OPE-Rack
---	Jettison S-IVB

TABLE 63

Time (sec)	Event
---	Perform service propulsion system (SPS) burn to inject into 260-n.mi. circular orbit
---	Release OPE

*At 98% thrust of S-IVB

Fig. 139 illustrates the launch operations sequence. Picture (1) shows the lift-off configuration. The vehicle shown is the Saturn IB comprised of the S-IB first stage, the S-IVB second stage, and the Instrumentation Unit (IU). The vehicle's payload consists of the CSM, the Apollo escape tower, the SLA, and NASA-approved experimental payloads. The OPE is located inside the SLA using the existing SLA hardpoints and OPE attachment hardware (the rack). Complex 37 (Pad A or B) at ETR is the launching site. An initial (parking) orbit of 80 by 120 n. mi. is reached about 10 minutes after liftoff, prior to which the expended first stage and the escape tower are jettisoned.

Picture (2) shows the orbital vehicle at 80 by 120 n. mi. during the initial portions of the CSM transposition event required to hard dock with the OPE. Transposition is required to align the CSM nozzle away from the OPE, to permit the CSM propulsion events necessary to complete the launch sequence. CSM/SLA release is provided by existing Apollo hardware and procedures.

Picture (3) shows the transposed CSM docked to the OPE via its docking collar and the CSM docking probe. The docking collar is essentially the same as will be used by the Lunar Module on other flights, and so the same (Apollo) docking mechanics and command procedures apply. A microswitch (on docking collar) closure will verify that a hard dock has been achieved. After these two events (docking and hard-dock verification), the integrated OPE-rack, SLA forward and aft skirts, IU, and S-IVB are jettisoned. Jettison is provided by OPE pyrotechnic devices similar to those to be employed on Lunar Module flights. The jettison command is initiated by the command pilot.

The time after liftoff that the S-IVB and SLA are jettisoned will be an important consideration in OPE final design. This is so because prior to jettisoning, the spacecraft radiator (facing the S-IVB) and the spacecraft solar paddles (partially shadowed by the SLA) will be partially ineffective. Their functioning, however, will not be absolutely required to support the spacecraft payload systems during the period from liftoff to liftoff plus 15—30 minutes. Jettison of the SLA and S-IVB is therefore required prior to 15—30 minutes after liftoff. If this requirement cannot be met by the prevailing AAP operations, an evaporative cooling heat exchanger may have to be added to the spacecraft and about 10 percent additional area may have to be added to the solar paddles.

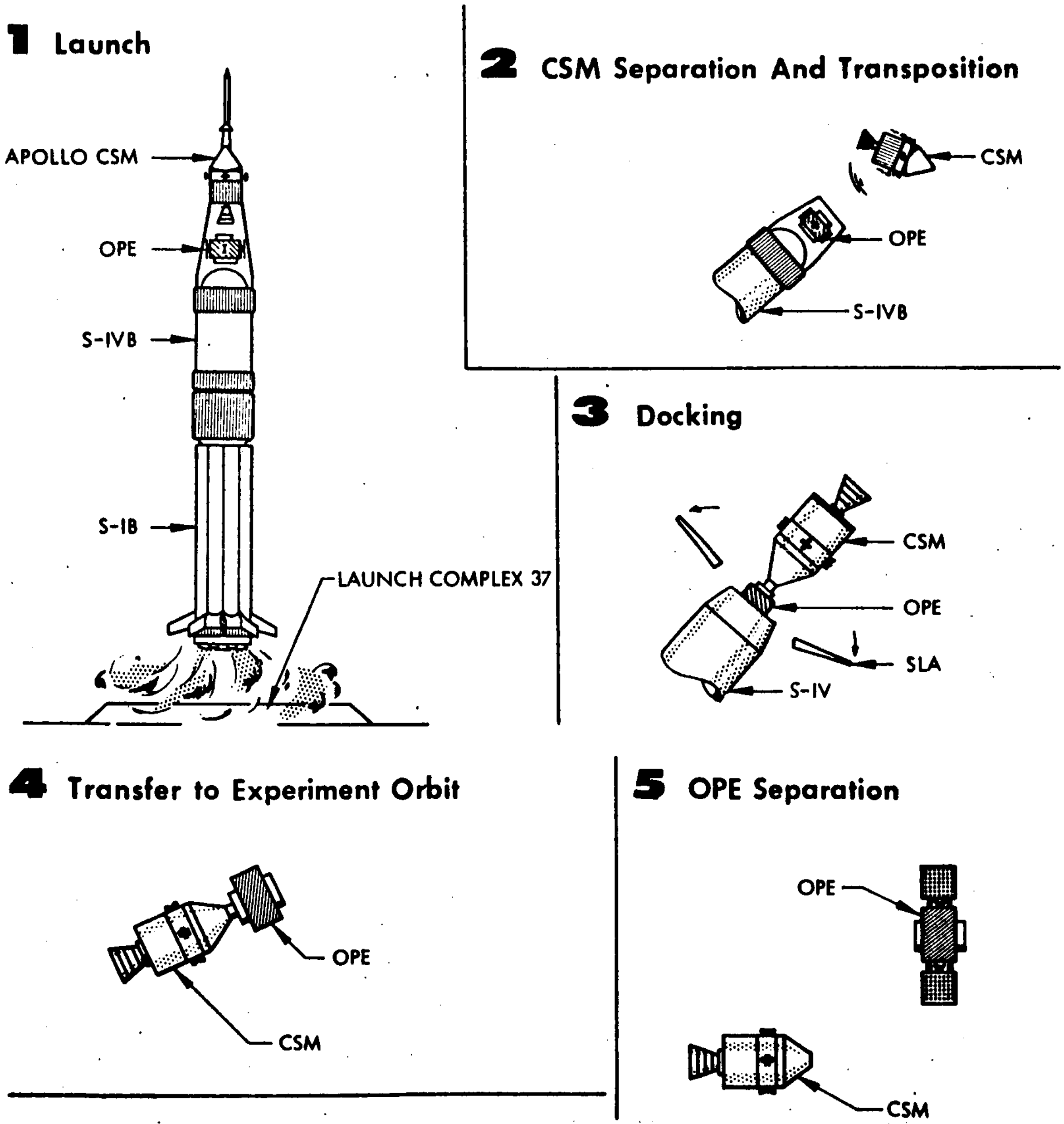


Fig. 139 Launch Operation Sequence

The requirement for jettisoning the S-IVB and SLA by 15—30 minutes after liftoff can be accommodated by the AAP, providing no other experiments requiring the use of an Orbiting Workshop (modified S-IVB) are aboard. That such experiments are not aboard has been the assumption, but its validity will not be ascertained until the assignment of specific experiments to specific Cluster B vehicles is made.

Picture (4) shows the OPE/CSM during the coplanar transfer from 80 by 120 n. mi. to 260 by 260 n. mi. CSM SPS propulsion to 250 n. mi. (46 minutes) is first performed, and terminal phase maneuvers at 250 to 260 n. mi. (35 minutes) are then conducted to minimize the injection errors.

Picture (5) shows the OPE separated from the CSM at the desired circular orbit altitude of 260 n. mi. Separation is performed and controlled by the CSM. The CSM flys "out of the hole" after its docking probe is released.

Launch data requirements. - Knowledge of the acceleration, vibration, and noise levels actually experienced by the primates during powered flight is required to assure that the initial telemetered data will be meaningful. This is so because of uncertainties concerning the expected severity of these levels and consequently their medical and biological effects upon the primates. Uncertainties exist at this time because:

- With respect to the expected environmental levels, flight-to-flight variations from nominal have not been considered.
- Trajectory ground rules used are subject to expected variations.
- The specific configuration of the Saturn 1B to be used (i.e., with/without Minuteman strap-ons) is undefined.

In-flight measurements of vibration, noise, and acceleration must therefore be required at this time. The continuous measurement of these environments during powered flight, at several points common to the OPE and the SLA, are specifically recommended. The measurement of energy levels in discrete ranges of frequencies, and not the measurement of complete spectra, would suffice. Data storage on the OPE telemetry recorders, for later transmission to earth, should be investigated during the OPE final design phase. The transmission of any OPE data to earth during powered flight is not required.

In-Orbit Operations

Within the capabilities of the MSFN, a specific amount of spacecraft status and experiment data, which is a function of station contact time and communication bandwidth, will be transmitted directly to earth. OPE data handling and processing involves the MSFN ground stations, the NASA communications network (NASCOM), the Mission Control Center, data reduction centers, and data distribution and archiving centers. Certain of these centers will also be utilized in the command and control of the spacecraft. In this section, a survey of existing and planned facilities is documented, a facilities analysis made, AAP augmentation requirements determined, and in-orbit operational modes analyzed. A brief experiment contingency analysis is presented last.

Network description. - The Manned Space Flight Network consists of the Mission Control Center (MCC), 14 primary ground stations distributed around the world, five tracking ships, eight aircraft, and four ground stations with limited capabilities. The NASA Communications Network (NASCOM) provides voice, data, and teletype circuits between MCC and other elements of the MSFN.

Mission Control Center: The Mission Control Center (Houston) will consist of two Mission Operation Control Rooms (MOCR's) manned by flight control personnel. Each MOCR will have a set of six Staff Support Rooms (SSR's) so that two separate missions can be supported by the MCC. The SSR's are:

- Flight Dynamics SSR
- Vehicle Systems SSR
- Life Systems SSR
- Flight Crew SSR
- Networks SSR
- Operations and Procedures SSR

The technical specialists located in these areas are responsible for supporting their counterparts in the MOCR. They perform data analysis, determine long-term performance trends, compare these trends with baseline data, and transmit this information together with recommendations to the MOCR personnel. Flexible and varied combinations of display data are provided by computer driven display generation equipment controlled from the consoles in the MOCR and SSR's. Large wall displays in the MOCR and support rooms provide television and plotting data for group presentation.

Current planning for the OPE-spacecraft flight calls for the provision of OPE support at either of the following locations:

- Annex to Vehicle Systems SSR

- A new "Experiment Support SSR" at MSC, reasonably accessible to the MOCR but in a nearby building, if necessary

Either location conforms to the "equal but separate" philosophy believed required by the AAP for the support of experiments.

Manned Space Flight Network: The MSFN for the in-line Apollo program, as presently constituted*, will consist of 14 ground stations, five ships, and eight aircraft. Four other ground stations and two ships will be available for limited mission support and the three MSFN deep space stations will be backed up by the DSIF (Deep Space Instrumentation Facility) stations at adjacent locations. The 14 primary ground stations are shown at approximate locations on the map of Fig. 140. Stations at Kano (Nigeria), Grand Turk Island, Tananarive (Malagasy Republic), and Canton Island are not scheduled for S-band equipment, but do have limited VHF and recording capability.

*NASA-GSFC, "Technical Manual, Manned Space Flight Network, Apollo Ground Systems," Preliminary, MS-401, Feb 1966

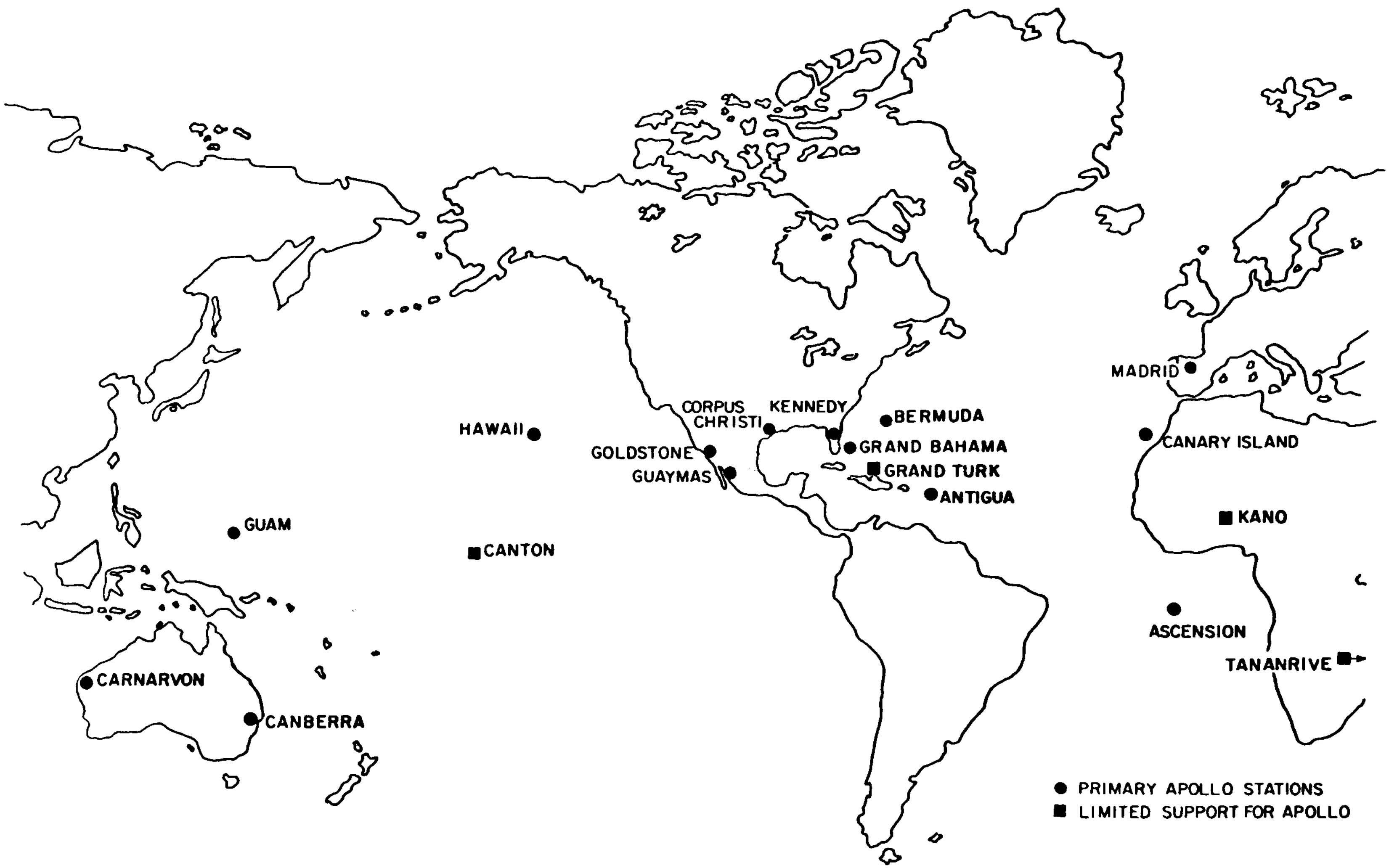


Fig. 140 Primary Apollo MSFN Ground Stations

Some stations have separate buildings for the S-band equipment, with computers, telemetry processing, teletype, and communications interfaces being in another building; the functional areas are the same, however. Present plans call for Guaymas, Ascension, and Carnarvon to be equipped with a display system. This requires an operations room with a layout similar to Fig. 141.

The equipment complement at the ground stations varies somewhat at the various locations. Typical equipments at the ground stations available for use by the OPE-spacecraft will be described in the following paragraphs.

In general, dual USB stations* have the capability of receiving two PM and two FM modulated S-band down-link carriers simultaneously and transmitting two up-link carriers. A typical equipment list for a dual station is shown in Table 64. Some dual stations may have only three S-band receivers. Others may have additional VHF receivers and diversity combiners. Four of the five dual stations are equipped with C-band radar and can track a beacon-equipped spacecraft to assist in determining ephemeris.

Single USB stations** have less S-band capability, but the remaining equipments are quite similar to those at dual stations. There is only one ranging system, two receivers, one signal data demodulator, and one data subcarrier oscillator. With these differences noted, Table 64 is representative of a typical equipment complement at a single USB station. Three of the six single USB stations have C-band tracking radar. The equipment extras at the dual USB stations are not required for the OPE mission.

A change (at all stations) from C-band to X-band tracking radar is planned. The OPE is equipped with X-band radar for OPE-CSM communications.

MSFN equipments for communications and data handling: The pertinent characteristics of those equipments which have direct bearing on OPE communications and data handling are summarized in the following paragraphs. Pertinent data on each of the following equipments are presented sequentially:

- S-band receivers
- VHF receivers (not required by OPE preliminary design)
- S-band up-link and down-link transmission
- S-band up-link transmitter
- Magnetic tape recorders
- Computer system equipments
- Data modems
- Antennas
- RF and data handling equipments
- Apollo television monitor
- OPE television reception and display

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* Hawaii, Guam, Carnarvon, Ascension, Cape Kennedy.

** Antigua, Corpus Christi, Canary Island, Grand Bahama, Guaymas, Bermuda.

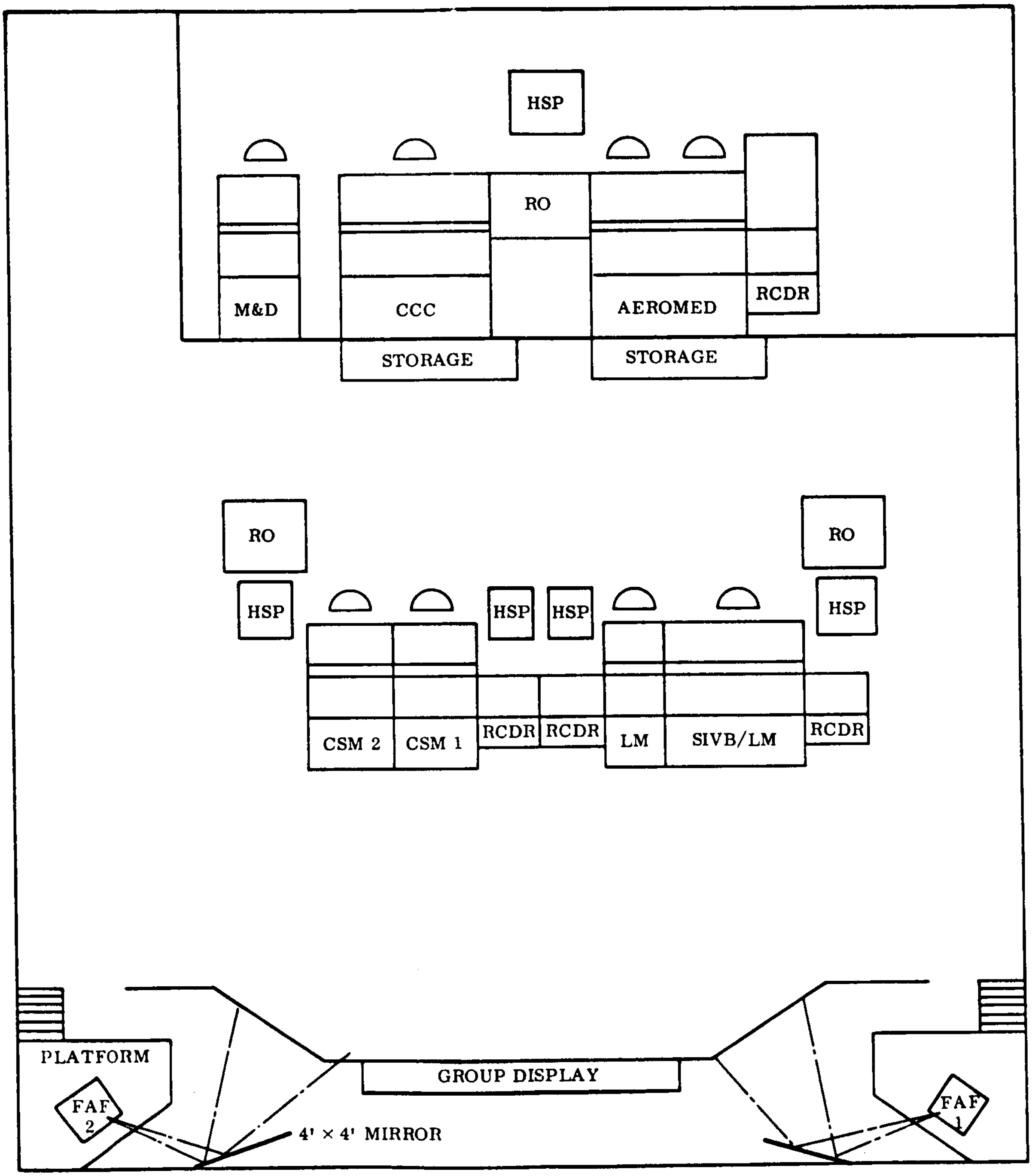


Fig. 141 Operations Room Layout for Ground Stations Having Display Systems

TABLE 64

TYPICAL DUAL USB STATION INSTRUMENTATION

● S-Band Equipment	
1	Antenna System Electronics
1	Acquisition Aid
2	Ranging System
1	Collimation System
4	Receivers
2	Signal Data Demodulators
1	Tracking Data Processor
2	Data Subcarrier Oscillators
1	Transmitter
1	Timing System
1	30' USB Antenna
● VHF Equipment	
6 or more	VHF Receivers (500 kc BW)
2	DCA-510 Diversity Combiners
2	AN/GRT-3 Transmitters
1	Crossed Dipole VHF Array
1	Quad-Helex VHF Array
● UHF Equipment	
2	AN/FRW-2A Transmitters
2	240 D-2 Power Amplifiers
2	Quad-Helix UHF Array
● Telemetry Equipment	
2	PCM Decoms
1	PCM Simulator
1	TDS-90/90 FM/FM Decom
1	TDS-90/30 FM/FM Decom

TABLE 64 (Cont.)

TYPICAL DUAL USB STATION INSTRUMENTATION

● Recorders	
2	MINICOM 22 Wide Band
1	CEC VR 3600
1	FR-1100 Ampex Narrow Band
2	FR-600 Ampex Voice Recorders
2	Visicorders – 906B
2	958 Sanborn Recorders
2	Brush Multichannel Recorders
● Data Processing	
2	642B Univac Computers
2	1540-A2 Mag Tape Unit
2	1232 I/O Console
2	1259 TTY Adaptor
2	2010 DTU (Dual)
1	HS Printer
1	Computer Address Matrix
	Other Periphery Equipment as required

Characteristics of the S-band ground receivers are given in Table 65. As indicated in the preceding subsection, there are two receivers at a single S-band site and three or four at a dual site.

TABLE 65

S-BAND GROUND RECEIVER CHARACTERISTICS

• Frequencies (fixed in advance)	
CM (PM)	2287.5 Mc
CM (FM)	2272.5 Mc
LM (PM or FM)	2282.5 Mc
S-IVB (FM)	2277.5 Mc
• IF Bandwidth	
PM	3.3 Mc
FM	10.0 Mc
• Noise Figure	2.0 db (max.)
• Dynamic Range	-80 to -157.5 dbm
• TLM Subcarrier	1.024 Mc
• Voice Subcarrier	1.25 Mc
• TLM Subcarrier Demod. Rate	1000—200,000 bps
• Subcarrier Demod. Capability	1 set of PM 1 set of FM = Single USB Station
	2 set of PM 2 set of FM = Dual USB Station

VHF receiver characteristics are given in Table 66. OPE VHF transmission is not now planned. Many of the ground stations are being equipped with dual diversity receivers which results in a nominal 2.8 db improvement in performance over a single receiver. All stations applicable to low-earth orbit missions will have VHF reception capability.

TABLE 66

VHF GROUND RECEIVER CHARACTERISTICS*

• Demodulation Capacility	FM/FM PCM/FM SS/FM PCM/PM AM PAM/FM/FM FM/FM/FM
• Freq. Range	215 - 260 Mc
• IF BW	12.5 kc - 3.3 Mc (Plug-in Units)
• RF Multicoupler	8 Channel
Noise Figure	<9 db
VSWR	<1.5 to 1
Isolation	>55 db between any two outputs
Input/out impedance	50 ohms

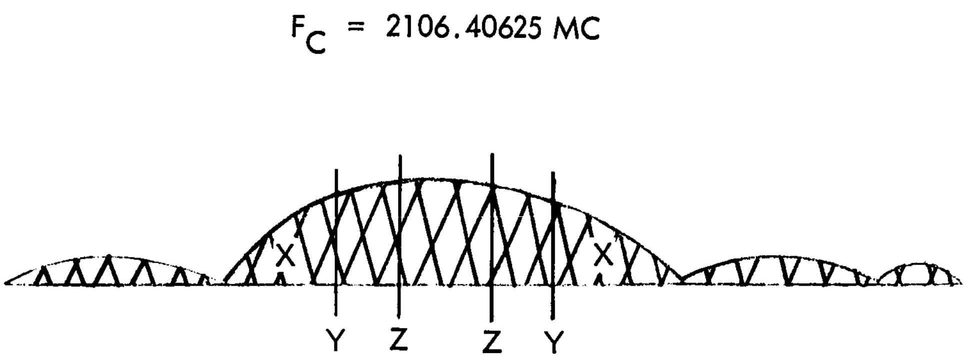
*NASA GSFC, "Manned Space Flight Network, Apollo Ground Systems," MG 401, Section 6, Feb 1966.

Representative (not necessarily all-inclusive) ground-to-spacecraft transmission modes for the AAP are presented in Table 66. Ground transmission to the OPE spacecraft must use Mode D-2. The carrier frequency is 2106.40625 Mc. The up-link S-band RF spectrum for normal mode communications to the OPE is shown in Figure 142.

TABLE 67

APOLLO TRANSMISSION MODES, GROUND-TO-SPACECRAFT

Mode	Channel	Modulation
A-2	VHF Voice	AM
B-2	C-Band Radar	Pulse Radar
C-2	S-Band : Command Ranging Carrier	PCM/PM/PM PM ---
D-2	S-Band: Command Voice Ranging Carrier	PCM/FM/PM FM/PM PM ---



X = UP RANGING CODE SPECTRUM
Y = 70 KC UP-DATA (CAN ALSO BE USED FOR VOICE IN A CONTINGENCY)
Z = 30 KC VOICE

Fig. 142 Up-Link S-Band Spectrum

Representative (not necessarily all-inclusive) spacecraft-to-ground transmission modes for the AAP are presented in Table 68. OPE spacecraft transmission to the ground must necessarily use Mode E-1 which provides, unlike all other modes, FM television transmission. The carrier frequency is 2287.5 Mc. Three types of modulation (PCM/PM/FM) are available and will be used. The down-link S-band RF spectrum for normal mode communications from the OPE is shown in Fig. 143.

TABLE 68

APOLLO TRANSMISSION MODES,

SPACECRAFT-TO-GROUND

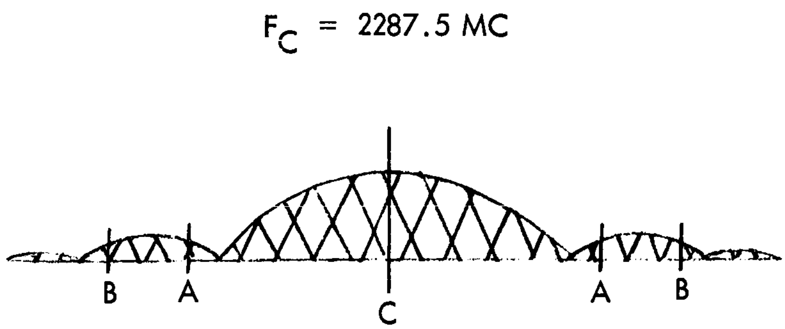
Mode	Channel	Modulation
A-1	VHF Voice	AM
B-1	C-Band Radar	Pulse Radar
C-1	S-Band: Telemetry* Telemetry** Ranging Carrier	PCM/PM/PM PCM/PM/PM PM ---
D-1	S-Band: Telemetry Voice Ranging Carrier	PCM/PM/PM FM/PM PM ---
E-1	S-Band: Television Telemetry	FM PCM/PM/FM
F-1	S-Band: Emergency Key	AM/PM

* 51.2 Kilobits realtime data

** 51.2 Kilobits recorded data

A summary of the important ground-station transmitter characteristics is presented in Table 69. The transmitter capabilities exceed the OPE requirements.

Since all spacecraft transmissions are recorded during a pass over a particular ground station, adequate tape recording capability is extremely important. All stations will be equipped with at least two wideband machines, one narrowband and one voice recorder. Important characteristics of the three types of magnetic tape recorders are presented in Table 70. The recorder which best supports the OPE is selected in subsequent paragraphs to this section.



A = 1.024 MC REAL-TIME CSM TELEMETRY (HIGH OR LOW) SUBCARRIER
B = 1.25 MC REAL-TIME CSM VOICE SUBCARRIER
C = RANGING CODE TURN-AROUND SPECTRUM

<u>TELEVISION – FM AT BASEBAND</u>

Fig. 143 Down-Link S-Band Spectrum

TABLE 69

S-BAND TRANSMITTER CHARACTERISTICS

• Power Output	1 to 10 kW for both 85' and 30' stations 2 kW max per channel for dual frequency operation of 30' stations
• Ranging Channel	
Type	PRN Code
Mod.	PM on carrier
Bit rate/clock rate	992.883 kbs
Bandwidth	1.75 Mc
• Voice Channel	
Subcarrier Frequency	30 kc
Mod.	FM on subcarrier
Freq. Response	100 - 3000 cps
• Up-Data Channel	
Subcarrier Freq.	70 kc
Mod.	FM on subcarrier
Data Rate	2 kc bi-phase mod. signal & 1 kc sync tone
Information Rate	200 bps
Sub-Bit Rate	1000 bps
• Back Up Voice on Up-Data Subcarrier	

TABLE 70

MAGNETIC TAPE RECORDERS

Wideband Recorder/Reproducer Model 22 (Mincom)

- Tracks: 14

  Channel Mix: 14 direct record or 7 direct and 7 FM
- Tape Speeds: 120, 60, 30, 15, 7-1/2, 3-3/4 ips
- Total Harmonic Distortion

  FM: 3%

  Direct: 1.0% or less, at 120 ips, third harmonic of 1 kc
- Intermodulation Distortion: 3%
- Frequency Response

Tape Speed	Direct	FM	Center Carrier Frequency
120 ips	400 cps to 1.5 Mc	dc to 500 kc	900 kc
60 ips	400 cps to 750 kc	dc to 250 kc	450 kc
30 ips	400 cps to 375 kc	dc to 125 kc	225 kc
15 ips	400 cps to 185 kc	dc to 62 kc	112.5 kc
7-1/2 ips	400 cps to 93.75 kc	dc to 31 kc	56 kc
3-3/4 ips	400 cps to 46 kc	dc to 15 kc	28 kc

- Output Signal: 1 v RMS into 50 ohm load
- Tape Size: 1-in.
- Rewind Time: > 6 min for 9200-ft reel

Narrow Band Recorder Reproducer Model 1540 (Mincom)

- Tracks: 14
- Tape Speeds: 60, 30, 15, and 7-1/2 ips
- Total Harmonic Distortion: 3% or less
- Frequency Response

Tape Speed	Direct	FM	Center Carrier Frequency
60 ips	300 cps to 250 kc	dc to 20 kc	108 kc
30 ips	200 cps to 125 kc	dc to 10 kc	54 kc
15 ips	150 cps to 63 kc	dc to 5 kc	27 kc
7-1/2	150 cps to 31 kc	dc to 2 kc	13.5 kc

TABLE 70 (Cont.)

- Output Level: 1 volt RMS into 50 ohm load
- Tape Size: 1-in.

Voice Recorder (Ampex)

- Tape Transport: Ampex FR-1100
- Electronics: Ampex GS-80 Recorder/Reproducer
- Tracks: 7 (on 1/2-in. tape)
- Frequency Response and S/N Ratio

Tape Speed	Direct	S/N Ratio
7-1/2 ips	(±4 db) 100 to 15,000 cps	50 db
3-3/4 ips	(±4 db) 100 to 8,000 cps	45 db
1-7/8 ips	(±4 db) 100 to 4,000 cps	40 db

- Microphone Input Impedance: 150 to 500 ohms, balanced or unbalanced
- Line Input: Balanced or unbalanced 600 ohm line
- Line Output: Balanced or unbalanced 600 ohm line
- Output Level: + 4 dbm from normal level tape

The ground-station-computer system formats data for communications equipment, flight controller readout displays, and for transmission of commands to spacecraft systems. For communications use, the system is required to select, convert, and format the received telemetry information into teletype or 2.4 kilobits/second digital data forms that are compatible with the existing communications circuits. Command functions are stored in the computer memory for readout when the spacecraft is in contact with the ground station. Two Univac 624B modified computers are utilized, one for telemetry processing and the other for commands. The machines are identical and one machine can perform both functions on a time-shared basis if the other machine is unavailable. The telemetry data processor can handle three simultaneous PCM inputs and format into 2.4 kbs bit streams for transfer to the MCC. Since the OPE will transmit 51.2 kbps data, slow playback with a transfer delay time of 51.2/2.4 seconds (21 seconds) is necessitated. Peripheral devices in the computing system include:

- 1540 Magnetic Tape Unit
- 1232 Input/Output Console
- Console Computer Interface Adapter (at those stations which have ground console)
- SB-1299 Switchboard
- High Speed Buffer, Translator, and Printer
- 1000 Interface System Adapter
- 2010 Data Transmission Unit
- 1259 TTY Adapter

Table 71 lists the computer equipment at the various MSFN stations. The principal characteristics of the major components of the computer system are presented in Table 72. If OPE station computations are required, the 2010 Data Transmission Unit could be used because its serial data capability (80 kbps) meets/exceeds the OPE requirements (51.2 kbps).

Data modems are located at each ground station and they serve to condition the data suitably for transmission via the communications channels. There are two types of modems, the 205A wireline modem and the HF Radio data modem. Each modem is capable of operating at 600, 1200, or 2400 bits/second with a bit-error rate of 10^5 at specified S/N ratios (12 db for the 205A and 17 db for the HF Radio data modem).

The tracking antennas at the ground stations exhibit high directivity and, in the S-band and VHF case, can track on received signals, be slaved to another antenna, or pointed on the basis of a stored computer program. The UHF command antenna would be slaved to one of the other antennas or programmed by a computer. Pertinent characteristics of each of the antenna types are summarized in Table 73. The quad-helix UHF antenna will be used to support the OPE mission.

A summary of RF and data handling equipments at the various MSFN stations is presented in Tables 74 and 75.

The Apollo television monitor (Code Designation: S-553-P-8) accepts a demodulated signal from the spacecraft receiving system and processes these signals for viewing on a cathode ray tube. The equipment operates with three different data formats. Special test equipment is supplied with the equipment to generate test

TABLE 71 – NETWORK DATA PROCESSING EQUIPMENT LOCATIONS

STATION / Equipment:	CP-642B Modified Computer	1540-A2 Magnetic Tape Unit	1232 I/O Console	1259 TTY Adapter	Peripheral Communications System (PCS): UNIVAC 2010 DTU (Dual)	Peripheral Communications System (PCS): UNIVAC 1000 ISA	Peripheral Communications System (PCS): Console Keyboard Adapter (CAM)	SB-1299 Distribution Switchboard	Mounted in CCIA cabinet, if any: HS Printer Buffer & Computer Interface	Mounted in CCIA cabinet, if any: HS Printer Translator	Motorola TP-4000 HS Printer Unit	CCIA: 1218 Computer	CCIA: Multiplexer	Existing 1218 Computer
Merritt Island and CNV	2	2	2	2	2	1	1	2	1	1	1			1
Grand Bahama Island	2	2	2	2	2	1	1	2	1	1	1			1
Bermuda	2	2	2	2	3	1	1	2	1	1	1			1
Antigua	2	2	2	2	2	1	1	2	1	1	1			1
Grand Canary Island	2	2	2	2	2	1	1	2	1	1	6	2		1
Ascension Island	2	2	2	2	2	1	1	2	1	1	6	2	1	
Madrid, Spain	2	2	2	2	2	1	1	2	1	1	1			
Carnarvon	2	2	2	2	2	1	1	2	1	1	6	2	1	1
Canberra	2	2	2	2	2	1	1	2	1	1	1			
Guam	2	2	2	2	2	1	1	2	1	1	1			
Hawaii	2	2	2	2	2	1	1	2	1	1	1			
Goldstone	2	2	2	2	2	1	1	2	1	1	1			
Guaymas	2	2	2	2	2	1	1	2	1	1	6	2	1	1
Corpus Christi, Texas	2	2	2	2	2	1	1	2	1	1	1			1
MCC-H, Houston	1	1	1	1	1	1	1			2	10			
ETG (GSFC)	2	2	2	2	2	1	1	2	1	1	6	2	1	1
Ship No. 1	3	4	3	3	3	1	1	5	1	1	6	2	1	
Ship No. 2	3	4	3	3	3	1	1	4	1	1	6	2	1	
Ship No. 3	3	4	3	3	3	1	1	4	1	1	6	2	1	

TABLE 72

STATION COMPUTER CHARACTERISTICS

642-B Computer

- General purpose, medium scale, parallel, binary machine
- Random access magnetic core memory
- 2 μsec read-write cycle
- Storage capacity 32,768 words, directly addressable; can be increased to 131,072 words, directly addressable
- 16 input channels, 16 output channels under full buffer control
- 18 unique interrupts

1540 Magnetic Tape Unit

- Tape transport for 1/2-inch tape
- Maximum tape speed – 120 ips
- Recording densities; 200, 556, or 800 frames/inch
- Tapes compatible between transports within the system and with IBM 727, 729II, 729IV, and 729VI Magnetic Tape Units.
- Duplexing capability with either computer

Univac 1232 Input/Output Console

- Tape photo-electric reader can read 5, 6, 7, or 8 level oiled or dry paper and mylar tape at 30 ips or 300 characters/second
- Tape punch can punch 5, 6, 7, or 8-level tape at 11 ips or 110 characters/second
- Keyboard input/output capability is 10 characters/second, with 72 characters/line

2010 Data Transmission Unit

- Interface between computer and communication equipment
- 8 or 10 bit parallel transfer
- Serial data capability of up to 80,000 bits/second (determined by communications MODEM clock rate)

TABLE 73

GROUND STATION ANTENNA CHARACTERISTICS

Freq. Band	Type	Size	Beamwidth (deg)	Gain	Pol.	Ellipticity
S-Band	Parabola	30'	1.05 ± .25	43 db	RCP or LCP	1 db
S-Band	Parabola	85'	0.25 ± .05	51 db	RCP or LCP	1 db
VHF	Quad Helix 9-turn	12' sq. ground plane	20	18 db	RCP or LCP	–
VHF	18 crossed dipoles	12' × 18' ground plane	18 AZ 12 Vert	18 db	RCP or LCP vert or horiz	–
UHF	Quad Helix 11-turn	Individual circular ground screens	20	18 db	LCP	–

TRACKING CHARACTERISTICS:

30' Antenna:	4° sec velocity 5° sec^2 acceleration Keyhole cone - 10° North-South Axis Pointing accuracy -±0.6 min
85' Antenna:	3° sec velocity Other parameters as for 30° dish except keyhole orientation East-West Axis
VHF/UHF Quad Helices:	25° sec slew rate 10° sec max. tracking rate 10° sec^2 acceleration in AZ 5° sec^2 acceleration in EL. Pointing accuracy ± .5°
VHF Crossed Dipoles:	30° sec slew rate 20° sec max. tracking rate 5° sec^2 angular acceleration Pointing accuracy ± .5°

TABLE 74

APOLLO MSFN STATIONS RF CAPABILITIES

Station	S-Band		VHF	UHF	C-Band Radar	S-Band Radar
	Single	Dual	Receivers	Command Trans		
1. Antigua	2R, 1T		3 WB	2	FPQ-16	
2. Hawaii		4R, Dual T	4 WB, 4NB	2	FPS-17	Verlort
3. Guam		3R, Dual T	3 WB	2	—	—
4. Corpus Christi	2R, 1T		4 WB, 4NB	2	—	Verlort
5. Canary Island	2R, 1T		1 WB, 4 NB	2	MPS-26	Verlort
6. Grand Bahama	2R, 1T		2 WB, 1 NB	2	FPS-16, TPQ-18	—
7. Guaymas	2R, 1T		4 WB, 4NB	2	—	Verlort
8. Carnarvon		4R, Dual T	20 WB	2	FPQ-6	Verlort
9. Ascension		4R, Dual T	3 WB	2	FPS-16, TPQ-18	—
10. Cape Kennedy		4R, Dual T	4 WB, 2ND	8	FPQ-6, TPQ-18, FPS-16	—
11. Bermuda	2R, 1T		9 WB, 4NB	2	FPS-16	Verlort
12. Madrid*		3R, 2T	—	—	—	—
13. Canberra*		3R, 2T	—	—	—	—
14. Goldstone*		3R, 2T	—	—	—	—
Tracking Ships		2R, Dual T	12	2	FPS-16	—
Aircraft	2R, 1T		18	—	—	—

WB = Wideband (up to 1.5 mc)
NB = Narrowband (100 Kc max)
* = 85' Antenna USB Stations

TABLE 75

APOLLO MSFN STATIONS DATA HANDLING CAPABILITIES

Station	Tape Recorders	PCM Decom.	Computer	Display Consoles
1. Antigua*	2 mincom 22, 1 VR 3600	5	2-642B Mod.	2 Remote status ind.
2. Hawaii	2 mincom 22, 1 VR 3600	3	2-642B Mod.	Gemini consoles & displays
3. Guam	2 mincom 22, +narrowband	2	2-642B Mod.	Systems monitor console
4. Corpus Christi	2 mincom 22, 1 VR 3600	3	2-642B Mod.	Gemini consoles & displays
5. Canary Island	2 mincom 22, 1 VR 3600	2	2-642B Mod.	Gemini consoles & displays
6. Grand Bahama	2 mincom 22, 1 VR 3600	4	2-642B Mod.	Remote status indicators
7. Guaymas	2 mincom 22, 1 VR 3600	2	2-642B Mod.	Apollo display system
8. Carnarvon	1 mincom 22, 1 VR 3600	2	2-642B Mod.	Apollo display system
9. Ascension	2 mincom 22, 1 VR 3600	2	2-642B Mod.	Apollo display system
10. Cape Kennedy*	2 mincom 22, 2 VR 3600	5	2-642B Mod.	Control center consoles
11. Bermuda	2 mincom 22, 1 VR 3600	3	2-642B Mod.	Operat & Impact pred. displays
12. Madrid	2 mincom 22	2	2-642B Mod.	Systems monitor console
13. Canberra	2 mincom 22	2	2-642B Mod.	Systems monitor console
14. Goldstone	2 mincom 22	2	2-642B Mod.	Systems monitor console
Tracking Ships	2 mincom 22	2	2-642B Mod.	Apollo Display Sys.
Aircraft	3 WB Recorders, 1 Audio			Master Control Console

*Formats 40.8 KB/S for transmission to Kennedy and/or Houston

patterns for each operational mode. Current planning* calls for installation of the Apollo television monitor at the following MSFN locations:

- Cape Kennedy
- Bermuda
- Antigua
- Ascension
- Madrid**
- Carnarvon
- Guaymas
- Canberra**
- Hawaii
- Goldstone**
- Guam
- Corpus Christi
- MCC (Houston)
- GSFC
- 3 Tracking Ships

Stations denoted by an asterisk (**) are equipped with 85-ft diameter antennas and are normally use-limited to deep space missions only. A television scan converter at Cape Kennedy (and Goldstone) will permit viewing on standard television receiving sets. Characteristics of the Apollo television monitor are presented in Table 76. The monitor can receive three different data formats, as illustrated in Figs. 144 and 145. Common characteristics for the three formats are:

- Blanking for the horizontal and vertical retrace electron beam
- Aspect ratio of 4 to 3
- Total number of lines/frame which includes blanking time

Characteristics of the three data formats are summarized in Table 76. An external test pattern generator generates the patterns listed in Table 77 for all three data formats. The test signals (linear within ±0.5 percent) are:

- Composite signal, applied to signal input jack
- Individual sync and display signals

During the OPE spacecraft mission, Format No. 2 and Format No. 3 will be used to view the following television of the primates:

- Mode 1. Five, 5-minute samples per day of 525 vertical lines per frame at 10 frames per second, to yield a primate resolution of 0.10 inch.

- Mode 2. One minute per daylight hour (totalling approximately 14 minutes per day) of 1500 vertical lines per frame at 1 frame per second, to yield a primate resolution of 0.01 inch.

Although the above-mentioned values of resolution, number of lines, and frame rate were selected by the payload designers, some explanation of their derivation is necessary here because of the following operational factors involved:

- Values of resolution were selected fixed values.
- Values of number of lines per frame are characteristics of the selected airborne video camera.
- Values of frame rate are compatible with the Apollo television monitor

* NASA-GSFC, "Manned Space Flight Network, Apollo Ground Systems," MG-401, Feb 1966

TABLE 76

APOLLO TELEVISION MONITOR CHARACTERISTICS

Input signal level:	Linear amplification, within ± 1% of any signal within the range of 0.15 to 2.5 v (p-p).
Polarity:	(+) or (-).
Video bandwidth:	Flat within ± 1 db from 0.05 cps to 600 kc with dc restoration.
Display:	One frame of information at a time, for photographic purposes. When in the "Manual" mode, there is no raster until a pushbutton switch is actuated. Starting with the next vertical sync signal, one frame of information is displayed. The raster is inhibited until the pushbutton switch is actuated a second time.
Vertical sync output:	A 0 to -6 v ± 0.6 v (p-p) pulse signal, 2.5 ± 10% ms in duration occurs each time a vertical sync signal (pattern) is received. Rise time of the pulse is faster than 1 μsec. Flatness of the pulse is within ± 0.6 v.
Cathode-ray tube:	Masked for displaying a 6 × 8-inch picture of the received signal.
High voltage:	Differentiates 8 gray scales of format No. 1 or 2 data when operating in an area where the illumination intensity is 75 foot candles from an overhead light source. The voltage is as low as possible while meeting these requirements. Small shades and filters may be used to optimize the display.
Linearity:	Signals displayed in the masked area of the CRT are linear within ± 1%.
Synchronization:	Horizontal and vertical synchronization with a low signal-to-noise ratio, "free wheel" at the sync rate when the sync signals are not readily identified.
Front panel controls:	POWER OFF/ON FORMAT SELECT CONTRAST BRIGHTNESS
Filter:	A removable filter over the face of the CRT removes the annoying light spectrum from the CRO phosphor.
Camera attachment:	A camera and hood can be attached to the monitor for photographing the information displayed on the CRO. The camera is easily removable from the monitor, and the monitor can be used without the camera.

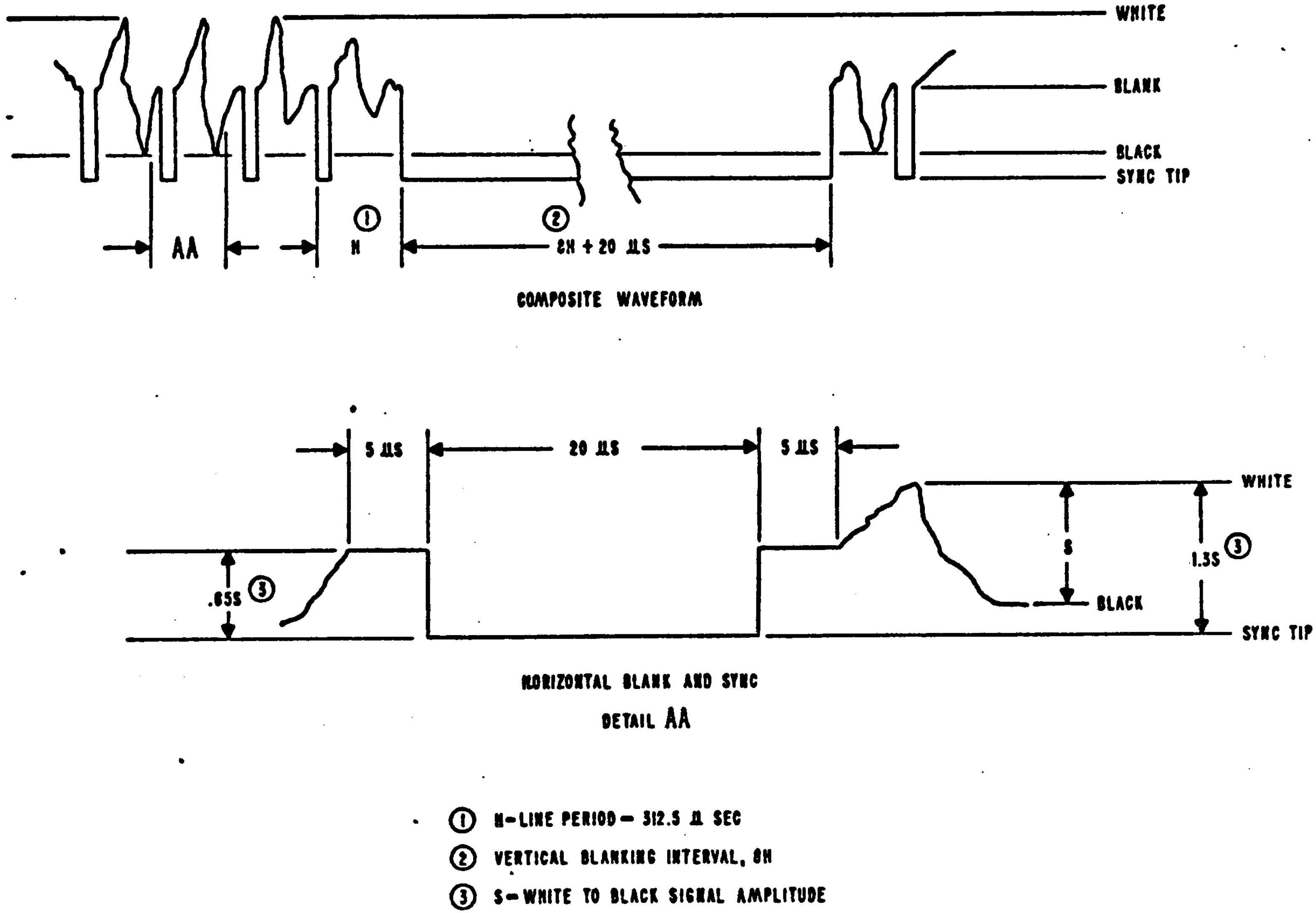
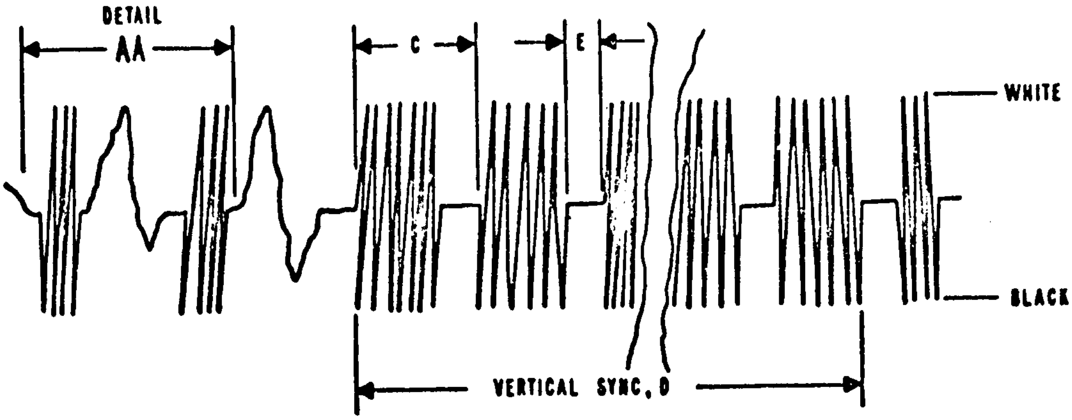
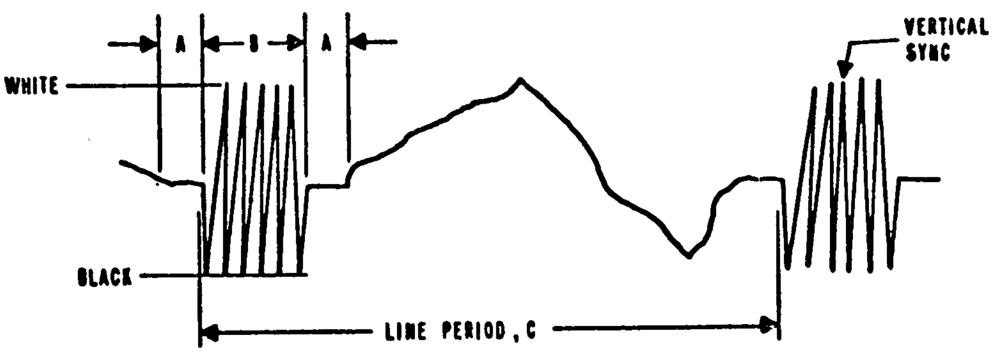


Fig. 144 Apollo Television Sync Format No. 1



COMPOSITE WAVEFORM



DETAIL AA

PRAMETER	FORMAT NO. 2	FORMAT NO. 3
LINE FREQUENCY	320 LPF	1280 LPF
FRAME RATE	10 FPS *	0.625 FPS
SYNC FREQUENCY	409.6 KCS	409.6 KCS
A	5 ± 1 µSEC	20 ± 4 µSEC
B	20 µSEC	80 µSEC
C	312.5 µSEC	1250 µSEC
D	8C = 2.5 MSEC	32C = 40 MSEC
E	30 TO 50 µSEC	160 TO 200 µSEC

Fig. 145 Apollo Television Sync Format No. 2 and No. 3

TABLE 77

TELEVISION DATA FORMAT CHARACTERISTICS

Data Format No. 1:	
Sweep	320 lines per frame, not interlaced
Frames	10 frames per second
Horizontal sync	Medium-white level for 4 to 6 μsec followed by a step to a value of 1.3 times the white-to-black signal level for 14 to 18 μsec, then at a medium-white level again for 4 to 6 μsec
Vertical sync (frame blanking interval)	2.75% (max) of the total frame period (nominally 8 lines)
Waveform	Medium white level for a period of 4 to 6 microseconds at the beginning, stepping to a value of 1.3 times the white-to-black signal level for a period of 2500 ± 50 μsec, and returning to a medium-white level for a period of 4 to 6 μsec
Data Format No. 2:	
Sweep	320 lines per frame, not interlaced
Frame	10 frames per second
Horizontal sync	Burst of 409.6 kc signal, 20 μsec duration, with p-p amplitude of white-to-black signal. The tone burst is preceded and followed by a pedestal of medium-white level for 5 ± 1 μsec
Vertical sync	Burst of 409.6 kc signal for 271.5 to 291.5 μsec duration. This burst is repeated seven times, at the line interval rate (8 lines at 312.5 μsec per line). P-to-p sync amplitude is the same as the white-to-black signal.
Data Format No. 3:	
Sweep	1280 lines per frame, not interlaced
Frames	0.625 frames per second
Horizontal sync	Burst of 409.6 kc signal, 80-μsec duration, with p-p amplitude of white-to-black signal. The tone burst is preceded and followed by a pedestal of medium-white level for 20 ± 4 μsec
Vertical sync	Burst of 409.6 kc signal for 1086- to 1170-μsec duration. This signal is repeated 31 times at the line interval rate (32 lines at 1250 μsec per line or 40 ms). P-to-p sync amplitude is the same as the white-to-black signal.

TABLE 78

EXTERNAL TEST PATTERN GENERATOR CHARACTERISTICS

• Gray scale pattern	Signals for displaying ten different shades of gray on each line
• Horizontal lines	Minimum of ten horizontal lines per frame
• Vertical lines	Minimum of ten individual vertical lines per frame. Also, a series of lines (at least two series per frame) to evaluate the horizontal resolution capabilities of the monitor. The series of lines pattern includes groups of the following: 625, 312, 156, 78 and 39 elements per line. Other combinations may be proposed for or in addition to the above combinations.
• Cross Hatch Pattern	"Horizontal Lines" and "Vertical Lines" patterns described above, simultaneously so as to present a series of squares.

TV data bandwidth and resolution computations are summarized in Table 79. The table indicates a transmission bandwidth requirement of 1 Mc (Mode 1) and 364 kc (Mode 2). Mode 1 and Mode 2 data can both be tape recorded at the remote stations using the Wideband Recorder/Reproducer Model 22 at a center carrier frequency of 900 kc. Mode 1 data must necessarily be recorded at half speed (60 ips) because of bandwidth limitations. All ground recorded video data will be transferred to MCC-H for analysis by the biomedical monitoring team headed by the Principal Investigator (PI). He will require compatible display equipments if not already available.

It is recommended that the ground stations be required to verify that video data had been correctly recorded prior to the transfer of data to the PI. Verification would be achieved by displaying both modes of data on the Apollo television monitor. The three data formats of the monitor are characterized in Table 79. Mode 1 data can be displayed on Format No. 3, but with some degradation because of the bandwidth limitations. In this respect, it is emphasized that "verification" and not "analysis" is required at the remote stations only. Mode 2 data can be displayed without any inherent reductions in image quality on Format No. 2. The following computational points are highlighted:

- Resolution of the monitor equals 6 inches (minimum dimension of CRT display screen) divided by N_v or N_h, whichever is smaller.
- Values of F, N_v, and BW are quoted monitor characteristics (Tables 76 and 77), and values of N_h are computed values.

NASA communications network: The NASA Communications Network (NASCOM)* unifies the following tracking and data acquisition networks to improve communications efficiency and reliability:

- MSFN
- STADAN (Satellite Tracking and Data Acquisition Network)
- DSN (Deep Space Network)

Under present operational philosophy, the total NASCOM communications resources become available as required for the support of any mission. This allows alternate communication channels to be available in the event of a particular channel malfunctioning. The proposed NASCOM circuits for the Apollo program** are shown in Fig. 146. Note on the circular inset that ALDS (Apollo Launch Data System) circuits are used between KSC and MSC and that LIEF (Launch Information Exchange Facility) are used between KSC and MSFC. Details are shown in Fig. 147. Note that duplex television circuits are provided between Kennedy and Houston, but only simplex television is provided from Kennedy to Huntsville.

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* NASA-GSFC, "Manned Space Flight Network — Augmentation Study for the AES," Part I, 1 Sept 1965 pp. 37-43

** NASA-GSFC, "Data System Development Plan, NASCOM," Rev. 2, 6 Aug 1966, Part VIII, 2-1 to 2-17.

TABLE 79

SPACECRAFT/GROUND TELEVISION TRANSMISSION AND DISPLAY

$$BW = \left(\frac{1}{3v \cdot 3h}\right)(F)(.7N_v)\left(\frac{N_h}{2}\right)$$

where BW = Bandwidth, cps
3v = Efficiency of vertical (assumed 90%)
3h = Efficiency of horizontal (assumed 90%)
F = Frames per second
N_v = Number of vertical lines
N_h = Number of horizontal lines

	OPE Data Requirements		Apollo Televsion Monitor Capabilities		
Item	Mode 1	Mode 2	Format 1	Format 2	Format 3
F	1	10	10	10	0.625
N_v	1500	525	320	320	1280
N_h	1600	160	435	435	1090
BW	1.03 Mc	364 kc	600 kc	600 kc	600 kc
	(Primate)		Image on Display Screen)		
Resolution	0.01 In.	0.1 In.	0.20 In.	0.02 In.	0.006 In.

Mode 2 → Format 2: DISPLAY (0.1 In.)

Mode 1 → Format 3: DISPLAY (Some Image Degradation) (0.01 In.)

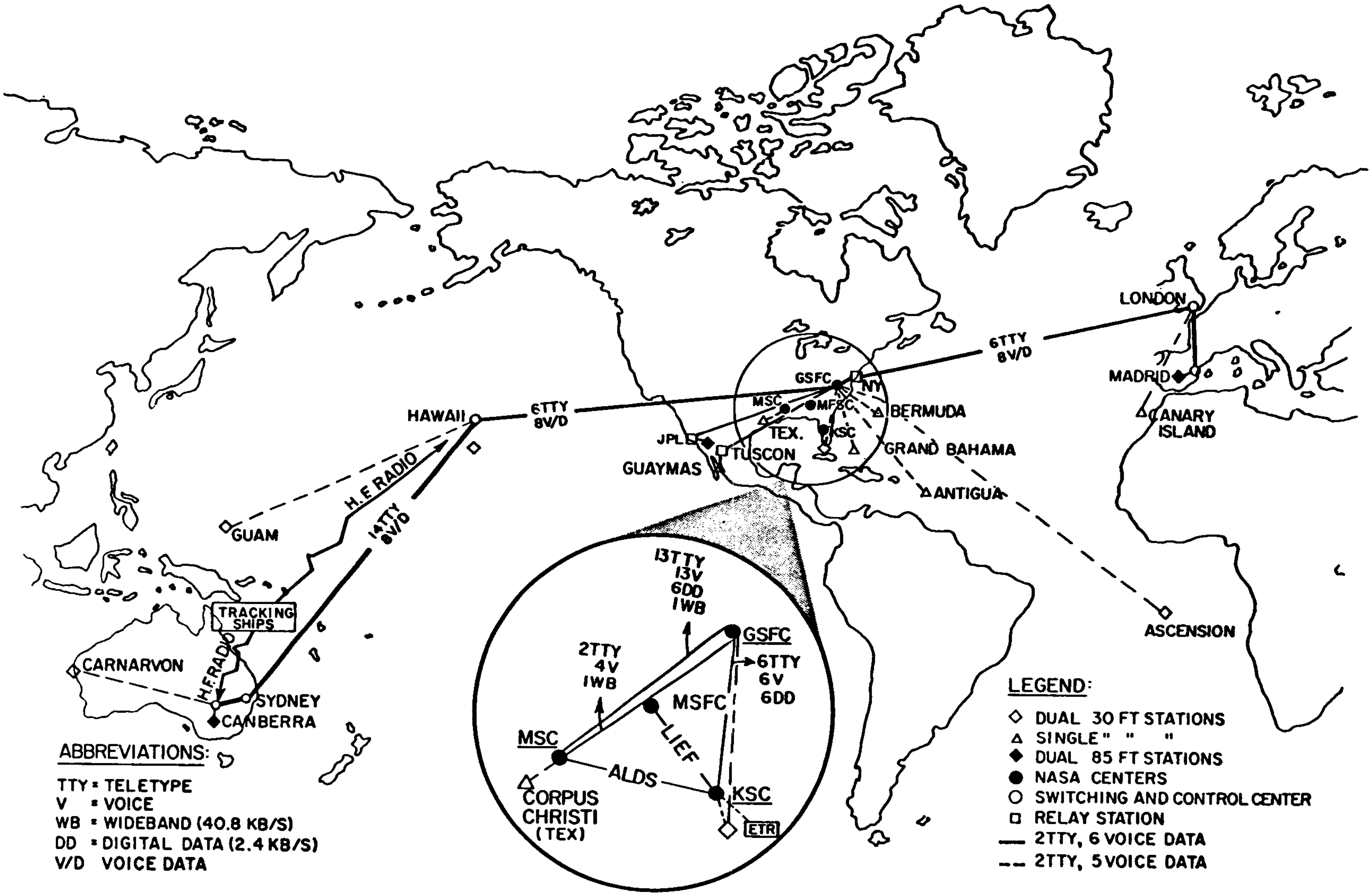

Fig. 146 Proposed Apollo NASCOM Circuits

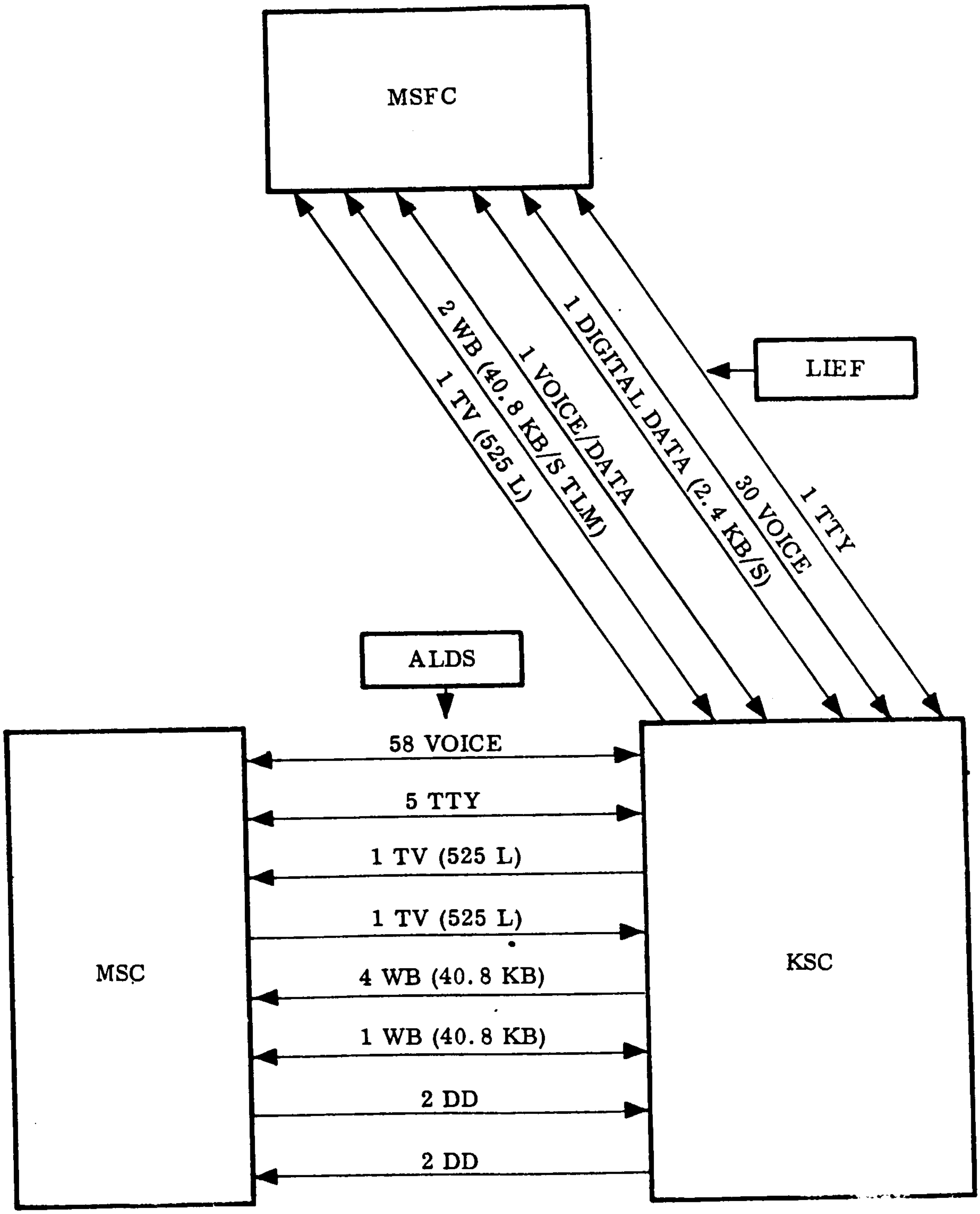

Fig. 147 Apollo Launch Data System and Launch Information Exchange Facility Block Diagram

NASCOM circuits to most of the ground stations are limited to two teletype lines, two voice lines, one biomedical line, and two digital data (up to 2.4 kbps) lines. In general, all circuits are routed to GSFC and from there to the Mission Control Center. Voice, biomedical, or digital data can be routed interchangeably over a standard telephone line which does allow for considerably versatility. Routing television data over a standard telephone line, between two stations not provided with television circuits, would require additional equipments at both stations. The OPE Mode 1 television-data requirements correspond to a 1 Mc bandwidth (computed in Table 79). For a bandwidth of commercial telephone lines equal to 3 kc, the delay in transmitting one frame of television would be about 5.5 minutes.

Current preliminary planning* calls for an additional telephone circuit from MCC to most MSFN sites to meet the anticipated requirements of the AAP and later missions.

Communication satellites: Present NASA plans for utilization of a portion of the capacity of the Intelsat II satellites in synchronous orbit over the Pacific** and Atlantic Oceans will have an impact on NASCOM capabilities. Three remote ground stations will be located near the MSFN facilities at Carnarvon, Ascension, and Canary Islands. Three ship-based stations (the tracking ships Mercury, Redstone, and Vanguard) will be on the Indian Ocean, Atlantic Ocean, and North Central Pacific Oceans. The service will provide six full duplex voice/data channels plus two teletype channels between each of the aforementioned remote stations and GSFC.

Implementation of communication satellite circuits could result in all MSFN stations being remoted to MCC, thus eliminating the need for a flight control or primate-monitoring team at any location except MCC.

It is of interest to note that a tropospheric scatter system is now operational between Carnarvon and Geraldon, Australia. This link can be used for the transmission and reception of four voice/data channels. One of these channels can be used to transmit and receive up to four 100-word/minute teletype channels. This system was installed to serve as a backup in the event service was interrupted on the regular land line circuits.

Orbiting relay stations: NASA is currently studying orbiting relay data stations which, if available, might permit direct relay of television and other signals from the spacecraft to MCC. It is questionable whether such capability will be available by 1970 (proposed year of OPE launch), but progress of the program should be carefully monitored.

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* NASA, "Program Support Requirements, Apollo Applications Program," Preliminary, 21 Dec 1966, pp. 2060 and Item TIGD.

** One of the Intelsat II satellites is presently on station over the Pacific at approximately 176° E. Longitude.

Spacecraft/ground communication links. - For a spacecraft in a 260-n. mi. circular orbit, the maximum slant range for the communication links is about 1,100 ni. mi. The 260-n. mi. altitude is the planned OPE injection altitude. If link calculations based on this distance show adequate margins, then spacecraft/ground communications should be entirely adequate. Downlink and uplink communications are analyzed in succeeding subsections. Rendezvous communications are presented last.

Downlink communication signals: The spacecraft (specifically the Data Management Subsystem) is designed for television transmission (stored or realtime) upon command using the Apollo Mode E-1 S-band FM link, previously described. A summary of S-band FM link calculations is provided in Table 80. The table shows that adequate margins will be achieved. During tracking or uplink commands, television signals cannot be transmitted to the ground.

Phase modulation (PM) is the primary mode for tracking and communication. Switching of the spacecraft transmitter (unified S-band transceiver) from FM to PM is executed by ground command. Realtime animal vocalization or biodata (sampled for 5 minutes six times daily) are modulated on a subcarrier that is 1.25 Mc removed from the center frequency of the carrier. When spacecraft/ground acquisition and lock are established, realtime transmission of one of the signals is initiated by command. The PM mode is also used for tracking the spacecraft. PM data cannot be transmitted during tracking periods.

A pulse code modulation (PCM) subcarrier that is 1.024 Mc removed from the center frequency is used to transmit the following signals:

- Recorded and realtime biodata and engineering data which are sampled for 10 seconds every 12 minutes
- Recorded and realtime psychomotor data

A ground command will initiate the transmission of these signals.

Figure 148 shows a typical contact over a ground station. Station acquisition and lock is shown on the left, and signal fade occurs on the right. Total contact time is 7 minutes (estimated nominal value). The figure indicates that data can be transmitted, commands received, or spacecraft tracking may take place on a single carrier frequency. The PM and PCM transmission times which are shown are for illustrative purposes only. The figure presupposes that in-flight calibrations were commanded and received during an earlier station contact.

Uplink command signals: The spacecraft is provided with a digital data system that receives, detects, decodes, and transfers command signals from the ground to on-board components. A summary command link calculation is provied in Table 81. The table shows that adequate margins will be achieved.

A summary of the command functions required is presented in Table 82. Over 100 stored and realtime command functions are required for normal spacecraft operation. During contingency situations, additional command functions will be required to restore normal or partial operation.

TABLE 80

S-BAND SPACECRAFT/GROUND COMMUNICATIONS LINK (FM)

Transmitter Carrier Power (20 w)	13.0 dbw
Transmitter Circuit Losses(1)	-5.3 db
Transmitter Antenna Gain	-3.0 db
Propagation Loss (2282.5 Mc, 1100 n.mi.)	-165.4 db
Receiver Antenna Gain	44.0 db
Receiver Circuit Loss	-0.5 db
Received Carrier Power	-117.2 dbw
Equivalent Receiver Noise Density	-203.9 dbw
Television (Baseband)	
Receiver IF Bandwidth (10.0 Mc)	70.0 db
Receiver Noise Figure	1.7 db
Equivalent Receiver Noise Power	-132.2 dbw
Predetection S/N specified(1)	10.0 db
Desired Carrier Power	-122.2 dbw
Margin	+5.0 db
Telemetry (1.024 Mc Subcarrier)	
First Detector Threshold C/N(2)	8.0 db
Desired Carrier Power	-124.2 dbw
Margin	+7.0 db
Voice/Biomed (1.25 Mc Subcarrier)	
First Detector Threshold C/N(2)	8.0 db
Desired Carrier Power	-124.2 dbw
Margin	+7.0 db

Notes:

(1) Grumman, "LEM-MSFN S-Band System Signal Performance and Interface Specification," LSP-380-17, 6-7-65

(2) Based on the relative amplitudes of the subcarriers with respect to the main carrier.

ASSUMPTIONS

- APOLLO UNIFIED S BAND NETWORK
- SELECTED DMS SPACECRAFT SUBSYSTEM
- WHEN TRACKING OCCURS, TV DATA CANNOT BE TRANSMITTED

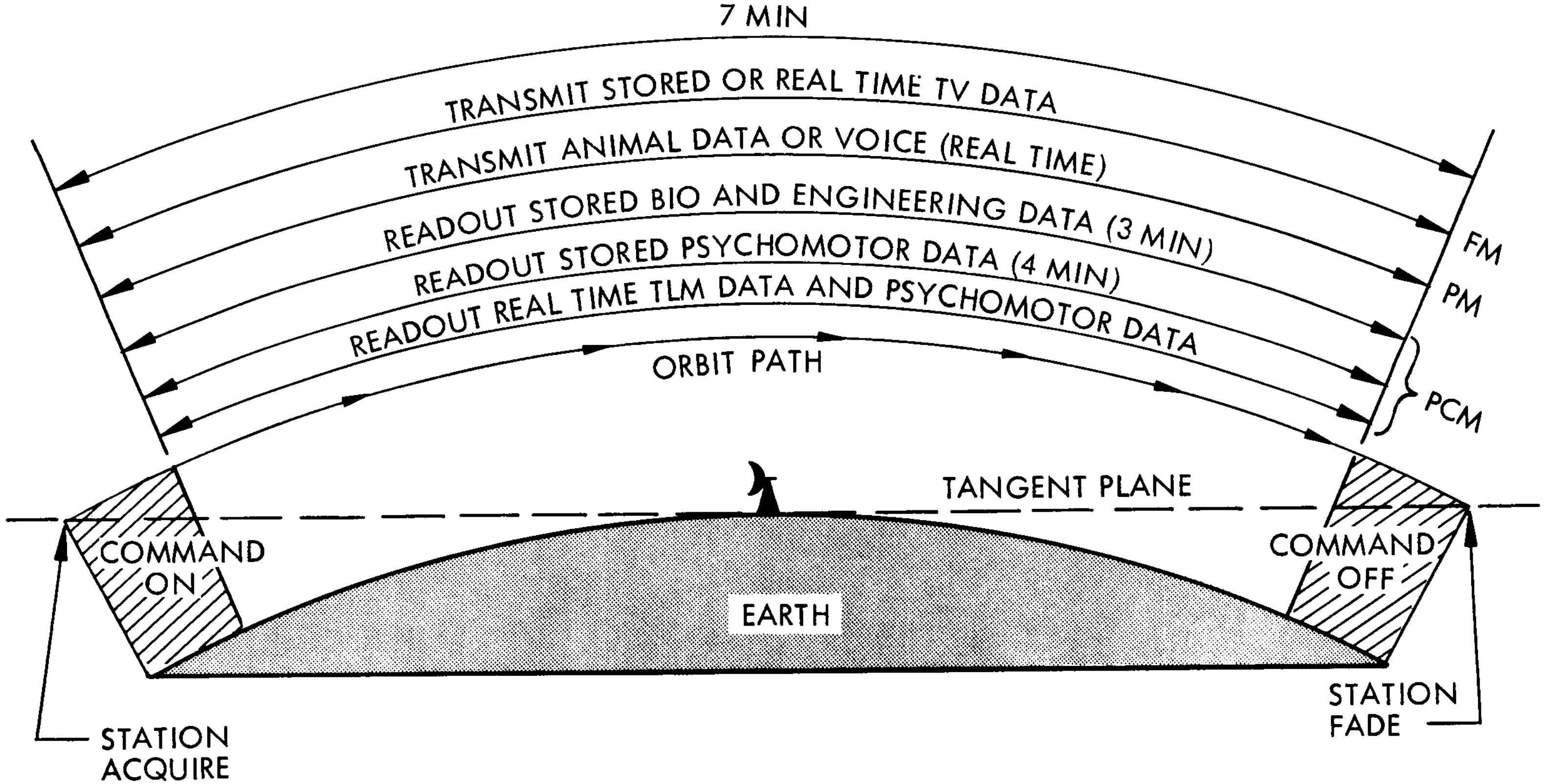


Fig. 148 Typical Station Contact

TABLE 81

GROUND/SPACECRAFT COMMAND LINK

Transmitter Carrier Power (10 kw)	+40.0 dbw
Transmitter Circuit Losses (Est.)	-3.0 db
Transmitter Antenna Gain[1]	+18.0 db
Propagation Loss (450 Mc, 1100 n.mi.)	-151.7 db
Receiver Antenna Gain	-9.0 db
Receiver Circuit Loss (Est.)	-3.0 db
Receiver Carrier Power	-108.7 dbw
Equivalent Receiver Noise Density (T_a = 2000°)	-195.6 dbw
Receiver Bandwidth (2.0 Mc)	63.0 db
Receiver Noise Figure	10.0 db
Equivalent Receiver Noise Power	-122.6 dbw
Required Predetection S/N Ratio	10.0 db
Desired Carrier Power	-112.6 dbw
Margin	+3.9 db

[1]NASA, GSFC, "Manned Space Flight Network, Station Handbook," MG 201, 15 Aug 1965, pp. 2.8-11.

TABLE 82

COMMAND FUNCTION LIST

Command Function and Subsystem/System Involved	Command System			
	No. 1		No. 2	
	Pulse	Relay	Pulse	Relay
Psychomotor:				
• Reset and Start Program	x		x	
• Animal Select		x		x
• Stop Program	x		x	
• Override Program Extension	x		x	
• Set TIM (Time Estimate) Period		x		x
• Set VIG (Vigilance) Click Frequency		x		x
• Set VIG Period Between Lever Actuations	x		x	
• Set ILK (Interlock) Total		x		x
• Select Audio Alarm and AVD Task		x		x
• Override Audio Alarm and AVD Task	x		x	
• Select Exercise Effort Level		x		x
• Select Exercise Actuation Level		x		x
• Enable Food Dispenser	x		x	
• Enable Water Dispenser	x		x	
• Set Food Total		x		x
• Set Water Total		x		x
• Override Lockout Food Dispenser		x		x
• Override Lockout Water Dispenser		x		x
Life Support:				
• Elevator Up	x		x	
• PO_2 Sensor Override		x		x
• PCO_2 Sensor Override		x		x
• Enable Cage Liner Spool	x		x	
• Open Preservative Valve (With Animal Select)	x		x	
• Shutoff Water Supply Tank No. 1	x		x	
• Shutoff Water Supply Tank No. 2	x		x	
• Crossfeed Water Supply	x		x	
• Enable Alternate Water Dispensing (Primate A)		x		x

TABLE 82 (Cont.)

Command Function and Subsystem/System Involved	Command System			
	No. 1		No. 2	
	Pulse	Relay	Pulse	Relay
Life Support (Cont.):				
• Enable Alternate Water Dispensing (Primate B)		x		x
• Open Preservative Vent Valve (With Animal Select)	x		x	
• Actuate Fixed Bleed 2-Gas Control	x		x	
• Actuate N_2 Atmosphere – ACS Isolation Valve		x		x
• Actuate 5 psi O_2 Control	x		x	
• Canister O_2 Supply Disable		x		x
• Operate Coolant Pump No. 1	x		x	
• Operate Coolant Pump No. 2	x			
• Operate Low Flow Fan No. 1: +y, -x Quad.	x			
• Operate Low Flow Fan No. 2: +y, -x Quad.	x			
• Operate Low Flow Fan No. 1: +x, -y Quad.	x			
• Operate Low Flow Fan No. 2: +x, -y Quad.	x			
• Actuate High Flow Fan No. 1: +x, +y Encl.	x			
• Actuate High Flow Fan No. 2: +x, +y Encl.	x			
• Actuate High Flow Fan No. 1: -x, -y Encl.	x			
• Actuate High Flow Fan No. 2: -x, -y Encl.	x			
• Actuate Cell Purge Fan: +x, +y Encl.			x	
• Actuate Cell Purge Fan: -x, -y Encl.			x	
• Actuate Life Cell Vent			x	
• Actuate Low Pressure Nitrogen Valve	x			
• Actuate Low Pressure Oxygen Valve			x	
• Adjust Nitrogen Fixed Bleed 2-Gas Control			x	
Data Management:				
• Readout Digital Recorder	x		x	
• Readout Psychomotor Recorder	x		x	
• Select UHF or S-Band Reception		x		x
• TV Camera Movement	x		x	
• X-Band Transponder On-Off		x		x
• Select TV Camera			x	
• Select Antenna			x	
• TV On-Off		x		x

TABLE 82 (Cont.)

Command Function and Subsystem/System Involved	Command System			
	No. 1		No. 2	
	Pulse	Relay	Pulse	Relay
Data Management (Cont.):				
• Telemetry On-Off		x		x
• Transmitter On-Off		x		x
• Select Direct or Stored Telemetry		x		x
• Stop Main Commutator		x		x
• Calibrate		x		x
• Override Rate Detector		x		x
• TV Recorder Read On-Off		x		x
• Select CO_2 Sensor		x		x
• TV Lights On-Off			x	
Spacecraft Status:				
• Remove Batteries from Bus		x		x
• Remove Nonessential Load from Bus		x		x
• Inverter Source Switching			x	
• Crossfeed O_2 Tank Supply			x	
• Crossfeed N_2 Tank Supply			x	
SUBTOTALS:	30	30	31	30
MARGIN (NUMBER OF COMMANDS):	2	2	1	2

When other-than normal conditions inside the lifecell are indicated from the received data, the acquiring ground station may be required to verify the reception and recording of data before commanding the spacecraft to erase the onboard tape and continue recording the data.

Data reduction. - In addition to computers at the Mission Control Center and at the MSFN stations which are used for operational support, off-line data processing capability is located at MSFC, MSC, and GSFC. Some of the computer capacity at KSC might also be available for off-line data processing tasks. The preliminary OPE mission plans called for major off-line data processing at GSFC, as is the case for the NASA/Ames "Biosatellite" missions. Data processing at MSC, adjacent to the Experiment Support SSR now appears best on the basis of cost and reliability; however, Pensacola would be best on the basis of convenience to the Principal Investigator. A final decision must await the identification of specific NASA resources and the capabilities of existing/planned facilities at Pensacola.

MSC: At Houston, the off-line data reduction center contains a total of three computers and associated peripheral equipment. A CDC 3800 is used as the central processor and performs the detailed data reduction. Two CDC 3200's are used as satellite computers to the CDC 3800 — one for input/output processing and the other for telemetry processing. Peripheral equipment is shared by all three computers. The CDC 3800 has a storage capability of up to 262,000 48-bit words with a maximum transfer rate of 1.25 times 10^6 words per second. The CDC 3200 computers have a storage capacity of 16,384 24-bit words (with modular additions possible) and a transfer rate of 250,000 words per second.

GSFC: Goddard's computer complex is eventually to consist of eight IBM 360 systems of varying capabilities. Five of these machines will be available for off-line functions. Presently, IBM 7094's are used, which have only 1/3 to 1/4 the capability of IBM 360/75 machines.

MSFC: At Huntsville, off-line data processing is eventually to consist of three Univac 1108 computers and associated peripheral equipment. The Univac 1108 is a general purpose computer with a 36-bit word length. Internal storage is normally 65,000 words. The storage cycle is 750 nanoseconds. A complete add operation can be completed in 2 to 4 microseconds. External drum storage capabilities are dependent upon peripheral equipment selected, but can be up to 176 times 10^6 words. Average access time to external storage is 4 to 92 milliseconds depending on the drum capacity. An identical installation will go into operation at Michoud. Full operation at both Huntsville and Michoud is scheduled for 1968.

Data flow. - The OPE mission-generated data are divided into two groupings — experiment data and spacecraft status data. Experiment data will include actual physical measurements from the primary sensors (and biosensors), as well as television and experiment subsystem information. Figure 149 shows the data flow path from acquisition to ground analysis and reporting which has been tentatively selected. Modifications to suit the Principal Investigator may be required.

Spacecraft — ground station: The data originates with the primary sensors aboard the spacecraft. The data consists of experiment and spacecraft-status data, including the measurements which have direct bearing on the primate's health.

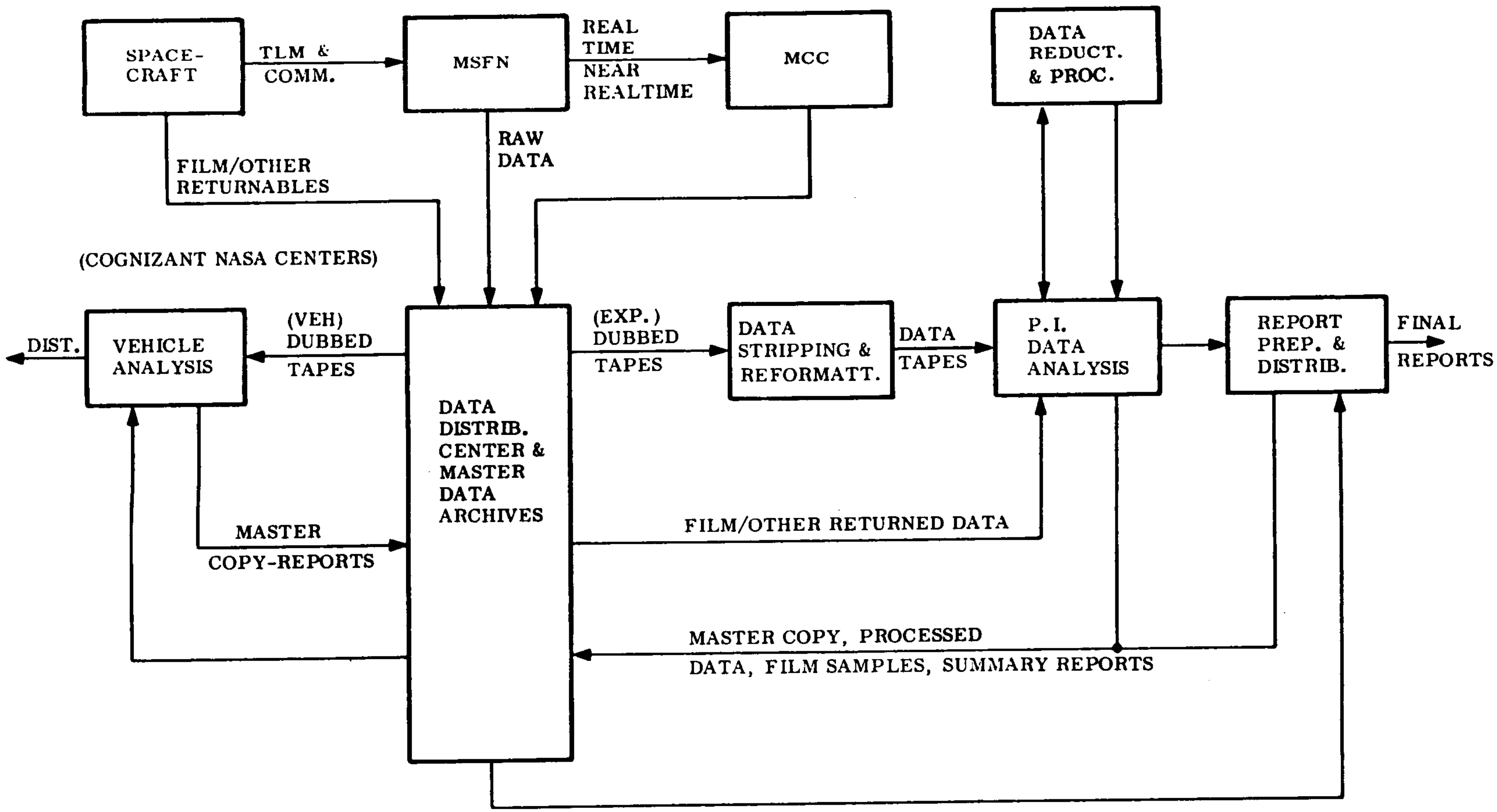


Fig. 149 Data Flow Block Diagram

All data, except the radiation dosimetry film packs, must be transmitted to ground via communication links. The radiation dosimetry film packs will be physically returned to earth with the primate retrieval canisters.

The data that must be transmitted to ground via the communication links are multiplexed and converted to a digital format. Important parameters, such as the primate activity "counts" and psychomotor performance indicators, are stripped from the data stream and stored for transmission during contact with ground stations. The data transmitted during a typical station contact was previously described and is illustrated in Fig. 148. Ground station capabilities for receiving and recording the data were summarized earlier. The scheduling of these transmissions by NASA, such that transmissions during certain earth times to a preselected set of stations will occur only, is anticipated.

NASCOM — MCC: Transmission of data to the MCC will be via the NASCOM network. The NASCOM transmission capabilities were previously summarized. It is recommended that priority on realtime data transfer be given to the so-called "Class A" items of data, shown in Table 83. These items of data have been judged by biomedical and engineering personnel as being the most critical of all data to be acquired. One transmission link would be assigned to this data only.

All raw data recorded on magnetic tape will be shipped within 24 hours after receipt to a so-called OPE (Spacecraft Operations Control Center) for storage, distribution, processing, and analysis. Urgency of analysis requirements will determine the method of shipment. Actual transport by courier might be desirable in some cases, but normal air mail or air express would be the routine method. It is recommended that dubs of all master tapes be made before shipping the masters.

MCC — Operations Control Center: Data received at MCC will include realtime flight data. The flight data are received specifically by the MOCR and analyzed by support personnel in the six SSR's. The processed flight data will be supplied to the OPE personnel located in the Experiment Support SSR. All other data pertinent to the OPE will be similarly transmitted.

The master tapes containing the raw data will be returned to the OPE Operations Control Center in the United States, located most probably at MCC. Tapes will be made up for the Principal Investigator by stripping out Class A data, merging other data (including flight data) needed for proper interpretation, and reformatting into one data package. Data storage requirements would be sharply reduced by storing only compressed data with redundancies removed, and by establishing a reasonable time limit after which the data would be destroyed. Raw data probably loses its significance after analysis has been completed and reports issued. The reports should be retained in the NASA archives to serve as a permanent record. It is recommended that the Principal Investigator have the responsibility for the analysis and reduction of all experimental data.

Quick-look experiment contingency analysis. - Within the capabilities of the MSFN, a specific amount of data (which are a function of station contact time and communication bandwidth) will be transmitted to earth. These data will consist of the measurements listed in Table 84.

TABLE 83

CLASSIFICATION OF DATA

Measurement	Class A	Class B	Class C
Biodata:			
Electrocardiogram	x		
Respiration Waveform		x	
Body Temperature		x	
Animal Activity	x		
Vocalization			x
Mass Measurement*	(x)		x
Psychomotor		x	
Video Mode 1	x		
Video Mode 2	x		
Environmental:			
Temperature (50% RH)	x		
Relative Humidity	x		
Cabin Total Pressure	x		
PO_2 Pressure	x		
PCO_2 Pressure	x		
Fan Pressure (low)	x		
Fan Pressure (high)	x		
Ion. Radiation Dosage			
Life Support:			
O_2 Consumption		x	
O_2 Tank Temperature	x		
O_2 Tank Pressure	x		
Food Consumption			
Water Consumption			
Water Consumption			
Water Flowmeter			
Water Tank Pressure			
Nitrogen Tank Temp.	x		
Nitrogen Tank Pressure	x		
Nitrogen Consumption		x	

Measurement	Class A	Class B	Class C
Spacecraft Systems:			
A. Thermal Control System			
Cabin Air DB (temp)			
Fan Outlet Air (temp) (high)			x
Fan Outlet and temp. (low)			x
Air Entering Cabin (temp)			
Fan Temp (2 fans) (high-low)			x
Atmosphere Storage Tank Temp (4 pls)			
LiOH Bed Air Exit Temp (2 pls)			x
Condensate Heat Exchanger Air Exit (2 pls)		x	
H_2O Tank Temp (2 pls)			
Cond. Ht. Exch. Coolant Inlet Temp (2 pls)	x		
Radiator Coolant Inlet Temp	x		
Radiator Coolant Outlet Temp (downstream of Grnd. Cooling Ht. Exch.)	x		
Solar Array Temp (4 pls) (Thermistor)		x	
Batt. Temp (2 pls) (Thermistor)		x	
Data Management System (6 pls)			
B. Electrical Power System			
Regulator Dissipation Voltage		x	
Array Current		x	
Bus Current	x		
Battery Voltage (2 pls)	x		
Charge Controller Status		x	
Bus Voltage	x		
C. Guidance and Control System			
Pitch Angle	x		
Roll Angle	x		
Yaw Angle Rate			x
Pitch Angle Rate			x
Roll Angle Rate			x
X Acceleration (two levels)			x
Y Acceleration (two levels)			x
Z Acceleration (two levels)			x
Altitude			x
Orbit Angle			x
Fuel Supply Pressure		x	
Fuel Supply Temperature		x	
Fan Speed	x		
Valve Position (on-off) 8 pts	x		
D. Spacecraft Miscellaneous Data Requirements			
S/C Separation			x
Switch Closures (15 pls) Deployment of Mechanical Systems			
Atmosphere Storage Tank Pressure (4 pls)			
H_2O Tank Pressure (2 pls)			
S/C Structure (4 pls) Integrity		x	

Legend:

A = Highly critical
B = Critical
C = Desirable

*Normally Class C. May be transferred in flight to Class A category if required.

TABLE 84

DATA MEASUREMENT SCHEDULE

Measurement	Dynamic Range	Sensor Accuracy (±%)	Times Per Day	Sample Duration (sec)	Number of Places
Biodata:					
Food Consumption	*	5	120	10	2
Water Consumption	*	5	120	10	2
Electrocardiogram	1-500 cps	1	120/6	10/300	2
Respiration Waveform	1-2 cps	1	120/6	10/300	2
Body Temperature	dc-1 cps	± 0.3°C	120	10	2
Animal Activity	0.1-100 cps	3	120	10	2
Vocalization	50 cps-12 Kc	*	*	*	2
Mass Measurement	dc-1 cps	1	4	30	2**
Psychomotor	*	*	120	10	2
Video Mode 1	40 cps-1 Mc	0.1 in.	5	300	2
Video Mode 2	40 cps-.4 Mc	0.01 in.	17	60	2
Radiation Dosimeter		*	4	10	6
Life Support:					
Cabin Temperature (50% RH)	50-100°F	0.5	120	10	2
Relative Humidity	30-85%	2	120	10	1
Cabin Total Pressure	12.7-16.7 psi	2	120	10	1
Oxygen Partial Pressure	137-228 mm Hg	2	120	10	1
Carbon Dioxide Partial Pressure	1-7.6 mm Hg	2	120	10	1
Low-Speed Fan Pressure	0-5 in H_2O	2	120	10	2
High-Speed Fan Pressure	0-1 in H_2O	2	120	10	2
High-Speed Fan Outlet Air Temperature	60-110°F	1	120	10	2
Low-Speed Fan Outlet Air Temperature	40-70°F	0.5	120	10	2

*Variable and/or fixed by ground command.
**20 sensors each place.

TABLE 84 (Cont.)

Measurement	Dynamic Range	Sensor Accuracy (±%)	Times Per Day	Sample Duration (sec)	Number of Places
Life Support (Cont.)					
Cabin Entry-Air Temperature	50-100° F	0.5	120	10	2
Sorbent Bed Air Exit Temperature	60-150° F	10	120	10	2
Condensate Heat Exchanger Air Exit Temperature	32-65° F	0.5	120	10	1
Condensate Heat Exchanger Coolant Inlet Temperature	0-150° F	2	120	10	1
Radiator Coolant Inlet Temperature	0-150° F	2	120	10	1
Radiator Coolant Outlet Temperature	-60-150° F	2	120	10	1
Oxygen Consumption (Flow Meter)	0.05-0.5 lb/day	2	120	10	1
Oxygen Tank Temperature	50-100° F	2	120	10	2
Oxygen Tank Pressure	0-6000 psi	2	120	10	2
Nitrogen Tank Temperature	50-100° F	2	120	10	2
Nitrogen Tank Pressure	0-6000 psi	2	120	10	2
Nitrogen Consumption (Flow Meter)	0.05-0.5 lb/day	2	120	10	1
Coolant Flow Meter (per loop)	50-150 lb/hr	2	120	10	2
Pump Outlet Pressure	0-75 psi	2	120	μsec	1
Water Separator Diff. Pressure	0-10 psi	2	120	μsec	1
Low Pressure Nitrogen Sensor	0-50 psi	2	120	μsec	2
Purge Fan Relay Energize	On/Off	—	*	μsec	1
Data Management System:					
Temperature	-50-100°C	2	120	μsec	6
Spacecraft Status:					
Water Tank Temperature	40-100°F	2	120	10	2
Water Consumption	*	5	120	10	2
Water Tank Pressure	0-40 psi	2	120	10	2
Water Flow	Flow/no-flow	—	120	10	2
Skin Temperature	±300°F	2	120	μsec	24

*Variable.

TABLE 84 (Cont.)

Measurement	Dynamic Range	Sensor Accuracy (±%)	Times Per Day	Sample Duration (sec)	Number of Places
Spacecraft Status: (Cont.)					
Regulator Dissipation Voltage	0-40v	2	120	μsec	1
Array Current	0-15a	2	120	μsec	1
Bus Current	0-15a	2	120	μsec	1
Battery Voltage	0-40v	2	120	μsec	2
Charge Controller Status	On/Off	2	120	μsec	1
Bus Voltage	0-40v	2	120	μsec	1
Solar Array Temperature (Thermistor)	-250-250° F	2	120	μsec	4
Battery Temperature (Thermistor)	20-100° F	2	120	μsec	2
Pitch Angle	± 180°	5	120	μsec	1
Roll Angle	± 180°	5	120	μsec	1
Yaw Angle Rate	0-1 rpm	5	120	μsec	1
Pitch Angle Rate	0-1 rpm	5	120	μsec	1
Roll Angle Rate	0-1 rpm	5	120	μsec	1
X-Axis Acceleration*	± 10 g	2	120	μsec	1
Y-Axis Acceleration*	± 10 g	2	120	μsec	1
Z-Axis Acceleration*	± 10 g	2	120	μsec	1
Fuel Supply Pressure	0-6000 psi	2	120	μsec	2
Fuel Supply Temperature	50-100° F	2	120	μsec	2
Fan Speed	14-2200 rpm	2	120	μsec	4
Valve Position	On/Off	–	120	μsec	8
Spacecraft Separation	On/Off	–	1	μsec	1
Switch Closures**	Open/Close	–	1	μsec	15
Structural Integrity	(To Be Determined)			μsec	4

*Launch-to-Injection only.
**For deployment of mechanical systems.

The mode and time of data taking are also shown. On-board calibration equipment will be used to verify satisfactory equipment operation. Of critical importance are the data required to diagnose and correct malfunctions (defined as any out-of-specification performance) which may be detrimental to a primate's well-being. The detection and transmission of this critical data are analyzed in the succeeding subsection. Contingency equipments and commands are identified and discussed.

In-flight calibration and contingency commands: As shown in Fig. 150 , calibration of experiment apparatus takes place at the sensor input. A composite signal will be gated into the signal conditioner and will consist of a set of square waves, ramp signals, and other square waves of half amplitude. When commanded and subsequently received on the ground, the signal will give data frequency response, phase shift, and nonlinear amplification. The signal source is supplied as integrated circuits for high reliability and low-power consumption. The signal generator will be turned on or off with the multiplexer. One multiplexer channel is assigned for continuous use by the composite signal, and its output will be compared against the same signal that was inserted at the signal conditioner.

As indicated in Fig. 151, calibration of the spacecraft apparatus will be afforded by supplying fixed voltages to certain commutated channels of the multiplexer. These reference voltages are transmitted to earth with the data. This calibration method gives a calibration check of the multiplexer and all equipments following it. It is a simpler method than that used for the more critical experiment data.

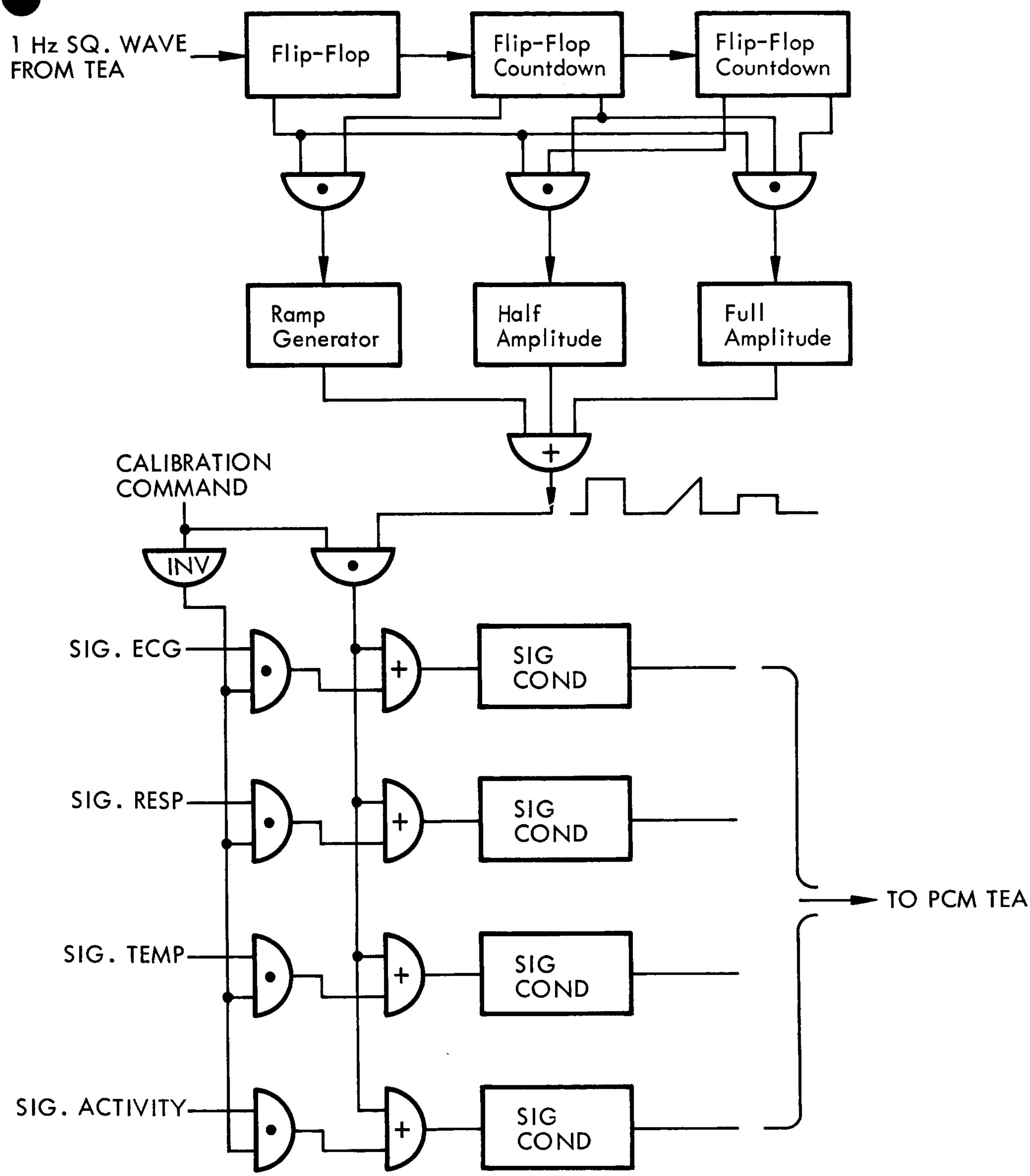
At least on a weekly basis, the on-board atmosphere monitoring instrumentation will be calibrated via ground command by flowing gas of known composition(s) through the instrument from an on-board gas supply tank(s). This will provide the calibration factor(s) for O_2, CO_2, H_2O and N_2 data, as well as to allow adjustment of the O_2 controller set-point via ground command.

As summarized in the above paragraphs, in-flight calibrations of experiment and spacecraft status are automated and will be performed by simple comparison to known source tests. If the comparison indicates either normal equipment operation or equipment malfunction, the transmission of data and its recording are uninterrupted, in the general case. All data scheduled for transmission will be transmitted whether or not the calibrations prerequisite for data interpretation are available. Reactions to equipment malfunctions are to the extent possible, automated. Alternate times and modes of data taking are also automated.

If the comparison (data to known source) indicates that uncorrectable malfunctions of major consequence to the primate's well-being are occurring, the acquiring ground station is required to verify reception of data and then to command the spacecraft to take one of the following actions:

- Erase the onboard tape and continue recording stored or realtime data
- Redump at next available ground station

The first of these two actions will be taken only if the acquisition of correctly recorded data, bit-for-bit, has first been verified. These requirements are intended to assure that diagnostic data required to determine the cause of malfunction will be made available.



TYPICAL FOR ALL EXPERIMENTAL SIGNALS

Fig. 150 In-Flight Calibration Equipment for Experiment Data

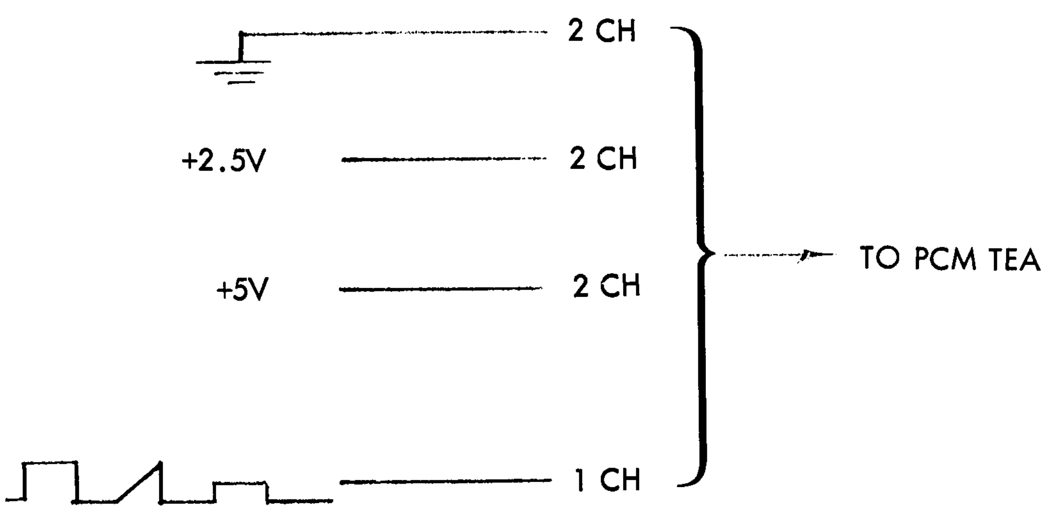


Fig. 151 In-Flight Calibration Equipment for Spacecraft – Status Data

The onboard equipments which are provided to detect major biomedical malfunctions are described in the succeeding paragraphs. Contingency commands required to assure the acquisition of diagnostic data are also described.

Rate detector: If a primate's heart rate, respiration rate, or body temperature exceeds certain upper or lower limits, a command is automatically generated to turn on the tape recorder and record certain bio and environmental data at an increased sampling rate. This automated command would override the time and mode of normal data taking entered into the onboard programmer. Equipment to override this automated command via a ground command are also provided. The rate detector (Fig. 152) senses the three data items as analog signals. The limits of these data, beyond which the rate detector automatically issues an override command, will be established in conjunction with the Principal Investigator.

ECG and respiration signals are changed to rate signals by amplifying the biosignal, then connecting the amplified signal to a Schmitt trigger. It triggers a one-shot of fixed width and amplitude that is then integrated into an analog signal. Minimum and maximum limits are detected which are used to control the command to the tape recorder and required equipment for recording. The temperature signal (primary) is already an analog signal and so further conditioning is not required.

Heart rate, respiration rate, and body temperature can be interrelated and are reflective of conditions such as activity change, illness, and environmental factors, e.g., CO_2, O_2, temperature, and humidity. Consequently, ECG, body temperature, respiration rate, activity, video, and environmental parameters, i.e., temperature, humidity CO_2, O_2 and N_2 will all be sampled at an increased rate and recorded under the contingency situation. Based on the diagnosis resulting from examination of this data, a variety of corrective actions can be taken as indicated in the command function list.

Rendezvous Operations

Rendezvous operations for time phasing and requirements, extravehicular astronaut analysis, and deorbit and reentry are discussed in the following paragraphs.

Time phasing and requirements. - The out-of-plane relationship between the final OPE orbit and the cluster orbit (defined as the angular separation between their orbit planes) was approximated in the Preliminary Mission Analysis Section. It was found that the angular separation could be high after 12 months of orbital flight. Subsequent sections discussed the CSM propulsive energy required to reduce both the plane separation and the altitude differential between the two vehicles to zero in order to execute the rendezvous sequence. It was found that the total propulsion requirement could exceed the CSM's total propulsion capability unless corrective measures were taken. Figure 39 was presented as one possible solution. As implied by this figure, acceptable requirements of the CSM can be achieved by reasonable adjustments in the OPE weight and/or drag.

It is clear that other, perhaps more desirable, methods will result from Ephemeris computer computations which additionally consider the effects of variations in the

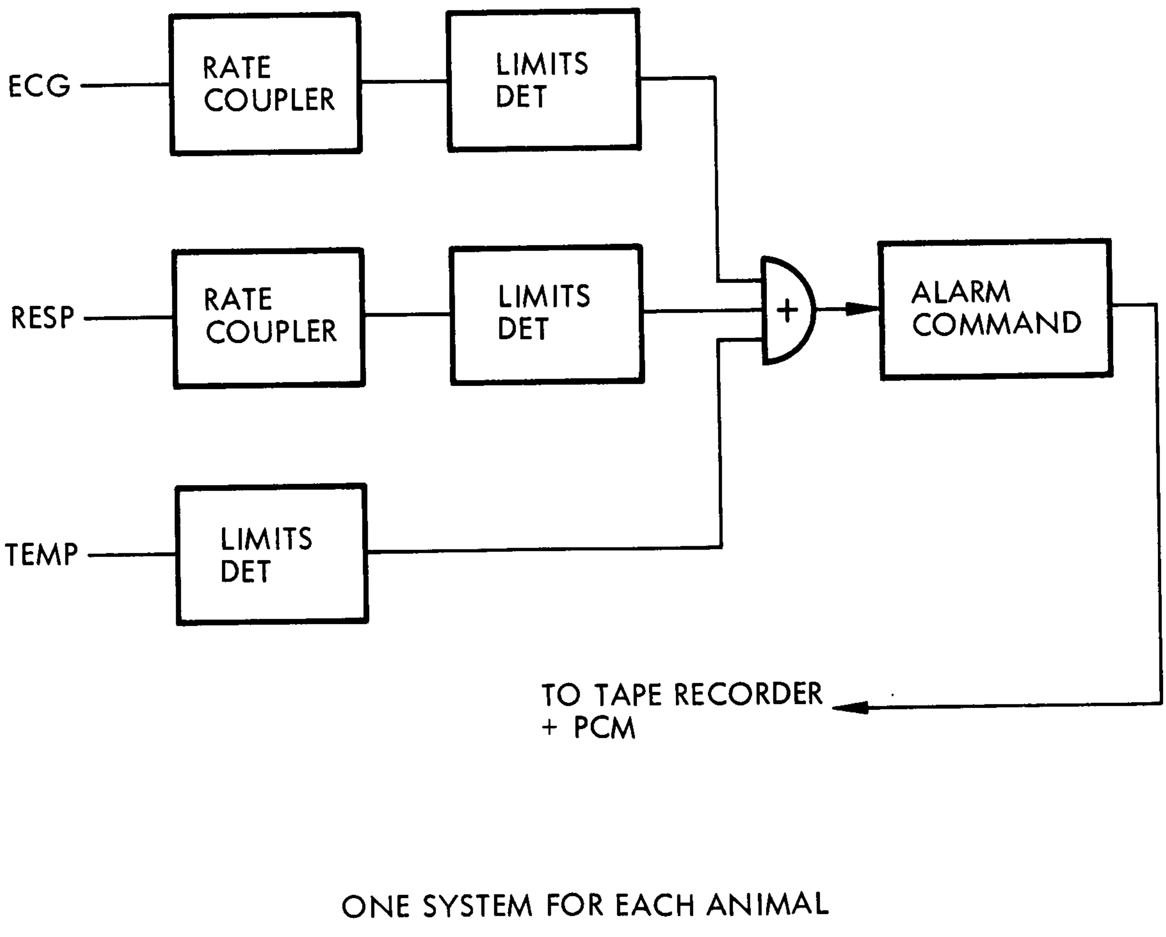


Fig. 152 Rate Detector Block Diagram

launch conditions (launch time, orbit inclination angle), the orbit conditions (altitude, eccentricity), and the total in-orbit duration of the OPE. Such computations are therefore required. This work must be left to future program phases when the ranges in these variables which are acceptable to both the AAP and the Langley Research Center can be established. Of course, the degree of flexibility which the AAP might allow is dependent upon the specific mission requirements of the AAP vehicle and the requirements of other experiments aboard.

For the proposed Cluster A series of AAP flights, the apparent margin between the propulsion requirements, and the propulsion capabilities varies from +3 percent to -93 percent, depending upon the specific flight and propulsion system being considered. Large-scale modifications to the Service Module to meet the requirements of the various AAP experiments are therefore expected. The capabilities as well as the requirements are therefore not specifically defined at this time.

When rendezvous is accomplished, the final CSM manuevers may take the form of station keeping with the OPE spacecraft or docking to it. In one case, the astronaut will perform the retrieving function by a tethered EVA excursion from the CM to the OPE spacecraft. In the other, he will use the EVA hatch in the side of the CM and translate along the mated CM/OPE spacecraft to the retrieval work station. The spacecraft is designed to permit either method of operation, which of the two to be employed will depend on the techniques and equipment to be used for similar AAP operations which prevail at the time of retrieval.

<u>Extravehicular astronaut (EVA) analysis.</u> - There are four steps in the analysis of the astronaut activities that must be performed:

- Identification of the key tasks
- Examination of the relationship between the key tasks and normal vehicle operations (preliminary compatibility analysis)
- Establishment of the preliminary time-lines
- Determination of astronaut ability to accomplish tasks and the effects on crew requirements (final compatibility analysis)

Identification of key tasks: The overall task is to retrieve the two Primate Retrieval Canisters (PRC) from the orbiting spacecraft and to stow the PRC's aboard the Command Module for safe return to earth. A work station for the astronaut is designed into the spacecraft, for his use in stabilizing himself against the spacecraft while performing the retrieval steps, as shown in Fig. 153 . These steps or subtasks are:

- Grasp handrail for body positioning maneuver
- Place feet in foot restraints
- Fasten tethers to attach points
- Adjust tethers to desired work distance

The handrails are located on each side of the access hatch as shown in Fig. 130. They are constructed of tubular aluminum and are approximately 18 inches long by 1-3/4 inches in diameter. The tether attach points are 22-1/2 inches from the top of the spacecraft, easily within the astronaut's reach on each side of the access hatch.

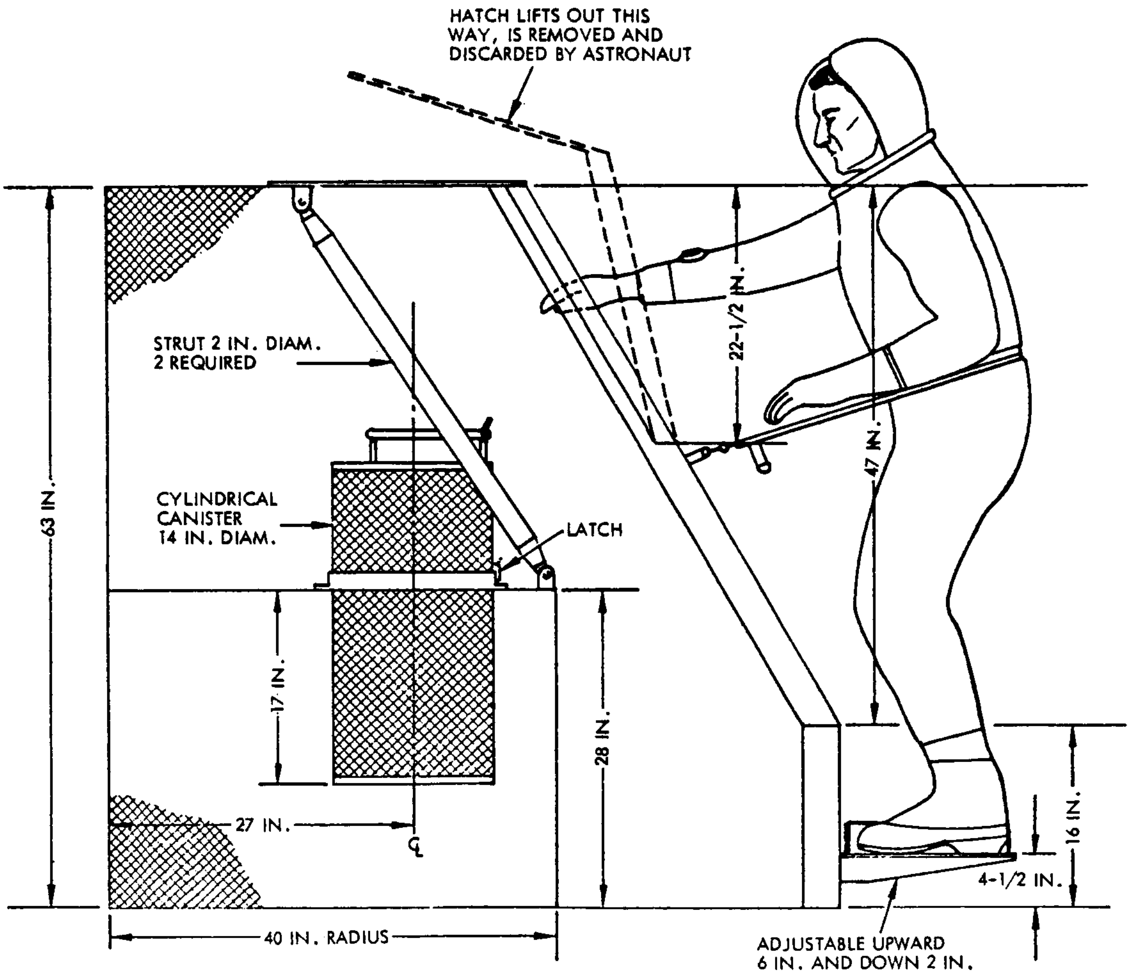


Fig. 153 Astronaut at Work Station During EVA

The foot restraints (Gemini-type assumed) are firmly attached to the side, 16 inches below the side angle. With the astronaut situated against the spacecraft in this manner, he is capable of performing the actual PRC retrieval sequence. The subtasks he must now perform are:

- Grasp handle to activate cowling-release mechanism
- Discard cowling away from work station
- Activate vacuum release valve
- Disconnect life support and hydraulic lines
- Tether PRC with harness safety line
- Activate PRC release latch
- Pull (disengage) PRC elevator detent
- Extract PRC from placement well (payload)
- Insert and secure PRC to translation eyelet
- Remove harness safety line
- Remove body tethers and feet from restraints
- Grasp translation rail and translate to opposite work station, sliding PRC along guide rail eyelets

At this point the astronaut is situated against the spacecraft adjacent to the second PRC. The subtasks delineated above which lead to the securing of the second PRC are then repeated. The astronaut is now ready to complete the following subtasks:

- Activate PRC mating device to form one unit
- Remove PRCs from translation eyelet
- Remove harness safety line
- Remove body tethers and feet from restraints
- Translate back to Command Module, ingress, stowage and tie-down PRCs.

The ingress, stowage, and tie-down sequences are described in the next subsection.

Preliminary compatibility analysis: The equipment incorporated into the work stations and the retrieval techniques used are compatible with the normal EVA operations. Compatible waist and equipment tethers are provided. The latching and securing mechanisms are designed for ease of activation, accessibility, and visibility. The PRC rolling eyelet, used in translating between work stations, allows the concurrent use of both hands for greater mobility and stability. This translational aid is comprised of a ball bearing eyelet and a protruding PRC attach ring. The handrails and foot restraints are rigid, thereby giving control to the upper and lower torso of the astronaut.

Preliminary time lines: The estimated times necessary to perform individual EVA activities are totalled in Table 85. The total time required is approximately 10 minutes.

Final compatibility analysis: A final compatibility analysis cannot be provided at this time because of astronaut-oriented unknowns. Unknown factors include:

- Astronaut self-propulsion techniques and devices
- Position/attitude of the spacecraft relative to the Command Module

Table 85

ASTRONAUT EVA TIME-LINES

No.	Crew Task/Event	Astronaut Activity (%)	Equip-ment Interface	Visual Feedback	Notes	Est. Time	Cum. Time
	Begin retrieval task procedure with subject grasping handrail of OPE Capsule						
1.	Grasp handrail for body positioning manuever	A-100	Handrail				
2.	Place feet in Gemini-type foot restraints	A-100	Dutch shoes				
3.	Fasten tethers to attack points	A-100	GT-12 tethers			1.05	1.05
4.	Grasp handle activating cowling release mechanism	A-100	Cowling release mech.				
5.	Discard cowling away from work station	A-100			The cowling can be discarded away from the vehicle without causing any adverse results		
6.	Activate vacuum release valve	A-100	Relief valve	Pressure		0.63	1.68
7.	Disconnect life support and hyd. line	A-100	Quick disconnect fittings		This manuever may require a two-handed operation; however, it is feasible for it not to be		
8.	Tether cylinder with harness safety line	A-100	Tethers rings				
9.	Activate cylinder release latch	A-100	Safety release latch	Status flag			
10.	Disengage retrieval elevator detent by pulling	A-100				1.40	3.08
11.	Remove cylinder from placement well	A-100				1.40	3.08

Table 85 (Cont.)

No	Crew Task/Event	Astronaut Activity (%)	Equip-ment Interface	Visual Feedback	Notes	Est Time	Cum Time
12.	Insert and secure to translation	A-100	Safety ring attachment				
13.	Remove harness safety line	A-100	Tether				
14.	Removes body tethers and feet from restraints	A-100	Tether and foot restraints				
15.	Grasp translation rail and translate to opposite work station, sliding cylinder along guide rod eyelet	A-100	Trans. rails and sliding eyelet		The rolling eyelet concept may involve problems of control, if translation involves a two-handed effort.	2.60	5.68
16.	Repeat steps 1 to 11 for next task	A-100				3.08	8.76
17.	Astronaut activates cylinder mating device forming one unit	A-100	Auto. long. male and female				
18.	Removes cylinder from translation eyelets	A-100	Tether				
19.	Repeat steps 13 and 14. End of Task	A-100				1.67	10.33
20	Translate back to CSM and increase into the spacecraft	A-100	Trans. rails or manuevery devise				

TOTAL EVA TASK TIME	10.33	Mins.
	619.80	Secs.

- Distance from spacecraft to Command Module
- Astronaut suit configuration
- Total EVA time allowed

It is expected that these data will be resolved as AAP planning, techniques, and EVA equipment become more definitized.

Deorbit and reentry. - The Command Module has two access openings or hatches which can be used for entry of the OPE primate retrieval canisters. These are the Tunnel Access Hatch and the Crew Access Hatch, illustrated in Fig. 154. The existence of adequate clearances is shown in Fig. 155. This figure also indicates the recommended stowage position. Stowage in the aft section of the CM, under the footrest of the astronaut's couch, is the recommended position for each canister. The manipulations required to complete the stowage and tie-down sequence is indicated. This recommended stowage position was based upon the presumed requirement for minimum CM modification. Four CO_2 absorbers, normally located under the astronaut's couch, must be relocated.

The CO_2 absorbers to be displaced weigh approximately 154 lb and occupy a volume of 6,750 in.^{3}. Studies to yield other stowage positions suggested that considerably greater CM modifications would be required.

The stowage position selected will provide adequate clearances between the astronaut couch and the retrieval canister when the couch is reclined to the maximum position during reentry, as shown in Fig. 156. An adjustable ballast mechanism will be required to return the CM center of gravity to the allowed range.

The basic CM return-weight capability is approximately 188 lb, but the possible capability is approximately 533 lb. The potential for off-loading expended packages accounts for the divergency between these two numbers. Studies to increase either number are underway by the AAP since increased experiment return requirements have been anticipated. Whether or not off-loading will be required to return the 175-lb OPE retrieval canisters cannot be stated with certainty at this time.

Figure 157 and Table 86 are presented for information only. The figure shows the data packages inside the CM, some of which are expended and need not be returned to earth. The table presents a selected list of packages which appear to be "least required" for earth return. The table indicates that a 470-lb increase in the CM basic return capability is possible. This increase would yield a possible return capability of 658 lb (188 + 470 lb). An OPE canisters growth weight of 250 lb would leave 250 lb for other experiments plus a weight margin of about 24 percent.

Only one astronaut task is required for each canister after it is stowed and tied down in its reentry position. For each canister, and immediately prior to reentry, an astronaut is required to depress the "electronic timer" button located atop the canister dome. This will initiate the continuous tape recording of signals from biosensors implanted in the primate. The task involves a simple motor manipulation while the astronaut is reclined in his couch prior to reentry. Once this task is completed, the canister is completely self-sufficient requiring no CM or astronaut support.

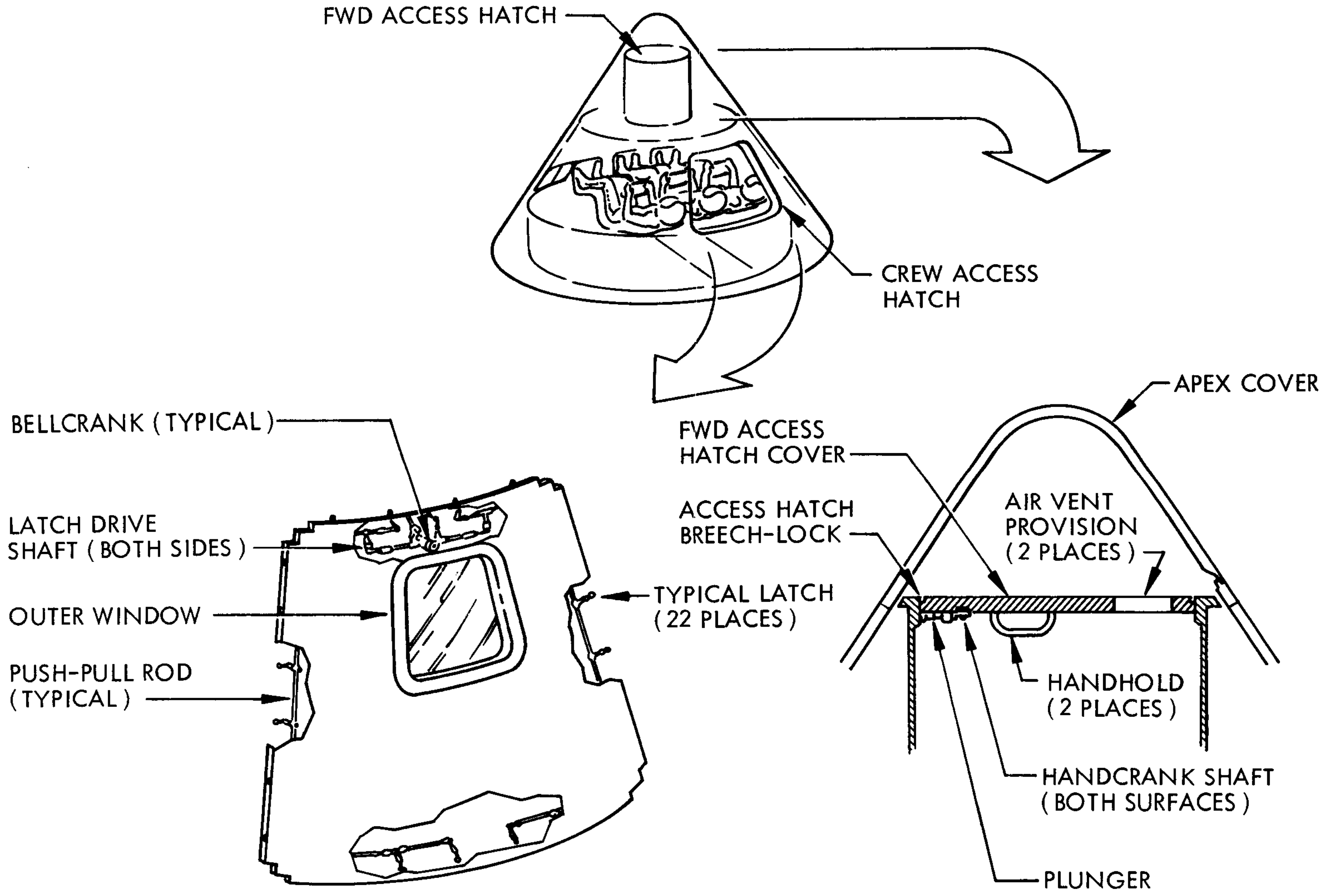


Fig. 154 CM Access

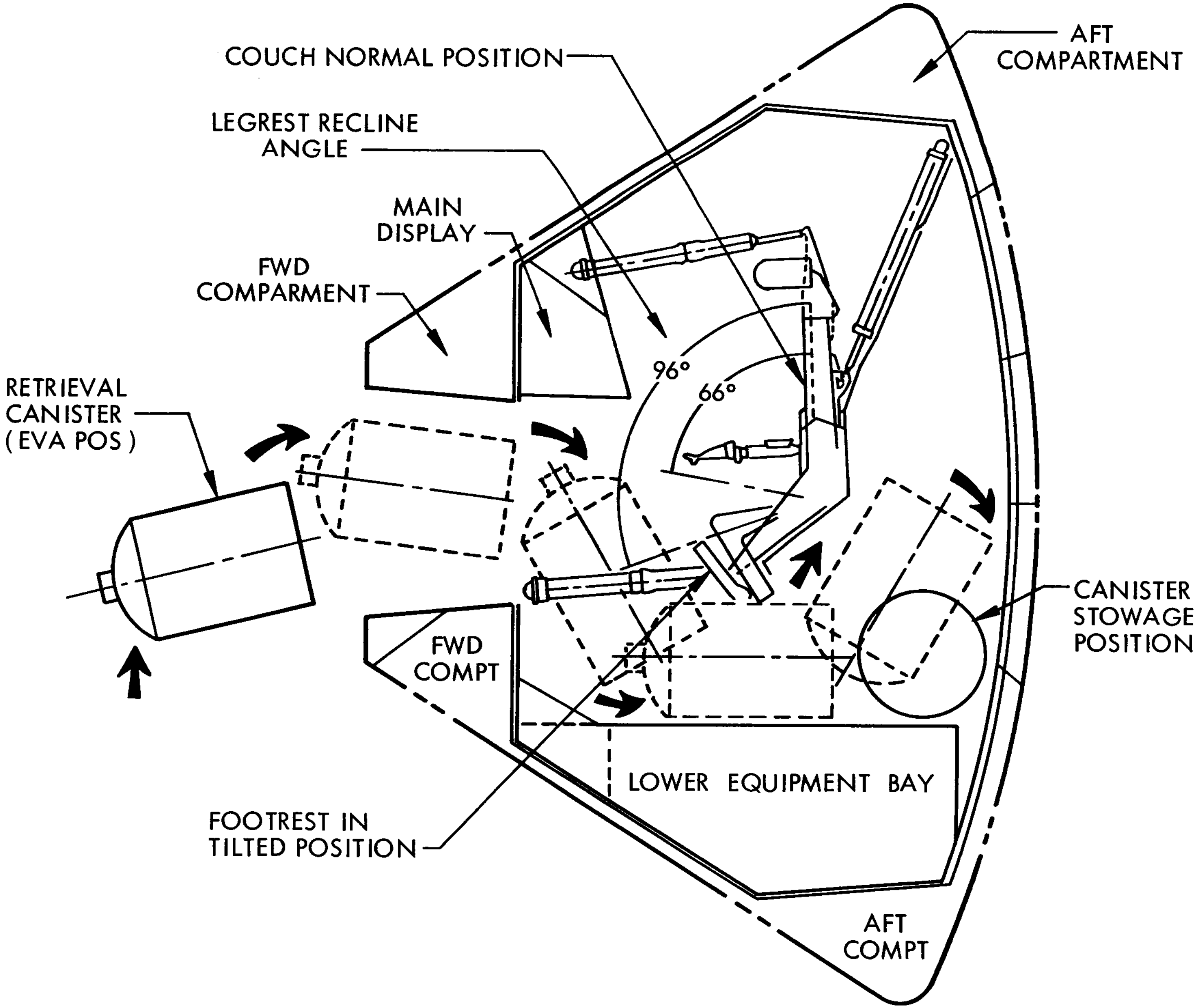


Fig. 155 Canister Stowage Inside CM

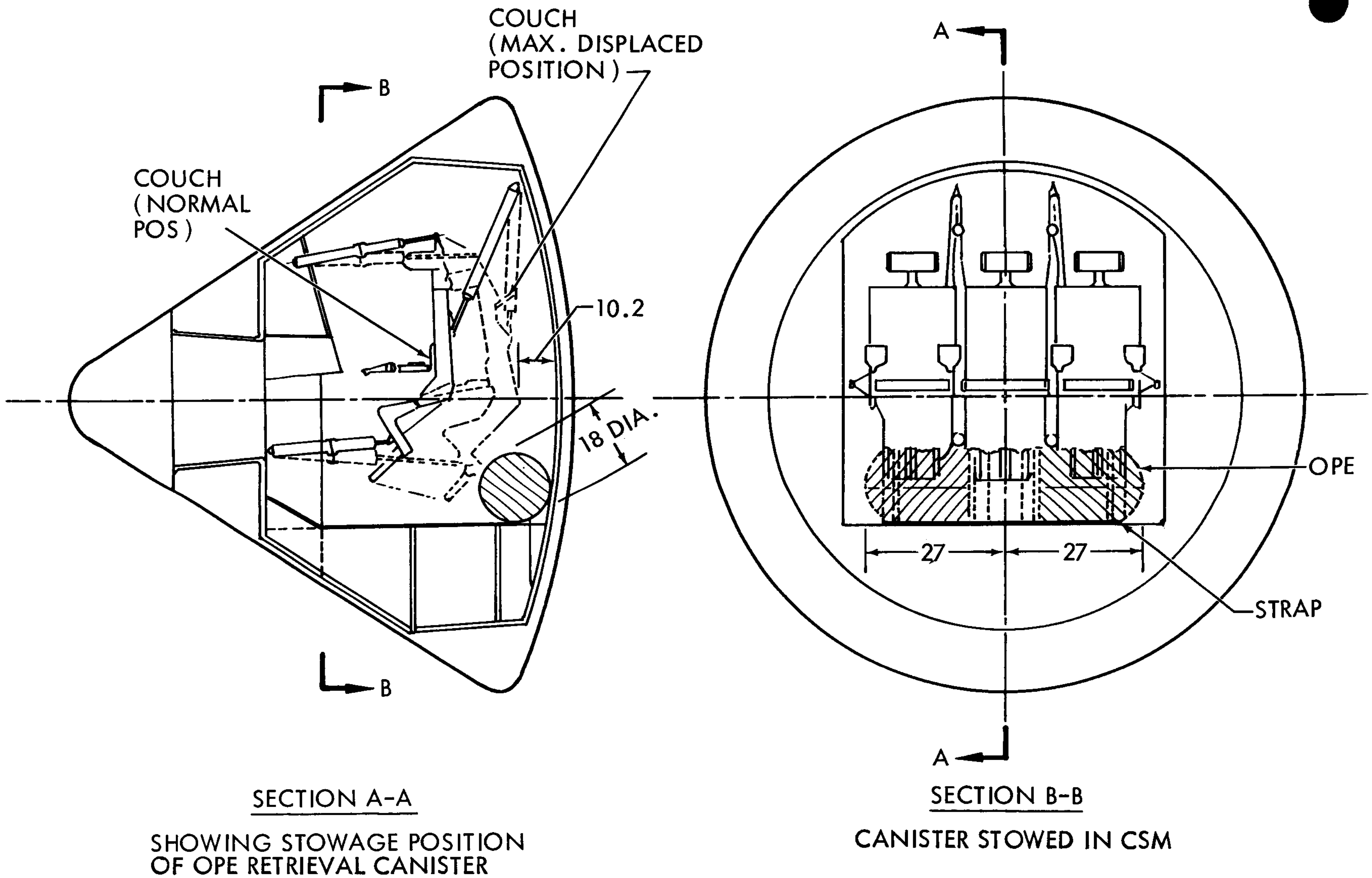
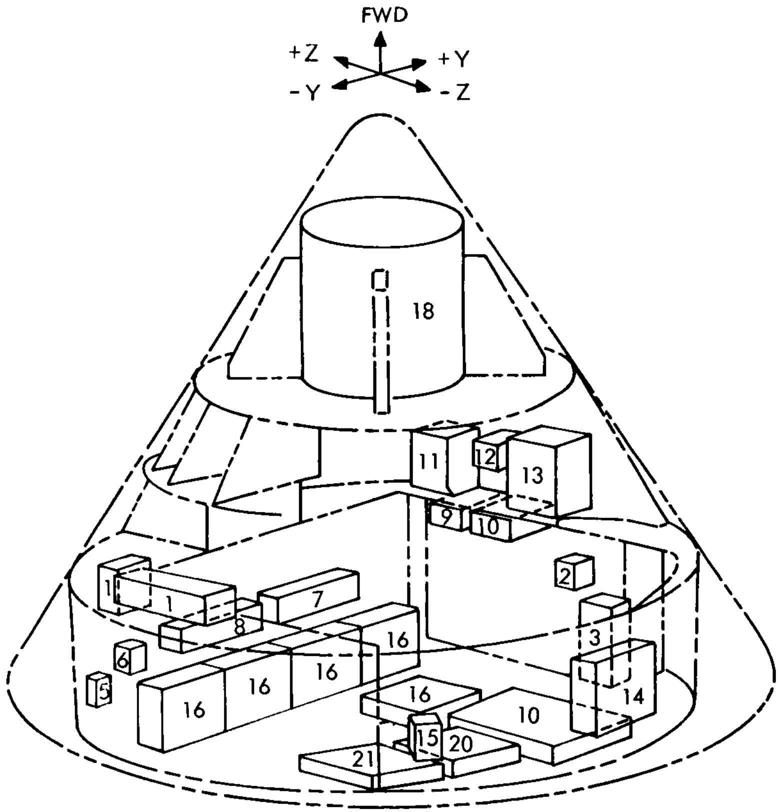


Fig. 156 CM/Retrieval – Canister Interface



		Wt (lb)	Vol in.^{3}	Pkg
1	Food Container, S065, S006	35	2940	2
2	LDEC, S009	24	1000	1
3	Extra Food	20	1625	1
4	Food Container M488, M498, T020, T021	27	2220	4
5	Sequence Camera, S070	10	275	1
6	Still Camera, D019, M479	15	725	2
7	Rock Box A, D018, M609, M487, M489, T017	52	1640	5
8	Rock Box B	52	1640	1
9	Vacuum Cleaner and Waste Bag, M052	3.5	350	1
10	Waste Container, D021, D022	9.0	1625	4
11	NASA Bio Instruments	5.0	210	1
12	PGA Cables, S006	4	270	1
13	Sanitary Supplies, M053, S063, S069	18.5	1125	3
14	CWG, LCG, Tools M018, M050, M051, M492	12	3160	3
15	TV Camera	13.4	650	1
16	CO_2 Absorbers, M469, S018, T002	154	6750	5
17	Data Storage	4	140	–
18	Tunnel	150	10500	1
19	Thermo Meteoroid Storage	420.0	–	
20	Pressure Garment			1
21	Pressure Garment			1

Fig. 157 CM Data Packages

TABLE 86

CANDIDATE CM PACKAGES FOR OFF-LOADING

Item*	Name	Weight (lb)
1	Food Container	35
4	Food Container	27
7	Rock Boxes	52
8	Rock Boxes	52
16	CO_2 Absorbers	154
18	Tunnel	150
		470 lb

*Refer to Fig. 157

Postflight Operations

The primate retrieval canisters will be returned to the Postflight Examination Laboratory designated by the Principal Investigator for post-flight analysis. The allowed transportation time is 44 hours, assuming a total time for in-orbit EVA, de-orbit, reentry, and recovery of four hours. Each canister is equipped with 48 hours of expendables for passive environmental control.

Examination of the primates at the recovery site (shipboard) may be required at the discretion of the Principal Investigator. Breaking of the canisters to extract the primates will only be performed by the Principal Investigator and/or his biomedical monitoring team.

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DEVELOPMENT PLAN

This section describes a program for the development of the Orbiting Primate Experiment (OPE) that will achieve flight-ready hardware by mid-1970. In order to accomplish this goal, full advantage is taken of past and present Bioastronautic activities and the efforts of the next phase are so ordered that a minimum of time is taken in the accomplishment of the necessary tasks.

Figure 158, the OPE Summary Master Schedule illustrates major design, manufacture and test activities spanned over the program duration with the estimated magnitude of each line item indicated in manmonths. Preliminary Engineering, Manufacturing, and Integrated Test Schedules are included in their respective sections of this plan.

As shown on the Summary Master Schedule, during Phase II overlapping or parallel effort in design of the flight article during the fabrication and test of the payload lab test model is dictated to achieve the desired flight date of mid-1970.

The payload laboratory test model detailed design and fabrication will be initiated on the basis of the drawings and specifications completed under the Phase I preliminary design study. This unit will be delivered to the Principal Investigator. Additional development efforts are planned to (1) resolve critical spacecraft design problems and (2) verify the zero-g adequacy of the retrieval and waste management portions of the payload design via zero-g aircraft flight tests on functional mockup type hardware. The engineering mockup is required as a design tool to resolve space allocation and interface problems. The schedule timing is such that results of testing the laboratory model by the Principal Investigator can influence the flight article design.

With regard to the flight hardware, early identification and test of critical elements of the system is of paramount importance. Proof of the ability of critical components to function satisfactorily is sought at the earliest possible date and the preparation of flight hardware proceeds on a calculated risk basis after qualification of critical components/subsystems, but before qualification test of the system has been completed. Delivery of a back-up unit of the flight hardware is made 2 months after acceptance of the flight article. The flight qualification model is refurbished following qualification testing in order to serve as the ground test unit for a parallel ground/flight test operation.

Management Control Plan

Development of a Management Control Plan covering Phase II of the OPE Program which carefully defines the work effort required is provided to achieve the most efficient use of time and manpower. This plan will also provide maximum program visibility to management to allow rapid recognition, decision, and resolution of problems as they arise.

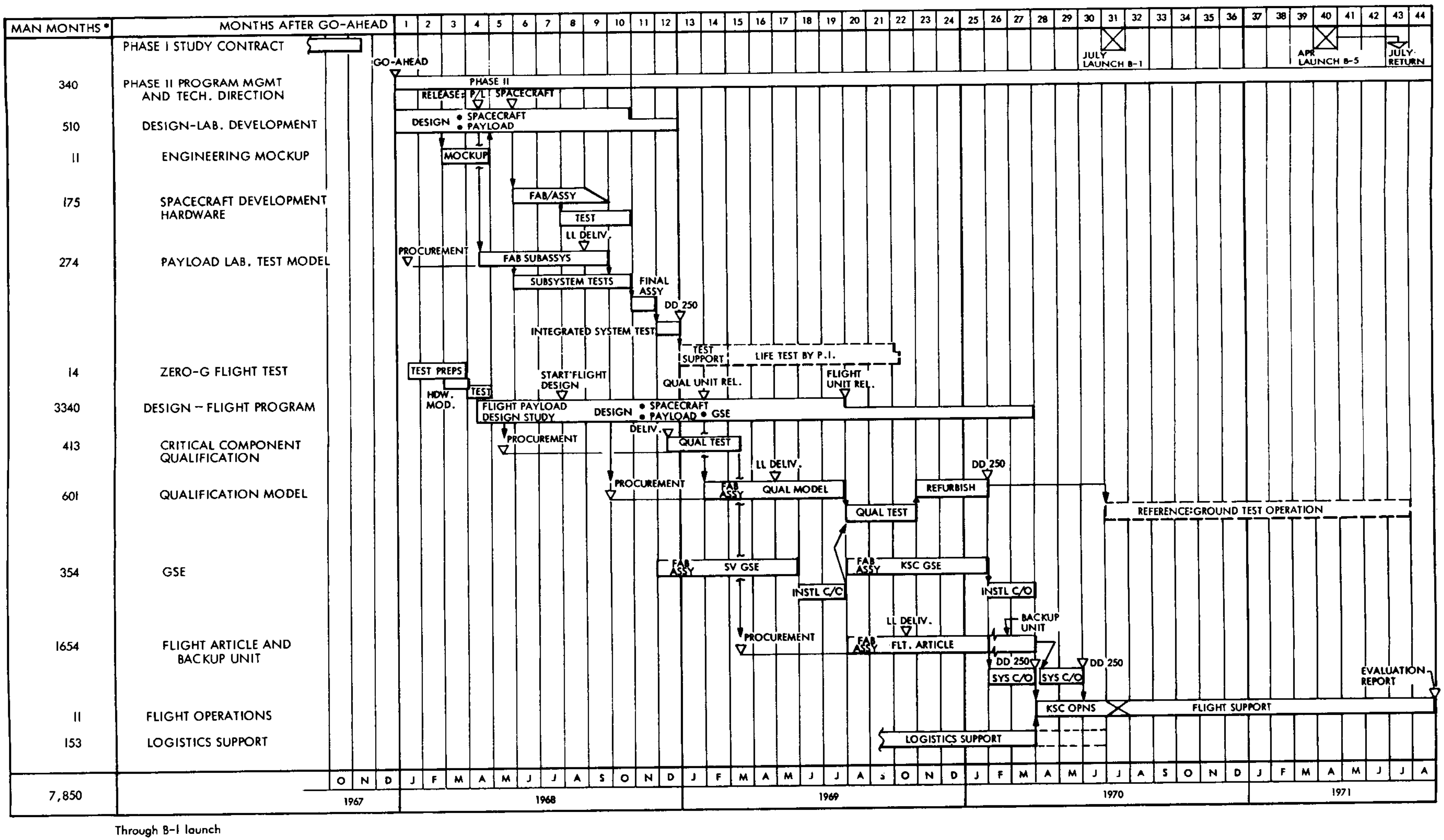

Fig. 158 OPE Summary Master Schedule

The basic process from which all management actions derive is depicted in Figure 159. The established methods and techniques for implementing these steps are described in subsequent sections.

Work breakdown structure (WBS). – Figure 160 illustrates a summary-level WBS for Phase II of the OPE program. With the elements of work organized into packages and the work package defined, the responsibility for accomplishment established, and the start and completion dates ordered, task control can be maintained. A complete Work Breakdown Structure is required and will further detail the Summary WBS. Responsibilities for contract requirements are programed into a computer for subsequent monitoring of performance.

Task control. – Jurisdiction of task control will be placed under the direct supervision of the Assistant Program Manager. Support for this activity is drawn from Industrial Accounting personnel who monitor the budget-control system which is keyed to the Work Breakdown Structure discussed below.

Initial organization of the Program content based upon a complete WBS will form the basis of the accounting structure. Cost accumulation charge numbers assigned to match the WBS provide management visibility of budget status. Schedule monitoring and status reporting is also based upon the same work packages.

For hardware end-items, accurate technical job content definition and estimated man-hours will be handled utilizing the Engineering Job Analysis (EJA) form. A control number assigned to this form will be used to identify the work package from its inception in engineering through the manufacturing and test activities. Approval by the Program Manager authorizes the work described. Changes within the scope of the contract are handled through the use of the EJA, while changes outside the scope of the contract will require Engineering Change Proposal activities with negotiation and approval of the proposed change.

Completion of job packages result in closure of the appropriate cost collection numbers to preclude further charges being assessed to the account.

All related Management Control activities of task definition, budget allocation and negotiation, and schedule control working to a common structure of work packages provide one of the keys to on-schedule, on-budget performance.

Task control will be performed continuously to verify conformance to plan. Schedule and cost will be monitored by schedule and cost analyses. A series of program-control and financial computer systems facilitates the management of work in process. Reports that show for each task the budget variance to date, forecasts for the next three months, and the total budget estimate to completion will be generated for management and operating organizations. An integrated series of reports monitors the status of end-item requirements. Analytical reports of all cost elements and Contract Cost Summary reports will be generated in line-code sequence. Unit performance reports will be specially tailored to monitor schedule, manpower, and budget performance at the operating level.

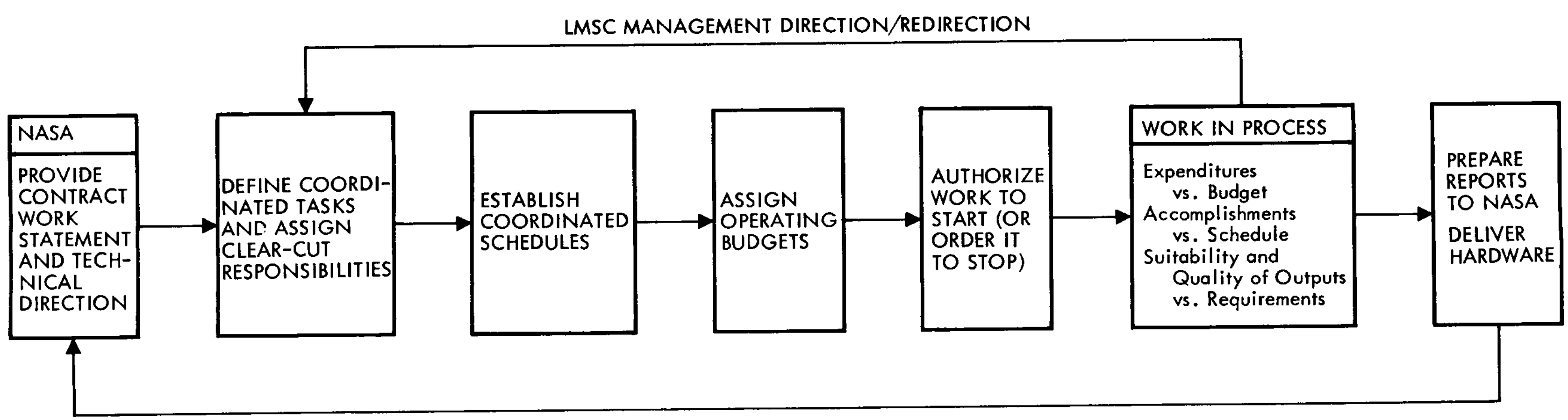

Fig. 159 Management Process

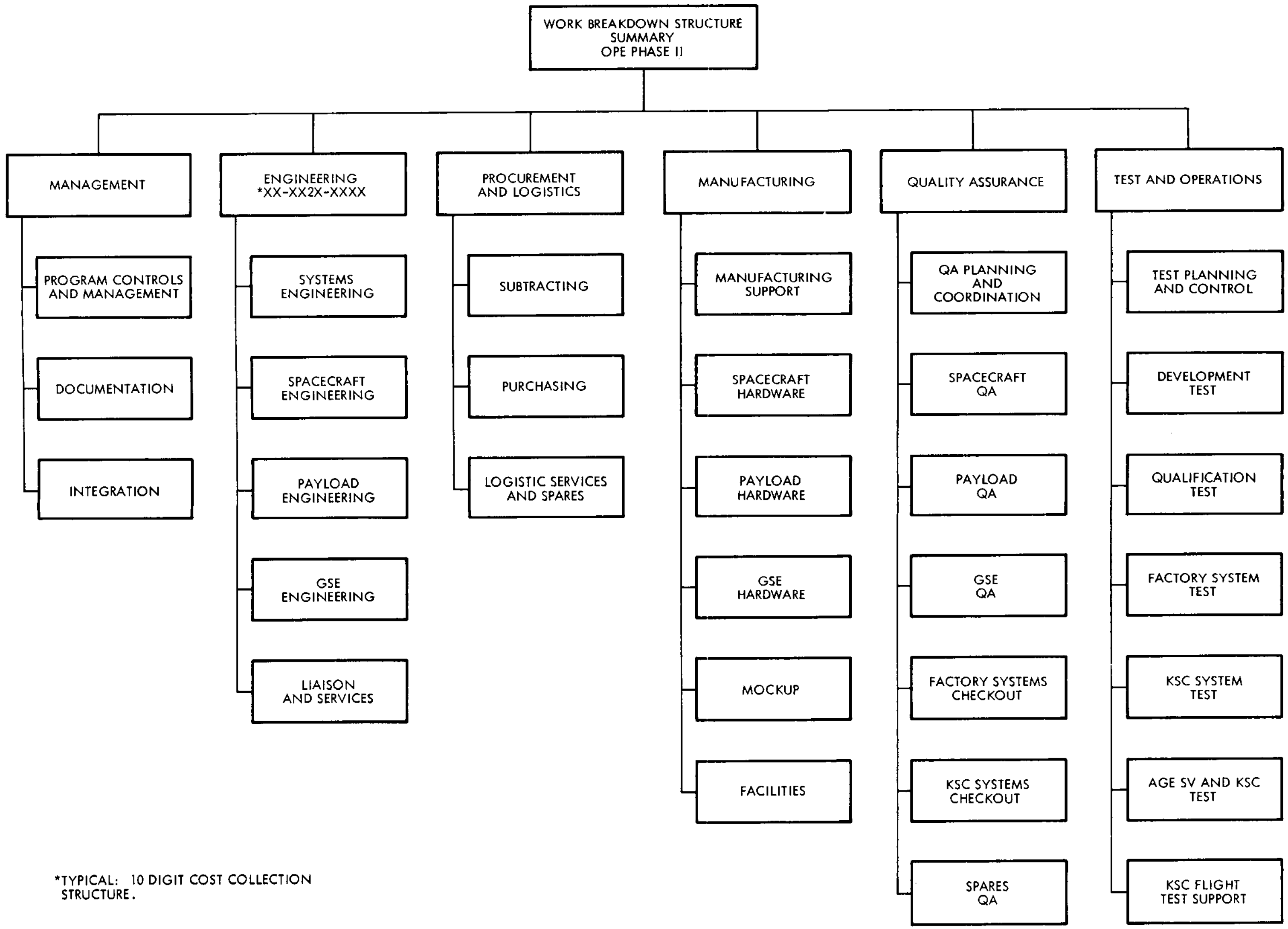

Fig. 160 Summary Work Breakdown Structure

Advanced information systems currently under development include one that generates graphic displays from budget system files; the users of these displays can tailor the outputs of their requirements by use of parameter cards. A direct measurement and prediction program will produce budget planning charts and reports using advanced mathematical techniques to forecast future expenditures. An integrated financial management system (STARFIRE) is now being programed that will optimize accounting controls and financial reporting and provide real-time capability through the latest remote input-output equipment, random processing, and high-capacity memory and storage.

Engineering. – This function will monitor unreleased engineering requirements by maintaining and reporting the initial configuration of the top assemblies, subassemblies, and critical components.

Make or buy determinations. – LMSC will act in the best interests of NASA by comparing supplier capability with Lockheed capability and by considering such factors as costs, technical requirements, development status, schedule, volume, and quality. Recognized suppliers who specialize in fabrication of required equipment or who offer unique services or products will be selected as sources of those services or products, if schedule compliance, cost, and technical competence are determined to be satisfactory.

Integrated procurement system. – This system provides computerized assistance in financially controlling inventories. Status is maintained by a network of automatic inputs of all requirements, order, receipt, and disbursement activity. A remote real-time inquiry capability provides instant inventory status as well as the status of all orders released into the procurement cycle. This computerized system will be utilized on the OPE program.

Integrated manufacturing system. – This system mechanically produces order-writing ledgers, shop orders, and kit lists. Job-assignment, acceptance, and order-move data are collected by remote input devices and are fed to the real-time order-location system and the off-line shop-order control programs. These programs provide Manufacturing with reports on labor performance, order status, machine utilization, and open order and location. In addition, labor cost is calculated and passed to the financial system. This integrated system will be utilized in controlling the manufacturing requirements of the OPE program.

Program Plan

Preliminary plans for the evolution of the Orbiting Primate Experiment from the present stage of preliminary design, through the construction and test of the payload test laboratory model to completion of the actual one year duration orbital flight are presented below. Preliminary plans are presented for the overall program followed by preliminary engineering, manufacturing and test plans. These are followed by outlines of reliability and quality assurance, and documentation plans, an outline of facility requirements and a summary of advanced technology and advanced development requirements. It is further planned to complete these plans in detail within 60 days commencement of Phase II.

A go-ahead for Phase II of 1 January 1968 has been assumed, which allows a time span for development of 30 months; with the launch of the experiment (OPE) occurring in the 31st month aboard the selected S/AAP vehicle. Current schedule information indicates a July 1970 flight for Vehicle 216 which was chosen as the launch vehicle on the basis of weight capability. Changes in the scheduled launch of this vehicle could require corresponding revision in the program for the OPE.

The early effort of Phase II will accomplish the design, fabrication and test of the payload laboratory test model, as well as spacecraft development test hardware. Utilization of the data package (drawings and specifications) delivered at the conclusion of Phase I and the information derived from related LMSC Independent Research and Development efforts will allow early procurement of critical long lead time payload components for assembly into the affected subsystems. Test of these subsystems precede the final assembly of the payload lab test model. The previously proven elements of the payload laboratory test model will follow the more critical elements with lesser priority in the design and fabrication cycle. Assembly of all elements of the laboratory test model will be followed by system and acceptance test and delivery to NASA for test activities by the principal investigator. Duration of these tests may extend to one year, to give elements of the system a life cycle proof test. During this period, LMSC will provide necessary on-site support and will utilize test results to refine qualification and flight system designs as appropriate. Development tests are also planned on spacecraft hardware in time to provide data for the final design of the qualification test model.

Zero-g aircraft tests of critical items such as the waste management and animal retrieval subsystems are planned early in the program to verify preliminary design concepts. Such testing can be accomplished by modifying hardware already available at LMSC.

Concurrent with the initial release of design of the laboratory test model, further payload design studies will be carried forward in the critical flight subsystem areas. A product of this effort will include the selection and placement on order of long lead time procurement items for qualification test of critical components.

Initiation of detail design of the flight test article must start by the end of month seven which is well into the development test span of the critical components and subsystems of the payload. A 6-month design span is allowed prior to fabrication of the qualification test model. Initial flight unit releases must start at the end of month 19, allowing a minimum of 6 months to fabricate and assemble the flight test article.

Systems and acceptance tests of two months duration conclude with DD 250 at end of month 27. A back-up flight unit will be fabricated with the flight article. The back-up unit will complete assembly and test two months later. KSC operations of 3 months have been allowed with the back-up unit at KSC one month before flight.

Refurbishment of the qualification test article is planned with delivery and shipment of the article to NASA at month 25. This model will then be available as an operating unit to function on the ground while the flight article is in orbit.

Ground Support Equipment peculiar to the program, identified in the Final Mission Analysis section, will be provided to support both systems test at the factory and the KSC operations.

The overall development schedule as depicted in Fig. 158 while not ideal in allowing completion and reporting of test activities before starting all elements of ensuing design and fabrication work can be achieved by an efficient industry/NASA team. This schedule is based on meeting the mid-1970 flight date described earlier.

A network of activities and events will be developed within the constraints of the Program Master Schedule for processing by the PERT computer system to determine slack time, schedule inconsistencies, and critical paths. The results of detailed task definition and PERT schedule analyses will be used to expand and refine the Program Master Schedule.

Figure 161 indicates a preliminary listing of milestones for Phase II of the OPE Program. In the preparation of the complete program plans at the outset of Phase II, the listing will be expanded to include NASA milestones and AAP interfaces which have a major bearing on schedule achievements.

Singled out for special mention and emphasis is milestone Number 11, the completion of systems test and DD-250 of the Payload Laboratory Test Model. Failure to meet this milestone would have a severe impact on the flight program with the initial qualification model designs just one month from release. Also, on schedule completion of milestone Number 6, the procurement of critical components for qualification testing which have been identified in the Payload design studies, appears essential.

An earlier and important milestone which will contribute to the overall conduct of the program is the completion in detail of all program plans for Phase II. In addition to those plans included herein in preliminary and outline form, would be:

- Configuration Management Plan
- Logistic Support Plan
- Training Plan

Engineering plan. – This section outlines the Engineering effort required in Phase II of the OPE Program. A preliminary Engineering Schedule, Fig. 162 , portrays the major division of the Engineering tasks as spanned over the Program time period.

The design concepts and preliminary designs of Phase I will be carried further into the detailed design and development of deliverable equipment. The steps to accomplish this work are outlined in the subsections below.

Systems engineering: A Systems Engineering group responsible for compatibility between all elements of the spacecraft, payload, and the support equipment will be established. Definition of spacecraft/payload requirements through to the submittal of flight data evaluation reports will be handled by this group.

1. Interface Control Document – An Interface Control Document (ICD) will be prepared, which will cover all mechanical, electrical, power, thermal, data subsystem and ground support equipment interfaces. Coordination of this document with the Principal Investigator, OPE Contractor and NASA/Langley will be required.

2. Experiment operations analysis – Support of Flight Operations will be provided in the analysis of data received from both the orbital payload and the ground operating experiment.

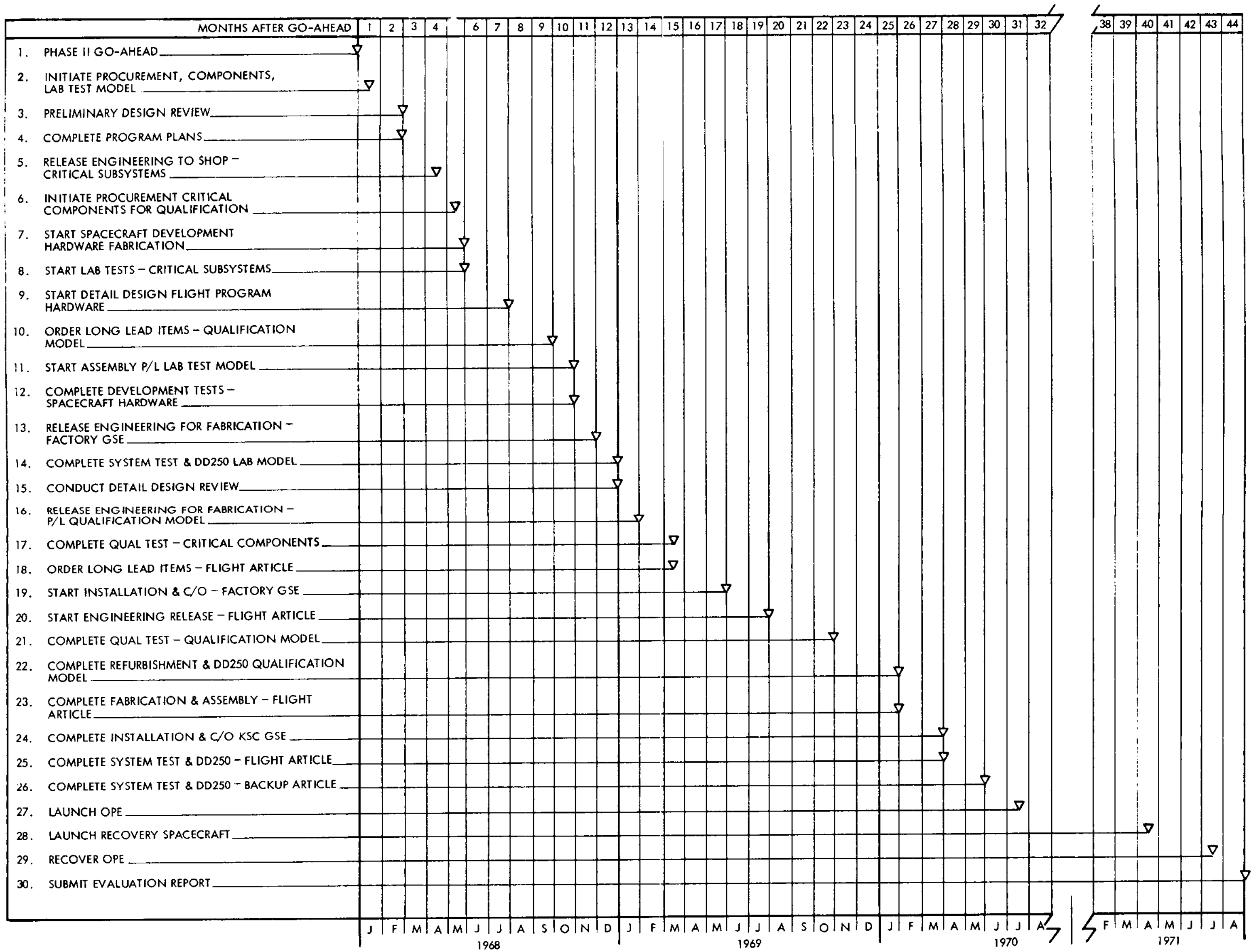

Fig. 161 OPE-Major Milestones Phase II

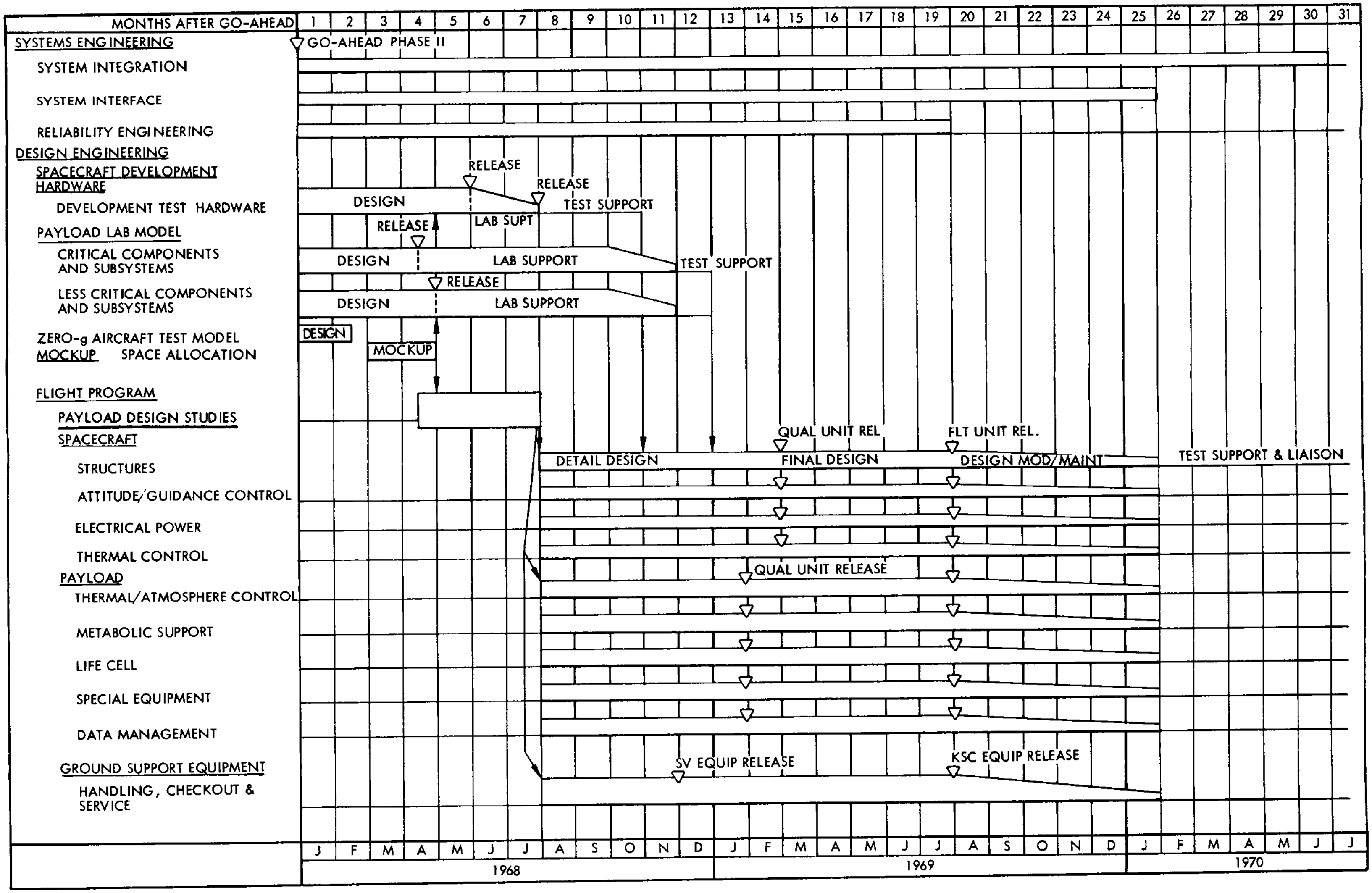

Fig. 162 OPE-Engineering Schedule

3. System requirements – Technical requirements for the design and test efforts of Phase II will be prepared in conformance with the Work Statement and submitted for approval. This will be an up-dated version of the System Requirements and Design Criteria document prepared in Phase I.

4. Design reviews – Adequacy of design, analyses, manufacturing, testing and documentation will be assessed in scheduled design reviews. Preparation of a Design Review Plan at the initiation of Phase II, to establish procedures, schedules, and reporting requirements will be required. Reviews at the System, Subsystem and equipment levels will be specified.

Design Engineering: Design Engineering functions will be performed by three distinct but coordinated sections:

- Payload design
- Spacecraft design
- Ground Support Equipment design.

1. Payload design – Emphasis in the early months of Phase II will be on the development of the critical subsystems of the payload laboratory test model. Design of the payload subsystems will proceed from the outset of Phase II based upon the data package prepared under the Study Contract and supplemented with the results of related LMSC Independent Research and Development efforts. Release of engineering to support the fabrication of development test hardware considered to be of a critical nature includes the waste management system, the mass measurement device, the elements of canister retrieval and the animal feeder three and one-half months after contract award. The cognizant designers will follow the construction of hardware through fabrication and test while commencing effort under the Flight Program of further payload design studies. Directly from these studies and utilizing the early test results of the development hardware, detail design of payload subsystems will proceed.

2. Spacecraft design – Continuity in the utilization of Engineering personnel will also be employed in the design of the spacecraft. Beginning at the outset of Phase II, design of the spacecraft and its subsystems results in releases to the shop for fabrication of development test hardware at the end of the fifth month. At this time a Thermal Test model is indicated to be required as well as a spaceframe for structural test, ascent and docking load tests. A large or full scale inertial model to determine actual capture times, recovery from transient disturbances, fuel consumption, etc., is also required. Again, close liaison with the shop will be maintained throughout the fabrication and development test by the assigned engineering personnel. Detail design of the spacecraft subsystems is indicated to commence at the beginning of month eight.

Design releases for the qualification test model of the payload and spacecraft are at the end of months 13 and 14 respectively. Final design releases for lot fabrication of two units, the flight article and one backup, commence at the end of month 19 in order to complete the hardware in time for the specified launch date.

Maintenance of design will continue through hardware delivery, and test support and liaison throughout the experiment operation as required.

3. Ground Support Equipment design – Early definition of all Ground Support Equipment requirements will be made in Phase II with the detail design provided to support

the fabrication of GSE for factory handling and test of the qualification test model and flight hardware. Engineering release for GSE for use at Kennedy Space Center will be required at the beginning of month 20 to support manufacture.

Reference to the Final Mission Analysis Section should be made for a listing of GSE requirements.

Engineering of GSE will consist of three functional sections:

- Handling
- Checkout
- Servicing.

Manufacturing Plan

The purpose of this preliminary plan is to outline the LMSC manufacturing approach for development of hardware for the Orbiting Primate Experiment. It is based on preliminary master schedules and current level of design definition as presented in this report.

A Phase II contract award will require fabrication and assembly of the following items:

Qty	Item	Comment
1	Mockup	To verify space allocation and producibility factors
1 set	Development Hardware	Including development assemblies for thermal control, structure and antennas, attitude control, human factors, and zero-g aircraft test
1	Laboratory test model	Flight qualifiable unit
1	Qualification model	
1	Flight unit	
1	Flight back-up unit	
2	Ground support equipment	

Schedules. — A preliminary manufacturing schedule is presented in Fig. 163. It is necessarily predicated on early release of engineering definition to permit initiation of manufacturing efforts for the mock-up, for development hardware and for the payload laboratory test model. Close adherence to schedule is mandatory. Continuing coordination between engineering and manufacturing planning will be maintained until release of flight unit design. Much of the planning and long lead procurement will be based on prereleased engineering drawings.

Tool design for the laboratory test model should begin in the second month, and a parallel effort for development hardware fabrication and test tools should begin in the third month. Peak loading for tool design will occur in the third through the fifth months. Maximum load for tool fabrication will be in the fifth and sixth months. For the balance of the program, the tool requirements are spread over the first nine months of the second year and are to be initiated from prereleased drawings.

Fabrication and assembly requirements for development hardware and the laboratory test model are roughly parallel, with the maximum manufacturing effort occurring in the sixth and seventh months of the schedule. An exception is the modification of existing hardware for zero-g flight test which will be handled on an informal basis during the third month. The flight program will develop its peak shop load in the last quarter of the second year. The schedules for assembly of the payload and spacecraft

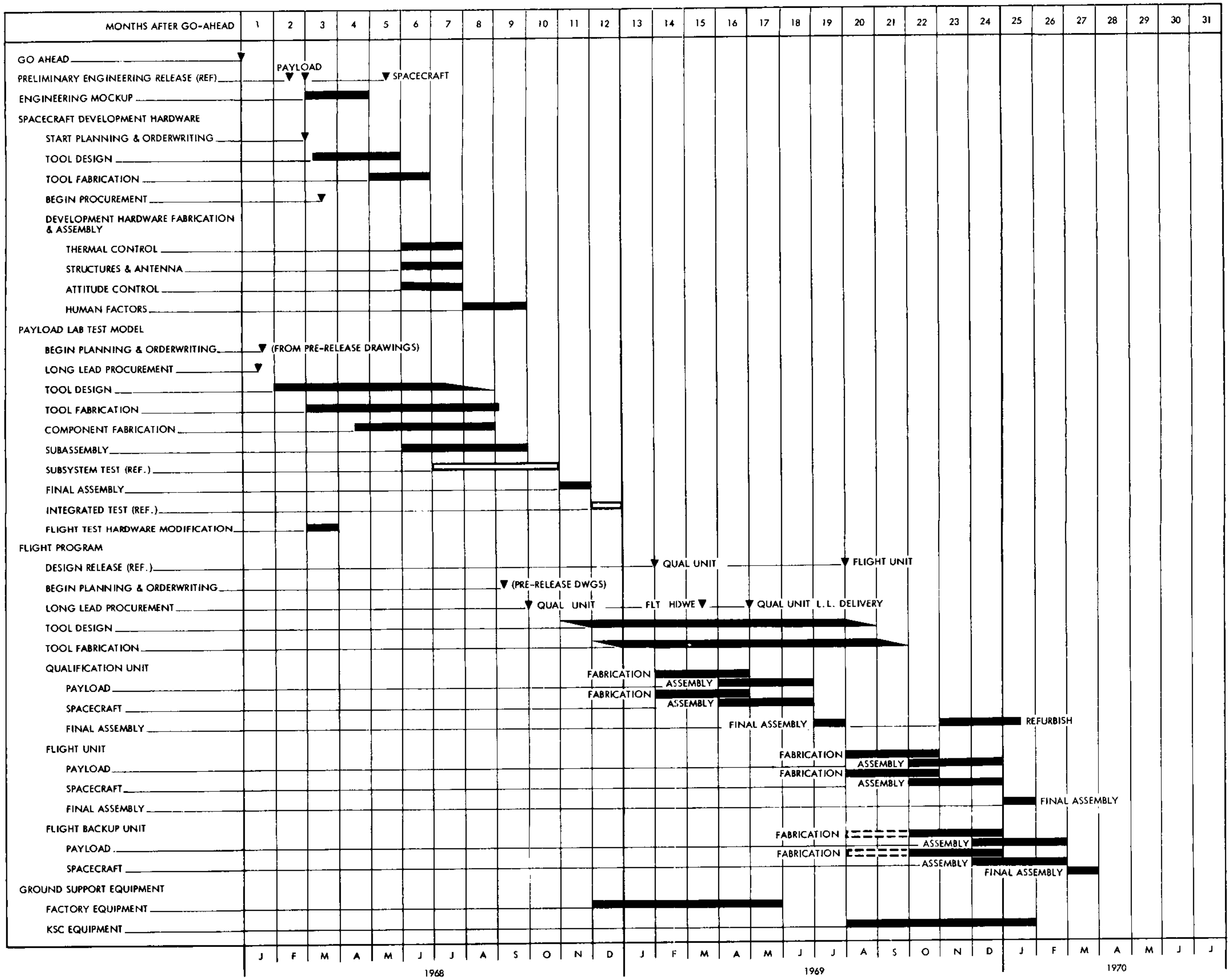

Fig. 163 Preliminary Manufacturing Schedule

units indicate a one month overlap for the flight and back-up articles. Special consideration will be given to assembly planning to avoid duplicate assembly tooling requirements.

Manufacturing policies. — The following paragraphs outline the basic policies which apply to the manufacture of the orbital primate experiment as presently defined.

Production planning: The schedule requirements of the Phase II program place production planning in the role most critical to success of the manufacturing portion of the program. Accordingly, this activity will include:

- Maximum interface with engineering design, procurement, integrated test, and quality assurance, throughout the entire program period, to ensure schedule compliance and coordination of the respective functions.

- Preplanning, concurrent with design engineering, to define sequence and content of assemblies, ensure producibility, and establish tooling requirements. Particular emphasis will be placed on those manufacturing activities which must be committed prior to, or immediately following, the formal release of engineering drawings.

- Participation in design change board activities to reflect the requirements of the manufacturing plan and to ensure planning compliance with program requirements.

- Provision for liaison planning coverage in tooling, mock-up development, and manufacturing areas as necessary in support of manufacturing activities.

- Preparation of production shop orders on a minimum paperwork basis (snap-out forms) for all fabrication and subassembly operations. Shop orders will specify processes and sequence, and will provide for inspection verification.

Manufacturing Process Control will be invoked by the manufacturing planning paper. This system provides configuration control from raw stock to final assembly, and complete traceability through the log book system.

The production planning function will utilize documentation procedures which have been refined over a ten year period in the fabrication and assembly of spacecraft.

Tooling: One complete set of tools will be fabricated for manufacturing use. Tool concepts will be established in preplanning. Detail tool planning will be carefully coordinated with scheduled sequence of component fabrication and hardware assembly. To the extent practical, tools required for the development hardware and for the payload test model will be employed, with modifications as required, for the flight program hardware.

A minimum tooling policy will apply, with maximum utilization of optical tooling and dimensional control. Where practical, physical mating of tools will be employed, in lieu of producing master gages. The use of replaceable components rather than full interchangeability of parts is recommended, to eliminate requirements for master or control tooling. Assembly tools will be designed with a minimum of detail part and subassembly locators. For the limited quantities required, close liaison and the use of the most skilled shop personnel will provide a more economical approach to hardware assembly. Provision will be made to trim mating parts for fit-up prior to assembly, to reduce trim fixture requirements. Standard fabrication equipment and set-ups will be specified in lieu of component fabrication tools wherever practical. Manufacturing aids necessary for handling or protection of components or assemblies will be closely coordinated with test and GSE requirements. When possible, the item will be designed for multipurpose application.

All tooling for the Orbiting Primate Experiment will be provided through the well established LMSC system, tool planning will be prepared by specialists assigned to the program and will be coordinated before release. Design, fabrication, and inspection of tools will be performed by existing support organizations in accord with previously negotiated schedules. Planning liaison will be maintained during the period of support organization activity.

The established LMSC tool design standards, drawing format, procedures, and tool code system will be employed. All tooling, except "shop aids" will be designated type X (experimental) and will be controlled under the procedures defined for such tooling. All coded tools will be the property of NASA and will be dispositioned in accordance with Contracting Officer direction.

Manufacturing control: Manufacturing will fabricate, test, and/or perform assembly operations in accordance with authorizing documents, which will reflect configuration conformance to released engineering drawings. Configuration control will be documented by Log Book, and will be serialized in terms of the using end item. Changes will be accomplished and documented by applicable serialized Operations Order.

Manpower budgets are allocated to affected manufacturing organizations and time phased in accordance with approved operating schedules. Actual expenditures will be monitored and performance against budget will be reported and reviewed weekly with the doing organization. Budgets for manufacturing are negotiated with the program management.

A work order, work authority structure will be established to accumulate costs by segment. Schedule/cost/performance reports, based on hardware-complete status, will be issued and reviewed with program management to identify problem areas and permit timely corrective action.

Work performed in the development assembly area is under the direct control of program management and, therefore, will not be subject to time standards control. On those operations performed by supporting manufacturing organizations, time standards will be utilized to measure productivity.

Specifications governing manufacture: Phase II contractual requirements and program engineering specifications will be carefully reviewed for reference to manufacturing requirements. Where definition is not provided, additional manufacturing guidance will be developed, as appropriate. Specifications, standards, and procedures will be selected, in descending order of priority, from among the following groups of documents:

NPC/NASA-Langley
DOD
Federal
Industry
Company

The resultant list will be submitted for Program Manager approval, and, when approved, will form the basis for fabrication and assembly of the hardware.

Manufacturing operations: Component fabrication will be performed in the LMSC central manufacturing shops which normally provide this service. A similar support function for electronic units is provided by the LMSC electronic manufacturing organization. Specialized shop and laboratory capabilities are available within the company for other required manufacturing support. The support organizations are responsible to program planning for performance against budget and schedule commitments.

Development assembly will be performed in a program controlled assembly area, where progress may be readily observed by responsible engineering and management personnel.

<u>Logistics services</u>. — Logistics support will be provided for manufacturing, test operations, and launch. A logistics analysis will be made early in the program to establish provisioning lists, which will include spares, supplies, materials, pyrotechnics and similar items which will be required at various points in the program schedule. Design and schedule change documentation will be monitored and analyzed for logistics impact. Adjustments will be made as necessary. Expendable materials and components will be identified, and inventories will be established on a minimum/maximum basis at the manufacturing site, and at the launch site.

The established LMSC system of logistics control will be employed for the Phase II program. Schedules and Status Reports are provided to program management on a routine basis. This system is part of the R&D Division Production Control System, and employs key punch inputs to a 407 unit which is programmed to produce the desired reports.

Logistics personnel, within the production control organization, will perform the initial analysis and establish the logistics program. Once established, routine part-time assistance will be provided to monitor and respond to program changes and design releases. This support will continue to be available to the launch date. In the event that a concurrent experiment is conducted on the ground at KSC, provisioning support will be provided.

Integrated Test Plan

The Integrated Test Plan encompasses all categories of testing necessary in the development, qualification, and acceptance of the Orbiting Primate Experiment.

These categories include:

- Component, subsystems development and qualification tests.
- Component, subsystems manufacturing and acceptance tests.
- Component Life and Reliability Tests.
- Systems environmental, and compatibility tests.
- Prelaunch tests.

The Integrated Test Plan is a basis for the coordination of planning and conduct of all testing effort and contains only that depth of detail necessary to identify and define the tasks and support required.

Figure 164, OPE Integrated Test Schedule, illustrates the major test activities throughout Phase II.

Test concepts. — Program requirements for an on-station life of 1 year with a high reliability factor demand the establishment of a test program that will ensure that potential flight malfunctions will be located, identified and corrected prior to launch. To accomplish this task, the OPE Test Program will be conducted to satisfactorily demonstrate the use of adequate designs, components and manufacturing processes and to ensure compliance with the enviromental, qualification, subsystems and systems specifications.

Subsystems and systems testing of the OPE production units will be performed to validate the individual spacecraft and to demonstrate a flight worthiness sufficient to perform its mission at or above the established reliability goal.

Finally, prelaunch tests will be conducted to verify OPE readiness for launch.

Test management. — LMSC will be responsible to NASA/Langley Research Center for management and technical surveillance of all testing described in the Integrated Test Plan. Testing will be performed primarily at LMSC facilities. OPE Engineering will have primary responsibility for detailed planning and performance of the tests. Technical control of OPE test activities will be achieved through design reviews, configuration management, test reports, program audits, and related techniques.

Test procedures: Formal test procedures will be prepared to cover all tests presented in the Integrated Test Plan, except for development tests. Primarily, preparation will be the responsibility of the test agency, with approval by Engineering and Product Assurance. Testing will not commence until approval is received. Product Assurance will witness and verify compliance with the test requirements and test conditions specified in the approved OPE Test Procedures.

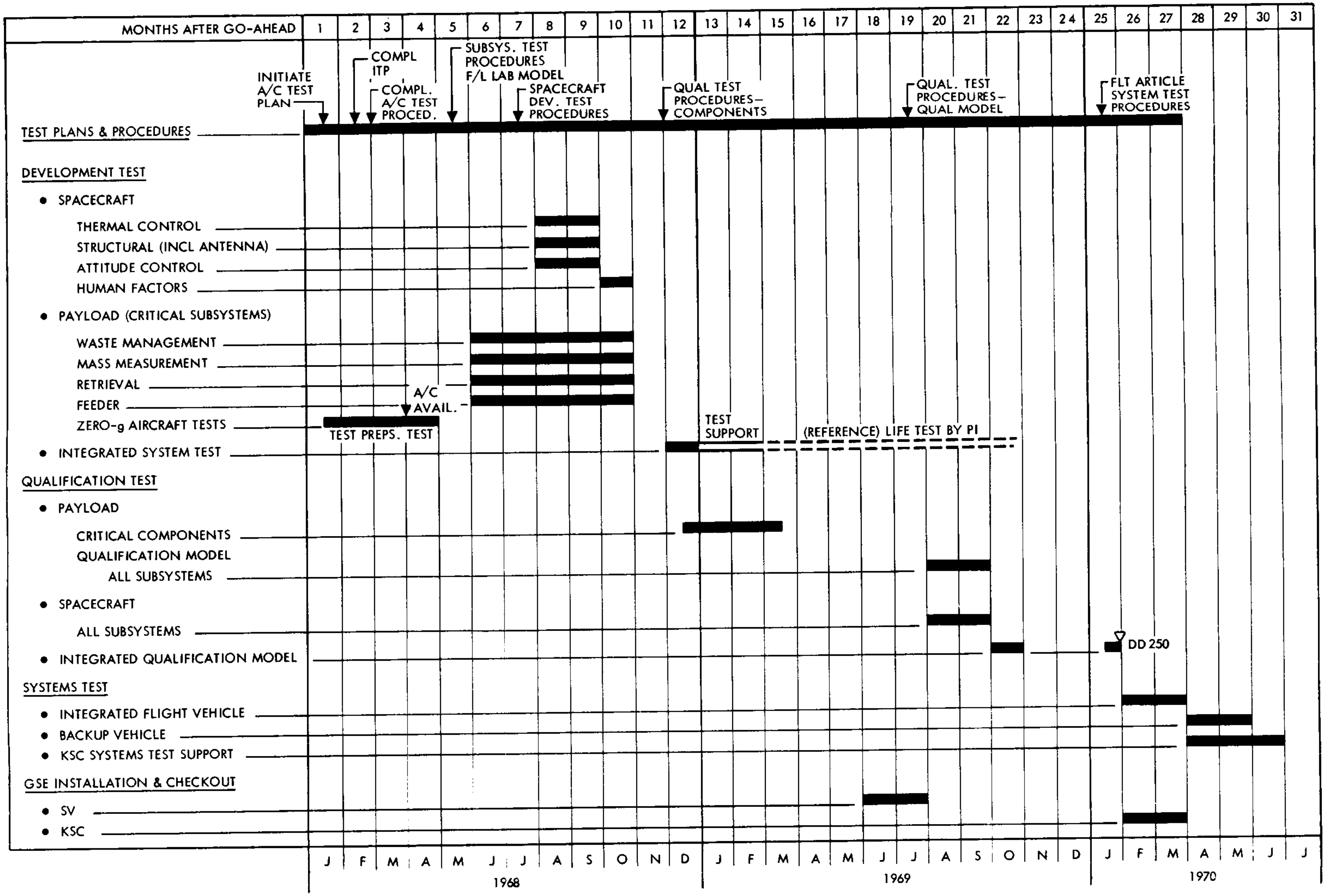

Fig. 164 OPE-Integrated Test Schedule

Operating time log: An operating time log will be maintained to show total accumulated operating time for each system and subsystem of a flight item configuration. The log will consist of a list of all parts that have a use or ground operating lifetime limitation, as determined by design engineering, and will provide data on cumulative time, remaining useful operating time, and a replacement schedule.

Failure reporting and corrective action: A system of failure reporting and corrective action will be applied to all phases of the OPE Test Program. Failures that occur during progressive development testing will be documented and reported to the responsible engineering organization. Failures that occur on tests being conducted to support the qualification program, qualification tests, or acceptance tests, will be reported through the formal discrepancy reporting system.

Final system test: The Final Systems Test conducted by LMSC on the OPE and other deliverable items will be witnessed by the local NASA representative. Systems Test Procedures (including functional diagrams and schematics of OPE hardware, test equipment, and test equipment/OPE interfaces) will be supplied to NASA before checkout begins.

Prelaunch installation and checkout: LMSC will perform a complete checkout of the OPE Spacecraft at the launch site. GFE suppliers will participate to the extent required. LMSC Operations will conduct the OPE Installation and Integrated Prelaunch Checkout. LMSC will adhere to NASA approved launch site operating procedures as applicable to the OPE Program.

OPE log book: The OPE Log Book will be maintained by OPE Product Assurance and will consist of the following information:

- Operating Time Log
- Configuration Record
- Replacement Record
- Weight and Balance
- Acceptance Data
- Failure and Corrective Action Record
- Open Items
- Shortages and Ship Separate Lists
- Spares List
- Nonconformance Status
- Operations and Inspection Records
- Qualification Status List
- Records of Limited Life Items
- Test Procedures and TPCN's

Test data control: A test data control system will be employed on all phases of the OPE Test Program to ensure that data is recorded, analyzed, and stored so that it can be readily retrieved and reported efficiently. Any data which may cause a change in the design or modify the reliability goals will be transmitted to OPE Design or Systems Engineering and OPE Reliability.

The test data will be recorded manually on test procedure data sheets and in the Vehicle Log Book. When required, additional data will be recorded on oscillographs, chart recorders, or magnetic tape recorders. The records will be verified by inspection, processed, and the necessary plots and/or visual displays made from the magnetic tape playbacks.

Data requiring computer analysis will be forwarded to OPE Engineering for processing in the Data Reduction and Processing Equipment System.

Data Analysis at the launch test facility will be conducted by the LMSC test engineer, LMSC data analysis group and OPE Systems or Design Engineering.

The test data, oscillograms, and magnetic tapes will be available for review during the test period, and will be retained by the testing organization for at least two years after launch.

Development test. — The objective of the development tests will be to evolve functional hardware which demonstrates capability to meet program objectives. Development tests will provide information necessary to evaluate feasibility of the design concept, and demonstrate that the equipment will operate as expected. Scope of the Development Test Program may be relatively fluid, with changes being made as information is obtained.

Since the development test program encompasses all the test activity preceding the formal qualification testing, it is expected that several levels of detail and formality will be necessary. Breadboard tests may be conducted with a minimum of formality and documentation. Materials tests will be performed in accordance with test procedures which are accepted as good practice in the aerospace industry. Each major development demonstration test will be performed in accordance with a formal Test Specification. Development demonstration test procedures will be applied to components for verification and demonstration of the final design. These tests will encompass a series of functional performance and use environmental tests to establish envelopes, constraints and capabilities, and verify operation in the use environments.

Critical subsystems: Initial design verification development tests will be required on the following subsystems:

- Payload

  Waste Management

  Mass Measurement

  Retrieval of the Primate

  Feeder System

- Spacecraft

  Thermal Control Capability

  Structural Integrity

  Attitude Control

  Human Factors (Retrieval Methods)

  Antenna/Solar Array Positioning

Zero-g verification: Zero-g aircraft tests will be conducted on a preliminary laboratory model to verify design concepts such as waste management and animal retrieval which are notably gravity affected. These tests will be conducted through NASA authorization of Air Force flight test facilities.

Qualification tests — general requirements: Early qualification tests will be conducted on critical components. These tests will be conducted early enough in the Program to allow results to be factored into the qualification test model fabrication effort. Payload and spacecraft models are qualified individually and concurrently. Following this the payload and spacecraft will be mated and tested as an integral unit.

1. Test articles — The specimens tested shall be designated as Qualification Hardware and shall be identical in configuration and production processing to the Flight Hardware. Any differences existing between the Qualification Hardware and the Flight Hardware will invalidate flight qualification except where written approval is granted to qualify the Flight Hardware by similarity. Normally, this approval will be given for minor modifications. Any major modifications will necessitate retesting in whole or in part.

2. Test specification — The test specification and changes thereto for qualification testing will be prepared by OPE Engineering and submitted to NASA for approval. If scheduling problems occur such that testing must commence before this approval is received, the experimentor may do so at the risk of possible retesting to meet requirements.

3. Sequence of tests — The sequence of tests normally will follow the same order in which the environments will be encountered during the mission. If scheduling of test equipment and other factors work an undue hardship in attempting to adhere to this criterion, the sequence of tests may be altered subject to approval, noting that changes in test sequence may require changes to the test specification.

4. Functional tests — Prior to conducting any functional tests, the hardware will be inspected for compliance with the applicable design, procurement or acceptance test specifications. A functional test, to determine whether the hardware is performing within specification tolerances, will be conducted before and after each environmental exposure. The same functional test will be performed during the exposure period if the equipment is required to operate in that environment during the mission. If the tests are run in series with no significant time interval between tests, the functional test after an envrionmental exposure may serve as verification of proper performance before the succeeding environmental test.

5. Failure definition — The following conditions will be classified as failures:

- Performance outside of specification tolerances during and/or after any of these tests, whichever is appropriate.
- Any physical damage or degradation which impairs the operational design characteristics of the component.
- Any loose, bent, cracked or otherwise damaged or improperly adjusted parts resulting from the test inputs.

6. Testing level — Tests will be conducted at the system level of assembly if practical. If not practical because of size and complexity of the equipment, the testing will be done at the next lower level of assembly.

7. Test mounting — Equipment will be mounted in a manner simulating the actual mounting in the spacecraft for all environmental tests wherein the equipment is expected to be affected by the spacecraft mounting.

8. Qualification approval — A final environmental qualification test report will be submitted to NASA for approval by the OPE Program office. Disapproval will be a constraint upon flight qualification. Consequently, the qualification test program will be scheduled such that there is sufficient time to allow for failures, rework during testing, preparation of the final test report, and review after submittal.

Qualification approval may be granted upon presentation of documentation that the equipment was tested to equivalent or more rigorous levels of environmental testing. This will be considered on a case-by-case basis. Equipment may be qualified via a combination of previous testing and some retesting for those environments which were not of a sufficiently high level or were omitted previously.

9. Qualification status — The contractor will include in the Monthly Progress Report a status report of the qualification testing. The report will indicate which tests must be performed on each component and which of the required tests are completed at the time of statusing.

10. Inspection — Government Source Inspection will be required for all environmental qualification testing.

Reliability tests: Reliability testing will be controlled by the Reliability Plan. Where life testing is desirable, but not practical due to time requirements, an accelerated test will be performed.

Component acceptance tests: Component acceptance tests are a variety of tests from piece part inspection through module level tests.

Testing of items, components or subsystems that are subcontracted by LMSC will comply with the Detailed Specification and controlled as outlined in the Quality Program Plan.

Receiving inspection of all items, components and systems manufactured outside of LMSC will be controlled by Product Assurance as outlined in the Quality Assurance Plan.

Manufacturing tests will include all tests necessary to verify correct assembly and/or function of parts, components, subassemblies and complete modules. Manufacturing tests will be controlled by the Integrated Test Plan.

Environmental testing of structural components and assemblies will be done during development and qualification test spans. Acceptance testing of structural components and assemblies will consist of "in process" and final inspection of the item for workmanship and drawing conformance and, if necessary, alignment checks.

Acceptance testing of electronic components is of particular importance. Experience indicates that exposure to both thermal and vibrational environments can be valuable aids in the detection of defects in workmanship on marginal electronic components.

The specific acceptance test sequence for any particular unit will be determined for the design involved; however, a general environmental test sequence can be applied as follows: A functional test will be performed at the upper and lower temperature limits and again tested at room temperature. The final steps will be to subject the unit to random vibration. When practical, the functional test will be done and data gathered during the vibrational environment.

After completion of environmental testing, the unit will be subjected to a final acceptance test. Following this the unit will be ready for installation in the spacecraft for systems level testing.

Systems acceptance tests: These tests will satisfactorily demonstrate the performance of integrated subsystems to the design requirements. Subsystem tests will be designed to verify proper operation of integrated component assemblies and to take data that cannot be monitored or measured to required accuracies during simulated flight.

Integrated system test is a simulated mission profile within the limits of the test environment. It will include prelaunch checks through "on station" functions.

Reliability and Quality Assurance Plans

This plan outlines the several tasks, which form the means by which the reliability of the spacecraft/payload, its probability of success in meeting mission objectives, and the measures necessary for achievement of reliability objectives as may be defined by NASA are to be assured.

Reliability tasks. – Although the reliability program to be prosecuted will not necessarily be limited to the following tasks, those described are considered to be the minimum necessary for adequate evaluation of the reliability of the design of the spacecraft and payload, and the subsequent determination of the probability of success in meeting mission objectives.

1. Reliability prediction – Analyses will be made of spacecraft and its subsystems, in order to assess their inherent reliability. Where the design proves to contain components whose reliability may compromise attainment of the mission reliability objectives, trade-off studies will be conducted to effect the best compromise between reliability improvement, and the weight and cost penalties associated with any changes. Such changes are anticipated to be minimal on the basis of the preliminary reliability investigations conducted during the preliminary design phase.

2. Parts selection and application – During the design phase, assistance will be afforded to design engineering in the selection of the best available parts, and their application in a properly derated fashion. Where special parts testing, or handling, such as power aging may be necessary, information will be provided to design engineering, upon the most effective methods of conducting such operations.

3. Failure modes & effects analysis – Reliability engineering will conduct at least one failure modes & effects analysis directed toward determining experiment success. The modes of failure will be identified, their causes will be explored, the effects of such causes upon the specific equipment will be analyzed, as well as the effects on other elements of the system, and finally the probabilities of the occurrence will be calculated. Recommendations will be made to design engineering as to the most effective ways to obviate any such failures or areas of critical weakness. These analyses will be updated until the time of design freeze, to assess the effects of any changes upon the overall reliability of the system. Experiment hazard levels are specified under Reliability Considerations of the section entitled System Description.

4. Design review – Reliability personnel will attend design reviews to present results of reliability activity, recommend changes, and assist in the overall assessment of the validity of the spacecraft/payload design.

5. Life safety analysis – Since preservation of the life of the primates is one of the major aims of the program, a study will be made to identify those equipment failures whose consequence may be early mortality of either or both of the animals. It is anticipated that this analysis will be run in conjunction with the Failure Modes & Effects Analysis, but differs from it in intention in that the possible failures of equipment are coded per a Criticality Listing. Code 1 is a failure which can cause demise of the animals; Code 2 is a failure, the occurrence of which may not cause immediate death but which will necessitate aborting the Mission due to the death of the animals being expected at a

near future date, and Code 3 denotes a failure or failures not causative of death, but which will either delay the completion of the Mission, or cause it to be terminated sooner than planned. Probabilities of any such occurrences will be computed and changes to the design will be made to minimize them insofar as possible.

6. Failure reporting and corrective action – Failure reporting and corrective action will be the joint responsibility of Reliability and Quality Assurance personnel assigned to the Program. The reporting vehicle is the Failed Equipment & Discrepancy Report (FEDR). This system of reporting is a closed loop in which the following actions take place:

- When a failure occurs, a FEDR form is initiated by Quality Assurance. The date, time and type of failure is reported on the form, together with details of the equipment or part failing, and its operating condition at the time of failure.

- The failure, or failed equipment is removed from the system, and delivered to design engineering for analysis. Reliability personnel assist design in an attempt to discover the cause of failure.

- Action is then recommended to repair, replace, or redesign the equipment in question. The form is completed documenting the details of all failures encountered during the course of the Program.

- Copies of the FEDR are circulated to all parties concerned for information/action.

7. Drawing and specification review – During the course of the Program, reliability engineers assigned will review and sign off all drawings and specifications for accuracy and adequacy of the reliability content. Recommendations will be made to engineering with respect to any changes believed necessary to improve the reliability content of these documents.

8. Reliability testing – Reliability will participate in any test planning activity conducted. The intent of such participation will be to recommend any special reliability testing as may be necessary to improve the final reliability of the hardware involved. Such testing may include step stress and overstress testing, as well as accelerated life simulation testing. As a matter of policy, the reliability input to the test program will be aimed toward obtaining the maximum amount of data useful to verify reliability for the minimum investment of special tests and test time.

9. Reliability reporting – Reliability reporting will provide those reports as may be contractually required for the program, and in addition any further reports which may be required by the Program Management. In general the reporting format will be structured using NASA NPC 250-1 as a guide, and will include but not be limited to the following:

- Prediction reports.

- Assessment and allocation reports.

- Reliability inputs to design reviews.
- Failure modes & life safety analysis reports.
- Inputs to trouble & failure reports (FEDR inputs).
- Reliability monthly progress reports.
- Final reliability report. (Program Summary)
- Any special reports required.

10. Schedule – At the outset of the Program, reliability engineering will furnish to the Program Office, a detailed schedule which time-orients the significant reliability activity. This schedule will form an input to the Master Program schedule, and set forth all reliability activity, and its time to accomplish each task from its inception to completion. Budgetary estimates will be provided also, in manhours to accomplish each task per the scheduled time spans.

Quality assurance plan outline: This Plan sets forth the Quality Assurance requirements for the program, which must be implemented to ensure that the materials, workmanship, and performance are to specified standards, and that the components have been manufactured to approved drawings and specifications. For the conduct of all of the activity described herein, NASA NPC 200-2,3. shall be used as a guide, to the extent that they apply.

1. Quality assurance policy – A Quality Assurance Program Plan shall be prepared delineating OPE program requirements and the methodology used for their implementation. This is accomplished by:

   - Use of company-wide QA policies and objectives.
   - Preparation of specific procedures for individual program requirements.

2. Quality assurance program management and controls – The following controls as a minimum will be effected:

   - Quality Planning, to detail all aspects of quality assurance activity.
   - Work Instructions to cover all aspects of quality activity and responsibility.
   - Adequate records pertaining to all quality activity during the program.
   - Failure analysis conducted on significant failures.
   - Production processes and workmanship standards prepared, or applied where already exist.
   - Corrective action taken in the event of any failure or discrepancy discovered throughout the conduct of the program.

3. Facilities and standards – Facilities and standards will be provided to the program. These will include but not be limited to the following:

- Drawings, Documentation & Changes: All drawings, and specifications will be reviewed prior to release, as will all changes and revisions.
- Measuring and Testing Equipment: All such equipment will be calibrated by the Measurement Standards Laboratory, and the calibration status will be maintained current by this laboratory.
- Customer Access to Inspection Equipment: Such access will be provided to gauges, measuring, and test devices, as required to verify product quality and Inspection accuracy.

4. Control of purchases – Suppliers who have been audited by Product Assurance will be used for the purchase requirements for this program. Such Suppliers are listed in the Directory of Approved Suppliers (DAS). Use of other suppliers than these will be investigated by and must be authorized by QA.

In control of Purchases the quality engineer assigned to the program is responsible to prepare Receiving Instructions for Inspection of all supplies and services procured. Included in his overall responsibilities are the following activities:

- Control of raw materials. To ensure that all such materials are inspected to verify their conformance with applicable specifications.
- Inspections and tests. To ensure that before acceptance of any article purchased, the supplier has performed sufficient inspection and testing to prove that his product has met all quality requirements as specified.
- Review of purchase orders. To ensure that they contain adequate and accurate quality assurance requirements imposed on the supplier.
- Nonconforming items. To ensure the proper control by vendors of any or all nonconforming items discovered in the receiving inspection of his shipment or shipments.

5. Manufacturing controls – In order that the quality of all items manufactured for use in the program is properly controlled, quality assurance will assure the following:

- That all incoming material is inspected and tested for complete conformance with specified requirements.
- That fabrication processes are properly inspected to assure that the produced hardware meets all quality standards and requirements.
- That all completed articles are inspected and tested to assure their final acceptance by the customer.

- That handling and storage of the final system and all of its associated elements is conducted in accordance with requirements.

- That nonconforming materials and hardware items are adequately reviewed, and disposition of them is made in the most expeditious and correct manner (Material Review Board, MRB).

- That all items are properly tagged or stamped with the indication of their inspection status.

All of the foregoing tasks will be fully detailed in the Quality Assurance program plan which will be prepared and released within 60 days after Phase II contract award. The plan will be subject to customer approval in accordance with the requirements of NPC 200-2.

Documentation Plan

This plan describes the techniques and disciplines to be employed during the Phase II Orbiting Primate Experiment Program to ensure timely supply of all contractually required documentation (data).

The primary objectives of the Lockheed OPE Program Documentation function, as further iterated in subsequent paragraphs of this plan are as follows:

- Ensure the generation of only that data essential to contract performance.

- Ensure the timely delivery of all contractually required data including that data generated by LMSC suppliers.

- Ensure continuous currency of all data.

- Ensure data quality.

- Provide a direct channel of communication between NASA Langley Research Center (LRC) OPE Program Office and the LMSC OPE Program Office for all matters pertaining to contractually required items of documentation.

Documentation management. – The disciplines of documentation management are implemented and enforced at LMSC through the establishment of a Documentation Management Officer (DMO), reporting to the OPE Program Manager. The LSMC DMO provides a counterpart working relationship with the NASA DMO and is the focal point for all data matters, both within the LMSC OPE Program and between LMSC and LRC.

The LMSC DMO is responsible for the performance of the following functions:

- Analysis and/or formulation of the Contract Data Requirements List (CDRL).

- Preparation of new Data Requirements Descriptions as required.
- Assignment and recording of responsibilities for generation of contractually required documentation.
- Scheduling of documentation preparation and delivery, and monitoring progress against those schedules.
- Issuance of documentation preparation bulletins and consultation with document preparing organizations.
- Supplier documentation management.
- Support to program reviews.
- Documentation change control.
- In-process and final review of documentation.
- Documentation delivery and distribution.
- Documentation accounting.

Preparing organization. – Organizations responsible for specific documents will submit them through the DMO for review and delivery. If any events occur that may jeopardize the scheduled delivery of a document, the DMO shall be immediately notified in order that necessary steps may be taken to ensure schedule recovery.

Documentation Review Board. – With the concurrence of the LMSC OPE Program Manager, the DMO may establish and chair a Documentation Review Board to assist in the execution of his duties. The Board will meet on an as-required basis and be composed of representatives from the following organizations as required:

- Each document generating organization
- OPE Procurement
- OPE Configuration Management
- OPE Contract Administration

Documentation requirements. – Documentation requirements are given in the following paragraphs.

Contract Data Requirements List: The OPE documentation requirements will be based upon the Contract Data Requirements List (CDRL), NASA Form 1106, or equivalent, furnished as a part of the Request for Proposal for the Phase II effort. The documentation items will be carefully examined and evaluated against program requirements and data indigenous to LMSC's normal operation. Changes to the CDRL will be recommended only when such evaluation indicates that an alternate requirement is justified in terms of economic savings or technical utility. Detailed justification will accompany any request for change to the CDRL. Additions to the CDRL will be proposed by LMSC where additional documentation is deemed essential to the conduct of the OPE Program.

Definition of Documentation: As used herein "Documentation" is defined as the technical, administrative, and financial records resulting from, or required for, the communication of concepts, plans, descriptions, requirements, and instructions related to the OPE Program. Specifications, standards, drawings, lists, plans, procedures, reports, and manuals are included in this definition. Documentation will be categorized as follows:

- Type I documentation shall be submitted to NASA for approval. Implementation of Type I documentation will not proceed until after approved by NASA or until 20 days after receipt by NASA/LRC, whichever is earlier. NASA approval is considered to be granted if LMSC has not received written notice of disapproval within 20 days after receipt of the document by NASA/LRC.

- Type II documentation will be submitted for coordination, surveillance, information review and/or management control.

- Type III documentation will be retained by LMSC and submitted to NASA only upon request.

Data Requirement Description (DRD): It is assumed that the OPE Program will utilize NASA Form 1107, or equivalent, for providing a unique description of the format and content of each item of documentation listed on the CDRL. The Contractor will prepare DRD's for recommended documentation items.

Source of requirements: The following primary documents will be analyzed and evaluated in the process of preparing recommended contract documentation items:

- Contract NAS 1-6972 Final Report - OPE, Phase 1.

- NPC 200-2, Quality Program Provisions for Space Systems Contractors.

- NPC 200-3, Inspection System Provisions for Suppliers of Space, Materials, Parts, Components, and Services.

- NPC 250-1, Reliability Program Provisions for Space Systems Contractors.

- NPC 500-1, Apollo Configuration Management Manual

- NPC 500-6, Apollo Documentation Administration Instruction

- NPC 500-8, Apollo Documentation Requirement Descriptions

Whenever feasible DRD's selected from the latter source will be utilized without change or with modifications as required to accommodate program requirements.

Documentation preparation ground rules: In the interest of economy, the following general rules will be observed by the LMSC OPE Program in the generation of contractually required documentation.

- Plans, Procedures & Reports – These documents will be prepared on 8 1/2 x 11 inch paper utilizing standard typewriters. Reproducible copies will be transparent Ozalid or other equivalent process. Non-reproducible copies will be of the type utilized internally to LMSC and are of sufficient legibility to satisfy their intended purposes.

  Revisions to documents over 30 pages in length are normally accomplished by the submission of revised pages and a letter of instructions for performing the updating. On periodic reports, if there is no change during the reporting period, a letter stating this fact is considered sufficient.

  Specifications, whose revision procedure will be covered by Configuration Management requirements are excepted from the above general rules.

- Drawings and Associated Lists – Non-reproducible copies of drawings will be of blueprint, blueline or white print type of sufficient legibility to serve their intended purpose. Reproducible copies of drawings will be furnished in the form of microfilm aperture cards of the type and quality used internally within LMSC, or of the transparent ozalid type whichever is specified by contract.

- Document Identification – Each document will have a unique identifying number assigned and statused in accordance with established LMSC procedure. In addition, each document will be identified to the applicable contract number and the LMSC R&D code identification number.

  Where data is applicable to more than one contract, it will be identified to the contract which controls the rights to such data. For example, data generated under Phase I of the OPE Program and subsequently utilized for the Phase II contract, need not be reidentified to the Phase II contract number.

- Due Dates – Documentation will be scheduled to allow receipt by the addressee not later than the due date specified on the CDRL. However, it is assumed that documents sent air mail and post marked on the due date is acceptable for contractual purposes.

Documentation management procedures. – In addition to the preparation of the CDRL and DRD's as described under "Documentation Requirements" herein, the following functions will be performed by or under the direction of the LMSC OPE DMO.

Documentation task assignments: All documentation items listed on the CDRL will be assigned to responsible document generating organizations and will be recorded for management information. Documentation assignments will normally be made in writing and will include the document identification, a copy of the corresponding DRD, reference to any other pertinent instruction, a document completion date, and in-process monitoring dates such as, "Draft available for review", "Ready for final approval", etc. Such assignments will be directed to the lowest definable organizational entity responsible for the document, and will be maintained current by the LMSC DMO to reflect program changes or redirections.

Master documentation schedule: A master documentation schedule will be prepared in conjunction with the documentation task assignment described above. This schedule will provide a composite view of the documentation workload phasing and will be a valuable tool for budget control and program monitoring. Each document will be identified and its due date defined, together with identifiable in-process monitoring point as required to provide early identification of problem areas in order that corrective management action may be taken before actual program impact occurs. The Master Documentation Schedule will be monitored and periodic or special reports will be made to the LMSC OPE Program Manager as required.

The above requirement must not be construed as requiring a separate scheduling and monitoring function under the LMSC DMO. In all cases, the same scheduling and reporting techniques will be used as are used for other program control activities.

Documentation preparation instructions: Established LMSC manuals such as the Drafting Practice Manual, the Specification Manual, etc., together with the DRD's described above provide basic instructions for the format and content of authorized documents. Certain departures from the standard requirements of these documents are often required or directed during the course of contract performance. A series of consecutively numbered "Documentation Management Bulletins" will be generated as required to disseminate such information to document preparing personnel. The format of these bulletins is flexible and they may be used to convey any appropriate subject matter pertaining to document preparation or associated administrative tasks.

Program review support: All program reviews, are concerned to varying degrees with documentation. The LMSC DMO, in conjunction with the LRC DMO, will establish those documents which will be required for each review. This is normally accomplished concurrently with establishment of the review agenda. The LMSC DMO assures that the agreed upon documentation is available in the proper status and quantity to support the review. In the case of formal documentation reviews, the LMSC DMO acts as LMSC co-chairman in the conduct of the review. The LMSC DMO additionally assures that any data changes directed or agreed to during program reviews are properly authorized, scheduled and monitored.

Documentation change control: The LMSC DMO, supported by the Documentation Review Board will be responsible for reviewing all changes to the OPE Program and its resulting design to assure that the impact of such changes upon the associated documentation is considered from a cost and schedule standpoint. Committments to the Configuration Control Board (CCB) regarding such impact are made by the LMSC DMO.

Documentation quality: All documents, except drawings and associated documentation receiving engineering check, and certain items of financial and administrative data will be reviewed by or in conjunction with the LMSC DMO prior to submittal to NASA/LRC. This review is conducted for the purpose of insuring proper format and content, identification and marking, as well as adherence to established standards of documentation quality and legibility. In addition to the final review, random in-process reviews will be conducted to prevent the necessity of any redocumentation requirements prior to delivery.

Document delivery and distribution: The LMSC DMO will assist NASA/LRC personnel in establishing standard distribution lists for each document included on the CDRL. The LMSC DMO will also prepare all document transmittal letters for signature by the OPE Program Contract Administrator. Upon notification from the LMSC DMO, the OPE Program Documentation Release Center will obtain the necessary copies of the document(s), confirm status, and assemble the package for shipment. The DMO will coordinate and prepare answers to all correspondence regarding documentation.

Documentation data accounting: To the extent possible, mechanized documentation release records, schedule monitoring inputs and documentation records will be employed to provide the LMSC DMO with current status of all OPE program documentation.

Data storage: The originals of all released documents will be stored in LMSC vaults. In addition a reproducible copy will be made and forwarded to a remote disaster storage location as a matter of company policy.

Supplier documentation management. — The disciplines and procedures described in the previous section are also imposed upon all suppliers generating documents in support of the OPE Program.

Supplier data requirements list (SDRL): The LMSC OPE Program DMO, in conjunction with the Documentation Review Board, will establish the documents required from each supplier on a Supplier Data Requirements List (Fig. 165) which is appended to and made a part of the subcontract or purchase order. Document content, format, and preparation instructions for each line item on the SDRL will be conveyed to the supplier by means of DRD's (Fig. 166) as described above. Prime contract DRD's will be modified as necessary to fit the peculiarities of the particular procurement involved. For example, a given supplier may be required only to furnish specific inputs to a document generated by LMSC for supply to NASA/LRC.

Supplier Liaison: The LMSC OPE Program DMO will maintain close contact through LMSC Procurement with all major suppliers to assure that documentation requirements are understood and that supplier-generated documentation is prepared in a timely manner.

Supplier documentation review and accounting: All supplier-generated documents will be reviewed and validated by the OPE Program Office prior to onward transmittal to NASA/LRC. Supplier documents will be reviewed by or under the direction of the LMSC DMO for form, content, and general contract compliance, and by the cognizant LMSC technical organization for technical adequacy. Supplier documents will be released into the LMSC documentation system for accounting and storage purposes.

Government and associate contractor documentation. — The LMSC OPE Program will define on DRD's that data covering GFE or associate contractor-furnished equipment which is essential to the LMSC portion of the OPE program. Such DRD's will be forwarded to NASA/LRC for action as necessary.

SUPPLIER DATA REQUIREMENTS LIST

PAGE OF

PROPOSAL REFERENCE OR SUPPLIER NAME:	SUBCONTRACT/P.O. NUMBER:	PRIME CONTRACT NUMBER:	APPLICABLE DCD/SCD:	ITEM DESCRIPTION:

ITEM NO.	TITLE	CLASS, TYPE, GROUP OR CATEGORY	AUTHORITY	FORM NUMBER	QTY. OF REPRO.	QTY. OF COPIES	REQUIRING OFFICE	FREQUENCY OF DELIVERY	PRELIMINARY DUE DATE	FINAL DUE DATE

PREPARED BY:	ORGN.	EXT.	DATE:	APPROVED BY:	ORGN.	EXT.	DATE:

FORM LMSC 2621

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION

Fig 165 Supplier Data Requirements List

DATA DESCRIPTION

TITLE	NUMBER	REV.
PURPOSE	DATE	
	PAGE OF	
	REFERENCED AUTHORITY REGULATION OR SPEC.	
	NUMBER	REV.
INTERRELATIONSHIP	COORDINATING DEPARTMENT	
PREPARATION INSTRUCTIONS		

FORM LMSC 2671

LOCKHEED AIRCRAFT CORPORATION

Fig 166 Data Description

Facility Requirements

The manufacture and test of hardware for the Orbiting Primate Experiment will require a broad range of capabilities, supported by an equally broad variety of facilities. A detailed listing of specific equipment requirements is premature at this time, and would necessarily include many items of conventional nature, capable, in part, of substitution through the use of alternate manufacturing and test approaches. The final selection of manufacturing is, for example, dependent upon the manufacturing specifications to be invoked. The following project facility requirements are therefore defined in terms of general manufacturing and test capabilities required.

Required fabrication facilities. — The following lists the fabrication facilities required for the OPE program.

Type of Shop	Capability Required
General Machine Shop and Precision Machine Shop	Machining of aluminum, stainless steel, and titanium.
Sheet Metal Fabrication	Aluminum and stainless steel roll forming, compound curvature components, spinning and precision sizing of circular mating edges to 7 ft diameter wire screen components.
Plastic Fabrication	Fiberglass lamination by vacuum bag and/or autocrave methods, use of washout plaster molds or equivalent method.
Multilayer Insulation Shop	Make up of multilayer insulation blankets of thin foils, techniques for fitting and assembly to structure, attachment, and thermal control at blanket penetrations.
Tube and Plumbing Shop	Precision forming of aluminum and stainless steel tubes. Precision sizing, trimming and flaring at tube ends. Orbital welding and/or brazing.
Tank Fabrication	Forming of hemispheres, and barrel sections, welding of high pressure tank assemblies.
Chemical Milling Facility	Thickness reduction of thinwall aluminum sheet metal components — both single and compound curvatures to 7 ft diameter.
Fabrication of Structural Frame Members	Roll and/or stretch forming of extruded shapes and tubes. Hot forming of titanium tubes.

Type of Shop	Capability Required
Structures Welding	Automatic and semi-automatic welding of thin aluminum shell structures with circular and linear seams. Manual welding of fittings, bracketry, and extrusions. Welding of complex tubular structures of titanium in inert gas atmosphere.
Electronic Systems, Cable and Harness	Advanced techniques for systems, fabrication assembly and checkout.
Cleaning, chemical processing, and surface finishing	Cleaning of oxygen systems. Sterilization of water systems. Cleaning in preparation for processing. Chemical processes for surface protection of aluminum and stainless steel. Application of protective and thermal control coatings.

LMSC Fabrication facilities. — Through previous participation in many missile, spacecraft, and Biotechnology programs, LMSC has developed a broad facility base which provides ample capability to support Phase II of the Orbiting Primate Experiment. With minor exceptions, all facilities which may be required for this program are located within the central LMSC complex of buildings at Sunnyvale, California, are readily accessible to each other and can be made available for the Phase II program.

The program schedule, with initial fabrication of components beginning in the fourth month, allows a reasonable lead time for negotiation of facility commitments at the time of Phase II proposal preparation.

Use of some Government-owned equipment is anticipated for Phase II manufacture. Existing facility contracts administered by LMSC at Sunnyvale are permissive of rent-free use of government owned equipment on a non-interference basis. The offset competitive advantage for rent free use of this equipment will be defined in detail in both the facilities plan and the cost proposal of Phase II RFP response.

LMSC's policy is that, wherever possible, R & D component fabrication will be accomplished in the existing central manufacturing facilities. Other facilities, established for support of the standard Agena, Missiles Systems, and certain R & D Division projects, are also available for certain types of work on a non-interference basis. In addition, a number of special-purpose development laboratories and shops in the R & D Division may be utilized to provide unique capabilities when needed.

The central LMSC manufacturing facility is located primarily in building 103. Within this area are included the general machine shop, the precision machine shop, sheet metal fabrication, the tube and plumbing facility, tanic fabrication, structural frame fabrication, structures welding, cleaning, chemical processing and surface finishing. All facilities are more than adequate to meet the project requirements as presently defined. It will be necessary to perform weld procedure development for the titanium tube structure welding, as the joints indicated are peculiar to this project.

The plastic fabrication shop is in Building 126, adjacent to the main shops, and is fully equipped to produce the fiberglass components required. Solar cell panel assembly and multilayer insulation shops are located in Building 150, which also houses the manufacturing research laboratories. The electronics manufacturing facility in Building 151, includes a number of related shops with complete capability for aerospace electronics systems manufacture. A large chemical milling facility, with 20 ft diameter capability is being established in Building 159 and is scheduled to be in operation several months prior to the first requirements of this program.

Assembly requirements. – A project-controlled assembly area is considered necessary to accomplish the general types of associated assembly tasks. This area must be adequately equipped to support the following activities:

Mockup Assembly

Development Hardware Assembly

Served, Bonded Storage

Equipment Checkout and Calibration

Experiments Environmental Support

Bench Sub-assembly

Final Assembly and Installation for

- Modification of LMSC primate cage for zero-g flight test
- Payload lab test model
- Qualification unit
- Flight unit
- Flight back up unit

Systems Integration

Functional Test and Data Acquisition

The facility should be air conditioned, with a minimum ceiling height of 18 ft. Overhead or portable hoist equipment will be required.

LMSC assembly facilities. – A project assembly area of at least 3,000 square ft will be allocated for this project. It will be within, or in close proximity to, the Biotechnology area in Building 151. All necessary assembly support equipment is available.

Committment of the assembly area will be made in response to the Phase II RFP, and preparation will begin at contract go-ahead. Activation of the area is anticipated in the second month of the contract.

Test facility requirements. – In addition to the experimental laboratory area planned for the OPE Program in Building 151, major test facilities and equipment are concentrated in the Development Test Laboratory, Building 102. This complex includes over 60,000 square feet of especially designed and constructed laboratory with extensive LMSC and Government-owned test equipment capable of simulating a full spectrum of natural and induced environments associated with hardware handling, launch and orbital flight conditions. This capability is complemented by a complete range of instrumentation, data acquisition, and data analysis equipment.

In selecting the facilities and equipment for performance of the Integrated Test Program emphasis shall be placed upon location, capability, capacity, schedule performance and program cost. All equipment and facilities utilized shall conform to the requirements of applicable NASA specifications including Section 9, NPC 200-2, "Quality Program Provisions For Space System Contractors".

The following listing covers tests to be conducted and general facility requirements to accomplish these tests.

Type of Test	Capability Required
Vibration (All axes)	Unit to handle up to 5,500 lb article over vibration range as follows: Sine 20 Hz @ 0.8 g 1000 Hz @ 16.5 g Random $0.2\ g^2$/cps @ 2 Hz $4.5\ g^2$/cps 200 to 600 Hz $2.25\ g^2$/cps @ 1000 Hz
Acoustic	Sound level to approximately 150 db with input of 100 to 500 cps
Static Load	Reaction support structure with multiple load and data channels
Separation	Handle 5,500 lb article through simulated separation from Apollo docking collar.
Transportation and Flight Shock	Sufficient to meet MIL-STD-810 on articles up to approximately 100 lb mass.
Acceleration	Drive 5,500 lb article at up to 4.5 g each axis

Type of Test	Capability Required
EMI & Anechoic	Sufficient to meet requirements of MIL-STD-826
Antenna Pattern	Sufficient range and electronic recording equipment.
Thermal	
Ascent	Transient thermal test of components/subassemblies. Source temperature up to 320 F. Vacuum to 10^{-4} Torr.
Orbital	Accept entire vehicle. Provide infrared simulation of absorbed heat rates. Vacuum level at least 10^{-4} Torr.
Pyrotechnic Shock	Simulate firing of pyrotechnic devices in vicinity of critical components where applicable.
Toxic Material	Test outgassing of selected materials under spacecabin environmental conditions.

In summary, LMSC test facilities are adequate to conduct all subsystem and/or component testing. Facilities are limited in capability required to test large heavy vehicles in the areas of dynamic vibration, acceleration and separation. Sufficient upgrading of these capabilities is not expected until a proposed large Vehicle Test Complex is constructed. Current plan is to break ground in June of 1968 with plans for completion by the end of 1969. Therefore, certain portions of the large vehicle testing will be conducted outside LMSC, based upon existing program schedules. Wylie Laboratories and/or United Test Laboratories can satisfy requirements not met by LMSC.

Advanced Technology and Advanced Development

Identification of those aspects of the OPE Program which require development of technology beyond the present state of the art are limited to the Payload portion of the experiment. In the present concept of the experiment system and subsystems as described in the data package, only the preservation fluid appears to fall within the category of advanced technology. Based upon the current status in the laboratory, it is estimated that the preservative fluid will have been proven in ample time for its possible use in the OPE experiment.

Several other areas of the Payload however can be deemed to be in the advance development stage. None of these would appear to warrant their segregation from the proposed development program or to require separate program or funding plans.

Included in the identification of advance development are the following items of the presently conceived OPE Payload subsystems:

Special Equipment Subsystem

- Mass measurement device

Life Cell

- Waste management subsystem
- Animal retrieval subsystem

Data Management System

- Implanted TLM system
- Implanted activity counter

Items indicated under the Data Management System are assumed to be GFE. The other items listed are covered within the development plan presented herein.

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CONCLUSIONS AND RECOMMENDATIONS

The results of this preliminary design study show that the Extended Weightlessness Experiment is feasible within the constraints imposed by the Saturn-Apollo program. The optimum carrier vehicle was found to be the Saturn-Apollo flight 216 scheduled as part of the AAP Cluster B. Launch date was established as mid-1970. Flight qualified hardware to meet the experimental objectives can be completed in time to meet the 1970 flight date providing the program described herein is initiated by January 1968.

The Orbiting Primate Spacecraft utilizes a maximum of available technology. For example, well-proven solar cell power and cold-gas attitude control concepts are used. Life support is provided in the simplest possible manner using high-pressure gas storage, lithium hydroxide for CO_2 removal, and other approaches proven by previous LMSC experience to be highly reliable. Only one area, animal preservation, was identified as a new technology requirement. Data management can be handled by minimum modification to Apollo/LM airborne electronics and to existing Saturn/Apollo communications network facilities. Significant development efforts, but no new technology, are required in several areas, particularly in bioinstrumentation, waste management, and retrieval. The design of the payload allows maximum flexibility for incorporation of design improvements as they evolve in the early hardware development phase of the program.

Fabrication and qualification of the spacecraft and payload can be accomplished with existing LMSC facilities in the time span required for a mid-1970 flight with the exception of full-scale acceleration testing of the integrated spacecraft/payload combination; the latter tests can be performed by suitable subcontract arrangement with one of several qualified outside test laboratories.

A preliminary reliability assessment shows the overall system reliability to be 0.942. However, the reliability of the airborne data management subsystem itself is assessed at 0.9497. With this subsystem out of the calculations, the overall system reliability increases to 0.99189. The most critical items of the data management subsystem appear to be the S-band transceiver and the power amplifier. It is therefore recommended that during final design of the data management subsystem, redundancy of these items be carefully reconsidered to improve the overall reliability.

As shown in the development plan, it is recommended that early flight tests on a zero-g aircraft be conducted to verify proper zero-g functioning of both the waste management and retrieval concepts described herein. This work can be accomplished by modifying "boiler-plate" hardware presently in existence at LMSC.

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SUPPLEMENT

ALTERNATE MISSION MODES AND EXPERIMENTS

In response to Modification 3 to the basic contract, alternate mission modes and payload configurations were studied to investigate the adaptability of the OPE hardware to perform a number of primate experiments other than the basic one involving two unrestrained animals. This investigation was carried to sufficient depth to allow an assessment of the impact of such experiments on the spacecraft, primarily in terms of weight, volume, power, thermal control and data management. The objective of this study was to assess the versatility of the basic design when applied to other potentially valuable experiments. The material presented in this section is not intended as a recommendation or evaluation of these other experiments. The various approaches explored are discussed in the following sections.

Prime Experiment

As described in foregoing sections of this report, the prime experiment consists of sustaining two unrestrained, 13-pound Rhesus monkeys in orbital flight for up to one year to determine the effects of extended weightlessness. For this purpose, the orbiting vehicle was developed as an independent, self-sustaining spacecraft. This section describes alternate mission modes for the prime experiment.

Mission modes. – Consideration of the Statement of Work led to three reasonable alternate mission modes:

- A slightly modified version of the independent spacecraft which is docked to the MDA of the OWS;
- a major reconfiguration of the independent spacecraft into modules which are appended mainly to the exterior of the AM of the OWS
- a similar reconfiguration of the independent spacecraft into modules which are installed internally in the OWS. Alternate mission mode characteristics are summarized in Table 87.

The S-IVB OWS is a conversion of a spent S-IVB stage into a habitable space structure for use by man in the performance of mission activities. The configuration of the OWS is undergoing considerable evolution, but, in general, an open grid deck, partitions, and some of the overhead are pre-installed in the liquid hydrogen tank of the S-IVB stage. An airlock module (AM) is installed within the Spacecraft-LM Adapter (SLA) in the space that the LM descent stage normally occupies. The function of the AM is to provide shirtsleeve conditions for IVA to and from the OWS and to other modules, and also to maintain pressurized integrity while permitting EVA. In addition, the AM is fitted with necessary equipment to integrate and service the entire cluster of modules. Adjacent to the AM is the MDA which has five docking ports permitting several vehicles or modules to cluster together simultaneously.

TABLE 87

CHARACTERISTICS OF ALTERNATE MISSION MODES

Item	Mission Mode		
	Docked	In AM/MDA	In OWS
Permits visitation by astronauts during the experiment	Yes	Yes	Yes
Permits total mass to be launched on more than one vehicle	Yes	Yes	Yes
Cluster provides meteoroid protection	Some	Some	Yes
Cluster provides thermal protection	*	*	*
Cluster provides electrical power	*	*	*
Cluster provides attitude control	Yes	Yes	Yes
Cluster provides data management	No	No	No
Permits adjustments and/or repairs to be effected	No	Some	Yes
Permits staggered retrieval sequence	Yes	Yes	Yes
Permits resupply of expendables	Yes	Yes	Yes
Reduces rendezvous problem	Yes	Yes	Yes
Reduces retrieval EVA problems	Some	Yes	Yes
Reduces structural requirement (launch truss, etc.)	No	Yes	Yes
Reduces separation requirements	No	Yes	Yes
Requires less overall mass to be launched	Yes	Yes	Yes
Reduces animal handling	No	No	No
Requires less man-hr to initiate experiment	No	No	No
Provides 0.001 g environment	†	†	†

†May exceed basic limit. Requires further analysis.
*Existing cluster system must be augmented to provide service.

This combination of modules is placed in orbit by a Saturn S-1B vehicle. After the S-IVB stage has completed its propulsion functions, the excess propellants are expelled and the OWS is then passivated, or made ready for habitation. Solar arrays are deployed from external pods on the S-IVB surface. A meteoroid bumper is deployed from the outer skin of the S-IVB stage. A quick opening hatch is removed from the LH_2 tank dome. The LH_2 tank is pressurized. Equipment is then deployed internally to complete the habitability process.

The Orbiting Primate Experiment spacecraft provides the following services to the payload:

Meteoroid Protection
Thermal Protection
Electrical Power
Attitude Control
Data Management

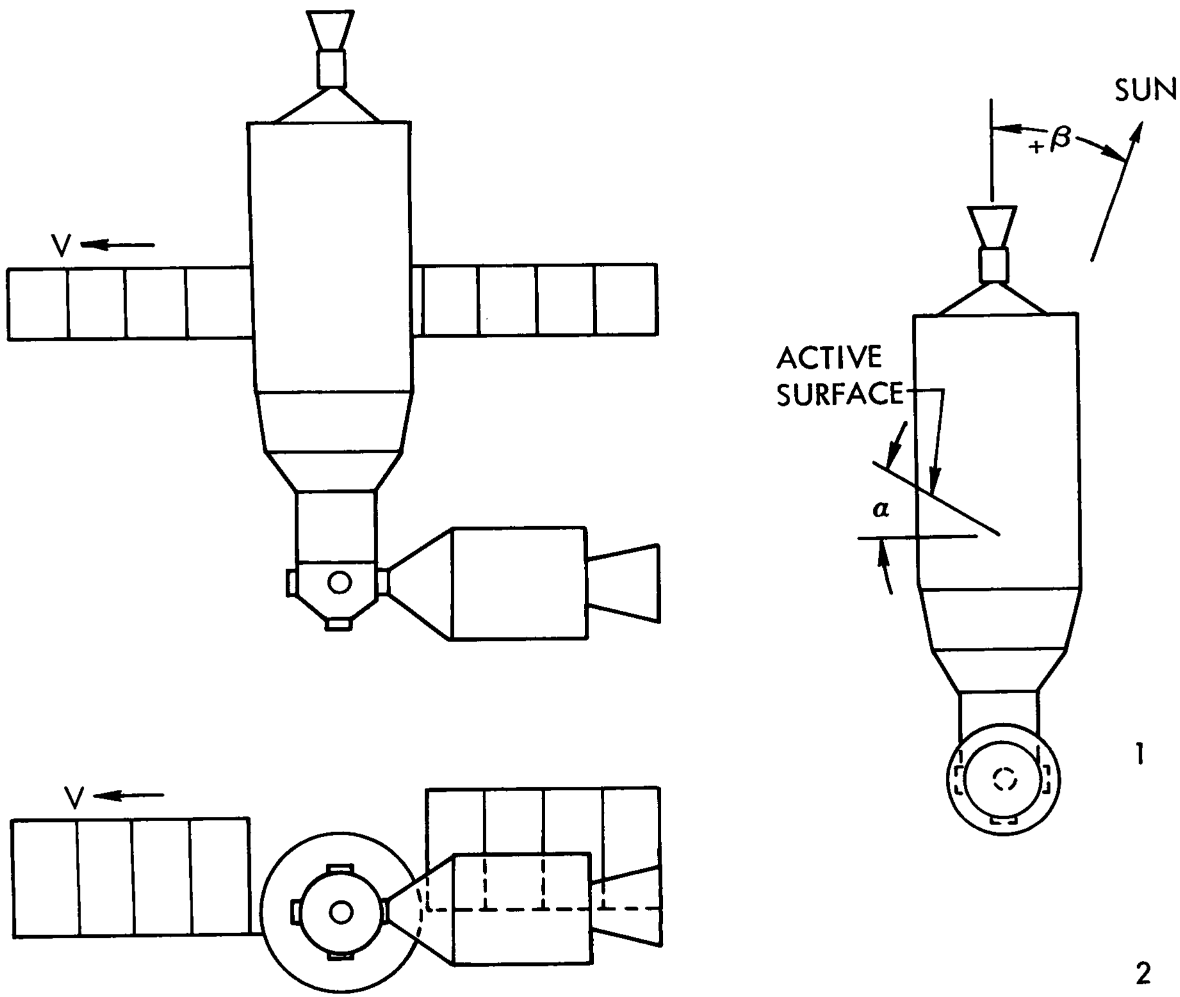
These services are discussed in pages 243 to 318. When the independent spacecraft is reconfigured into the alternate mission modes, these services undergo considerable change and in some cases are provided by the cluster.

A meteoroid shield is deployed from the OWS outer surface thereby effectively deleting the need for individual meteoroid protection for modules contained within. Modules clustered about the AM or MDA enjoy a 50 percent reduction of their view factor of space, thus reducing the amount of meteoroid protection required by 36 percent.

All alternate modes require space radiator services from the OWS since they are either inside or their radiator is rendered useless by view obstruction. The OWS radiator is sized to reject up to 10,000 Btu/hr. The weight of the radiator and coolant systems is 660 lb. Thus, any additional demands for heat rejection must be accomplished by augmenting this system at the rate of 0.066 lb/(Btu/hr). A thermal blanket surrounding the AM area provides protection from incident sunlight.

The OWS electrical power system consists of a 1047 ft^2 solar array which has limited tracking capability, 8 - 27 amp-hr secondary batteries, and when the CSM is present, 3 fuel cells and 3 - 40 amp-hr batteries. When the CSM is absent, the electrical power system produces anywhere from 1880 to 3475 w, depending upon the season and tracking angle, which is assumed to be automatic (see Fig. 167). Since the OPE is using power continuously for a year, even during dormant unmanned periods when no CSM is present, the 1880-w output of the solar array/secondary battery system must be used to derive the performance factor for possible system uprating. Assuming 1.5 lb/ft^2 as a reasonable solar array specific weight, the penalty for augmenting the array is:

$$\frac{1047 \text{ ft}^2 \text{ solar array}}{1880 \text{ w}} \times 1.5 \frac{\text{lb}}{\text{ft}^2 \text{ solar array}} = 0.836 \frac{\text{lb}}{\text{w}}$$



β	α	POWER (W)
−60	40	3475
−30	40	2425
0	0	1880
+30	40	2425
+60	40	3475

1. CONTINUOUS BUS POWER CAPABILITY OF SOLAR ARRAY/SECONDARY BATTERY SYSTEM AT 260 N.M., 25°C TEMPERATURE, AND NO SHADOWING.
2. TOTAL ACTIVE ARRAY AREA; 1047 FT^2
3. VEHICLE GRAVITY GRADIENT STABILIZED, SIVB ENGINE AWAY FROM EARTH

Fig. 167 Mission A Configuration/Flight Plan/Bus Power

In addition to augmenting the solar array, a secondary battery (177 lb) is required which is identical in all respects to the installation in the independent spacecraft. The overall factor for the electrical power installation then becomes:

$$0.836 + \frac{177}{400} = 1.28 \text{ lb/w}$$

Nonautomatic tracking solar arrays would be heavier.

The OWS is maintained in stable attitude for 7.5 hr after launch by the S-IVB stage auxiliary propulsion system. An additional system provides rate stabilization for docking of various modules and to assist and maintain gravity gradient capture.

The OWS flies in a low inclination orbit at 260 x 260 nm altitude. It is gravity gradient stabilized with the J-2 engine away from Earth. The solar arrays are extended in the orbit plane with the sensitive surface normal to the orbit plane. As various modules dock and undock, the attitude of the cluster shifts in response to the center of gravity movement. A gross change of attitude occurs when the LM/Apollo Telescope Mount and men are present. At this time, the entire cluster is sun-oriented during the daylight portion of each orbit, and gravity gradient stabilized during each night pass. The torque to accomplish this comes from the RCS of both LM and the CSM. Studies have been conducted which propose a control moment gyro momentum sink which will provide this torque and then be desaturated at night by maneuvering against the gravity gradient field.

The OPE in any of its alternate forms will make a negligible change in the mass properties of the cluster; therefore, no attitude control system augmentation is required.

The OWS data managment system is VHF with Unified S-Band (USB) when men are present. This function is provided by the CSM DMS. The OPE requires USB continuously. Therefore, the entire USB system must be carried on alternate mission modes to cover periods when the CSM is absent. Dual antenna requirements, however, can be dropped because of the Earth-orientation of the cluster. The only difficulty occurs in the time sharing of the MSFN by both OPE and OWS.

Docked configuration.— The docked OPE is shown in Figs. 168, and 169. The following variations from the independent spacecraft are made:

- A male docking fitting and support structure is added.
- The attitude control system is removed.
- The solar arrays are removed.
- The space radiator is removed.
- One-half of the water supply is removed.
- One-half of the water reclamation storage capability is removed, and a heater and insulation added to the remaining tanks.
- The X-band system is removed.
- One-half of the S-band and UHF antennas are removed.
- The OWS solar array is augmented.
- The OWS space radiator is augmented.
- The retrieval canister is augmented to permit usage as an ascent animal transporter.
- The outer skin thickness is reduced from 0.025 in. to 0.016 in.

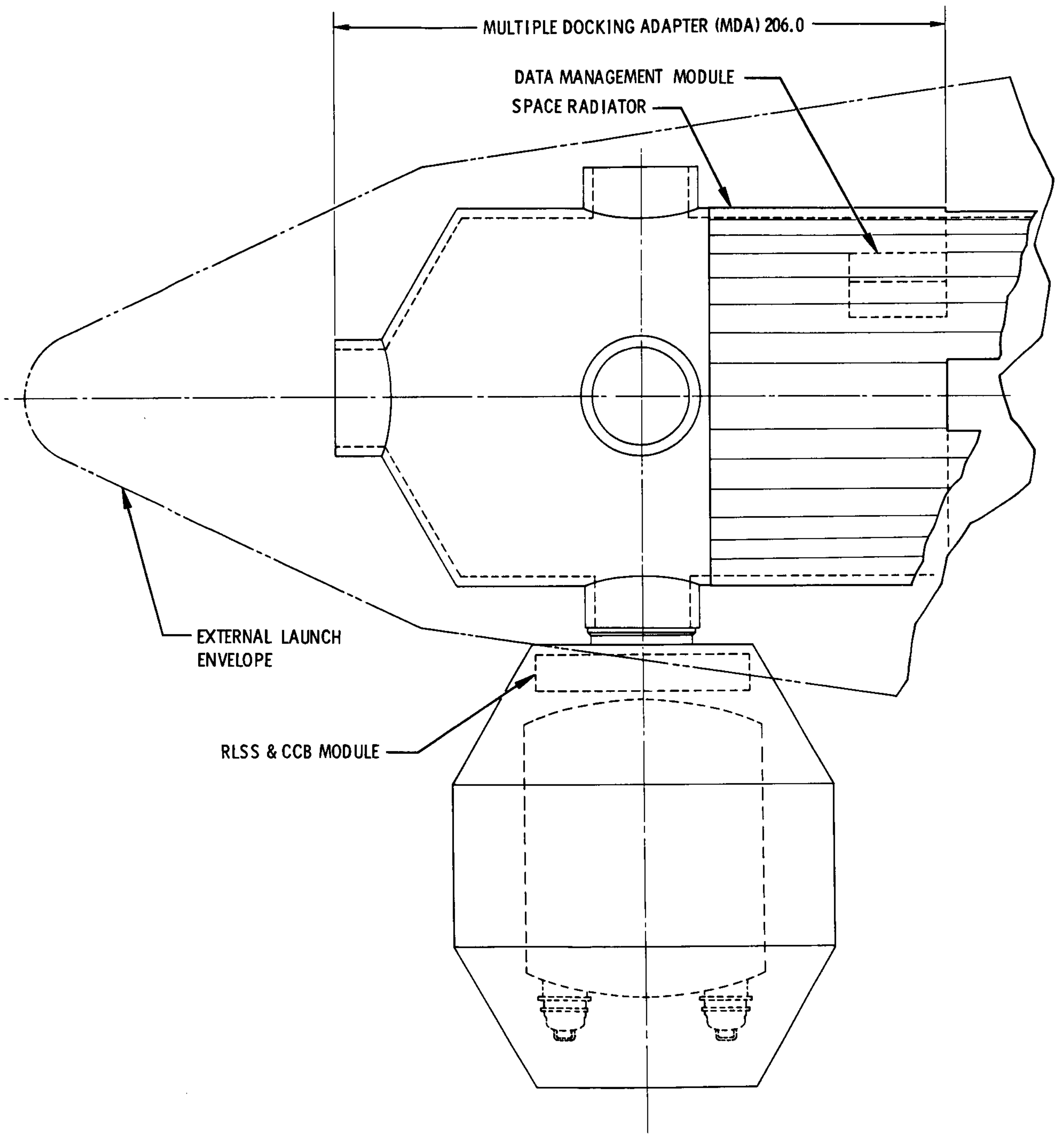


Fig. 168 Prime and Additional Experiments Docked to MDA

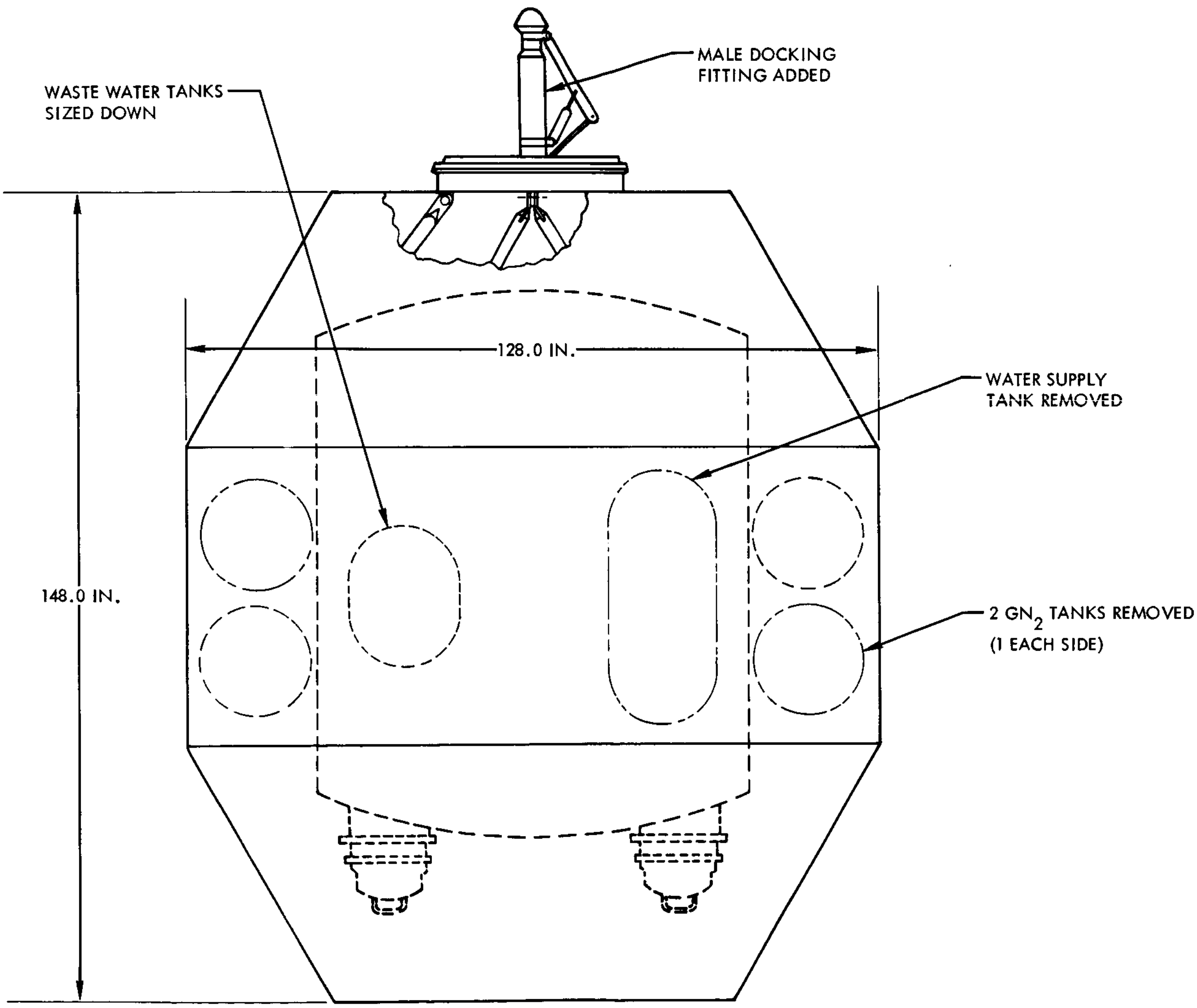


Fig. 169 Prime Experiment S/C for Docked Mode

AM/MDA configuration: This alternate involves a major reconfiguration of the OPE. The logical way to modularize is to make physical separations where a minimum of interfaces occur. At the same time, the weight and volume of the modules should be reduced to easily handled units. A further restriction is imposed by any peculiar payload envelopes or passageway constrictions. Although this alternate does not involve passage through the MDA/AM/OWS hatches, it was arbitrarily decided that in the interest of flexibility the major modules would be sized to pass through the 40.30 in. diameter quick opening hatch in the dome of the OWS LH_2 tank. These modules would then be valid for the OWS-installed alternate and represent a more flexible building block approach to the problem.

A study of the lifecell module and Thermal and Atmosphere Control systems (Figs. 64 and 65) led to a subdivision as shown in Fig. 170. The animals are placed in separate lifecell modules as shown in Fig. 171, and the distribution of Thermal and Atmosphere Control system elements is as indicated. Each life cell module contains the cage assembly, behavioral panel, exercise unit retrieval system, purge and high-flow fans, lights and TV, waste management system, and other elements shown in Fig. 64. Certain features of commonality such as the enclosure and waste management headers, and the combined high-flow return ducting low-friction effect required special treatment. For instance, the headers are now incorporated in the Environmental Control System (ECS) module, and the high-flow return ducts in each lifecell module were doubled that of before. The module is 38.75 in. in diameter by 124 in. long and will pass through the 40.30 diameter OWS hatch as well as the AM hatches. In order to obtain the 38.75 in. outer diameter the cage diameter decreased from 34 to 30 in.

The ECS module is 28.5 in. in diameter by 86 in. long as shown in Fig. 172, and contains the elements as shown in Fig. 170. A longitudinal bulkhead equally dividing the LiOH, charcoal, and catalyst canisters maintains the redundant process loops. The bulkhead is perforated at both end transitions to permit either side to process the atmosphere of both life cell modules. Each low-flow fan assembly has two outlets which are routed to the separate life cell modules.

The feeder module described in Fig. 74 is repackaged into a pressurized container shown in Fig. 173, which is 15 in. diameter by 51 in. long.

Other modules which include gaseous oxygen and nitrogen, liquid preservative and drinking water, inert reclaimed water storage tanks and data management system components, possess no unique packaging problems. They can be subdivided and reshaped to accommodate almost any carrier vehicle configuration. One point to be noted is the requirement to segregate reclaimed water until it has been Earth-laboratory examined for potability.

The modules are clustered about the exterior of the 65.12-in.-diameter portion of the AM and in the number one and two trusses, as shown in Fig. 174. The lifecell modules are arranged so that the animal transportation canisters penetrate the floor of the structural transition section of the AM. This permits animal handling in a shirt-sleeve environment. Thermal control and electrical power are obtained by augmenting the OWS systems. The lifecell and ECS modules are necessarily prebuilt on the AM, but all other modules could be orbitally assembled. This version foregoes the possibility of an animal social contact window. Two alternate versions which retain this feature were phantomed in section AA of Fig. 174. One version places both monkeys in a single lifecell module with internal retrieval canisters. The other version utilizes the single animal lifecell described above, but requires the retrieval canister to be outside.

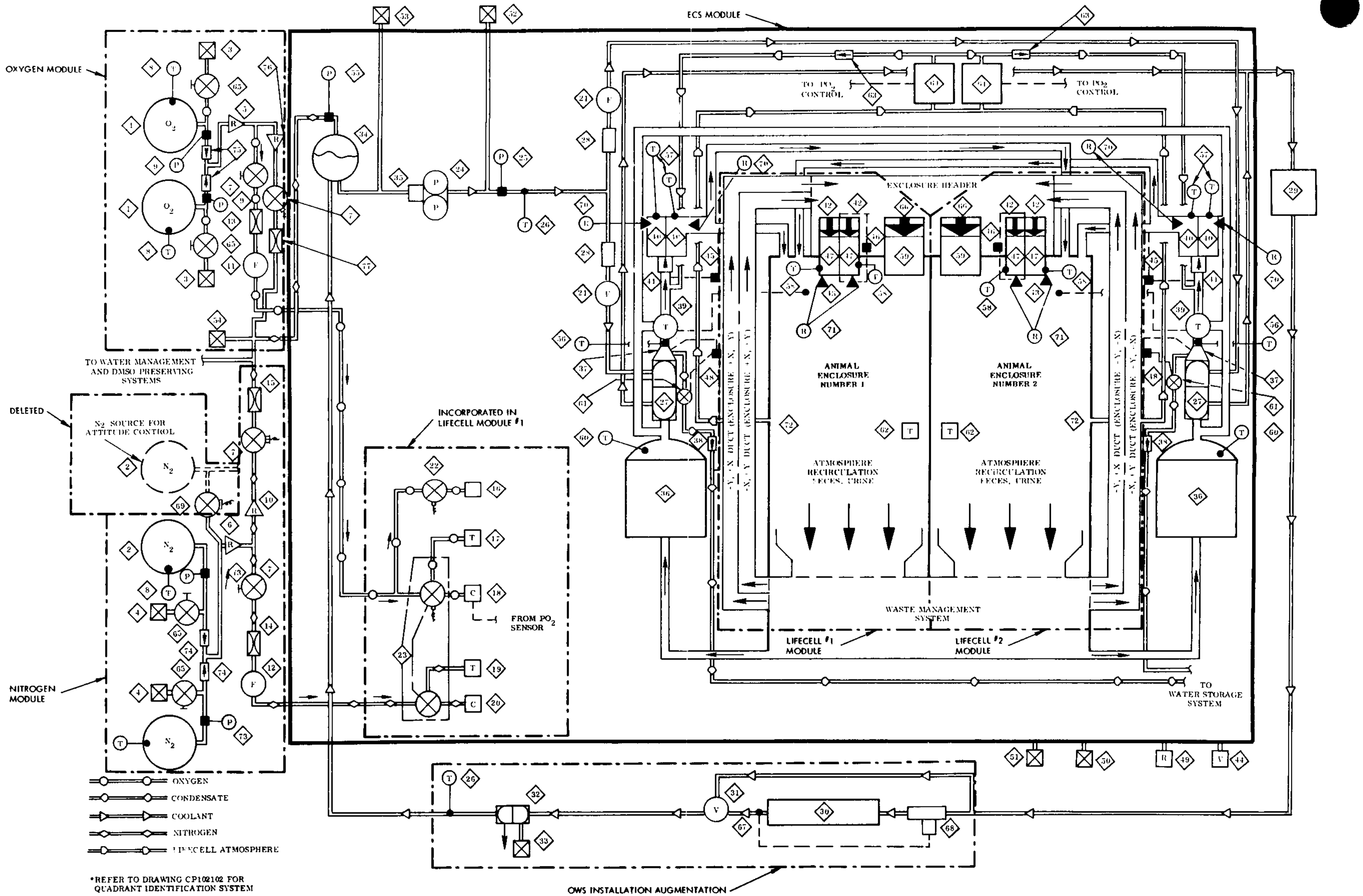


Fig. 170 Life Cell and ECS Modularization

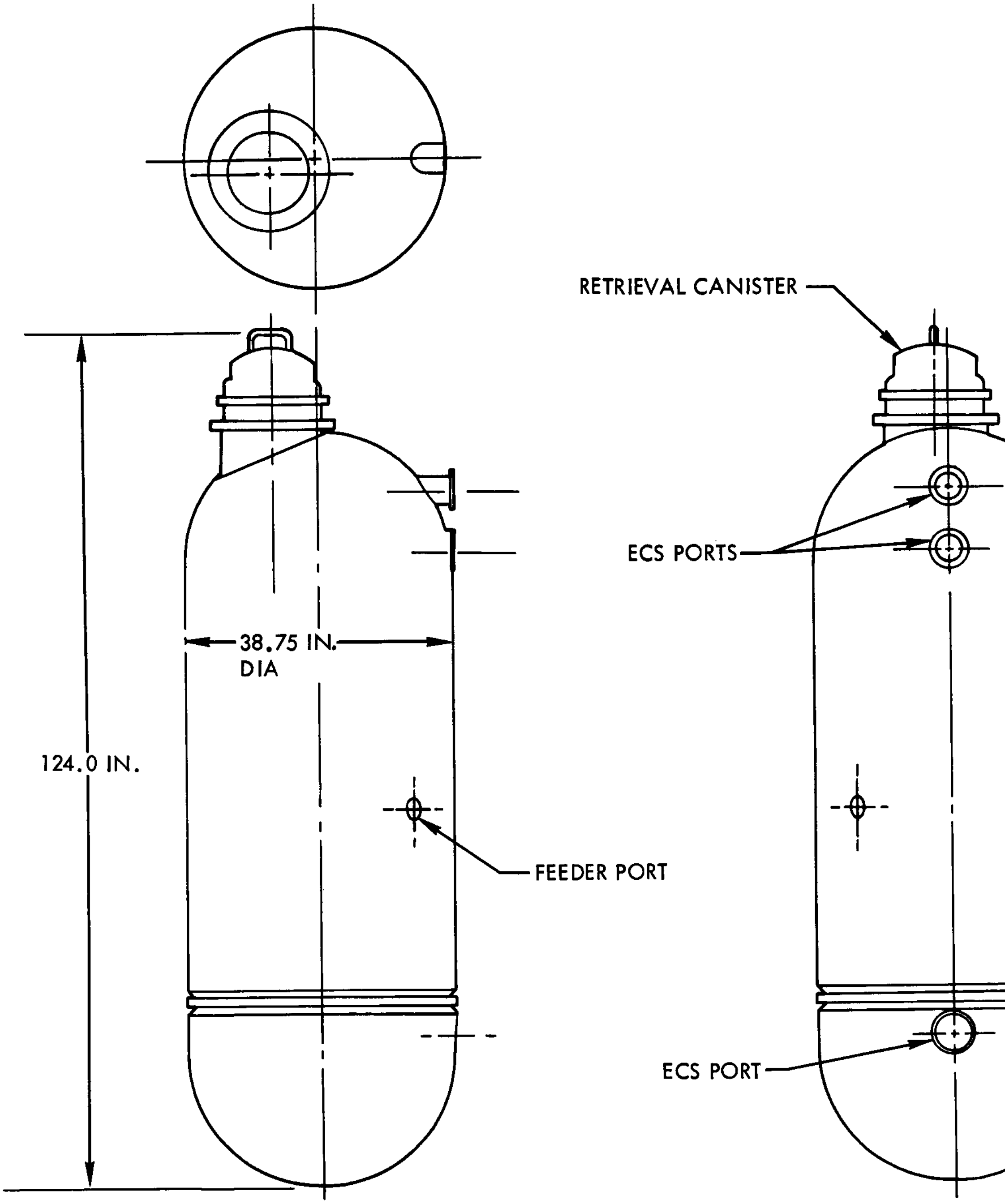


Fig. 171 Life Cell Module for 13-lb Unrestrained Rhesus

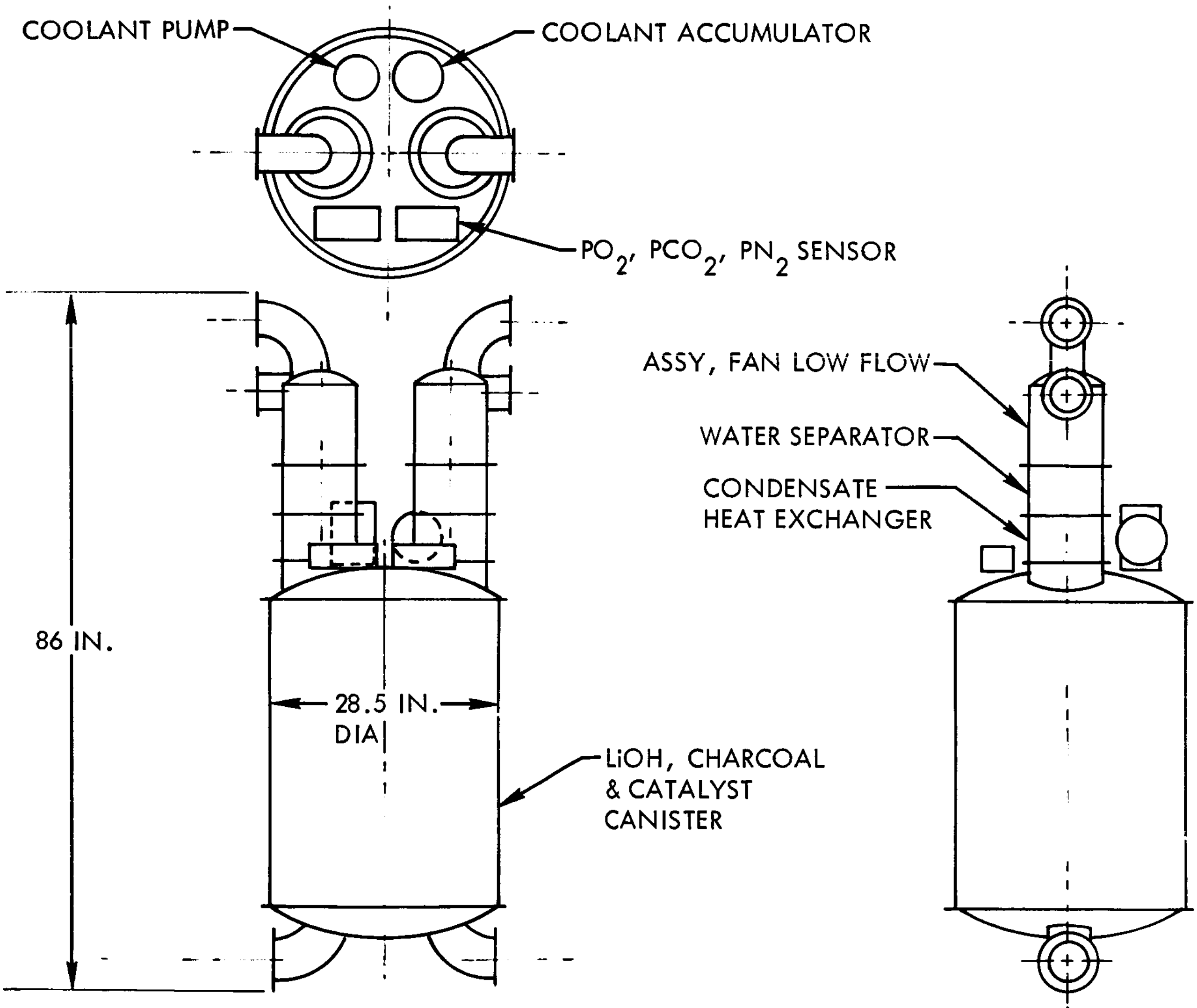


Fig. 172 ECS Module for 13-lb Unrestrained Rhesus

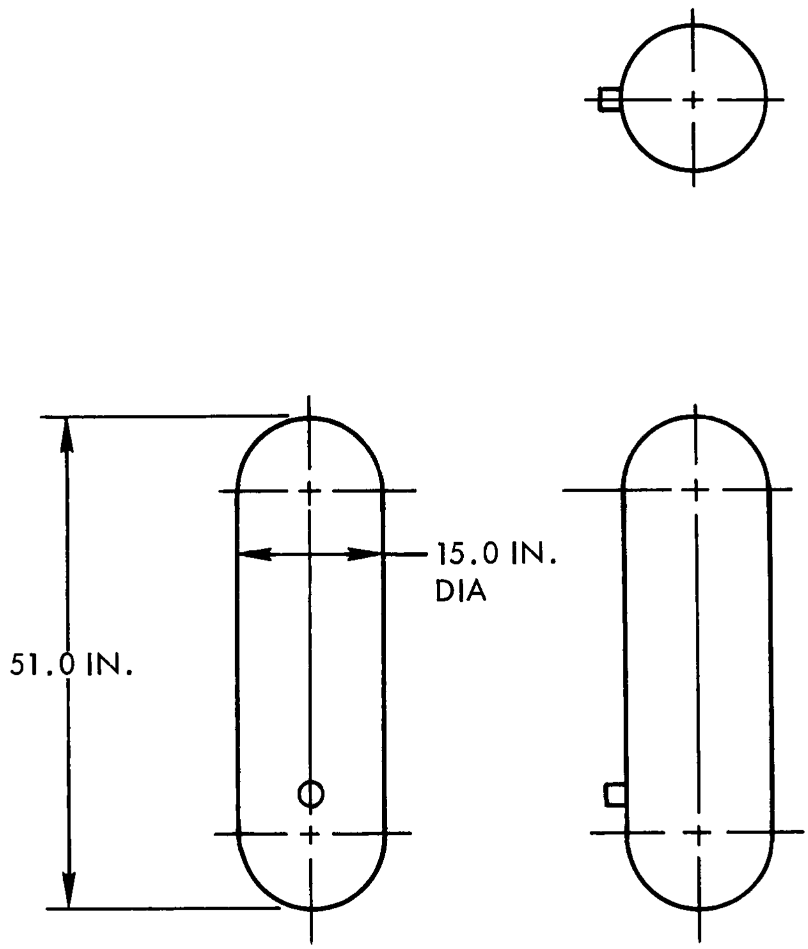


Fig. 173 Feeder Module

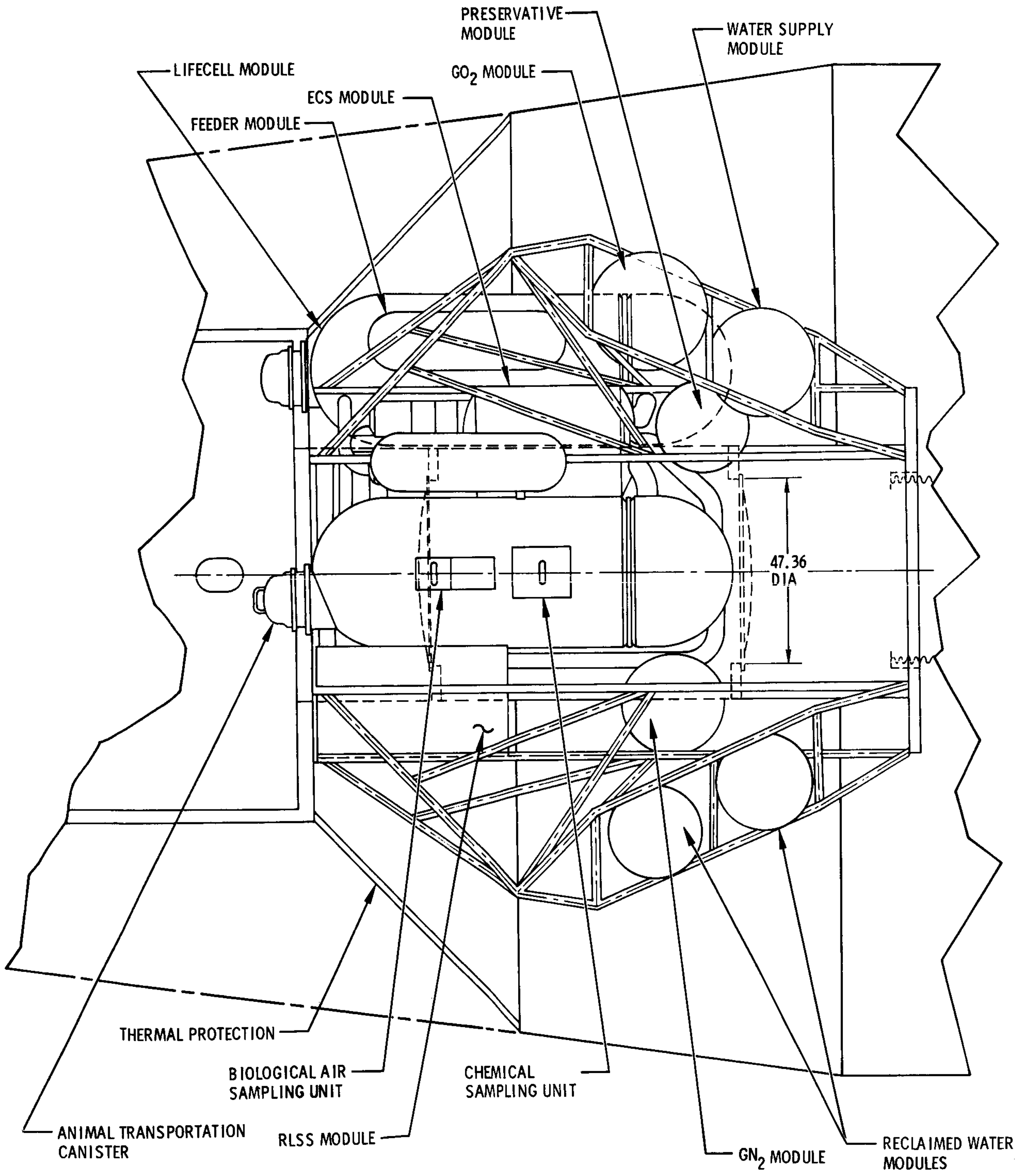


Fig. 174a Prime and Additional Experiments Installed in AM/MDA

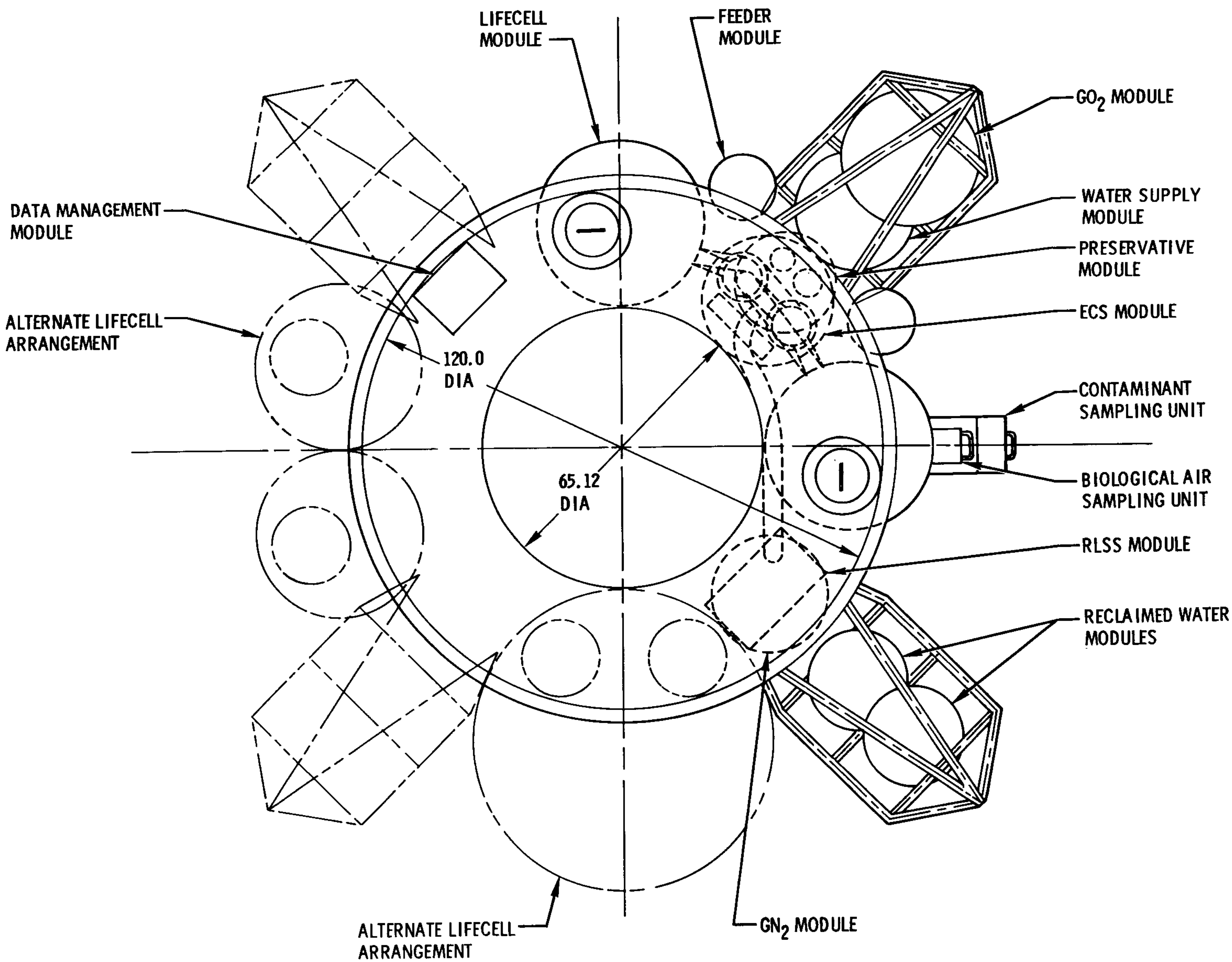


Fig. 174b Prime and Additional Experiments Installed in AM/MDA (Cont.)

OWS configuration. – The same modules described for the AM/MDA version are used in the OWS as shown in Fig. 175. The combined arrangement is emplaced on top of the grid deck which forms the overhead of the waste management and food preparation compartments of the OWS. Since the OWS atmosphere is at 5 psia, a need still exists for pressure-tight modules. Thermal control and electrical power are again obtained by augmentation of the appropriate OWS systems.

Operations. – In the docked mode, the OPE is launched on a manned launch vehicle in a dormant condition with no animals installed. The animals are contained in the augmented retrieval canisters which could be included in the same launch.

After the OWS is injected into orbit, the CSM/OPE effects rendezvous and docks the OPE to a port on the MDA. The CSM then detaches from the OPE and docks to another MDA port. The astronauts activate all the OPE systems. When the OPE is ready, the animals are inserted into the lifecell.

After 90 days, a sample of reclaimed waste water is returned to Earth for examination* and the reclamation process output is shunted to a different tank. At the end of 180 days, a decision must be made to either use the reclaimed water from the first tank or resupply with new water. If the decision is to use reclaimed water from the first tank, then a sample of the second tank is returned to Earth for examination and a similar decision is made at 270 days. This procedure in no way violates the principle of the prime experiment and illustrates one advantage of the alternate mission modes. At the conclusion of the experiment, the animals are retrieved.

The astronaut can interface with the experiment in several ways:

- Replace failed components/subassemblies.
- Replace tapes.
- Return recorded data to Earth.
- Return one or both animals.
- Bring up new animals.

When the OPE is installed in the AM/MDA, the lifecell and ECS modules are launched with the OWS, and the animal transportation canisters are launched on a manned vehicle. The distribution of the launching of other modules is a subject for mission study. After all modules have been brought together and activated, the animals are inserted. The same procedure for water sampling and potability verification which was outlined for the docked version is followed. In addition to the interfaces mentioned for the docked version, the astronauts can also perform major maintenance of the experiment by temporarily containing the animals in the transportation canisters and replacing or adjusting

*The water processing equipment provided in the independent spacecraft includes all of the elements of an air-evaporation water recovery system and presumably the water output is potable. This cannot be easily verified with the independent spacecraft, but can be verified in the cluster configuration by ground-analysis of a sample returned by a visiting astronaut. The water will be stored at 200F to prevent microbial degradation.

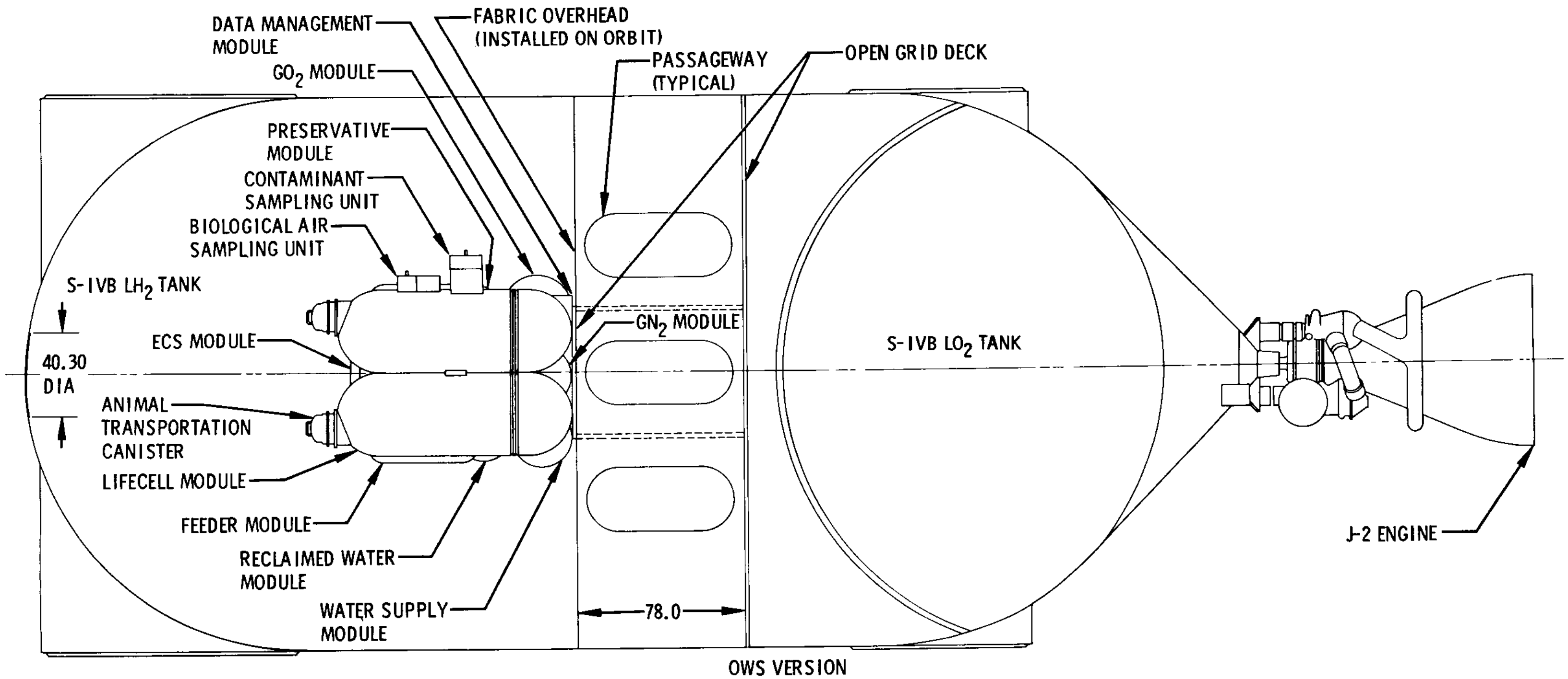


Fig. 175a Prime and Additional Experiments Installed in OWS

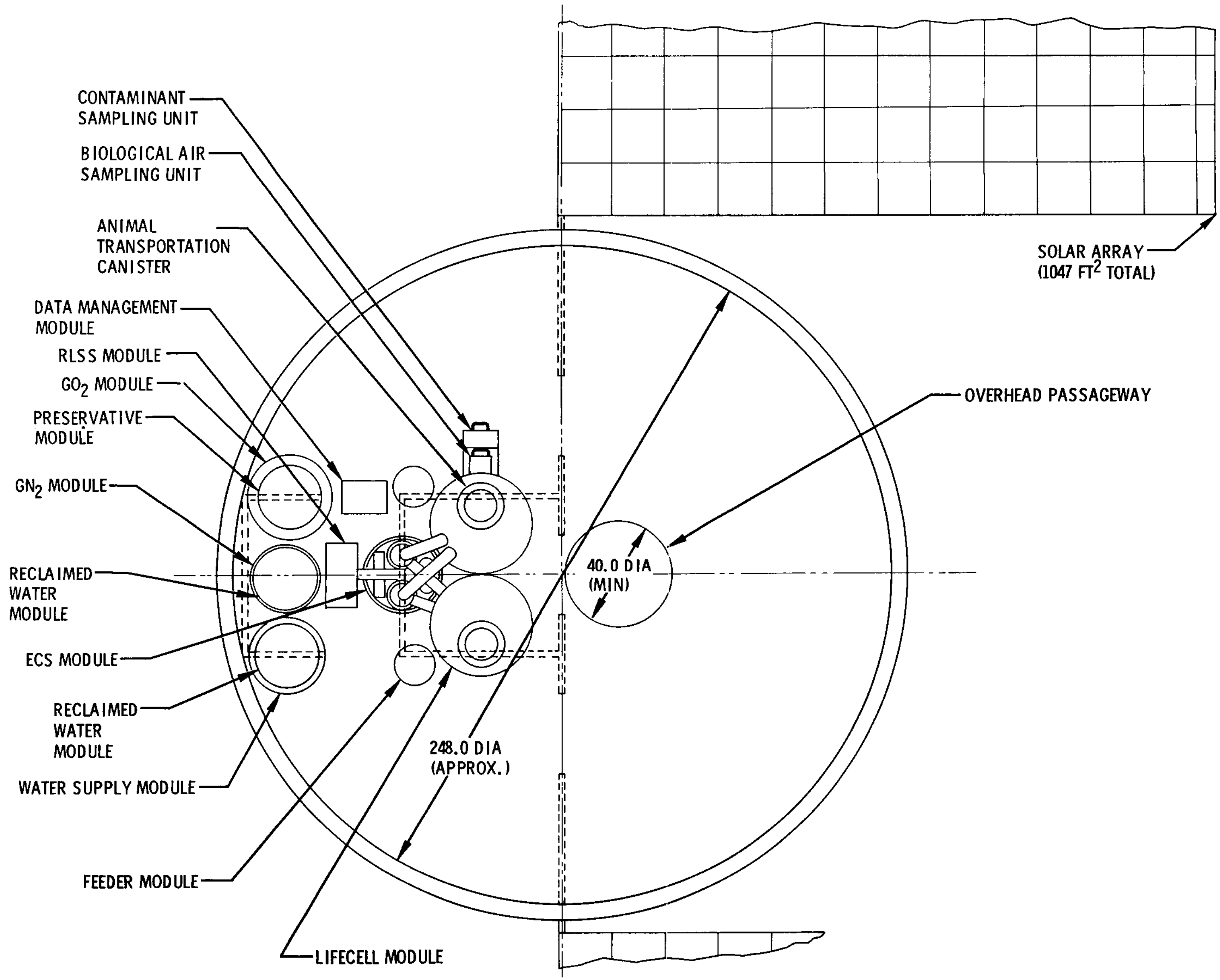


Fig. 175b Prime and Additional Experiments Installed in OWS (Cont.)

malfunctioning equipment. Also, photographic film cassettes of the animals could be changed and returned to Earth by the simple inclusion of a camera in the life-cell or a window installation for a hand-held camera.

When the OPE is installed in the OWS, the lifecell modules, due to their large diameter, must be launched in the MDA of the OWS. The animal transportation canisters must accompany a manned launch. All other modules may be launched whenever the mission analysis dictates, since their existing dimensions or subdividability permit ingress from the outside into the OWS. After orbital assembly and activation, the animals are inserted. The same procedure for water sampling and potability verification which was outlined for the docked and AM/MDA versions is followed. All the interfaces mentioned for the docked and AM/MDA versions may also be accomplished by the astronauts in the OWS version somewhat more readily due to the shirtsleeve environment of the OWS. If the animals could be subjected to the 5 psia of the OWS, the replacement or refurbishment of the OPE could be greatly enhanced.

Effect on system parameters. – For each of the three alternate mission modes, the power requirement is reduced from that of the independent version by 23 w to give a continuous power requirement of 377 w. Heat rejection loads are increased from 785 to 1273 Btu/hr. These values are reflected in the weight summaries of Tables 88 and 89 which show the total weight of the docked version to be 5086.9 lb and the AM/MDA version 4446.6 lb. The weight breakdown of the OWS version is the same as that of the AM/MDA version except that the 25-lb meteoroid protection requirement is deleted. The weight of the OWS version is therefore 4421.6 lb.

Having the OPE docked to the MDA raises the possibility of time-sharing any available data management capability. Utilization of the MDA data acquisition, encoding, and recording equipment can only be considered in a secondary or back-up mode, since this equipment has a very low sampling rate on its commutated PAM input (0.5 SPS x 60 channels) and a single channel 1.6 KBPS PCM encoder. Both the PAM commutator and the PCM encoder drive voltage controlled oscillators (VCO's). The VCO outputs are recorded on an FM recorder with 32 minute recording time and 8 minutes of playback time. The MDA (as presently proposed) does not contain RF transmission equipment, and as a consequence, the output of the FM recorder is routed to VHF transmitters in the OWS. Not only are these data sampling rates incompatible with the OPE requirements, but the TV signals generated by the OPE cameras could not be transmitted on the low (125 KHz) bandwidth VHF transmitters in the OWS.

Another possibility, that of having USB transmitters available either as part of the MDA or via a cable to one of the two CSM units, has been recommended by LMSC as part of the Integrated Medical and Behavorial Laboratory Measurement System (IMBLMS) conceptual design. Since data does not have to be transmitted from the OPE during ascent and descent, such USB equipment (via a docked interface connector) could be used in place of the transmitter that would normally be carried by the OPE carrier. The IMBLMS, as presently conceived by LMSC, has a data management system with extensive growth capability. In its ultimate or fully expandable condition, it will contain high-resolution/low-scan rate TV equipment including a video recorder, a high-speed data sampling and recording system, a special processor for data routing, evaluation and compression, and will interface with the Up-Link command system in the Airlock

TABLE 88

DOCKED VERSION WEIGHT SUMMARY

Independent spacecraft launch weight		+5,500.0
Plus male docking fitting and structure		+92.4
Less attitude control system		-282.0
GN_2 and tank and support	-267.0	
Pneumatics and sun sensors	-11.0	
Electronics	-4.0	
Less solar arrays		-108.0
Less space radiator		-28.3
Less one-half of water supply		-413.4
Water	-386.0	
Tank and support	-27.4	
Less one-half of water storage capability		-11.1
Plus water storage tank insulation		+10.4
Less X-band system (including antenna)		-30.8
Less duplicate S-band and UHF antenna		-7.3
Plus retrieval canister augmentation 2 at 10.0		+20.0
Plus S-IVB solar array augmentation (377 w) (0.836 lb/w)		+315.0
Plus S-IVB space radiator augmentation (1273 Btu/hr) (0.066 lb/Btu/hr)		+84.0
Less meteoroid protection reduction		-54.0
Total weight docked version OPE		5,086.9

TABLE 89

MDA/AM VERSION WEIGHT SUMMARY

ECS module			519.0
Ducting		15.0	
LiOH + charcoal canister		23.0	
Condensate heat exchanger	2 at 2.0	4.0	
Check valves		1.0	
Low flow fan assy	4 at 6.0	24.0	
Coolant pump	2 at 3.5	7.0	
Coolant accumulator		12.0	
Coolant plumbing		3.0	
Coolant		10.0	
Temp. control valves		2.0	
Reclaimed water plumbing		3.0	
Support structure		47.0	
Charcoal	2 at 23.0	46.0	
LiOH	2 at 156.0	312.0	
Misc.		10.0	
Lifecell Module	2 at 478.9		957.8
High flow fans	2 at 3.75	7.5	
Cabin purge fans	2 at 6.00	12.0	
Vent valve assy.		2.0	
Relief valve assy.		2.0	
Atmosphere supply plumbing		1.0	
Ducting		5.0	
Waste management		67.0	
Psychomotor		17.0	

TABLE 89 (Cont.)

Retrieval canister (less preservative & tank)		84.5	
Drinking system (less water and tank)		14.8	
Pressure vessel		60.0	
Metabolic support structure		32.5	
Cage complex		42.0	
Data mgmt. sys. (inside lifecell)		51.6	
Mass measurement		40.0	
Support structure		40.0	
Feeder Module	2 at 154.5		309.0
Canister		14.0	
Mechanism		24.5	
Food (46,643 gm)		103.0	
Support structure		13.0	
Tankage modules – Oxygen module			704.0
Oxygen		246.0	
Tank and lines		389.0	
Regulator		5.0	
Support structure		64.0	
– Nitrogen module			279.4
Nitrogen		95.0	
Tank and lines		154.0	
Regulator		5.0	
Support structure		25.4	
– Preservative module			293.7
Preservative		250.0	
Preservative tank and lines		17.0	

TABLE 89 (Cont.)

Support structure		26.7	
– Water module			451.0
Water		386.0	
Tank and lines		24.0	
Support structure		41.0	
Reclaimed water module	2 at 21.7		43.4
Tank	2 at 20.2	40.4	
Support structure	2 at 1.5	3.0	
Data management module (outside lifecell)			209.2
Box		16.8	
S-band transceiver		20.2	
Amplifier/diplexer		16.8	
Digital updata link		22.4	
PCM and timing equipment assy.		35.0	
TV recorder		20.0	
Work panel recorder		8.0	
Data storage recorder		11.0	
Signal processor assy.		11.5	
Animal data commutator		1.5	
Temperature bridges		1.5	
Rate sensor (medical)		1.5	
Rate sensor (attitude)		1.0	
Cold plate		15.0	
Support structure		12.0	
Misc. (wire, plugs, brackets)		15.0	
Antennas			4.3
S-band		1.3	

TABLE 89 (Cont.)

UHF	2.0	
Mounting provisions	1.0	
Electrical power (377 w) (1.28 lb/w)		482.8
Thermal control		168.0
Radiator (1273 Btu/hr (0.066 lb/Btu/hr)	84.0	
Insulation	30.0	
Finishes	54.0	
Meteoroid protection		25.0
Total weight MDA/AM Version OPE		4,446.6

Module. Should the IMBLMS be available in the previously described state of growth, it could easily fulfill the OPE's PCM-Timing Equipment Assembly, the Animal Data Commutator and out-of-limits monitor, Digital Recorder, Signal Processor Assembly and Video Recorder functions. Some of the TV electronics could be time-shared and, in addition, primate voice recording capability would always be available.

Some schedule conflict could occur, however, it is presently felt that the 10-sec sample of OPE data (except video) every 12 min should have little impact on IMBLMS. The possibility of preprocessing OPE data would reduce downlink and ground cluttering.

Assuming a nonaxial docking configuration and USB equipment in the OPE, it is necessary to orient the OPE on the Earth side of the cluster to prevent shadowing of the antenna patterns.

Should USB equipment be provided as part of the OPE, it is not inconceivable to consider this same equipment to fulfill the need created by the IMBLMS. After the OPE mission is completed, the carrier could remain on-orbit and continue to satisfy these wide-band transmission requirements. However, because of ground station limitations, operation of the added USB equipment will have to be time-shared with similar CSM and LM USB transmissions.

From a data management viewpoint, having the OPE installed in the MDA is not significantly different than the case of docking the OPE to the MDA. The need for an interface connector as part of the docking mechanism is eliminated while the need for USB antennas (provided the OPE has integral USB equipment) must be satisfied by the addition of these antennas to the surface of the MDA.

Having the OPE installed in the OWS presents the possibility of using the data management system integral to the OWS. The PCM portion of this system can handle OPE sampled data requirements except those exceeding 80 samples per second. Real-time capability exists from 80 SPS through 640 SPS, but this data cannot be recorded for delayed transmission due to the limited (5.12 KBPS) input capability of the OWS recorder. Unfortunately, the PCM portion of the OWS data management system, besides falling short in the higher sampling rate requirements is presently delegated to handling OWS housekeeping data, plus AM and MDA flight control data. A commutated PAM system with a 60-channel input and a commutation rate of 1 channel per sec is provided for experiment data and is completely inadequate for OPE. The transmitters serving the OWS data acquisition equipment, as previously mentioned, are bandwidth limited to 125 KHz and could not handle the OPE video data. Therefore, the OPE will require its own DMS even when installed in the OWS.

All of the discussion previously presented relative to the IMBLMS and USB equipment and relative to the need for additional antennas is applicable to the OWS situation.

Experiments in Addition to Prime Experiment

Additional experiments which may be added to the vehicle include a regenerative life support subsystem (RLSS), a contaminant analysis unit, a chemical sampling unit, and a biological sampling unit. These items are described in the following paragraphs.

Regenerative life support subsystem. – The regenerative life support subsystem is a modularized package which may be added to the vehicle to perform the functions of the basic thermal and atmosphere control subsystem. The regenerable subsystem can process the lifecell atmosphere to remove carbon dioxide and trace contaminants; maintain lifecell temperature, pressure, and humidity conditions; and provide oxygen. The basic nonregenerative subsystem is also provided for use in the event of regenerative system malfunction. In this way, malfunctions in the regenerative subsystem will not endanger the prime experiment. The regenerative subsystem is shown schematically in Fig. 176 with components identified in Table 90. The subsystem consists of:

- A contaminant, temperature, and humidity control unit including fans, charcoal, heat exchanger, water separator, and temperature control valve
- A CO_2 removal unit including silica gel and Molecular Sieve beds
- A CO_2 reduction unit including vacuum pump, carbon dioxide accumulator, Sabatier reactor assembly, condensing heat exchanger, and a high-temperature fluid system
- An electrolysis unit including a control unit, oxygen pump, oxygen accumulator, and regulator and relief valve assembly
- A catalytic oxidizer including pre- and post-sorbent beds

The regenerative subsystem can be used either in total as shown in Fig. 176 or in part, with minimum adaptation to account for deleted elements. Optional configurations are suggested below.

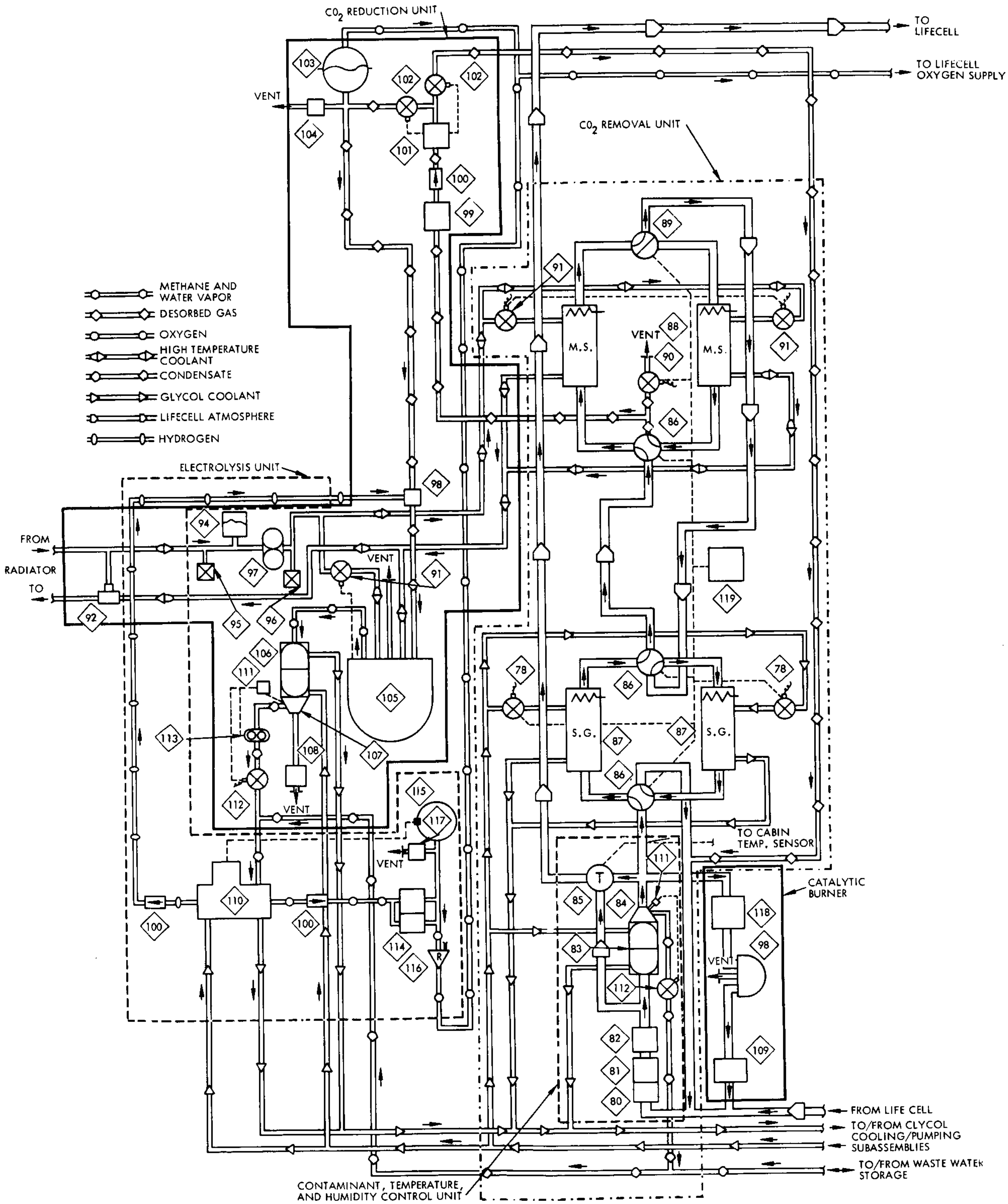


Fig. 176 Regenerative Life Support Subsystem Schematic

TABLE 90

REGENERATIVE LIFE SUPPORT SUBSYSTEM COMPONENTS

Item No.	Description
78*	Valve, shutoff, solenoid, coolant
79*	Valve, shutoff, solenoid, gas, low flow
80	Assembly, check valve and filter
81	Assembly, fan, RLSS flow
82	Assembly, canister, charcoal
83	Assembly, condensate heat exchanger
84	Assembly, water separator
85	Valve, bypass temperature control
86	Valve, sorbent BEP
87	Assembly, canister, silica gel
88	Assembly, canister, molecular sieve
89	Valve, outlet molecular sieve
90	Valve, vacuum desorption
91	Valve, solenoid, shutoff, high-temp. coolant
92	Valve, coolant supply temperature
93	Valve, mixture control
94	Accumulator, high-temp. coolant
95	Coupling, high-temp. coolant disconnect, servicing, outlet
96	Coupling, high-temp. coolant disconnect, servicing, inlet
97	Module, coolant pump package
98	Assembly, catalytic burner
99	Module, vacuum pump
100	Valve, check, gas

*In basic thermal and atmosphere control subsystem – not shown on schematic.

TABLE 90 (Cont.)

Item No.	Description
101	Assembly, sensor controller, partial pressure, O_2
102	Valve, shutoff, solenoid, gas
103	Accumulator, CO_2
104	Valve, relief, CO_2
105	Assembly, reactor, sabatier
106	Assembly, condensate heat exchanger
107	Assembly, water separator
108	Assembly, CH_4 vent disconnect and pressure control
109	Assembly, canister, catalytic burner post sorbent
110	Assembly, electrolysis cell and controller
111	Sensor, differential pressure, water separator
112	Valve, water control
113	Assembly, water pump
114	Module, Oxygen pump and bypass valve
115	Accumulator, O_2
116	Assembly, regulator, O_2
117	Valve, relief, O_2
118	Assembly, canister, catalytic burner, presorbent
119	Assembly, timer CO_2 removal unit

1. Electrolysis only - maintains sufficient oxygen for breathing and leakage by electrolyzing collected suitably purified waste water; hydrogen is vented overboard.

2. Temperature and humidity control, and contaminant and CO_2 removal - controls life cell temperature, humidity, CO_2 and contaminant concentrations. CO_2 is desorbed to space. Temperature and humidity control are interrelated and humidity control is desirable upstream of Molecular Sieve unit.

3. Combine 1 and 2.

4. Add Sabatier reactor to configuration 3 - CO_2 from Molecular Sieve is desorbed and fed to Sabatier reactor together with hydrogen from electrolyzer. Methane and water are produced in reactor. Methane is vented overboard; water is electrolyzed.

5. Add elevated temperature contaminant removal to configuration 4 - Processes additional small portion of system flow; reduces chemical and biological contaminant levels in life cell.

In the experiment studies presented herein, only configuration 5, the complete regenerative package, has been considered. This configuration is described in the following paragraphs.

Contaminant, temperature, and humidity control unit: The inlet flow to the RLSS module comes from the downstream side of the waste management wicking material. During operation of the RLSS, flow through the basic thermal and atmosphere control system is prevented by two solenoid-operated shut-off valves. The fan module contains four redundant fans, each one capable of providing 26 cfm: 1.7 to the CO_2 removal unit, 0.1 to the catalytic burner, and the remainder to the life cell. The pressure rise required is estimated at 1 in. H_2O. The charcoal bed contains sufficient phosphoric-acid impregnated charcoal for the full one-year removal of ammonia, low-boiling hydrocarbons, and a variety of other contaminants which are well absorbed on charcoal.

In view of this, the charcoal has been off-loaded from the LiOH canisters in the basic subsystem. Fan and charcoal bed reliabilities are satisfactory for the entire mission. If other portions of the RLSS are inoperative, the fan and charcoal canister will operate in parallel with the primary life support subsystem. This precludes having to provide redundant charcoal beds which would represent a significant weight penalty.

In a manner similar to the basic thermal and atmosphere control system, the flow passes through a condensing plate-fin heat exchanger, past it, or a combination of the two, depending on the signal from the cabin temperature sensor/controller. Glycol-water at 45° F is supplied to the heat exchanger to cool the gas and condense water vapor. Condensed water is removed by a hydrophobic/hydrophilic water separator. Water is extracted from the separator by opening a solenoid valve to the water storage system which operates at a pressure slightly below that of the life cell. The solenoid valve is opened by a differential pressure switch sensing the pressure drop across the hydrophilic sump. A portion of the extracted water is transferred to the electrolysis cell feed system and the remainder is transferred to water storage.

Following temperature and humidity control as described above, 1.7 cfm of the air flow proceeds to the regenerative CO_2 removal unit, 0.1 cfm flows to the catalytic burner, and the remainder is returned to the life cell.

Catalytic burner: The flow to the catalytic burner assembly passes through a chemically basic presorbent canister to remove potential catalyst poisons, over a 0.5 percent palladium on alumina catalyst at 750° F, through a basic postsorbent to remove undesirable products of combustion, and then back to the fan inlet. Using the above process, CO, CH_4, H_2 and other contaminants not removed by the acid-impregnated charcoal or Molecular Sieve are converted to CO_2 and/or water.

CO_2 removal unit: The air processed by the CO_2 removal unit first passes through the cooled silica gel bed for drying to a -15° F dew point, then through the cooled Molecular Sieve bed for CO_2 removal, and finally over the electrically heated (300° F) silica gel bed to adsorb water from it. In this process, there is no net water adsorption. The electrically heated Molecular Sieve bed (350° F maximum) is desorbed to the CO_2 reduction unit normally, but it can be desorbed to space if necessary because of a malfunction of the CO_2 reduction unit. A separate high-temperature fluid loop (Coolanol 35) and radiator combination is used for cooling the Molecular Sieve beds. After the silica gel bed is dried and the Molecular Sieve bed is desorbed, they are cooled by diverting 45° F fluid through coils imbedded in them.

The unit operates on a two-hour cycle to reduce the heat-up, cool-down and valve actuation frequencies. Adsorbing beds are exposed to water vapor and CO_2 for the full two hours. Desorbing beds are heated and desorbed for 1.5 hr and then cooled for 0.5 hr prior to alternation of the beds by the flow diverter valves.

The four-bed system is used to prevent water poisoning of the Molecular Sieve beds. The Molecular Sieve preferentially adsorbs water vapor. Progressive accumulation of water in the Molecular Sieve will degrade its capability for adsorbing carbon dioxide, and it is difficult to desorb water from the Sieve. Silica gel can be easily desorbed at 300° F; consequently, it is more practical to remove moisture from the process gas with silica gel prior to its delivery to the Molecular Sieve, and return this moisture to the atmosphere by heated desorption after the gas has passed through the Molecular Sieve. The flow rate of 1.7 cfm was based on an inlet CO_2 concentration of 3.8 mmHg and a CO_2 production rate of 0.67 lb/day (2-13 lb Rhesus at 1.5 BMR).

CO_2 reduction unit: The desorption pump in the CO_2 reduction unit compresses effluent gas from the desorbing Molecular Sieve at less than 2 psia, to the CO_2 accumulator pressure of 35 psia. Pump discharge gas first passes over an oxygen sensor/controller and, until oxygen concentration is reduced to approximately 12 mmHg, the effluent gas is discharged to the cabin. This is done to eliminate, insofar as possible, oxygen and nitrogen from the CO_2. With this approach, CO_2 purity to the Sabatier reactor is approximately 95 percent. The CO_2 accumulator discharges gas to the Sabatier reactor through a mixture control valve. Excess CO_2* is vented overboard through a relief valve. The CO_2 is mixed with hydrogen prior to entering the Sabatier reactor. The mixture is maintained approximately 10 percent hydrogen-rich to achieve the optimum Sabatier reactor conditions. The reactor operates at a pressure of 10 psia, a temperature

*If the electrolysis system is generating only sufficient oxygen for leakage and metabolic consumption, then it is not generating sufficient hydrogen to react all of the CO_2.

of 500° F, a space velocity of 550 hr^{-1} and achieves a CO_2 conversion efficiency of approximately 98 percent. The reactor is started by heating it electrically. Once this is done, energy from the reaction maintains temperature. After start-up, the Coolanol-35 system accepts excess heat from the reactor. Hydrogen and CO_2 passing over the hot ruthenium catalyst are converted primarily to methane and water. Reactor effluent gases are cooled to approximately 50° F and condensed water vapor is separated and delivered to the electrolysis cell. Methane and other noncondensible gases are vented overboard. The Coolanol-35 system has a flowrate of 0.71 lb/hr at a head rise of 5 psi. The heat rejection rate is 170 Btu/hr, requiring a radiator surface area of 5.5 ft^2.

Electrolysis unit: The electrolysis unit uses a 30 percent KOH circulating electrolyte. Gas/liquid phase separation is accomplished by using an absorbent-matrix porous-electrode combination. The electrolyte is maintained at 75° F by the circulating glycol system, and the system operates at approximately one-atmosphere pressure. Water passed through an ion-exchange resin to render it neutral or slightly basic is supplied to the electrolysis cell automatically to make up electrolyzed water. The bladdered waste water storage tank acts as an accumulator for this function. The water electrolysis rate is controlled in response to the pressure in the oxygen accumulator. The oxygen produced is pumped from the electrolysis unit pressure of approximately one atmosphere to the accumulator pressure of approximately 40 psia. The pump runs continuously and is provided with a suction pressure control regulator to maintain the gas side of the electrolysis system at the desired level. The oxygen produced is regulated to 35 psia; this is slightly higher than the regulated pressure from the normal oxygen supply. As long as the electrolysis system is operating, it will be the sole supply of oxygen to the life cell. Hydrogen from the cell is fed at constant pressure to the H_2/CO_2 mixture control valve. The mixture control valve admits sufficient CO_2 to react with the available hydrogen.

Mass balance: A mass balance for the CO_2 removal unit, CO_2 reduction unit, and electrolytic cell is shown in Fig. 177. As indicated on this figure, a total of 0.305 lb/day of CO_2 is vented overboard, and make-up water amounting to 0.362 lb is required to produce the required oxygen. This water is stored in addition to the drinking water supply.

Contaminant analysis/chemical sampling unit. – The analysis of chemical atmospheric contaminants in the lifecell is an important consideration to verify that no toxic or explosive condition prevails during the course of the experiment. To accomplish this goal, the following four-fold approach is recommended:

- Mass spectrometer analysis of preselected compounds
- Periodic mass spectral analysis over a wide mass range
- Collection of gas samples for return to Earth and subsequent chemical analysis
- Concentration and collection of chemical contaminants on charcoal, which can also be returned to Earth for analysis

The first approach will provide, through telemetry, a near real-time assessment of the general contaminant level through examination of specific expected compounds. The second approach will provide periodic mass spectra which, following interpretation on the ground, will allow a more complete analysis of the atmosphere. The third and fourth approaches provide a backup to the mass spectrometer and will allow a postflight detailed analysis to be performed through the use of extensive laboratory instrumentation.

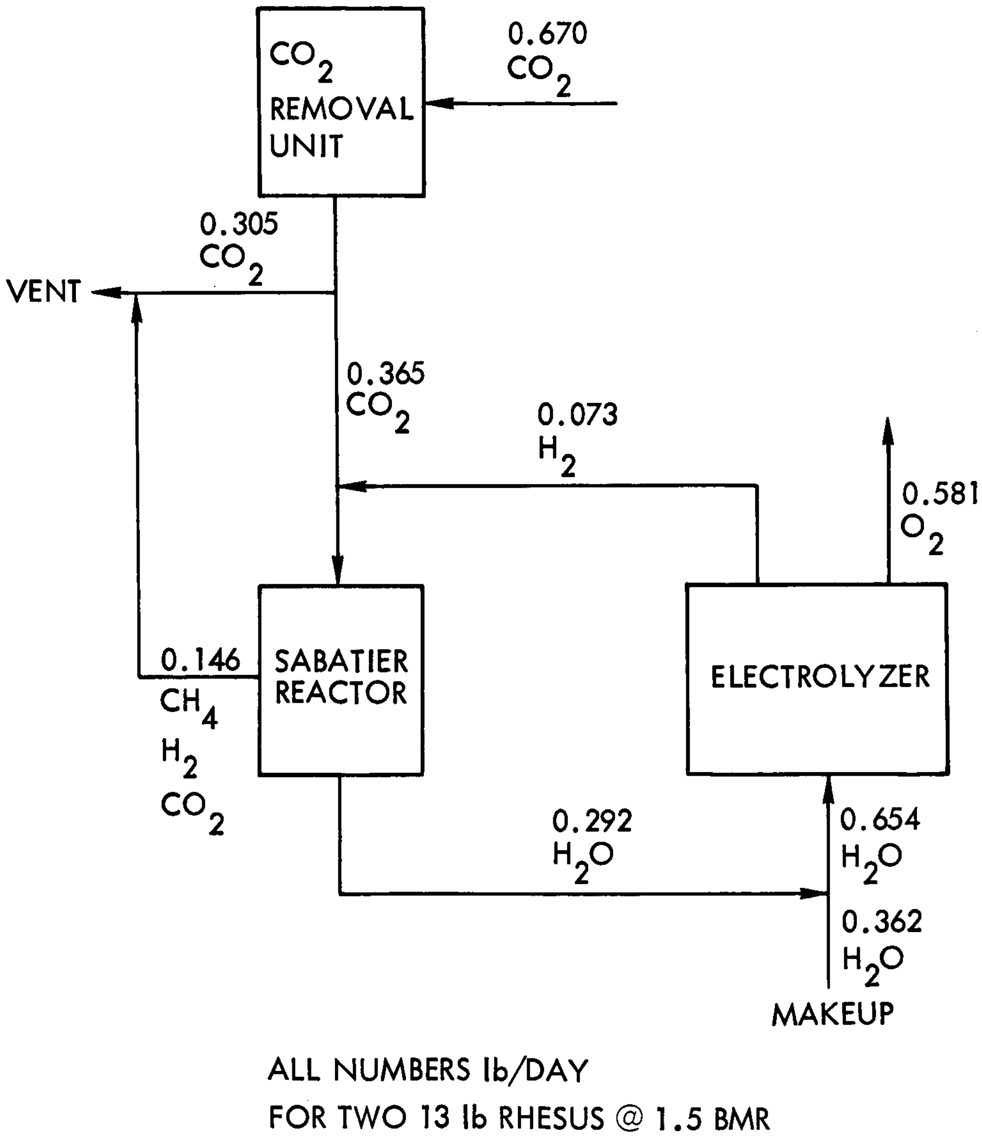


Fig. 177 Regenerative Life Support Subsystem Mass Balance

Mass spectrometer: The mass spectrometer was selected as the flight instrument because of its ability to monitor both specific and total atmospheric contamination. A flight-type mass spectrometer suitable for this purpose is under development for the NASA Langley Research Center (Reference 1). This unit, a miniaturized double-focusing, permanent magnet device, analyzes a small sample of life cell atmosphere which is admitted to an ionizing source through a capillary-type viscous-flow pressure divider and a platinum-aperature molecular leak. In passing through the ion source, the atmospheric gas molecules are bombarded by the ionizing electron beam to create positive ions representing each of the constituents in the atmosphere. By utilizing the double-focusing principle, the ions are accelerated, focused, and directed into the analyzer sector where they are separated into individual ions according to their mass/charge ratio. A dual-mode operation allows selection of twelve mass positions for simultaneous monitoring of individual contaminants and rapid scan capability for detection and measurement of varying and unpredicted contaminants.

In the monitoring mode, detectors are positioned along the instrument's collector plane at positions coincident with the particular contaminant masses of interest. Up to twelve mass positions are monitored simultaneously. A separate amplifier handles each output.

A preliminary selection of contaminants to be monitored is shown in Table 91. The contaminants indicated are of concern from either an explosive or toxic point of view, have been identified in previous animal capsule experiments (Ref. 3), and can be adequately resolved by the selected mass spectrometer. Most of these contaminants have mass spectral responses at several M/e values (Ref. 2) as indicated in Table 89; however, the mass spectrometer will be adjusted to monitor each compound at the one selected M/e shown.

In the scanning mode, the instrument sweeps a wide mass range using a single detector. From the mass spectrum obtained, the partial pressures of a number of contaminants in addition to those measured in the monitoring mode can be computed and, thus, the contaminant concentrations can be determined. A scan of 2 - 150 M/e is required to cover the majority of the trace contaminants anticipated. The signal output from the mass spectrometer is channeled through the main commutator and can be transmitted in real time or recorded for subsequent transmission. The mass spectrometer and its supporting equipment are shown in Fig. 178. A standard gas sample is provided sufficient in size to calibrate the mass spectrometer once each week. This gas sample will contain known quantities of the specific contaminants which the spectrometer is set up to monitor. A vent valve is provided to allow checkout and calibration prior to launch, closure of the unit during prelaunch and ascent to prevent atmospheric contamination, and venting to space vacuum once a sufficient altitude is reached.

Chemical sample collection: Twelve sample collection chambers, each having a volume of 35 $in.^3$, are provided to collect a whole gas sample once per month during a one-year mission. Twelve additional canisters are provided, each containing 0.3 lb of granular charcoal to collect concentrated contaminant samples. Valves are provided on the inlet and outlet sides of each chamber and each canister, as shown in Fig. 179. Life cell gas will flow through one chamber and one charcoal canister continuously. The charcoal will be downstream of the sampling chamber. The chamber will be glass or glass-lined to prevent adsorption of contaminants. Periodically the chamber and charcoal canister will each be sealed-off and a new assembly placed on line. The entire unit will be designed and installed for astronaut recovery.

TABLE 91

PRELIMINARY CONTAMINANT SELECTION MASS SPECTROMETER MONITORING MODE

Contaminant	$\frac{M/e}{(\%\ Response)^*}$	Selected M/e
Hydrogen	$\frac{2}{(100)}$	2
Methane	$\frac{16}{(100)}$, $\frac{15}{(76)}$	15
Ammonia	$\frac{17}{(100)}$, $\frac{16}{(80)}$	17
Carbon Monoxide	$\frac{28}{(100)}$	28
Nitrogen Dioxide	$\frac{30}{(100)}$, $\frac{46}{(37)}$, $\frac{16}{(22)}$	30
Hydrogen Sulfide	$\frac{34}{(100)}$, $\frac{32}{(44)}$, $\frac{31}{(42)}$	34
Acetonitrile	$\frac{41}{(100)}$, $\frac{40}{(52)}$, $\frac{39}{(19)}$, $\frac{38}{(12)}$	40
Methyl Mercaptan	$\frac{47}{(100)}$, $\frac{48}{(90)}$, $\frac{45}{(47)}$	47
Butyric Acid	$\frac{60}{(100)}$, $\frac{27}{(41)}$, $\frac{73}{(27)}$	60
Ethyl Mercaptan	$\frac{62}{(100)}$, $\frac{29}{(85)}$, $\frac{47}{(77)}$, $\frac{27}{(67)}$	62
Sulfur Dioxide	$\frac{64}{(100)}$, $\frac{48}{(49)}$	64
Benzene	$\frac{78}{(100)}$, $\frac{52}{(20)}$, $\frac{51}{(21)}$, $\frac{50}{(17)}$	78

*Magnitude of mass peak at specified M/e value. Maximum peak height = 100.

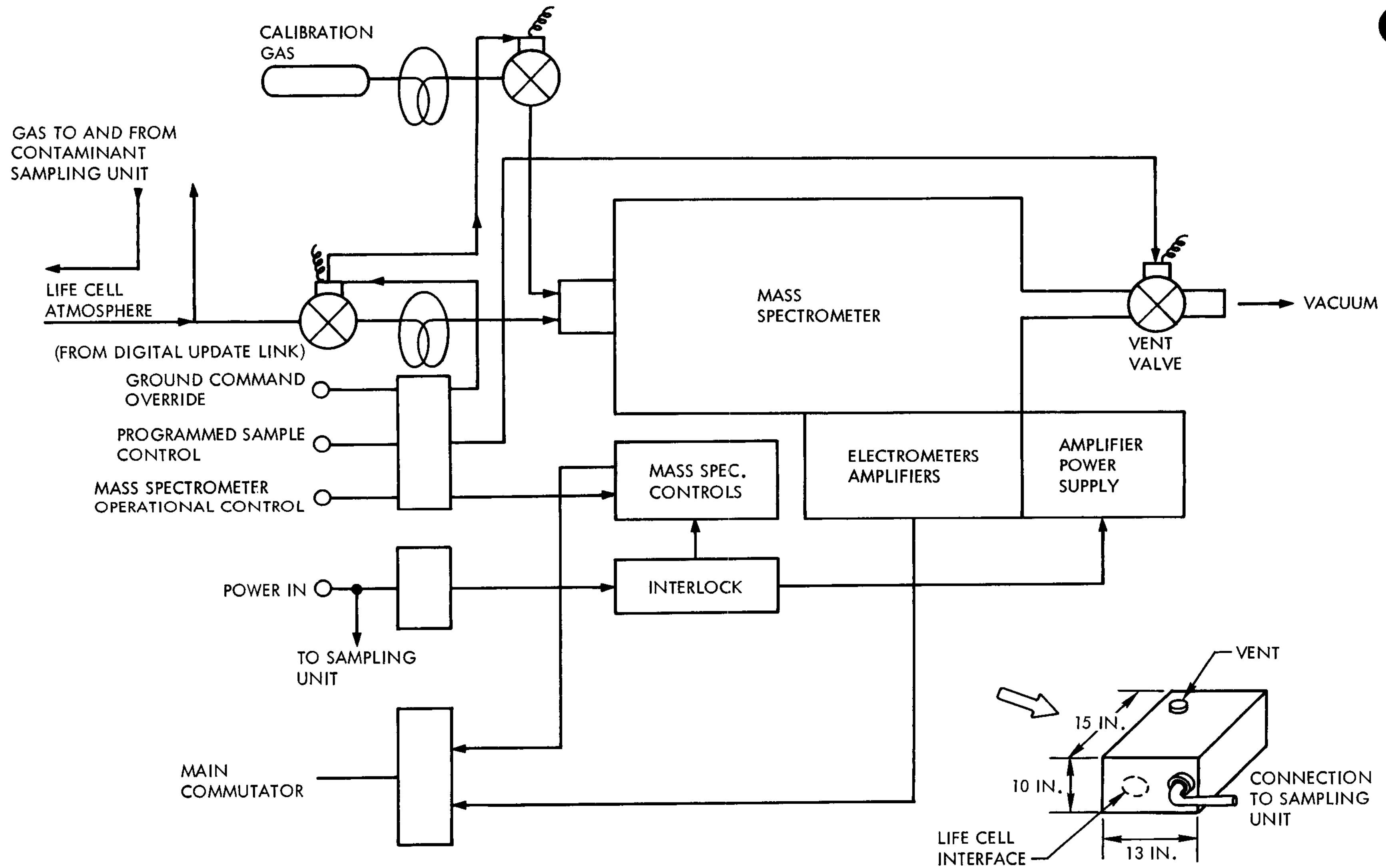


Fig. 178 Mass Spectrometer and Associated Equipment

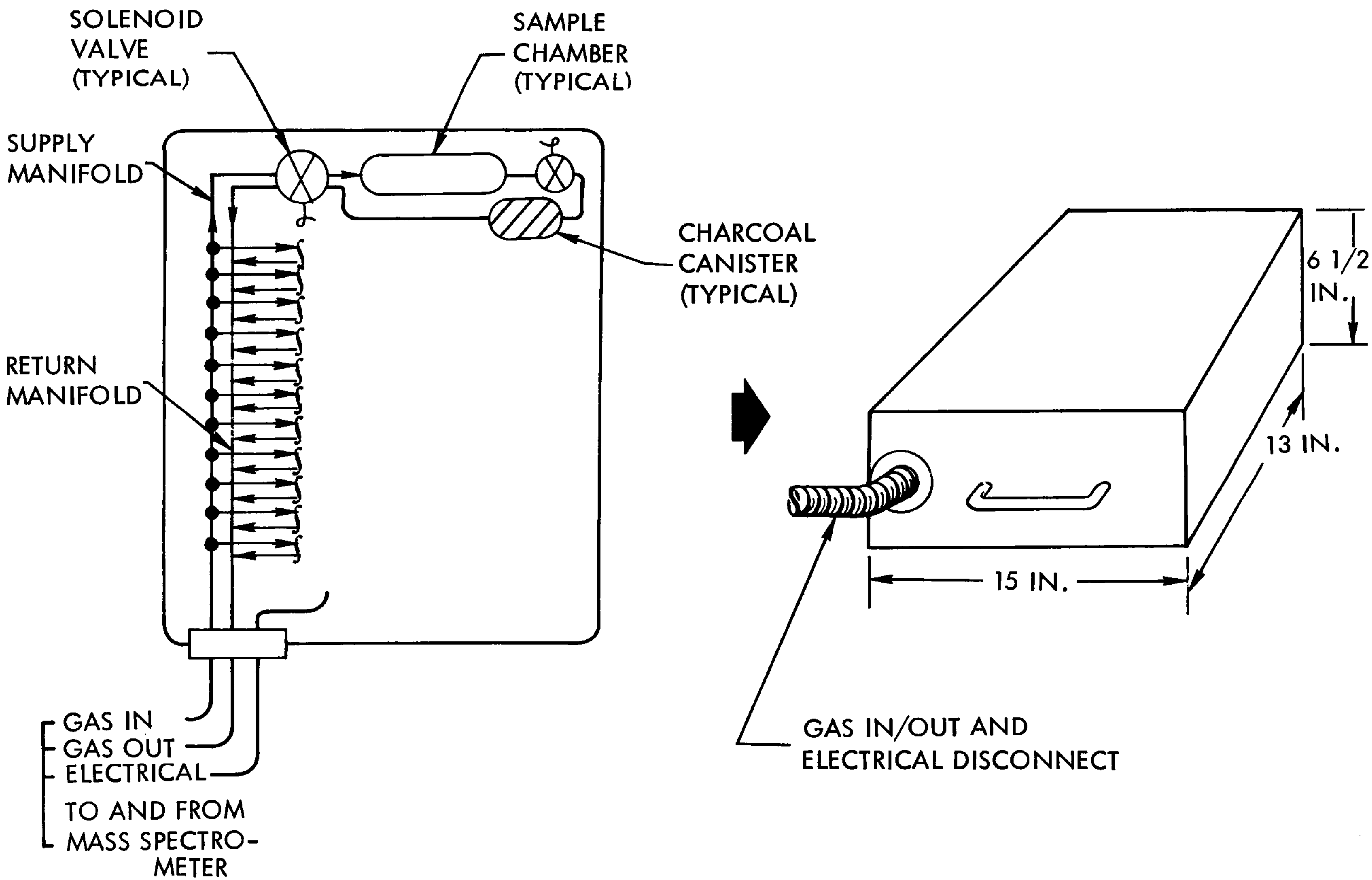


Fig. 179 Chemical Sample Collection Chamber

Biological air sampling. – Sampling of the life cell atmosphere to determine the types and quantities of micro-organisms present as a function of experiment duration is important to establish the level of this stress on the primate. Consideration was given to flow-through filter, liquid-trapping, and gel-impingement techniques. The filter collection system relies on micro-porous media to trap micro-organisms for subsequent culturing to establish types and quantities. This approach is quite suitable for the zero-gravity application but suffers some sample kill due to air drying. The liquid trapping scheme avoids the air-drying problem but introduces a gas-liquid phase separation requirement. The gel-impingement technique avoids both the air-drying and zero-g phase separation problems. Primarily for these reasons, it was chosen as the preferred technique. This technique relies on a 90-deg change in air flow direction to separate micro-organisms from the air sample. The separated micro-organisms impinge on and adhere to an Agar surface placed at 90 deg to the inlet air stream. The Agar provides sufficient moisture to prevent drying and acts as an inert carrier for the micro-organisms. The exposed Agar surface is stored at a temperature of approximately -10° F in a sealed container to prevent micro-organism growth and drying.

Figure 180 shows an apparatus to accomplish biological air sampling in this manner once per day for a one-year mission. In this apparatus, approximately 380 3/4-in. square Agar surfaces 0.020 in. thick are placed on an impermeable tape 2 in. wide and 0.002 in. thick. The remainder of the tape surface is covered with an adhesive. A 0.002 in. thick Mylar film is placed on the adhesive side, thereby covering and sealing the Agar squares. Eighty feet of this composite tape is wound on the supply reel. As this tape is transported through the sampling unit, the Mylar cover is stripped off, the Agar surfaces are exposed to the air impingement head, and the Mylar rejoins and reseals the tape prior to reaching the takeup reel as shown in the inset on Fig. 180.

The cassette for the takeup reel is insulated and provided with fluid passages to maintain a temperature of approximately -10° F. This cassette is detachable and is recovered for microbiological analysis, after return to earth.

The air sample is withdrawn from the life cell just after the air has passed through the waste management system and prior to return to the cage.

Mission modes. – The RLSS, the Containment Analysis/Chemical Sampling Unit, and the Biological Sampling Unit may be flown in any of the four mission modes: independent, docked to the MDA, installed in the AM/MDA, or installed in the OWS.

The RLSS is packaged in a module of 50 in. x 25 in. x 12 in. In the independent or docked modes, the module is located at one end of the life cell to simplify interface connections as shown in Fig. 168. In the AM/MDA installation, the module is mounted to the AM truss structure as shown in Fig. 174. In the OWS mode, the module is assembled with the remainder of the prime experiment as shown in Fig. 175.

The Contaminant Analysis/Chemical Sampling Unit contains the mass spectrometer and its supporting equipment together with the atmosphere sampling unit. The combined package is contained in a rectangular box 13 in. x 15 in. x 16.5 in. The gas sampling assembly portion (13 in. x 15 in. x 6.5 in.) can be easily detached for retrieval by an Apollo astronaut. In the independent and docked modes, the entire unit is mounted at the lower end of the life cell adjacent to the regenerative life support subsystem module. The gas sampling assembly is accessible and removable through an access door in the spacecraft skin. This configuration is shown in Fig. 168. Other installations are shown in Figs. 174 and 175.

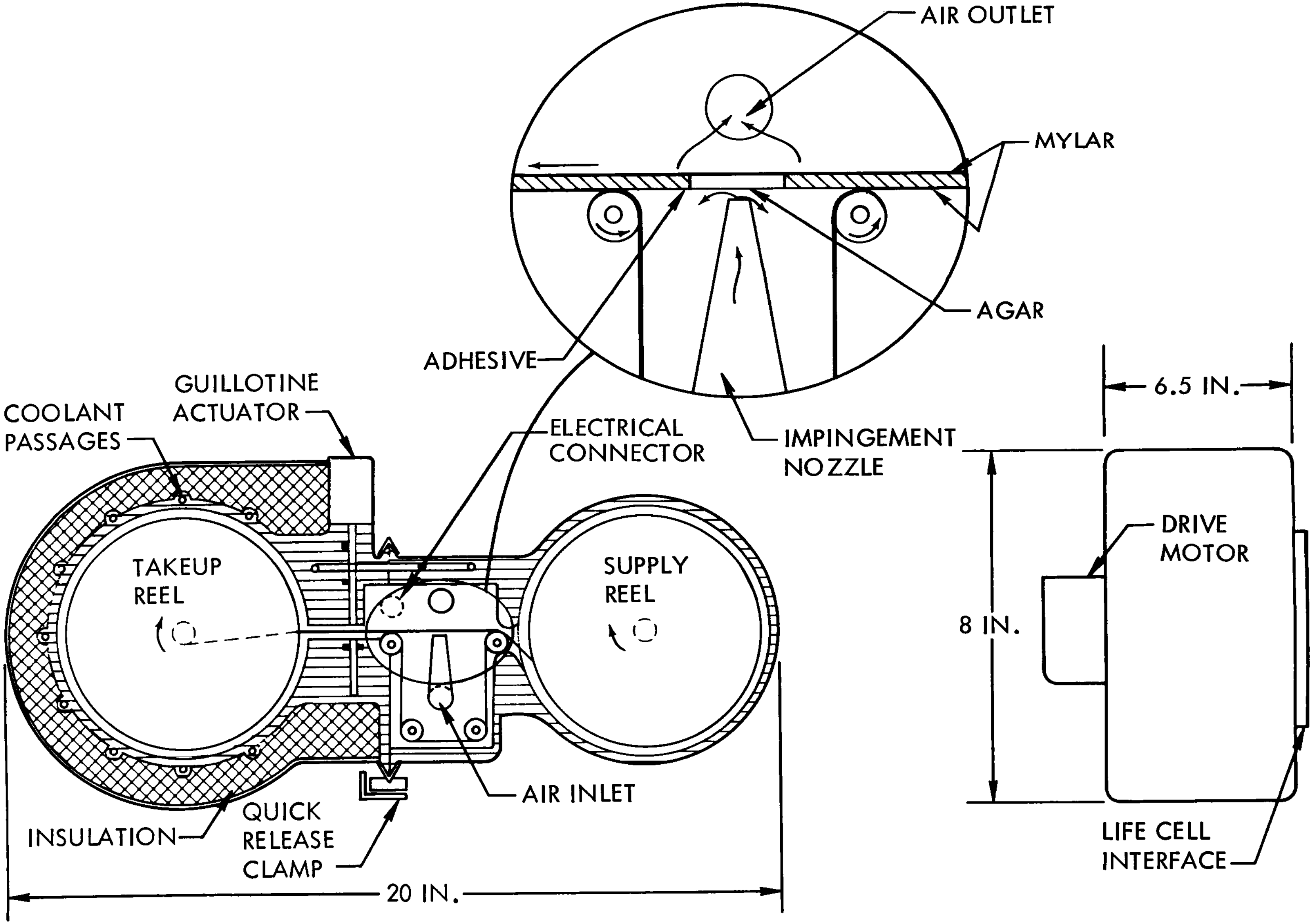


Fig. 180 Biological Air Sampling Unit

The Biological Sampling Unit is mounted external to the life cell. Interfaces with the life cell are 1/2-in. air inlet and outlet ports, 1/4-in. coolant inlet and outlet ports, and 28-VDC power. Application of power once a day will advance the tape to expose and store one Agar surface. Air and coolant flow in and out of the unit continuously. The installations for the various mission modes are shown in Figs. 168, 174, and 175.

Operations. — In the independent mode, the RLSS will be operating prior to launch and will provide the functions of temperature and humidity control, CO_2 and trace contaminant removal, and oxygen supply as long as it remains operable. Time of successful RLSS operation adds directly to mission duration. A minimum mission duration of six months is achievable with no successful RLSS operation. Moderate diagnostic data and override capabilities are included to enhance RLSS experiment success. The system can operate in some partially-successful modes and still provide useful functions. For example, the electrolysis unit alone can electrolyze waste water to make oxygen thereby saving stored gas, and the CO_2 removal unit can operate alone to avoid use of the LiOH stores. On the independent spacecraft mission, this would not increase mission time but would provide more information on the performance of the RLSS elements which remain operable. Complete switch-over to the basic thermal and atmosphere control subsystem is accomplished by deenergizing all but the RLSS fans, energizing the basic subsystem, opening the air flow valves to the LiOH, opening the glycol-water valves to the basic subsystem heat exchangers, closing the glycol water valves to the RLSS heat exchangers, and switching the temperature control function to the basic subsystem. As mentioned in the RLSS subsystem description, the contaminant removal function remains with the RLSS throughout the mission. Sufficient redundancy is included in the RLSS fan module to ensure successful full-mission operation.

In the docked mode, the RLSS will be checked out prior to launch, shut down for launch, and restarted prior to animal insertion.

When installed in the AM/MDA mode, the RLSS will be checked out and operated prior to launch but shut down during launch to reduce battery requirements. Startup and final checkout will be accomplished on orbit by the Apollo crew prior to animal insertion. In this modular configuration, the possibility exists to replace an inoperative RLSS experiment package at a later point in the mission through EVA.

When installed in the OWS, the entire RLSS package will be checked out, operated and deactivated prior to launch. The deactivated package will be carried inside the MDA during ascent. Following OWS activation on orbit, the package will be installed in the OWS. In this modular configuration, located in a shirtsleeve environment, increased possibilities exist for replacement of components and subassemblies by the astronaut.

The Contaminant Analysis/Chemical Sampling Unit operates automatically on command from the data management subsystem programmer. Once each day, the mass spectrometer is operated in each of the monitoring and scanning modes. Once each week, the mass spectrometer is calibrated in each mode using a sample from the calibration gas reservoir. Once each month, on a one-year mission, a new gas sampling chamber and charcoal canister are placed on the flow line and the previous units are sealed off. On shorter missions, the gas sampling unit would be operated more frequently. Ground commands can be given to override the mass spectrometer program so that calibration, monitor, and scan modes can be selected when desired, or the program changed with regard to these functions.

For all mission modes, upon completion of the mission, the astronaut retrieves the gas sampling assembly by removing the access door; disconnecting one connector containing gas in, gas out, and power; and actuating a quick-release fastener. The unit requires no interface, other than mounting, with the Apollo Command Module.

In the independent mode, the Biological Sampling Unit will be operated once per day during the prelaunch period. Coolant from a ground source will maintain adequate temperature control of the stored samples during this period. During flight, coolant from the space radiator will be cold enough to maintain refrigeration. Thermal lag is relied upon for the ascent phase. Prior to recovery, the life cell is depressurized, and the retrieving astronaut removes an access hatch and actuates a spring-loaded guillotine to sever the tape and seal off the coolant passages on the recoverable unit. A quick-release clamp is then opened allowing the cassette of stored tape to be removed. The cassette is stowed in the Apollo command module. Refrigeration is not initiated again until the unit has been returned to earth. This relatively short exposure to a moderate temperature is not a problem. Following return to earth, the Agar sections will be analyzed to establish microbial air contamination as a function of time throughout the experiment.

For other than the independent mode, the Biological Sampling Unit will be placed in operation only after the animal is inserted.

Effect on system parameters. – Each of the above additional experiments will have an impact on the weight, power, thermal requirements, and DMS of the basic prime experiment. These impacts are discussed below.

RLSS: The RLSS occupies 8.7 ft^3, weighs 170.5 lb, requires an average of 300 w of electrical power and contributes an additional 852 Btu/hr to the radiator heat load. A weight and power statement for the RLSS is included in Table 92.

Interfaces between the RLSS and the life cell consist of air in and out, glycol in and out, and oxygen into the life cell. Other connections are required to the waste water storage system, to and from the Coolanol-35 radiator and a vacuum vent. Power is supplied directly to the RLSS module.

Data from the module include 10 temperatures, 10 pressures, 4 valve positions, one partial pressure, one voltage, and one current measurement. Ground commands are included to position CO_2 removal unit valves, backflush water separators, vacuum desorb the Molecular Sieve, deenergize RLSS functions, and switch-over to the basic thermal and atmosphere control subsystem.

The main commutator provided by the PCMTEA equipment in the prime OPE has sufficient sampling capability to include the additional data from the RLSS experiment.

The digital up-link command system equipment will also be capable of providing the necessary commands to the RLSS. Essentially, one command will be set aside to select either the RLSS or the basic nonregenerative subsystem. This technique will allow all commands previously assigned to the basic system to also be used with the RLSS.

Contaminant Analysis/Chemical Sampling Unit: The total weight of the Contaminant Analysis/Chemical Sampling Unit is 45 lb. It occupies 1.9 ft^3. Peak power is 39 w (28 VDC); average power is negligible. The unit requires commands from the spacecraft programmer to open the vacuum vent, to calibrate the mass spectrometer, and operate

TABLE 92

REGENERATIVE LIFE SUPPORT SUBSYSTEM WEIGHT AND POWER STATEMENT

Item	Qty.	Wt. (lb)	Power (watts) Avg.	Power (watts) Peak
Contaminant, Temperature and Humidity Control Unit				
Filter	1	1.5		
Check Valve	4	0.8		
Fans	4	14.0	15	15
Charcoal Bed	1	46.0		
Temp. Control Valve	1	1.5		
Heat Exchanger/Water Separator	1	4.0		
ΔP Switch	1	0.1		
Water Control Valve	1	0.4	--	5
Plumbing/Ducting		1.0		
Subtotal		(69.3)	(15)	
CO_2 Removal Unit				
Bed Control Valves	4	4.0	--	20
Silica Gel Canisters/ Heaters	2	3.8	90	120
Mol. Sieve Canisters/ Heaters	2	3.0	19	25
Vacuum Desorption Valve	1	0.5		
Coolant Control Valves	4	1.6		
Timer	1	2.0	1	1
Plumbing and Ducting		4.0		
Subtotal		(18.9)	(110)	
CO_2 Reduction Unit				
Desorption Pump	1	3.0	10	10
Check Valve	1	0.2		
pO_2 Sensor/Control	1	5.0	1	1
Solenoid Valve	2	0.5	--	10
CO_2 Accumulator	1	3.0		
Relief Valve	1	0.2		
Ground Test QD	2	0.4		
Mixture Control Valve	1	2.0		
Sabatier Reactor	1	0.6	--	23
Heat Exchanger/Water Separator	1	2.0		
Vent Regulator	2	1.0		
Δ P Switch	1	0.2		
Water Control Valve	1	0.4	--	5
Pump	1	1.5	10	10
Plumbing/Ducting		3.0		
Subtotal		(23.0)	(21)	

Item	Qty.	Wt. (lb)	Power (watts) Avg.	Power (watts) Peak
Coolanol-35 System				
Service QD	2	0.5		
Pump	1	7.0	10	10
Coolant Control Valve	1	0.4	--	5
Temp. Control Valve	1	0.4		
Accumulator	1	1.0		
Plumbing/Ducting		2.0		
Subtotal		(11.3)	(10)	
Electrolysis Unit				
Electrolysis Cell	1	10.5	121	121
Oxygen Accumulator	1	3.0		
Check Valves	2	0.5		
Oxygen Pump	1	3.0	10	10
Suction-Pressure Reg.	1	0.4		
Electrolysis Controller	1	5.0	2	2
Relief Valve	1	0.4		
Pressure Regulator	1	0.4		
Plumbing/Ducting		3.0		
Subtotal		(26.2)	(133)	
Catalytic Burner				
Sorbent Beds	2	3.5		
Burner	1	2.0	10	10
Plumbing/Ducting		0.5		
Subtotal		(6.0)	(10)	
Packaging/Mounting		(15.8)		
Total		(170.5)	(300)	

Note: As a result of using the RLSS, weight and power changes occur in the basic thermal and atmosphere control system as follows:

Item	Wt. (lb)
Charcoal	-46.0
Valves	+1.5
Plumbing/Ducting	+1.0
Net	-43.5

Item	Power (watts)
Low-Flow Fans	-25
Net	-25

it in the "monitor" and "scan" modes, and to open and close gas sampling valves. Ground commands are provided to override the programmed "calibration," "monitor" and "scan" modes, and to change the program as required.

The mass spectrometer associated with the Contaminant Analysis/Chemical Sampling Unit will require the addition of a switching matrix to the input of the PCMTEA. The mass spectrometer, when operated in its scanning mode, will require a one channel sampling rate of approximately 2200 samples per second. This rate can be achieved with the PCMTEA commutator by paralleling the 22 channels having the 100 sample per second capability. Since these same 22 channels, or a portion thereof, will be required to sample other data points, the DMS will require the addition of a PCMTEA commutator input switching matrix. The switching matrix will, effectively, permit the time sharing of the PCMTEA capability. It should be noted, however, that while the PCMTEA is programmed to sample the mass spectrometer in its scanning mode, other simultaneous measurements requiring the 100 sps capability will have to be forestalled or assigned to other channels with different sampling rates. The proposed input switching matrix could perform the necessary reassignment.

It is anticipated that the operation of the Contaminant Analysis/Chemical Sampling Unit (less than five minutes per day) will not cause any schedule interference with either the prime OPE or the RLSS experiment. Digital up-link commands required by the unit will, however, necessitate the addition of extended decoding/encoding capability to the on-board equipment

Biological air sampler: The entire Biological Air Sampling Unit weighs 24 lb, occupies 0.6 ft^3, and requires a maximum of 10 w of 28 VDC power. The average power consumption is negligible due to the extremely low duty cycle. The on-board programmer must initiate operation once per day. There is no output data from the unit.

Addition of the biological air sampler experiment to the prime OPE would have a very minimal effect on data management. There is no output data, and only the addition of a start point (once per day) to the onboard programmer is required for implementation.

Weight, power and heat rejection summary: To integrate additions to the prime experiment, certain penalties must be paid. The electrical power system and thermal rejection systems must be augmented to handle the increased loads shown in Table 93.

When the electrolyzer is being used on all AAP cluster versions of the OPE, makeup water will be required if reclaimed water is used for drinking.* This is not the case for the independent spacecraft version. No provision exists for water potability analysis in this mission mode, therefore the water cannot be consumed. The reclaimed water is considered suitable for use in the electrolysis system, however, as an added precaution it is passed through an ion exchange column to render it neutral or slightly basic.

*A check of water potability at 3 months via a sample returned by the astronaut from the AAP cluster will determine if it is suitable for drinking.

TABLE 93

ADDITIONS TO PRIME EXPERIMENT

Module	Weight (lb)	Average Power Req'd (w)	Thermal Rejection Req'd (Btu/hr)
Regenerative Life Support Subsystem	170.5	300	852
Contaminant/Chemical Sampler	45.0	1	3
Biological Air Sampler	24.0	0	35
Total Additions	239.5	301	890

These weight penalties, together with the basic weight of the modular additions, must be accommodated in the case of the independent spacecraft, by offloading expendables. However, if the appropriate expendables are offloaded, the mission need not be foreshortened. For instance, the regenerative life support system (RLSS) performs the same function as the oxygen module, and the lithium-hydroxide beds. If only these expendables are offloaded and the regenerative life support system does not fail, the mission will proceed for a full year.

The weight penalties for modular additions to the independent spacecraft are:

Electrical power (301 w) (0.714 lb/w)	215.0
Thermal rejection (890 Btu/hr) (0.082 lb/Btu/hr)	73.0

and for the AAP cluster are:

Electrical power (301 w) (1.28 lb/w)	385.0
Thermal rejection (890 Btu/hr) (0.066 lb/Btu/hr)	58.7
Electrolysis water makeup (0.362 lb/day) (180 day) (1.1) (2)	143.2

Thus, a total of 527.5 lb of expendables must be offloaded to maintain an independent spacecraft gross weight of 5500 lb. This is equivalent to shortening the prime mission by 205.5 days. Thus, the mission could be reduced to 173.5 days (5.72 months) less ground launch hold time if the RLSS fails on the first day. However, if the RLSS continues running for 205.5 days, then the original mission duration is assured.

Weight summary of modular additions incorporated in OPE

Independent Spacecraft Version	5500.0
Docked Version	5385.8
Prime experiment docked version	5086.9
Modular additions	239.5
Electrical power penalty	385.0
Thermal rejection penalty	58.7
Less offloaded expendables	-527.5
Electrolysis water makeup	143.2
MDA/AM Version	4745.5
Prime experiment MDA/AM version	4446.6
Modular additions	239.5
Electrical power penalty	385.0
Thermal rejection penalty	58.7
Less offloaded expendables	-527.5
Electrolysis water makeup	143.2

OWS Version	4720.5
Prime experiment OWS version	4421.6
Modular additions	239.5
Electrical power penalty	385.0
Thermal rejection penalty	58.7
Less offloaded expendables	-527.5
Electrolysis water makeup	143.2

Experiments in Place of Prime Experiment

Investigations were conducted to explore methods of obtaining more detailed measurements of the biological reaction of primates to the space environment. Because, in the prime experiment, the primates are unrestrained, the number of direct biological measurements possible is restricted to those which can be obtained by means of surgically implantable short-range transmitters. Even here one is limited to the number of channels of information that can be obtained, because the transmitters themselves displace the various organs somewhat and can easily result in localized pressure necrosis and possibly death of the primates.

It is, therefore, believed that surgical implantation of individual transmitters for ECG, temperature, and respiration, and of a backup activity-indicator magnet, constitutes the maximum burden that the primates can accommodate simultaneously, and even this is subject to empirical evaluation. Substituting other parameters, such as neck-muscle EMG and EEG, for those presently selected, would, perhaps, be possible, but requires considerable development which does not appear justified at this time.

Because the primates in the prime experiment are unrestrained, no method is feasible to separate urine and feces for possible biochemical analysis. Neither can any blood be obtained for analysis. If an entirely new experiment is developed in which restrained primates – preferably of much larger size than those selected for the prime experiment – are employed, considerably more biological parameters, in addition to those selected for the prime experiment, could be obtained. Therefore, in place of the prime experiment of two unrestrained Rhesus monkeys, two new experiments to obtain more extensive biological data are postulated; these are:

- An 8-month duration flight of four restrained 13-lb Rhesus monkeys
- An 8-month duration flight of two restrained 40-lb chimpanzees

Both experiments are within the life support and metabolic capabilities of the basic system described for the prime experiment due to the combined effect of a shorter mission duration (8 months versus 12 months) and a reduced metabolic rate for restrained animals (1.1 versus 1.5 BMR). The four-Rhesus experiments, in addition to providing more data than is available from the prime experiment, also provides more statistical meaning than the prime experiment by doubling the number of test subjects. The two-chimpanzee experiment provides a better probability of successful mission completion in that the larger animals can more easily accept the physical burden of the more complex bio-instrumentation required. As in the prime experiment, both new experiments would be returned to Earth for postflight analysis at the end of the mission.

Bioinstrumentation. – For both of the above new experiments, instrumentation is provided to obtain the following types of measurements:

- Neurological
- Cardiovascular

- Respiratory
- Temperature
- Hematologic
- Urinary
- Endocrine
- Metabolic
- Behavioral

A block diagram of the cardiovascular, pulmonary, neurological, metabolic and behavioral measurement system is shown in Fig. 181. All of the instrumentation is described in subsequent paragraphs.

Neurological measurements: By means of surgically implanted cortical and depth electrodes, terminating in a subminiature multipin socket attached to the primate's cranium, it will be possible to obtain a record of the animal's cerebral electrical activity or electroencephalogram (EEG). It is considered feasible also to run two fine-insulated wires subcutaneously from this socket to one of the posterior neck muscles such as the left or right semispinalis capitis, to obtain an electromyogram (EMG).

Apart from the ability to determine the level of cerebral activity, such as alertness, drowsiness, and sleep, and to monitor such states as hypoxia, and carbon dioxide toxicity, from the EEG records by themselves, the neck muscle EMG is a good indicator of muscular activity and relaxation. It is recommended that the cortical electrodes include at least one frontoparietal and one parietooccipital lead, although leads from both hemispheres are preferred for comparative purposes.

Another important measurement in this category is the electro-oculogram (EOG) to measure the position of the eyeball within its socket. A recent publication describes a method to surgically attach permanent electrodes which are located at the outer cauthi of the eyes to measure horizontal eye movement in animals. These electrodes are attached through the temporal bone and are mounted flush with the internal surface of the eye sockets, where they pick up changes in the projected corneoretinal potential difference as a result of eye movement. There is a possibility that vertical eye movements can also be obtained in a somewhat similar fashion, or maybe by means of simple screw-type electrodes attached close to the supra- and infraorbital borders of one of the eyes. The usefulness of this electrode configuration would have to be assessed in the laboratory because of the distinct possibility of severe muscle artifacts of larger amplitude completely obliterating the corneoretinal signal. If this turns out to be the case, only the horizontal component of the eye movement will be taken.

It is recommended that the leads from the EOG electrodes run subcutaneously to the multipin socket on top of the cranium. Each one of the EEG, EMG, and EOG electrode pairs requires a separate high gain differential preamplifier with high common mode rejection, located as close as possible to the primate. The main amplifiers can be located at any convenient location in the vehicle.

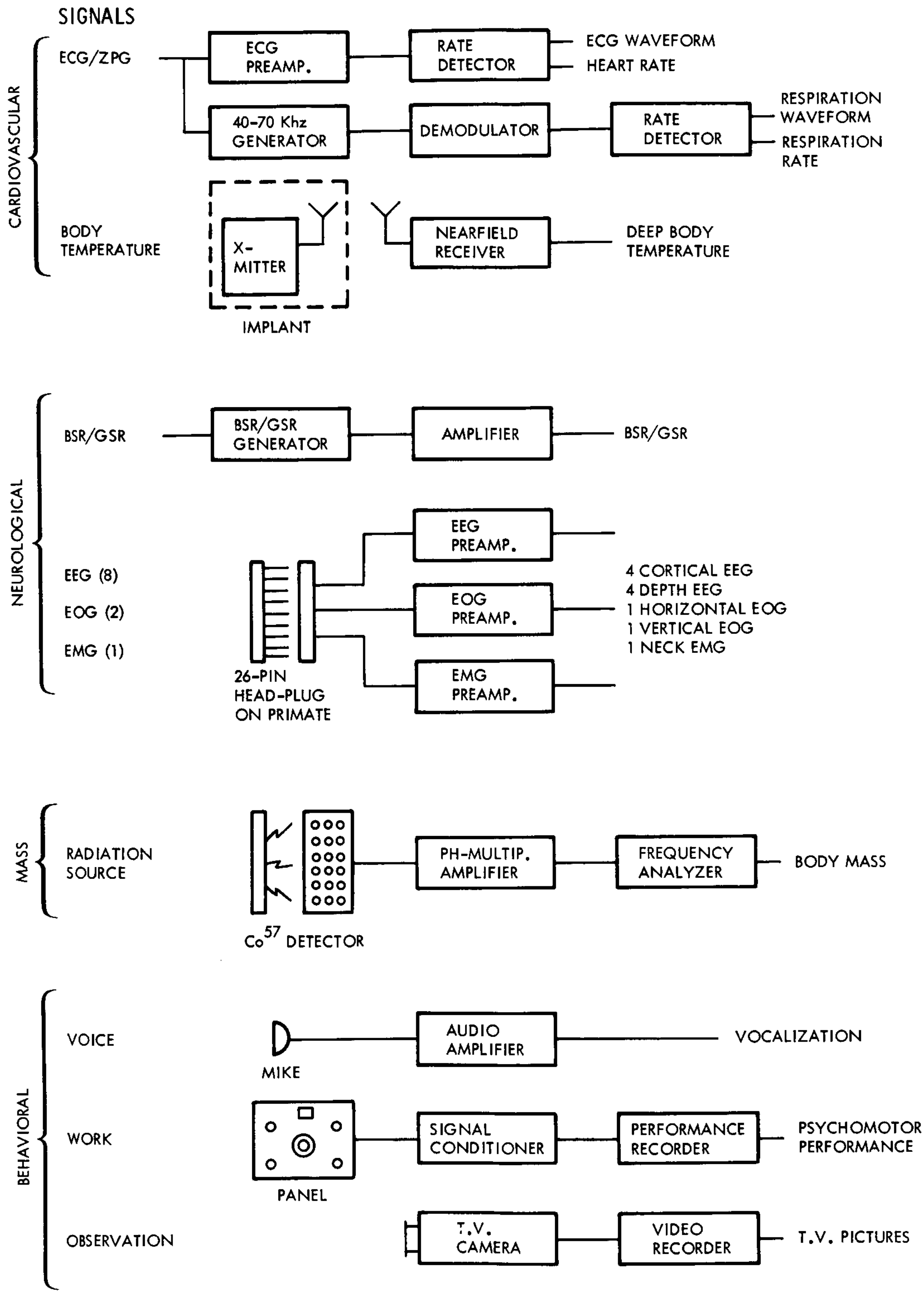


Fig. 181 Cardiovascular, Pulmonary, Neurological and Behavioral Measurements System Diagram

The EEG and EMG preamplifiers will be a.c. coupled (0.1 - 30 Hz, and 30 to 300 Hz, respectively), while the EOG preamplifier(s) are d.c. coupled (0 - 50 Hz). Signal input levels for EEG cover a range from 2 to 200 microvolts, EMG from 25 to 5000 microvolts, and EOG from 50 to 3500 microvolts, but the upper limits for the latter two signals could be limited to 1 millivolt as no precise quantitative measurement is required, and the large majority of signals will fall below this level.

A parameter of psychophysiologic interest in stressful environments is provided by measuring changes in electrical resistance of the skin. Two different components must be distinguished for a more complete evaluation of this phenomenon. They are the relatively steady-state base resistance (Basal Skin Resistance, BSR), which is a function of mental alertness, and the small short-term transient changes (Galvanic Skin Responses, GSR), which occur as the result of inner body and environmental stimuli. Both are modulated by impulses emanating from the autonomic nervous system. They are of particular interest when viewed with simultaneously recorded EEG, EOG, heartrate, and bloodpressure.

In primates, these parameters can be acquired most successfully from electrodes attached to the foot soles. In view of the long mission duration, these electrodes will consist of form-fitted "socks" of conductive silver-cloth, instead of the metal plate electrodes commonly used in comparatively short-term laboratory or orbital experiments. To maintain a uniform electrode contact area, the silver-cloth electrodes will be covered with custom-made stockinets.

The DC-coupled BSR, and the AC-coupled GSR signals require two separate on-board recording channels. Following delayed or real-time transmission of these signals to Earth, they can be analyzed in their analog form, or by means of an automatic GSR analyzer, such as the unit developed by N. R. Burch, et al., at Baylor University College of Medicine.

Cardiovascular measurements: The ECG will be obtained in a modified Lead I configuration from two small electrodes, subcutaneously implanted bilaterally at the level of the fifth or sixth costal interspaces in the mid-axillary lines, with a reference electrode located elsewhere. The ECG signal will be carried by hardline directly to an ECG amplifier/signal conditioner, from which both the full ECG waveform and the cardiac rate can be obtained, and no telemetry link is required. This system provides the most reliable method of actual ECG recording.

The use of a completely unattached system, a so-called field-effect monitor, for the acquisition of cardiac electrical activity, where the sensing element is located in the backrest of the primate chair, offers the distinct advantage of avoiding the infection hazard. This method, developed by Dr. W. A. Shafer at General Dynamics, San Diego, may shortly be evaluated for astronaut monitoring by MSC, Houston. This measurement method does not provide a classic ECG, but the waveform so obtained resembles the ECG in certain respects. The time relationship between the various complexes can be accurately measured, even though the waveform itself shows considerably more individual characteristics than the ordinary ECG, and has, therefore, been called a "signature" waveform. LMSC recommends that this method of cardiac electrical activity measurement be studied further since the observations of changes in the signature waveform are of considerable value. This feature is not included in the designs described herein.

While the recording of a phonocardiogram would be a very desirable measurement, it is believed that the attachment of a cardiac microphone would prove to be very difficult. This could only be accomplished by means of a rather tight-fitting elastic band or harness to keep the microphone correctly located. Even if the microphone were provided with a foam plastic ring, the continuous pressure would eventually cause local tissue necrosis, which could also occur at other locations along the band or harness. Furthermore, the persistent pressure of the band or harness on the animal's ribcage would result in changes in the respiration waveform which could then not be differentiated from those possibly taking place as the result of weightlessness and other spaceflight stresses.

The insertion of a cardiophone catheter in the right ventricle of the heart on a chronic basis cannot be recommended. A possible method would be to surgically implant such a catheter subcutaneously and intercostally over the apex of the heart, with the shielded and teflon-coated connecting cable exiting through a stab wound on the primate's side. However, these shielded cables are relatively thick and less flexible than those used for the ECG. This possibility would require considerable laboratory experimentation before it could be recommended.

By means of indwelling arterial and venous catheters in the femoral artery and the saphenous vein, for instance, a permanent A-V shunt can be established. This offers the possibility of obtaining a full arterial pressure waveform indicative of systolic and diastolic blood pressures. It is recommended that a Statham miniature blood pressure transducer (Model SP 37) be built into the multipurpose blood cuvette to be discussed below. This very recent development weighs only four grams, has a total volume displacement of 12×10^{-4} mm^3/mmHg, requires an excitation voltage of 7.5-v ac or dc, gives an output of 50 μV/V/cmHg ± 1%. It functions over a temperature range of 0 - 200°F, and has a pressure range of 0 - 300 mmHg.

Periodic in-flight calibration will have to be provided. This will consist of two servo-driven stopcocks, which temporarily disconnect the A-V shunt from the primate's circulatory system, and an automatic pressure cycler, providing a pressure pulse of 300 mmHg into the disconnected section of the A-V shunt upon ground command. Since the pressure transducer is incorporated into a multipurpose cuvette (see Hematologic Measurement System) which requires periodic flushing, it is recommended that the calibration procedure is immediately preceded by a relatively high velocity flush of the cuvette assembly, either with distilled water or a mild hemolytic detergent solution.

To prevent blood coagulation and coating within or on the catheters, stopcocks, cuvette, and electrodes, a small motor driven pump has to inject a very small amount (0.01 ml) of heparin sodiumiin saline solution into the upstream side of the A-V shunt or cuvette at frequent intervals, such as once every two to three minutes throughout the flight. The exact concentration of heparin to be employed, as well as the required frequency of administration on a long-term continuous basis in the Macaca mulatta, have to be experimentally determined in terms of excluding the possibility of spontaneous epistaxis or internal hemorrhages.

Respiratory and miscellaneous measurements: A double pair of subcutaneously implanted electrodes will be used to obtain the transthoracic Lead I ECG, as well as an impedance pneumogram (ZPG). This also provides a measurement of respiration. By monitoring the entire signal instead of only the derived rate information as was proposed for the prime experiment, a complete waveform, indicating depth and duration of each respiratory movement, is acquired. This greatly enhances the diagnostic value of the measurement. As in the case of the hardline ECG, no telemetry link is necessary. The method requires a very stable 40 - 70 Khz sine wave generator/ demodulator with associated coupling networks to separate the ECG and ZPG signals.

It is not considered feasible to make other direct respiratory measurements on the primates. Oxygen consumption will be determined from the oxygen and nitrogen flow meter readings, oxygen and nitrogen supply tank pressures, and partial O_2 pressure in the life cell. The availability of the full respiratory waveform provides a good cross-correlation between the life cell pCO_2 reading and respiration depth and rate.

It is felt that the best method to measure deep body temperature remains the surgically implanted temperature transmitter which requires only a single receiving antenna for each restrained primate. Externally applied disc thermistors only give an indication of local skin temperature, which is of little value. Also, it is very difficult to maintain stable skin contact beyond a few hours or possibly days. The use of a hardline connected implanted thermistor sensor is not recommended as this would require yet another skin puncture.

Because the primates would be almost completely restrained, there is no need to measure gross body activity. Thus, the automatic gain control amplifier and activity counter circuits previously recommended in conjunction with the body temperature transmitter, are not necessary. Also, the teflon-coated implanted magnet and its associated induction coil plus amplifier, proposed as a back-up system for activity measurement in the prime experiment, need not be included. Primate activity will be restricted to movements of the arms and head, and these can be monitored periodically with a single high-resolution TV camera. The high resolution is necessary to clearly observe such factors as skin and hair condition and facial expressions. The second, low-resolution camera included in the prime experiment is not required for the alternate experiment.

Hematologic measurements: It was mentioned previously that the near total restraint of the primates in the alternate experiment offered the possibility of implanting an arterial and a venous catheter which will be used to establish a permanent A-V shunt. Apart from allowing continuous blood pressure measurements in the manner already discussed, the A-V shunt offers the opportunity to perform various in-line hematologic measurements by means of a multipurpose cuvette.

This cuvette is visualized as a rectangular block of inert plastic with a small-bore hole through its longest axis. Perpendicular to this hole are drilled various other holes, most of which have a direct connection with the central hole, while others approach it but do not actually establish a connection with it. Various ion-specific electrodes are sealed into these holes in such a fashion that their sensitive surface is flush with the surface of the central bore. Whereas the cuvette assembly forms part of the A-V shunt, a continuous readout can be provided from each of the electrodes. Each electrode requires its individual solid state reference voltage source and/or amplifier which preferably should be left "on" continuously throughout the flight to maintain maximum stability. The output signals from these amplifiers are fed to a commutator, which periodically and sequentially samples each channel. This information is stored on tape for future transmission to Earth upon ground command via the Data Management System (DMS), PCMTEA, and telemetry link. The necessity of periodic flushing of the blood cuvette assembly was already discussed earlier.

Using the above system, it is possible to obtain the following measurements:

pH
Sodium
Potassium
Chloride
Calcium
pO_2
pCO_2

Ordinarily, the last two measurements are performed on capillary blood, but this is not feasible in the OPE experiment. Nevertheless, the measurement of arterial oxygen saturation is an important parameter in its own right, while the occurrence of more than trace amounts of carbon dioxide in the arterial blood would indicate serious impairment of the alveolar gas exchange mechanism. It is conceivable that such a situation could also develop as the result of the formation of excessive amounts of ionized constituents or free radicals, either in the vehicle atmosphere or directly in the bloodstream, due to long-term exposure to the relatively high levels of ionizing solar and cosmic radiations encountered.

Another important measurement which can be obtained with the multipurpose cuvette is that of the hematocrit utilizing the impedance technique employed in the Yellow Springs Electronic Micro-Hematocrit instrument. The presently used sensor can be mounted in the same way as the ion-specific electrodes, and only the electronics have to be repackaged and possibly miniaturized. The necessity of hematocrit measurements was amply borne out in the Mercury and Gemini Programs where significant changes were observed in the astronauts.

An approximation of total blood volume can be accomplished by injecting an accurately premeasured volume of T-1824 (Evans Blue) into the bloodstream upon ground command. The color concentration after ten minutes of circulation is compared at a wavelength of 620 mμ against the transmittance immediately prior to the injection. The dye disappears slowly from the blood, so that an interval of four weeks should be observed between two measurements. By comparing the light transmittance in-flight against a pre-flight determination on the experimental primates, performed several weeks prior to launch, as well as against those of control primates of the same initial weight, fairly accurate measurements of blood volume changes are possible. It is not believed that the Evans Blue injection will interfere with the other measurements, but this must be experimentally determined, so that correction coefficients for these other measurements can be established if significant changes should be found.

The single colorimeter channel can also be incorporated in the cuvette block. A small light source is positioned on one side of and perpendicular to the central bore, while a photoelectric cell is situated directly across the light on the opposite side of the bore. It is suggested that a second photocell be mounted immediately adjacent to and oriented towards the light source. This photocell is part of a simple solid state circuit which automatically compensates for any decrease in light output due to aging of the light bulb. Such a circuit was developed a few years ago by Lagerwerff.

A modification of measuring RBC survival by means of Cr^{51} tagging appears quite possible for inclusion in the OPE, if the small additional radiation burden is not considered to constitute a deterrent. To minimize the complexity of the on-board instrumentation, it is recommended that the injection with the prepared whole blood be given as shortly before launch as possible at the launch complex. Assuming the RBC survival time in the Rhesus monkey to be approximately equivalent to the average of 120 days in man, this would leave a period of more than three months, during which the decreasing radio-activity in the primates' blood can be followed while they are in orbit if no decrease in RBC survival takes place.

In ordinary clinical practice, measuring RBC survival requires the drawing of periodic blood samples, the radio-activity level of which is the determined in a well-type scintillometer/frequency analyzer. This method is too complex to be considered for the OPE in view of the required automation of the procedure. The method for measuring the radio-activity of the primates' blood is suggested in the following paragraphs.

Instead of periodically measuring the blood's radio-activity from drawn samples, it is measured continuously "in-line," utilizing the A-V shunt. Before or after the blood passes the cuvette mentioned above, it flows through a shielded detector made up of scintillating plastic material, and containing a small photomultiplier tube. The radio-activity emitted by the Cr^{51} labelled RBCs causes this plastic to scintillate; this phenomenon is observed, amplified, and converted into electrical pulses by the photomultiplier and its associated circuitry. To further enhance the sensitivity of the detector block, all sides are silver mirrored except the area in contact with the photomultiplier.

A single channel frequency analyzer, passing only the pulses caused by the Cr^{51} emission, feeds these pulses to a binary counter, which counts these pulses for periods of 60 min and transfers the total count at the end of each hour to the DMS.

After the previous hour's count has been recorded, the binary counter automatically resets to zero, and starts the count for the next 60-min period. The counting periods are two to six times longer than is customary, while the total volume of blood passing through the detector during each 60-min period is several hundred times larger than the 10 or 20 ml samples used for counting radio activity in clinical practice. This will result in very high counts during the first few weeks, but as the number of remaining Cr^{51} labelled RBCs gradually decreases, then by the same token the accuracy of the measurements will be appreciably enhanced.

Should the high initial counts require more than an eight-bit binary word — the maximum that can be accommodated by the DMS — an extra storage capacitor to increase the time constant of the counting circuit can provide any desired attenuation factor. Upon ground command, this capacitor can be disconnected from the storage register to increase the counting sensitivity to maximum during the latter part of the mission. Simultaneous measurements on the control animals, compared against those performed by the conventional method, can be used to translate the counts obtained by this new technique.

The argument is valid that the volume of blood actually passing through the detector is not constant and varies as the result of such factors as stroke volume, heart rate, pressure, and particulate content of the blood. However, assuming that no clogging of the A-V shunt occurs, the near total restraint of the subjects is apt to minimize changes in these parameters very considerably, while the continuous counting procedure recommended will tend to average out minor fluctuations in the hourly counts when viewed against the overall duration of the experiment. RBC survival must be considered to be so important a measurement that even if performed with a method that may have several shortcomings, it is better than not to be able to take the measurement at all, or to use a technique which, due to its complexity, has a very low reliability factor.

Hematologic measurements system: A diagram of the hematologic measurements system is presented in Fig. 182. The position of the timer-controlled, motor-driven stopcocks S1 and S2 is shown in the "normal" position, whereby the arterial blood passes through the blood cuvette assembly, where the various measurements are taken. Timer 1 activates micropump 1 every 2 to 3 minutes to dispense a small amount of heparin solution into the inlet side of the system, via check valve V1. This check-valve, as well as check valves V3 and V4, are set to operate at a pump pressure in excess of 250 mmHg to prevent blood from entering the various pumps and reservoirs. Once every 2 to 4 weeks, pump P3 is activated upon ground command to dispense 1 ml of 0.25% indigo carmine solution solution into the blood stream, to measure renal permeability. The readout for this test is accomplished with the photo-electric cell/threshold detector included in the urinary measurement system (see Fig. 183).

Pump P4 is activated by ground command once every 4 to 6 weeks to measure total blood volume. Upon activation, P4 dispenses 1 ml of 0.5% Evans Blue (T-1824) solution into the blood stream. After 10 to 15 minutes, the concentration of dye in the blood is determined from the output of the 620 mμ colorimeter cell included in the blood cuvette assembly.

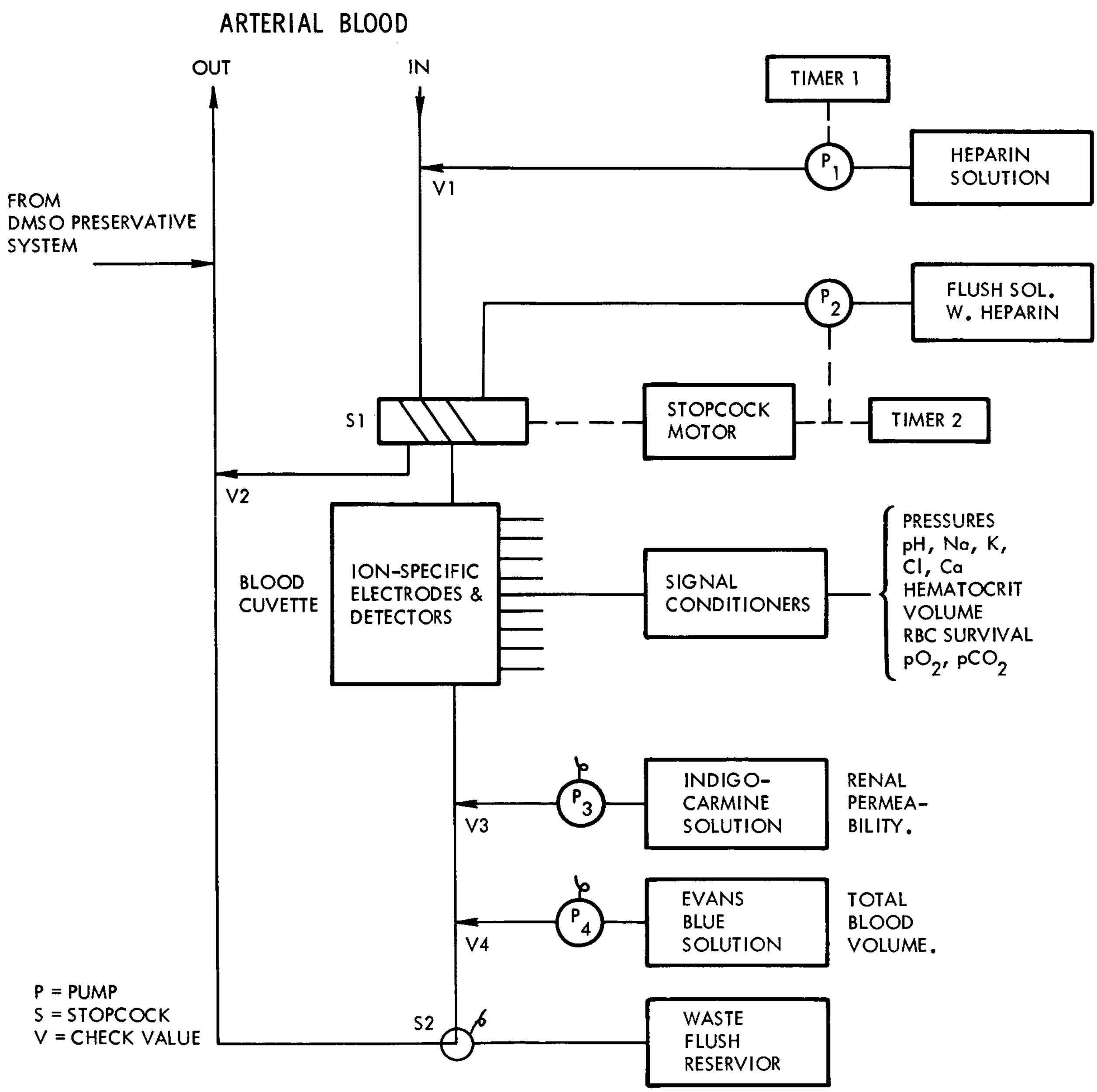


Fig. 182 Hematologic Measurements Block Diagram

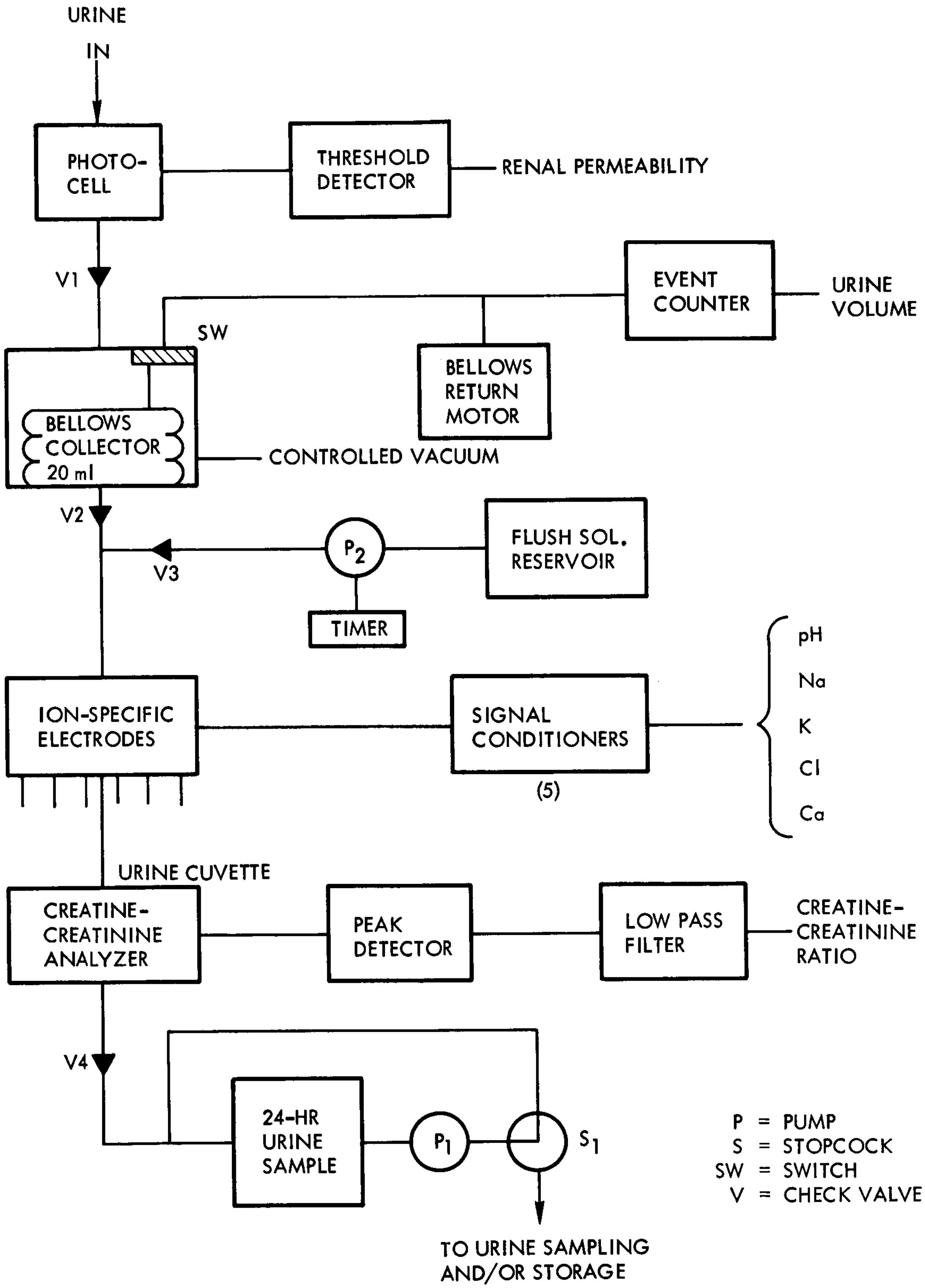


Fig. 183 Urinary Measurements Block Diagram

Timer 2 operates motor-driven stopcocks S1 and S2 and pump P2 in sequence once every week. When S1 is turned through 180 deg to the "flush" position, the incoming blood bypasses the blood cuvette assembly via check valve V2, set to operate at a pressure above 100 mmHg. Simultaneously, a flow path is established between P2 and the blood cuvette assembly, while the downstream side of the latter is opened to a waste flush storage reservoir via S2. Pump P2 will then flush the blood cuvette assembly for 10–15 seconds under a pressure of 5 $lb/in.^2$ with fluid from the flush solution reservoir.

Following the flushing, P2 will be shut off and S1 and S2 are returned to the normal position, placing the blood cuvette assembly on-line again.

Urinary measurements: Using the same principle of the multipurpose cuvette discussed in the preceding chapter, the following urinary measurements appear feasible:

pH
Sodium
Potassium
Chloride
Calcium
Renal permeability

In addition, the measurement of urine volume, and Creatine-creatinine ratio are also possible, but require special instrumentation.

Chronic urine collection can most readily be accomplished when male primates are used. A retention catheter can be inserted, the bulb of which is located in the surgically enlarged urether at the level of the prostate. This prevents kinking of the distal end of the catheter, which so frequently occurs during chronic experiments in the laboratory when the bulb is located entirely within the bladder. This same difficulty is also encountered when female monkeys are catheterized, because the shortness of the urethra makes positioning of the bulb inside the bladder mandatory.

The catheter terminates into a 10–20 ml volume bellows-type receptacle, located in a small airtight housing, maintained at a partial vacuum. The receptacle is provided with two one-way valves. Every time the receptacle is full, a microswitch activates a piston, driven by a small electric motor, which flushes the contents of the receptacle through a multipurpose cuvette. Simultaneously, this flushing activity generates an event pulse, which is stored by the DMS on tape for later transmission, thus providing an accurate measure of urinary volume. After having passed through the cuvette, the urine flows either into a waste storage reservoir together with germicide to prevent growth of microorganisms, or is diverted to a cryogenic urine sample preservation device at periodic intervals.

During the time that the urine is flowing through the cuvette, ion-specific electrodes measure pH, sodium, potassium, chloride, and calcium; this information is simultaneously recorded on the data tape.

The measurement of renal permeability is performed as follows: At periodic intervals upon ground command, a small premeasured amount of indigo carmine solution is injected into the primate's bloodstream by means of a special motor driven pump attached to the A-V shunt assembly. To measure the time interval between the injection and the first appearance of the dye in the urine, a simple light-source photocell-detector is located upstream from the urinary volume measurement assembly. As soon as the dye appears in the urine, a marked decrease in the photocell's output will take place. Both the exact injection time and the time of the dye's appearance can be recorded.

To obtain uniformity of measurements, it is recommended that a threshold detector be employed to record the latter event, i.e., that the dye concentration must reach a certain minimum value before the time of this event is recorded. This method obviates the need to make a continuous record of the photocell's output during the intervening time period. Although this measurement is very simple, it must be ascertained that the indigo carmine dye does not degrade the performance of the ion-specific electrodes, and does not cause discoloration of the detector cell wall.

The determination of the creatine-creatinine ratio will be made with an instrument similar to that developed by JPL for the NASA Biosatellite Program (NASA Tech Brief 67-10245). This instrument is completely automatic in operation and can probably be used without modification. It is recommended that the instrument be placed downstream from the urinary cuvette in order to prevent damage to the ion-specific electrodes because the instrument uses hydrochloric acid as one of its reagents. The actual measurements are performed colorimetrically at a wavelength of 490 $m\mu$.

To remove deposits, the entire urinalysis system has to be flushed periodically in the same manner as the blood analysis system.

Urinary measurements system: Figure 183 represents the diagram of the proposed urinary measurements system. The first sensor consists of the photoelectric call/threshold detector for the measurement of renal permeability after injection of indigo carmine into the bloodstream described previously.

The urine passes through one-way check valve V1, set to operate at a pressure of 5-10 mmHg, into the bellows-type urine volume measurement system. The bellows, having a volume of 20 ml, is contained in an airtight chamber, which is maintained at a slight and constant negative pressure. When the bellows is filled, it will activate a microswitch which will register one count on the event counter and, simultaneously, turn on the bellows-return motor. This motor thus empties the bellows via one-way check valve V2 set to operate at a pressure of 100 mmHg.

The urine now flows through the urine cuvette assembly, where pH, Na, K, Cl, and Ca, measurements are taken, and then into the creatine-creatinine ratio analyzer. After this measurement has been taken, the processed urine passes through check valve V4 to the 24-hr urine sample tank. Pump P_1 circulates the urine to mix the sample and, at 24-hr intervals, discharges the urine to sampling and/or storage through valve S_1.

Once a week the timer activates pump P_1 which flushes the cuvette assembly for 10 to 15 seconds with a cleansing solution from the flush solution reservoir via check valve V3 set to operate at pressure of 150 mmHg. The pressure to be delivered by pump P_2 is approximately 5 lb/in^2.

Endocrine measurements: In the previous discussion, mention was made of a low temperature urine sample preservation device. This device becomes necessary because at this time no OPE-adaptable methods can be recommended to perform any of the endocrine measurements in orbit. It is, therefore, necessary to return preserved urine samples to Earth for laboratory evaluation.

If accurate endocrine measurements are to be made, it is necessary to determine the total output on a 24-hr basis, since the concentration of the different hormones in the blood and urine may vary widely throughout the day. For this reason, a 24-hr mixing tank is provided upstream of the urine samples.

The availability of urine samples also enables verification of several of the in-flight measurements taken from the same urine samples, a fact which greatly enhances the value of the entire experiment. One 15 ml urine sample can be taken and stored every week.

Once the urine samples have been retrieved, the number of endocrine measurements that can be performed on these samples will be restricted to those hormones that can survive the long-term low-temperature storage, and will also be restricted by the relatively small sample volumes.

Metabolic measurements: Only a limited number of gross metabolic measurements can be performed on board the OPE. These include water and food intake, urine volume, oxygen consumption, and body mass.

Water and food delivery amounts are known on the basis of actuation of the food and water lip devices. When the primate takes a drink of water from the bite-activated dispenser as a reward, it can be assumed that this amount is actually ingested. Such is not necessarily the case with the food pellets, however. After having taken a pellet from the food dispenser, the monkey may decide to play with it, rather than to consume it. In the process of such play, the food pellet may float or be thrown away. To minimize food wastage, the daily ration should be limited to a fraction above the minimum daily requirement.

As has been shown in the section on urinary measurements, the urine output can be measured with sufficient accuracy.

It is not considered feasible to measure fecal output in flight. Only by returning all fecal material to Earth, can the total fecal output during the flight be determined. However, feces can be sampled in flight, frozen, and returned to Earth for post-flight analysis.

As previously discussed under Respiratory Measurements, the measurement of metabolic oxygen consumption can be determined indirectly.

The mass measurement system will be essentially the same as that recommended for the prime experiment. This system is based on the change in nuclear radiation transmission through the primate's body as its mass changes. In the seated configuration visualized for this experiment, the radiation detectors will be incorporated in the seat and back area of the restraint chair or couch. The accuracy of the system is expected to fall within the 1 to 3 percent range.

In the area of metabolic measurements, a series of interesting radio-isotope studies, other than the Cr^{51} labelled RBC survival, are considered feasible. These consist of ingestion during the flight of specially prepared food pellets containing specifically labelled constituents, followed by whole body radiospectrometric scans and autoradiographic analyses of individual tissues following retrieval of the primates. Even if the animal should die in orbit some time after ingestion of these special food pellets, the preserved cadaver would yield a host of valuable information regarding various metabolic processes.

To determine dietary utilization of labelled amino acids (S^{35}), the neurosecretory cells and the pyramidal cells of the brain's cortex require post-flight analysis by means of either tissue extraction and gas flow counting of protein or by autoradiographic technique.

Turnover of calcium in the bones can be determined by using Ca^{45} in the food pellets. Pre- and post-flight bone X-ray densitometry, supported by post-flight autoradiography, of the os calcaneus, for instance, will determine the overall directionality of calcium ion movement over the entire flight period.

Labelled Vitamin E, C^{14} normally accumulates very markedly in bone marrow and in the red pulp of the spleen. Subnormal amounts of radioactive carbon in these organs postflight would therefore be indicative of disturbed Vitamin E uptake.

The localization of the very stable Vitamin B12 molecule in most endocrine glands (except the adrenal medulla) under Earthbound conditions, invites the use of cyanocobalamin Co^{58}. Vitamin B12 also localizes itself in the renal cortex and in a small segment of the brain stem.

Although the use of these isotopes in flight is highly recommended by LMSC, measures must be taken to prevent their use from possibly interfering with such measurements as RBC survival and mass; this prevention is accomplished by appropriate filters in the frequency analyzers employed for the latter measurements.

This area is suggested for further study as it will yield much more information with a minimum of impact on weight, volume, power, and other system parameters. This feature has not been included in the designs presented herein.

Behavioral measurements: The behavioral task subsystem utilized in the substitute experiments is identical to that recommended for the prime experiment. It has been described in considerable detail in previous sections of this report.

General considerations: Because we are dealing with restrained animals in whom a permanent external A-V shunt and an indwelling urinary catheter have been established, it is mandatory that the primates remain in the restrained condition following surgery, during insertion into the life cell, throughout the flight, during EVA extraction from the life cell, during reentry, and during their transportation back to the ground-based laboratory. To minimize the total volume of blood in the A-V shunt, as well as to circumvent the necessity of repeatedly having to connect and disconnect the in-line equipment, which would increase the risk of emboli formation, all analysis equipment has to be located in close proximity to the primate, i.e., attached to the restraining device.

The electronics for the amplification and signal conditioning of the low-level bioelectric potentials, such as EEG, EOG, neck-EMG, ZPG, and ECG, are also located near the animal to prevent signal loss due to possible electrical interference from other equipment.

The requirements for the urinalysis equipment are less stringent. This equipment can be positioned at some distance away, even though as a matter of principle, lengthy lines should be avoided if possible. This, however, requires the connecting and disconnecting of a quick-connect fitting between the primate's restraining device during insertion and removal of the experiment.

It is visualized that electrical power inputs, signal outputs, and urine and fecal disposal are established automatically when the primate's restraining device is inserted into the life cell and are again broken when the animal is pushed up into the retrieval capsule prior to retrieval. In this manner the astronaut who is to retrieve these capsules during EVA operation is not involved in any additional manipulations. Provisions are made for continued recording of at least ECG and ZPG on a small tape recorder inside the retrieval capsule during pre-insertion and recovery operations.

Because of the presence of the A-V shunt, preservation of a dead animal can be accomplished in a much simpler manner than is the case with unrestrained animals. Upon ground command a relatively small amount of dimethyl sulfoxide and formaldehyde solution is injected directly into the animal's vascular tree on the venous side. The command signal simultaneously opens the preservative supply valve and stopcock S2 (Fig. 182). As preservative solution is being introduced into the animal under pressure, the blood is dumped into the waste flush reservoir.

It is important to realize that although all suggested measurements lie well within the state-of-the-art, presently available laboratory instruments are far too bulky and heavy and consume far too much power to be directly usable for the OPE. A considerable amount of miniaturization and various modifications are necessary to develop suitable hardware. The methods suggested for such measurements as total blood volume, RBC survival, renal permeability, and urine output, although based on existing and/or feasible techniques, require entirely new development efforts. Finally, laboratory experiments will have to be performed to determine the validity of these new measurement techniques against standard laboratory procedures and to draw up cross-correlation graphs between the two methods for each one of these measurements.

The effort, time, and cost involved in this development is substantial and should be started as soon as possible to meet the AAP flight schedules.

The use of much larger primates than the 12 to 14 lb Macaca mulatta, e.g., adult chimpanzees (Pan troglodytes) should result in an increased probability of experiment success. In the small primates, the diameter of even major blood vessels, such as the femoral arteries and the saphenous veins, is relatively small, requiring even smaller diameter catheters for their cannulation. Not only are these narrow bore catheters more fragile and more prone to mechanical damage, including kinking and being closed by external pressure (high g-loads during lift-off and reentry), but they are much more susceptible to sludging and subsequent clogging. Also, fixation of these small catheters is more difficult.

Experiment configurations. – The major design impact of substituting extensively instrumented animals in place of the prime experiment is in the area of the animal confinement instrumentation and support methods. The design approach was to provide lifecell modules in which the animals could be restrained, instrumented, and supported for all mission modes. The configurations of these modules for both the 13-lb Rhesus and the 40-lb chimpanzee are described in the following paragraphs.

Rhesus restraint module: To successfully achieve the degree of bioinstrumentation required, it is necessary to restrain the primate. The restraint module also is designed to be compatible with the retrieval canister both in terms of size and interface.

As shown in Fig. 184, the design concept is based on a pallet formed to the shape of a seated 13-lb primate. The basic framework of the pallet provides a suitable structure upon which the various bioinstrumentation electronic and measurement units are mounted. Urine and feces transport equipment is also mounted to this pallet. This concept allows the primate to be secured and connections to be made between the animal and the bioinstrumentation and waste management transport equipment.

The restraint module is attached to the door which forms the bottom of the retrieval canister. The module slides in place on two round tubes, which are integral with the door, and locked in place. A bayonet socket fitting between the module and the door provides additional stability to the module/door interface.

Urine and fecal plumbing connections are made between the door and the module prior to placement of the retrieval canister shell over the assembly. Electrical and fluid interfaces are made at the upper end of the module between the module and the retrieval canister.

The restraint module is a fiberglass and metal structure. Bioinstrumentation equipment and electronic packages are constructed in modular form for installation on the back of the restraint seat. These modules are designed to plug in, electrically and hydraulically, to a central harness and plumbing system. This central system interfaces with the retrieval canister umbilical lines.

This design concept allows the fully instrumented primate to be installed in the restraint module a considerable time prior to launch. All bioinstrumentation can be checked out and final adjustments made prior to placement of the module in the retrieval canister.

Rhesus retrieval canister: The retrieval canister design follows the same general concept as that used for the prime experiment. However, several changes have been made to provide for the restraint module and the additional bioinstrumentation.

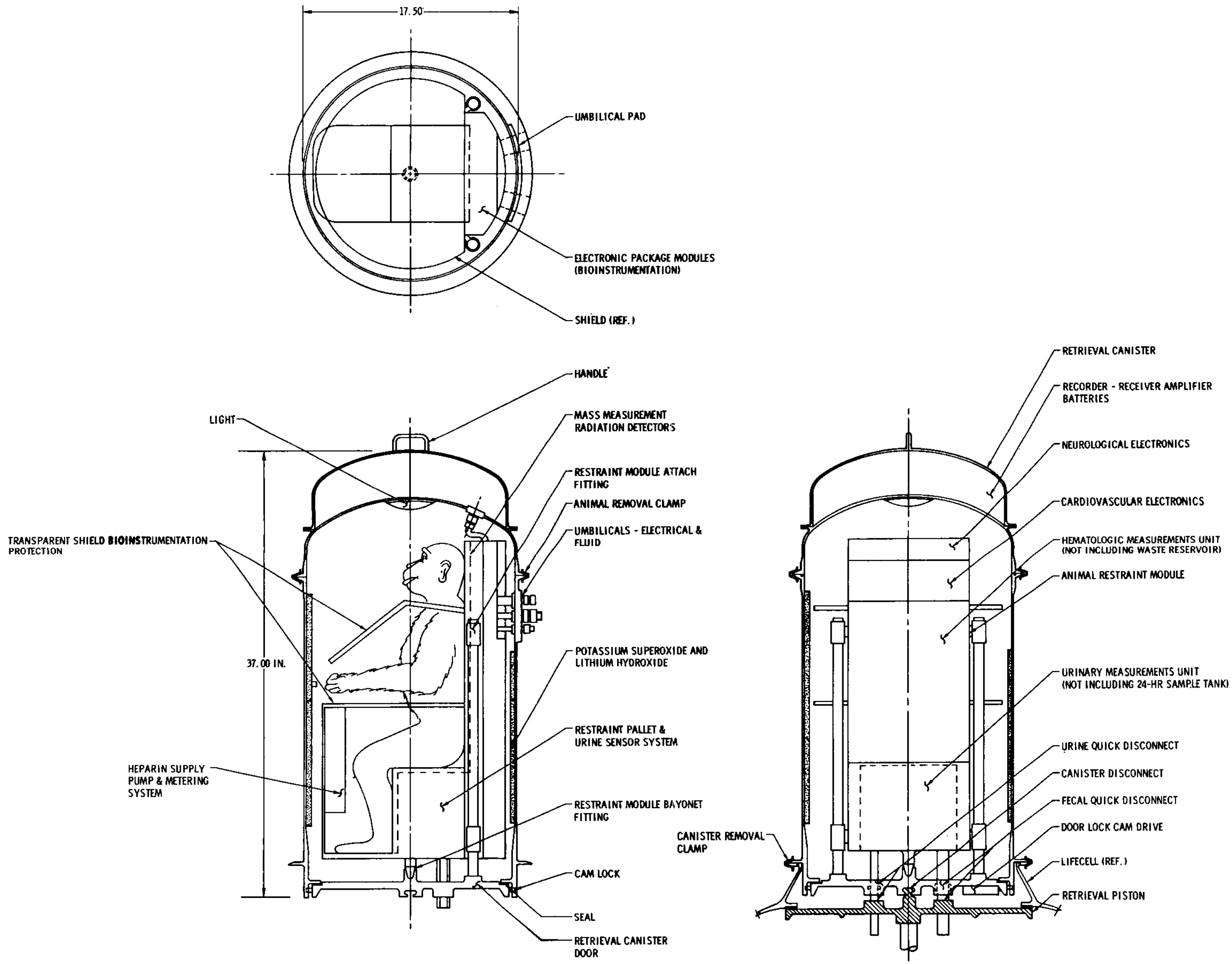


Fig. 184 Rhesus Restraint Module and Retrieval Canister

The basic shape of the canister is maintained, but the overall dimensions have been increased to a diameter of 17.5 in. with an overall length of 37.0 in. The greatest design change is in the retrieval canister door. In most of the mission modes under consideration the retrieval canister functions as an animal transport device for ascent, insertion and return to earth. Consequently, the retrieval canister door must be capable of multiple opening and closing cycles. It also must have integral quick-disconnect couplings for the transfer of feces and urine through the door after insertion into the lifecell.

The door design utilizes a face seal between the door and a lip on the retrieval canister. Sufficient force is supplied by a door-lock cam drive to effect a seal between the two surfaces. The door is closed and locked prior to launch, making the retrieval canister a transport lifecell system.

The amount of potassium superoxide and lithium hydroxide has been increased from a 48-hr supply to a 96-hr supply. This will allow the canister to be carried into orbit onboard the CM, as well as return from orbit ensuring the animal an adequate O_2 supply and CO_2 removal. The use of the hydrophobic screen and moisture absorption material is no longer required because the concept uses a closed waste management system.

A larger umbilical pad is provided to interface with the additional electrical connections. A battery within the heparin pump and metering module supplies power to ensure pump operation while in transit from the medical complex to the launch vehicle and during ascent, insertion, and CM return operations.

Rhesus lifecell: The lifecell for the restrained Rhesus experiment is presented in Fig. 185. The design concept for the lifecell supports the requirements of both the retrieval canister and the restraint module. The lifecell is 26.0 in. in diameter and 50.5 in. long. The upper end of the lifecell is designed to interface with the retrieval canister. The lower portion of the lifecell has a removable dome, providing access to the lifecell interior. The lower dome also contains the fecal storage system, telescoping pneumatic actuator, and provisions for waste product storage and preservation. Provisions are also made for installation of all primate support equipment. These items include the behavorial panel, food pellet dispenser, video monitoring system, radiation mass measurement unit, drinking device, retrieval piston, lighting, and fluid and electrical interfaces.

The Rhesus is installed in the lifecell through the following sequence. The animal is first placed into the restraint module, and all bioinstrumentation equipment is checked out. The animal/restraint module combination is placed into the retrieval canister, all interfaces are connected, and the retrieval canister door is locked in place by actuating the door-lock cam drive. The retrieval canister dome is installed, and the complete system is leak checked. This system is then ready for insertion in the lifecell.

The telescoping pneumatic actuator in the lifecell is energized, causing the retrieval piston to interface with a seat and seal in the upper dome of the lifecell. This action seals the lifecell, forming a pressure-tight vessel. The retrieval canister/restraint module assembly is then aligned with the lifecell and the retrieval canister

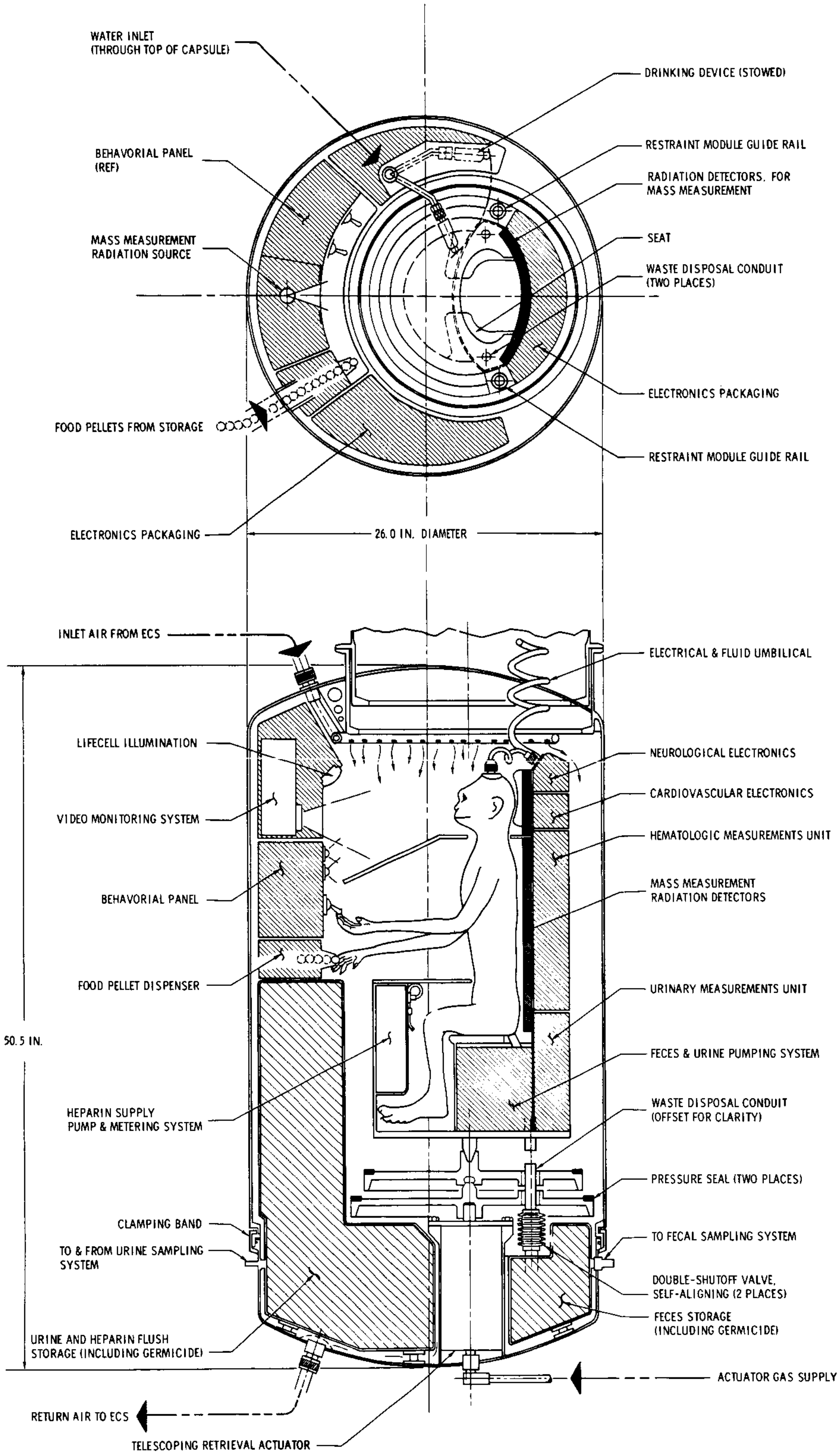


Fig. 185 Rhesus Restraint Module in Life Cell Module

is moved into place. This engages and opens the quick-disconnect couplings between the retrieval canister door and the retrieval piston. A lock between the doors is provided through a ball and socket connection at the door center. The "V"-band coupling is installed to make the retrieval canister/lifecell combination an integral system. Electrical and fluid connections are made at the umbilical pad between the canister and the spacecraft.

The retrieval canister door is then unlocked by actuating remotely the door-lock cam drive. The pneumatic actuator is energized and the restraint module is slowly lowered into the lifecell. When the actuator has nearly completed its full stroke, the fecal and urine outlet tubes interface with their respective storage containers. With the actuator fully retracted, the restraint module is in proper position within the lifecell for interface with the support equipment located within the lifecell. Electrical and fluid lines extend from the umbilical connection in the retrieval canister to the restraint module. A take-up reel is provided for control of these lines during extension and retraction of the pneumatic actuator.

The pellet feeder is located externally to the lifecell with a pressure seal at the lifecell/feeder interface. The pellets are transported to the dispenser on the lifecell interior. The basic feeder is of the same design as for the prime experiment.

Drinking water is stored externally to the lifecell with the lip device mounted on a swivel fitting within the lifecell interior. The swivel permits the device to swing out of the way during the insertion and retrieval processes.

Management of both urine and feces is accomplished by direct connections to the primate. Urine is directed to the analysis, sampling, and storage containers from an indwelling catheter. A germicide is added to the urine prior to storage and the tank is fitted with a relief vent. Fecal control is maintained by utilizing a modified version of the LMSC-developed "chimpants." The chimpants are molded rubber pants which closely fit the primate's body. A tube integral with the pants and centered on the anus, directs feces to a peristaltic pump for transport to the fecal sampling unit and then to the waste storage container where germicide is added to minimize biodegradation. The fecal storage tank has a relief vent.

Urine and feces sampling units: To perform postflight endocrine studies, it is necessary to retrieve suitably stored samples of urine and feces. Urine and feces samples will be taken once a week during the 8-month mission. Provisions are made for one urine and one feces sampling unit for each animal.

The urine sampling unit is shown in Fig. 186. Urine flows through the unit continuously, but is prevented from entering the collection bottles until the outflow shutoff valve is closed and the cam/roller assembly is rotated. Rotation of this unit enables flow to the hypodermic needle and depresses it through the collector septum. Further rotation of the cam and roller forces urine, trapped in the tube above the cam, into the urine container. Approximately 15 ml of urine is collected at each sampling. The urine container is conically shaped to hold the fluid in a zero-g environment. Air in the container vents through a hydrophobic screen, which also prevents any

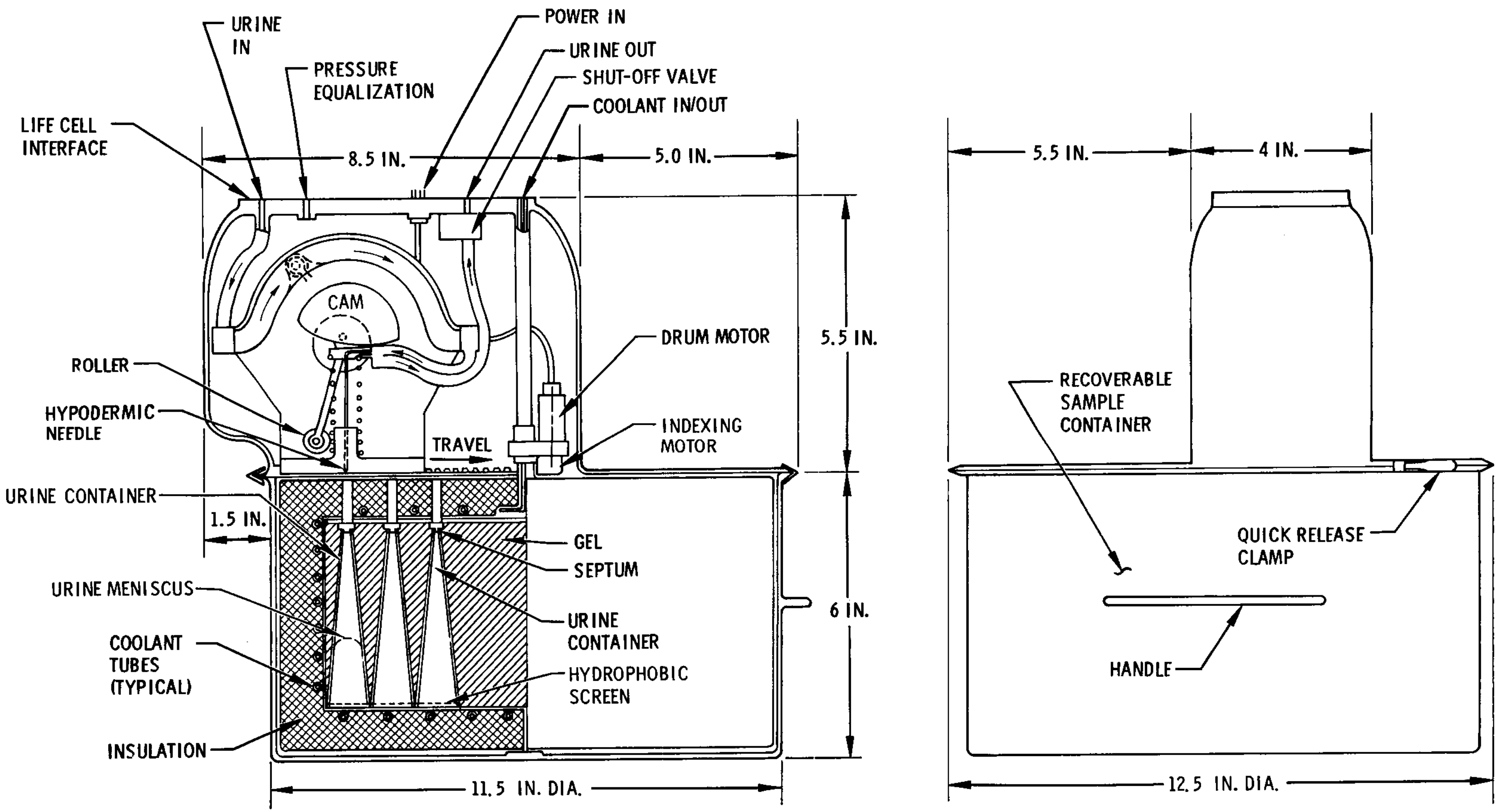


Fig. 186 Urine Sampling Unit

floating urine particles from leaving the container. The urine is frozen rapidly since the collector bottles are maintained at a low temperature. Following the filling of one container, the unit indexes to position the hypodermic needle over the next container. Through rotation of the collector drum and translation of the cam/roller assembly, all containers can be filled.

To obtain a pooled, 24-hr urine sample, the animal's urine output for this period will be stored in a mixing reservoir. This mixed sample will be discharged to the urine storage tank through the urine sampling unit so that when the urine sampling unit is activated it obtains a 24-hr composite sample.

The feces sampling unit uses the delivery pressure of the feces pump to force an aliquot (approximately 15 ml) of fecal material into a cylindrical container located in a rotating drum similar to that used for the urine sample containers. A gate valve in the feces pump discharge line diverts one fecal sample into a cylindrical chamber each time the drum is indexed. The fecal material is stored at low temperature.

The behavioral task subsystem is the same as that used for the prime experiment. However, if a different program is required, sufficient volume is available for additional task and programming equipment.

One high-resolution video camera and sufficient lighting are provided to view the animal above the transparent shield.

Environmental control in the lifecell is achieved by a sufficient flow of air from top to bottom and through insulation of the lifecell from environmental extremes. Oxygen and nitrogen supply and control, thermal and humidity control, and CO_2 and contaminant removal are achieved with the modularized thermal and atmosphere control system. The system concept is identical to that for the prime experiment. The thermal and atmosphere control module contains two separate loops; either one can provide acceptable environmental control for all four primate lifecells.

Chimpanzee restraint module and portable lifecell: The 40-lb chimpanzee restraint module is similar in design to the Rhesus module. While the design concept remains the same, certain changes are made to provide for the larger animal. An inboard profile of this module is shown in Fig. 187. The module is a structural system which provides mounting racks for all bioelectronic and bioplumbing systems. The module will be contoured to fit the body profile of the chimpanzee. A restraint is employed to retain the animal in the couch, with shields to protect the bioinstrumentation. A restraint suit of rachel nylon knit with integral restraint straps is placed over the animal's body and interfaces with restraint buckles integral with the restraint module. The bioinstrumentation, urine 24-hr sample tank/pumping system, fecal sampling and pumping system, heparin supply pump and metering system, and instrumentation and video camera shields are all integral with the restraint module. Structural fittings are provided on the module for attachment to the lifecell structure.

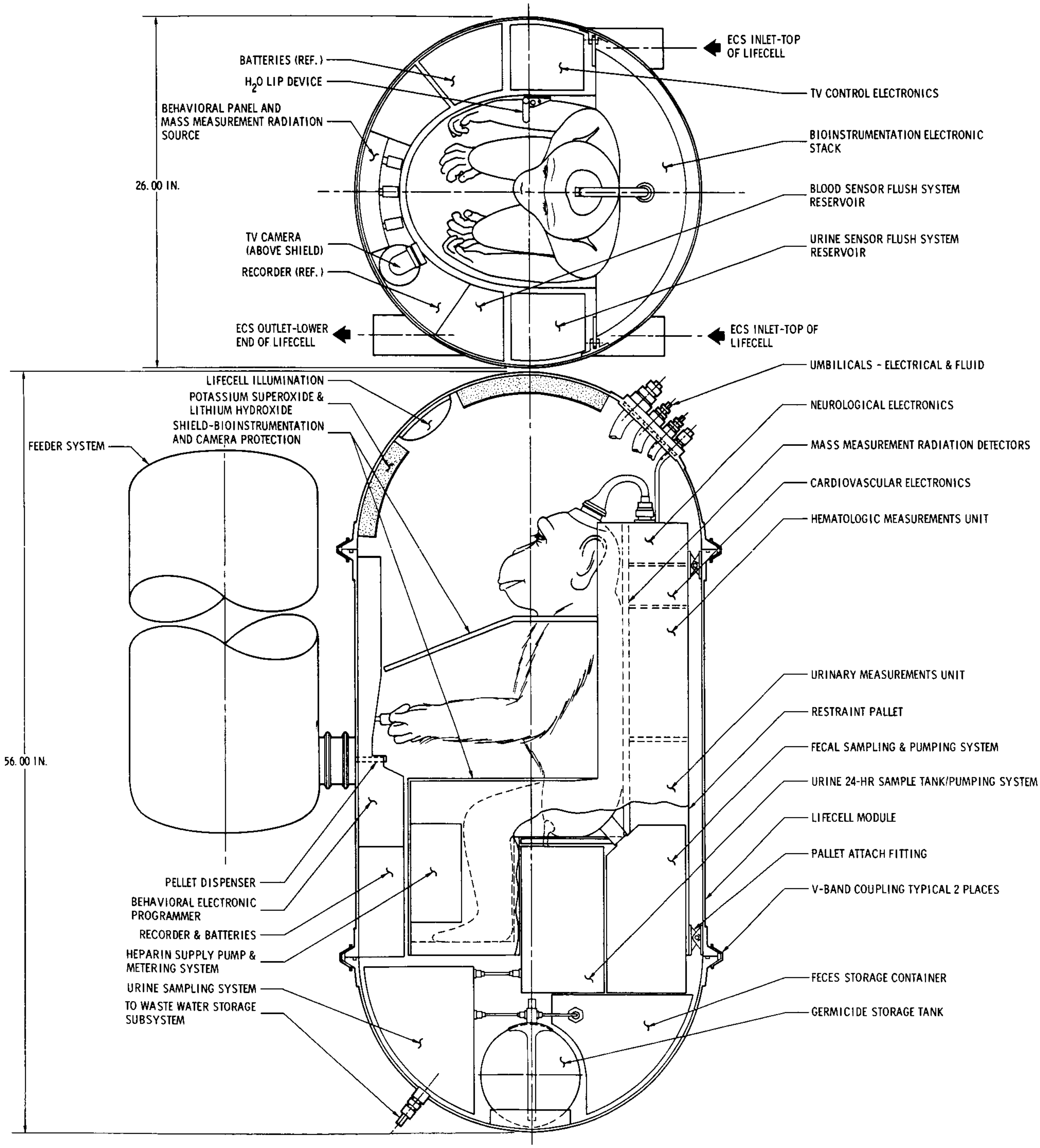


Fig. 187 Chimpanzee Life Cell Module

A portable lifecell houses the primate and the additional support systems. Due to the size of the animal, the retrieval canister concept was dropped in favor of a portable lifecell.

The lifecell is a cylindrical pressure vessel with hemispherical ends. The diameter is 26.0 in. and the overall length is 56.0 in. Machined rings are provided at both ends to retain a silicone rubber O-ring seal and a "V"-band coupling. This provides access to the lifecell interior for installation and servicing of the support equipment and animal insertion.

The upper dome of the lifecell contains the potassium superoxide and lithium hydroxide supply for O_2 generation and CO_2 removal during the ascent and recovery operations. An actuator system will expose or seal off the KO_2 and LiOH to the interior lifecell atmosphere on command. Lifecell illumination is also provided by a light fixture located in the dome. Electrical and fluid umbilicals are also provided.

The cylindrical lifecell section provides mounting provisions for the behavioral panel, mass measurement radiation source, behavioral programmer, urine and blood sensor flush systems, feeder and water supply interface ports, TV camera control electronics, recorder, batteries, and food and water dispensers.

The lower lifecell dome contains the urine sampling system, feces storage container, and germicide storage tank. A port is also provided for transfer of fluid waste products through the lifecell wall to the waste-water storage system located externally to the lifecell.

Mounting lugs are provided on the lifecell exterior for mounting to the spacecraft structure. ECS inlets are provided at the top of the lifecell with an exit at the bottom to provide atmosphere distribution. This permits the fresh atmosphere to enter near the animal's head with the flow leading the lifecell near the waste management system. The dual-loop modular environmental control system serves both lifecells. One loop of this system is capable of supporting both lifecells.

The feeder operation is the same, in principle, as that devised for the prime experiment. However, to keep the lifecell as small as possible, the feeder is mounted externally to the lifecell. An airlock is provided between the lifecell and the feeder. A "V"-band coupling with O-ring seals completes the connection.

The water lip device is mounted to the restraint module and the water supply tanks are mounted externally to the lifecell. The water system is similar in design to that of the prime experiment.

Urine control is achieved by using a catheter to transport urine from the primate to the 24-hr sample tank and pumping system, where it is further transported to the urine sampling system. After sampling, the urine is treated with a germicide and stored in the waste water subsystem external to the lifecell. The fecal control system employs the use of the chimpants for the transport of feces from the primate to the fecal sample and pumping system. The feces is then transferred into the fecal storage container where germicide is added. This design is similar to that described for the restrained Rhesus experiment.

Use of the chimpanzee permits a wide selection of behavioral programs. Space has been provided for a large, multitask behavioral panel and electronic programmer. A typical program may consist of the following components:

- Continuous Avoidance
- Temporal Delay Task
- Match to Sample
- Sample Reaction Time
- Auditory Discrimination
- Choice Reaction Time

These programs can be arranged in many ways to meet the requirements of the principal investigator.

Mission modes. – For all mission modes, two possible approaches were explored for experiments in place of the prime experiment: an experiment involving four 13-lb Rhesus and an experiment involving two 40-lb chimpanzees. In both cases, to counteract a substantial growth in launched weight, the amounts of metabolic expendables (food, water, oxygen, and LiOH) were reduced. As a result, both experiments are planned for a reduced mission duration of 8 months, a compromise which is considered to be outweighed by the increase in biological data to be obtained.

Four-Rhesus experiment: In the independent and docked modes, the four lifecell modules are clustered about the ECS module. The axes of all five modules are parallel to each other and normal to the spacecraft axis (Fig. 188). The other modules also are located within the meteorid shield and thermal protection blanket, and are supported by a tubular structure. For the docked mode, all of the variations from the independent spacecraft which were mentioned in the foregoing section describing the prime experiment in the docked mode are repeated.

When installed in the AM/MDA, the modules are clustered about the exterior of the 65.12-in.-diameter portion of the AM and in the number one and two trusses, as shown in Fig. 189. The lifecell modules are arranged so that the animal transportation canisters penetrate the floor of the structural transition section of the AM. This permits insertion and retrieval of the animal canisters in a shirtsleeve environment. Thermal control and electrical power are obtained by augmenting the OWS systems. The lifecell and ECS modules are necessarily prebuilt on the AM, but all other modules could be assembled in orbit.

When installed in the OWS, the same modules described for the MDA/AM version are used (Fig. 190). The combined arrangement is placed on top of the grid deck that forms the overhead of the waste management and food preparation compartments of the OWS. Since the OWS atmosphere is at 5 psia, a need still exists for pressure-tight modules. Thermal control and electrical power are again obtained by augmentation of the appropriate OWS systems.

For all mission modes, the urine and fecal sampling units are located externally to the lifecells, one urine and one feces sampler per lifecell.

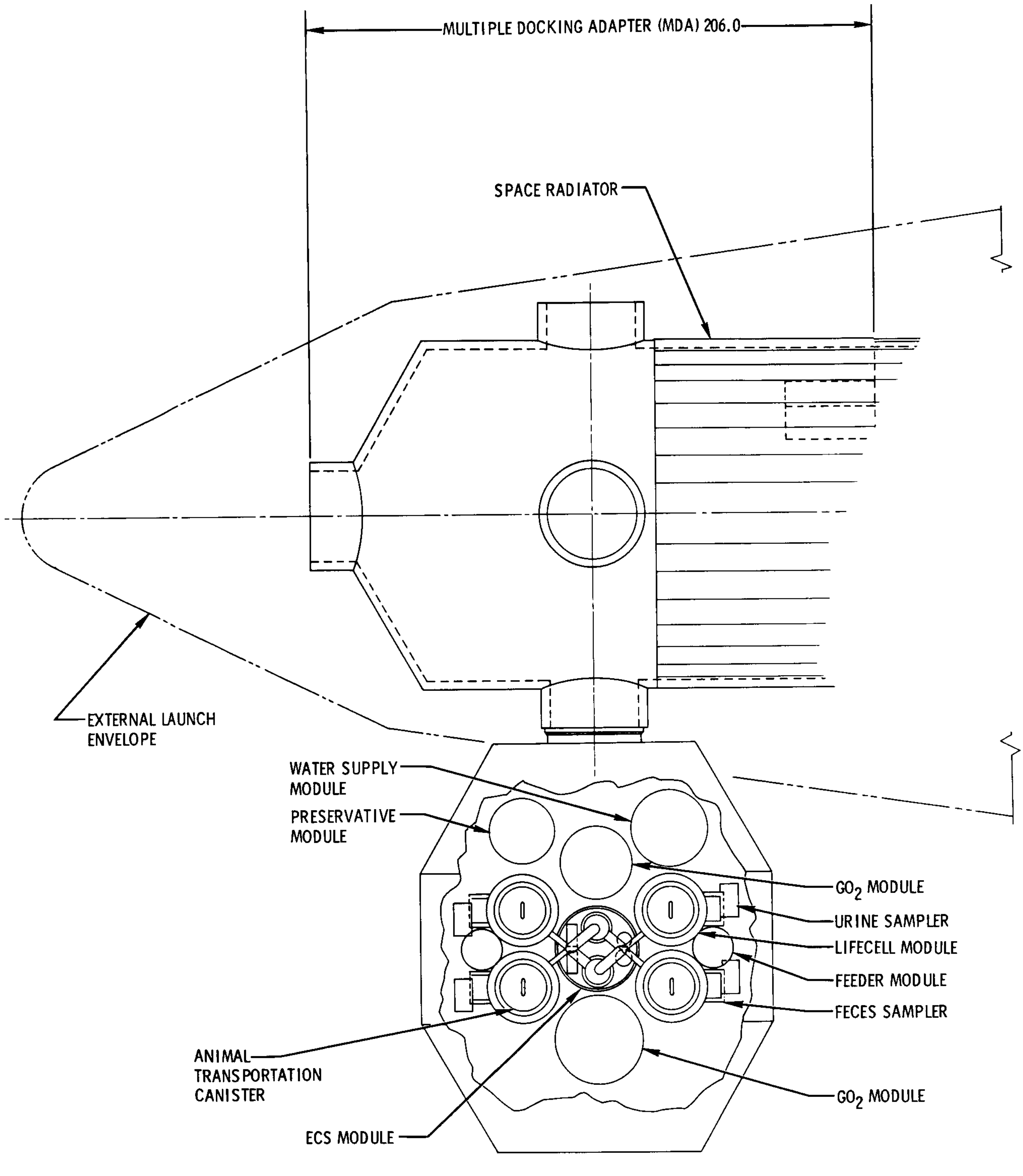


Fig. 188 Four Rhesus Modules in Docked Configuration

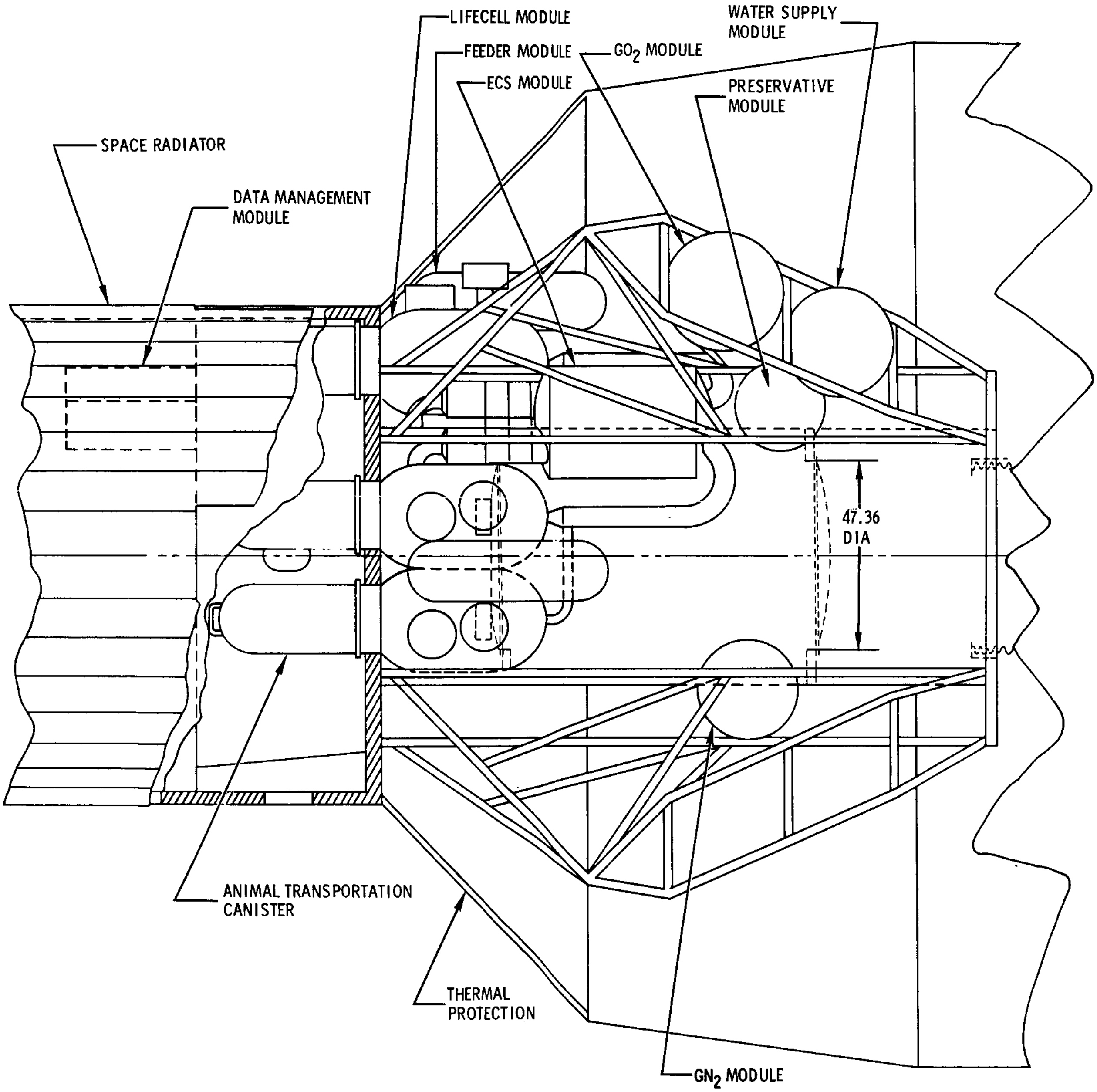


Fig. 189a Four Rhesus Modules Installed in AM/MDA

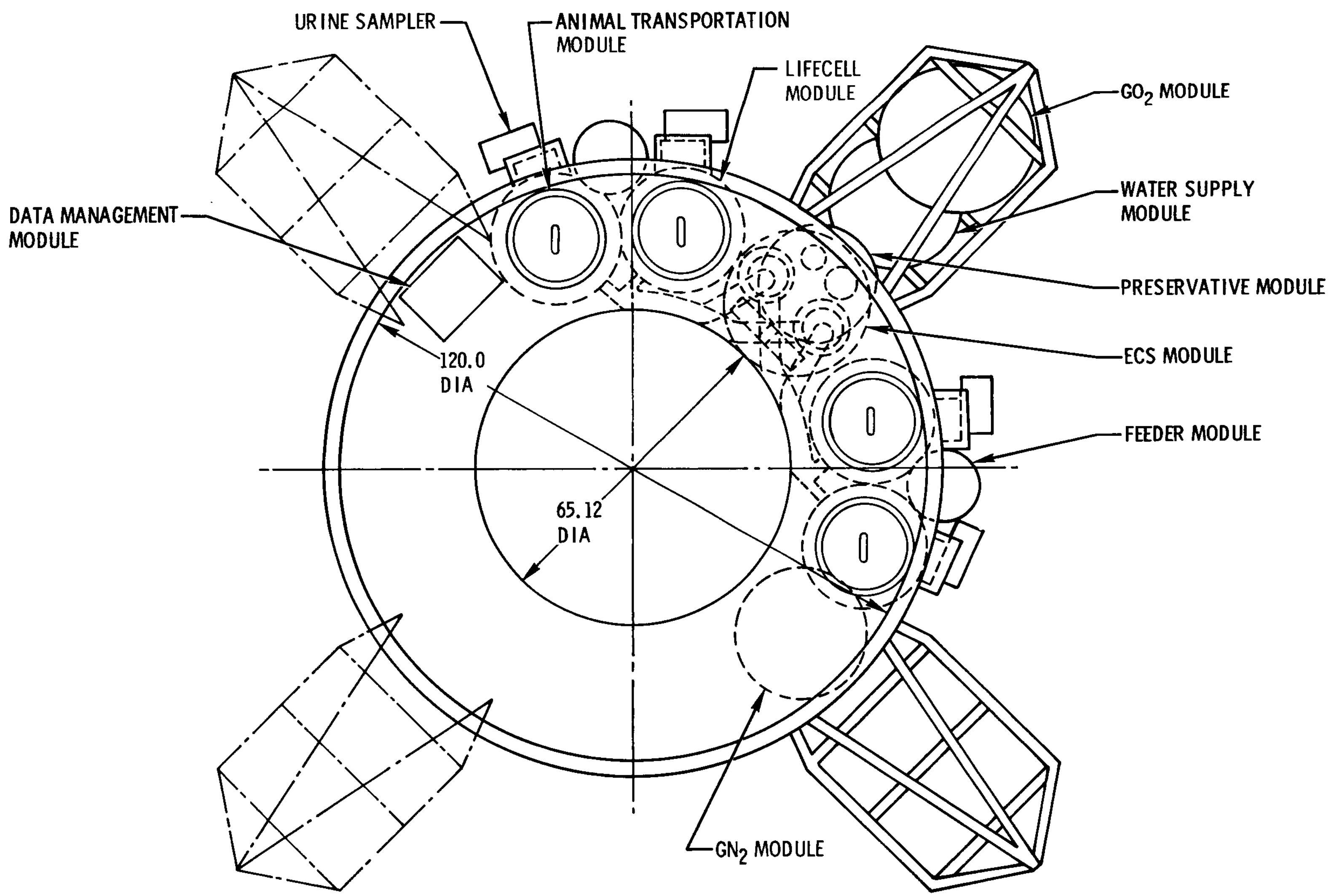


Fig. 189b Four Rhesus Modules Installed in AM/MDA (Cont.)

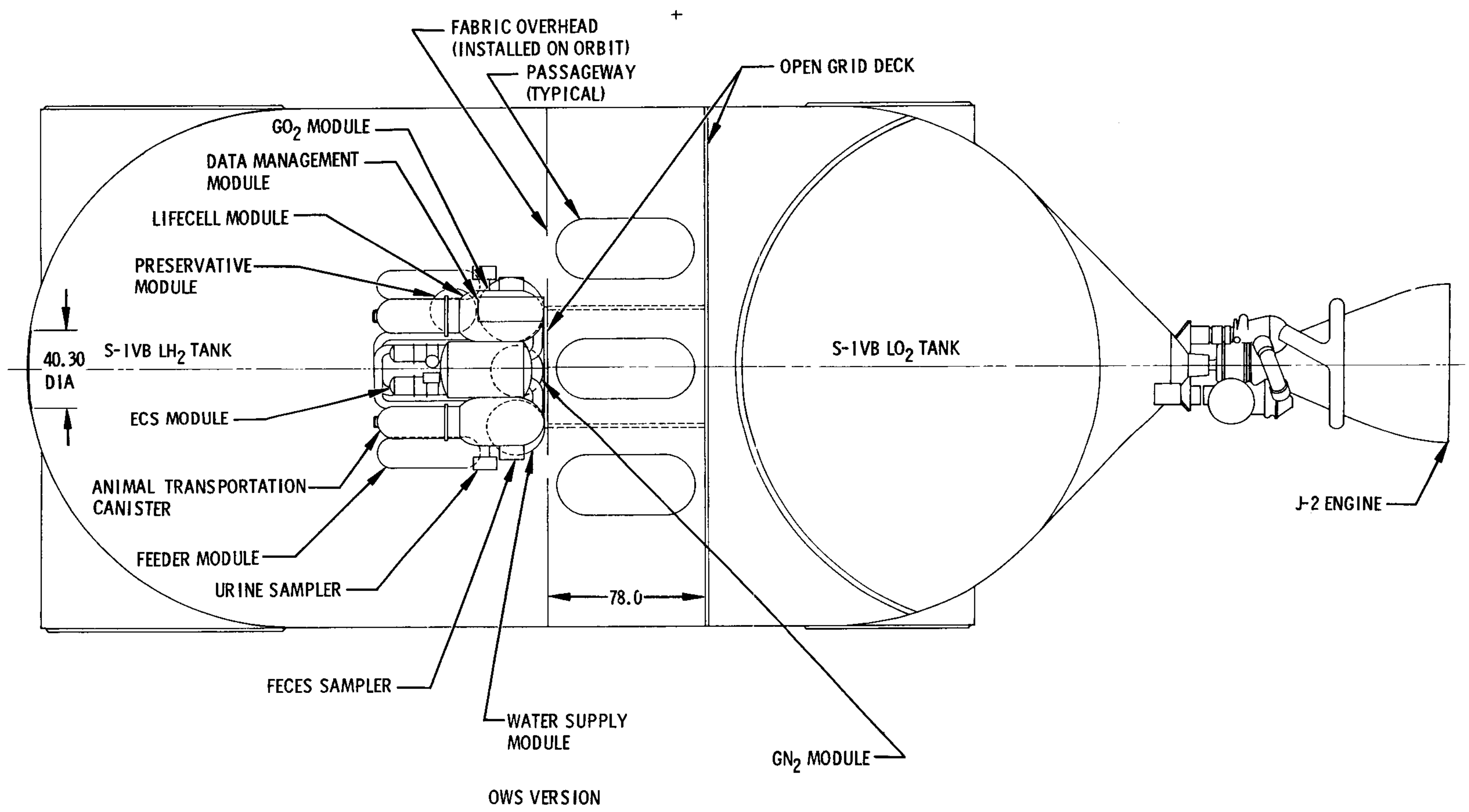


Fig. 190a Four Rhesus Modules Installed in OWS

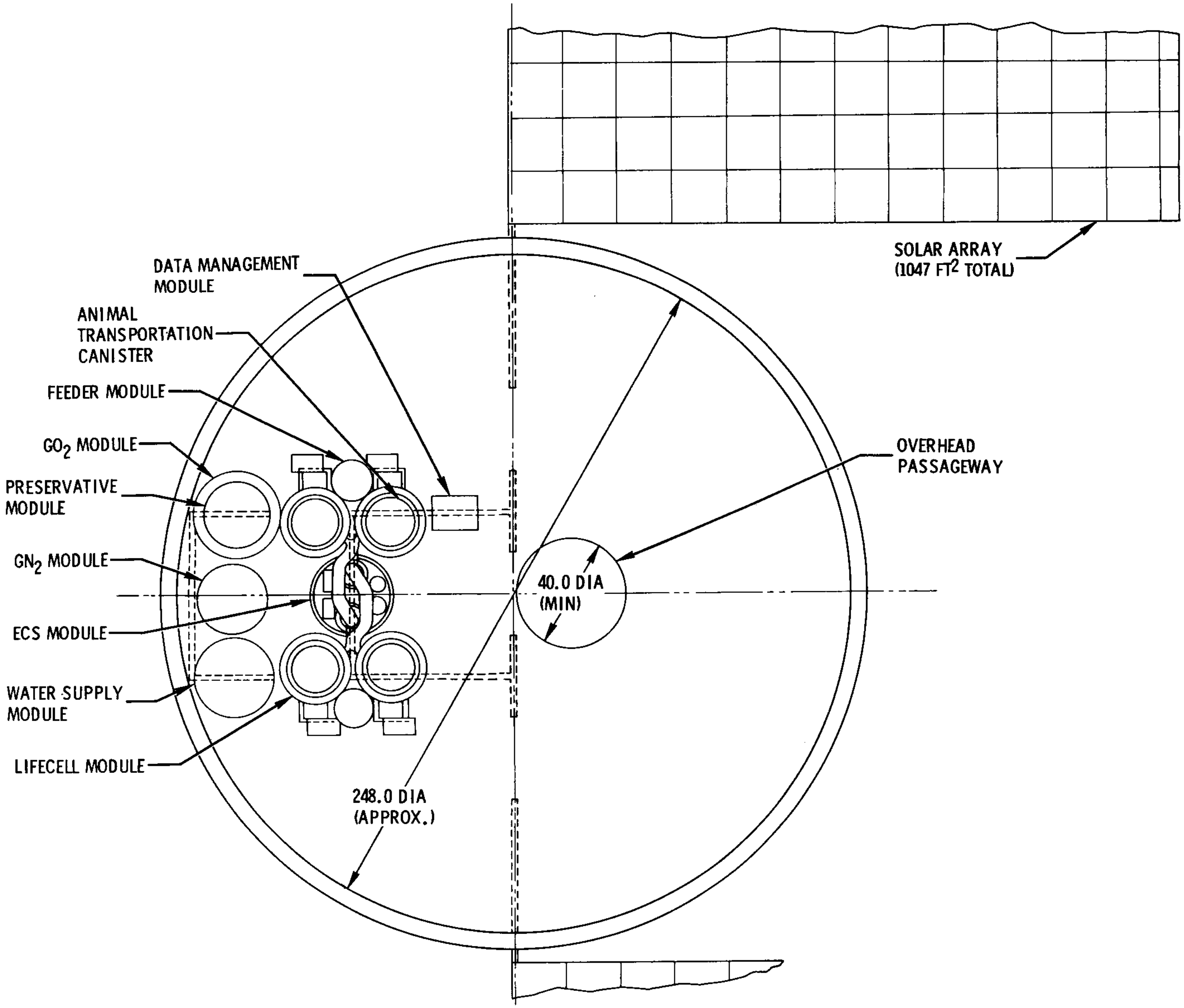


Fig. 190b Four Rhesus Modules Installed in OWS (Cont.)

Two-chimpanzee experiment: For all mission modes, the configuration and placement of the two-chimpanzee experiment are similar to those of the four-Rhesus experiment. In this case, however, each animal is contained in a portable lifecell module, which also serves as the transport container. The layout of the modules for the independent and docked modes is shown in Fig. 191. When installed in the AM/MDA or the OWS, the configurations are as shown in Figs. 192 and 193, respectively. The urine and feces sampling units are installed inside the chimpanzee lifecell, since this lifecell is recoverable.

Operations. – The basic operations for both the four-Rhesus experiment and the two-chimpanzee experiment are identical. In either case, the experiment in the independent mode is launched and placed in a solo orbit in the same manner as the prime experiment.

In the docked mode, the OPE spacecraft is launched in a dormant mode on a manned launch vehicle with no animals installed. The animals are launched in their retrieval canisters, in the case of the Rhesus monkeys, or in the portable lifecells, in the case of the chimpanzees. Both spacecraft and animals could be included in the same launch vehicle if weight and volume constraints of the launch vehicle permit.

After the OWS is injected into orbit, the CSM/OPE spacecraft effects rendezvous and docks the OPE spacecraft to a port on the MDA. The CSM then detaches from the OPE and docks to another port. The astronauts activate all the OPE systems. When the OPE is ready, the Rhesus canisters are inserted into their lifecells in the OPE spacecraft, or the chimpanzee portable lifecells are inserted into OPE spacecraft. At the conclusion of the experiment in the docked mode, the astronaut can interface with the experiment in several ways:

- Replace failed components
- Replace tapes
- Return recorded data to earth
- Return chemical and biological samples
- Return one or all animals
- Bring up new animals

When installed in the AM/MDA, the lifecells and ECS module for the Rhesus experiment are launched with the OWS and the animal retrieval canisters containing the animals are launched on a subsequent manned vehicle. In the case of the chimpanzee experiment, the ECS is launched with the OWS, and the animals in their portable lifecells follow on a subsequent manned launch. The distribution of other supporting equipment modules between the two launch vehicles will depend on prevailing weight and volume availabilities. After the ECS and all supporting modules are assembled and activated, the animals are inserted.

In addition to the interfaces mentioned for the docked version, the astronauts can also perform major maintenance of the experiment by temporarily containing the animals in their retrieval canisters or portable lifecells and replacing or adjusting malfunctioning equipment. Also, photographic film cassetes of the animals could be changed and returned to earth by the simple inclusion of a camera in the lifecell or a window installation for a hand-held camera.

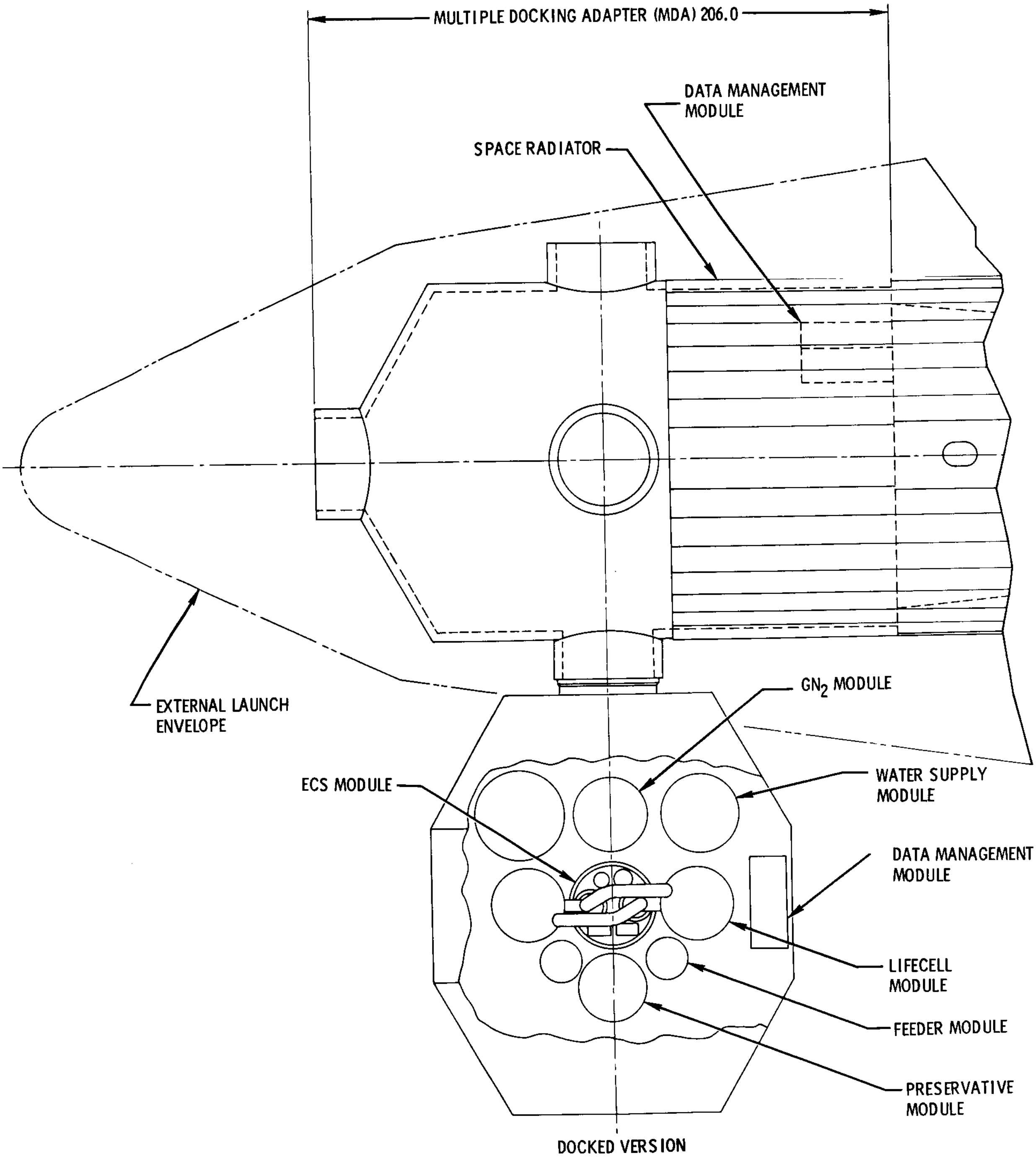


Fig. 191 Two Chimpanzee Modules in Docked Configuration

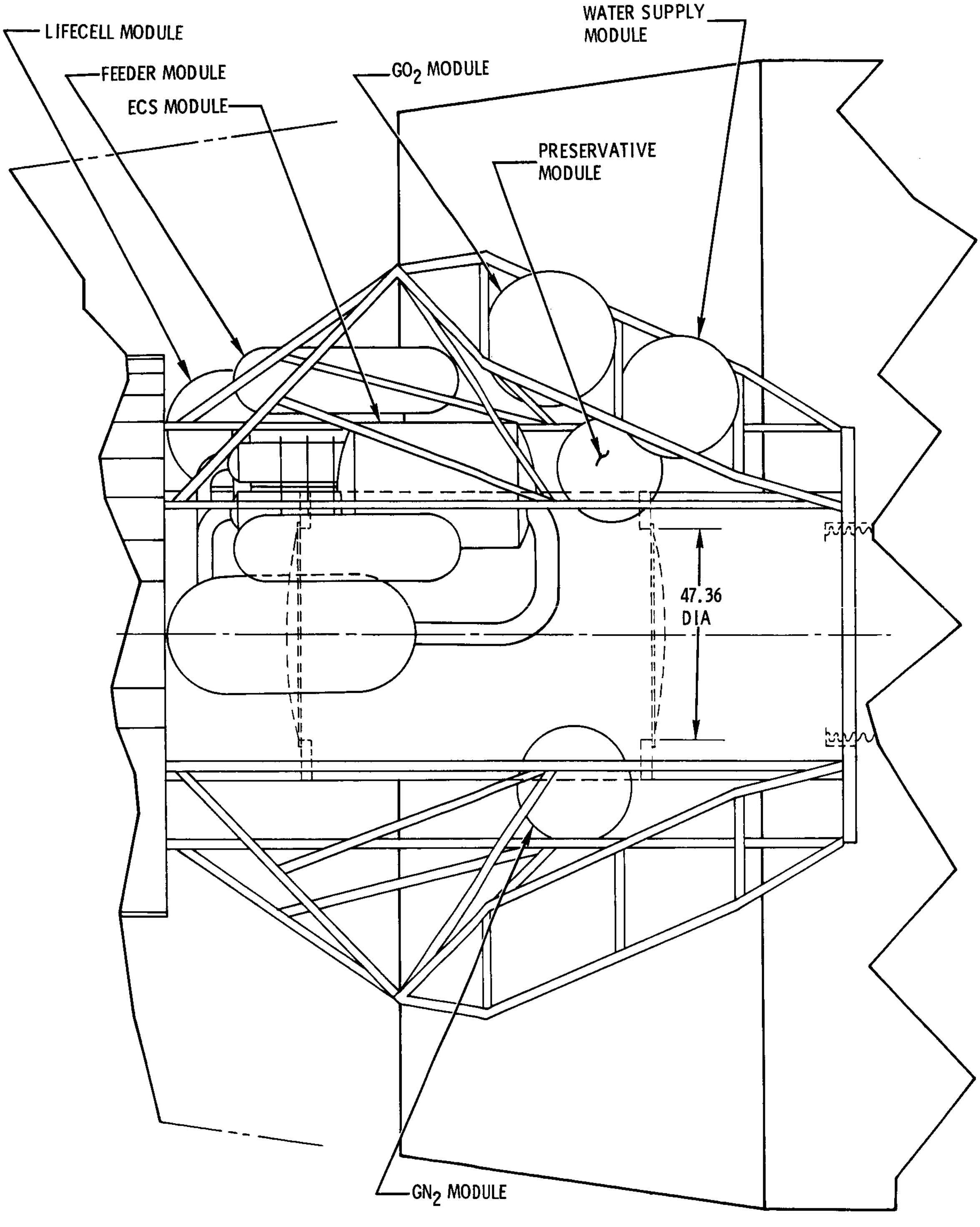


Fig. 192a Two Chimpanzee Modules Installed in AM/MDA

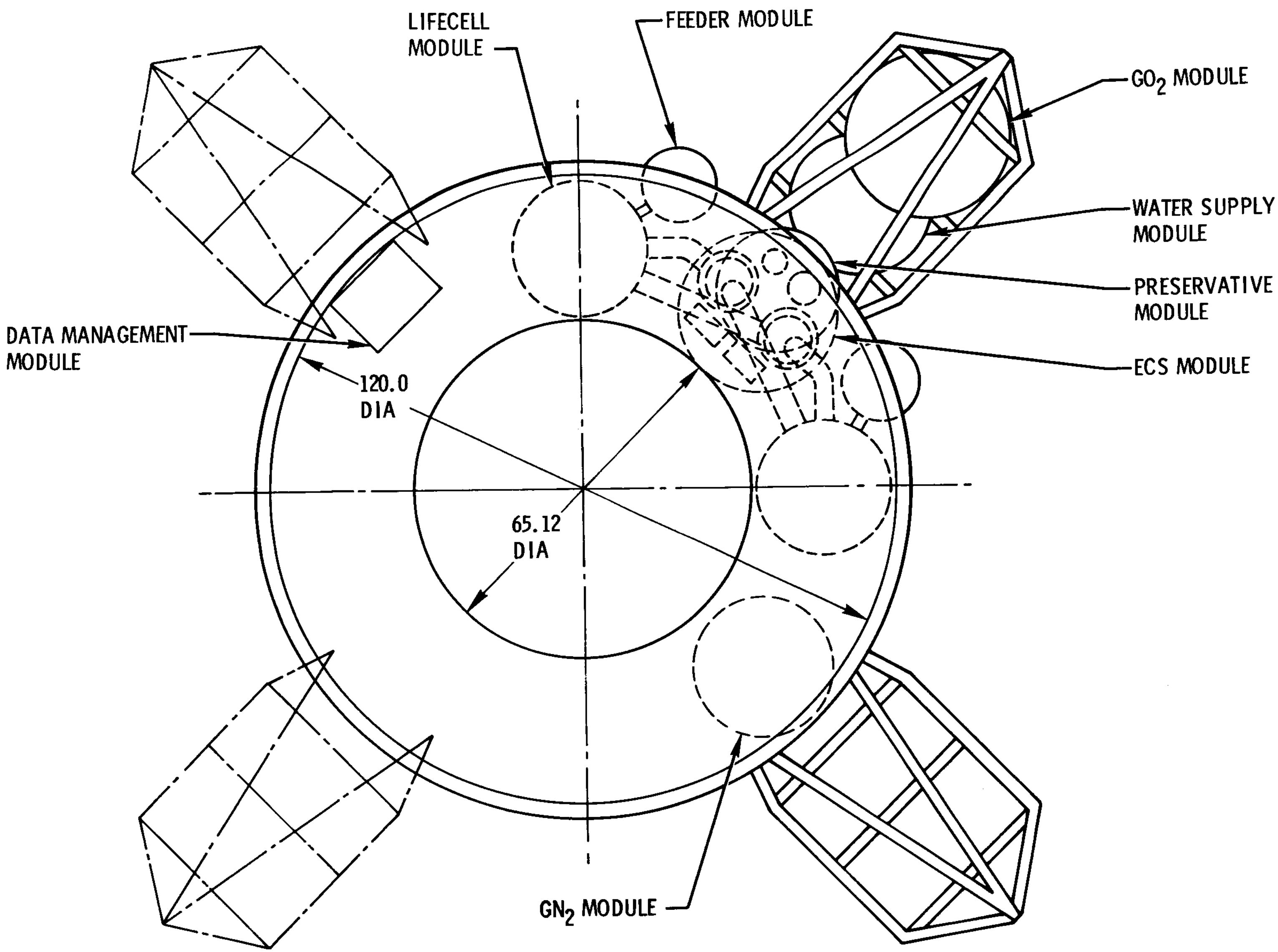


Fig. 192b Two Chimpanzee Modules Installed in AM/MDA (Cont.)

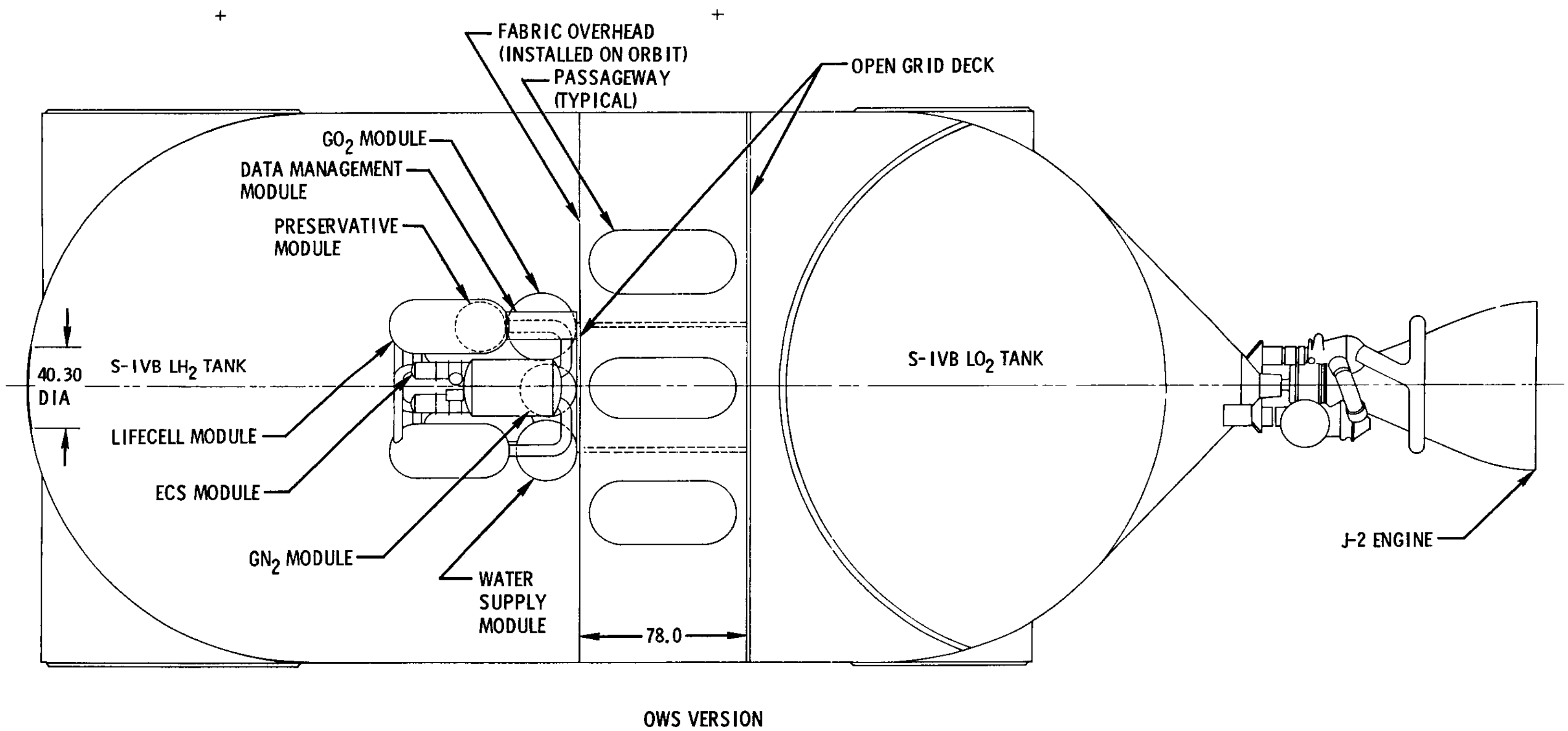


Fig. 193a Two Chimpanzee Modules Installed in OWS

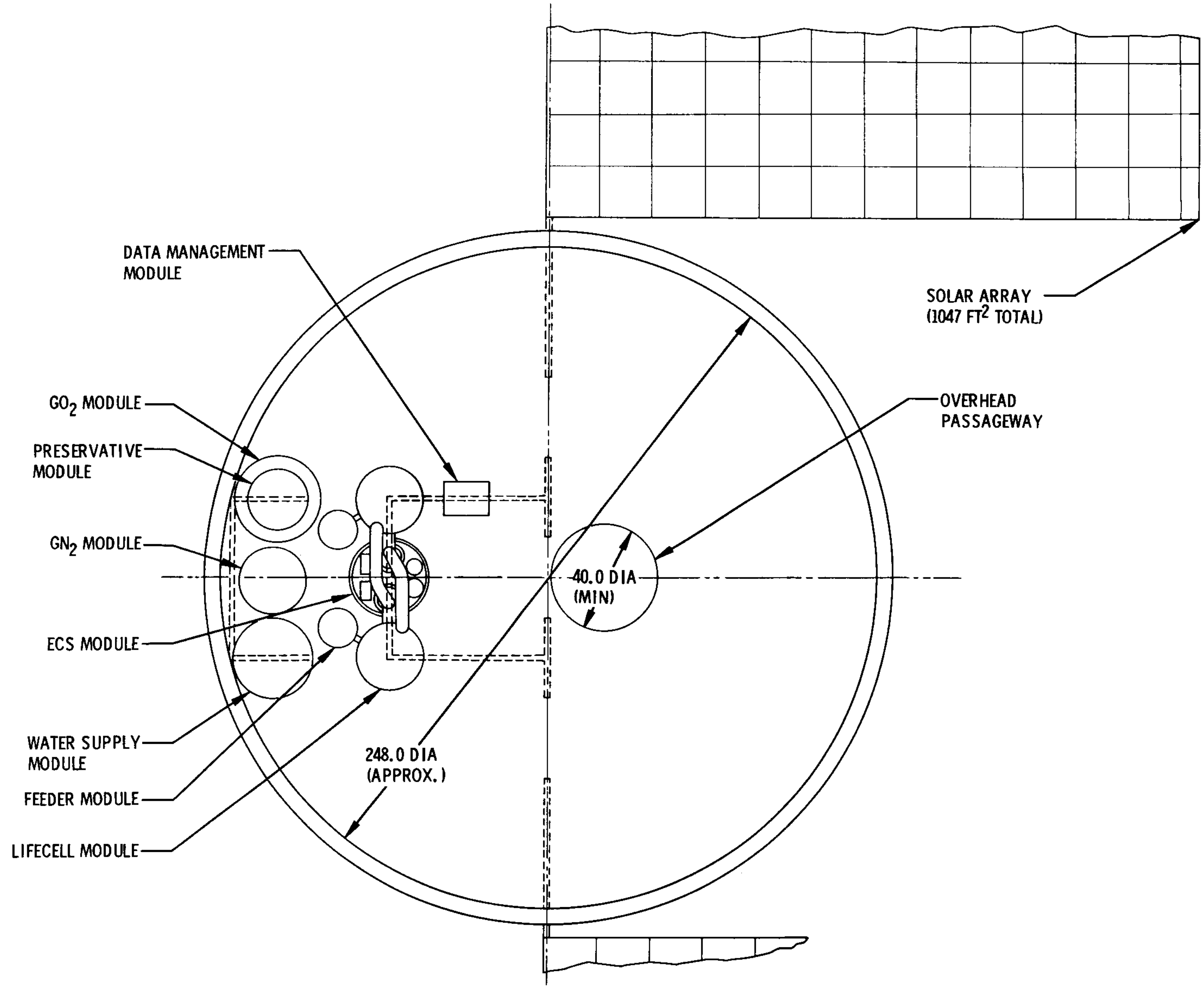


Fig. 193b Two Chimpanzee Modules Installed in OWS (Cont.)

In all mission modes, the animals are retrieved in their retrieval canisters (Rhesus) or portable lifecells (chimpanzee) which are stowed in the CM for return to earth at the conclusion of the mission.

Effect on system parameters. – The estimated weight of the prime experiment is 5500 lb, the average power requirement is 400 w, and the thermal rejection load is 785 Btu/hr. The DMS for this experiment is described in foregoing sections of this report. For both new experiments, each alternate mission mode has its individual effects on these prime experiment parameters.

Four-Rhesus experiment: For this experiment, the addition of two more animals and the added instrumentation results in increases in all parameters. The power requirement increases to 473 w for the independent mode and to 440 w for other modes. Thermal rejection loads increase to 1165 Btu/hr for the independent mode and to 1757 Btu/hr for the other modes. Weight statements for all modes are presented in Tables 94, 95, and 96.

The Rhesus experiment, with its four restrained and heavily instrumented monkeys, would have (except for TV data) considerable impact on the data management system. The TV system would only use one camera per primate instead of two per primate as described in the unrestrained primate experiment, thus maintaining the same total of four cameras which would be time-shared on the video recorder.

Although near-field telemetry requirements for ECG and respiration measurements are eliminated, the measurements themselves are still required. Added to the basic unrestrained primate measurement (except for activity monitoring) are neurological measurements (including EMG with frequency response requirements of 0-300 Hz necessitating additional paralleling of PCMTEA 200 SPS input channels), hematologic measurements, new cardiovascular measurements, and urinary measurements. One PCMTEA unit plus an expanded animal data commutator should be able to collect all necessary data from one primate. Data from all four primates could not be taken simultaneously without additional equipment. However, the one PCMTEA plus expanded animal data commutator could be (with no added switching network) timeshared. This method would result in 10 sec of data every 12 min for primate no. 1, followed by 10 sec of data every 12 min from primate no. 2, etc. The additional measurements plus the fact that the number of animals is doubled will result in approximately four times as much data as the prime OPE and, as a consequence, will require four times as much storage capability or necessitate shorter periods between ground dumps. Additional ground dumps would only tend to overtax the MSFN's predicted data innundation occurring from AAP, Apollo-in-Line-Programs, OPE, etc. It would appear more desirable to increase the data recorders capability.

LMSC recommends that the Apollo data recorder (five digital channels and nine analog channels) be used in place of the recorder recommended for the prime OPE. The added analog recording capability would be very suitable for recording data in the 0- to 500-CPS range, using FM recording techniques besides providing a new vocal recording capability. The Apollo recorder is a three-speed unit which provides a maximum recording time of 4 hr, weighs 40 lb, consumes 40 w of power, and occupies 1254 in.^{3} of volume. Four of the recorders specified for the prime OPE

TABLE 94

WEIGHT SUMMARY – FOUR 13-LB RHESUS IN INDEPENDENT MODE

Item		Weight (lb)
Animal Transport Module 4 @ 143.8 lb		575.2
Lifecell Module 4 @ 166.5		666.0
ECS Module		519.0
Oxygen Module		645.0
Nitrogen Module		503.0
Preservative Module		50.0
Water Module		1001.0
Antennas		12.6
Electrical Power (473 w)(0.714 lb/w)		338.0
Thermal Control		316.6
Radiator (1165 Btu/hr)(0.077 lb/Btu/hr)	89.6	
Insulation Blanket	143.0	
Finishes	54.0	
Tube Insulation	20.0	
Access Provisions	10.0	
Outer Skin		190.0
EVA Provisions		13.0
Docking Truss		92.4
Internal Structure		110.8
Launch Truss		203.3
Separation System		38.8
Urine Sampling System		78.0
Feces Sampling System		78.0
Urine & Feces Sampling Systems Radiator		30.0
Data Management System		251.1
Feeder Module 2 @ 151.6 lb		303.2
Waste Water Modules		22.3
X-Band System		28.8
Attitude Control Subsystem		11.0
Total		6077.1

TABLE 95

WEIGHT SUMMARY OF FOUR RHESUS IN DOCKED MODE

Item	Weight (lb)	
Animal Transport Module 4 @ 143.8 lb		575.2
Canister	31.0	
Recorder-Receiver Amp. & Batts.	5.0	
Cardiovascular Electronics & Av. Plumbing	0.3	
Hemotological Electronics & Sensor Sys.	3.5	
Animal Restraint Module	10.0	
RBC Survival Electronics	0.9	
Urine Analysis Electronics & Sampling Sys.	6.0	
Creating-Creatinine Ratio Analyzer	15.0	
Disconnects	2.0	
Door Lock	2.0	
Umbilical Pad & Umbilicals	4.5	
Shield	1.5	
Handle	0.2	
Restraint Module Attach Fitting	0.5	
Animal Removal Clamp	1.0	
KO_2 & LiOH	3.6	
Canister Removal Clamp	1.5	
Canister Door	1.5	
Heparin (9360 cc @ sp. gr. = 1.0 + tank)	23.6	
Animal	13.0	
Neurological Electronics	0.3	
Implants	0.2	
Nearfield Receiver	0.7	
Vocalization Microphone & Amplifier	1.0	
Support Structure	10.0	
Urine Flush (1000 cc H_2O + 250 cc germicide tank)	5.0	

TABLE 95 (Cont.)

Item	Weight (lb)	
Lifecell Module 4 @ 166.5 lb		666.0
Mass Measurement Sys.	40.0	
Psychomotor	7.0	
TV 1 @ 4.0 lb	4.0	
Canister	16.0	
Waste Management Sys.	18.5	
Elevator	10.0	
Misc.	25.0	
Low Flow Fan	6.0	
Support Structure	20.0	
Data Mgmt. Sys. (inside Lifecell)	20.0	
ECS Module (same as prime Section 2.4.2)		519.0
Oxygen Module		645.0
Oxygen [(.191)(4) + 0.06] [(8)(31)] [1.1]	225.0	
Tank & Lines (1.52)(225.0) + 15	357.0	
Regulator	5.0	
Support Structure	58.0	
Nitrogen Module		199.0
Nitrogen (0.24)(8)(31)(1.1)	65.5	
Tank & Lines (1.46)(65.5) + 15	110.5	
Regulator	5.0	
Support Structure	18.0	
Preservative Module (intravenous)		50.0
Preservative (4) (10.0)	40.0	
Tank & Lines	5.0	
Support Structure	5.0	
Water Module		1001.0
Water (0.792)(4)(8)(31)(1.1)	865.0	
Tank & Lines	45.0	
Support Structure	91.0	

TABLE 95 (Cont.)

Item	Weignt (lb)	
Antennas (same as prime Section 4.3)		4.3
Electrical Power (440)(1.28 lb/w)		563.0
Thermal Control		343.0
Radiator (1757 Btu/hr)(0.066 lb/Btu/hr)	116.0	
Insulation Blanket	143.0	
Finishes	54.0	
Tube Insulation	20.0	
Access Provisions	10.0	
Outer Skin System		136.0
Skin	96.0	
Stiffeners	25.0	
Access Provisions (Doublers, etc.)	15.0	
EVA Provisions		13.0
Docking Truss 2 @ 92.4 lb		184.8
Internal Structure		110.8
Launch Truss		203.3
Separation System		38.8
Urine Sampling System 4 @ 19.5 lb		78.0
Feces Sampling System 4 @ 19.5 lb		78.0
Urine & Feces Sampling Systems Radiator		30.0
Data Management Sys.		260.6
Box	16.8	
Support Structure	12.0	
S-Band Transceiver	20.2	
Amplifier/Diplexer	16.8	
PCM & Timing Equipment Assy.	35.0	
TV Recorder	20.0	
Work Panel Recorder	8.0	
Data Storage Recorder	40.0	

TABLE 95 (Cont.)

Item	Weight (lb)	
Signal Processor Assy.	11.5	
Animal Data Commutator	1.5	
Temperature Bridges	1.5	
Rate Sensor (Medical)	1.5	
Rate Sensor (Attitude)	1.0	
Miscellaneous	15.0	
Digital Update Link 2 @ 22.4 lb	44.8	
Cold Plate	15.0	
Feeder Module 2 @ 151.6 lb		303.2
Food (0.182)(8)(31)(1.1)(2)	97.6	
Mechanism	27.0	
Canister	14.0	
Support Structure	13.0	
Total		6002.5

provide a 20 min recording capability (each), have a total weight of 44 lb, consume a total of 60 w (provided they were run simultaneously), and occupy a total of 1284 $in.^3$ of volume.

Another area of major impact will occur with the up-link command system. The prime OPE has a margin of two commands over the command systems' capability. The added instrumentation and measurement capability proposed for the four restrained primates will necessitate doubling the up-link command capability.

Figure 194 is a simplified block diagram of the data management system extended to handling data from four restrained primates.

The above discussion applies to all mission modes of the four-Rhesus experiment. For all modes other than independent, there exists the possibility of using the uplink command capability available in the AM. However, this command capability would have to be doubled unless the IMBLMS was available. Preprogramming of the IMBLMS processor could be used to sample and evaluate the data from the four primates, however the additional measurement requirements plus two additional primates could create a scheduling problem with the use of the IMBLMS.

TABLE 96

WEIGHT SUMMARY OF FOUR RHESUS
IN AM/MDA OR OWS[a]

Item		Weight (lb)
Animal Transport Module 4 @ 143.8 lb		575.2
Lifecell Module 4 @ 166.5 lb		666.0
ECS Module		519.0
Oxygen Module		645.0
Nitrogen Module		199.0
Feeder Module 2 @ 151.6 lb		303.2
Preservative Module		50.0
Water Module		1001.0
Data Management Module		260.6
Antennas		4.3
Electrical Power (440 w) (1.28 lb/w)		563.0
Thermal Control		200.0
Radiator (1757 Btu/hr) (0.066 lb/Btu/hr)	116.0	
Insulation	30.0	
Finishes	54.0	
Urine and Feces Sampling Systems Radiator		30.0
Meteoroid Protection		25.0
Urine Sampling System 4 @ 19.5 lb		78.0
Feces Sampling System 4 @ 19.5 lb		78.0
Total		5197.3

[a] The OWS version is the same as the AM/MDA version except that the 25-lb meteoroid protection requirement is deleted. Total in OWS mode is 5172.3 lb.

Should the four Rhesus be installed in the OWS, a portion of the OWS data system might possibly be utilized to process part of the primate data. Unfortunately, this would result in primate data on two different downlinks and could present a correlation problem during ground evaluation.

Two-Chimpanzee experiment: For this experiment, electrical power requirements increase to 438 w for the independent mode and 405 w for all other modes. Thermal rejection loads increase to 970 Btu/hr for the independent mode and 1562 Btu/hr for all other modes. The weight summaries for the various modes are shown in Tables 97, 98, and 99.

The effect on the DMS of handling data for the two-chimpanzee experiment in all modes is similar to that described for the four-Rhesus experiment except that the quantity of data to be acquired is reduced. The instrumentation equipment shown in Fig. 194 required for each animal would be applicable to only two animals. The command capability would still have to be doubled over that of the prime experiment.

TABLE 97

WEIGHT SUMMARY FOR TWO 40-LB CHIMPANZEES IN INDEPENDENT MODE

Item	Weight (lb)	
Portable Lifecell 2 @ 340.1 lb each		680.2
ECS Module		519.0
Oxygen Module		696.8
Nitrogen Module		503.0
Preservative Module		40.0
Water Module		1162.7
Reclaimed Water Module		22.3
Data Management Module		251.1
Antennas		12.6
Electrical Power (438 w) (0.714 lb/w)		313.0
Thermal Control		301.6
Radiator (970 Btu/hr) (0.077 lb/Btu/hr)	74.6	
Insulation Blanket	143.0	
Finishes	54.0	
Tube Insulation	20.0	
Access Provisions	10.0	
Outer Skin		190.0
EVA Provisions		13.0
Docking Truss		92.4
Internal Structure		110.8
Launch Truss		203.3
Separation System		38.8
Urine Sampling System		39.0
Feces Sampling System		39.0
Urine and Feces Sampling Systems Radiator		17.0
Feeder Module 2 @ 159.8 lb each		319.6
V-Band System		28.8
Attitude Control Subsystem		11.0
Total		5575.0

TABLE 98

WEIGHT SUMMARY FOR TWO CHIMPANZEES
IN DOCKED MODE

Item	Weight (lb)	
Portable Lifecell 2 @ 340.1 lb each		680.2
Canister	30.0	
Recorder-Receiver Amp and Batts	5.0	
Cardiovascular Electronics and Av. Plumbing	0.3	
Hemotological Electronics and Sensor System	3.5	
Animal Restraint Module	15.0	
RBC Survival Electronics	0.9	
Urine Analysis Electronics and Sampling System	6.0	
Creatine-Creatinine Ratio Analyzer	15.0	
KO_2 and LiOH	5.8	
Heparin (9360 cc @ sp. gr. = 1.0 + Tank)	23.6	
Animal	40.0	
Neurological Electronics	0.3	
Implants	0.2	
Nearfield Receiver	0.7	
Vocalization Microphone and Amplifier	1.0	
Support Structure	30.4	
Mass Measurement System	45.0	
Psychomotor	10.0	
Waste Management System	15.0	
Low Flow Fan	6.0	
Data Mgmt. System (Inside Lifecell)	40.0	
Miscellaneous	25.0	
Disconnects	3.0	
Umbilical Pad and Umbilicals	5.0	
Shield	2.0	
Handle	0.4	
Restraint Module Attach Fitting	1.0	
Animal Removal Clamp	1.0	

TABLE 98 (Cont.)

Item	Weight (lb)	
Portable Lifecell (cont.)		
TV	4.0	
Urine Flush (1000 cc H_2O + 250 cc Germicide + Tank)	5.0	
ECS Module (same as prime Section 2.4.2)		519.0
Oxygen Module		696.8
Oxygen [(0.416) (2) + 0.06] (8) (31) (1.1)	243.5	
Tank and Lines (1.52) (243.5) + 15.0	385.0	
Regulator	5.0	
Support Structure	63.3	
Nitrogen Module		199.0
Nitrogen (0.24) (8) (31) (1.1)	65.5	
Tank and Lines (1.46) (65.5) + 15.0	110.5	
Regulator	5.0	
Support Structure	18.0	
Preservative Module		40.0
Preservative 2 @ 15.0 lb each	30.0	
Tank and Lines	5.0	
Support Structure	5.0	
Water Module		1162.7
Water (1.87) (2) (8) (31) (1.1)	1002.0	
Tank and Lines	55.0	
Support Structure	105.7	
Data Management Module		260.6
Antennae		4.3
Electrical Power (405 w) (1.28 lb/w)		518.0
Thermal Control		330.0
Radiator (1562 Btu/hr) (0.066 lb/Btu/hr)	103.0	
Insulation Blanket	143.0	
Finishes	54.0	
Tube Insulation	20.0	
Access Provisions	10.0	

TABLE 98 (Cont.)

Item	Weight (lb)	
Outer Skin System		136.0
Skin	96.0	
Stiffeners	25.0	
Access Provisions (Doublers, etc.)	15.0	
EVA Provisions		13.0
Docking Truss 2 @ 92.4 lb each		184.8
Internal Structure		110.8
Launch Truss		203.3
Separation System		38.8
Feeding System 2 @ 159.8 lb each		319.6
Food (0.397) (8) (31) (1.1)	108.3	
Inert Wt.	51.5	
Urine Sampling System 2 @ 19.5 lb each		39.0
Feces Sampling System 2 @ 19.5 lb each		39.0
Urine and Feces Sampling Systems Radiator		17.0
Total		5511.9

TABLE 99

WEIGHT SUMMARY FOR TWO CHIMPANZEES IN AM/MDA AND OWS MODES[a]

Item	Weight (lb)	
Portable Lifecell Module 2 @ 340.1 lb each		680.2
ECS Module		519.0
Oxygen Module		696.8
Nitrogen Module		199.0
Preservative Module		40.0
Water Module		1162.7
Data Management Module		260.6
Antennae		4.3
Electrical Power		518.0
Thermal Control		187.0
Radiator (1562 Btu/hr) (0.066 lb/Btu/hr)	103.0	
Insulation	30.0	
Finishes	54.0	
Meteoroid Protection		25.0
Urine Sampling System 2 @ 19.5 lb each		39.0
Feces Sampling System 2 @ 19.5 lb each		39.0
Urine and Feces Sampling Systems Radiator		17.0
Feeding System 2 @ 159.8 lb each		319.6
Total		4707.2

(a) The OWS mode weight is the same as the AM/MDA mode except that the 25-lb meteoroid protection requirement is deleted. Total weight is 4682.2 lb.

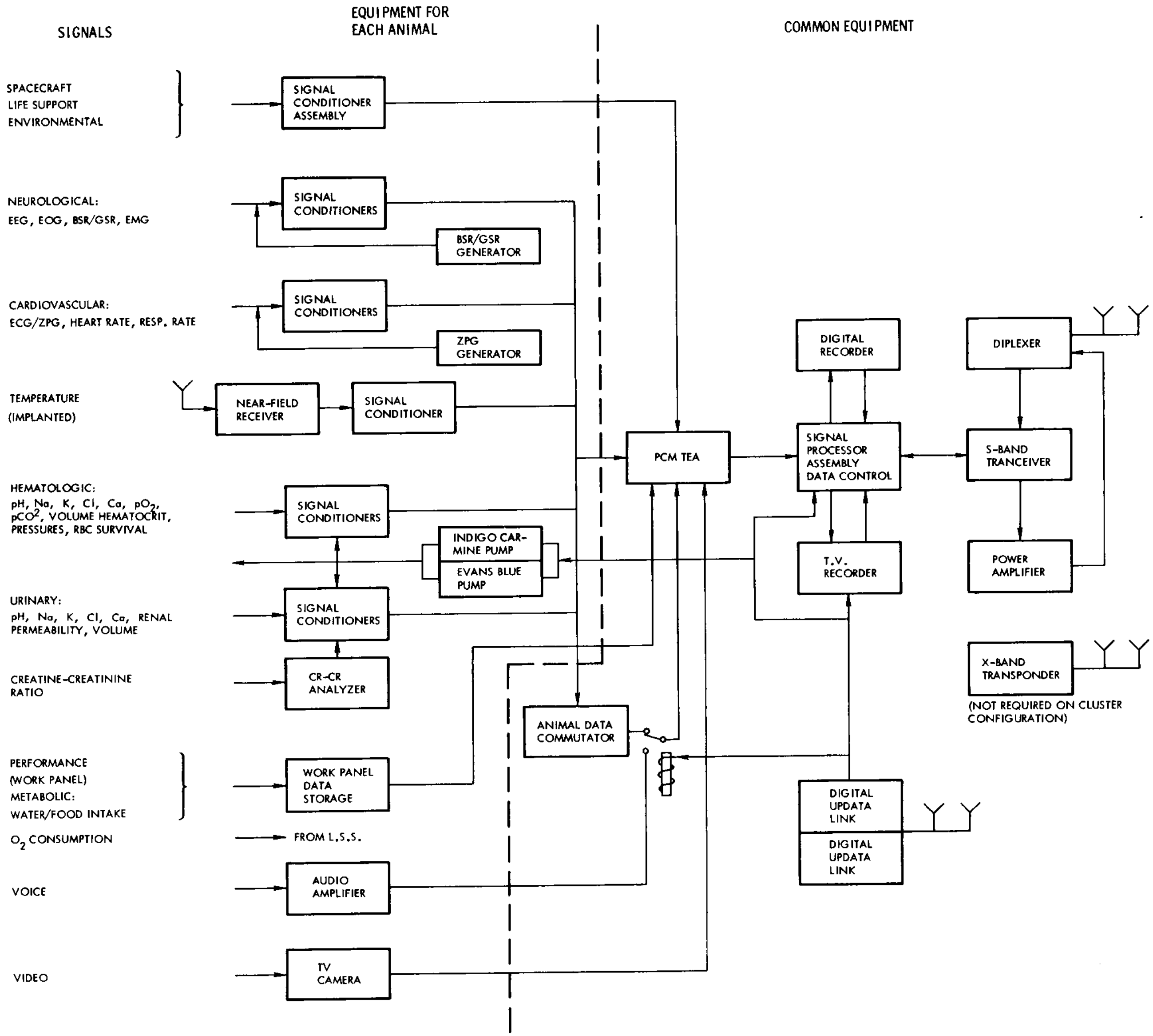


Fig. 194 Extended OPE DMS

Summary Comparison

Pertinent quantitative characteristics of the alternate experiments and mission modes investigated are shown in the summary matrix of Table 100. As seen in the matrix, reductions in total launched weight for all experiments are realized when the individual experiment is flown in a mode other than the independent mode. These reductions are most significant when the experiment is flown in the AM/MDA or the OWS. Power requirements, in general, are less than those for the independent versions,and the thermal rejection loads are greater. For all modes other than independent mode, the launched weights are based on the assumption that water reclaimed through the ECS will prove to be potable. Otherwise, additional drinking water modules will be required to sustain the mission for its programmed duration.

In discussing comparisons, it is helpful to relate all remarks to one element of the summary comparison matrix presented in Table 100. This element is the prime experiment in the independent mission mode, i.e., the prime/independent mode. Converting the prime/independent version to the prime/docked version entails a 23-w power reduction due to the removal of attitude control and to solar-cell degradation allowance. This reduction holds for all other AAP cluster versions. In this same conversion, the DMS thermal load is converted from passive rejection through the spacecraft skin to active rejection via the AAP cluster radiator. This conversion prevails for all other AAP cluster versions. The net weight change is the result of reduction effected by removal of the attitude control system and a portion of the drinking water supply and of an increase resulting from augmentation of the cluster electrical and thermal rejection systems.

The conversion from a prime/docked to a prime/AM/MDA version allows removal of a considerable amount of structure. This removal is somewhat offset by the weight increase associated with conversion of a multianimal lifecell into individual animal lifecells.

Comparing the in-addition-to-prime/independent version with the prime/independent version, a power increase of 301 w is encountered because of the addition of the regenerative life-support subsystem and the contaminant analysis and chemical and biological air sampling units. This addition accounts for the 890 Btu/hr thermal load increase. The weight is held at 5,500 lb, however, by offloading expendables. The conversion of the in-addition-to-prime/independent version to the AAP cluster versions involves the same causes and increments as those previously specified for the prime/independent version. The weight difference between the prime experiment and the in-addition-to-prime experiment, when converting from the independent mode to the docked mission modes, is due to the greater cluster electrical-power augmentation required for the docked mode experiment.

The in-place-of-prime (four-Rhesus)/independent version requires more power than the prime/independent version, chiefly because of the increased animal instrumentation and metabolic loads. This increased electrical load is also the major reason for the increase of thermal load. The weight difference is the result of reconfiguration of the lifecell into four single-animal units, and re-sizing of the electrical-power and thermal-rejection systems, together with the adjustment of all expendables for an 8-month mission.

TABLE 100

EXPERIMENT/MISSION MODE COMPARISON MATRIX

Experiment	Mission Mode			
	Independent	Docked to MDA	Installation in AM/MDA	Installation in OWS
Prime				
Two unrestrained 13-lb Rhesus for 1 yr	5,500[a] 400[b] 785[c]	5,087[a] 377[b] 1,273[c]	4,447[a] 377[b] 1,273[c]	4,422[a] 377[b] 1,273[c]
In addition to Prime				
RLSS, contaminant analysis, and chemical and biological air sampling plus Prime experiment for 6 to 12 mo	5,500[a] 701[b] 1,675[c]	5,386[a] 678[b] 2,163[c]	4,745[a] 678[b] 2,163[c]	4,721[a] 678[b] 2,163[c]
In Place of Prime				
Four restrained 13-lb Rhesus for 8 mo	6,077[a] 473[b] 1,165[c]	6,003[a] 440[b] 1,757[c]	5,197[a] 440[b] 1,757[c]	5,172[a] 440[b] 1,757[c]
In Place of Prime				
Two restrained 40-lb chimpanzees for 8 mo	5,575[a] 438[b] 970[c]	5,512[a] 405[b] 1,562[c]	4,707[a] 405[b] 1,562[c]	4,682[a] 405[b] 1,562[c]

[a] Weight (lb)
[b] Average power (w)
[c] Thermal load (Btu/hr)

Conversion of the in-place-of-prime (four-Rhesus)/independent version into AAP cluster versions involves the same mechanical changes as those required for the prime or in-addition-to-prime versions. The greater weight saving for the prime or in-addition-to-prime versions is attributable to the lifecell being already compartmented into single-animal units whereas the prime experiment involved a major lifecell conversion.

The in-place-of-prime (two chimpanzee)/independent experiment involves the same changes, but to a lesser degree, as those for the in-place-of-prime (four-Rhesus) version.

From the standpoint of data management, the impact on airborne and ground equipment itself is relatively small for all modes of the prime experiment and of other additional experiments. In the case of the prime experiment, any of the modes other than the independent version reduces the airborne equipment by elimination of the X-band transponder system. For the additional experiments, all apparent problems can be solved by minor airborne equipment additions and a reasonable amount of time-sharing of available data channels. The experiments which may be used in place of the prime experiment, however, have a significant impact on the DMS in the form of a doubled command capability requirement, and the addition of complex bio-instrumentation, of which much is still in an early state of development.

All modes other than the independent version, from an operations standpoint, offer the distinct advantage of eliminating a separate rendezvous by the retrieving CM. In addition, they provide a means for making use of astronaut presence to effect repair and maintenance to a degree which increases as the mission mode proceeds from docked to AM/MDA to OWS. At the same time, the requirement for astronaut EVA decreases as the mission modes proceed through the same order.

In the independent mode, all expendables and equipment must be onboard a single-launch vehicle at the time of launch since there is no possibility for resupply, repair, or maintenance after orbital injection.

A significant operational impact of the experiments in place of the prime experiment is the possible overtaxing of the limited return capability of the CM. The four-Rhesus experiment involves the return of four retrieval canisters which are larger in size that the two canisters to be returned for the prime experiment. In the two-chimpanzee experiment, the animals must be returned in portable lifecells which impose even greater weight and volume penalties than the four-Rhesus canisters. The experiment in addition to the prime experiment has little effect on return requirements since the only additional items to be returned are the samples from the chemical and biological air-sampling units.

Other operational characteristics of the alternate experiments and mission modes are summarized in Table 100.

TABLE 101

OPERATIONAL FACTORS MATRIX

Experiment	Operational Factor	Mission Mode			
		Independent	Docked	Installation in AM/MDA	Installation in OWS
Prime Two unrestrained 13-lb Rhesus for 1 yr	Resupply	None	Possible →		
	EVA	For retrieval only	For insertion, retrieval, and maintenance	For maintenance only	None required
	Maintenance capability	None	Minimum	Moderate	Maximum
	CM return requirements	Minimum →			
In addition to Prime RLSS, contaminant analysis, and chemical and biological air sampling plus Prime experiment for 6 to 12 mo	Resupply	None	Possible →		
	EVA	For retrieval only	For insertion, retrieval, and maintenance	For maintenance only	None required
	Maintenance capability	None	Minimum	Moderate	Maximum
	CM return requirements	Low →			
In Place of Prime Four restrained 13-lb Rhesus for 8 mo	Resupply	None	Possible →		
	EVA	For retrieval only	For insertion, retrieval, and maintenance	For maintenance only	None required
	Maintenance capability	None	Minimum	Moderate	Maximum
	CM return requirements	Moderate →			

TABLE 101 (Cont.)

Experiment	Operational Factor	Mission Mode: Independent	Docked	Installation in AM/MDA	Installation in OWS
In Place of Prime	Resupply	None	Possible		
Two restrained 40-lb chimpanzees for 8 mo	EVA	For retrieval only	For insertion, retrieval, and maintenance	For insertion, retrieval, and maintenance	None required
	Maintenance capability	None	Minimum	Moderate	Maximum
	CM return requirements	Maximum			

Conclusions and Recommendations

The prime experiment payload and spacecraft design lends itself well to experiment additions (i.e., regenerative life-support and contaminant, and chemical and biological sampling) and experiment substitutions (i.e., highly instrumented Rhesus or chimpanzee experiments). In addition, the prime experiment itself is adaptable to a modularized design in which all systems could be installed in the AM/MDA or OWS of the AAP cluster with significant savings in launch weight. Highly instrumented primate experiments in place of the prime experiment are feasible and attractive in terms of the large amount of in-flight biological and behavioral data yield, but impose large weight and volume penalties on the return CM.

The array of mission modes and experiments possible with the basic hardware and its derivatives provides a large range of possibilities from which to select in terms of (1) the type of data most desired, such as in-flight biodata, postflight biological analysis capability, or regenerative life-support and (2) the dependence on the AAP cluster desired and/or required.

The experiments studied provide data in the biological, behavioral, and life-support areas. The highly instrumented animals will yield a maximum of biological and behavioral data on orbit but the unrestrained, less-instrumented animal has a high probability of live return to earth for postflight study with its attendent biological/behavioral data yield; it is therefore difficult to assess the relative value of the various experiments studied. Clearly, however, the regenerative life-support and contaminant, chemical and biological sampling experiments significantly increase the yield of the life-support data.

In proceeding through the AAP-cluster mission modes considered, i.e., docked to the MDA, in the AM/MDA, and in the OWS, dependence on the cluster progressively increases with an attendent increase in the probability of experiment success through increased possibilities for astronaut involvement in observation, adjustment, and/or repair.

The prime experiment with its additions in the OWS results in a very attractive combination of success probability and experiment yield. It can be accomplished with a relatively low launch weight, has slight impact on other subsystems and on the CM return capability, and to a maximum degree, lends itself to astronaut caretaking.

Among the independent spacecraft configurations studied, this experiment also appears very promising. It can be accomplished within the original weight target of 5,500 lb for the prime experiment and will yield approximately 6 mo of weightless animal testing even if the RLSS fails early in the mission. Any successful RLSS operating time adds directly to the 6-mo experiment duration.

Supplement References

1. NASA LRC, Development of a Two-Gas Atmosphere Sensor System (Mass Sectrometer), Book I of III, SDS Data Systems, NASA LRC Report, NASA CR-66T72, 31 March 1966

2. Lockheed Missiles & Space Company, Catalog of Infrared and Mass Spectra and Chromatographic Retention Time Indices of Trace Contaminants Identified in Closed Environmental Systems, by E. H. Hawasaki, O. T. Leong, and R. C. Tuttle, LMSC 672406, Sunnyvale, California, 24 January 1967

3. -----, Gas Monitoring for Air Force Biosatellite Test and Chemical Analysis of Room Temperature CO Converter Charcoal and Lithium Hydroxide Beds, by E. H. Kawasaki, O. T. Leong, and R. C. Tuttle, LMSC 670573, Sunnyvale, California, 22 June 1966

Appendix A

AAP-LOCKHEED SPECIAL TASKS REVIEW

Lockheed personnel working under Contract NAS 8-21103 for the Marshall Space Flight Center have included a primate experiment (T009) in their AAP payload integration studies. Reference 3 is a compilation of the material presented to MSFC personnel on 26 May 1967 in response to special augmentation tasks assigned to Lockheed under the existing contract. Illustrations contained in this Reference which relate to the OPE spacecraft are reproduced here for information.

The reproductions are presented here as Figs. A-1 through A-3. Figure A-1 shows the primate experiment scheduled aboard Flight B4. The experiment is a dependent module version, of the OPE installed inside the cluster. The three preferred locations of the experiment are highlighted in Fig. A-2.

The locations favored by MSFC are the MDA (Multiple Docking Adapter) and the S-IVB Workshop as illustrated in Fig. A-3.

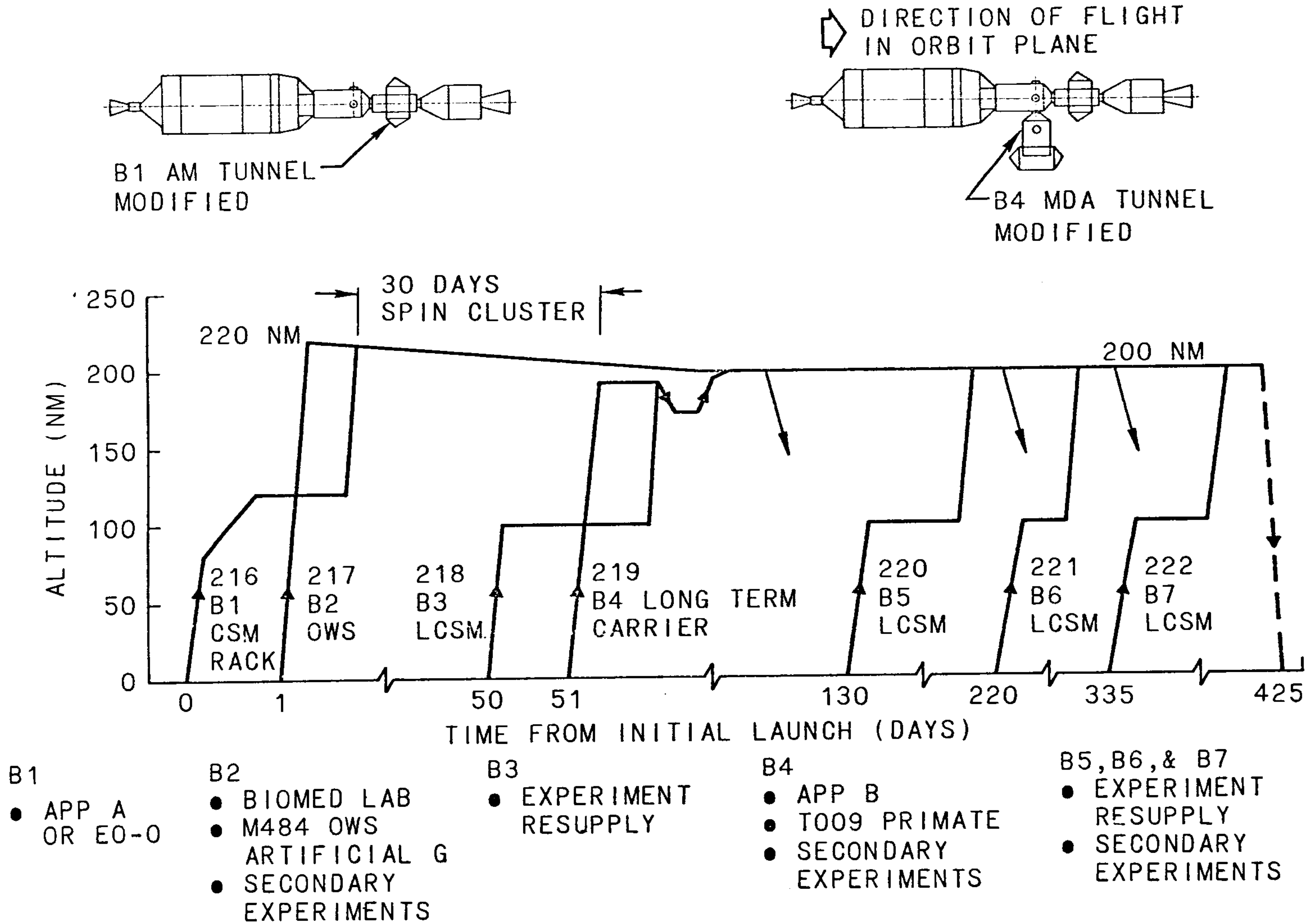


Fig. A-1 Mission Profile

INDEPENDENT MODULE

- COMPLETELY SELF CONTAINED
- RENDEZVOUS/RESUPPLY
- RENDEZVOUS/RESUPPLY/RETRIEVE

DEPENDENT MODULE

- INTERIOR OWS
- DOCKED
- TETHERED MODULE
- INTEGRATION IN MDA (ADDED)

Fig. A-2 Candidate Primate Experiment Modules

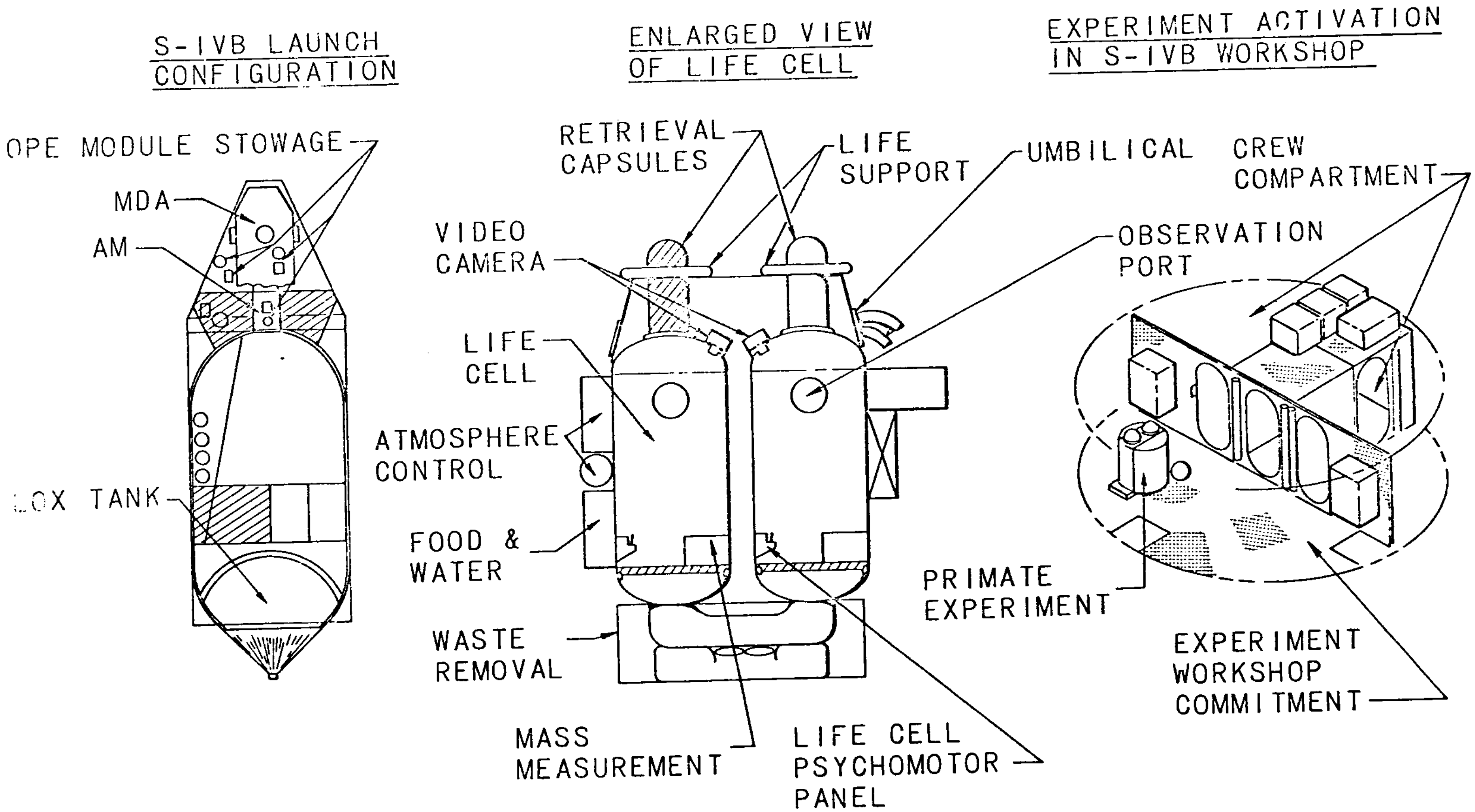


Fig. A-3 Primate Experiment

Appendix B

MSFC-AAP GUIDELINES, 11 APRIL 1967

The Reference 4 MSFC guidelines for AAP payload integration studies represent a sharp departure from the previously supplied guidelines. The Cluster A, B, C, and D organization of flights is now replaced, according to Reference 4, by the "1968 Cluster", the "Reuse Cluster" and the "New Cluster". Flights in the first two of these groupings are scheduled for 1968-1969, and include Flight 216. The "new Cluster" is defined for contingency purposes only, to be launched in 1970 based upon the failure of the inoperable condition of the "1968 Cluster". Its inclination angle would fall in the range from 28 deg to 50 deg. The lists of "1968 Experiments" and "Candidate New Experiments" do not include the primate experiment.

Appendix C

ORBITING PRIMATE EXPERIMENT FAILURE MODES EFFECT AND ANALYSIS

This appendix contains a series of tabulations and block diagrams to define hazard levels, individual and total subsystem reliability, and the results of failure modes effect and analysis.

Subsystem Reliability Summary

Airborne Data Management		0.9497
Integrated Environmental and Thermal Control		
Gas Control	0.9998	
Glycol Coolant Loop	0.9992	0.9993
Atmosphere Circulation	0.9996	
Electrical Power Subsystem	0.9988	
Drinking Water Subsystem	0.9995	
Feeding Subsystem	0.9994	
Attitude Control Subsystem	0.9987	
Animal Retrieval (Dead Preserved)	0.9972	
Waste Water Management	0.9994	
Total Subsystem Reliability =	0.942	

(1) The values given for each component are at 60 percent confidence level (general for space qualified hardware).

(2) No better confidence level than 50 percent can be assumed for the predicted subsystem values, as in the absence of performance data, there is an equal chance of them being in error as being correct.

Integrated Environmental and Thermal Control

	Reliability
Gas Control System	0.9998
Atmosphere Circulation	0.9996
Glycol Coolant Loop	0.9992
Overall Reliability	0.9993

Definition of Hazard Levels

1. Circumstances which would;

   a) Cause death to the primates, not attributable to extended weightlessness, or
   b) Prevent preservation and retrieval of the dead primates

2. Failures which would reduce the quality or quantity of data below an acceptable level

3. Failures which would reduce the mission duration to six months or less, but with adequate data and animal retrieval

4. Failures which would reduce the mission duration to six months, or more, with good quality data

FAILURE MODE AND EFFECTS ANALYSIS
INTEGRATED ENVIRONMENTAL AND THERMAL CONTROL

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.9996	1	1. Oxygen partial pressure too low or falling to zero	1. Oxygen regulator (Part No. 5) fails					
			(a) Closed	Fails to deliver oxygen to the system	(a) Atmosphere will degrade to 100 percent nitrogen	(a) Death of primates	1 in 2,500	None
	2		(b) Partially closed	Fails to deliver sufficient oxygen to maintain 3.5 psi partial pressure of oxygen	(b) Atmosphere has low partial pressure of oxygen	(b) May impair performance of primates or cause death from anoxia		
0.99999	1		2. Blocked oxygen flow meter (Part No. 12)					
			(a) Completely	Inoperative. No oxygen to gas control system	As above (a)	As above (a)	1 in 20,000	None
0.99995	4		3. Partial pressure of oxygen sensor (Part No. 64) failure	Gives no signal to oxygen flow control valve (Part No. 18)	Preferred mode of gas control inoperative, command to total pressure control system	Will not endanger mission success	1 in 25,000	None
0.9995	4		4. Oxygen flow control valve (Part No. 18) does not operate	Fails to regulate oxygen as required by partial pressure sensor (Part No. 64)	As in (3) above	Will not endanger mission success	1 in 2,000	None
0.99995	4	2. Oxygen partial pressure too high and rising	1. Signal from oxygen partial pressure Sensor (Part No. 64) not given to shut off oxygen flow control valve (Part No. 18)	Sensor inoperative	Oxygen partial pressure rising to give an eventual 100 percent oxygen atmosphere. Command to total pressure control mode	Will not endanger mission success	1 in 25,000	None
			2. Control valve fails to close on signal from oxygen partial pressure sensor	Does not obey command	As above	As above	1 in 2,000	None

FAILURE MODE AND EFFECTS ANALYSIS
INTEGRATED ENVIRONMENTAL AND THERMAL CONTROL (Cont.)

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.9996	2	3. Total pressure less than 14.7 psia	1. Nitrogen regulator (Part No. 6) fails					
			(a) Closed	No nitrogen flow	Life-cell pressure dropping toward 3.5 psia	No nitrogen flow shown by flow meter, ground command to 5 psia pure oxygen control mode	1 in 2,500	None
	2		(b) Partially closed	Insufficient nitrogen flow to maintain 14.7 psia total pressure	Gradual decrease in life-cell pressure to some lower equilibrium pressure	May require ground command to 5 psia pure oxygen control mode		
0.99995	2		2. Blocked nitrogen flow meter					
			(a) Completely	(a) No nitrogen flow	(a) Rapid drop in life-cell pressure	(a) No flow shown by flow meter, ground command to 5 psia pure oxygen control mode	1 in 20,000	None
	2		(b) Partially	(b) Insufficient nitrogen flow to maintain 14.7 psia total pressure	(b) Gradual decrease in life-cell pressure	(b) May require ground command to 5 psia pure oxygen control mode		
0.9995	2		3. Nitrogen flow control valve (Part No. 20) malfunction				1 in 2,000	None
			(a) No flow	As (a) above	As (a) above	As (a) above		
			(b) Restricted flow	As (b) above	As (b) above	As (b) above		
0.99999	1	Total pressure greater than 14.7 psia	1. Plumbing failure in oxygen system	High rate of O_2 gas discharged into life-cell	Atmosphere would go 100% oxygen and pressure in excess of 15 psia would open cabin relief valve (Part No. 49)	Aborted with probable loss of primates in a matter of days	1 in 100,000	None
0.99999	1		2. Plumbing failure in nitrogen system	High rate of N_2 gas discharged into life-cell	Atmosphere would go 100% nitrogen unless oxygen was being controlled to 3.5 psi by partial pressure sensor (Part No. 64). Excess gas would open cabin relief valve (Part No. 49)	Aborted with loss of primates. If oxygen partial pressure sensor was operative the primates would live. In this case the mission would be shortened by a time factor based on the characteristics of the flow limiter (Part No. 14)	1 in 100,000	None

FAILURE MODE AND EFFECTS ANALYSIS
INTEGRATED ENVIRONMENTAL AND THERMAL CONTROL (Cont.)

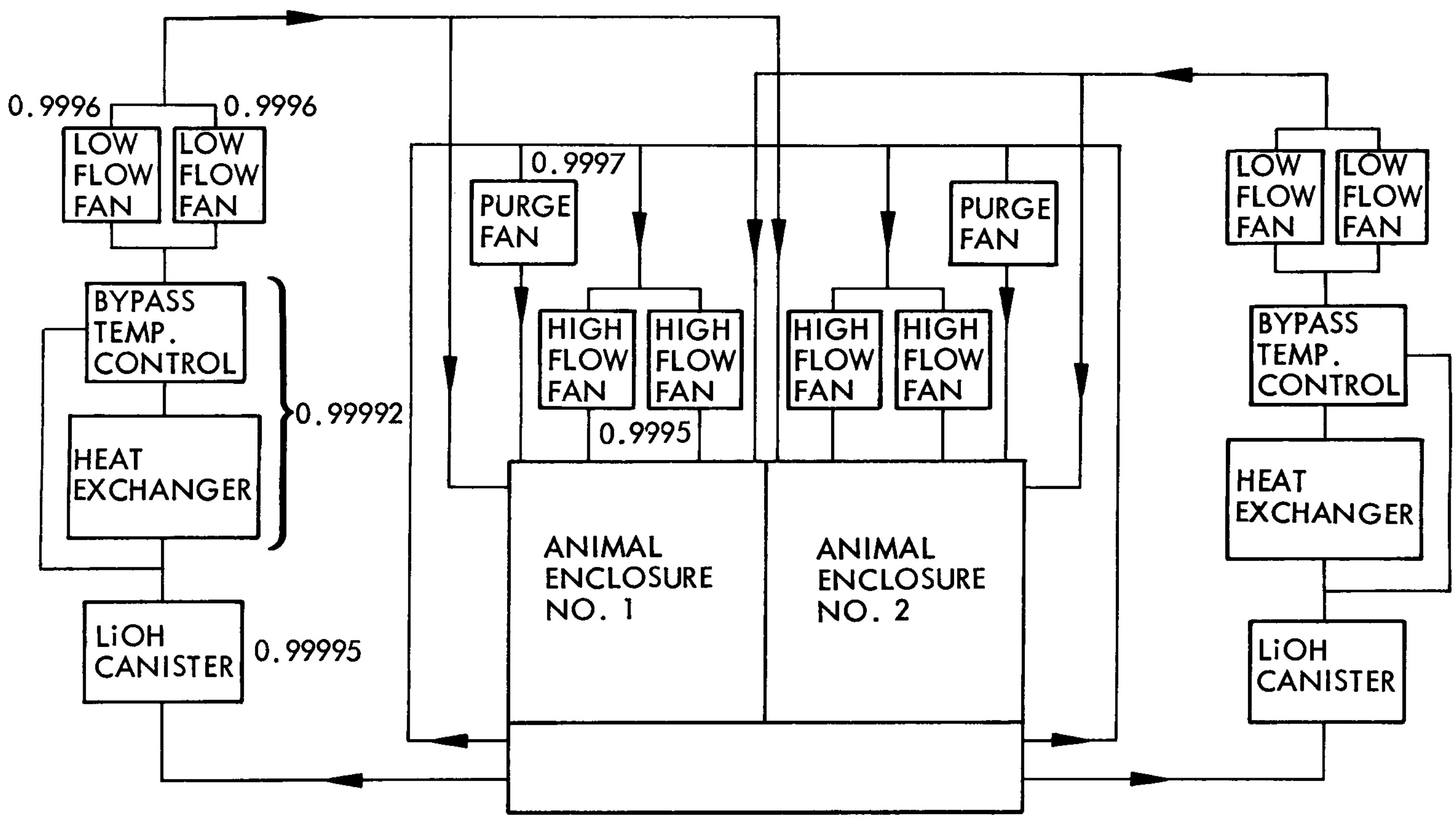
Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.999	1	5. Total loss of circulation	1. Loss of power	All circulation fans inoperative	Buildup of CO_2 and trace contaminants to a lethal level	Death of primates due to causes other than extended weightlessness	1 in 1,000	See electrical
0.9995	4	6. Reduced circulation	1. Failure of one high flow fan (Part No. 47) either mechanically or electrically	Inoperative	None. Switch to alternate redundant high-flow fan	Will not affect mission success	1 in 2,000	Already redundant
	4		2. Failure of two high-flow fans – one in each life-cell	Two high-flow fans inoperative	None. Switch to alternate redundant high-flow fans in each life-cell	Will not affect mission success		
	4		3. Failure of two high-flow fans – both in same life-cell	As above	Circulation will be reduced to that supplied by low-flow fan in one life cell. The other will have normal flow. The low-flow life-cell may have more trouble with floating waste	Both primates should be unaffected and mission success should not be impaired		
	2		4. Failure of three high-flow fans	Three high-flow fans inoperative	Remaining high-flow fan will maintain circulation in one life-cell	Mission success may be endangered due to loss of gyroscopic stiffness and effect on the attitude control		
0.9996	4	7. Loss of differential pressure across fans in low-flow ducts	1. Failure of one low-flow fan (Part No. 40)	Inoperative	None. Switch to alternate redundant low-flow fan	Will not affect mission success	1 in 2,500	None; already redundant
			2. Failure of two low-flow fans – one in each loop	Two low-flow fans inoperative	None. Switch to alternate redundant fans in each loop	Will not affect mission success		
			3. Failure of three low-flow fans	Three low-flow fans inoperative	Remaining low-flow fan will maintain a reduced flow through the life-cells. The flow through the LiOH canisters will be unequal but adequate	Mission success may be endangered due to loss of gyroscopic stiffness and effect on attitude control		
			4. Failure of two low-flow fans in same loop	Two low-flow fans inoperative	One loop operating with one or both fans will maintain low-flow in both loops and through both LiOH canisters	Mission success not endangered		

FAILURE MODE AND EFFECTS ANALYSIS
INTEGRATED ENVIRONMENTAL AND THERMAL CONTROL (Cont.)

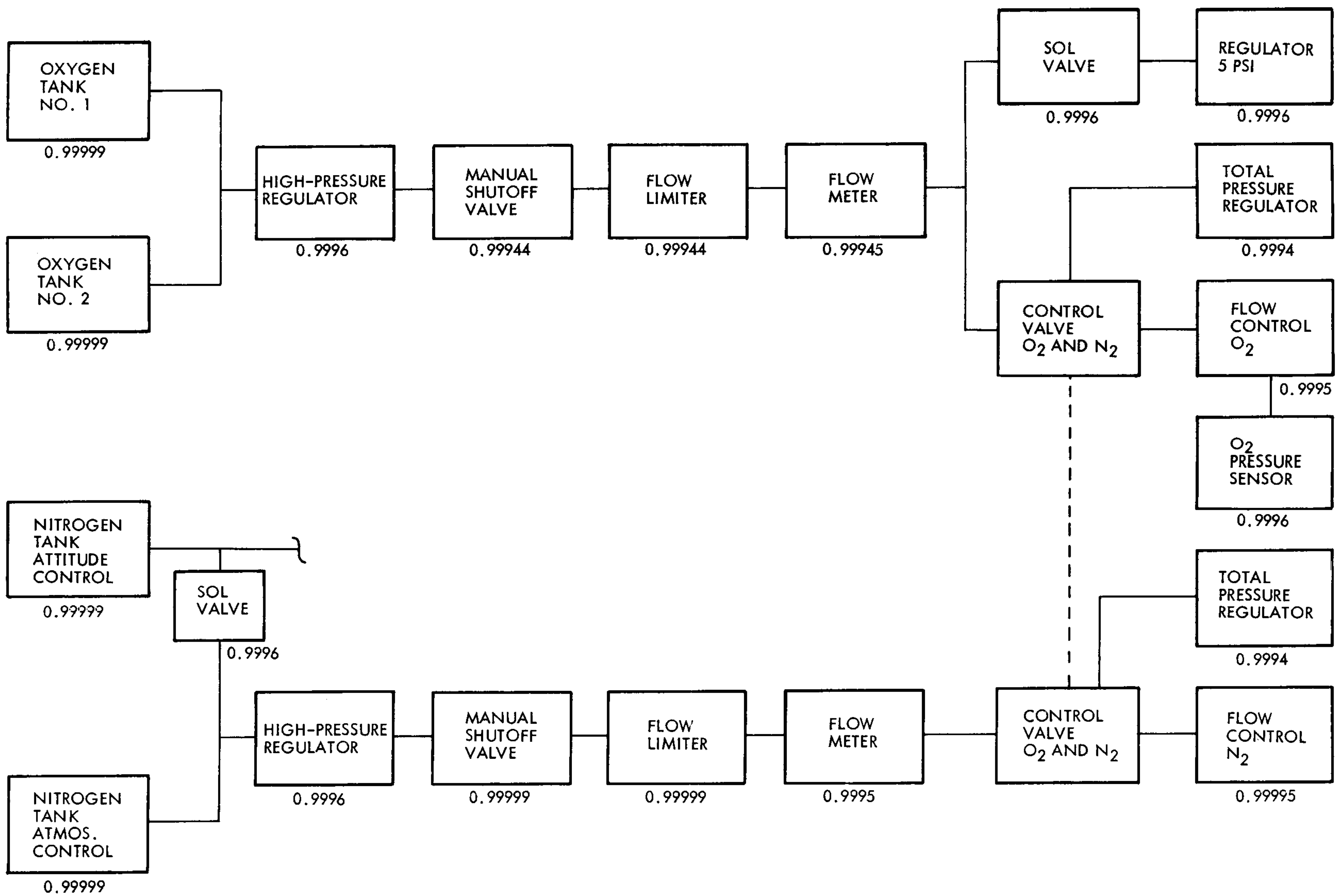
Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.9996	4	8. Atmosphere, temperature and humidity rising	1. Coolant pump failure mechanical or electrical (Part No. 24)	Inoperative	Temporary rise in temperature and humidity. Switch over to redundant pump by ground command	Will not endanger mission success	1 in 2,500	Already redundant
0.9999998	1		2. Failure of both coolant pumps (Part No. 24)	Both pumps inoperative	Gradual rise in life-cell temperature to intolerable level	Probable death of primates due to causes other than extended weightlessness	1 in 5,000,000	
0.9999999	4		3. Faulty heat exchanger (Part No. 27)					
			(a) Clogged tubes	Restricted flow of coolant	None. The heat exchanger in the other loop can handle the extra load	Will not affect mission	1 in 10,000,000	None
	4		(b) Clogged microporous filter	Water build-up	None. The heat exchanger in the other loop can handle the extra humidity	Will not affect mission		
0.99999	1		4. Failure of vernatherm radiator by-pass control (Part No. 31)				1 in 100,000	1
			(a) Complete by-pass of coolant	Inoperative	Temperature would rise in life cells to an intolerable level	Probable death of primates due to cause other than extended weightlessness		
0.99999	1		(b) No by-pass of coolant	Inoperative	Temperature would decrease in the life cells to an intolerable level	Probable death of the primates due to cause other than extended weightlessness	1 in 100,000	
	2		(c) Partial by-pass but no modulation	Inoperative	Temperature would cycle about some mean value	Possible death or impairment of primates performance. Shortened mission with probable retrieval dead or alive		

FAILURE MODE AND EFFECTS ANALYSIS
INTEGRATED ENVIRONMENTAL AND THERMAL CONTROL (Cont.)

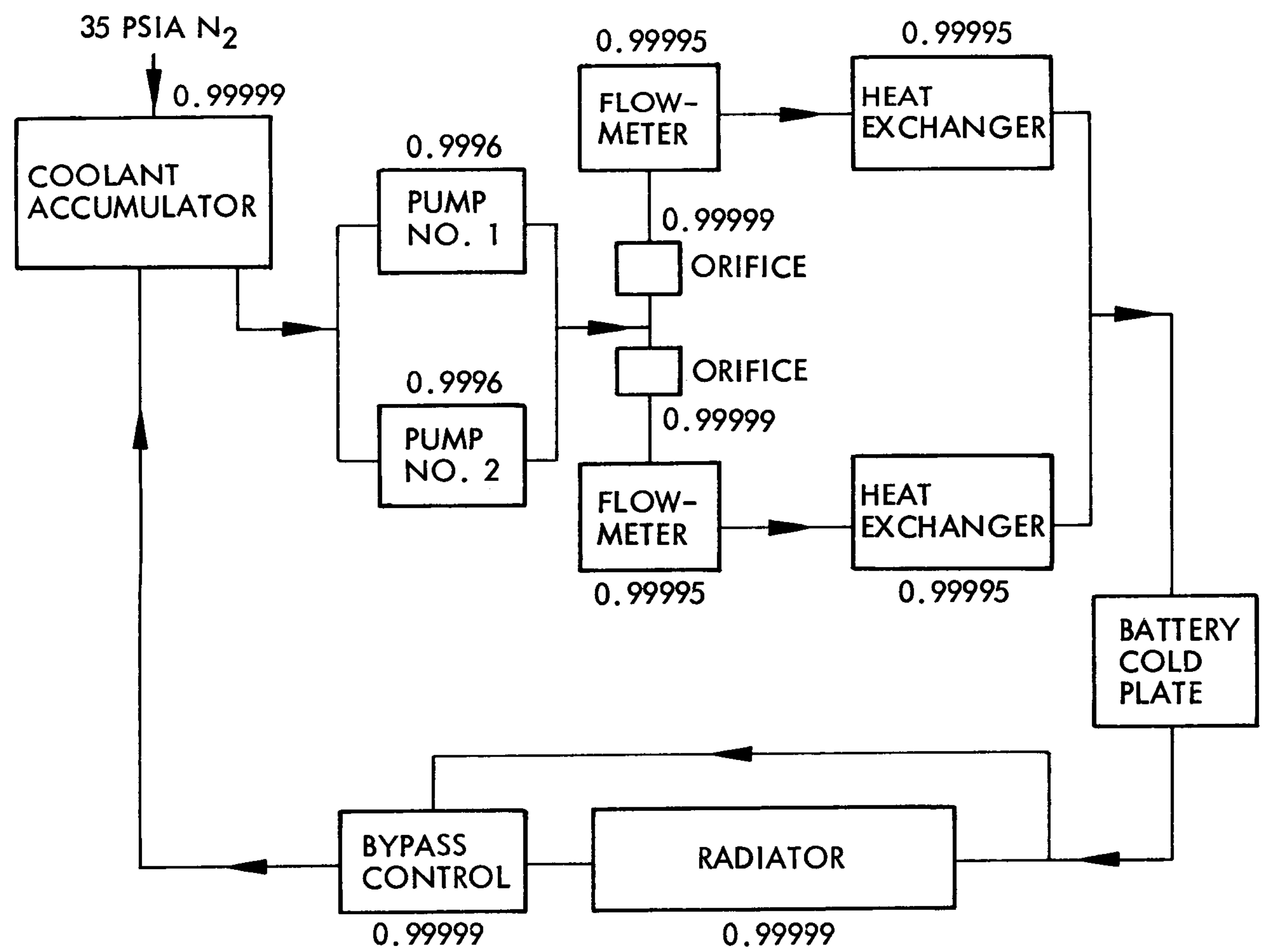
Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
	4		5. Radiator (Part No. 30) degraded by micro-meteorite bombardment	Alteration of absorption and emissivity coefficients	Wider variations of temperature and humidity	Will not affect mission success		None
0.99999	2	9. Coolant pump outlet pressure fluctuating	1. Ruptured diaphragm in coolant accumulator (Part No. 34)	Nitrogen gas will mix with coolant fluid	Pumps will behave erratically and cavitation can be expected. Pump life will be shortened	Mission would be shortened in anticipation of complete pump failure and loss of temperature control	1 in 100,000	None



Atmosphere Circulation – Reliability 0.9996



Gas Control System – Reliability 0.9998

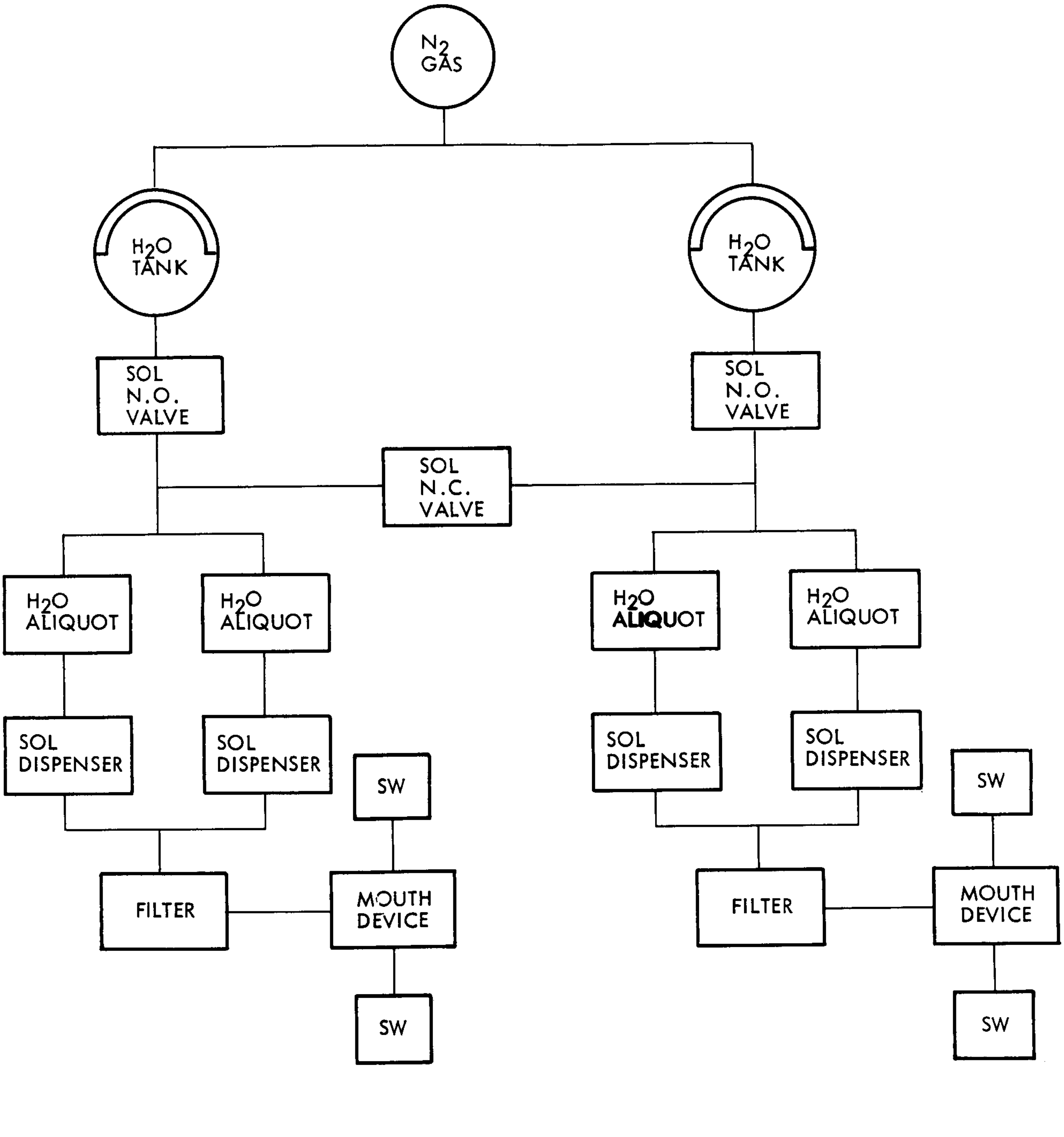


Glycol Coolant Loop – Reliability 0.99992

FAILURE MODE AND EFFECTS ANALYSIS
DRINKING WATER SUPPLY*

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.9993	3	1. No water delivered to one primate	1. Failure of normally open two-way solenoid valve downstream of water tank in closed position	Inoperative	Loss of available drinking water. Switch to other tank by opening the normally closed solenoid valve between tanks	Reduces mission time capability	1 in 1,400	None
	4		2. Aliquot accumulator dispensing mechanism stuck	Inoperative	None. Switch to spare accumulator	Will not affect mission		1
0.99999	4		3. 3-way 2-position solenoid valve fails electrically or mechanically	Inoperative	None. Switch to redundant spare valve	Will not affect mission	1 in 100,000	1
	3		4. Blocked filter; fungoid type growth on surface	No flow	Inoperative	Loss of primate due to cause other than extended weightlessness		1 H_2O purifier (silver ion or chlorine)
0.99999			5. Lip switch fails to make contact	None. Will probably operate next time around	Temporary loss of one aliquot portion	Will not affect mission	1 in 100,000	1
			6. Mouth device becomes plugged	No flow of water	Requires operation of a device, by ground command, to clear obstruction	Will not affect mission if effective unplugging device is used		1

*Overall reliability 0.99905

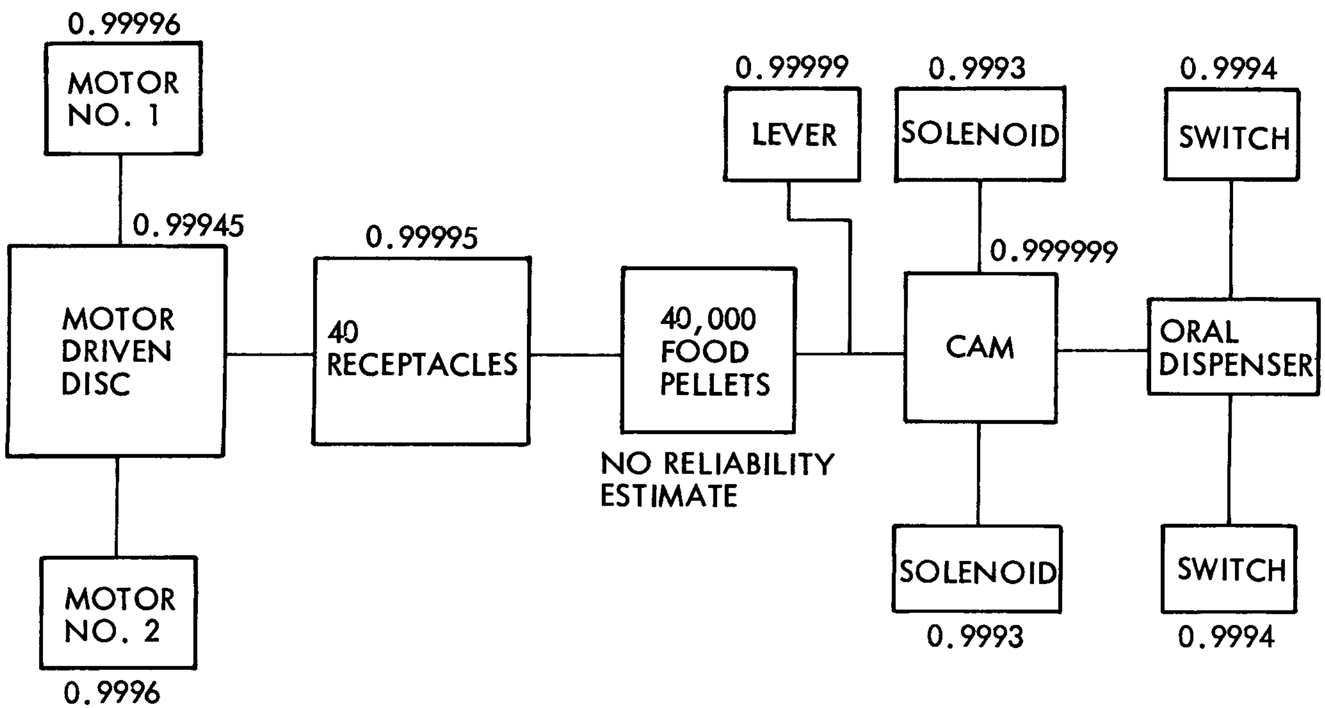


Drinking Water Supply and Dispensing System – Reliability 0.99905

FAILURE MODE AND EFFECTS ANALYSIS
FEEDER SYSTEM*

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.9995	4	1. Food supply system not working	1. Failure of plate to rotate either by					
			(a) Mechanical fault	Inoperative	No food dispensed to one primate	Primate will die. Mission data reduced to one primate	1 in 2,000	None
			(b) Electrical motor failure	Inoperative	As above	As above	1 in 2,000	1
			2. Failure of solenoid to operate lever mechanism	No food pellet dispensed	None. Switch over to spare redundant solenoid	Will not affect mission		1
			3. Food pellets sticking together	Food receptacles not properly filled	Feeder system erratic	Could affect mission success	Low if fully fully devel-oped system	

*Overall reliability 0.9994 (does not include figure for pellets sticking to receptacle or to each other)

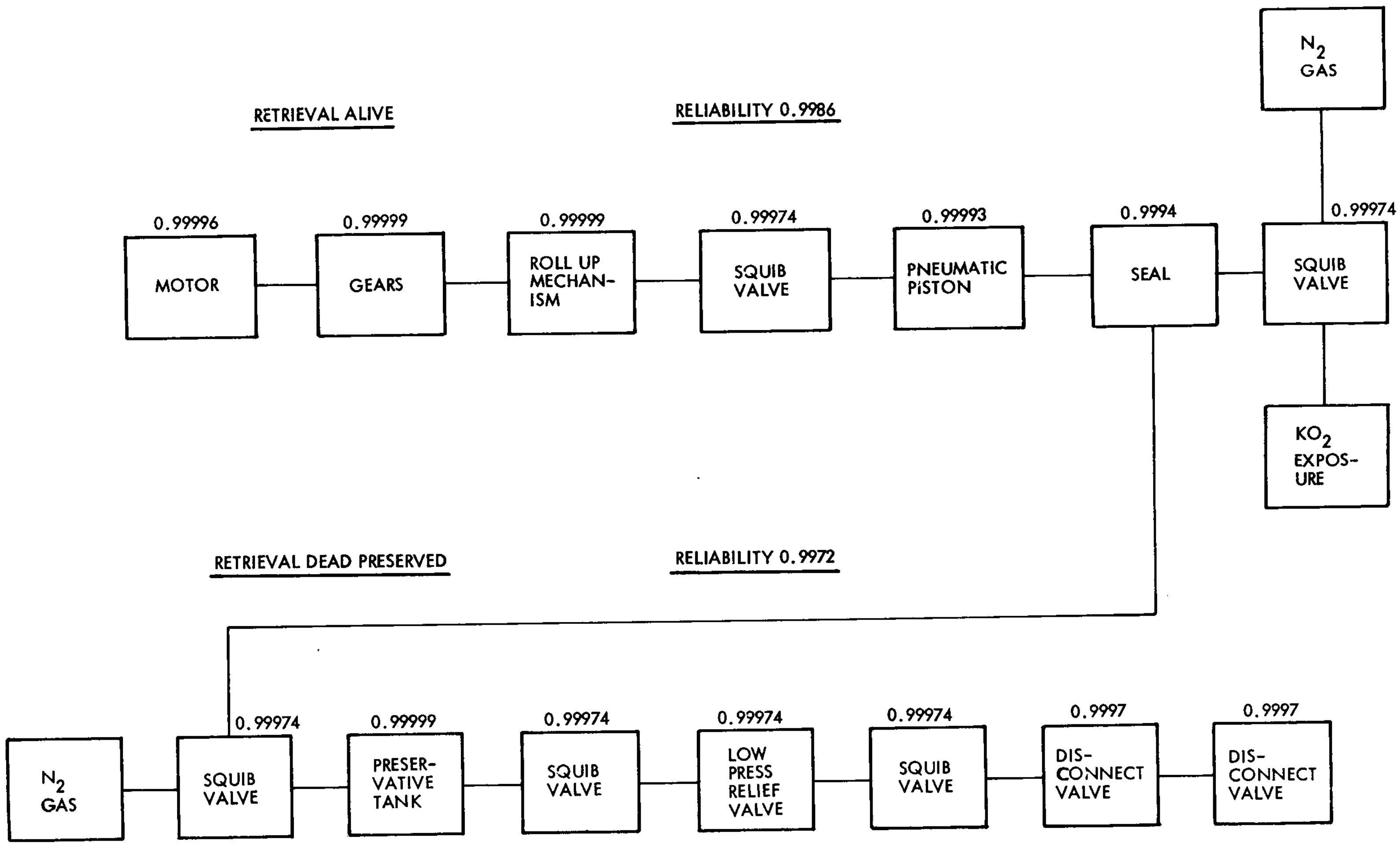


Feeder and Food Supply –
System Reliability 0.9994 (Excludes Food Pellets)

FAILURE MODE AND EFFECTS ANALYSIS
ANIMAL RETRIEVAL*

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.99995	1	1. Failure to get primate, dead or alive, into canister	1. Failure of motor-driven roller to roll up the thin steel cage wall	No confinement	Primate not guided into canister	Failure to retrieve primate or primates	1 in 20,000	None
0.99993			2. Failure of telescoping pneumatic actuator to drive domed plate into canister	Canister not properly sealed	Retrieval of primate not possible	As above	1 in 14,300	None
0.99974			3. Failure of squib to actuate pneumatic gas pressure system valve	No confinement of primate in canister	As above	As above	1 in 3,000	Already redundant
0.99974		2. Failure to preserve dead primate	1. Squib actuated valve fails to supply gas pressure to preservation tank	Some fluid might enter the canister by surface tension effects, but reliable preservation is unlikely	Preserving fluid not adequately surrounding the dead primate	Primate can be retrieved, but preservation doubtful	1 in 3,000	Already redundant
0.99974			2. Squib actuated valve fails to open. No fluid into canister	Fails in closed position	No preservation of dead primate	Failure to retrieve primate in a preserved condition	1 in 3,000	Already redundant
0.99974		3. Failure to off-load preservation fluid	1. Preservative vent valve squibs do not ignite	Dud squibs	Canister full of fluid. Disconnect would create thrust unless equipped with valve disconnects	The extra weight of retrieved canisters would modify reentry characteristics	1 in 3,000	Already redundant

*IV Retrieval System	Reliability
1. Retrieval alive	0.9986
2. Retrieval dead preserved	0.9972

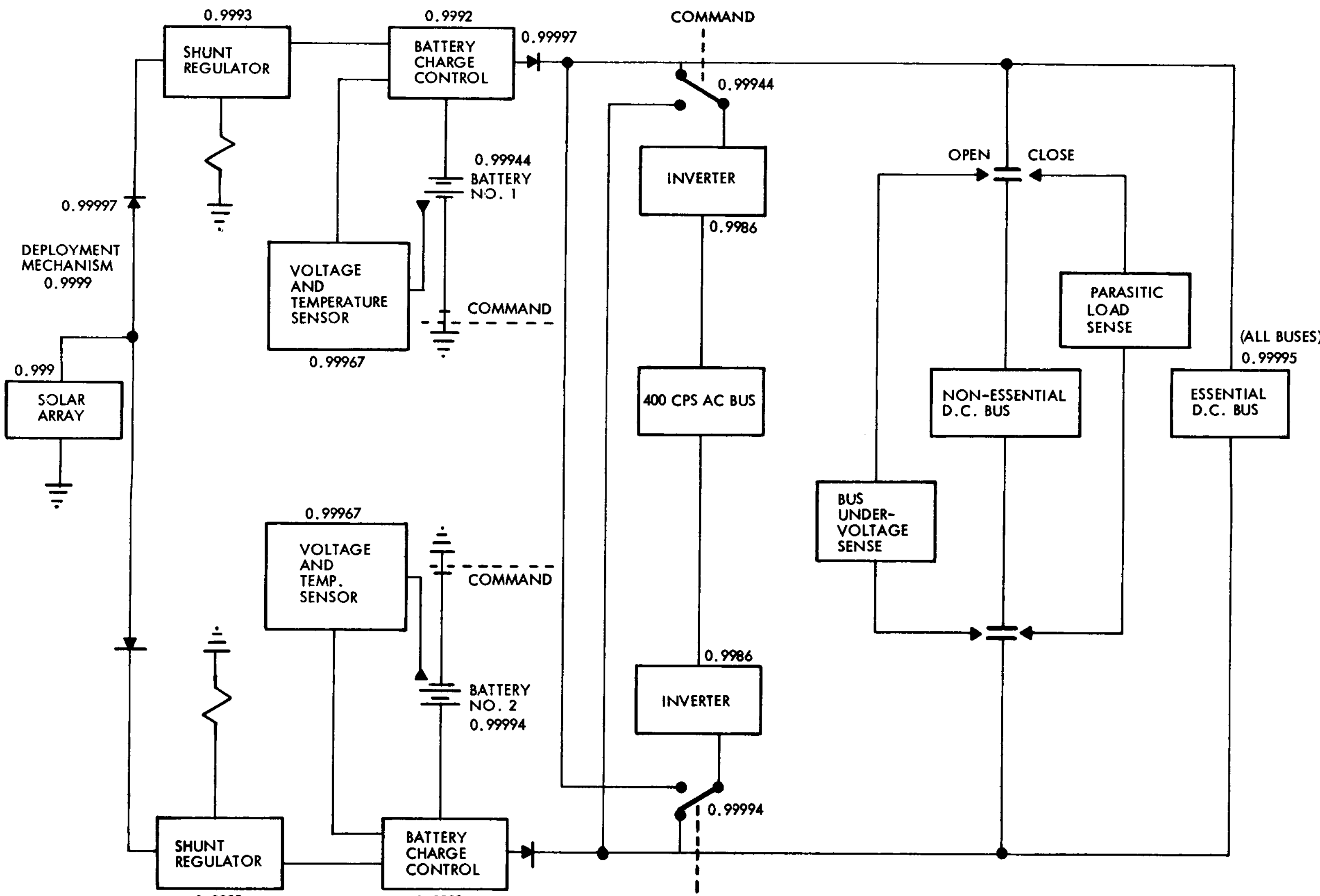


Retrieval, Canister Fluid and Life Support System

FAILURE MODE AND EFFECTS ANALYSIS
ELECTRICAL POWER*

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.999	1	Loss of power	1. Failure of solar array	No electrical output	Inoperative	Aborted	1 in 1,000	None. The solar array has built-in redundancies
0.999	2		2. Degraded solar array	Reduced electrical output	Degrades performance. Noncritical loads will be switched off	Reduced quality of data		
0.9999	4		3. Diode failure	Will not pass current	Loss of one leg of the actively redundant battery charging system	Will not affect mission success. Single battery loop can take entire electrical load	1 in 10,000	1
0.9993	4		4. Shunt regulator failure	Loss of control of bus voltage	Degraded bus voltage, switch battery of the line, run on other actively redundant battery loop	As above	1 in 1,400	
0.9992	4		5. Battery charge control (with voltage and temperature sensors)	Will not limit charging current at high battery temperature	Life expectancy of battery will be reduced. No obvious effects	Will not endanger mission success as electrical system still has active redundancy with adequate capacity in each loop	1 in 1,250	1
0.99944	4		6. Battery failure	No output	None. Inverter can be operated from other regulator-battery set. Switch dead battery off the line	Will not affect mission success	1 in 2,000	1
0.9986	4		7. Inverter failure	No output	None. Switch battery to redundant inverter	Will not affect mission success	1 in 700	1

*Overall reliability 0.9988

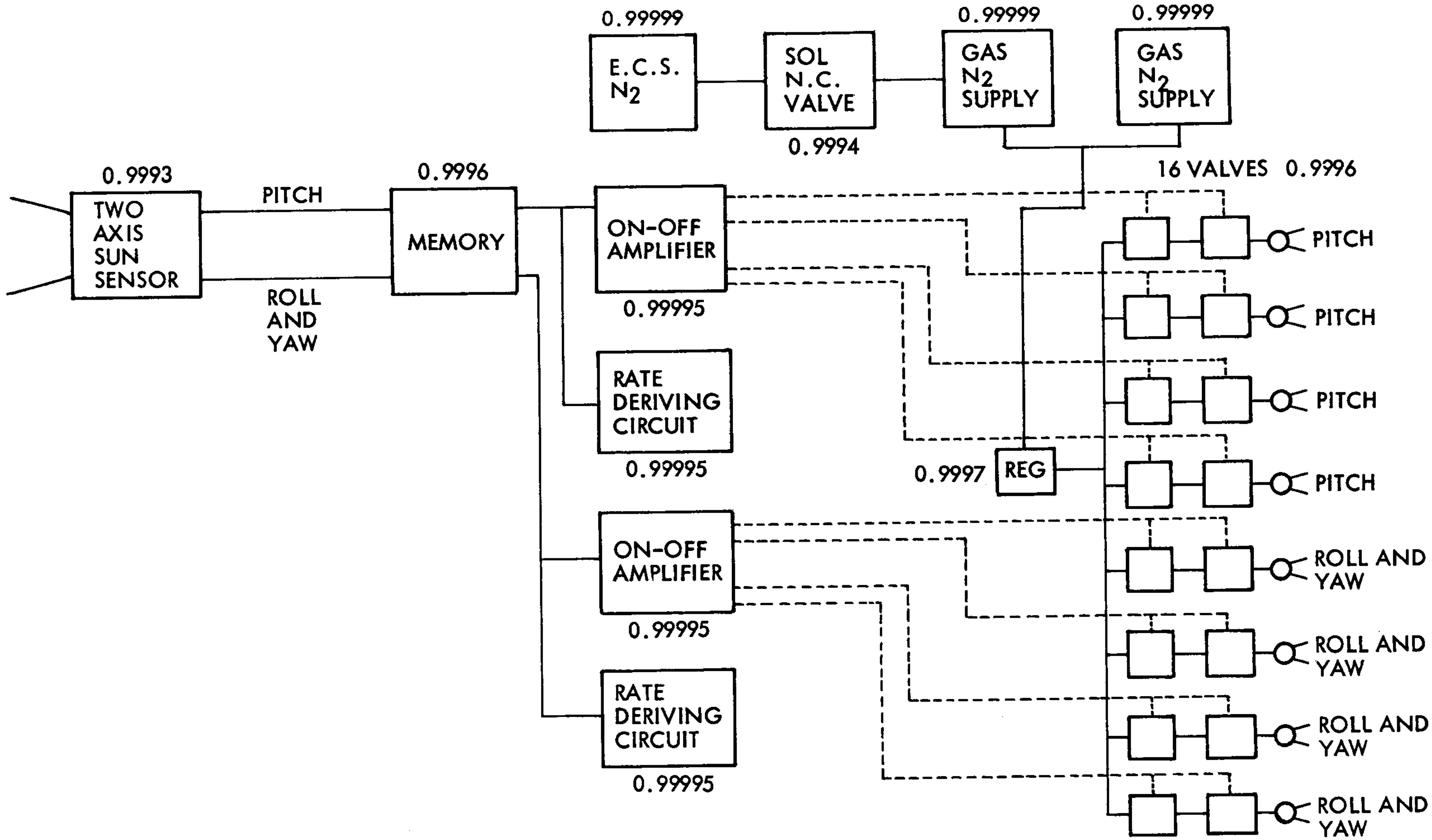


Electrical Subsystem – Reliability 0.9988

FAILURE MODE AND EFFECTS ANALYSIS
ATTITUDE CONTROL*

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.9997	1	1. Loss of control capability	1. Gas regulator	Supplies no gas or inadequate gas to the jets	Loss of attitude control. Solar array cannot be sun oriented	Aborted due to gradual loss of electrical power and all power operated subsystems	1 in 3,300	None
0.99995	1		2. On-off amplifier in pitch circuit	No output signal to pitch control valves	Loss of pitch control. Solar array cannot maintain sun orientation	As above	1 in 20,000	None
0.99995	1		3. On-off amplifier in roll and yaw circuit	No output signal to roll and yaw control valves	Loss of roll and yaw control. Solar array cannot maintain sun orientation	As above	1 in 20,000	None
0.9996	4		4. Control valve					
			(a) Fails open	Loss of on-off capability	None. Redundant series valve will take over	Will not endanger mission	1 in 2,500	1
			(b) Fails closed	As above	None. Parallel redundant valves will take over	Will not endanger mission		1
0.9993	1	2. Loss of control capability	5. Two axis sun sensor	Failure to give signals for pitch or roll and yaw	Loss of attitude control. Solar array not sun oriented	Aborted due to gradual loss of electrical power	1 in 1,400	1
0.999999 (Red)	1		6. Memory	Momentum wheel affected by loss of fan or fans	Satellite may go into spin during satellite night which may not be corrected each morning	An uncontrollable spin rate would abort the mission	1 in 10^6 with redundancy	Momentum wheel high flow fans have redundancy

*Overall reliability 0.9987

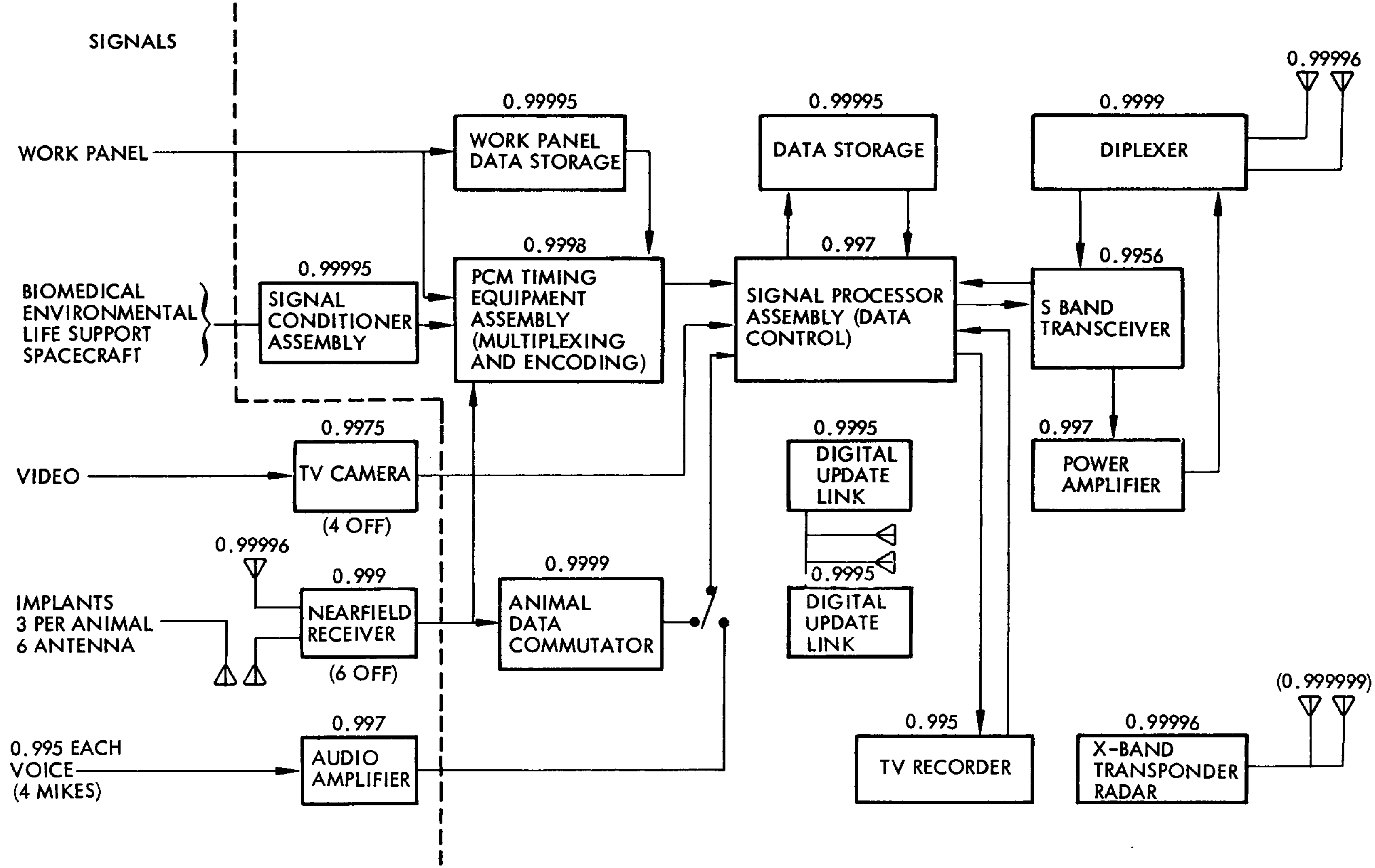


Attitude Control System _ Reliability 0.9987

FAILURE MODE AND EFFECTS ANALYSIS
AIRBORNE DATA MANAGEMENT*

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.99996	4	1. Loss of data	1. Antenna	No signals transmitted from one antenna	Loss of 50% of data transmission	Reduced data acquisition (quantity)	1 in 25,000	None
0.9999	4		2. Diplexer	As above	As above	As above	1 in 10,000	None
0.997	1		3. Power amplifier	Delivers no input to the diplexer and antennas	Loss of all data	Mission aborted	1 in 330	1
0.9956	1 or 2		4. S-band transceiver	Delivers no input to the antennas or at best degraded inputs in quality and quantity	Poor data in quantity and/or quality	Cause for abort depending on degree of degradation	1 in 227	1
0.997	4		5. Signal processor assembly data control	Processing loss of some data channel or channels	Loss of some data output	Will not affect mission success	1 in 330	None
0.99995	2 or 3		6. Data storage	Failure to store or readout data when required	Loss of some data	Reduction in quantity of data	1 in 20,000	None
0.9995	4		7. Digital update link	Failure to receive update commands	None. Redundant link will take over. Use can be made of the S-band transceiver as well	Will not affect mission success	1 in 2,000 each link	1
0.9999	4		8. Animal data commutator	No output from implants	Loss of real time bio-data	Little effect stored bio-data will be read out		
0.99995	4		9. Signal conditioning	One of numerous channels may be lost	Loss of biomedical, environmental, life-support or spacecraft data	Will not affect mission success unless CO_2 and/or O_2 partial pressure data is lost	1 in 20,000	
0.9975	4		10. TV equipment					
			(a) Camera	Loss of picture	Very little, lose one picture output signal	Reduces data but should not affect mission success	1 in 400	Already redundant
			(b) Controls	Loss of all picture channels (four)	All TV data lost			
0.992	4		11. Microphone or audio amplifier channel	No sound on one of four channels	Very little, system is redundant	Will not affect mission success	1 in 125 per voice channel	Already redundant

*Overall reliability 0.9497

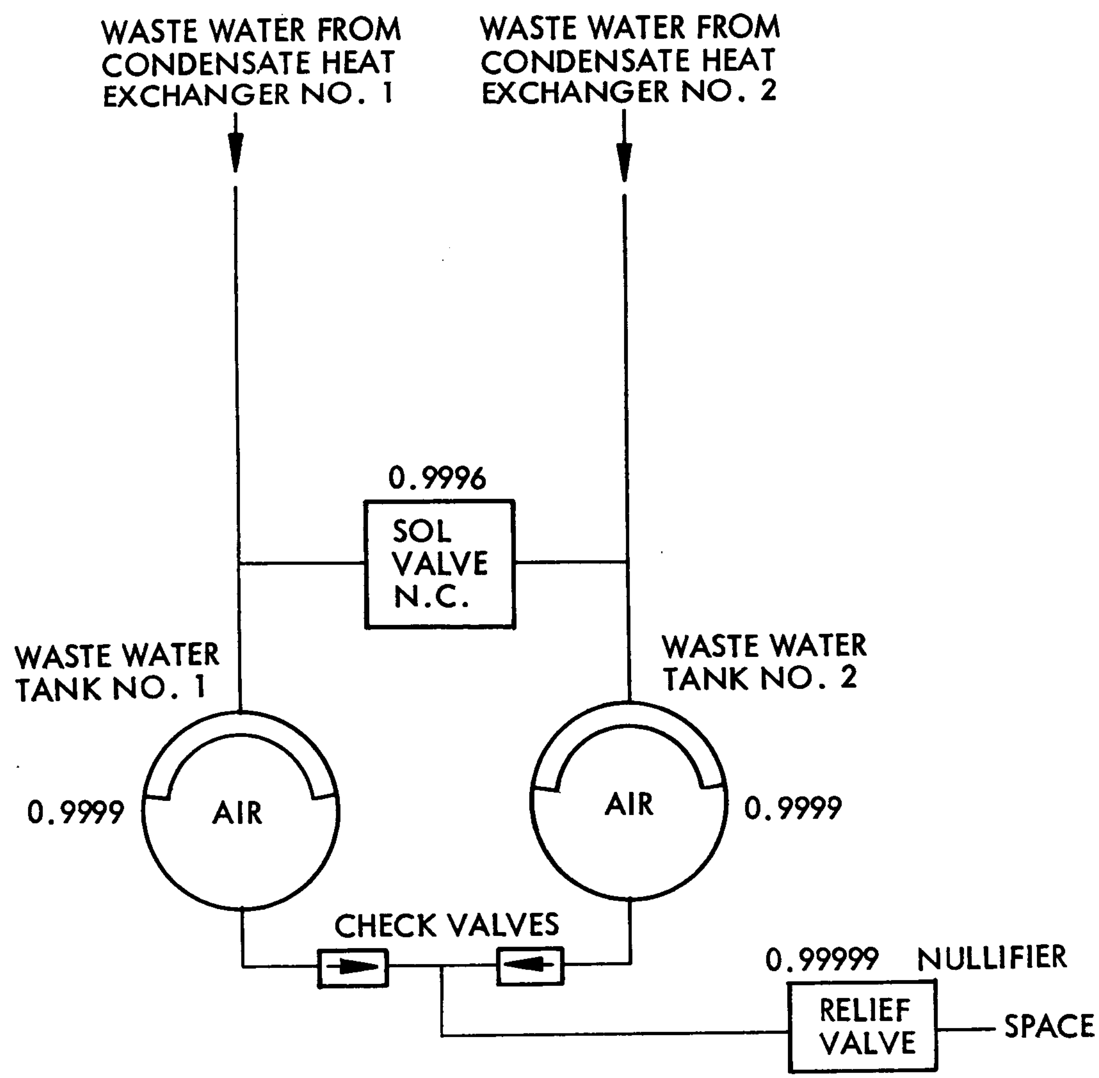


Airborne Data Management System – Reliability 0.9497

FAILURE MODE AND EFFECTS ANALYSIS
WASTE WATER STORAGE*

Reliability	Hazard Level	Mode of Failure	Cause of Failure	Effect on Component	Effect on Subsystem	Effect on Mission	Problem of Occurrence (Chance)	Recommended Redundancy
0.9999	4	Unequal filling of waste water tanks	1. Ruptured tank diaphragm	Water not stored but discharged to space	CG shift	Increased attitude control gas usage	1 in 10,000	None
0.9996	4		2. Solenoid valve fails to open and one primate dies	One waste water tank accumulates more water than the other	As above	As above	1 in 2,500	None
0.99999	2		3. Relief valve					
			(a) Fails open	Diaphragms reverse with subsequent loss of differential pressure across microporous filter	Water not collected in tanks, will accumulate in life cells	Will shorten mission duration	1 in 10,000	None

*Overall reliability 0.9994



Waste Water Storage – Reliability 0.9994

Appendix D
GLOSSARY

AAP:	Apollo Application Program
AGC:	Automatic Gain Control
AGE:	Aerospace Ground Equipment
ALDS:	Apollo Launch Data System
AM:	Airlock Module
A/S:	Ascent Stage
ATM-B:	Apollo Telescope Mount B (Type)
BMR:	Basal Metabolic Rate
BSR:	Basal Skin Resistance
C-BAND:	3900–6200 Mc
CM:	Command Module
CRT:	Cathode-Ray Tube
CRT:	Combined Readiness Test
CSM:	Command/Service Module (28 Days)
DMS:	Data Management System
D/S:	Descent Stage
DSIF:	Deep Space Instrumentation Facility
ECG:	Electrocardiogram
ECS:	Environmental Control System
EED:	Electro-Explosive Device
EEG:	Electroencephalogram
EMG:	Electromyogram
EMI:	Electromagnetic Interference
EOG:	Electro-Oculogram
ETR:	Eastern Test Range
EVA:	Extravehicular Activity

FM:	Frequency Modulation
G&N:	Guidance and Navigation
GFE:	Government Furnished Equipment
GSE:	Ground Support Equipment
GSFC:	Goddard Space Flight Center
GSR:	Galvanic Skin Response
IMBLMS:	Integrated Medical and Behavioral Laboratory Measurement System
IU:	Instrument Unit (Saturn Vehicle)
IVA:	Intravehicular Activity
JPL:	Jet Propulsion Laboratory
KBPS:	Kilobits Per Second
KSC:	Kennedy Space Center
L-CSM:	Logistic (Type) – CSM (96 Days)
LIEF:	Launch Information Exchange Facility
LM-A:	Lunar Module Ascent
LM/RACK:	Lunar Module (Carrier)
LMSC:	Lockheed Missiles & Space Company
LM/SHELTER:	Lunar Module Shelter
LRC:	Langley Research Center
LSS:	Life Support System
MCC:	Mission Control Center
MCC-H:	Mission Control Center – Houston
MCC-K:	Mission Control Center – Cape Kennedy
MDA:	Multiple-Docking Adapter
MOCR:	Mission Operation Control Room
MODEM:	Modulator-Demodulator
MSC:	Manned Spacecraft Center
MSFC:	Marshall Space Flight Center
MSFEB:	Manned Space Flight Experiment Board
MSFN:	Manned Space Flight Network
M&SS:	Mapping and Survey System

NASCOM:	NASA Communications Network
NM:	Nautical Miles
NRD:	National Range Documents (i.e., PRD, PSP, OR and OD)
NRZ:	Non-Return to Zero
OD:	Operations Directive
OPE:	Orbital Primate Experiment
OR:	Operations Requirements
OTB:	Ordnance Test Building
OWS-2:	Orbital Workshop Number 2
PAM:	Pulse-Amplitude Modulation
PCM:	Pulse-Code Modulation
PCMTEA:	Pulse-Code Modulation Timing Equipment Assembly
PLC:	Primate Life Cell (Payload)
PM:	Phase Modulation
PRD:	Program Requirements Document
PSP:	Program Support Plan
RACK:	Equipment Support Structure
RBC:	Red Blood Cell
RCM:	Refurbished Command Module
RCS:	Reaction Control System
RLSS	Regenerative Life Support System
RTG:	Radioisotope Thermoelectric Generator
S/AAP:	Saturn/ Apollo Applications Program
S-BAND:	1550–5200 Mc
SAB:	Spacecraft Assembly Building
SF:	Safety Factor
S-IB:	First Stage of Saturn IB
S-IC:	Saturn V First Stage
S-II:	Saturn V Second Stage
S-IVB:	Saturn V Third Stage or Saturn IB Upper Stage
SLA:	Spacecraft LM Adapter

SPS:	Service Propulsion System
SSR:	Staff Support Room
UHF:	Ultra-High Frequency (300–3000 Mc)
USB:	Unified S-Band
VHF:	Very High Frequency (30–300 Mc)
X-CSM:	Extended (Type) – CSM (56 Days)
ZPG:	Impedance Pneumogram

LIBRARY CARD ABSTRACT

This report describes the results of a preliminary design and mission analysis study for a spacecraft to orbit two Rhesus monkeys for a period of up to one year. The study includes the establishment of design criteria, review of Apollo Applications Program candidate launch and retrieval vehicles, and the tradeoff analyses which led to the preferred approaches for all subsystems. The preliminary designs of all spacecraft subsystems are documented in the form of drawings and written descriptions. A development plan is also included which outlines the steps required to continue the program through the flight phase.