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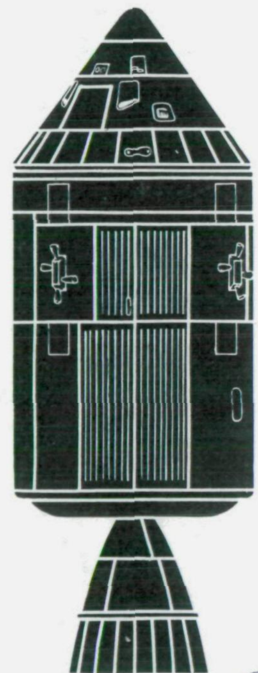

Final Report

PRELIMINARY DEFINITION PHASE
APOLLO EXTENSION SYSTEM

Service Propulsion System

NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION

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FINAL REPORT

PRELIMINARY DEFINITION PHASE APOLLO EXTENSION SYSTEM

[redacted]

Service Propulsion System

8 December 1965

Approved by

~~U. S. Government Agencies and Contractors Only~~

H. P. Burns
Chief Engineer

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NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

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FOREWORD

This document is submitted by the Space and Information Systems Division (S&ID) of North American Aviation, Incorporated, to the National Aeronautics and Space Administration Manned Spacecraft Center in partial fulfillment of the final reporting requirements of Contract NAS9-5017, "Preliminary Definition Study for Utilization of CSM for AES."

Reports being submitted under the subject contract are listed below. Data resulting from subcontractor studies or provided by other sources external to S&ID are included in the appropriate volumes. The reader is urged to refer to other documents in the final report series for further information not contained in this document.

Report No.	Title
SID 65-1145	Master Program Plan (Preliminary)
SID 65-1146	Manufacturing Plan (Preliminary)
SID 65-1147	Facilities Plan (Preliminary)
SID 65-1148	General Test Plan (Preliminary)
SID 65-1150	Configuration Management Plan (Preliminary)
SID 65-1151	Baseline Ground Operations Requirements Plan
SID 65-1517	Program Summary
SID 65-1518	Technical Summary
SID 65-1519	Subsystems Summary
SID 65-1520	Guidance and Control System
SID 65-1521	Communications and Data System
SID 65-1522	Instrumentations, Displays, and Controls
SID 65-1523	Environmental Control and Life Support Systems
SID 65-1524	Thermal Analysis
SID 65-1525	Power Generation and Distribution Systems
SID 65-1526	Cryogenic Storage System
SID 65-1527	Service Propulsion System
SID 65-1528	Reaction Control System
SID 65-1529	Spacecraft Design Summary
SID 65-1530	Structural Loads and Criteria
SID 65-1531	Structural Analysis
SID 65-1532	Mass Properties
SID 65-1533	Earth Recovery System
SID 65-1534	Systems Analysis Summary
SID 65-1535	Reliability Summary
SID 65-1536	Experimenters Design Guide

TECHNICAL REPORT INDEX/ABSTRACT

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DESCRIPTIVE TERMS

APOLLO EXTENSION SYSTEMS (AES)
AES-SPS CONFIGURATION A
AES SERVICE PROPULSION SYSTEM (AES-SPS)
AES-SPS CONFIGURATION B
AES PROPELLANT UTILIZATION GAGING SYSTEM (AES-PUGS)

ABSTRACT

THIS REPORT PRESENTS THE RESULTS OF THE PRELIMINARY DEFINITION PHASE STUDY OF THE AES SERVICE PROPULSION SYSTEM. THE STUDY APPROACH CONSISTED OF COMPARING THE AES AND APOLLO MISSION PROFILES AND PROPULSION PERFORMANCE REQUIREMENTS TO DEFINE DIFFERENCES. A TRADEOFF STUDY WAS PERFORMED TO DETERMINE WHAT IS THE MINIMUM MODIFICATION TO OFF LOAD PROPELLANT.

THE RECOMMENDED AES SPS IS ESSENTIALLY THE SAME AS THE APOLLO BLOCK II SPS WITH THESE EXCEPTIONS: (1) ADDITION OF A ZERO GRAVITY LEAK DETECTION HELIUM AND PROPELLANT GAGING SYSTEM, (2) PROVISION FOR EXTERNAL NITROGEN SERVICING FOR THE SPS ENGINE, (3) A CHANGE OF THE HELIUM PRESSURIZATION SYSTEM TO ELIMINATE BACKFLOW LEAKAGE OF PROPELLANT TO THE REGULATORS, AND (4) A MINOR REARRANGEMENT OF THE NON-FLOW COMPONENTS OF THE HELIUM PANEL TO PERMIT CHANGES IN THE PANEL CONFIGURATION REQUIRED BY AES SM SPACE ALLOCATIONS.

FOR MISSION REQUIRING LESS THAN 21,000 POUNDS OF PROPELLANT, A CONFIGURATION WHICH DELETES THE STORAGE TANKS AND POSSIBLY ONE HELIUM TANK IS RECOMMENDED. THIS CONFIGURATION RESULTS IN A SAVING OF 586 POUNDS IF BOTH BLOCK II HELIUM TANKS ARE RETAINED OR 864 POUNDS IF ONE HELIUM TANK IS DELETED. THE DELETION OF ONE HELIUM TANK WITH 21,000 POUNDS OF PROPELLANT RESULTS IN A 10 PERCENT DECAY IN CHAMBER PRESSURE AND HENCE THRUST OVER THE LAST 30 SECONDS OF THRUST.

Report No.	Title
SID 65-1537	Experiment Identification Descriptions
SID 65-1538-1	Mission Description—Flight 211
SID 65-1538-2	Mission Description—Flight 507
SID 65-1538-3	Mission Description—Flight 509
SID 65-1538-4	Mission Description—Flight 511
SID 65-1538-5	Mission Description—Flights 214/215
SID 65-1538-6	Mission Description—Flights 216/217
SID 65-1539	Ground Support and Logistics
SID 65-1541	Interface Methodology
SID 65-1542	Functional Flow Diagrams—Lunar Polar Orbit Reference Mission
SID 65-1543-1	Allis-Chalmers Fuel Cell Study—Technical Summary
SID 65-1543-2	Allis-Chalmers Fuel Cell Study—Program Analysis
SID 65-1544-1	Land Landing System—Technical Summary
SID 65-1544-2	Land Landing System—Program Analysis
SID 65-1545-1	Phase I Experiments Integration—Program Analysis
SID 65-1545-2	Phase I Experiments Integration—Cost Data
SID 65-1546	Final Briefing
SID 65-1547	Performance Analysis Phase II Flights
SID 65-1571	Program Costs

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ILLUSTRATIONS

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INTRODUCTION

This report presents the results of the Preliminary Definition Phase (PDP) study of the use of the Apollo Block II service propulsion system (SPS) for the Apollo Extension System (AES) program. It includes a statement of AES requirements for the system based on four representative reference missions, a preliminary evaluation of the capability of the Block II SPS to meet these requirements, and a description of the system design changes required. The study ground rules required that the Apollo Block II SPS be used with minimum change. It was also assumed that the Block II system will have been fully qualified for the performance and environmental requirements of Apollo 14-day earth orbital and 10-day lunar missions.

The study approach consisted of comparing the AES and Apollo mission profiles and propulsion performance requirements to define differences. The most significant difference is the AES increase in mission duration to 45 days. This required study emphasis of long term space environmental exposure effects and the definition of system changes for coast life extension. These factors involved both materials life in the space environment and a reassessment of SPS propellant and helium fluid leakage implications.

The comparison of performance requirements revealed no differences between the AES and Apollo missions except that a large number of AES earth orbit missions require only a fraction of the Block II propellant capacity. Since a study ground rule required that potential weight savings be identified, a trade-off study of reduced tankage configurations was conducted and resulted in a recommended alternate design to be used on these missions.

The report is organized in a manner which follows the chronological order of the study. The AES propulsion requirements are developed together with a definition of the Block II SPS capabilities. Since numerous design changes have been made to the Block II SPS, Appendix A contains a Block II specification revised to reflect all changes which occurred up to 1 October 1965. The SPS trade-off analyses and environmental effects studies are summarized. Finally, a concise description of the AES SPS is presented.

For a description of other subsystems which interface with the SPS, the reader is referred to the reports listed in the Foreword. The Subsystems Summary report, SID 65-1519, provides a brief description of the contents of all subsystem reports.

I. AES REQUIREMENTS

The requirements imposed on the service propulsion system (SPS) by the AES program are derived from the characteristics of four representative AES reference missions. A more detailed description of these reference missions is contained in the Systems Analysis report, SID 65-1534. These missions, briefly described below, were used as the basis for all analysis and trade-offs conducted during this study. As can be seen from the reference missions, it is mandatory to retain the Block II SPS tankage as the basic AES configuration. Several other missions, which have been defined only in a preliminary manner, require considerably less SPS propellant than the Block II and logically lead to an alternate selection of a reduced propellant tankage configuration.

Maximum duration of the reference missions is 45 days, whereas the Block II requirement is a maximum of 14 days. The longer duration places more stringent requirements on the SPS. The longer exposure to propellants and combined environments is an area that requires extensive analysis and eventually demonstrated capability by test.

AES MISSION REQUIREMENTS

The requirements imposed on the SPS by the AES program are derived from the characteristics of the four AES reference missions.

REFERENCE MISSION NUMBER 1

The mission experimental phase is conducted in a 200 nautical mile, earth polar orbit with a three man crew. The total flight duration is forty-five days. The SPS is used for two maneuvers with ideal velocity requirements as noted:

Final orbit injection	176.3 fps
De-orbit	360.1 fps

REFERENCE MISSION NUMBER 2

The mission experimental phase is conducted in a synchronous (19,321 nautical mile altitude) equatorial earth orbit with a three man crew. The total flight duration is forty-five days. The SPS is used for two maneuvers with ideal velocity requirements as noted:

Final orbit injection	0 to 3000 fps (SC weight dependent)
De-orbit	4905 fps

REFERENCE MISSION NUMBER 3

The mission experimental phase is conducted in an 80 nautical mile, lunar polar orbit with a three man crew. The lunar orbital phase has a duration of 28 days, and the total mission duration is thirty-four days. The SPS is used for several maneuvers with ideal velocity requirements as noted (a non-free-return trajectory is assumed):

Translunar midcourse corrections	68 fps
Lunar orbit insertion	2700 fps
Deviations	93 fps
Transearth injection	5540 fps
Transearth midcourse corrections	216 fps
Deviations	382 fps

REFERENCE MISSION NUMBER 4

The CSM is used to deliver the LEM taxi to the surface of the moon for a lunar stay-time of up to 14 days. The flight has a three-man crew with one crewman remaining in the CM during the lunar orbital phase. The lunar phase is at 80 nautical miles in a low inclination orbit. The total mission duration is 20 days. The SPS is used for several maneuvers with ideal velocity requirements as noted:

Translunar midcourse corrections	68 fps
Lunar orbit insertion	3445 fps
LEM Rescue	680 fps
Deviations	94 fps
Transearth injection	3088 fps
Transearth midcourse corrections	40 fps
Deviations	62 fps

METEOROID EFFECTS

The AES meteoroid environmental requirement and an evaluation of SPS tank and lines protection is described in report SID 65-1534, Systems Analysis Summary.

RADIATION EFFECTS

The AES radiation model requirement is defined in report SID 65-1534, Systems Analysis Summary. The effect on SPS materials is to be determined on the basis of different solar proton events: one with a 0.1 percent probability of occurrence and one with a 1.0 percent probability. The referenced report also includes calculation of zone dosages at various locations of the SPS.

PERFORMANCE REQUIREMENTS

With respect to primary performance characteristics, AES missions have been designed to fall within the anticipated maximum propulsion capability of the Block II SPS. These primary characteristics are shown in Table 1.

PROPELLANT REQUIREMENTS

The AES Phase II usable propellant requirements for each flight were estimated (see report SID 65-1547, Performance Analysis, Phase II Flights) and are presented in Figure 1.

The Block II maximum usable propellant capacity of 39,700 pounds is required for some flights. This slightly exceeds the nominal requirement of 38,000 pounds for the Block II LOR mission. On six other flights, notably low inclination earth orbital missions, the propellant requirement is less than 10,000 pounds.

SPECIFIC IMPULSE

The current prediction of attainable three sigma low specific impulse for the Block II SPS is 311.7 seconds. The AES requirement is constrained to this value although higher performance would increase AES CSM utility.

THRUST

The nominal thrust predicted for the Block II SPS is 20,000 pounds. The AES requirement is constrained to this value. Additional thrust is of no advantage except for suborbital abort and flights using SPS for earth orbital injection where an increase in payload would be possible.

DUTY CYCLE

The maximum number of in-flight starts required for AES missions considered to date does not exceed 15. For these missions, the minimum coast time between long burns is not less than two hours. Neither of these duty cycle characteristics is a firm requirement at this time, however, since AES missions are not yet final. Therefore, SPS capability should permit flexibility in this area. The maximum coast period between firings is 45 days.

MINIMUM IMPULSE AND REPEATIBILITY

The AES Phase II requirements (see Table 1) for SPS minimum impulse, shutdown impulse, and impulse repeatibility have been assumed to be identical to the Block II requirement. This is subject to verification during guidance and control requirements definition in future studies.

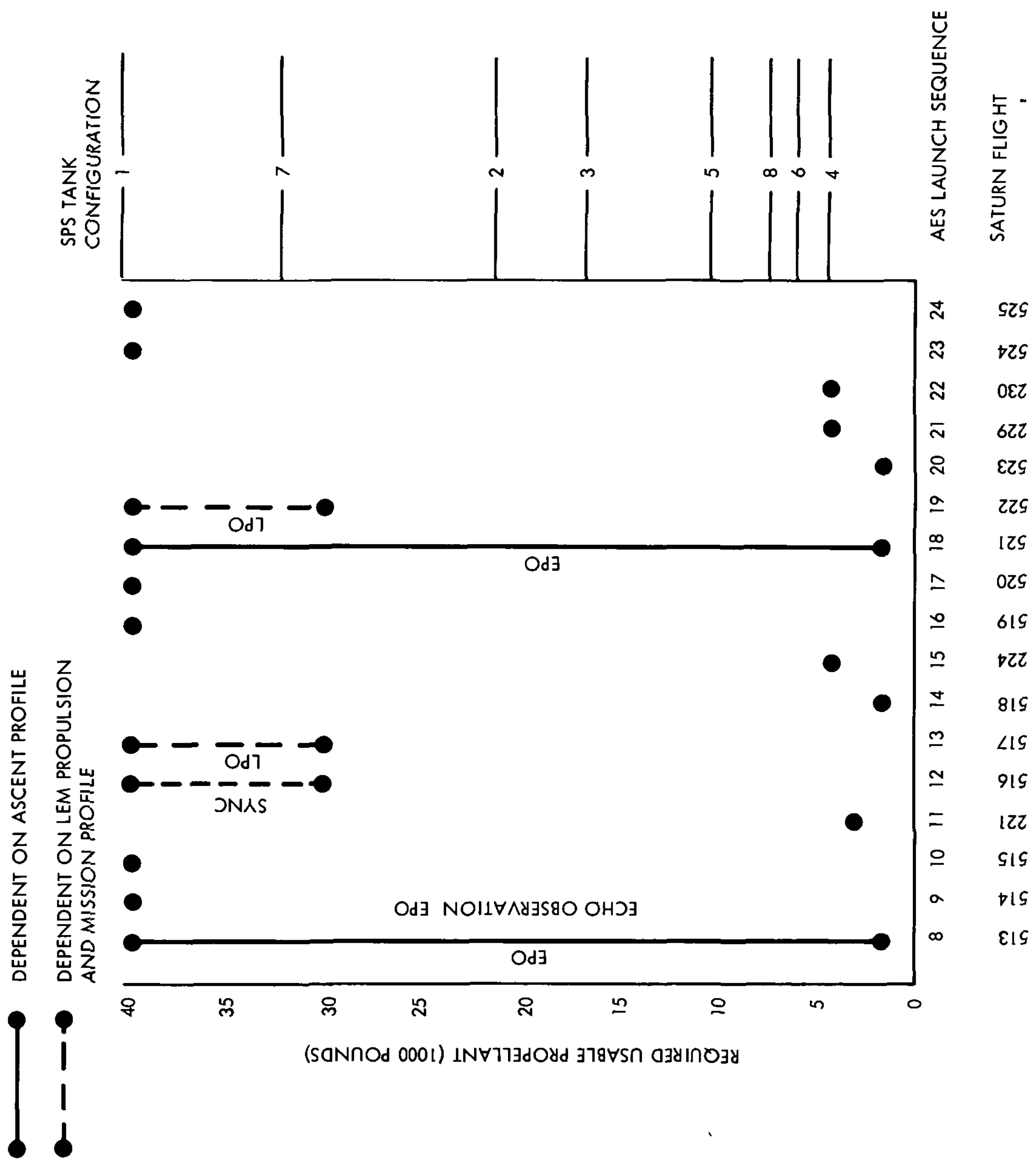

Figure 1. AES Phase II Mission SPS Propellant Requirements

Table 1. System Requirements

Reliability Goal (LPO Mission)	
Mission Success	0.99868
Crew Safety	0.9999
Dry Weight (including tanks)	2,830 lbs
Usable Propellant (maximum)	39,700 lbs
Thrust (nominal)	20,000 lbs
Specific Impulse (3-sigma minimum)	311.7 sec
Minimum Impulse per Start	5000±200 lb sec
Shutdown Impulse	8,830 to 14,200±300 lb sec

II. BLOCK II SERVICE PORPULSION SYSTEM

The initial baseline configuration of the SPS for AES is the Apollo Block II design. Since a current definition of that design does not exist in an unamended document, it is defined in Appendix A for AES program purposes. Appendix A consists of the applicable sections of the Block II CSM Technical Specification (SID 64-1344) and CSM Master End-Item Specification (SID 64-1345) revised to reflect current information, including the recent change from a 2 to a 1.6 O/F mixture ratio.

To further assist in establishing an AES baseline, the following sections contain descriptions of the changes from Block I to Block II and a report on the present status of Block II development.

BLOCK I TO BLOCK II CHANGES

The major differences between Block I and Block II service propulsion systems are:

Block II tanks are shorter than those of Block I reducing propellant capacity from 45,000 pounds to 39,700 pounds; this resulted in tank weight reduction and corresponding changes to the gaging system probes. Block II tank shells and heads are reduced in thickness from those of Block I. Block II tanks are designed to a limit pressure of 225 psia instead of the 240 psia for Block I tanks. The relief valve and burst diaphragm are revised accordingly.

Since less propellant is to be expelled, less helium is required. By reducing the design loading pressure, a reduction in wall thickness was effected on Block II helium vessels. Outside helium tank dimensions are identical.

The propellants are injected into the thrust chamber at a nominal mixture ratio (weight rate of oxidizer to fuel) of 2.0 to 1 for Block I and 1.6 to 1 for Block II.

The Block II gaging equipment is the same as Block I with the following exceptions: (1) displays are changed to utilize electroluminescence; (2) propellant remaining is displayed in percent; and (3) point sensor locations are at the same percent (of full) point as in Block I.

BLOCK II DEVELOPMENT STATUS

The development status of the Block II service propulsion subsystem is summarized as follows: The basic subsystem design for the Block II spacecraft has been released and development is in progress.

The in-house laboratory development tests are complete with the exception of tank-oxidizer compatibility and oxidizer tank door seal evaluation tests currently in progress. Qualification test status is currently as outlined in Table 2.

Full qualification of the pressurization and propellant supply components is limited to couplings and connectors.

The release of revised designs and specifications for the remaining Block II components has extended qualification completion dates; in most cases, the dates are yet to be determined. The lines and helium panel are to be qualified during major ground tests. The final qualification of propellant retention reservoirs and screens will be accomplished during early Block I flight tests.

The qualification status of the Block II propellant utilization gaging system is not yet determined as a result of new design requirements, including a tank size change and a change to 1.6:1 O/F mixture ratio.

The engine is currently in Block I qualification at Aerojet-Sacramento, and Arnold Engineering Development Corporation. Block II qualification is scheduled for completion in December, 1966. All engine components, while requiring some component level qualification, will attain final qualification as part of the complete engine assembly.

Table 2. Block II Service Propulsion Subsystem Development Status

Component	Qual Complete	Est. Qual Completion	Vendor Qual	NAA Lab Test	Remarks
Pressurization Equipment					
Helium Pressure Vessel	No	TBD	X		Tank wall thickness reduced
Components					
Test Point Coupling	Yes		X		
Helium Fill and Drain Coupling	Yes		X		
Solenoid Valve	No	TBD	X		Operating temperature -20 F (was +30 F)
Helium Regulator	No	TBD	X		Internal leakage reduced from 80 std. cc/min to 40 std. cc/min. Addition of inlet filter.
Check Valve (Fuel)	No	TBD	X		Operating temperature -105 F He (was -65 F)
Check Valve (Oxidizer)	No	TBD	X		New seal material for temperature and propellant compatibility
Relief Valve	No	TBD	X		New pressure setting
He Tank Connection	Yes			X	Also major ground tests (AF 001)
Lines	No	TBD			To be qualified during major ground tests for Block II configuration

Table 2. Block II Service Propulsion Subsystem Development Status (Continued)

Component	Qual Complete	Est. Qual Completion	Vendor Qual	NAA Lab Test	Remarks
Fuel Supply Equipment					
Tanks	No	TBD	X		Tanks shortened
Tank Door & Seal Assy.	No	TBE	X		Inner seals changed to Raco (were butyl o-rings)
Components					
Fill & Drain Coupling	Yes		X		
Fuel Vent Coupling	Yes		X		
Residual Drain Coupling	Yes		X		
Test Point Coupling	Yes		X		
Heat Exchanger	No	TBD	X		Adapt to mixture ratio of 1.6:1 Add test port
Flex Connector	No	TBD	X		Adapt to mixture ratio of 1.6:1
Fuel Retention Equipment					
Reservoir	No	TBD	X	X	Also flight tests
Umbrella Screens	No	TBD	X	X	Also flight tests
Lines	No	TBD			To be qualified during major ground test for Block II configuration

Table 2. Block II Service Propulsion Subsystem Development Status (Continued)

Component	Qual Complete	Est. Qual Completion	Vendor Qual	NAA Lab Test	Remarks
Oxidizer Supply Equipment					
Tanks	No	TBD	X		Tanks shortened
Tank Door & Seal Assy.	No	TBD	X		Inner seals changed to Raco (were butyl o-rings)
Components					
Fill & Drain Coupling	Yes		X		
Oxidizer Vent Coupling	Yes		X		
Residual Drain Coupling	Yes		X		
Test Point Coupling	Yes		X		
Heat Exchanger	No	TBD	X		Adapt to mixture ratio of 1.6:1 Add test port
Flex Connector	Yes		X		
Oxid Retention Equipment					
Reservoir	No	TBD	X	X	Also flight tests
Umbrella Screens	No	TBD	X	X	
Lines	No	TBD			To be qualified during major ground test for Block II configuration

Table 2. Block II Service Propulsion Subsystem Development Status (Continued)

Component	Qual Complete	Est. Qual Completion	Vendor Qual	NAA Lab Test	Remarks
PUGS Equipment					PUGS - Propellant Utilization and Gaging System
Display Panel	No	TBD			New design requirements
Control Unit	No	TBD			New design requirements
Sensing Probe	No	TBD			New design requirements
PU Valve	No	TBD			New design requirements
Engine Assembly	No	12/15/66	X		Final qualification by complete assembly
Nozzle Extension					
Thrust Chamber Assy.					
Injector & Manifold					
Ablative Chamber					
Main Propellant Valve Assy					
Pneumatic Actuators					
Valve					
Fill Vent Couplings	Yes		X		
Vent & Drain Couplings	Yes		X		

Table 2. Block II Service Propulsion Subsystem Development Status (Continued)

Component	Qual Complete	Est. Qual Completion	Vendor Qual	NAA Lab Test	Remarks
Drain & Vent Lines	No	12/15/66	X		
Valve Actuation Equipment					
Pressure Vessel (GN_2)					
Components					
Solenoid Pilot Valve					
Pressure Regulator					
Relief Valve					
Fill & Vent Coupling					
Lines					
Gimbal Assembly					
Gimbal Ring & Bearings					
Actuators					
Electrical Harness					

III. SERVICE PROPULSION SYSTEM TRADE-OFFS

This section presents an evaluation of the Apollo Block II SPS for use on AES Phase II missions. It reviews the requirements enumerated in the preceding section and describes alternative and recommended changes needed to meet these requirements.

Since the AES Phase II missions have been designed to fall within the maximum performance capability of the Block II SPS, no evaluation of its primary performance characteristics is required. Two exceptions are an evaluation of burned weight savings obtainable on off-loaded missions through reduced propellant tankage, and an investigation of the feasibility of supplementing SPS capability through the use of LEM propulsion.

REDUCED PROPELLANT TANKAGE

Several AES Phase II missions require considerably less than the maximum propellant load and can be benefited by the burned weight and volume savings attainable by eliminating excess tankage on these flights instead of merely off-loading propellant. The weight saving attained increases with the reduction in capacity of the configuration selected as is indicated by Figure 2. However, the complexity and cost of the changes required varies widely over this range. For this reason the study investigated discrete design points of the curve in detail. Configurations studied required only available hardware and minimum related changes and yet covered a wide variety of propellant loadings.

A total of eight tank configurations were considered that utilize hardware available from the Apollo and LEM programs. Of these, the simplest configuration (involving only the removal of the propellant storage tanks and related minor changes to gaging and plumbing) was selected. This configuration has a capacity for 53 percent of the maximum usable propellant load and saves 586 pounds of burned weight with minimal change cost.

In the following paragraphs, the initial trade study is discussed followed by a supplementary study of the several helium vessel options possible with the selected propellant tank configuration.

INITIAL TRADE STUDY

The eight configurations covered in the initial study are shown in Figure 3. Table 3 contains detailed descriptions of each case and the associated changes required. The table also shows the results of evaluation of the trade factors involved: weight and volume savings, usable propellant, cost, reliability, and test program impact. Center of gravity excursion limits were also evaluated.

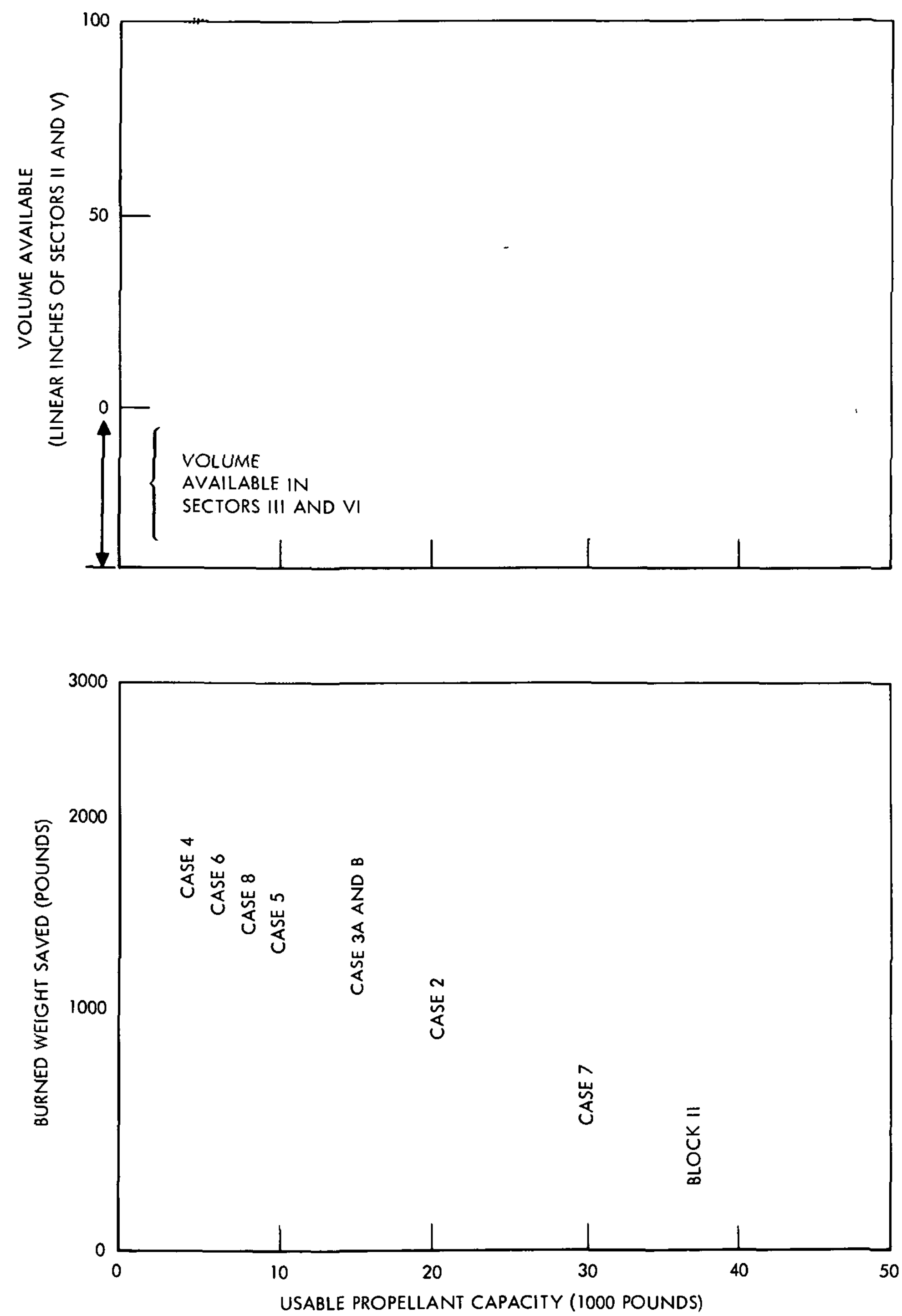

Figure 2. Weight and Volume Saving Versus Usable Propellant Capacity

CASE 3A

OXIDIZER BAY III
FUEL BAY IV

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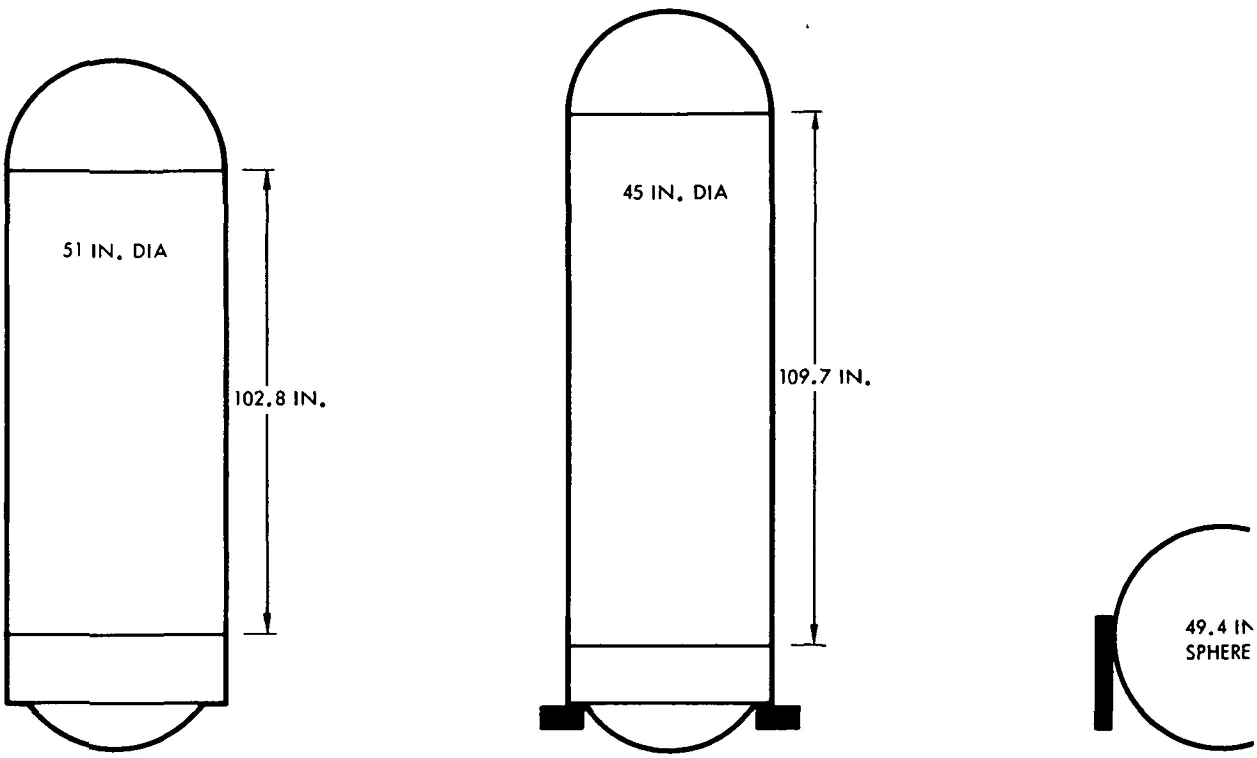


CASE 2

OXIDIZER BAY II
FUEL BAY V

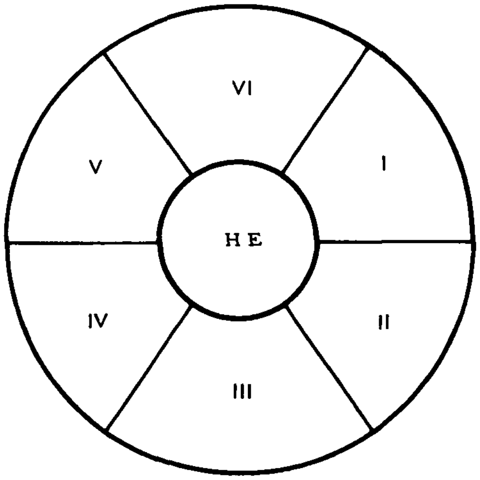
CASE 3B

OXIDIZER BAY II
FUEL BAY V
ADAPTER REQUIRED

CASE

OXIDIZER B
FUEL BA
ADAPTER REQ

Fig 3 1



SM BAYS LOOKING FORWARD

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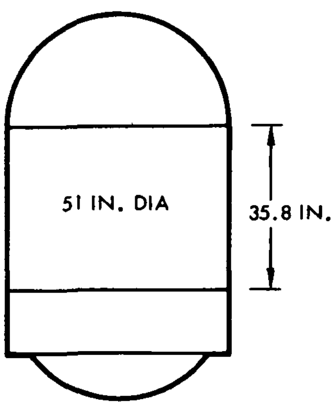

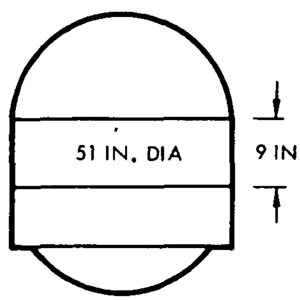

4

.Y II
Y V
UIRED

CASE 5

OXIDIZER BAY II
FUEL BAY V

CASE 6

OXIDIZER BAY II
FUEL BAY V

Fig 3

2

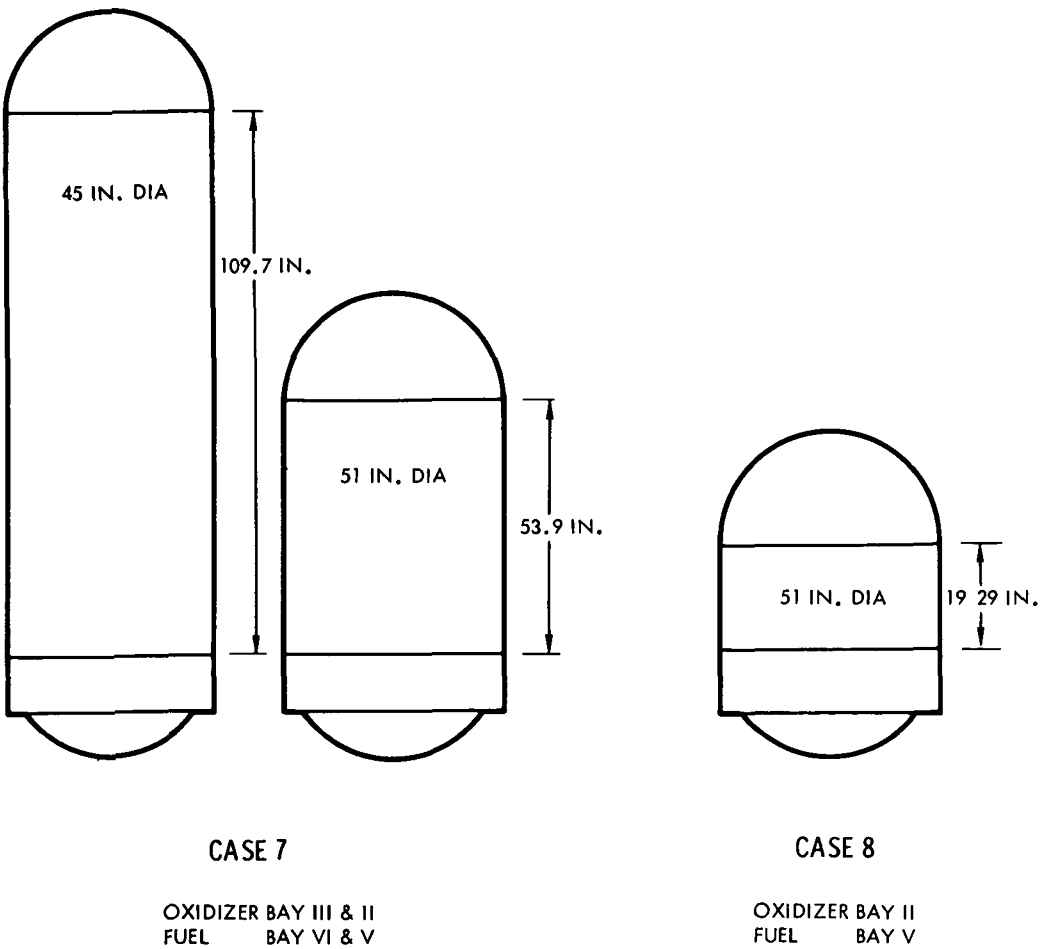

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Figure 3, Reduced Tankage Configurations

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Tank Configuration	1	2	3A	
Description	Block II	51-inch sumps	45-inch sumps Sectors III and VI	45 Se
Useable Propellant MR 1 6 (Pounds)	38,000 (nominal	21,040	16,550	16,
*Residual Propellant	543	435	419	41
Storage Tank Type Diameter (Inches) Cylinder Length (Inches) Oxidizer Location (Section) Fuel Location (Sector)	 Block II 45 III VI	 None — — — —	 None — — — —	 No — — — —
Sump Tank Type Diameter (Inches) Cylinder Length (Inches) Oxidizer Location (Sector) Fuel Location (Sector)	 Block II 51 102 8 II V	 Block II 51 102 8 II V	 Block II storage 45 109 7 III VI	 Bl 45 10 II V
Helium Tanks Number and Type Diameter (Inches) Fill Pressure (PSIG) Qualified Pressure Helium Weight (Pounds)	 Two Block II 40 7 OD 3585 3685 88 5	 One Block I 40 7 OD 3920 4000 47 0	 One Block II 40 7 OD 3200 3685 39 4	 On 40 32 36 39
Tank Internals Retention Reservoir Retention Screens Gaging Screens Tank Door	 Block II 51-inch, 58-inch height Block II 51-inch Block II Block II	 Block II 58-inch height Block II Block II sump omit storage Block II	 Block II 58-inch height Block I fuel or adapt Block II Change Block II sump Omit Block II storage Block II	 Bl 5 Bl o Ch On Bl
Tank System Weight Saving Including Residuals (Pounds)	None	864	1057	10
Volume Saving	None	Sectors III and VI	Sectors II and V	Se
Hardware Changes	None	Remove 45-inch storages Reroute helium plumbing Remove one helium bottle Adapt gauging control unit	Remove 51-inch sumps Reroute all plumbing Adapt internals Remove one helium bottle	Re Mo Ad Re Ad Re
C G Limits X-Y Plane X-Z Plane	 -0 5 to 2 0° 0 6 to 3 0°	 0 1 to 2 85° 0 to 1 9°	 0 1 to 0 8° 0 2 to 3 8°	 0 0
Reliability of Tanks (45 Days) Failures per 10^6 Missions	204	102	102	10
GSE and Ground Operations Changes	None	Minor ACE modification for PUGS	------------	--
Development Testing and/or Requalification	45-day environment	45-day environment Requalify helium tank	45-day environment Major ground-test verification firings	45
Nonrecurring Δ Price (dollars)		969,000	Not estimated	No
Recurring Δ Price/Unit (dollars)		- 213,000	Not estimated	No

*Includes all structure, propellants, residuals, engines, and lines, etc
Loading tolerance not included

Table 3 (1)

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Table 3. Tank Tradeoff Matrix

3B	4	5	6	7	8
inch sumps tors II and V	Two LEM ascent tanks	10,000-pound 51-inch sumps	Minimum tank 51-inch sumps	80% Block II Reduced 51-inch sumps	Two LEM descent tanks
550	4270	10,000 (nominal)	5830	30,400	7500
	393	405	397	527	400
ie	None — — — —	None — — — —	None — — — —	Block II 45 III VI	None — — — —
ck II storage 7	LEM ascent 49 4 ID Zero II V	Block II shortened 51 35 8 II V	Block II shortened 51 9 0 II V	Block II shortened 51 53 9 II V	LEM descent 51 19 29 II V
Block II 7 OD 0 5 4	Two LEM ascent 22 OD 3500 3500 11 6	One LEM descent 33 3 ID 3400 3500 23 7	One LEM descent 33 3 ID 2400 3500 17 4	Two Block II 40 7 OD 2740 3685 67 9	One LEM descent 33 3 ID 2800 3500 20
ck II -inch height ck I fuel adapt Block II nge Block II sump it Block II storage ck II	Block II shortened Block II adapted Block II sump adapted LEM adapted	Block II (may be shortened) Block II Block II shortened Block II	Block II shortened Block II Block II shortened Block II	Block II shortened Block II Block II shortened Block II	Block II shortened Block II Block II shortened Block II
7	1545	1236	1341	246	
tors IV and VI	Sectors III and VI and Δ of 105 inches in Sectors II and V	Sectors III and VI and Δ of 67 inches in Sectors II and V	Sectors III and VI and Δ of 93 8 inches in Sectors II and V	Δ of 48 9 inches in Sectors II and V	Sectors III and VI and Δ of 83 5 inches in Sectors II and V
nove 51-inch sumps e 45-inch tanks tank supports oute helium plumbing pt internals nove one helium bottle	Remove Block II tanks Add two LEM ascent tanks Add tank supports Reroute helium plumbing Adapt internals Remove Block II helium bottles Add two LEM ascent bottles	Remove 45-inch storage Shorten 51-inch tanks Reroute helium plumbing Remove helium bottles Add LEM descent helium bottle Adapt gauging system	Remove 45-inch storage Shorten 51-inch tanks Reroute helium plumbing Adapt internals Remove helium bottles Add LEM descent helium bottle	Shorten 51-inch sumps Adapt gauging system	Remove Block II tanks Add two LEM descent tanks Adapt tank supports Reroute plumbing Adapt internals Remove Block II helium bottles Add two LEM descent bottles
to 2 7° to 1 9°	0 1 to 1 0° 0 2 to 1 8°	0 2 to 2 0° 0 1 to 1 9°	0 1 to 1 3° 0 1 to 1 9°	Not Determined	Not Determined
	Not known	1729 with LEM He 102 with Apollo He	1729 with LEM He 102 with Apollo He	204	Not known
					→
day environment	45-day environment + LEM tank requal + internal requal	45-day environment + short tank requal + internal requal	45-day environment + short tank requal + internal requal	45-day environment + short tank requal + internal requal	45-day environment + LEM tank requal + internal requal
estimated	4,278,000	3,024,000	3,220,000	Not estimated	Not estimated
estimated	-220,000	-244,000	- 244,000	Not estimated	Not estimated

~~CONFIDENTIAL~~

The first case is the SPS Block II initial baseline configuration. It consists of 45-inch storage tanks in Bays III and IV of the SM, 51-inch sump tanks in Bays II and V, and two Block II 40.7-inch helium vessels.

Cases 2, 5, and 6 are all discrete examples of the same basic configuration, which utilizes the 51-inch sump tanks only. In each case, the tanks are located in SM Bays II and V, with Bays III and VI empty. Case 2 uses full length Block II sump tanks. Case 5 uses Block II sump tanks with the cylindrical sections shortened to 35.8 inches. Case 6 has cylindrical sections shortened to 9 inches, the minimum possible using Apollo hardware. The initial trade study indicates a helium requirement for Case 2 of one Block I helium vessel qualified to 4000 psia. Cases 5 and 6 each require one LEM descent-stage helium vessel. Each of these three cases require that the transfer lines be removed and the helium supply lines be rerouted to the sump tanks. In addition, the propellant gaging system will have to be adapted, and the helium standpipe shortened for Cases 5 and 6. For Case 6 the retention can be shortened. Finally, for Cases 5 and 6, provisions must be made in the SM for installation of LEM descent helium vessels in place of Apollo vessels.

Cases 3A and 3B are essentially the same in that they both consist of full length Apollo Block II 45-inch storage tanks used as sump tanks with the Apollo 51-inch sump tanks removed. Both configurations require that the retention screens and reservoir be modified and propellant gaging system be adapted to the use of a 45-inch tank for a sump. They both utilize one Block II helium vessel. The difference is that Case 3A locates the propellant tanks in Bays III and VI while Case 3B locates the propellant tanks in Bays II and V (the Block II location for sump tanks). Case 3A requires that the propellant feed line plumbing be completely rerouted. Case 3B requires only that the helium supply lines be rerouted, but new tank supports must be provided in Bays II and V.

One LEM ascent-stage tank for each propellant is provided in Case 4. The LEM ascent tanks are 49.4 inch spheres and are located in SM Bays II and V, with Bays III and VI empty. Two LEM ascent-stage helium vessels are also required. This case requires the most extensive hardware modifications of all configurations considered. The Apollo Block II retention reservoir must be shortened, the retention screens and gaging system must be adapted, and the LEM tank door must be made compatible with the AES propellant and helium line connections. The helium supply line must be rerouted to the propellant tanks, and support provisions must be made in the SM for both the propellant tanks and the helium vessels. In addition to the hardware changes, it is noted that the LEM ascent-stage tanks must be pressurized prior to launch for structural integrity and this violates a basic Apollo SM structural criterion.

Case 7 provides 80 percent of the nominal usable propellant of the Apollo Block II baseline configuration. It is the same as Case 1 in that it uses four tanks, but the 51-inch sump tanks have a cylindrical section length reduced to 53.9 inches. This requires that the gaging system be modified and the sump tank standpipe shortened.

Case 8 considers one unmodified 51-inch LEM descent-stage tank for each propellant. The tanks have a cylindrical section length of 19.29 inches and are located in SM Bays II and V. One LEM descent helium vessel is required for pressurization. The Apollo retention reservoir, standpipe, and gaging system must be adapted to these tanks. The helium plumbing must be rerouted to the propellant tanks, and structural changes made to support the propellant tanks and helium vessels.

Many additional combinations of available tankage hardware were considered, such as using two LEM ascent tanks for each propellant, but these were not included in the trade-off matrix because they would either require much more extensive hardware changes and system redevelopment, or were already covered by a more favorable case.

TANK SIZING METHOD

The usable propellant capacities and the helium requirements for the eight AES SPS tankage configurations are shown in Table 3. Except for Case 1, which is the reference design, the propellant capacities and helium requirements were obtained by design calculations. Apollo Block II data were used where applicable and the results are considered approximate but conservative. The following explains the basic approach used for these calculations.

The unpressurized volume of the propellant tank under consideration was calculated if this information was unavailable. Calculations of residual volumes included: empty tank vapor, retention reservoir inside pull-through, retention reservoir outside pull-through, gaging system tolerances, and GSE loading tolerances. Residual volumes and estimated minimum ullage were subtracted from the total tank volume. The tank volume remaining was considered usable propellant volume. Since the propellant tanks are of equal size for fuel and oxidizer and since the mixture ratio of 1.6:1 does not give exactly an equal volume ratio for Aerozine 50 and nitrogen tetroxide, the fuel volume requirement is considered controlling and there is excess ullage in the oxidizer tanks.

The helium requirement for each configuration was obtained from an iterative end-point analysis. The helium required to displace propellant was calculated considering the volume of the residuals, tank expansion due to pressure and temperature, and helium compressibility. The weight of helium remaining in the helium vessel at final cut-off is an unknown even though the volume and final pressure of the helium vessel is known since the final temperature of helium in the vessel must be determined. The final temperature of the helium in the vessel was obtained from three equations solved simultaneously. The first equation described the initial helium vessel loading conditions. The second equation specified the final cut-off helium vessel conditions. The third equation specified the thermodynamic process of isentropic expansion of the helium in the helium vessel. In addition, heat is transfered from the helium vessel walls to the expanded helium. This was also considered, but since the end temperature depends on the initial helium pressure, an iterative solution was necessary. The results of this calculation give the required helium loading pressure and thus the helium weight initially loaded into the helium vessel.

A sample calculation detailing this procedure for Case 5 is given in Appendix B. For this study, the method given above was considered satisfactory since the trade-off study was of a preliminary design nature and the results are considered conservative. During future studies, a rigorous analysis requiring machine calculations will be accomplished for the configurations chosen for the AES SPS design.

DISCUSSION OF TANK TRADE-OFF FACTORS

Referring to Table 3, the usable propellant for the AES SPS tank configurations considered vary from the Apollo Block II maximum loading of 39,700 down to 4,270 pounds, with discrete increments at 30,400 pounds, 21,040 pounds, 16,550 pounds, 10,000 pounds, 7,500 pounds, and 5,830 pounds. These loadings are obtained by using Apollo or LEM tankage hardware with and without modification. Propellant tankage, helium vessel, and tank internal descriptions are also shown. Tank system weight and volume savings, as given in Table 3, are approximately represented in Figure 2. A brief notation of required hardware changes is also included so that system change complexity is identified.

The c.g. limits for each configuration are included in the matrix but did not enter into the trade-off evaluation since all of the configurations fall within the SPS gimbal capability. The reliability estimates shown are based on the best information presently available, but are approximately due to the lack of Block II reliability data on qualified Block II or LEM hardware. GSE changes are considered minor and do not materially affect the trade-off evaluation. All of the configurations, including the Block II baseline configuration, must be partially requalified for the 45-day AES environment, but LEM hardware and all Apollo hardware that is modified must be requalified for functional certification.

The recurring cost saving is approximately the same for each case estimated, while the non-recurring costs vary considerably. Case 2 costs about $1100 per pound of weight saved while Cases 4, 5, and 6 cost between $2400 and $2800 per pound of weight saved.

DISCUSSION OF GAGING SYSTEM CHANGES

Although not included in the matrix of Table 3, a trade-off study of propellant quantity gaging system changes for the reduced tankage configuration also was conducted. Tankage configurations considered were Case 2, and the shortened propellant tank Cases 5, 6, and 7.

Gaging Changes for Case 2

Case 2 tankage configuration deletes two 45-inch storage tanks and the corresponding gaging sensors. Deletion of the sensors affords a weight reduction of 21.5 pounds. Power consumption will also be reduced by about 0.5 watt for the two sensor removals.

There are three different possible modifications to the control unit. The simplest change would be no modification at all to this unit. To accommodate the two missing sensors, a sensor simulator would be employed. This would be contained within a 3 x 4 x 2 inch envelope and weigh approximately 0.75 pounds. The net weight reduction would be 19.75 pounds. This is the recommended configuration for tankage Case 2 since it requires minimum change.

The second approach would be to delete the primary fuel storage servo and servo amplifier, and primary oxidizer servo and servo amplifier. To provide a proper signal to the display servo amplifiers, a fixed voltage corresponding to an empty fuel and oxidizer tank signal would be substituted for the primary fuel storage servo and primary oxidizer storage servo. Removal of the servos and their associated amplifiers would result in a decrease in weight of approximately 1.5 pounds per servo, a decrease in power consumption of approximately 3.25 watts per servo, and an increase in reliability. The auxiliary system would require a sensor simulator located inside the control unit, utilizing some of the space made available by the removal of the two primary servos. The resulting control unit would be 3 pounds lighter, and consume 6.5 watts less power than its Block II counterpart. Reprogramming would be required on all GSE equipment (BME, STU, ACE).

The third approach to modification of the control unit would afford a maximum weight reduction, but violates minimum change policy. This change would be to delete both primary storage servos, and replace both primary sump servos with microelectronic amplifiers. The microelectronic units would be operational amplifiers which provide analog signals proportional to propellant quantities. For telemetry outputs, these signals are demodulated and supplied to the telemetry lines.

By replacing the sump servos, and deleting the store servos, significant savings would be attained in weight and power consumption. Using the figure of 1.5 pounds per servo, six pounds could be saved, without counting savings possible by redesign of the control unit to remove the extra case and plate material, and internal wiring saved by the previously described change.

Power consumption would be reduced by approximately 3.25 watts per servo (since the electronic amplifiers and demodulators will dissipate less than 0.2 watt total) and this would afford a 13 watt power reduction for the system. Reliability would increase, primarily due to the deletion of the sump servo motors.

The control unit modified as indicated would perform all the functions of the control unit described in the second approach, but would be lighter, more reliable and consume less power. GSE restrictions are the same as the unit in the second approach.

<u>Gaging Changes for Cases 5, 6, and 7</u>

The second type of tankage configuration considered (shortening the fuel and oxidizer sump tank probes) requires the changes resulting from removal of the storage tank, plus a redesign of the sump probes. The probe design is almost entirely mechanical in nature, and consists of the following: (1) physically shorten probes; (2) relocate point sensors; and (3) reprofile probes to accommodate new tank geometry. Shortening the probes affords a weight reduction of 8 pounds for the fuel probe and 7 pounds for the oxidizer probe. This weight reduction is based on 36-inch cylindrical length oxidizer and fuel sump tanks. Vibration testing may be necessary to check the shortened unit design for structural integrity.

Reprofiling is required to compensate for changes in the tank geometry accompanying tank shortening. This consists of feeding the new height-volume relationships into a digital computer to generate the proper profile data for each probe. Then each unit must be fabricated and profiled per the computer results to verify the profile pattern. Testing with reference fluids is required in this phase of the operation.

Besides the probes, and relocation of point sensors, the only other component requiring modification is the control unit. Choice of the changes described in preceding paragraphs must also be made for this configuration, because one fuel and one oxidizer sensor are deleted. In addition, control unit modifications are required to accommodate the probe shortening, probe reprofiling, and the point sensor redistribution.

Sensor shortening and reprofiling will result in capacitance versus mass readings which differ from Block II. To correct this, scale factors must be modified in the primary sump servo amplifiers. Unfortunately, the scale factor change will prevent the control unit from being directly interchangeable with Block II units.

Similarly, point sensor relocation will require corresponding changes in the D/A converter in the auxiliary system. No change is obtained in power consumption from the previous sump tank configuration.

REDUCED TANK CONFIGURATION TRADE-OFFS

The evaluation of the AES SPS propellant tankage and helium requirement was based on the primary considerations of mission propellant requirements, configuration change cost, and weight saved. The AES mission propellant requirements, shown in Figure 1, fortunately resulted in only two classes of loading—full up or less than 30-percent full.

Case 2, the recommended configuration, is applicable to all AES missions not requiring the full-up Block II propellant loading. This configuration provides for 21,000 pounds of usable propellant, or 53 percent of the maximum usable Block II propellant load, saves 586 pounds of burned weight, and most important, requires only minor modification to the Block II baseline. The cost per pound of weight saved for this configuration is the lowest of all configurations considered. This case was selected since it is more than adequate for all anticipated off-loaded mission requirements, provides usable propellant flexibility, and requires the least hardware change of all the cases considered.

Compared to Case 2, Cases 3A and 3B have a slightly larger weight saving; they, too, cover the anticipated off-loaded mission requirements, but with less flexibility and require more extensive hardware modification. These cases were eliminated in favor of Case 2. The two LEM ascent tanks, Case 4, do not meet the Apollo structural criteria in that they must be pre-launch pressurized, cover only part of the off-loaded mission requirements, and would require extensive hardware modification and related development.

Cases 5 and 6 are discrete examples of Case 2 and were eliminated in favor of Case 2 because of the greater amount of hardware change that is required. These cases serve to emphasize the maximum weight and volume savings available if the tanks are sized only for the minimum propellant required.

Case 7 was eliminated in favor of Case 1, the Block II baseline, since the penalties of cost and development due to hardware modification greatly outweight the weight and volume savings gained, and the mission requirements for the possible full-up missions would not be met. The two LEM descent stage tanks appear to be readily adaptable to the AES system, but the propellant capacity is covered by Case 5 which has less hardware modification.

HELIUM VESSEL TRADE-OFFS

Subsequent to the selection of Case 2 in the initial trade study, the use of one Block I helium vessel was rejected because of logistic considerations. This relates to the option to change configurations from AES to Block II at the Kennedy Space Center. If two types of helium vessels (Block I and II) are used, this may unnecessarily complicate the spares problem. The AES baseline reduced tankage configuration (referred to elsewhere in this report as AES-SPS Configuration B) was then changed to use two Block II helium vessels. The following is a discussion of all of the helium vessel options and leads to the recommendation that the baseline Configuration B use one Block II helium vessel.

<u>One Block II Vessel</u>

Configuration B propellant tank pressurized volume (usable plus ullage) is about 53 percent of the standard Block II configuration volume. Since one Block II helium vessel provides only 50 percent of the helium of a standard configuration, one Block II vessel is not adequate without a change in ground rules. The following changes were considered:

1. An increase in residual propellant by 5 percent would reduce the pressurized volume to 50 percent but incurs a penalty of 2000 pounds of burned weight.

2. Reduction of usable propellant from 53 percent to 50 percent and ullage pressurization (to 175 psia) on the stack during helium servicing permits the use of one Block II vessel. The propellant system would then remain pressurized at rated pressure down to the engine valve for the last 20 hours of the countdown or else the gas

servicing operation must be delayed in the count. Either procedure requires a change in the ground operations procedures. The first alternative adds hazard and the second evidently would interfere with other ground operations and would still result in hazard exposure for the remaining period of the count.

3. A decay in propellant tank pressure of 10 percent over the last 30 seconds of the SPS burn profile would permit the use of one Block II vessel (a 14 percent pressure decay was studied as a proposed Block II weight saving). This would result in approximately 5 percent decay in chamber pressure and thrust and about 0.5 second loss in specific impulse over the decay period. The SPS would still be operating within its design limits and the only penalty is the brief specific impulse loss which is equivalent to a burned weight penalty of less than 10 pounds. For off-loaded flights requiring less than about 17,000 pounds of propellant, no decay results since normal ground pressurization of the ullage space provides the necessary additional helium. Those flights using between 17,000 and 21,000 pounds of propellant will have chamber pressure decay in proportion to the loading.

One Block I Vessel

A Block I vessel carries roughly 10 percent more usable helium than a Block II vessel and will therefore meet the Configuration B requirement. Logistic considerations of Case H-2 include the fact that a Configuration B SM must be scheduled in advance to permit the vessel to be procured. Case H-3 uses a Block I vessel in both A and B configurations and the choice between the two may be delayed. The manufacturer, Airite, has stated that minimal charges would be involved in changing back to Block I during Block II production. The vessels are interchangeable except for maximum fill pressure which is 4000 psi on Block I and 3585 on Block II. No GSE change is required.

Two Block II Vessels

Two Block II vessels may be used and 40 pounds of helium may be off-loaded.

Helium Vessel Recommendation

Table 4 shows the possibilities considered and the burned weight penalty attributed to each. Since the full-up flights are more weight sensitive and there are more of these flights, the use of a Block I vessel in Configuration A (Case H-3) is questionable. Case H-4 incurs a severe weight penalty (278 pounds) on B flights. Case H-1 using P_c decay is recommended.

CONCLUSIONS

The final selection, designated as SPS baseline Configuration B, was chosen primarily for its simplicity. Other configurations of less capacity would cover the presently anticipated mission propellant requirements and with greater weight savings. However, the increased complexity of changes

Table 4. Configuration B Helium Vessel Weights

Case Title	One Block II Vessel	One Block I Vessel	One Block I Vessel	Two Block II Vessels
Case Number	H-1	H-2	H-3	H-4
Change in Ground Rules	Pc Decay	Logistic	Block II	None
"A" Configuration Vessels *	II, II	II, II	II, I	II, II
Helium Weight	88	88	88	88
Vessel Weight	670	670	727	670
Total	758	758	815	758
Penalty on "A" Flights	-0-	-0-	57	-0-
"B" Configuration Vessels	II	I	I	II, II
Helium Weight	44	48	48	48
Vessel Weight	335	392	392	620
Total	379	440	440	718
Penalty on "B" Flights	(61) **	-0-	-0-	278

* Configuration "A" is the standard 4-tank AES SPS.

** Case H-1 is 61 pounds lighter than Case H-2 minus I_{sp} penalty equal to less than 10 pounds.

involved required greater cost per pound of weight saved. The selected configuration is applicable to those AES missions requiring less than 21,000 pounds of SPS propellant. It uses only the two sump tanks, the two propellant storage tanks being omitted. Both of the Block II helium vessels are retained. However, the definition of AES baseline Configuration B could be changed to use only one Block II helium vessel. This will result in a 10 percent decay in chamber pressure and thrust over the last 30 seconds of flights requiring 21,000 pounds of propellant.

To summarize, the B configuration has a capacity for 53 percent of the maximum usable propellant load, saves 586 pounds (the saving is 864 pounds if the change to one helium vessel is made), and has the lowest cost per pound of burned weight saved.

USE OF LEM PROPULSION

Some consideration was given to the use of the LEM descent engine on a LEM laboratory for performing certain maneuvers with the CSM (for example, lunar injection and injection into synchronous orbit). Several probable problems with SPS propellant retention devices are possible using this scheme since the SPS is not designed for continuous negative acceleration in excess of that due to SM RCS.

The SPS sump tanks contain a propellant retention reservoir with two additional retention screens (see Figure 4). The anticipated capability of the retention screens in the negative X direction is no more than 0.02 g. The retention reservoir will hold enough propellant for about 6 seconds of SPS firing and represents a start slug.

Use of the LEM engine will impart a negative acceleration of approximately 0.1 g to 0.2 g. This can dislodge the propellants behind the screen and replace it with trapped helium. Subsequent SPS firings will ingest the helium (probably a large quantity), possibly triggering instability and restart problems. If the SPS engine is shut down by the CSM, it may never be restarted due to the trapped helium. If the SPS firing has been insufficient to resettle the propellants, the RCS does not impart enough acceleration to permit the settling of propellants through the screens.

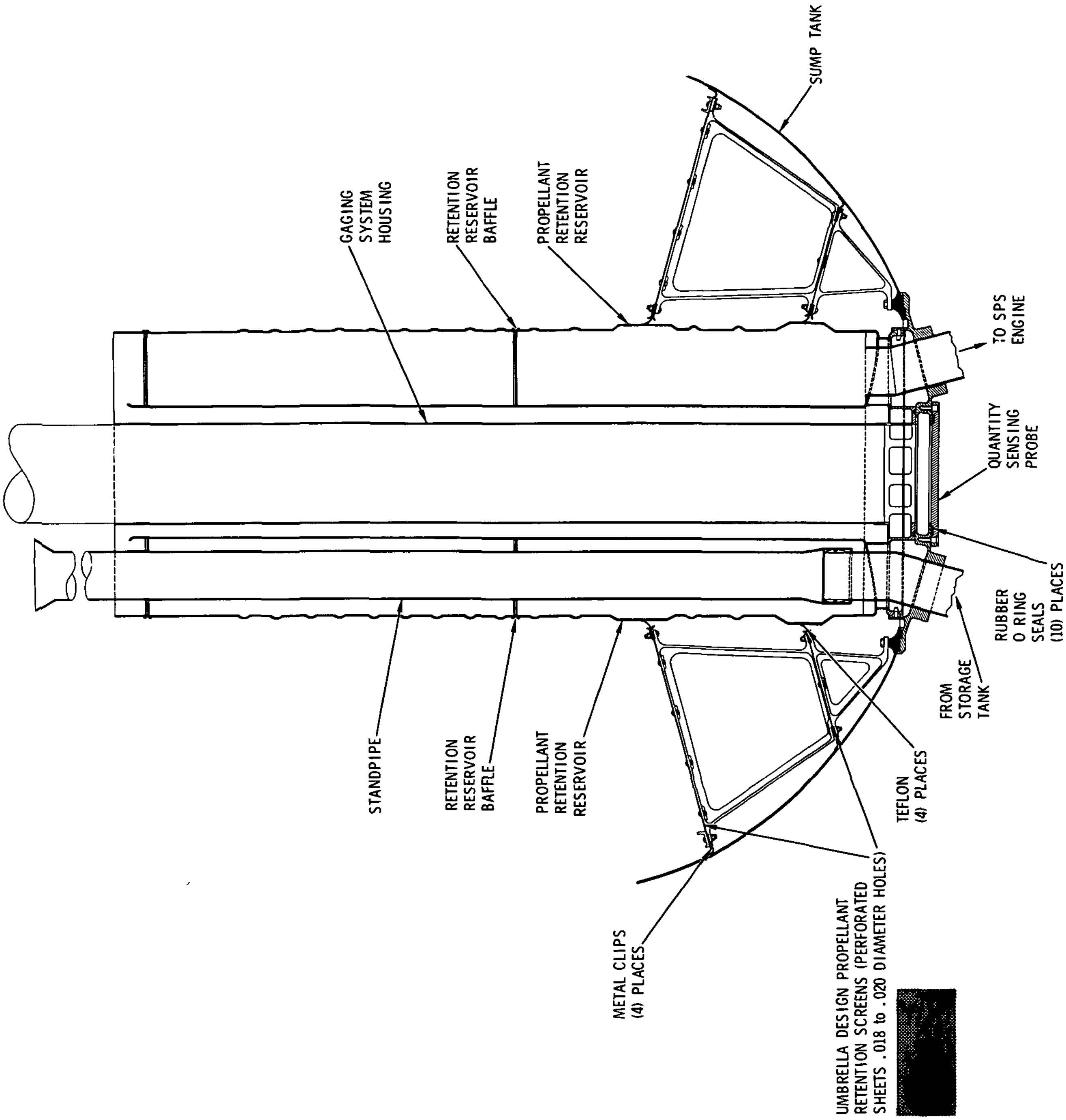

Figure 4. SPS Propellant Retention Reservoir

IV. LIFE EXTENSION ANALYSES

This section presents the analyses and the results of life extension studies. There are two topics of interest: (1) fluid leakage, and (2) materials life extension.

Analyses were made of the internal and external leakage. The critical areas of internal leakage are backflow leakage to the regulator, main propellant valve shaft seals, and helium leakage to ullage. The critical areas of external leakage are the nitrogen system and the propellant and helium system.

The materials life extension analyses were concerned with the effects of the space environment on metallic and non-metallic materials. For a definition of the radiation and meteoroid affects on the SPS, the reader is referred to the Systems Analysis Summary, SID 65-1534.

FLUID LEAKAGE

Extension of the Block II SPS to 45-day coast capability requires reassessment of fluid leakage from the system. The fluids to be considered are: (1) nitrogen used for engine valve actuation, (2) helium used for propellant pressurization, and (3) propellants. The types of leaks to be considered include: (1) internal leaks causing pressures to exceed design limits or causing materials compatibility and/or explosion problems, and (2) external leaks depleting gas or propellant. In this section, fluid leakage is considered from three viewpoints: (1) the problem of predicting in-flight leak rates (assuming no failures) over long periods from preflight acceptance tests, (2) a review of Block II acceptance test leak rates for adequacy on an extended mission, and (3) the measurement of fluid leakage continuously throughout a mission to permit timely abort action.

The Block II design philosophy used to establish standards for acceptance test leak rates has assumed that the increase in leak rate during a mission would not be excessive unless a failure with an assignable cause occurred. Moreover, acceptance test methods for liquids (propellants) utilize a gas substitute to improve accuracy in leak rate measurement. In a following section, this same philosophy is used for a reassessment of Block II leak rates for AES 45-day missions. However, it is apparent that for longer duration missions the prediction of in-flight leak rates on the basis of a short term acceptance test leaves room for considerable doubt. Progressive degradation in the performance of seals is a well known and common occurrence, usually compensated for by maintenance. Such deterioration is not generally covered as a failure by reliability assessment. In any event, an improvement in predicting long term performance of individual seals (which can vary grossly from that of qualification test articles) on the basis of pre-launch acceptance tests is required for the AES SPS. The problem primarily involves test technique and will be emphasized in future studies.

The study results indicate two potential problem areas that require additional analysis. These are: (1) propellant backflow leakage across the helium check valve and into the helium regulator region, and (2) propellant leakage from the shaft seals into the common gear housing of the propellant valve actuation system.

INTERNAL LEAKAGE

The areas of internal leakage which were evaluated are backflow leakage to regulators, main propellant valve shaft seals, and helium leakage to ullage.

Backflow Leakage to Regulator

A potential problem exists for the AES mission since a corrosive or explosive mixture of helium and propellants may be attained above the check valves and in region of the regulators. This plumbing is common to both fuel and oxidizer backflow above the check valves. There are three possible ways that propellant, liquid or vapor, might reach the check valve region:

1. When the helium solenoid valve is closed, helium leaking externally from the region above the check valve will cause a differential pressure that, with backflow helium leakage through the check valve, will allow a slug of propellant to move from the propellant tank towards the check valves.

2. Even without an external helium leak, the propellant vapor will diffuse into the helium and eventually this diffusion will reach the check valve region.

3. In zero gravity, films of propellant will climb the helium pressurization lines and may reach the check valves.

Only the first method is considered a problem for the AES mission. The time required for diffusion is of the order of 1500 hours before propellant vapor would reach the check valve region. The phenomenon of films of propellant climbing the helium lines in zero-g is line length dependent and it is doubtful that propellant would get to the check valves.

The AES mission is approximately 1000 hours and the specification external leak rate of the four check valves is 2.0×10^{-4} scc/sec of helium. Considering the volume of the helium system from the check valves to the top of the standpipe, a slug of propellant could conceivably reach the check valves due to the driving pressure differential after 500 hours of coast. Additional external helium leakage above the check valves will cause a pressure differential that, with backflow leakage across the check valves, will allow fuel and oxidizer to reach and mix in the common regulator region. It may be concluded that a potential problem exists in this area for the 45-day AES mission. This problem will be further analyzed to determine the amount of propellant backflowing through the check valve and the explosive or corrosive effect it may have in the regulator region.

Main Propellant Valve Shaft Seals

A potential explosive or corrosive problem is a possibility with the Block II Apollo configuration of the SPS engine ball valve shaft seal drainage system. The ball valve shaft seals have a specification leak rate maximum of 40 scc/hr of GN_2. This rate of GN_2 is equivalent to 2 cc/hr of propellant. Under these conditions, sufficient propellant leakage can result to permit the propellant to freeze in the unheated drain line as a result of free expansion to the vacuum environment. If enough propellant freezes in the drain line, blockage will occur. Propellant trapped in the blocked drain line can then either leak into the common gear compartment and cause a possible corrosive or explosive condition or build up pressure that will impair the integrity of the valve shaft seal.

This problem is common with the Apollo mission and Block II development will be closely monitored by AES. Once the problem is defined, design provisions such as line heaters will be considered as may be necessary.

Helium Leakage to Ullage

Helium leakage into the propellant tank ullage has been considered as follows:

Helium Solenoid Valve

Internal Leakage - 20 scc/hr

Assume - 45 days of leakage continually

Assume - minimum ullage volume for 45 days

Maximum pressure rise in propellant tanks - 0.9 psi

Helium Regulator

Internal Leakage - 10 scc/min

Volume between solenoid and regulator - 2 in^3

Time to relieve 4000 psi in above volume to ullage pressure at leak rate of 10 scc/min - 15 hours

Ullage pressure rise during above time (15 hours) when ullage is minimum - 0.4 psi

From the above it is shown that the result of the internal leak rate of the helium solenoid valve is an ullage pressure rise of less than 1 psi. It should be noted that no anticipated AES mission is run for 45 days at minimum ullage.

The internal helium regulator leakage would present a potential problem only if the solenoid valve was periodically cycled while maintaining a minimum ullage volume. The maximum pressure rise with minimum ullage, assuming a failed solenoid valve continually open, is approximately 0.6 psi per day.

EXTERNAL LEAKAGE

The areas of external leakage which were evaluated are the nitrogen system, the propellant system, and the helium system.

Nitrogen System

A review of the pneumatic valve actuation system leakage indicates that this system will be adequate for the 45-day AES missions. The system has been sized for 50 SPS restarts requiring approximately 0.5 lbm N_2 minimum. The Block II SPS engine Specification MC 901-0009 requires 36 restarts and the N_2 loaded is approximately 0.8 pounds at 2500±50 psig at 70°F.

Zero external leakage is required of the components. The maximum internal nitrogen leakage is 10 cc/hr (STP) on the solenoid valves, regulator, and across the actuator pistons. During coast, the two-way pilot solenoid valve is the controlling leak area. The leak will pass through the regulator and finally leak out through the relief port of the main three-way solenoid valve.

The loss of N_2 due to leakage from each system during a 45-day coast period is approximately 0.027 pounds of N_2. This is equivalent to about three percent of the required N_2 load.

Propellant System

Leakage rates through the SPS engine valve balls and shaft seals are as specified in the SPS engine specification. Another source of propellant leakage is through the propellant tank relief valve. However, since the burst disc must be failed before leakage can occur, its analysis is part of reliability assessment. A third source is through the tank door seal. The leakage rates are specified as follows:

Engine valve ball leakage = 120 scc/hr (as nitrogen)

Engine valve shaft leakage = 40 scc/hr (as nitrogen)

Propellant tank leakage = 820×10^{-6} scc/sec (as nitrogen)

This leakage was converted to an equivalent sonic orifice size using an upstream pressure of 240 psia. By assuming a discharge coefficient of 1.0 and an upstream pressure of 240 psi, the 45-day leakage of equivalent propellant was calculated to be less than 11 pounds.

Helium System

The specification leak rate considered for the calculation of the external loss of helium assumed that all sources leaked simultaneously for 45 days. The leak rates used are presented in Table 5. As can be seen from the table, less than 0.1 percent of the helium would be lost from the system on an AES mission. An increase by a factor of three to reflect three sigma uncertainty would still not require an increase in helium loading.

IN-FLIGHT LEAK DETECTION

Fluid leakage from the SPS during a 45-day mission is a problem that must be considered from the standpoint of both mission success and crew safety. For mission success, excess leakage must be prevented. For crew safety, excess leakage must be discovered at the earliest possible time so that the leak rate may be determined and abort action taken when necessary. Since the Block II outage control gaging system is inoperable during periods of coast (that is, during non-thrust or zero gravity periods), crew safety improvement requires a separate leak detection system for SPS fluids.

Although the zero gravity (coast) leak detection requirement has been established, the system for performing this function has not been determined. Many methods have been proposed, but it is not known at this time if any of the methods possess sufficient accuracy. For useful leak detection, the required accuracy is related to the quantity of contingency (reserve) propellant that may be aboard. If the gage error exceeds the amount of reserve propellant (the excess over the amount required for abort) the gage has no value. The required accuracy varies with mission flight plans but, in general, is approximately 2 percent of the total propellant loaded and 5 percent of the total helium loaded. This desired accuracy does not appear to be outside the realm of feasibility, but gaging capability will not be determined until the completion of the leak detection system feasibility studies.

Some of the characteristics of the SPS propellant system that must be considered to finally resolve the problem of leak detection accuracy are: thermal gradients, mass distribution gradients, random convection patterns for helium inflow, random liquid orientation and location during zero-g transfer line filling and draining for full and partially filled loadings, tank pressure fluctuations during firing due to regulator action and transfer line monometer effects, and helium inlet temperatures varying during blow down to 20°F less than propellant temperature.

It is planned that the leak detection system will either supplant or utilize, as is, the existing outage control gaging system. Requirements applicable to the study are outlined below.

Leak detection gaging is defined as gaging during non-flow periods under conditions of zero to 5.0 g acceleration for durations up to 45 days. Changes in the fluid quantity (from that at the start of the non-flow period) contained in each of the two propellant systems (oxidizer and fuel) are to be

Table 5. SPS Helium Components - External Leakage

SPECIFICATION	NAME	BLOCK II SPEC. LEAKAGE(scc/sec)	TOTAL HE LOSS 45 DAYS (lbsm)
MC 144-0023	T.P. Disconnect (Fuel side)(17 parts)	5×10^{-6}	123×10^{-6}
	Joints, Transducers and Ports (71)	5×10^{-6}	515×10^{-6}
MC 273-0009	Coupling - Helium Fill	5×10^{-6}	7×10^{-6}
MC 362-0019	Heat Exchanger, Ox	5×10^{-6}	7×10^{-6}
MC 362-0019	Heat Exchanger, Fuel	5×10^{-6}	7×10^{-6}
MC 284-0018	Helium Solenoid Valve #1	5×10^{-6}	7×10^{-6}
MC 284-0018	Helium Solenoid Valve #2	5×10^{-6}	7×10^{-6}
MC 284-0324	Helium Pressure Regulator #1	$16,700 \times 10^{-6}$	$24,200 \times 10^{-6}$
MC 284-0324	Helium Pressure Regulator #2	$16,700 \times 10^{-6}$	$24,200 \times 10^{-6}$
MC 284-0330	Relief Valve (Fuel side)	5×10^{-6}	7×10^{-6}
MC 284-0330	Relief Valve (Ox side)	5×10^{-6}	7×10^{-6}
MC 284-0327	Check Valve Quad (Fuel side)	5×10^{-6}	7×10^{-6}
MC 284-0327	Check Valve Quad (Ox side)	5×10^{-6}	7×10^{-6}
V37-343102	Tank Fuel Storage	820×10^{-6}	1190×10^{-6}
V37-343102	Tank Ox Storage	820×10^{-6}	1190×10^{-6}
V37-343102	Tank Fuel Sump	820×10^{-6}	1190×10^{-6}
V37-343102	Tank Ox Sump	820×10^{-6}	1190×10^{-6}
	TOTAL	$37,165 \times 10^{-6}$ scc	.053861 lbsm

measured and displayed. It is preferable but not required that the system discriminate between propellant and helium quantity changes. It is also preferable that the system function continuously, but discrete measurements on command are adequate provided the operation is not limited to a fixed number of measurements. It is preferable but not mandatory that the measurement indicate total fluid quantities instead of merely changes, since the information could be used as a check against outage control gaging measurement obtained at the termination of the previous flow period.

Outage control gaging is defined as gaging during flow of helium into and propellant out of the propellant system under conditions of 0.1 to 1.0 g acceleration for durations of 1.0 to 630.0 seconds. Oxidizer and fuel remaining are measured and displayed separately and continuously during flow. The excess or deficiency of oxidizer remaining with respect to a 1.6 to 1 oxidizer-to-fuel mixture ratio is computed from these measurements and displayed continuously.

Mission requirements applying to the complete fluid gaging system include 45-day space flight operations which may be all coast (non-flow) period, or firings (flow) may be interspersed at any time in the 45-day period. Firings of less than 5 seconds will not require outage control gaging. Leak detection gaging is required from completion of servicing on the launch stack through boost and coast until the first SPS firing and for every coast period thereafter.

MATERIALS LIFE EXTENSION

To extend the Block II SPS capability from 14 to 45 days requires review of SPS materials environmental exposure resistance. With one exception, the metallic materials used are considered to have inherent capability for 45 days since the basis for their selection was not significantly time limited. The exception is the use of titanium exposed to nitrogen tetroxide, a problem area currently receiving intensive study by the Apollo program. The materials to be reviewed by AES were consequently reduced to the non-metallics. A detailed review of the life capability of most of the non-metallics is contained in this section. Since engine subcontractor participation was not included in this preliminary definition phase, only a partial list of representative engine non-metallics were reviewed.

The study was limited to non-metallic materials exposed to Aerozine 50 and nitrogen tetroxide propellants over the SPS design temperature range plus exposure to radiation, as described by the report SID 65-1534, System Analysis Summary, and exposure to space vacuum.

The results of this study indicate, with a few exceptions, that the non-metallics of the Block II SPS are adequate for AES missions and environments. The exceptions are in the use of butyl rubber for the nitrogen tetroxide tank door seals (main door and sensor flange) and for the low temperature Aerozine 50 fuel/helium check valve. Since the material for the oxidizer/helium check valve has not been finally selected for Apollo, it has not yet been evaluated for AES. Radiation dosage expected for AES missions should have no appreciable affect since system materials are provided with the inherent shielding of component housings and spacecraft structure and equipment.

Propellant exposure demonstrations used for Apollo design verification and/or qualification tests have been limited to durations of 15 to 52 days, the period varying with the component. For this reason, supplementary tests of longer duration are recommended for AES component certification (see report SID 65-1148, General Test Plan).

The approach used for the study (see Figure 5) first required a listing of all non-metallic materials and compositions from Block II design drawings, specifications, and related data. Material compatibility with the propellants and radiation resistance data was then obtained from various test reports and published reports from industry. This data was then compared with the extended mission requirement of exposure for 45 days and with radiation dosage occurring during the mission. Radiation dosage was determined using both a 1.0 percent and a 0.1 percent probability of occurrence of solar proton events and by calculating the effect of inherent shielding of the particular component. Other environment criteria, such as temperature limits, were considered to remain within the Block II specification requirements.

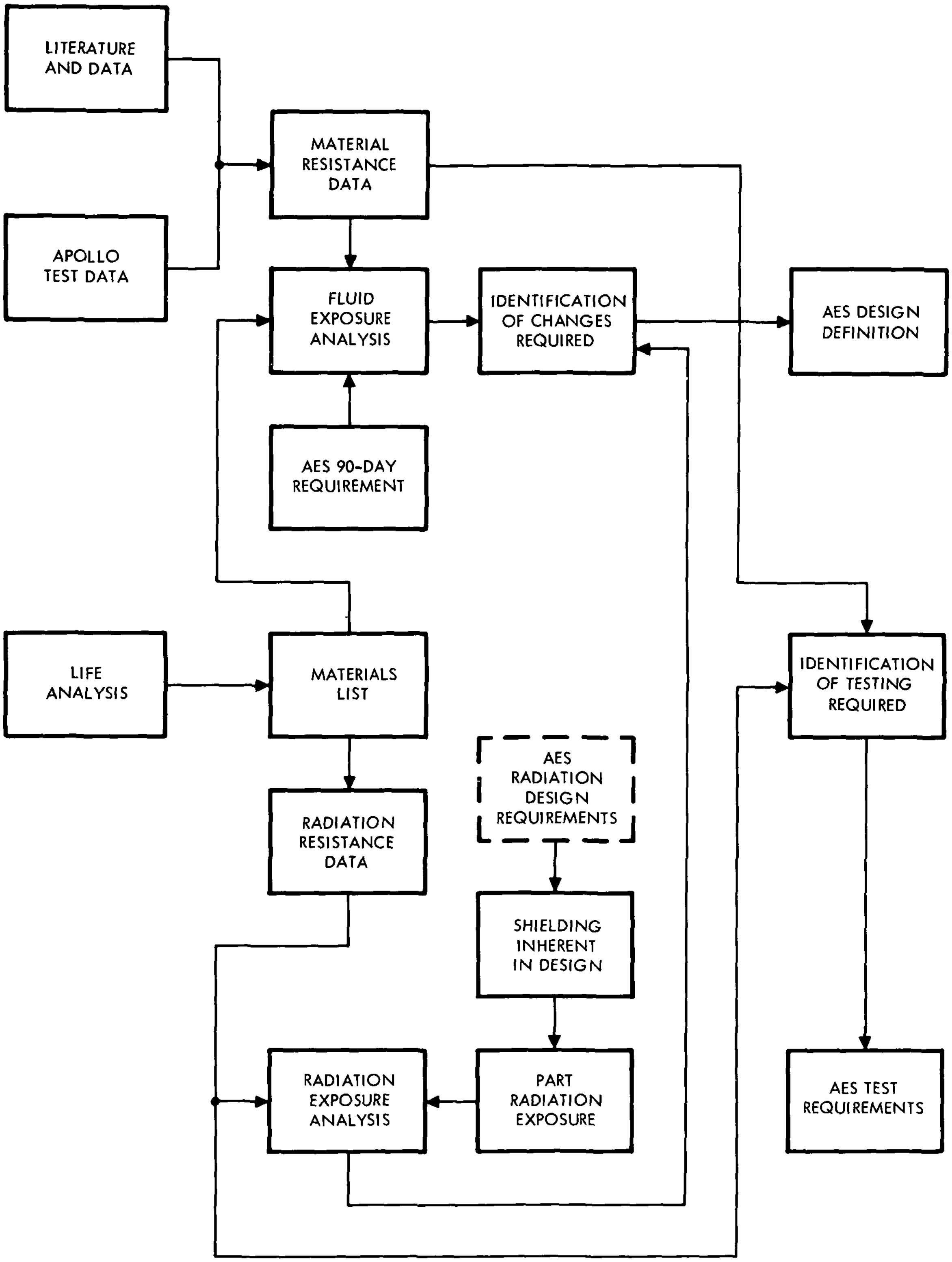

Figure 5. Task Flow Diagram, Nonmetallic Materials Life Study

PROPELLANT COMPATIBILITY

Table 6 is a list of components in the SPS feed system shown schematically in Figure 6. Table 6 also contains a partial list of SPS engine components. The engine list is not complete since engine subcontractor participation was deferred to the FDP. Tables 7 through 9 list the non-metallic materials used in these components, the material specification, where the material is used, the Apollo demonstration test reference, the materials' acceptability for AES, and the source reference from Table 15 used in evaluating the materials' acceptability.

RADIATION RESISTANCE

Tables 10 through 12 list the non-metallic materials exposed to radiation effects and compares anticipated part dosage with the acceptable dosage of radiation for the specific material. The SM preliminary zone dosages and part dosages have been established by using the AES mission radiation model. The model is based on solar proton events which have both a 1.0-percent and a 0.1-percent probability of occurrence in 45 days. The analysis takes into consideration the natural shielding of vehicle equipment and structure and component housing. Additional details on radiation levels used is in AES report SID 65-1534 in the section covering Radiation and Meteoroid Analysis.

The sources of Table 15 were used as a basis for material acceptable dosage levels and are representative of published data on materials radiation damage. Where the material acceptable dosage levels far exceed the expected zone dosage, the shielding and part dosages were not calculated. A general view of the effects of radiation on representative materials is presented in Table 13 obtained from source 13 and Table 14 obtained from source 10.

SPACE VACUUM EFFECTS

The pressure in external space is approximately 10^{-8} torr in earth orbit and 10^{-13} torr in lunar space. However, the pressures at the locations in the SPS of the materials of interest will be much higher, perhaps 10^{-3} torr, due to outgassing of the materials, and vaporization and sublimation of working fluids and lubricants. Moreover, these materials are all confined either by a component housing with minimal venting cross section or by a small diameter vent tube several feet long. In every case, the conductance is very low, and if venting were to reduce the pressure below 10^{-6} torr the mean free path of the remaining gases is huge compared to the escape path dimensions. For example, nitrogen at 20°C and 10^{-6} torr has a mean free path in excess of 50 meters. It is clear then, that ultra high vacuum effects are not applicable to materials interior to the SPS. Moderate vacuum levels (above 10^{-5} torr) will be encountered, but these are levels at which industrial vacuum materials data applies.

SPS components and non-metallic materials exposed to vacuum are listed in Table 16. These are identified as either static seals (exposed to moderate vacuum at the outer edge only) or moving seals such as valve

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Table 6. SPS Subsystem Components

QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPECIFICATION
			FUEL DISTRIBUTION	
1	ME 144-0023-0051	Lear-Siegler	Coupling, TP11	MC 144-0023
2	ME 144-0023-0031	Lear-Siegler	Coupling, Residual Drain	MC 144-0023
1	ME 273-0071-0001	Aeroquip Corporation	Flex Joint, Fuel Line	MC 273-0071 Ø
1	ME 362-0019-0011	United Aircraft Prod.	Heat Exchanger, He Low Pressure	MC 362-0019 Ø
1	ME 273-0020-0001	J. C. Carter	Coupling, Fill and Drain	MC 273-0012
1	V37-342102	NAA	Tank, Fuel Sump	
1	V37-343102	NAA	Tank Fuel Storage	
1	ME 273-0012-0001	J. C. Carter	Coupling, Fuel Vent	MC 273-0012
			OXIDIZER DISTRIBUTION	
1	ME 144-0023-0051	Lear-Siegler	Coupling, TP12	MC 144-0023
2	ME 144-0023-0011	Lear-Siegler	Coupling, Residual Drain	MC 144-0023
1	ME 273-0040-0001	Aeroquip Corporation	Flex Joint, Oxidizer Line	MC 273-0040
1	ME 362-0019-0001	United Aircraft Prod.	Heat Exchanger, He Low Pressure	MC 362-0019 Ø
1	ME 273-0022-0001	J. C. Carter	Coupling, Oxidizer Vent	MC 273-0012
1	V37-342102	NAA	Tank, Oxidizer Sump	
1	V37-343102	NAA	Tank, Oxidizer Storage	
1	ME 273-0018-0001	J. C. Carter	Coupling, Ox Fill and Drain	MC 273-0012
			PRESSURIZATION SYSTEM	
1	ME 284-0324-0002	B. H. Hadley	Regulator, Press. He (2-stage)	MC 284-0324 Ø
1	ME 284-0324-0012	B. H. Hadley	Regulator, Press. He (2-stage)	MC 284-0324 Ø
2	ME 144-0023-0011	Lear-Siegler	Coupling, TP9 and TP10	MC 144-0023
2	ME 144-0023-0031	Lear-Siegler	Coupling, TP7 and TP8	MC 144-0023
1	ME 144-0023-0041	Lear-Siegler	Coupling, TP6	MC 144-0023
1	ME 284-0330-0001	Calmec	Valve, Relief Oxidizer	MC 284-0330 Ø
1	ME 284-0330-0011	Calmec	Valve, Relief Fuel	MC 284-0330 Ø

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Table 6. SPS Subsystem Components (Continued)

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QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPECIFICATION
		PRESSURIZATION SYSTEM (Continued)		
1	ME 284-0327-0002	Accessory Products Co.	Valve, Quad Check, Fuel	MC 284-0327 Ø
1	ME 284-0327-0012	Accessory Products Co.	Valve, Quad Check, Oxidizer	MC 284-0327 Ø
4	ME 144-0023-0071	Lear-Siegler	Coupling, TP2, TP3, TP4, & TP5	MC 144-0023
1	ME 273-0009-0007	On-Mark	Coupling, Fill and Drain	MC 273-0009
2	ME 284-0018-0005	Accessory Products Co.	Valve, Solenoid, Shut-off	MC 284-0018
2	V37-347102	NAA	Vessel, Press. Helium	
		SPS ENGINE ASSEMBLY (Partial List)		
2	*	Aerojet-General	Coupling, Test Relief Valve	
2	*	Aerojet-General	Coupling, Regulator Outlet	
2	*	Aerojet-General	Coupling, Fill Vent	
1	ME 901-0192-0001	Aerojet-General	Nozzle Extension, Engine	MC 901-0009
1	ME 273-0011-0001	J. C. Carter	Coupling, Vent and Drain	MC 273-0011
1	ME 273-0024-0001	J. C. Carter	Coupling, Fill Vent	MC 273-0011
1	ME 321-0004-0001	Aerojet-General	Rocket Engine - SP	MC 901-0009

Ø New Block II Specifications

* Furnished with engine.

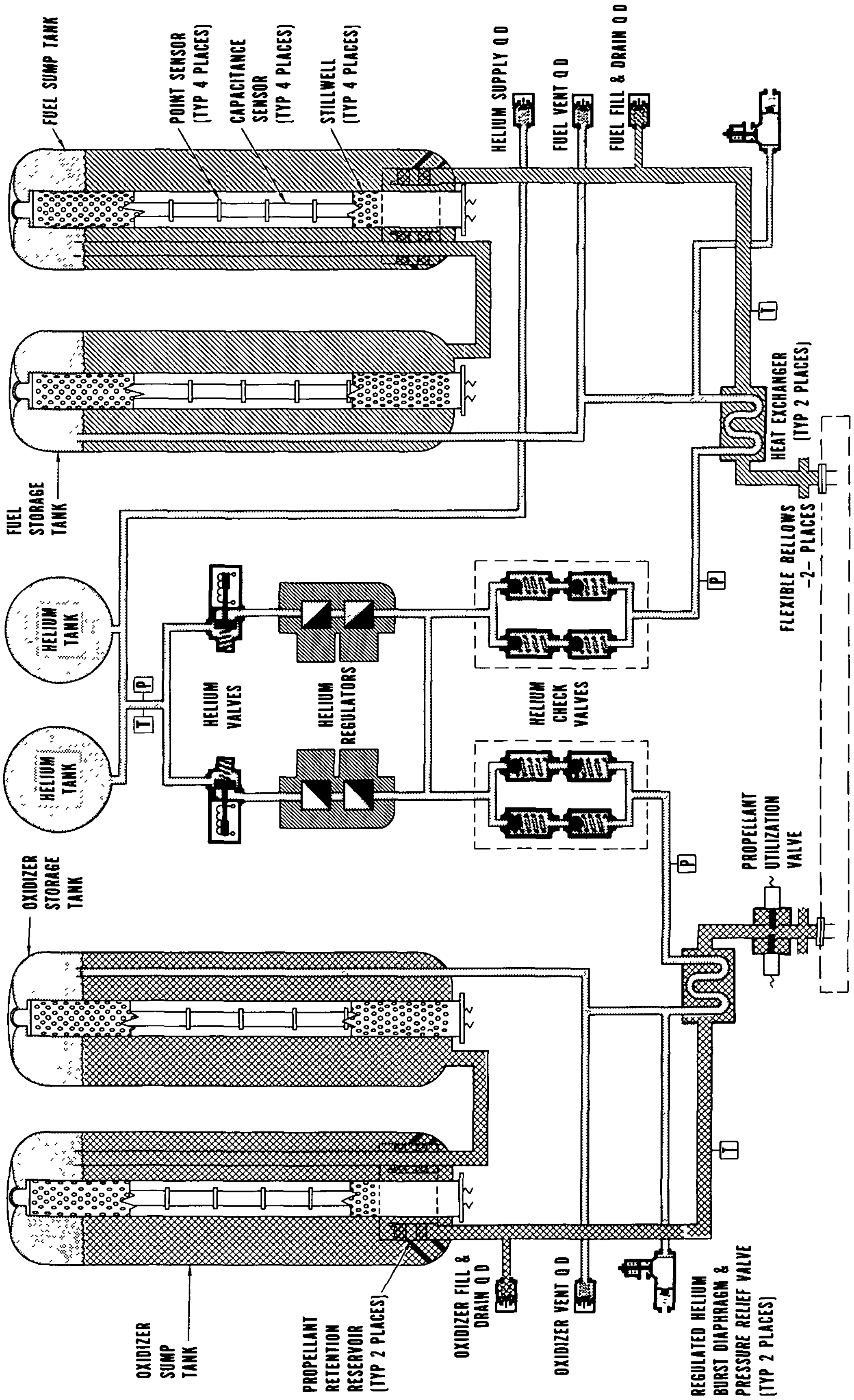


Figure 6. SPS Propellant Feed

Table 7. Propellant Compatibility - Engine Nonmetallics

Material	Specification	Where Used		Apollo Test Demonstration	Design Acceptable for AES		
		System/Part	Detail		Yes	No	Reference Basis
RTV-60		Bipropellant valve		*	X		
Teflon, TFE		Bipropellant valve			X		1, 2
Teflon, FEP		Bipropellant valve	Oxidizer ball seal		X		1, 2
Butyl Rubber		Bipropellant valve	Fuel ball seal		X		1, 2

*Per MIL-E-5151, para. 4.3.3.1.1 (total of 28 days' exposure); see Spec. MC 901-0009, Rocket Engine, SPS

Table 8. Propellant Compatibility Feed System—Nonmetallic

Material	Specification	Where Used		Apollo Test Demonstration	Design Acceptable for AES		
		System/Part	Detail		Yes	No	Reference Basis
Kynar		ME 144-0023 Coupling	Fuel and oxidizer	1	X		1, 2, 3, 4
		ME 273-0011 Coupling	Fuel and oxidizer	1	X		1, 2, 5, 7
		ME 273-0012 Coupling	Fuel and oxidizer	1	X		1, 2, 6, 8
Teflon	AMS 3651	ME 273-0011 Coupling	Fuel and oxidizer	1	X		Same as above
		ME 284-0330[5] Valve	Fuel and oxidizer	2	X		1, 2
		ME 273-0012 Coupling	Spacer, fuel, and oxidizer	1	X		Same as above
Nitroso rubber	3	ME 284-0327[6] Check valve	Seals (oxidizer only)	4			To be determined
Butyl rubber		ME 284-0327[6] Check valve	Seals (fuel only)			X	

[1]Fluid Exposure Test Duration for 38 days total
[2]Fluid Exposure Test Duration for 15 days total
[3]Material in Development (for Oxidizer only)
[4]To be completed (for Oxidizer only).
[5]Replaces ME 284-0027 for Block II
[6]Replaces ME 284-0025 for Block II

Table 9. Propellant Compatibility - Tank Nonmetallics

Material	Specification	Where Used		Apollo Test Demonstration	Design Acceptable for AES		
		System/Part	Detail		Yes	No	Reference Basis
Butyl rubber	SR 634-70[1]	V37-480201	Door seals, fuel	2	X		9
		V37-470201	Door seals, oxidizer	2		X	9
		Gauging system	Sensor flanges, fuel	2	X		9
			Sensor flanges, oxidizer	2		X	9

[1] Stillman Rubber Company
[2] Fluid exposure test duration 28 days (ATR 301-13)

Table 10. Radiation Resistance—Engine Nonmetallics

Material	Specification	Where Used		SM Zone Dosage Ergs/Gm	Minimum Shielding	Max. Part Dosage Ergs/Gm	Acceptable Dosage Ergs/Gm*	Reference Basis
		Component	Detail					
RTV-60		Bipropellant valve		5.4×10^3 (1%) 3.6×10^4 (0.1%)			1.3×10^8	10, 11
Teflon, TFE		Bipropellant valve					1.7×10^6	11
Teflon, FEP		Bipropellant valve	Ball seal				1.7×10^6	11
Butyl rubber		Bipropellant valve	Ball seal				0.5×10^8 2×10^8	10 11
Zinc chromate	MIL-P-8116	Structural assembly	Injector chamber joint					
Sealant	MS-22473 GD. E, Color PR	Bipropellant valve						
5-122, Teflon with Freon Carrier	AGC 44159	Engine assembly						
Thd. Compound anti-sieze	AGC 44053	Engine assembly						
Compound microseal	MIL-A-907	Engine assembly bipropellant valve						
Glass cloth	Teflon impregnated	Harness assembly					1.7×10^6	11
Glass		Harness connector					10^7	11, 13

*Beginning of moderate damage

Table 10. Radiation Resistance—Engine Nonmetallics (Cont)

Material	Specification	Where Used		SM Zone Dosage Ergs/Gm	Minimum Shielding	Max. Part Dosage Ergs/Gm	Acceptable Dosage Ergs/Gm*	Reference Basis
		Component	Detail					
Thermofit tubing		Harness assembly		5.4×10^3 (1%) 3.6×10^4 (0.1%)			10^9	11
Adhesive	AGC 44076	Bipropellant valve	GN_2 system					
Epoxy		Engine	Combustion chamber	4.3×10^6 (1%) 2.9×10^7 (0.1%)			2.5×10^{11}	11
Phenolic resin		Engine	Combustion chamber				2.7×10^8	11
Buna-N		Bipropellant valve	GN_2 system	5.4×10^3 (1%) 3.6×10^4 (0.1%)			2×10^8	11
Nylon		Bipropellant valve	GN_2 system				10^8	10, 11
Kel-F		Bipropellant valve	GN_2 system				1.3×10^8	11
Kynar		Bipropellant valve	GN_2 system				3×10^6	12
Lubricant	Dow-Corning FS-1265-10, 000CS	Gimbal	Bearing assembly				10^9	11
	DC-11	Bipropellant valve	GN_2 system				10^9	11
	Dow-Corning FS-1290	Gimbal	Actuator gear train				10^9	11

*Beginning of moderate damage

Table 11. Radiation Resistance—Feed System Nonmetallics

Material	Specification	Where Used: Component	Where Used: Detail	SM Zone Dosage Ergs/Gm	Minimum Shielding	Maximum Part Dosage Ergs/Gm	Acceptable Dosage, Ergs/Gm[1]	Reference Basis
Kynar		ME 273-0009 Coupling	Pressure cap	5.8×10^7(1%) 3.9×10^8(0.1%)	0.120 Alum	$< 2 \times 10^5$	3×10^6	12
		ME 144-0023 Coupling			0.030 Stl	5.8×10^5(1%) 3.9×10^6(0.1%)		
		ME 284-0027 Relief valve	Lock plugs		0.040 Stl	4.4×10^5(1%) 2.9×10^6(0.1%)		
		ME 273-0012 Coupling	Poppet seal		0.038 Stl			
		ME 273-0011 Coupling			0.035 Stl	4.9×10^5(1%) 3.3×10^6(0.1%)		
Teflon, FED	AMS-3651	ME 273-0009 Coupling	Environmental seal		0.045 Stl	4×10^5(1%) 2.5×10^6(0.1%)	1.7×10^6	11
Teflon, FEP	AMS-3651	ME 284-0324 Regulator		4.5×10^4(1%) 3×10^5(0.1%)	0.120 Alum	$< 2 \times 10^5$		
	ASTM D 1457 Type 4	ME 284-0018 Solenoid valve	Pilot poppet	1×10^5(1%) 6×10^5(0.1%)	0.120 Stl	$<6 \times 10^4$		
		ME 284-0025 Check valve	Poppet seal	5.8×10^7(1%) 3.9×10^8(0.1%)	0.090 Stl	$<9 \times 10^4$		
		ME 284-0027 Relief valve	Poppet seal		0.040 Stl.	4.4×10^5(1%) 2.9×10^6(0.1%)		
	AMS-3651	ME 273-0012 Coupling	Spacer		0.038 Stl			
Buna-N	SR380-95[2]	ME 284-0018 Solenoid valve	Main poppet	1×10^5(1%) 6×10^5(0.1%)	0.120 Stl	$<6 \times 10^4$	2×10^8	11
Kel-F	AMS-3650 Grede 300	ME 284-0018 Solenoid valve			0.120 Stl.	$<6 \times 10^4$	1.3×10^8	11
Silicon rubber	DC 675 per MIL-R-5847 Class IB GD 70	ME 284-0324[3] Regulator	O-rings	4.5×10^4(1%) 3×10^5(0.1%)	0.10 Stl	$<8 \times 10^4$	1.3×10^8	11

[1]Beginning of moderate damage
[2]Stillman Rubber Company
[3]Replaces ME 284-0020 on Block II

Table 11. Radiation Resistance—Feed System Nonmetallics (Cont)

Material	Specification	Where Used		SM Zone Dosage Ergs/Gm	Minimum Shielding	Maximum Part Dosage Ergs/Gm	Acceptable Dosage, Ergs/Gm[1]	Reference Basis
		Component	Detail					
Silcone rubber (Cont)	Parker Comp. N-380-8	ME 284-0018 Solenoid valve	O-rings	1×10^5(1%) 6×10^5(0.1%)	0.120 Stl	$<6 \times 10^4$		
	OMC No 6	ME 273-0009 Coupling	O-ring	5.8×10^7(1%) 3.9×10^8(0.1%)	0.045 Stl	$<2 \times 10^5$		
Butyl rubber	SR 634-70[2]	ME 284-0327[5] Check valve	Poppet seal	5.4×10^3(1%) 3.6×10^4(0.1%)	0.090 Stl.	$<9 \times 10^4$	0.5×10^8 2×10^8	10
Resistazine 88	[4]	ME 284-0327[5] Check valve	Poppet seal		0.090 Stl.	$<9 \times 10^4$	[4]	
Mylar	MIL-I-631 Type G	ME 284-0324[3] Regulator	Diaphragm	4.5×10^4(1%) 3×10^5(0.1%)	0.120 Alum.	$<2 \times 10^5$	4.4×10^8	11
Microseal	No 100-1	ME 284-0324[3] Regulator			0.120 Alum.	$<2 \times 10^5$		
Drilube (O-ring lubricant)	Type 822	ME 284-0324[3] Regulator	Lubricant		0.10 Stl	$<8 \times 10^4$	10^9	11
Glass (compression seal)		ME 284-0018 Solenoid valve	Wire lead seal	1×10^5(1%) 6×10^5(0.1%)	0.032 Stl	$<2 \times 10^5$	10^7	10, 11
Mystik tape	#7000 G		Insulator		0.032 Stl	$<2 \times 10^5$	10^{11}	11
Fiber glass cloth	NEMA GD G-7 Class H		Insulator		0.032 Stl	$<2 \times 10^5$	10^7	10, 11
Plastic tubing	Tygon		Wire leads cover				10^9	11
Vinyl			Sleeve				10^9	10, 11
Nylon cord	MIL-T-713A Type P, Class 2 waxed		Tubing tiedown				10^8	10, 11

[1]Beginning of moderate damage
[2]Stillman Rubber Company
[3]Replaces ME 284-0020 on Block II
[4]In development
[5]Replaces ME 284-0025 on Block II

Table 12. Radiation Resistance — Tank Nonmetallics

Material	Specification	Where Used		SM Zone Dosage Ergs/Gm	Minimum Shielding	Maximum Part Dosage Ergs/Gm	Acceptable Dosage, Ergs/Gm*	Reference Basis
		System/Part	Detail					
Butyl rubber	SR634-70[1]	V37-470201 V37-480201 Tank installation Propellant	Door Seals (O-rings) PUGS[2] Sensor flange	5.8×10^7 (1%) 3.9×10^8 (0.1%)	0.20 0.15	2.1×10^5 (1%) 1.4×10^6 (0.1%) 2.9×10^5 (1%) 1.9×10^6 (0.1%)	0.5×10^8 2×10^8	10 11
Synthetic rubber	MD 261-0002, 0014 O-ring	V37-460201 He system	Tank joint	5.4×10^3 (1%) 3.6×10^4 (0.1%)	0.70 Ti		0.5×10^8 2×10^8	10 11

* Beginning of moderate damage
[1] Stillman Rubber Company
[2] Propellant utilization and gauging system

Table 13. Radiation Stability of Typical Engineering Materials

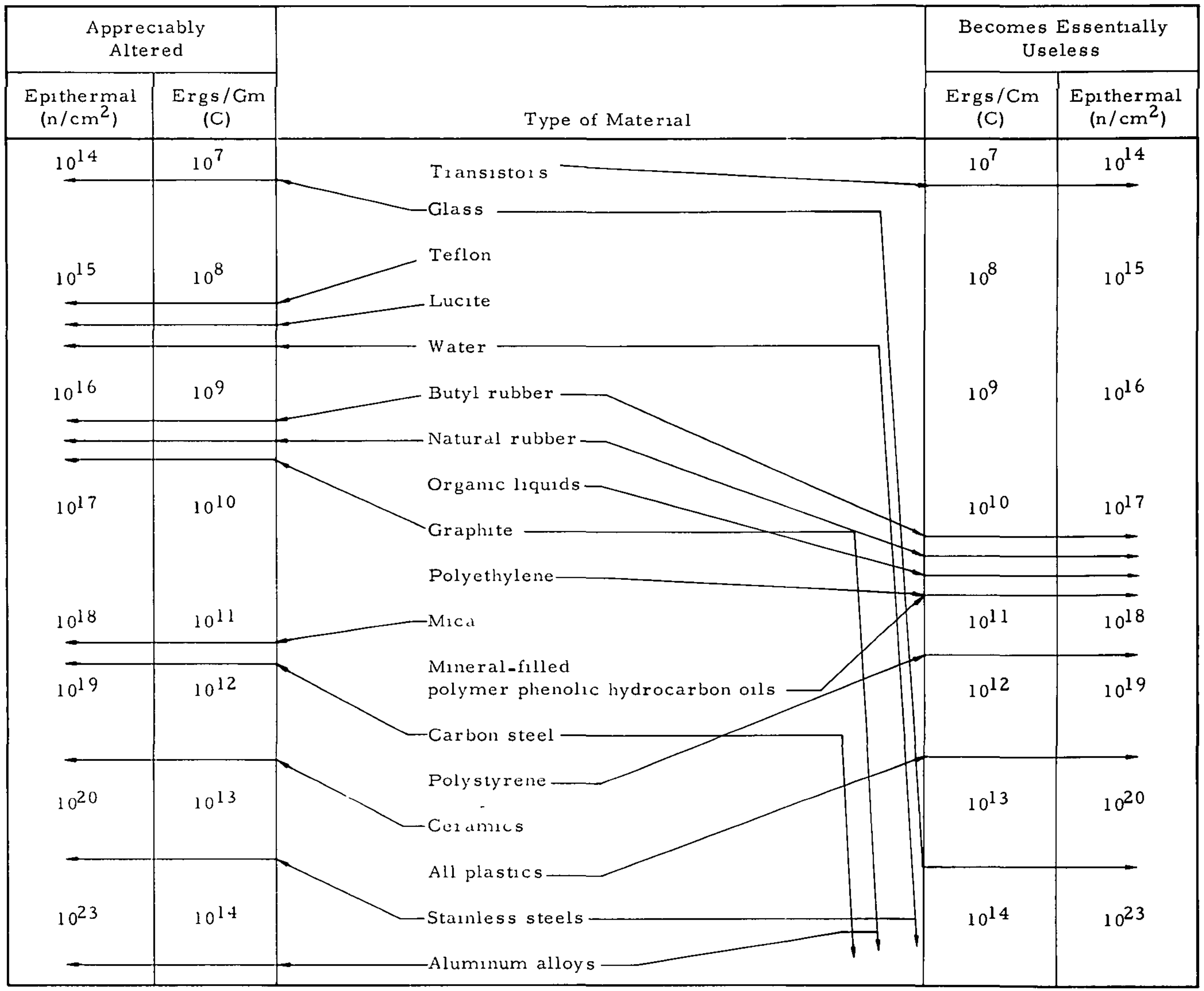


Table 14. Radiation Effects on Plastics and Rubber Materials

Material	Beginning of Moderate Damage ergs/gm (C)	Beginning of Serious Damage ergs/gm (C)
Plastics Materials		
Phenolic Laminates	10^{12}	10^{12}
Polystyrene	7.5×10^{10}	6.5×10^{11}
Polycater Laminates	7.5×10^{10}	5.5×10^{11}
Phenolics, Mineral-Filled	5×10^{10}	5×10^{11}
Silicones, Glass-Reinforced	5×10^{10}	5×10^{11}
Epoxy Resins	10^{10}	9×10^{10}
Phenolics, Unfilled	4.5×10^{9}	3×10^{10}
Polyvinyl Chloride	4.5×10^{9}	5×10^{10}
Amino Resins	2×10^{9}	9×10^{9}
Polyethylene	10^{9}	9×10^{9}
Cellulosics	8×10^{8}	5×10^{9}
Silicones	10^{8}	4.5×10^{9}
Polyamides	10^{8}	10^{9}
Polyesters, Unfilled	10^{8}	2.5×10^{8}
TFE Fluorocarbon	5×10^{6}	2.5×10^{7}
Rubber Materials		
Urethane	7.5×10^{9}	4.5×10^{10}
Natural	6×10^{9}	3×10^{10}
SBR	3.5×10^{9}	3×10^{10}
Nitrile	2×10^{9}	9.5×10^{9}
Neoprene	2×10^{9}	9.5×10^{9}
Acrylic	0.5×10^{9}	10^{10}
Silicone	8×10^{8}	6×10^{9}
Fluoroelastomers	8×10^{8}	4×10^{9}
Polysulfide	1.5×10^{8}	4.5×10^{8}
Butyl	0.5×10^{8}	4×10^{8}

Table 15. Material Resistance Data Sources

1. Titan II Storable Propellant Handbook, Revision B, March 1963

2. Aerojet-General Corporation, Liquid Rocket Plant, "Storable Liquid Propellants Nitrogen Tetroxide/Aerozine 50", Report LRP 198, Second Edition, June 1962

3. Lear-Siegler Design Verification Report 3620-2

4. Lear-Siegler Qualification Test Report TR1310.

5. J. C. Carter, Design Verification Report 3534

6. J. C. Carter, Design Verification Report 3534-1

7. J. C. Carter, Qualification Test Report 3534-2

8. J. C. Carter, Qualification Test Report 3534-4

9. NAA Engineering Development Laboratories Memo SNE 7-63-15, 16 July 1963

10. ARMA Report, IDEP No. 347.60.00.00-B5-03, April 1962

11. Radiation Effects Information Center, Battelle Memorial Institute, Report No. 21, September 1961

12. Pennsalt Chemical Corporation, Philadelphia, Pa.

13. Nuclear Environmental Testing, L. B. Gardner, Litton Industries, Proceedings of the Institute of Environmental Sciences, April 1960

14. NAA/S&ID Procurement Specifications

Table 16. Components/Materials Exposed to High Vacuum

COMPONENT	AFFECTED PART	MATERIAL
Relief Valve	Moving poppet Rupture disc	Teflon Aluminum
Engine Ball Valve	Moving seals Static seals	Teflon Teflon
Engine Ball Shaft	Moving seals	Teflon (Oxidizer) Butyl Rubber (Fuel)
Regulator	Diaphragm	Mylar
Propellant Tank Door	Static seal	Butyl Rubber
Helium Tank Connection	Static seal Static seal	Butyl Rubber Tin coated metal
Tank Probe Cover	Static Seal	Butyl Rubber
Engine Pneumatic Line	Static seal	Butyl Rubber

poppets and solenoid shuttles. None of these non-metallics is considered subject to serious degradation over a 45-day mission at the vacuum levels expected. The vapor pressures and outgassing rate data of typical elastomers are shown below.

Material	Vapor Pressure torr at 20°C	Outgassing Rate torr liters/sec cm^2
Silicone Rubber	10^{-5}	3×10^{-5}
Nylon	10^{-5}	- - -
Teflon	10^{-6}	5×10^{-6}
Kel-F	10^{-6}	5×10^{-6}
Silastic	10^{-5}	- - -
Neoprene Rubber	10^{-5}	2×10^{-4}
Butyl Rubber	No Data	1.5×10^{-6}
Mylar	No Data	- - -

The propellant tank relief valve, as presently designed, has a metal to metal stem slide bearing. However, it is planned that the Block II design will incorporate a Teflon sleeve bearing surface which will eliminate any cold welding problem. When the improved design is implemented, it should be reevaluated for AES application.

The components in the engine pneumatic (GN_2) pressurization system, such as the pressure regulator, solenoid valves (3 way), actuator, and relief valve are protected from vacuum exposure by check valves which maintain a slight back pressure on the components.

V. AES SERVICE PROPULSION SYSTEM

Two configurations (designated A and B) of different propellant capacities are recommended for the AES SPS. Configuration A uses all four of the Apollo Block II propellant tanks. Configuration B uses only two of the tanks and is for off-loaded missions requiring less than 53 percent of the Block II propellant capacity. Configuration B is identical to Configuration A except for: (1) deletion of two propellant storage tanks, and (2) plumbing and gaging changes necessitated by (1).

Both configurations are designed to provide the impulse necessary to change the linear momentum of the spacecraft for the normal and contingent AES mission maneuvers listed below.

1. Abort after LES jettison (abort to orbit or abort to entry)
2. Earth orbital plane changes
3. De-orbit from earth orbit
4. Translunar course corrections
5. Translunar aborts
6. Lunar orbit insertion
7. Lunar orbital plane changes
8. Lunar orbit abort
9. Transearth injection
10. Transearth course corrections
11. Earth orbit injection

The SPS will provide the spacecraft velocity increments for these maneuvers by providing an approximately constant thrust in the plus X direction through the center of gravity of the spacecraft. To provide the impulse required, the primary requirements of the SPS are as listed in Table 17.

CONFIGURATION A

The AES Phase II SPS is composed of the following integrated equipment as shown in Figure 7: engine, pressurization, propellant supply, propellant utilization and gaging system (PUGS), leak detection gaging, displays and controls, and electrical and instrumentation.

Table 17. AES SPS Performance

ITEM	CONFIGURATION A	CONFIGURATION B
Reliability (LPO Mission) Prediction		
Mission Success	0.99608	N.A.
Crew Safety	0.99736	N.A.
Dry Weight (including tanks)	2,830 lbs	2,140 lbs
Usable Propellant (maximum)	39,700 lbs	21,040 lbs
Thrust (nominal)	20,000 lbs	20,000 lbs
Instantaneous Specific Impulse	311.7 sec (-3 sigma at end of 750 sec of firing)	311.7 sec (-3 sigma at end of 750 sec of firing)
Number of starts	36	36
Minimum impulse per start	5,000 lb sec	5,000 lb sec
Engine minimum run-to-run tolerance	±200 lb sec	±200 lb sec
Shutdown Impulse	8,830 to 14,200 lb sec	8,830 to 14,200 lb sec
Shutdown tolerance	±300 lb sec	±300 lb sec

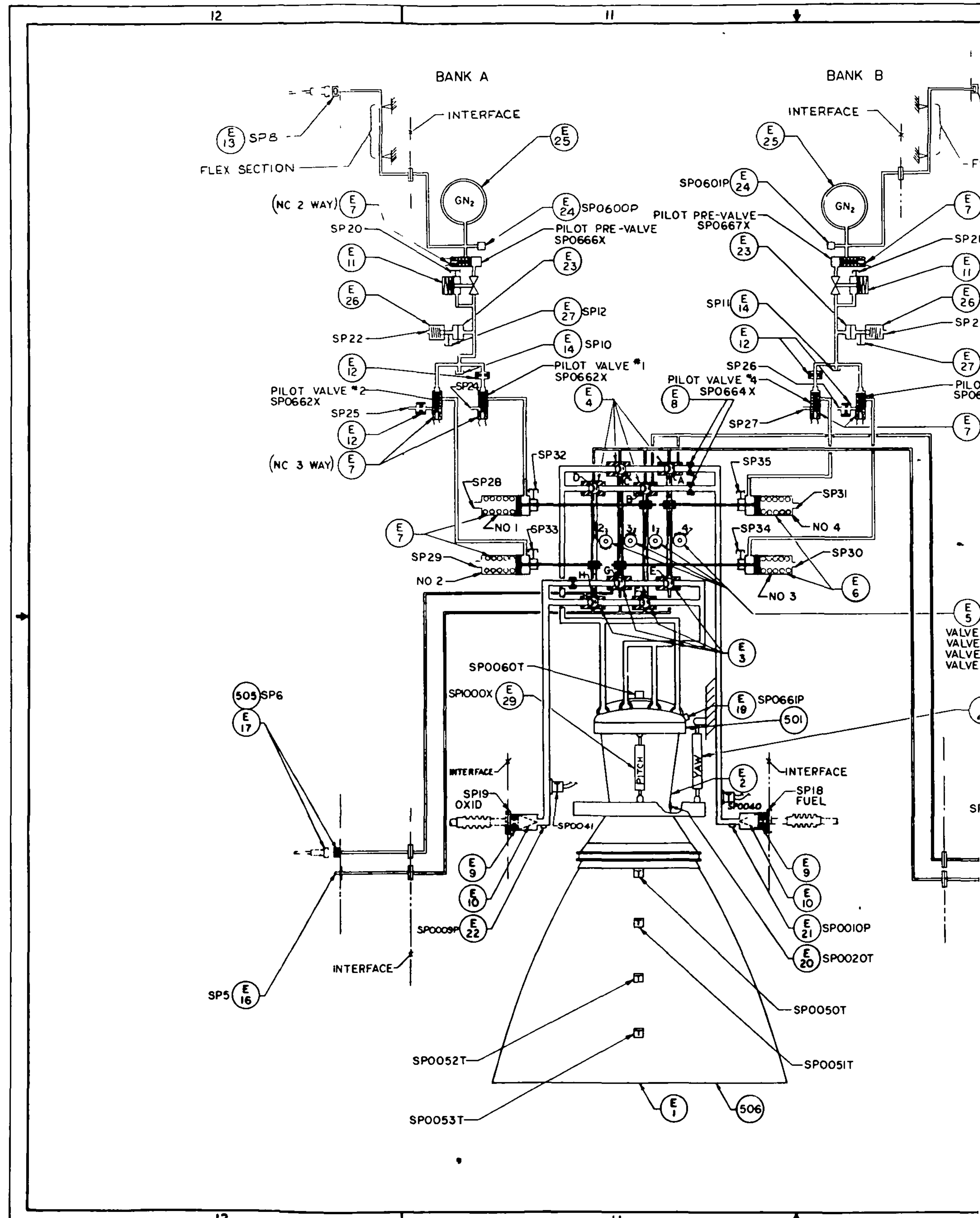

BANK A
BANK B
INTERFACE
FLEX SECTION
SP8
GN2
SP0600P
SP0601P
PILOT PRE-VALVE
SP0666X
SP0667X
(NC 2 WAY)
(NC 3 WAY)
SP20
SP22
SP12
SP10
SP11
SP26
SP24
SP25
SP27
PILOT VALVE #1
SP0662X
PILOT VALVE #2
SP0662X
PILOT VALVE #4
SP0664X
SP28
SP29
SP30
SP31
SP32
SP33
SP34
SP35
NO 1
NO 2
NO 3
NO 4
SP0060T
SP1000X
SP0661P
PITCH
YAW
SP19
OXID
SP18
FUEL
SP0041
SP0040
SP0009P
SP0010P
SP0020T
SP0050T
SP0051T
SP0052T
SP0053T
SP5
SP6
505
501
506

Fig 7 (1)

10	9	8

(E 13) SP 9

X SECTION

C 2WAY)

13

VALVE #3
X

C 3WAY)

SP0022, SP0026
SP0023, SP0027
SP0024, SP0028
SP0025, SP0029

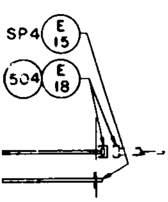
) SP1001X

ENGINE ASSEMBLY

E 1 NOZZLE EXTENSION
2 THRUST CHAMBER ASSEMBLY
3 PROPELLANT VALVE-OXIDIZER (NC)
4 PROPELLANT VALVE-FUEL (NC)
5 PROPELLANT VALVE POSITION POT (REDUNDANT COIL)
6 PROPELLANT VALVE PNEUMATIC ACTUATORS
7 PROPELLANT CONTROL PILOT VALVE-
SOLENOID ACTUATED NC
8 LOOP BALANCING ORIFICES
9 ENGINE SYSTEM BALANCING ORIFICES
10 PROPELLANT SCREEN
11 PRESSURE REGULATOR, PNEUMATIC (N_2)
12 FLOW CONTROL & TRIM ORIFICES
(5) 13 PNEUMATIC FILL & VENT GN_2 PRESSURIZING, SELF SEAL
TYPE, DISCONNECT CAPPED FOR FLIGHT
14 REGULATOR OUTLET TEST PORT SELF SEAL TYPE
DISCONNECT CAPPED FOR FLIGHT
15 PROPELLANT VALVE SEAL DRAIN (FUEL)
OPEN TYPE QUICK DISCONNECT
16 PROPELLANT VALVE SEAL DRAIN (OXIDIZER)
OPEN TYPE QUICK DISCONNECT
17 PROPELLANT INLET MANIFOLD FILL VENT (OXID)
SELF SEAL TYPE QUICK DISCONNECT
WITH CAP FOR ENGINE OPERATION
18 PROPELLANT INLET MANIFOLD FILL VENT, (FUEL)
SELF SEAL TYPE QUICK DISCONNECT
WITH CAP FOR ENGINE OPERATION
19 THRUST CHAMBER PRESSURE BOSS
20 THRUST CHAMBER WALL TEMPERATURE SENSOR
21 FUEL INLET PRESSURE BOSS
22 OXIDIZER INLET PRESSURE BOSS
23 BURST DISC
24 PNEUMATIC STORAGE PRESSURE BOSS
25 PNEUMATIC STORAGE TANK (GN_2)
26 RELIEF VALVE
27 RELIEF VALVE TEST PORT SELF SEAL TYPE,
DISCONNECT CAPPED FOR FLIGHT
28 GIMBAL ACTUATOR (YAW)
29 GIMBAL ACTUATOR (PITCH)

ENGINE SERVICE POINTS

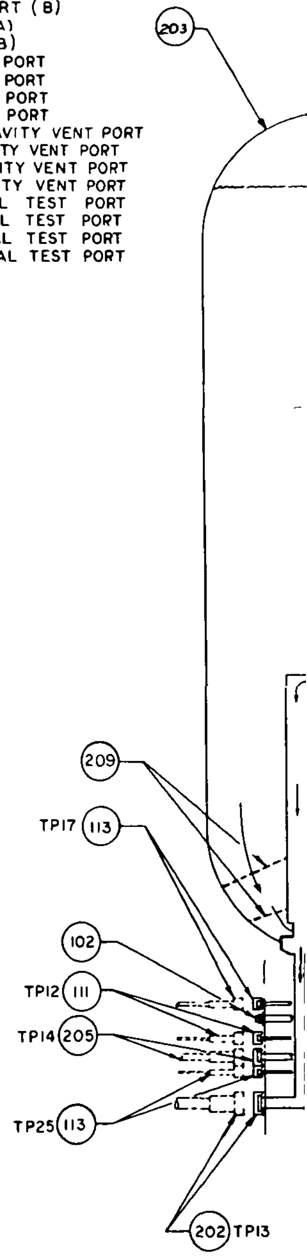
SP 4-PROP VALVE SEAL DRAIN (FUEL)
SP 5-PROP VALVE SEAL DRAIN (OXID)
SP 6-PROP INLET MANIFOLD FILL VENT (OXID)
SP 7-PROP INLET MANIFOLD FILL VENT (FUEL)
SP 8-PNEUMATIC FILL & VENT (GN_2) BANK A
SP 9-PNEUMATIC FILL & VENT (GN_2) BANK B
SP10-REG OUTLET TEST PORT (A)
SP11-REGULATOR OUTLET TEST PORT (B)
SP12-RELIEF VALVE TEST PORT (A)
SP13-RELIEF VALVE TEST PORT (B)
SP18-FUEL INLET
SP19-OXIDIZER INLET
SP20-REG AMBIENT SENSING PORT (A)
SP21-REG AMBIENT SENSING PORT (B)
SP22-RELIEF VALVE OUTLET (A)
SP23-RELIEF VALVE OUTLET (B)
SP24-PILOT VALVE NO 1 VENT PORT
SP25-PILOT VALVE NO 2 VENT PORT
SP26-PILOT VALVE NO 3 VENT PORT
SP27-PILOT VALVE NO 4 VENT PORT
SP28-ACTUATOR NO 1 SPRING CAVITY VENT PORT
SP29-ACTUATOR NO 2 SPRING CAVITY VENT PORT
SP30-ACTUATOR NO 3 SPRING CAVITY VENT PORT
SP31-ACTUATOR NO 4 SPRING CAVITY VENT PORT
SP32-ACTUATOR NO 1 SHAFT SEAL TEST PORT
SP33-ACTUATOR NO 2 SHAFT SEAL TEST PORT
SP34-ACTUATOR NO 3 SHAFT SEAL TEST PORT
SP35-ACTUATOR NO 4 SHAFT SEAL TEST PORT



VALVE NUMBER	BALL DESIGNATION	DESCRIPTION	LOOP
1	B	LOWER UPSTREAM FUEL	PRIMARY
	F	LOWER DOWNSTREAM OX	
2	D	LOWER DOWNSTREAM FUEL	
	H	LOWER UPSTREAM OX	
3	C	UPPER DOWNSTREAM FUEL	SECONDARY
	G	UPPER UPSTREAM OX	
4	A	UPPER UPSTREAM FUEL	
	E	UPPER DOWNSTREAM OX	

	ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
SP			264004-4800	LEAR SEIGLER	COUPLING, TEST, REL VALVE (GROUND)	
12 & 13		2	(1)	AEROJET-GENERAL	COUPLING, TEST, REL VALVE (FLIGHT)	
SP			264004-4600	LEAR-SEIGLER	COUPLING, REG OUTLET (GROUND)	
10 & 11		2	(1)	AEROJET-GENERAL	COUPLING, REG OUTLET (FLIGHT)	
SP 8	(5)	2	ME144 0023-0081	LEAR-SEIGLER	COUPLING, FILL VENT (GROUND)	MC144-0023
& 9	(5)	2	ME144-0023-0071	LEAR SEIGLER	COUPLING, FILL VENT (FLIGHT)	MC144-0023
	506	1	ME901-0192-0001	AEROJET-GENERAL	NOZZLE EXTENSION, SPS ENGINE	MC901-0009
SP6	505	1	ME273-0011-0002	J.C CARTER	COUPLING, ENG. VENT & DRAIN (GROUND)	MC273 0011
		1	ME273-0011-0001		COUPLING, ENG. VENT & DRAIN (FLIGHT)	
SP7	504	1	ME273-0024-0002	J.C CARTER	COUPLING, FILL VENT (GROUND)	MC273-0011
		1	ME273-0024-0001		COUPLING, FILL VENT (FLIGHT)	
	501	1	ME321 0004 0001	AEROJET-GENERAL	ROCKET ENGINE-SERVICE PROPULSION	MC901-0009
	ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC

ROCKET ENGINE SYSTEM



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10	9	8

FIG 7(2)

CONFIDENTIAL

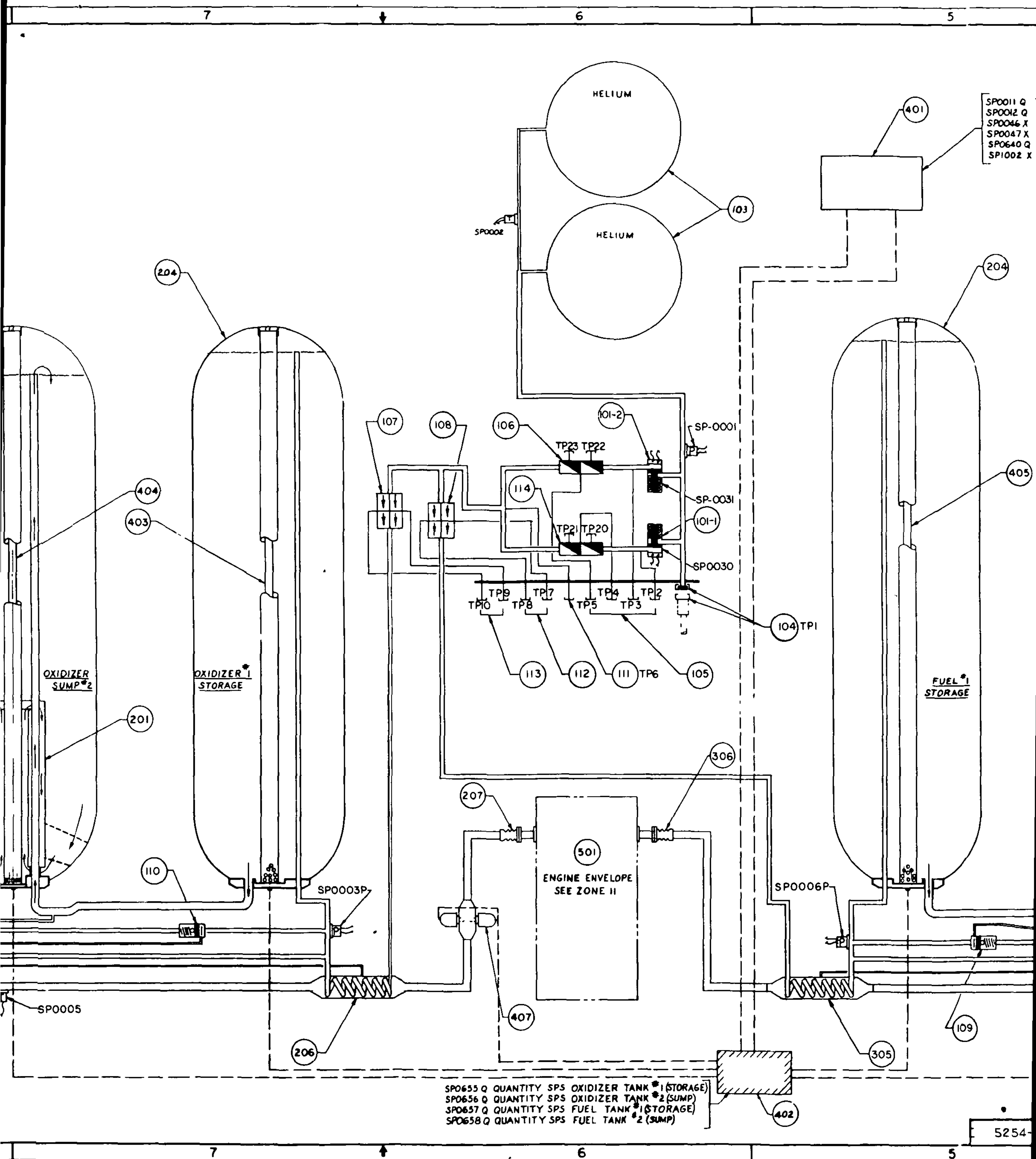

Fig 7 ③

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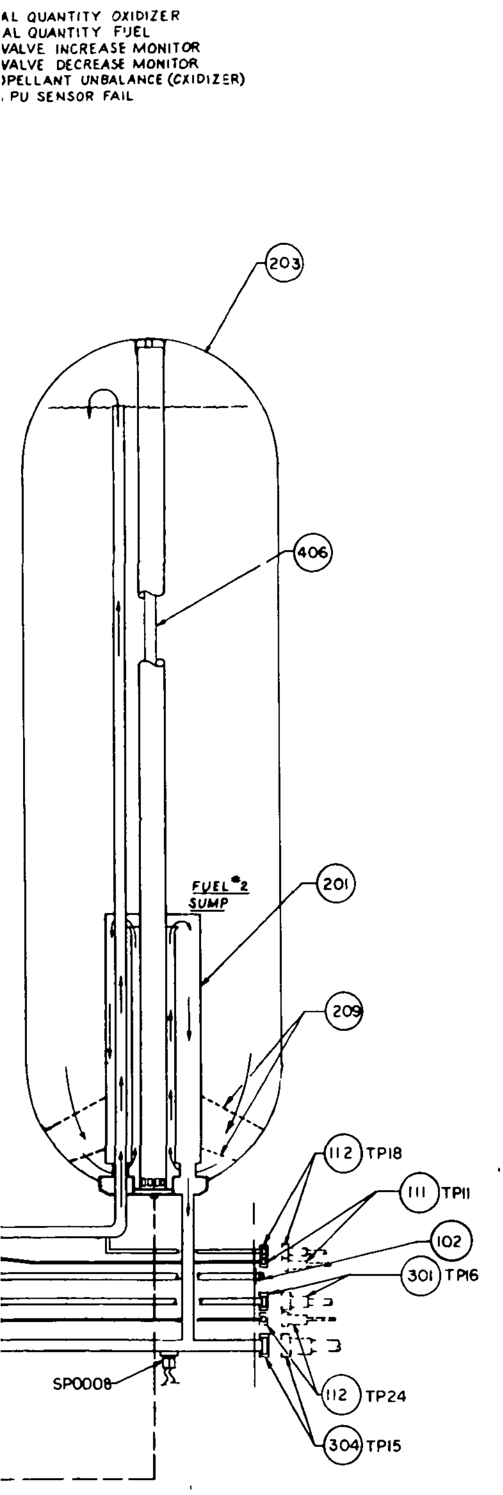


ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061	LEAR SIEGLER	COUPLING TP11 GROUND	MC144-0023
	1	ME144 0023 0051		COUPLING TP11 FLIGHT	
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	
112	2	ME144-0023-0041	LEAR SIEGLER	COUPLING, RESIDUAL DRAIN (GROUND)	MC144-0023
	2	ME144-0023-0031		COUPLING, RESIDUAL DRAIN (FLIGHT)	
306	1	ME273-0071-0001	AEROQUIP CORP	FLEX JOINT, FUEL LINE	MC273-0071
305	1	ME362-0019 0011	UNITED AIRCRAFT PROD	HEAT EXCHANGER, He LOW PRESS	MC362-0019
304	1	ME273-0020-0002	J.C. CARTER	COUPLING, FILL & DRAIN (GROUND)	MC273-0012
	1	ME273-0020-0001		COUPLING, FILL & DRAIN (FLIGHT)	
203	1	V37-342102	NAA	TANK, FUEL SUMP	
204	1	V37-343102	NAA	TANK, FUEL STORAGE	
301	1	ME273-0012-0002	J.C. CARTER	COUPLING, FUEL VENT (GROUND)	MC273-0012
	1	ME273-0012 0001		COUPLING, FUEL VENT (FLIGHT)	

FUEL DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061	LEAR SIEGLER	COUPLING TP12 GROUND	MC144-0023
	1	ME144-0023-0051		COUPLING TP12 FLIGHT	
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
113	2	ME144-0023-0021	LEAR-SIEGLER	COUPLING, OX RESIDUAL DRAIN (GROUND)	MC144-0023
	2	ME144-0023-0011		COUPLING, OX RESIDUAL DRAIN (FLIGHT)	
207	1	ME273-0040-0001	AEROQUIP CORP	FLEX JOINT OXIDIZER LINE	MC273-0040
206	1	ME362-0019-0001	UNITED AIRCRAFT PROD.	HEAT EXCHANGER He LOW PRESS	MC362-0019
205	1	ME273-0022-0002	J.C. CARTER	COUPLING OX VENT (GROUND)	MC273-0012
	1	ME273-0022-0001		COUPLING, OX VENT (FLIGHT)	
204	1	V37-343102	NAA	TANK, OXIDIZER STORAGE	
203	1	V37-342102	NAA	TANK, OXIDIZER SUMP	SID65-762
202	1	ME273-0018-0002	J.C. CARTER	COUPLING OX FILL & DRAIN (GROUND)	MC273-0012
	1	ME273-0018-0001		COUPLING, OX FILL & DRAIN (FLIGHT)	
201	1	V37-470240	NAA	RSVR PROPELLANT RETENTION	

OXIDIZER DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
114	1	ME284-0324-0012	B.H. HADLEY CO	REGULATOR, PRESS He (2 STAGE)	MC284-0324
113	2	ME144-0023-0021	LEAR-SIEGLER	COUPLING, TP 9 & 10 (GROUND)	MC144-0023
	2	ME144-0023-0011		COUPLING, TP 9 & 10 (FLIGHT)	
112	2	ME144-0023-0041	LEAR-SIEGLER	COUPLING TP 7 & 8 (GROUND)	MC144-0023
	2	ME144-0023-0031		COUPLING, TP 7 & 8 (FLIGHT)	
111	1	ME144-0023-0061	LEAR-SIEGLER	COUPLING, TP6 (GROUND)	MC144-0023
	1	ME144-0023-0051		COUPLING, TP6 (FLIGHT)	
110	1	ME284-0330-0001	CALMEC	VALVE, RELIEF OXIDIZER	MC284-0330
109	1	ME284-0330-0011	CALMEC	VALVE, RELIEF FUEL	MC284-0330
108	1	ME284-0327 0002	APCO	VALVE, QUAD CHECK, FUEL	MC284-0327
107	1	ME284-0327-0012	APCO	VALVE, QUAD CHECK, OXIDIZER	MC284-0327
106	1	ME284-0324 0002	B.H. HADLEY	REGULATOR, PRESS He (2-STAGE)	MC284-0324
105	4	ME144-0023-0081	LEAR-SIEGLER	COUPLING, TP 2, 3, 4 & 5 (GROUND)	MC144-0023
	4	ME144-0023-0071		COUPLING, TP 2, 3, 4 & 5 (FLIGHT)	
104	1	ME273-0009-0002	ON-MARK	COUPLING, FILL & DRAIN (GROUND)	MC273-0009
	1	ME273-0009-0007		COUPLING, FILL & DRAIN (FLIGHT)	
103	2	V37-347102	NAA	VESSEL, PRESSURE HELIUM	SID65-762
102	2	V17-460215	NAA	UNION, OVERBOARD RELIEF	———
101	2	ME284-0018-0005	APCO	VALVE, SOLENOID SHUT-OFF	

PRESSURIZATION SYSTEM (HELIUM)

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
407	1	ME284-0219-0001	SIMMONDS PREC. PROD.	VALVE ASSEMBLY P.U	MC901-0008
406	1	ME432-0183-0001		SENSING PROBE FUEL TANK #2 (SUMP)	MC610-0001
405	1	ME432-0182-0001		SENSING PROBE FUEL TANK #1 (STOR)	
404	1	ME432-0181-0001		SENSING PROBE OX TANK #2 (SUMP)	
403	1	ME432-0180-0001		SENSING PROBE OX TANK #1 (STOR)	
402	1	ME450-0008-0001		CONTROL UNIT	
401	1	ME181-0139-0001		DISPLAY PANEL C/M	MC610-0001

P.U. AND GAUGING SYSTEM

5 5 SUPPLIED BY NAA/SID AT INSTALLATION
4 REF SID 65-1527 REPORT
3 REF INSTL DWGS V37-460201 HELIUM V37-470201 OXIDIZER V37-480201 (FUEL) V37-470202 (GAUGING) V37-410001 (ENGINE)
2 TP20, 21 22 & 23 ARE AND10050-2 PORTS PER SCD ME284-0324-0002
1 1 AIRBORNE HALF OF DISCONNECTS SUPPLIED BY AEROJET-GENERAL AS PART OF ENG ASSY

NOTES UNLESS OTHERWISE NOTED

Figure 7. Service Propulsion Fluid Feed System for AES Phase II

FIG 7 (4)

5254-223

The SPS provides a nearly constant propulsive force through combustion and expansion of propellants supplied under pressure to a combustion chamber. The engine and its thrust chamber are gimbal-mounted with actuators for thrust directional control. The oxidizer is nitrogen tetroxide, and the fuel is a mixture of unsymmetrical dimethylhydrazine and hydrazine. The propellants are forced to the engine from their tanks by gas displacement at a controlled pressure, using helium from high-pressure storage vessels. Propellant gaging is provided to indicate fuel and oxidizer quantity and ratio imbalance during firing. Displays for monitoring critical SPS parameters and controls for operating certain SPS functions are contained within the CM.

ENGINE

The engine is a non-throttleable, gimballed, pressure-fed rocket engine consisting of an ablation-cooled combustion chamber, radiation-cooled nozzle extending from 6:1 to 52.6:1 area ratio, aluminum injector, thrust and gimbal mount, propellant valves, propellant valve actuation package, gimbal actuators, electrical harness, propellant lines, and drain and vent lines. The engine operates nominally at a chamber pressure of 96.0 psia and a vacuum thrust of 20,000 pounds at a minimum specific impulse of 311.7 seconds. The engine is capable of a minimum of 36 restarts with engine design life of 750 seconds.

Propellants are injected into the thrust chamber at a nominal mixture ratio of 1.6 to 1 by weight of oxidizer to fuel, and ignition occurs by hypergolic reaction. Control of propellant to the thrust chamber assembly is provided by a redundant set of series-parallel ball valves actuated by pneumatic pressure controlled by electrically operated solenoid valves. The pneumatic tank and lines upstream of the shut-off valve have a volume of 125 $\pm$ 5 cubic inches and are filled with GN_2 to a pressure and temperature equivalent to 2500 $\pm$ 5 psig at 70F.

Engine gimballing on the yaw and pitch axis is accomplished by electrical actuators (26 volt dc) with redundant motors, gear train, and magnetic clotches operating on a single ball-screw shaft.

PRESSURIZATION EQUIPMENT

The pressurization equipment consists of a high-pressure helium supply contained within two spherical vessels of internal volumes of 19.6 cubic feet each, and associated pressure regulators, isolation valves, check valves, and relief valves. Nominal vessel fill pressure is 3685 psi. The fuel and oxidizer storage tanks are pressurized from a common helium source. Two solenoid-operated poppet valves, installed in parallel downstream of the helium pressure vessels isolate the helium supply when the propulsion subsystem is not operating. Propellant pressure control is accomplished by two helium pressure regulator assemblies installed in parallel, one assembly downstream of each of the helium supply isolation valves. Each regulator assembly consists of a primary and a secondary regulator in series. The primary regulator reduces the variable upstream pressure to the normal downstream operating pressure of 185 psi. The secondary regulator in the assembly remains open as long as the primary regulator functions properly. A series-parallel installation of check valve assemblies, downstream of the helium pressure regulators, prevents backflow of propellants into the pressurization equipment. One oxidizer and one fuel pressure relief valve assembly, each consisting of a relief valve, a burst diaphragm and a filter screen, provide protection for the respective propellant tanks.

PROPELLANT SUPPLY EQUIPMENT

The propellant supply equipment contains an oxidizer and a fuel supply and distribution system. The oxidizer is contained within two hemispherically domed, cylindrical titanium tanks. The sump tank and storage tank have internal volumes of 160.5 cubic feet and 127.8 cubic feet, respectively, and contain a maximum of 24,430 pounds of usable oxidizer.

Normal working pressure in the tanks is 175 psia. The maximum working (limit) pressure is 225 psid. The two tanks are connected in series by a single propellant transfer line, with the upstream tank utilized as a storage tank and the downstream tank as a sump. Oxidizer is supplied to the engine through a propellant retention reservoir in the bottom of the sump tank and a line from the reservoir outlet to the engine. The fuel equipment is identical in size and configuration to the oxidizer equipment and contains a maximum of 15,270 pounds of usable fuel.

PROPELLANT UTILIZATION AND GAGING EQUIPMENT

The propellant utilization and gaging equipment consists of oxidizer and fuel tank primary and auxiliary quantity-sensing devices, an electrical control unit, an oxidizer flow control valve assembly, and a crew display panel. This equipment monitors the quantities of propellants remaining during periods of SC acceleration in order that the desired oxidizer-to-fuel consumption rate ratio can be manually on-off step controlled during firing, to assure that depletion of the oxidizer and fuel occurs almost simultaneously.

The primary quantity sensors are cylindrical capacitance probes mounted axially in each tank. Auxiliary gaging is accomplished by impedance point sensors which provide a step function impedance change when the liquid level passes their location. Oxidizer flow control is accomplished by manual control of a motor-operated, redundant, double-blade valve assembly which has three positions: increased, decreased, or normal oxidizer flow rate. No fuel flow control is provided. The control unit provides circuitry to perform all required computations for the gaging equipment. The function of the control unit is: (1) to compute total propellant quantities from individual tank quantities, (2) to compute propellant imbalance, (3) to compare continuously during SPS firing oxidizer-fuel ratio, and (4) to compare continuously, during SC acceleration, total propellant quantities

indicated by primary and auxiliary equipment. The crew display panel provides onboard indications required by the crew and switches for control functions and onboard testing.

DISPLAYS AND CONTROLS

The following CM control functions and displays are provided for the SPS for both Block II and AES Phase II.

Engine Control Switches

Gimbal actuator motor ON-OFF switches Y_1, Y_2, P_1, P_2
Two prepilor valve ON-OFF switches
Gimbal actuator trim thumb wheels, pitch and yaw
Thrust ON switch.

Propellant Management Switches

PU Valve, oxidizer flow
INCREASE-DECREASE switch
PU Valve, primary-secondary
Gate selector switch
PU sensor, primary-auxiliary switch for change from continuous sensing to point sensing
PU auto-test
Self-test switch
Two helium isolation valves
ON-OFF auto switch

SPS Caution and Warning Indications

Pitch gimbal drive fail
Yaw gimbal drive fail
Flight Combustion Stability Monitor (FCSM)
SPS propellant critical imbalance warning
SPS wall temperature high
SPS pressure (tank overpressure)

Propellant Equipment Quantity and Oxidizer-Fuel Usage Presentations

Oxidizer quantity meter
Fuel quantity meter
Unbalance meter

SPS Pressure and Temperature Presentations

GN_2 tank, pressure (engine valve actuation) (2)
Helium tank, pressure and temperature meter
Propellant tank, fuel pressure meter
Propellant tank, oxidizer pressure meter

Engine inlet, fuel pressure meter
Engine inlet, oxidizer pressure meter
Thrust chamber, chamber pressure meter

Other SPS Indications

Oxidizer flow increase indication
Oxidizer flow decrease indication
Helium isolation valves (2) OPEN-CLOSE indications
Engine valve OPEN-CLOSE indication (4)
Engine yaw gimbal position indication
Engine pitch gimbal position indication
Thrust ON indication

SPS GROUND SUPPORT EQUIPMENT

Ground support equipment required to support AES SPS test and launch operations includes servicing equipment, checkout equipment, handling equipment, and auxiliary test sets needed to simulate CM control subsystems for ground test operations. Table 18 summarizes AES SPS GSE showing the model designation and the facilities where the equipment is required.

SPS OPERATION

Prior to SPS operation, the engine prepilot valves upstream of the pilot valves are opened, the gimbal actuator electric motors are started, and the engine thrust vector position is aligned with the vehicle center-of-mass by the use of manual controls on the control panel. With the pre-pilot valves open, the gimbal motors running, and the thrust vector aligned, the SPS is activated upon receiving signals from the guidance and control subsystem or by manual override. When the SPS firing signal is received, a 28-volt direct current energizes the pilot engine pneumatic system pilot valves that admit pneumatic pressure to the propellant valve actuators. The actuators, attached by mechanical linkage to the main propellant valves, open the valves. Propellants flow into the injector and combustion chamber and ignite hypergolically to provide engine thrust. The 28-volt starting signal also simultaneously opens the helium isolation valves to provide continuous propellant tank pressurization during propellant consumption.

Engine shutdown is accomplished by removing the 28-volt dc signal from the propellant valve actuator pilot valves. When in the de-energized position, the pilot valves dump the pneumatic pressure, allowing the main propellant valves to close. The 28-volt signal is simultaneously removed from the helium isolation valves. Subsequent to engine shutdown, the gimbal actuator motors and the prepilot valves are manually de-energized.

During engine operation, thrust vector alignment is controlled by closed-loop operation with an integrated guidance and control subsystem. During all normal SPS firings, the THRUST ON signal is provided by the guidance and control subsystem, with the spacecraft crew performing the

Table 18. AES SPS Ground Support Equipment

Model No.	Descriptive Title	Use Location			
		Downey	WSMR/ PSDF	MSC	KSC
	MAJOR EQUIPMENT				
A34-243	Umbilical disc set	X	X	X	X
A14-028	Optical alignment set (engine installation)		X	X	X
A14-044-0001	Nozzle closure, 6:1 nozzle	X	X	X	X
A14-045-0001	Nozzle plug, leak and pressure, 140-psi	X	X	X	X
A14-051-101	CM substitution unit	X	X	X	X
A14-056-0002	SM gimbal locking fixture link	X	X	X	X
A14-060-0001	Nozzle extension closure			X	X
A14-125	Nozzle extension leak check test set		X		X
A14-140	Inertia simulator, nozzle extension, nonfireable	X	X		X
A14-036	Ground air circulation unit	X			X
A14-046	Crane control auxiliary	X		X	X
A14-208	Nozzle extension SPS engine inertia simulator, fireable		X		X
C14-017	SPS engine installation alignment fixture	X	X	X	X
C14-075-201	SPS pneumatic checkout unit	X	X	X	X
C14-089-101	Mobile data recorder, all systems	X	X	X	X
C14-163	SPS PUGS BME	X	X	X	X
C14-408	Thrust chamber alignment BME	X	X	X	X
C14-449	SPS control unit, He servicing	X	X	X	X

Table 18. AES-SPS Ground Support Equipment (Continued)

Page 2 of 4

Model No.	Descriptive Title	Use Location: Downey	WSMR/ PSDF	MSC	KSC
MAJOR EQUIPMENT (Continued)					
C14-455-101	SPS remote control relay unit, static firing	X	X	X	X
C14-468	Freon unit, engine leak check	X	X	X	X
C14-488	Control unit, fuel servicing	X	X	X	X
C14-489	Control unit, oxidizer servicing	X	X	X	X
C14-602	STU, SPS control unit, checkout and static firing		X		X
A14-231	Dummy panel, Helium access				X
H14-024-0001	SPS engine sling	X	X	X	X
H14-064	Helium tank sling	X	X	X	X
H14-072	Mission propulsion engine support	X	X	X	X
H14-089	Polarity checker, engine gimbal actuation		X	X	X
H14-179	Gimbal test support stand, SPS engine	X	X		
S14-002-101 -201	Oxidizer transfer unit, SPS servicing		X	X	X
S14-003	Mass spectrometer, helium leak test	X	X		
S14-008	Fuel transfer unit, SPS servicing				X
S14-008-101	Fuel transfer unit, SPS servicing	X			
S14-008-201	Fuel transfer unit, SPS servicing	X	X		X
S14-009-0001	Helium transfer unit, SPS servicing		X	X	X

Table 18. AES-SPS Ground Support Equipment (Continued)

Model No.	Descriptive Title	Use Location			
		Downey	WSMR/ PSDF	MSC	KSC
	MAJOR EQUIPMENT (Continued)				
S14-014	Fluid distribution equipment, house spacecraft servicing	X			
S14-022-0002	Helium booster unit, 6000 psi	X	X	X	X
S14-035	Fluid distribution equipment, static firing servicing				X
S14-058	Fuel ready storage unit	X	X		X
S14-059-0001	Oxidizer ready storage unit		X	X	X
S14-060	Toxic vapor disposal, fuel		X		X
S14-061	Toxic vapor disposal, oxidizer		X		X
S14-062-0001	Helium ready storage		X	X	X
S14-069	Moisture monitor unit, decontamination	X	X		X
S14-070	SPS engine decontamination unit		X		
S14-071	Mobile induction brazing unit		X	X	X
S14-045-101	Fluid distribution equipment, thermal vacuum			X	
S14-082	Fluid distribution equipment, Building 290	X			
S14-084-0001	Halogen leak detector unit	X	X	X	X
S14-099	Pressurizing unit, SPS tank protection, shipping and storage	X	X	X	X
S14-113	Helium regulator ambient sensing port		X		X
	HANDLING AND MISCELLANEOUS EQUIPMENT				
H14-124	Work stand - integration	X		X	X

Table 18. AES-SPS Ground Support Equipment (Continued)

Page 4 of 4

Model No.	Descriptive Title	Use Location			
		Downey	WSMR/ PSDF	MSC	KSC
	HANDLING AND MISCELLANEOUS EQUIPMENT (Continued)				
A14-043-0001	Protect cover, kit, nozzle extension, handling		X	X	X
H14-052	Positioning trailer, engine handling	X	X	X	X
H14-060	Fuel and oxidizer tank sling	X		X	X
H14-120	Support fixture, SPS engine checkout and maintenance	X	X		
H14-132	SPS engine dolly	X	X	X	X
H14-134	Spacecraft integrated test support base	X	X	X	X
H14-157	SPS nozzle extension handling fixture	X	X	X	X
S14-	Cable sets, spacecraft, GSE, and ACE	X	X	X	X
H14-105	Dolly, Service Module	X			

prefire operations described above. However, a manual THRUST ON switch is provided to allow manual firing of the rocket engine by the crew during certain emergency modes.

SPS SUBCONTRACTORS

Table 19 lists the Apollo Block II subcontractors and procurement specification numbers for SPS components and subassemblies.

Table 19. SPS Procured Components and Subassemblies

Component or Subassembly	Specification	Subcontractor
Valve, solenoid, normally closed, helium, 5/8 inch	MC 284-0018	Accessory Products
Regulator Unit, pressure helium, SPS (4500 psia)	MC 284-0324	B. H. Hadley
Check valve assembly series-parallel, helium	MC 284-0327	Accessory Products
Valve, relief, pressure, helium, SPS	MC 284-0330	Calmec Manufacturing
Heat exchanger, oxidizer-helium	MC 362-0019	United Aircraft Products
Heat exchanger, fuel-helium	MC 362-0019	United Aircraft Products
Coupling, helium, SPS, disconnect	MC 273-0009	On-Mark Couplings
Coupling, propellant fill and vent, SPS, disconnect	MC 273-0012	J. C. Carter
Coupling, system test point disconnect	MC 144-0023	Lear-Siegler
Flex connector, fuel	MC 273-0071	Aeroquip
Flex connector, oxidizer	MC 273-0040	Aeroquip
Utilization and gaging equipment, propellant, SPS	MC 610-0001	Simmonds Precision Products
Rocket engine, service propulsion system	MC 901-0009	Aerojet-General
Oxidizer tanks: Sump Storage	Drawings: V37-342102 V37-343102	General Motors, Allison Division
Fuel tanks: Sump Storage	Drawings: V37-342102 V37-343102	General Motors, Allison Division
Helium tanks	Drawing V37-347102	Electrada, Airite Division

CONFIGURATION B

Configuration B of the AES SPS is the same as described for Configuration A with exceptions as noted in the following paragraph. A system fluid schematic is shown in Figure 8.

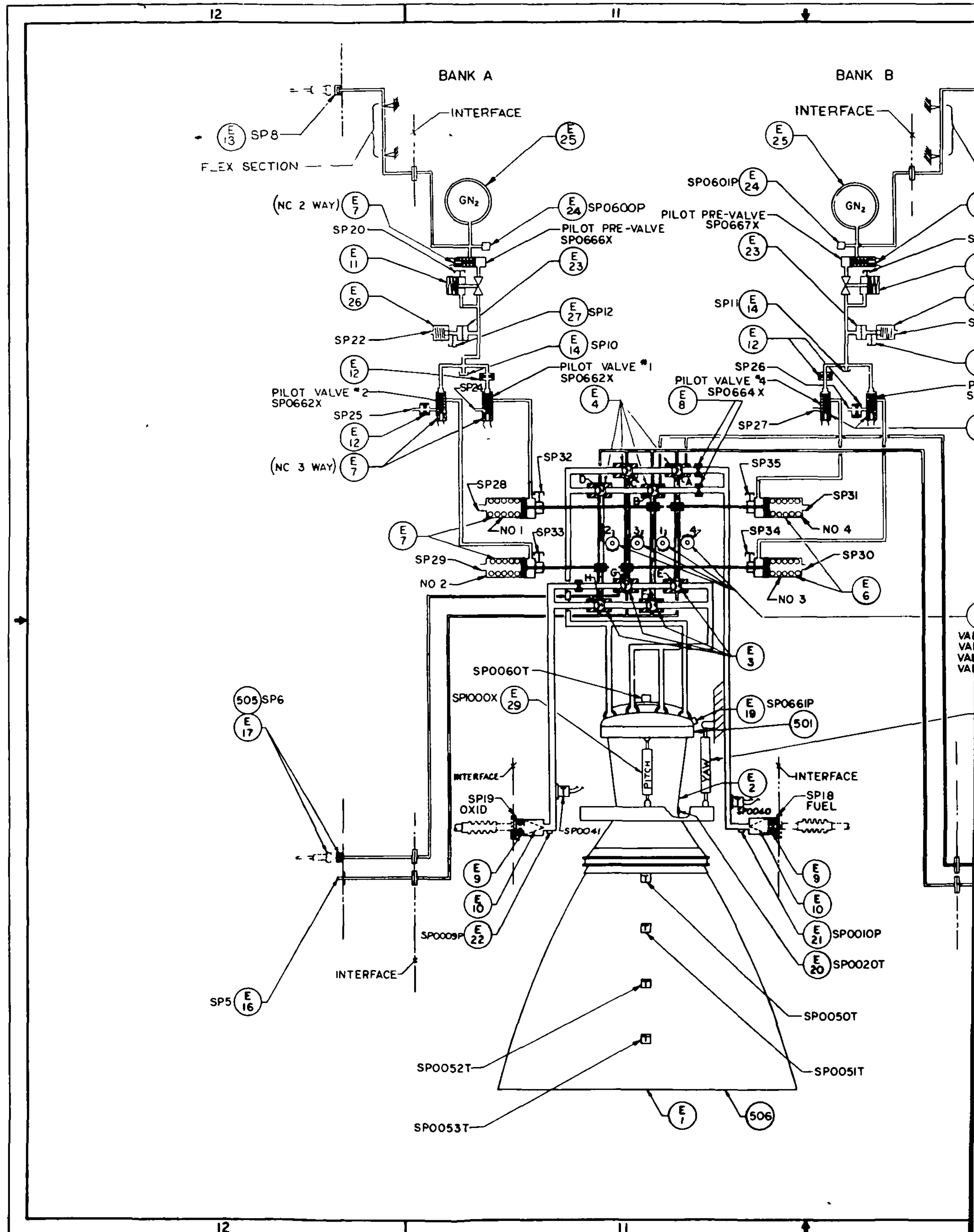
For Configuration B, the oxidizer is contained in a single Apollo Block II sump tank having an internal volume of 160.5 cubic feet and containing 12,948 pounds of usable oxidizer. Normal working pressure in the tank is 175 psig. Configuration of retention reservoir and lines downstream of the tank are identical to Configuration A. The fuel equipment is identical in size and configuration to the oxidizer equipment and contains 8092 pounds of usable fuel.

Due to the deletion of the two propellant storage tanks, Configuration B will require only two of the four tank sensors used on Configuration A. Some revision to the electronic and display portions of the PUGS as well as to the GSE may be required. System interfaces will remain unchanged if possible and changes to the PUGS/spacecraft/GSE are to be minimized.

The use of only two sump tanks requires that the transfer line be eliminated and the one inch helium supply line be adapted to the sump tank door propellant transfer line inlet. The transfer line drain and external disconnect coupling will remain for its original purpose but must be connected to the new helium supply line.

The GSE units used for checkout of the Configuration A SPS PUGS are to be used for Configuration B checkouts with modifications necessitated by the changes from 4 probes to the two-probe configuration. The following GSE will be affected:

1. C14-352 PUGS C/O Unit
2. C14-241 ACE
3. Console displays
4. Documentation referencing the above equipment.



12
11
BANK A
BANK B
INTERFACE
SP8
FLEX SECTION
GN2
(NC 2 WAY)
SP20
SP0600P
PILOT PRE-VALVE
SP0666X
SP0601P
SP0667X
SP12
SP22
SP10
SP11
PILOT VALVE #1
SP0662X
PILOT VALVE #2
SP0662X
PILOT VALVE #4
SP0664X
SP24
SP25
SP26
SP27
(NC 3 WAY)
SP32
SP28
NO 1
SP33
SP29
NO 2
SP35
SP31
NO 4
SP34
SP30
NO 3
SP0060T
SP1000X
SP0661P
PITCH
YAW
505 SP6
SP19
OXID
SP0041
SP18
FUEL
SP0040
SP0009P
SP0010P
SP0020T
INTERFACE
SP5
SP0050T
SP0051T
SP0052T
SP0053T
501
506

Fig 8 (1)

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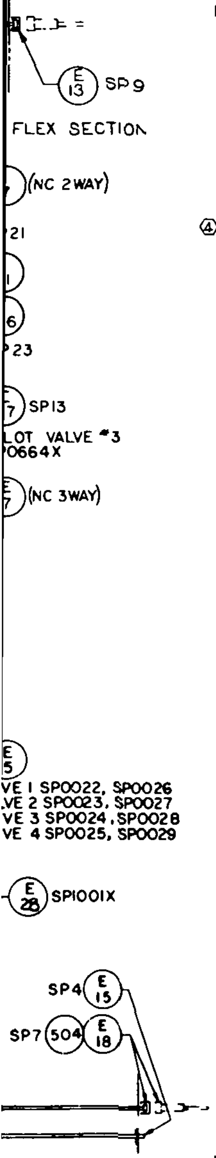
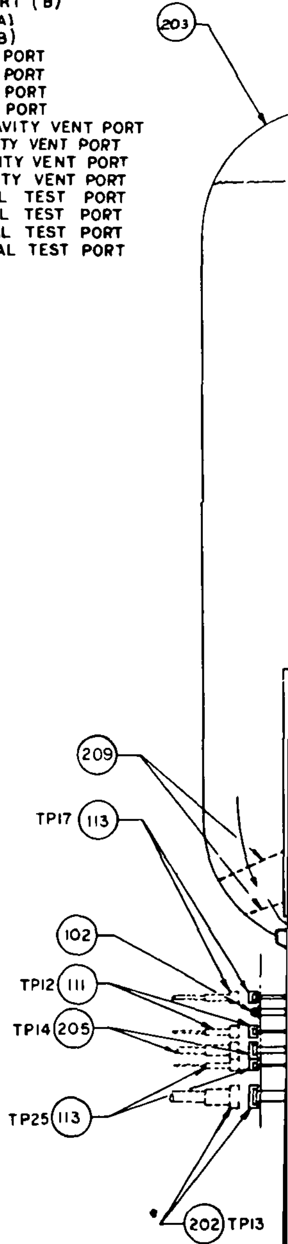


ENGINE ASSEMBLY

E 1 NOZZLE EXTENSION
2 THRUST CHAMBER ASSEMBLY
3 PROPELLANT VALVE-OXIDIZER (NC)
4 PROPELLANT VALVE-FUEL (NC)
5 PROPELLANT VALVE POSITION POT. (REDUNDANT COIL)
6 PROPELLANT VALVE PNEUMATIC ACTUATORS
7 PROPELLANT CONTROL PILOT VALVE-SOLENOID ACTUATED NC
8 LOOP BALANCING ORIFICES
9 ENGINE SYSTEM BALANCING ORIFICES
10 PROPELLANT SCREEN
11 PRESSURE REGULATOR PNEUMATIC (N_2)
12 FLOW CONTROL & TRIM ORIFICES
(4) 13 PNEUMATIC FILL & VENT GN_2 PRESSURIZING, SELF SEAL TYPE, DISCONNECT CAPPED FOR FLIGHT
14 REGULATOR OUTLET TEST PORT SELF SEAL TYPE DISCONNECT CAPPED FOR FLIGHT
15 PROPELLANT VALVE SEAL DRAIN (FUEL) OPEN TYPE QUICK DISCONNECT
16 PROPELLANT VALVE SEAL DRAIN (OXIDIZER) OPEN TYPE QUICK DISCONNECT
17 PROPELLANT INLET MANIFOLD FILL VENT (OXID) SELF SEAL TYPE QUICK DISCONNECT WITH CAP FOR ENGINE OPERATION
18 PROPELLANT INLET MANIFOLD FILL VENT, (FUEL) SELF SEAL TYPE QUICK DISCONNECT WITH CAP FOR ENGINE OPERATION
19 THRUST CHAMBER PRESSURE BOSS
20 THRUST CHAMBER WALL TEMPERATURE SENSOR
21 FUEL INLET PRESSURE BOSS
22 OXIDIZER INLET PRESSURE BOSS
23 BURST DISC
24 PNEUMATIC STORAGE PRESSURE BOSS
25 PNEUMATIC STORAGE TANK (GN_2)
26 RELIEF VALVE
27 RELIEF VALVE TEST PORT SELF SEAL TYPE, DISCONNECT CAPPED FOR FLIGHT
28 GIMBAL ACTUATOR (YAW)
29 GIMBAL ACTUATOR (PITCH)

ENGINE SERVICE POINTS

SP 4-PROP VALVE SEAL DRAIN (FUEL)
SP 5-PROP VALVE SEAL DRAIN (OXID)
SP 6-PROP INLET MANIFOLD FILL VENT (OXID)
SP 7-PROP INLET MANIFOLD FILL VENT (FUEL)
SP 8-PNEUMATIC FILL & VENT (GN_2) BANK A
SP 9-PNEUMATIC FILL & VENT (GN_2) BANK B
SP10-REG OUTLET TEST PORT (A)
SP11-REGULATOR OUTLET TEST PORT (B)
SP12-RELIEF VALVE TEST PORT (A)
SP13-RELIEF VALVE TEST PORT (B)
SP18-FUEL INLET
SP19-OXIDIZER INLET
SP20-REG AMBIENT SENSING POPT (A)
SP21-REG AMBIENT SENSING PORT (B)
SP22-RELIEF VALVE OUTLET (A)
SP23-RELIEF VALVE OUTLET (B)
SP24-PILOT VALVE NO 1 VENT PORT
SP25-PILOT VALVE NO 2 VENT PORT
SP26-PILOT VALVE NO 3 VENT PORT
SP27-PILOT VALVE NO 4 VENT PORT
SP28-ACTUATOR NO 1 SPRING CAVITY VENT PORT
SP29-ACTUATOR NO 2 SPRING CAVITY VENT PORT
SP30-ACTUATOR NO 3 SPRING CAVITY VENT PORT
SP31-ACTUATOR NO 4 SPRING CAVITY VENT PORT
SP32-ACTUATOR NO 1 SHAFT SEAL TEST PORT
SP33-ACTUATOR NO 2 SHAFT SEAL TEST PORT
SP34-ACTUATOR NO 3 SHAFT SEAL TEST PORT
SP35-ACTUATOR NO 4 SHAFT SEAL TEST PORT



VALVE NUMBER	BALL DESIGNATION	DESCRIPTION	LOOP
1	B F	LOWER UPSTREAM FUEL LOWER DOWNSTREAM OX	PRIMARY
2	D H	LOWER DOWNSTREAM FUEL LOWER UPSTREAM OX	
3	C G	UPPER DOWNSTREAM FUEL UPPER UPSTREAM OX	SECONDARY
4	A E	UPPER UPSTREAM FUEL UPPER DOWNSTREAM OX	

SP 12 & 13			264004-4800	LEAR SEIGLER	COUPLING, TEST, REL VALVE (GROUND)	
		2	(1)	AEROJET GENERAL	COUPLING, TEST, REL VALVE (FLIGHT)	
SP 10 & 11			264004-4600	LEAR-SEIGLER	COUPLING, REG OUTLET (GROUND)	
		2	(1)	AEROJET-GENERAL	COUPLING, REG OUTLET (FLIGHT)	
SP 8 & 9	(4)	2	ME144-0023-0081	LEAR-SEIGLER	COUPLING, FILL VENT (GROUND)	MC144-0023
	(4)	2	ME144-0023-0071	LEAR-SEIGLER	COUPLING, FILL VENT (FLIGHT)	MC144-0023
	506	1	ME901-0192-0001	AEROJET-GENERAL	NOZZLE EXTENSION, SPS ENGINE	MC901-0009
SP6	505	1	ME273-0011-0002	J.C CARTER	COUPLING, ENG VENT & DRAIN (GROUND)	MC273-0011
		1	ME273-0011-0001		COUPLING, ENG VENT & DRAIN (FLIGHT)	
SP7	504	1	ME273-0024-0002	J.C CARTER	COUPLING, FILL VENT (GROUND)	MC273-0011
		1	ME273-0024-0001		COUPLING, FILL VENT (FLIGHT)	
	501	1	ME321-0004-0001	AEROJET-GENERAL	ROCKET ENGINE-SERVICE PROPULSION	MC901-0009
	ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC

ROCKET ENGINE SYSTEM

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Fig 8 (2)

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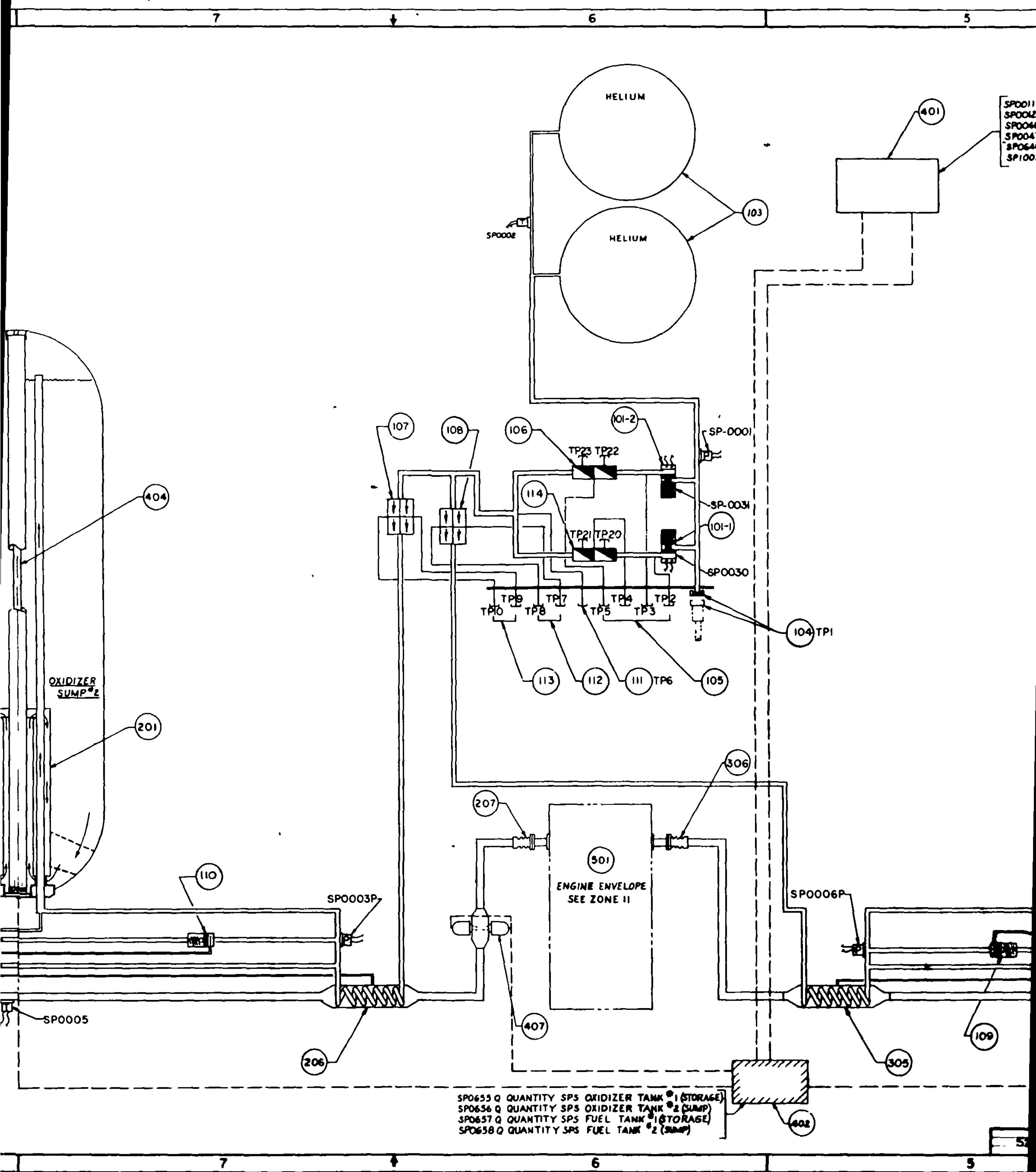

Fig 8
3

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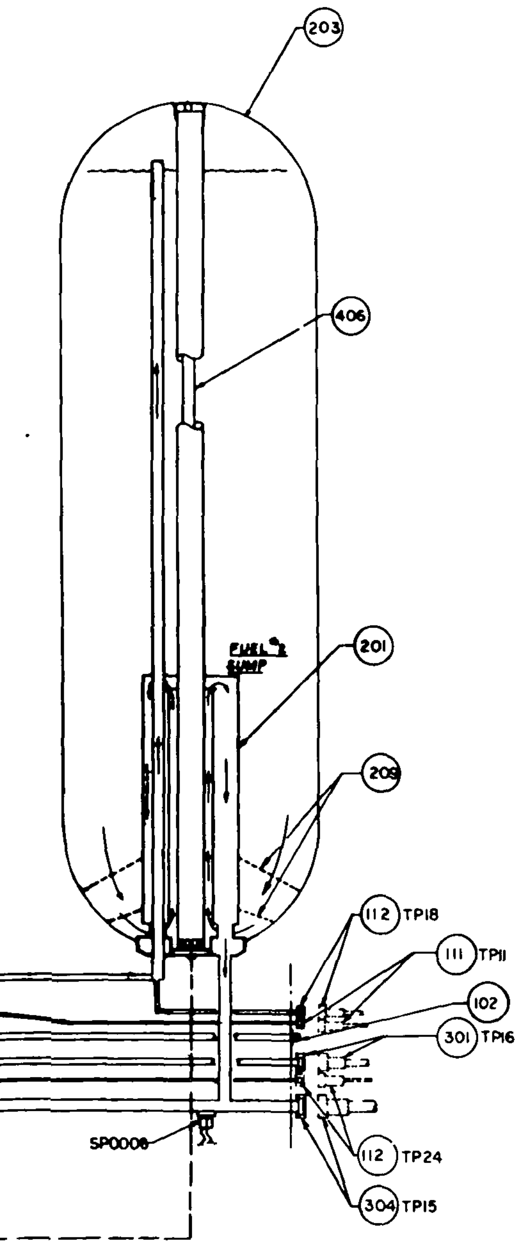
TOTAL QUANTITY OXIDIZER
TOTAL QUANTITY FUEL
PU VALVE INCREASE MONITOR
PU VALVE DECREASE MONITOR
PROPELLANT UNBALANCE (OXIDIZER)
SPS PU SENSOR FAIL



ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061	LEAR SIEGLER	COUPLING TPII GROUND	MC144-0023
	1	ME144-0023-0051		COUPLING TPII FLIGHT	
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	
112	2	ME144-0023-0041	LEAR SIEGLER	COUPLING, RESIDUAL DRAIN (GROUND)	MC144-0023
	2	ME144-0023-0031		COUPLING, RESIDUAL DRAIN (FLIGHT)	
306	1	ME273-0071-0001	AEROQUIP CORP	FLEX JOINT, FUEL LINE	MC273-0071
305	1	ME362-0019-0011	UNITED AIRCRAFT PROD.	HEAT EXCHANGER, He LOW PRESS	MC362-0019
304	1	ME273-0020-0002	J.C. CARTER	COUPLING, FILL & DRAIN (GROUND)	MC273-0012
	1	ME273-0020-0001		COUPLING, FILL & DRAIN (FLIGHT)	
203	1	V37-342102	NAA	TANK, FUEL SUMP	
301	1	ME273-0012-0002	J.C. CARTER	COUPLING, FUEL VENT (GROUND)	MC273-0012
	1	ME273-0012-0001		COUPLING, FUEL VENT (FLIGHT)	

FUEL DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061	LEAR SIEGLER	COUPLING TP12 GROUND	MC144-0023
	1	ME144-0023-0051		COUPLING TP12 FLIGHT	
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
113	2	ME144-0023-0021	LEAR-SIEGLER	COUPLING, OX RESIDUAL DRAIN (GROUND)	MC144-0023
	2	ME144-0023-0011		COUPLING, OX RESIDUAL DRAIN (FLIGHT)	
207	1	ME273-0040-0001	AEROQUIP CORP	FLEX JOINT OXIDIZER LINE	MC273-0040
206	1	ME362-0019-0001	UNITED AIRCRAFT PROD.	HEAT EXCHANGER He LOW PRESS	MC362-0019
205	1	ME273-0022-0002	J.C. CARTER	COUPLING, OX VENT (GROUND)	MC273-0012
	1	ME273-0022-0001		COUPLING, OX VENT (FLIGHT)	
203	1	V37-342102	NAA	TANK, OXIDIZER SUMP	SID65-762
202	1	ME273-0018-0002	J.C. CARTER	COUPLING, OX FILL & DRAIN (GROUND)	MC273-0012
	1	ME273-0018-0001		COUPLING, OX FILL & DRAIN (FLIGHT)	
201	1	V37-470240	NAA	RSVR PROPELLANT RETENTION	

OXIDIZER DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
114	1	ME284-0324-0002	B.H. HADLEY CO	REGULATOR, PRESS He (2 STAGE)	MC284-0324
113	2	ME144-0023-0021	LEAR-SIEGLER	COUPLING, TP 9 & 10 (GROUND)	MC144-0023
	2	ME144-0023-0011		COUPLING, TP 9 & 10 (FLIGHT)	
112	2	ME144-0023-0041	LEAR-SIEGLER	COUPLING TP 7 & 8 (GROUND)	MC144-0023
	2	ME144-0023-0031		COUPLING, TP 7 & 8 (FLIGHT)	
111	1	ME144-0023-0061	LEAR-SIEGLER	COUPLING, TP6 (GROUND)	MC144-0023
	1	ME144-0023-0051		COUPLING, TP6 (FLIGHT)	
110	1	ME284-0330-0001	CALMEC	VALVE, RELIEF OXIDIZER	MC284-0330
109	1	ME284-0330-0011	CALMEC	VALVE, RELIEF FUEL	MC284-0330
108	1	ME284-0327-0002	APCO	VALVE, QUAD CHECK, FUEL	MC284-0327
107	1	ME284-0327-0012	APCO	VALVE, QUAD CHECK, OXIDIZER	MC284-0327
106	1	ME284-0324-0002	B.H. HADLEY	REGULATOR, PRESS He (2-STAGE)	MC284-0324
105	4	ME144-0023-0081	LEAR-SIEGLER	COUPLING, TP 2, 3, 4 & 5 (GROUND)	MC144-0023
	4	ME144-0023-0071		COUPLING, TP 2, 3, 4 & 5 (FLIGHT)	
104	1	ME273-0009-0002	ON-MARK	COUPLING, FILL & DRAIN (GROUND)	MC273-0009
	1	ME273-0009-0007		COUPLING, FILL & DRAIN (FLIGHT)	
103	2	V37-347102	NAA	VESSEL, PRESSURE HELIUM	SID65-762
102	2	V17-460215	NAA	UNION, OVERBOARD RELIEF	———
101	2	ME284-[illegible]-0005	APCO	VALVE, SOLENOID SHUT-OFF	

PRESSURIZATION SYSTEM (HELIUM)

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
407	1	ME284-0219-0001	SIMMONDS PREC. PROD.	VALVE ASSEMBLY P.U	MC901-0008
406	1	ME432-0183-0001		SENSING PROBE FUEL TANK #2 (SUMP)	MC610-0001
404	1	ME432-0181-0001		SENSING PROBE OX TANK #2 (SUMP)	
402	1	ME450-0008-0001		CONTROL UNIT (MODIFIED)	
401	1	ME431-0139-0001		DISPLAY PANEL C/M	MC610-0001

P.U. AND GAUGING SYSTEM

4 4 SUPPLIED BY NAA/SID AT INSTALLATION
3 REF SID65-1527 REPORT
2 TP20, 21, 22 & 23 ARE AND10050-2 PORTS, PER SCD ME284-0324-0002
1 1 AIRBORNE HALF OF DISCONNECTS SUPPLIED BY AEROJET-GENERAL AS PART OF ENG ASSY ME321-0004-0001

NOTES UNLESS OTHERWISE NOTED

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Figure 8. Service Propulsion Fluid Feed System (Two Tanks)

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Fig 8 4

DESIGN CHANGES FROM BLOCK II TO AES

The required changes in SPS design from Apollo Block II to AES are listed in the following.

Changes will be made to the engine and its installation to allow for external servicing of its pneumatic system. The nature and extent of the changes are treated in subsequent sections.

Due to space allocations on the AES SM, the helium panel must be rearranged. However, the relationship of the mainstream components and lines will remain unchanged. Panel structure and test point configuration will be changed but will not require retest.

A feasibility study is planned covering the design of a new or supplementary fluid (propellant and helium) gaging system to allow continuous quantity readout capability for leak detection purposes. The present gaging system function of providing outage control information will be retained and performed by the new system or by the present Block II system depending on the outcome of the study.

Analysis has indicated that backflow propellant leakage through the helium check valve will be a problem during a 45-day AES mission even though such leakage can be tolerated during the Apollo Block II missions. System changes to eliminate this problem are required. Future studies will determine the extent and nature of such changes.

An alternate, reduced tankage configuration will be utilized for AES missions where propellant requirements are less than 53 percent of Block II capacity. Changes to the system will include: (1) removal of both the fuel and oxidizer storage tanks; and (2) rearrangement of propellant and pressurant lines and gaging system as required by (1) above.

POTENTIAL AES CHANGES FROM BLOCK II

Apollo test results for the present butyl rubber propellant tank door seal appear to have acceptable leak rates (maximum 5.5×10^{-4} scc/sec), but the effect of N_2O_4 on the O-ring (shore, volume swell, and elongation) are of such magnitude as to cause doubt for any usage beyond the 16 days of the Apollo test.

Implementation of a new door seal design is presently being accomplished for the Apollo propellant tanks. The design is a Teflon seal backed by a stainless steel "V" ring. This new design appears to be more acceptable for the AES mission, but final resolution in this area will be obtained after Apollo door seal testing and design revision has been accomplished.

The propellant valve shaft seal drain line may require some modification due to propellant leakage with possible freezing and blockage of the line. The propellant trapped in the blocked line can leak into the common gear housing, causing a possible corrosive or explosive condition to impair the integrity of the valve shaft seal. This potential problem is common with the Apollo and AES missions, and data may be obtained from Apollo tests.

After final selection of helium check valve poppet seal material for Block II, the material must be re-evaluated for AES 45-day propellant exposure resistance.

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INTERFACE DEFINITIONS

The following defines the interface requirements for the AES service propulsion system.

ELECTRICAL POWER INTERFACES

SPS power requirements for AES are the same as for the Block II SPS as shown in Table 20. To obtain mission power profiles, the power requirements for any phase of a mission are discussed below. The duration of each phase may be taken from AES mission timelines to establish the power profile and total energy required for any mission. Since the maximum SPS firing duration is 625 seconds and the maximum number of starts is 36, the maximum total energy (not including tests) required for a mission is shown in Table 21.

One minute prior to lift-off, during launch and up until one minute after launch cut-off, the SPS shall be armed. Any subsequent S-IVB firing shall also require arming one minute before and until one minute after. Arming during these phases requires zero ac power and 1418 watts dc power.

During the minute before and the minute after an SPS firing, or during an arming test, the SPS shall be armed and will require idle power for gimbal motors and prevalves. This requires zero ac power and 1418 watts dc power. For the duration of an SPS firing, the power required is 69.5 watts ac and 2180 watts dc (140 watts for the pilot valve is customarily budgeted to SCS but is considered SPS here).

TEMPERATURE LIMITS

AES mission thermal-attitude effects should result in SPS temperature not exceeding the following limits:

Propellant temperature limits in tanks

At lift-off	70+5 F
During flight	40+ to 80 F

Propellant temperature limits in lines (including at disconnect panel)

40 to 140 F

PUGS control unit temperature limits

ON (during SPS firings)	40 to 80 F
OFF (during boost and coast)	20 to 140 F (not to exceed 8 hours)

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Table 20. Power Requirements

Equipment	Power Requirements (watts)
PU Gaging Equipment	62.6 AC
PU Valve	6.9 AC
PU Gaging Equipment	14.0 DC
PU Valve (actuated)	5.6 DC
PU Valve (normal)	2.8 DC
Helium Solenoid Valves	125.0 DC
Gimbal Actuator Motors (idle)	1348.0 DC
Gimbal Actuator Motors (firing)	1825.0 DC
Prevalves	70.2 DC
Pilot Valve	140.0 DC

Back wall of thrust chamber

500 F maximum

Helium panel

+30 to +150 F

Relief valve

+30 to +150 F

Gimbal actuators

Operating	-10 to +200 F
Not operating	-10 to +140 F

Gimbal bearings

-10 to +200 F

Table 21. Maximum Mission Energy Requirements

Period	Duration (Hrs.)	Power (Watts)		Energy (Watts-hours)	
		AC	DC	AC	DC
Launch	.35	0	1418	0	496
Armed (26)	.868	0	1418	0	1230
Firing	.174	69.5	2180	12.1	370
Maximum Total Energy per Mission (Not including ground and in-flight tests)				12.1	2092

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ENGINE ACCESS AND SERVICING

The AES SM has limited space available and direct engine access on the launch stack as is provided in Block II may not be retained. However, access is provided by moving a hydrogen tank in Sector IV (see report SID 65-1529, Spacecraft Design Summary). This access permits engine electrical and pneumatic package functional checks, inspection and maintenance before the count down phase but does not permit engine nitrogen servicing since gas servicing occurs subsequent to hydrogen tank chill down. For this reason, the engine nitrogen servicing connections are to be relocated to an external point on the AES SPS. Since the engine and its pneumatic package are gimballed, a design utilizing two flexible 0.25-inch fill lines from the engine to two fill ports on the SM skin line are required. It is anticipated that this design will consist of free lengths of 0.25-inch tubing long enough to accommodate engine gimbal motion without excessive flexural stress.

Other methods for engine gas servicing that do not require flexible high-pressure lines were evaluated and discarded. These involved access through the SLA, and one of the following:

1. Fill connections supported by the engine and located on a shielded panel below the nozzle aft bulkhead closeout boot.

2. Fill connections located on the engine above the boot with **provision for arm access through the boot.**

3. Fill connections located on the engine with personnel access through the aft bulkhead.

The specific engine components which will require access prior to the countdown phase (for functional checks, inspection, and maintenance) by moving the hydrogen tank are listed as follows:

24 electrical connectors
2 instrumentation connectors
2 valve actuation packages (each replaceable as a unit)

Each valve actuation package consists of:

1 nitrogen vessel
1 pressure regulator
1 rupture disc assembly
1 relief valve
2 solenoid pilot valves
2 test points

EVALUATION OF AES GIMBAL LIMITS

The AES missions utilize the standard Apollo Block II SPS without changes. However, in support of AES studies, pallet weight limitation of 1700 pounds was identified. Since NASA is planning for heavier pallets, the following explains the basis for the pallet weight limitation and **describes what is involved in raising the limitation.**

The pallet weight limitation is imposed by the Block II thrust vector control (TVC) design limit on c.g. excursion of:

Pitch (X-Z plane)	-1/4 to +4 1/4 degrees
Yaw (X-Y plane)	-1/4 to +3 1/4 degrees

These TVC characteristics were established to match the c.g. excursion envelope of the several possible Block II vehicle configurations (with and without LEM, with and without pallet weight). The pallet weight used was 1500 pounds in conformance with Apollo CCA 317.

When the SPS mixture ratio was changed from 2.0 to 1.6, the propellant induced c.g. excursion was reduced substantially because the propellant tank arrangement became much more mass-symmetric. Accordingly, the TVC angular limits were reduced in a corresponding amount. Advantages accrued to the TVC system through this reduction were included in the arguments presented in favor of the mixture ratio change. However, the Block II requirement to accommodate a 1500 pound pallet was retained intact.

Although gimbal angle capability was reduced in this change, nothing in the way of TVC capability was given up. One of the primary requirements for TVC is to maintain stable vehicle attitudes during SPS firing under all applicable c.g. excursion and/or initial conditions. The angular limits of gimbal swing are merely secondary requirements that stem from this primary TVC requirement. There is no requirement to have a TVC system with gimbal angle capability in excess of the over travel requirements. In fact, to design for excess gimbal swing would ordinarily impose unnecessary and adverse requirements on TVC, particularly with respect to large signal stability.

The c.g. excursion limits cited above were obtained from the Block II gimbal limits and the fact that 2.25 degrees over travel past the c.g. is required for control throughout the SPS burn profile. Previous gimbal limits are compared with current limits in Table 22.

Table 22. SPS Gimbal Limits

Configuration	Block I	Original Block II	Current Block II
LEM	No	Yes & No	Yes & No
Mixture Ratio	2.0	2.0	1.6
Pallet Weight, pounds	None	0 to 1500	0 to 1500
Pitch travel, degrees	± 6	± 5	± 4.5
Yaw travel, degrees	± 7	± 6.5	± 4.5
Pitch Null Offset, degrees	0	0	+ 2
Yaw Null Offset, degrees	+ 4	+ 3	+ 1

To raise the pallet weight limitation requires an increase in plus yaw TVC c.g. tracking capability of approximately one and two degrees for 3200 and 5000 pound limitations, respectively. To make such a change first requires a redefinition of the pallet weight range to be accommodated, the coordinate envelope occupied by its centroid, and a description of its mass distribution or inertial properties envelope. Secondly, a design change investigation is required. With respect to the Block II service propulsion subsystem, it is not now believed that the gimbal angle capability will be a limiting factor. The gimbal actuator travel is adequate to permit a two degree increase and/or the null position could be moved. A study would be required to determine the best method. Changing actuator travel requires movement of its snubbing stops, a factory operation.

With respect to the Block II stabilization and control subsystem, its design is now frozen and a stability analysis is required whether the TVC limits are changed by moving the gimbal null point or by increasing the travel. This analysis would determine if the change has any impact on the actuator characteristics such as rate and acceleration, as well as what, if any, changes to the SCS design are required.

VI. SUMMARY

The service propulsion subsystem to be used for AES is essentially the same as the Apollo Block II SPS. The few changes required are:

1. The addition of a zero gravity leak detection helium and propellant gaging system to improve crew safety (Block II gaging system is inoperable during periods of coast).

2. Provision for external nitrogen servicing for the engine to permit more efficient space utilization in the SM.

3. A change in the helium pressurization system to eliminate backflow leakage of propellant to the regulator.

4. The use of an alternate, reduced propellant tankage configuration to save weight on off-loaded flights. This configuration eliminates the propellant storage tanks and uses only the sump tanks.

5. A minor rearrangement of non-flow components on the helium panel to permit changes in the panel configuration required by AES SM space allocations.

In addition, potential changes for AES exist in areas still under design revision and development for Apollo Block II, depending on the design selected by Apollo. These include the oxidizer tank door seal (a change in material is required), the engine valve shaft seal drain (a heater may be required), and the oxidizer and fuel check valves (a change in material may be required).

These changes are the result of preliminary evaluation of AES requirements. AES missions have been designed to fall within the maximum propulsion capability of the Block II SPS, and no change in SPS primary performance characteristics is required. However, the AES mission duration of 45 days, when compared to the Apollo Block II duration of 14 days, requires review of materials life, leakage effects, and coast-dependent failure effects on reliability. In addition, the AES SM incorporates minor changes from the Apollo Block II SM, and the SPS installation requirements required review.

Materials life was evaluated for 45 day exposure to propellants, radiation, and high vacuum. None of these environments, when extended from 14 to 45 days, changes the choice of Block II SPS materials except in areas where new materials are being selected for Block II.

The AES meteoroid environment requires protection of propellant tanks and lines and an investigation of engine nozzle extension failure modes with meteoroid penetration.

The increase in mission duration has several implications on SPS fluid leakage. Crew safety will be improved by the addition of a zero gravity leak detection system to provide information for timely abort action. For improvement of mission success, only leak prevention measures are useful. This area requires further study of design aspects of seals and seal redundancy, plus improvements in acceptance test technique for predicting production article leak rates over long missions. The Block II leak rate standards for acceptance test were reviewed and found to be adequate except for backflow leakage to the regulator.

For AES missions requiring less than a full propellant load, a reduced propellant tankage trade study was conducted. This resulted in a configuration (designated as SPS baseline configuration B) to be used as an alternate to the standard Block II propellant tank configuration which is required for some AES missions. The final selection, consisting of two unmodified Block II sump tanks, was chosen primarily for its simplicity. Other configurations of less capacity would cover the presently anticipated off-loaded mission propellant requirements and with greater weight savings. However, the increased complexity of their changes required much greater cost per pound of weight saved. The recommended configuration is applicable to those AES missions requiring less than 21,000 pounds of SPS propellant. It uses only the two sump tanks, the two SPS propellant storage tanks being omitted. Both of the Block II helium vessels are retained. However, it is recommended that the definition of baseline configuration B be changed to use only one Block II helium vessel. This will result in a 10 percent decay in chamber pressure and thrust over the last 30 seconds of flights requiring 21,000 pounds of propellant.

The recommended configuration has a capacity of 53 percent of the maximum usable propellant load, saves 586 pounds (the saving is 864 pounds if the change to one helium vessel is made), and has the lowest cost per pound of burned weight saved.

APPENDIX A

SPS BLOCK II BASELINE DESIGN DEFINITION

1.0 SERVICE PROPULSION SUBSYSTEM DESCRIPTION

1.1 SERVICE PROPULSION SUBSYSTEM AES MISSION FUNCTIONAL REQUIREMENTS

The SPS is designed to provide the impulse necessary to change the linear momentum of the spacecraft for the normal and contingent AES mission maneuvers listed below.

1. Abort after LES jettison (abort to orbit or abort to entry)
2. Earth orbital plane changes
3. De-orbit from earth orbit
4. Translunar course corrections
5. Translunar aborts
6. Lunar orbit insertion
7. Lunar orbital plane changes
8. Lunar orbit abort
9. Transearth injection
10. Transearth course corrections

The service propulsion subsystem will provide the spacecraft major velocity increments for the above maneuvers by providing an approximately constant thrust in the +X direction through the center of gravity of the spacecraft.

1.2 SERVICE PROPULSION SUBSYSTEM PERFORMANCE REQUIREMENTS

To perform the SPS mission objectives listed in paragraph 1.1, overall requirements of the subsystem are as listed in Table 23.

Table 23. SPS Performance Requirements

Item	Baseline Value
Dry Weight (including tanks)	2,830 lbs
Usable Propellant (maximum)	39,700 lbs
Thrust (nominal)	20,000 lbs
Instantaneous Specific Impulse	311.7 sec (-3 sigma at end of 750 sec of firing)
Number of starts	36
Minimum impulse per start	5000 lb sec
Engine minimum run-to-run tolerance	± 200 lb sec

1.3 SERVICE PROPULSION SUBSYSTEM DESIGN DESCRIPTION

The SPS provides a nearly constant propulsive force through combustion and expansion of propellants supplied under pressure to a combustion chamber. The engine and its thrust chamber are gimbal-mounted with actuators for thrust directional control. The oxidizer is nitrogen tetroxide, and the fuel is a mixture of unsymmetrical dimethylhydrazine and hydrazine. The propellants are forced to the engine from their tanks by gas displacement at a controlled pressure, using helium from high-pressure storage vessels. Propellant gaging is provided to indicate fuel and oxidizer quantity and ratio imbalance. Displays for monitoring critical SPS parameters and controls for operating certain SPS functions are contained within the command module.

1.3.1 Service Propulsion Subsystem Equipment

The SPS shall consist of the following equipment:

1. Helium pressurization equipment
2. Propellant supply and distribution equipment
3. Propellant utilization and gaging equipment
4. Rocket engine assembly
5. Instrumentation

6. Displays and controls

7. Servicing provisions.

The SPS shall be arranged as shown schematically in Figure 9. Refer to Table 24 for component nomenclature and quantity.

1.3.1.1 Helium Pressurization Equipment

The pressurization equipment shall consist of a high-pressure helium supply, contained within two spherical helium tanks of internal volumes of 19.6 cubic feet each, and associated pressure regulators, isolation valves, check valves, and pressure relief valves. The helium tanks shall have a limit pressure of 3685 pounds per square inch (psi), a proof pressure of 4914 psi, and a design fill pressure of 3585 psia. The tanks shall be connected to a common helium distribution line. All helium lines shall have a minimum proof pressure of 8800 psi and a burst pressure of 17,600 psi. Two solenoid-operated poppet valves, installed in parallel downstream of the helium storage vessels, shall provide for isolating the helium supply when the propulsion system is inoperative. Pressure regulation shall be accomplished by two pressure regulator assemblies, installed in parallel, one assembly downstream of each of the helium supply isolation valves. Each regulator assembly shall incorporate a primary and a secondary regulator in series. A series parallel installation of check valve assemblies, downstream of the helium pressure regulators, shall prevent backflow of propellants into the pressurization subsystem. One oxidizer and one fuel pressure relief valve assembly, each consisting of a relief valve, a burst diaphragm and a filter screen, shall be provided for the respective propellant tank systems. The pressure relief valves shall limit the pressures in the propellant tankage assemblies to 225 psi maximum.

1.3.1.1.1 Pressurization Assembly. The pressurization assembly shall consist of the following components:

1. Helium pressure-regulating assemblies

2. Helium shutoff valves

3. Fill and drain coupling assemblies

4. Testpoint coupling assemblies

5. Helium check valve assemblies

6. Helium pressure relief valve assemblies

7. Helium tanks.

1.3.1.1.1.1 Helium Pressure-Regulating Assemblies. Each regulator assembly shall incorporate a filter (40 microns or less) and a primary and secondary regulator in series. The primary regulator shall reduce the high

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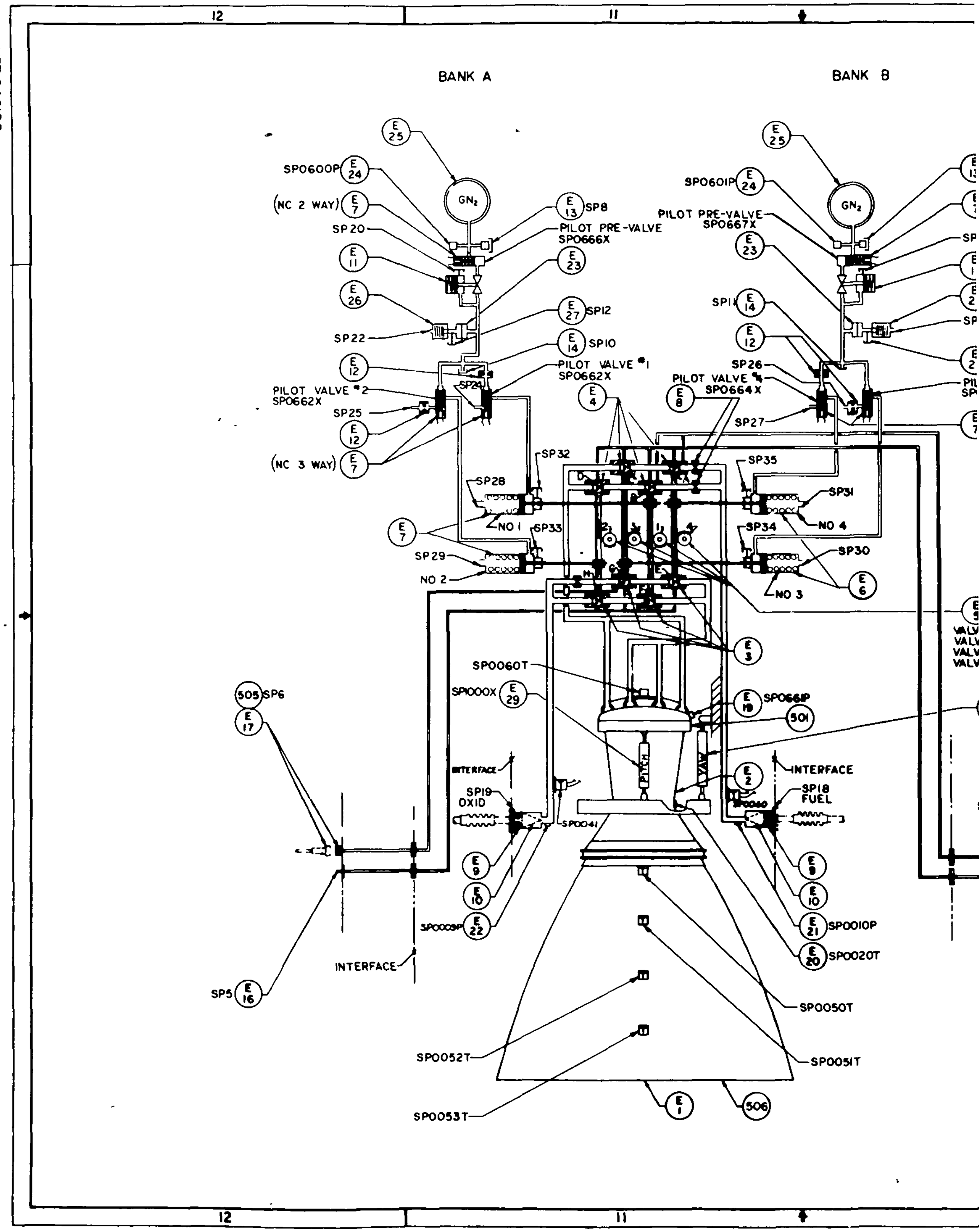

Fig 8

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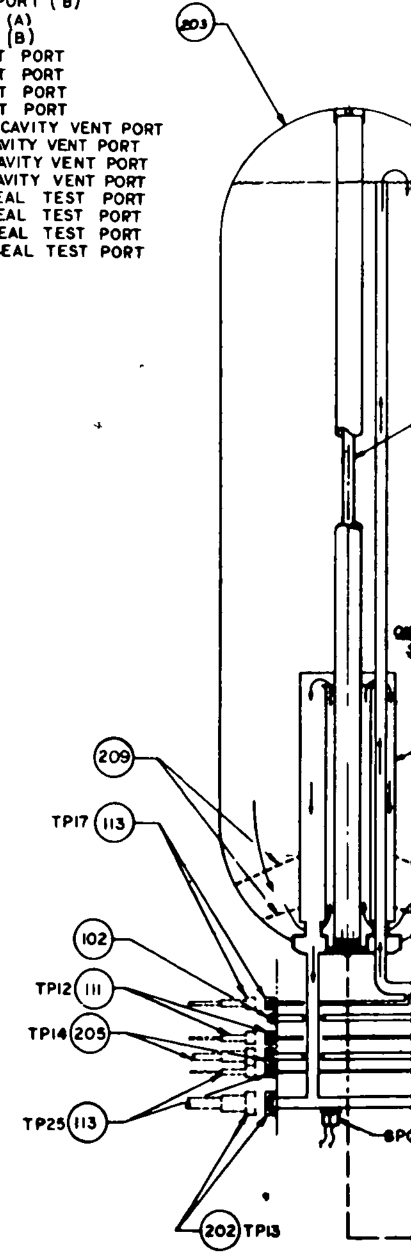
ENGINE ASSEMBLY

E1 NOZZLE EXTENSION
2 THRUST CHAMBER ASSEMBLY
3 PROPELLANT VALVE-OXIDIZER (NC)
4 PROPELLANT VALVE-FUEL (NC)
5 PROPELLANT VALVE POSITION POT (REDUNDANT COIL)
6 PROPELLANT VALVE PNEUMATIC ACTUATORS
7 PROPELLANT CONTROL PILOT VALVE-SOLENOID ACTUATED NC
8 LOOP BALANCING ORIFICES
9 ENGINE SYSTEM BALANCING ORIFICES
10 PROPELLANT SCREEN
11 PRESSURE REGULATOR, PNEUMATIC (N_2)
12 FLOW CONTROL ℄ TRIM ORIFICES
13 PNEUMATIC FILL ℄ VENT GN_2 PRESSURIZING SELF SEAL TYPE, DISCONNECT CAPPED FOR FLIGHT
14 REGULATOR OUTLET TEST PORT SELF SEAL TYPE DISCONNECT CAPPED FOR FLIGHT
15 PROPELLANT VALVE SEAL DRAIN (FUEL) OPEN TYPE QUICK DISCONNECT
16 PROPELLANT VALVE SEAL DRAIN (OXIDIZER) OPEN TYPE QUICK DISCONNECT
17 PROPELLANT INLET MANIFOLD FILL VENT (OXID) SELF SEAL TYPE QUICK DISCONNECT WITH CAP FOR ENGINE OPERATION
18 PROPELLANT INLET MANIFOLD FILL VENT, (FUEL) SELF SEAL TYPE QUICK DISCONNECT WITH CAP FOR ENGINE OPERATION
19 THRUST CHAMBER PRESSURE BOSS
20 THRUST CHAMBER WALL TEMPERATURE SENSOR
21 FUEL INLET PRESSURE BOSS
22 OXIDIZER INLET PRESSURE BOSS
23 BURST DISC
24 PNEUMATIC STORAGE PRESSURE BOSS
25 PNEUMATIC STORAGE TANK (GN_2)
26 RELIEF VALVE
27 RELIEF VALVE TEST PORT SELF SEAL TYPE, DISCONNECT CAPPED FOR FLIGHT
28 GIMBAL ACTUATOR (YAW)
29 GIMBAL ACTUATOR (PITCH)

ENGINE SERVICE POINTS

SP 4-PROP VALVE SEAL DRAIN (FUEL)
SP 5-PROP VALVE SEAL DRAIN (OXID)
SP 6-PROP INLET MANIFOLD FILL VENT (OXID)
SP 7-PROP INLET MANIFOLD FILL VENT (FUEL)
SP 8-PNEUMATIC FILL ℄ VENT (GN_2) BANK A
SP 9-PNEUMATIC FILL ℄ VENT (GN_2) BANK B
SP10-REG OUTLET TEST PORT (A)
SP11-REGULATOR OUTLET TEST PORT (B)
SP12-RELIEF VALVE TEST PORT (A)
SP13-RELIEF VALVE TEST PORT (B)
SP18-FUEL INLET
SP19-OXIDIZER INLET
SP20-REG AMBIENT SENSING POPT (A)
SP21-REG AMBIENT SENSING PORT (B)
SP22-RELIEF VALVE OUTLET (A)
SP23-RELIEF VALVE OUTLET (B)
SP24-PILOT VALVE NO 1 VENT PORT
SP25-PILOT VALVE NO 2 VENT PORT
SP26-PILOT VALVE NO 3 VENT PORT
SP27-PILOT VALVE NO 4 VENT PORT
SP28-ACTUATOR NO 1 SPRING CAVITY VENT PORT
SP29-ACTUATOR NO 2 SPRING CAVITY VENT PORT
SP30-ACTUATOR NO 3 SPRING CAVITY VENT PORT
SP31-ACTUATOR NO 4 SPRING CAVITY VENT PORT
SP32-ACTUATOR NO 1 SHAFT SEAL TEST PORT
SP33-ACTUATOR NO 2 SHAFT SEAL TEST PORT
SP34-ACTUATOR NO 3 SHAFT SEAL TEST PORT
SP35-ACTUATOR NO 4 SHAFT SEAL TEST PORT



VALVE NUMBER	BALL DESIGNATION	DESCRIPTION	LOOP
1	B	LOWER UPSTREAM FUEL	PRIMARY
	F	LOWER DOWNSTREAM OX	
2	D	LOWER DOWNSTREAM FUEL	
	H	LOWER UPSTREAM OX	
3	C	UPPER DOWNSTREAM FUEL	SECONDARY
	G	UPPER UPSTREAM OX	
4	A	UPPER UPSTREAM FUEL	
	E	UPPER DOWNSTREAM OX	

	ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
SP			264004-4800	LEAR-SEIGLER	COUPLING TEST REL VALVE (GROUND)	
12 ℄ 13		2	(1)	AEROJET-GENERAL	COUPLING TEST REL VALVE (FLIGHT)	
SP			264004-4600	LEAR-SEIGLER	COUPLING, REG OUTLET (GROUND)	
10 ℄ 11		2	(1)	AEROJET-GENERAL	COUPLING, REG OUTLET (FLIGHT)	
SP 8			264004-5200	LEAR-SEIGLER	COUPLING, FILL VENT (GROUND)	
℄ 9		2	(1)	AEROJET-GENERAL	COUPLING, FILL VENT (FLIGHT)	
	506	1	ME901-0192-0001	AEROJET-GENERAL	NOZZLE EXTENSION, SPS ENGINE	MC901-0009
SP6	505	1	ME273-0011-0002	J.C CARTER	COUPLING, ENG VENT (DRAIN (GROUND)	MC273-0011
		1	ME273-0011-0001		COUPLING, ENG VENT (DRAIN (FLIGHT)	
SP7	504	1	ME273-0024-0002	J.C CARTER	COUPLING, FILL VENT (GROUND)	MC273-0011
		1	ME273-0024-0001		COUPLING, FILL VENT (FLIGHT)	
	501	1	ME321 0004-0001	AEROJET-GENERAL	ROCKET ENGINE-SERVICE PROPULSION	MC901-0009
	ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC

ROCKET ENGINE SYSTEM

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Fig 9 (2)

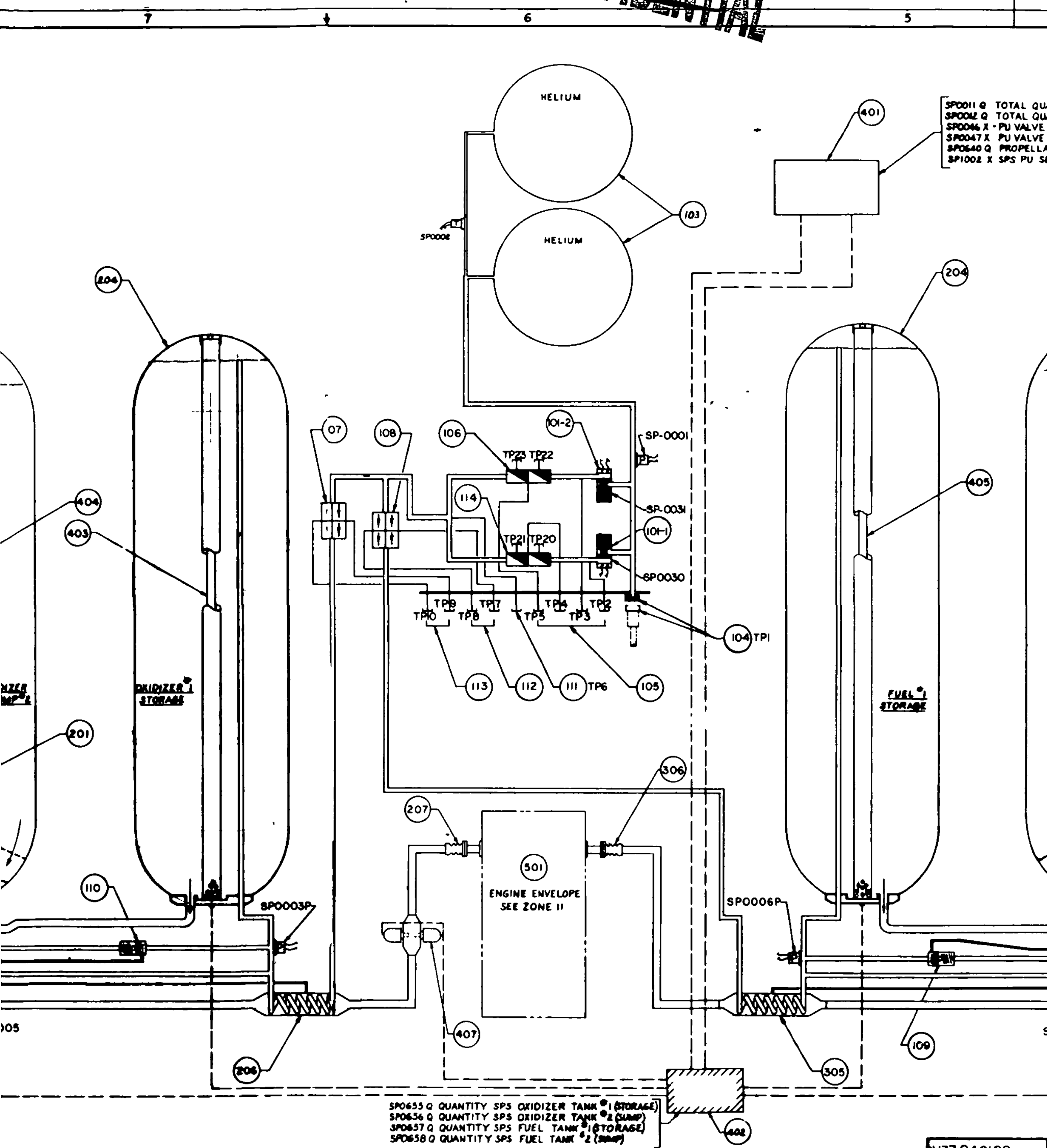
7
6
5
HELIUM
HELIUM
103
SP0002
401
SP0011 Q TOTAL QUAN
SP0012 Q TOTAL QUAN
SP0046 X - PU VALVE I
SP0047 X PU VALVE D
SP0640 Q PROPELLAN
SP1002 X SPS PU SEN
204
204
404
403
201
405
OXIDIZER #1 STORAGE
FUEL #1 STORAGE
07
108
106
101-2
SP-0001
TP23 TP22
114
SP-0031
TP21 TP20
101-1
SP0030
TP19 TP17 TP14 TP12
TP10 TP8 TP5 TP3
104 TP1
113
112
111 TP6
105
306
207
501
ENGINE ENVELOPE SEE ZONE 11
110
SP0003P
SP0006P
206
407
305
109
402
SP0655 Q QUANTITY SPS OXIDIZER TANK #1 (STORAGE)
SP0656 Q QUANTITY SPS OXIDIZER TANK #2 (SUMP)
SP0657 Q QUANTITY SPS FUEL TANK #1 (STORAGE)
SP0658 Q QUANTITY SPS FUEL TANK #2 (SUMP)
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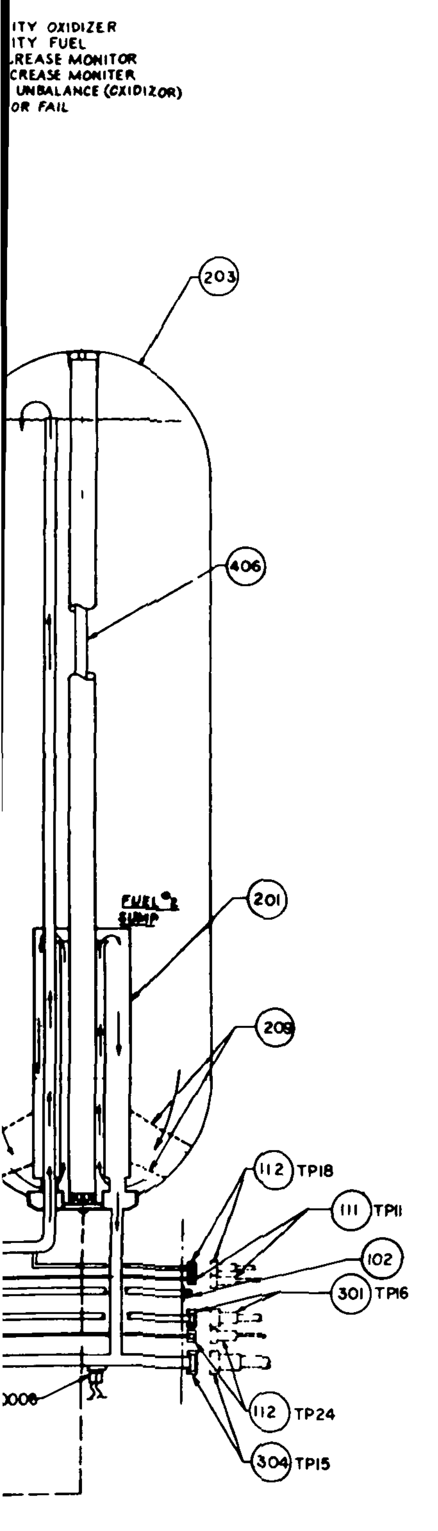
Fig 9 (3)



ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061	LEAR SIEGLER	COUPLING TP11 GROUND	MC144-0023
	1	ME144-0023-0051		COUPLING TP11 FLIGHT	
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	
112	2	ME144-0023-0041	LEAR SIEGLER	COUPLING, RESIDUAL DRAIN (GROUND)	MC144-0023
	2	ME144-0023-0031		COUPLING, RESIDUAL DRAIN (FLIGHT)	
306	1	ME273-0071-0001	AEROQUIP CORP	FLEX JOINT, FUEL LINE	MC273-0071
305	1	ME362-0019-0011	UNITED AIRCRAFT PROD.	HEAT EXCHANGER, He LOW PRESS	MC362-0019
304	1	ME273-0020-0002	J.C. CARTER	COUPLING, FILL & DRAIN (GROUND)	MC273-0012
	1	ME273-0020-0001		COUPLING, FILL & DRAIN (FLIGHT)	
203	1	V37-342102	NAA	TANK, FUEL SUMP	
204	1	V37-343102	NAA	TANK, FUEL STORAGE	
301	1	ME273-0012-0002	J.C. CARTER	COUPLING, FUEL VENT (GROUND)	MC273-0012
	1	ME273-0012-0001		COUPLING, FUEL VENT (FLIGHT)	

FUEL DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061	LEAR SIEGLER	COUPLING TP12 GROUND	MC144-0023
	1	ME144-0023-0051		COUPLING TP12 FLIGHT	
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
113	2	ME144-0023-0021	LEAR-SIEGLER	COUPLING, OX RESIDUAL DRAIN (GROUND)	MC144-0023
	2	ME144-0023-0011		COUPLING, OX RESIDUAL DRAIN (FLIGHT)	
207	1	ME273-0040-0001	AEROQUIP CORP	FLEX JOINT, OXIDIZER LINE	MC273-0040
206	1	ME362-0019-0001	UNITED AIRCRAFT PROD.	HEAT EXCHANGER, He LOW PRESS	MC362-0019
205	1	ME273-0022-0002	J.C. CARTER	COUPLING, OX VENT (GROUND)	MC273-0012
	1	ME273-0022-0001		COUPLING, OX VENT (FLIGHT)	
204	1	V37-343102	NAA	TANK, OXIDIZER STORAGE	
203	1	V37-342102	NAA	TANK, OXIDIZER SUMP	SID65-762
202	1	ME273-0018-0002	J.C. CARTER	COUPLING, OX FILL & DRAIN (GROUND)	MC273-0012
	1	ME273-0018-0001		COUPLING, OX FILL & DRAIN (FLIGHT)	
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	

OXIDIZER DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
114	1	ME284-0324-0002	B.H. HADLEY CO	REGULATOR, PRESS. He (2 STAGE)	MC284-0324
113	2	ME144-0023-0021	LEAR-SIEGLER	COUPLING, TP 9 & 10 (GROUND)	MC144-0023
	2	ME144-0023-0011		COUPLING, TP 9 & 10 (FLIGHT)	
112	2	ME144-0023-0041	LEAR-SIEGLER	COUPLING, TP 7 & 8 (GROUND)	MC144-0023
	2	ME144-0023-0031		COUPLING, TP 7 & 8 (FLIGHT)	
111	1	ME144-0023-0061	LEAR-SIEGLER	COUPLING, TP6 (GROUND)	MC144-0023
	1	ME144-0023-0051		COUPLING, TP6 (FLIGHT)	
110	1	[illegible]	CALMEC	VALVE, RELIEF OXIDIZER	[illegible]
109	1	ME284-0330-0011	CALMEC	VALVE, RELIEF FUEL	[illegible]
108	1	ME284-0327-0002	APCO	VALVE, QUAD CHECK, FUEL	[illegible]
107	1	[illegible]	APCO	VALVE, QUAD CHECK, OXIDIZER	[illegible]
106	1	[illegible]	B.H. HADLEY	REGULATOR, PRESS He (2 STAGE)	[illegible]
105	4	ME144-0023-0081	LEAR-SIEGLER	COUPLING, TP 2, 3, 4 & 5 (GROUND)	MC144-0023
	4	ME144-0023-0071		COUPLING, TP 2, 3, 4 & 5 (FLIGHT)	
104	1	ME273-0009-0002	ON-MARK	COUPLING, FILL & DRAIN (GROUND)	MC273-0009
	1	ME273-0009-0007		COUPLING, FILL & DRAIN (FLIGHT)	
103	2	V37-347102	NAA	VESSEL, PRESSURE, HELIUM	SID65-762
102	2	V17-440215	NAA	UNION, OVERBOARD RELIEF	——
101	2	[illegible]	APCO	VALVE, SOLENOID SHUT-OFF	

PRESSURIZATION SYSTEM (HELIUM)

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
407	1	ME284-0019-0001	SIMMONDS PREC. PROD.	VALVE ASSEMBLY P.U.	MC901-0008
406	1	[illegible]		SENSING PROBE FUEL TANK #2 (SUMP)	MC610-0001
405	1	[illegible]		SENSING PROBE FUEL TANK #1 (STOR)	
404	1	ME432-0181-0001		SENSING PROBE OX TANK #2 (SUMP)	
403	1	ME432-0180-0001		SENSING PROBE OX TANK #1 (STOR)	
402	1	[illegible]		CONTROL UNIT	
401	1	[illegible]		DISPLAY PANEL C/M	MC610-0001

P.U. AND GAUGING SYSTEM

3 REF INSTL DWGS V37-460201 HELIUM V37-470201 OXIDIZER V37-480201 (FUEL) V37-470202 (GAUGING) V37-410001 (ENGINE)
2 TP20, 21, 22 & 23 ARE AND10050-2 PORTS, PER SCD ME284-0324-0002
(1) 1 AIRBORNE HALF OF DISCONNECTS SUPPLIED BY AEROJET-GENERAL AS PART OF ENG ASSY

NOTES UNLESS OTHERWISE NOTED

V37-940109

Figure 9. Service Propulsion System, Fluid

Fig 9 (4)

Table 24. SPS Component Nomenclature and Quantity

Nomenclature	Quantity
Engine (AJ10-137):	
Chamber	1
Injector, baffled	1
Valve, Bipropellant	1
Pneumatic Valve Actuation Assembly	1
Nozzle Extension	1
Actuator, Yaw	1
Actuator, Pitch	1
Mount, thrust	1
Wiring Harness	2
Propellant Line	2
Fill Vent Line	2
Seal Drain Line	2
Pressurization Subsystem:	
Test Point Coupling	11
Valve, Relief Oxidizer	1
Valve, Relief Fuel	1
Valve, Quad Check, Fuel	1
Valve, Quad Check, Oxidizer	1
Regulator, Pressure Helium	2
Coupling, Fill and Drain	1
Vessel, Pressure	2
Union, Overboard Relief	2
Valve, Solenoid Shutoff	2

Table 24. SPS Component Nomenclature and Quantity (Cont)

Nomenclature	Quantity
Oxidizer Distribution Subsystem:	
Coupling, Oxidizer Residual Drain	1
Flexible Joint, Oxidizer Line	1
Heat Exchange, Helium Low Pressure	1
Coupling, Oxidizer Vent	1
Tank, Oxidizer Storage	1
Tank, Oxidizer Sump	1
Coupling, Oxidizer Fill and Drain	1
Reservoir, Propellant	1
Fuel Distribution Subsystem:	
Coupling, Residual Drain	1
Flexible Joint, Fuel Line	1
Heat Exchanger, Helium Low Pressure	1
Coupling, Fuel Vent	1
Tank, Fuel Sump	1
Tank, Fuel Storage	1
Coupling, Fill and Drain	1
Reservoir, Propellant	1
Gaging Subsystem:	
Valve Assembly	1
Sensing Probe, Fuel Sump	1
Sensing Probe, Fuel Storage	1
Sensing Probe, Oxidizer Sump	1
Sensing Probe, Oxidizer Storage	1

upstream pressure to the normal downstream operating pressure of 183.5 psi. The secondary regulator in the series shall remain open as long as the primary regulator functions properly, when operating it shall regulate to 188.5 psi nominal. Regulator lockup pressure shall be 195 psig maximum. The regulators are coded 106 and 114 in Figure 9. Design requirements are presented in Table 25.

1.3.1.1.1.2 Helium Shutoff Valves. The helium shutoff valves shall initiate and interrupt the flow of helium to the propellant tanks in conjunction with the opening and closing of the engine propellant valves, and provide isolation of the helium supply during ground checkout of the system. Two such valves shall be installed in parallel downstream of the helium storage vessels as shown schematically in Figure 9, coded 101. Design requirements for the valve are presented in Table 25.

1.3.1.1.1.3 Fill and Drain Coupling Assemblies. The coupling assemblies shall consist of two half-couplings: A male half-coupling (the airborne component), and a female half-coupling (the ground support component). The coupling assemblies shall permit helium flow from the ground support equipment tank(s) to pressurize the helium tanks in the spacecraft from zero psig to a maximum operating pressure of 3685 psig, and from the spacecraft tanks to the ground support equipment tank(s) for depressurization of the spacecraft tank. The airborne half-coupling, with and without its dust cap engaged, shall effectively seal the helium system, shown schematically in Figure 9, coded 104. The design requirements for the coupling assemblies are presented in Table 25.

1.3.1.1.1.4 Test Point Coupling Assemblies. The test point coupling assemblies shall be used in the operational checkout of the pressurization assembly. The coupling assemblies located in the assembly are shown schematically in Figure 9, coded 105, 111, 112, and 113. The design requirements for the coupling assemblies are presented in Table 25.

1.3.1.1.1.5 Helium Check Valve Assemblies. A parallel installation of check valve assemblies, downstream of the helium pressure regulators, shall prevent the backflow of propellants into the pressurization assembly. Each check valve assembly shall incorporate four independent check valves installed in a series-parallel combination. The valves shall permit helium flow in the downstream direction only. A single helium distribution line shall connect each of the assemblies directly to its respective oxidizer or fuel storage tank. The location of the check valve assemblies are shown schematically in Figure 9, coded 107 and 108. The design requirements for the valve assemblies are presented in Table 25.

1.3.1.1.1.6 Helium Pressure Relief Valve Assemblies. One oxidizer and one fuel pressure relief valve assembly, each consisting of a relief valve, a burst diaphragm, and a filter screen, shall be provided for the respective propellant tankage systems. The pressure relief valves shall limit the pressure loads applied to the propellant tankage systems if an excessive pressure rise occurs. In this event, the protective burst diaphragm shall burst and helium and propellant vapor shall be vented to space

Table 25. SPS Component Design Requirements

	Proof Pressure (psig)	Burst Pressure (psig)	Operating Pressure (psig)	Flow Rates	Design Loads (lb)	Creepage (inches)	Flexibility	Duty Cycle	Service Life	Electrical	Response	Engagement Disengage-ment Force
Pressurization subsystem helium pressure regulator	6000	8000	Inlet 4000 to 350 out-let 182 to 195	6 to 9 lb/min at 60°F				Zero to full flow	1500 duty cycles and 336 hours		2 seconds	
Pressurization subsystem helium shutoff valve	6750	9000	0 to 4500	Zero to 9 lb/min				Open Close Open	2500 duty cycles and 336 hours	2 amps at 30 vdc pull in 2>18 vdc	200 mil sec open 325 mil sec close	
Pressurization subsystem helium fill and drain coupling	6750	9000	4500		Axial 100 lb lateral 75 lb			Engagement - disengagement	100 duty cycles and 1500 hours			120 in-lb torque 35 lb-longitude
Pressurization subsystem test point coupling	Propellant couplings 540 helium couplings 6750	Propellant couplings 720 helium couplings 9000	Propellant couplings 360 helium couplings 4500					Engagement - disengagement	100 duty cycles and 1500 hours			30 in-lb torque 35 lb-longitude
Pressurization subsystem helium check valve	375	500	Cracking 1 5 to 3 0					Static Full Flow Static	4000 duty cycles and 360 hours			
Helium relief valve	375	500	Diaphragm 216±4 valve 221±4	3 lb/min at 225 psig 60°F				Diaphragm 0 to 205 to zero Valve 212 to 225 to 212	Diaphragm 1500 cycles valve 4000 cycles and 336 hours			
Helium tank	4914	5528	3585			0 125 circum at 3940 psig 120°F for 336 hrs		100 psi to 4400 to 100 every 5 minutes	100 duty cycles			
Propellant subsystem test point coupling	Propellant couplings 540 helium couplings 6750	Propellant couplings 720 helium couplings 9000	Propellant couplings 360 helium couplings 4500					Engagement - disengagement	100 duty cycles and 1500 hours			30 in-lb torque 35 lb-longitude
Propellant subsystem fill, vent, and drain coupling (engine)	540	720	360		Axial 75 lb lateral 50 lb			Engagement - disengagement	100 duty cycles and 1500 hours			50 in-lb torque 20 lb-longitude
Propellant subsystem fill, vent, and drain coupling (tanks)	360	480	240		Propellant axial 100 lb lateral 75 lb heli-um axial 150 lateral 100			Engagement - disengagement	100 duty cycles and 1500 hours			20 in-lb torque 35 lb-longitude
Propellant subsystem flexible bellows fuel	375	480	Normal 178 maxi-mum 225				Longitude 0 355 in axial offset 0 188 in angular ±2°	Surge of 135 psi in 100 mil-liseconds at 138 psi +240 psi	2500 duty cycles and 1500 hours			
Propellant subsystem flexible bellows oxidizer	375	480	Normal 178 maxi-mum 225				Longitude 0 355 in axial offset 0 188 in angular ±2°	Surge of 135 psi in 100 mil-liseconds at 138 psi +240 psi	2500 duty cycles and 1500 hours			
Propellant subsystem heat exchanger fuel	375	480	Normal 178 maxi-mum 2225					Surge of 135 psi in 100 mil-liseconds at 138 psi +240 psi	400 duty cycles and 336 hours			
Propellant subsystem heat exchanger oxidizer	375	480	Normal 178 maxi-mum 2225					Surge of 135 psi in 100 mil-liseconds at 138 psi +240 psi	400 duty cycle and 336 hours			
Propellant subsystem propellant tanks	300	338	175			0 125 circum at 225 psi 120°F for 336 hrs		10 psi to 240 every 2 minutes	100 duty cycles			

through the relief valves. The relief valves shall close and reseal the assembly after tank pressures have been returned to a safe operating level. Prior to the rupture, the burst diaphragms shall prevent propellants from entering the relief valves. Filter screens shall prevent foreign particles or burst diaphragm fragments from entering the relief valves after rupture of the burst diaphragm. The location of the relief valves in the stream is shown in Figure 9, coded 109 and 110. Design requirements for the valves are presented in Table 25.

1.3.1.1.1.7 Helium Tanks. The high-pressure helium supply shall be contained within two spherical pressure vessels with internal volumes of 19.6 cubic feet each. Each vessel is fabricated from two hemispheres, connected by fusion welding. The tanks are shown schematically in Figure 9, coded 103. Design requirements are presented in Table 25.

1.3.1.2 Propellant Supply and Distribution Equipment

The propellant equipment shall consist of an oxidizer and a fuel supply and distribution system. The oxidizer supply shall be contained within two hemispherically domed, cylindrical tanks of an internal volume of 160.5 cubic feet for the sump tank and 127.8 cubic feet for the storage tank. Normal working pressure in the tanks shall be 175 psi as compared to a limit pressure of 225 psi and a proof pressure of 300 psi. The two tanks shall be connected in series by a single propellant transfer line, with the upstream tank utilized as the storage tank and the downstream tank used as the sump tank. All propellant lines shall have a minimum proof pressure of 480 psi and a burst pressure of 960 psi. The sump tank shall supply oxidizer to the engine through a single supply line and, by means of a propellant retention reservoir in the bottom of the sump tank, shall retain sufficient propellant at the tank outlet to provide restart in a zero-g condition. The oxidizer tanks will contain a maximum of 24,430 pounds of usable oxidizer plus allowances for loading tolerances, residuals and required ullage. The fuel tanks shall have the same internal volumes as the oxidizer tanks and will contain a maximum of 15,270 pounds of usable fuel plus allowances.

1.3.1.2.1 Propellant and Pressurant. The following fluids shall be utilized in the SPS:

1. Oxidizer - Nitrogen tetroxide (N_2O_4) conforming to Specification MIL-P-26539.

2. Fuel - An equal mixture by weight of hydrazine (N_2H_4) and unsymmetrical dimethylhydrazine (UDMH) conforming to Specification MIL-P-27402.

3. Pressurant - Helium conforming to Specification MIL-P-27407.

1.3.1.2.2 Nominal Propellant Engine Inlet Conditions.

1. Starting mode supply pressure

Fuel	178 psia
Oxidizer	176 psia

2. Steady-state supply pressure

Fuel	169 psia
Oxidizer	160 psia

3. Propellant temperature 70 F

1.3.1.2.2.1 Operating Propellant Supply Pressure. Propellant supply pressure at the inlet to the engine valves is 178±4 psia for fuel and 176±4 psia for oxidizer. During normal starting modes the fuel and oxidizer pressures shall be within ±4 psi of each other. During normal steady-state engine operation the fuel inlet pressure will be 7 to 11 psi higher than the oxidizer inlet pressure. During normal steady-state engine operation the fuel is furnished to the propellant interface at 169±4 psia and the oxidizer at 160±4 psia.

1.3.1.2.2.2 Operating Propellant Supply Temperature. The bulk temperatures of propellants furnished to the engine are to be within these ranges:

1. Oxidizer plus 40 degrees to 80 degrees F
2. Fuel plus 40 degrees to 80 degrees F

Throughout this range, the fuel temperature is to be within plus or minus 10 degrees F of the oxidizer temperature. As the engine feed lines are exposed to a wider temperature environment, propellants will be delivered to the engine within the following ranges for short periods on start.

1. Oxidizer plus 30 degrees to 135 degrees F
2. Fuel plus 40 degrees to 135 degrees F

1.3.1.2.3 Fluid Leakage. Engine external or internal fluid leakage which impairs or endangers functioning of the engine or vehicle shall not be permitted. Drain ports shall be provided as necessary to dispose of permissible external leakage.

1.3.1.2.4 Propellant Assembly. The propellant assembly shall consist of the following components:

1. Test point coupling assemblies

2. Fill, vent, and drain coupling assemblies (engine)

3. Fill, vent, and drain coupling assemblies (tanks)

4. Flexible bellows (UDMH/hydrazine)

5. Flexible bellows (nitrogen tetroxide)

6. Heat exchanger (UDMH/hydrazine)

7. Heat exchanger (nitrogen tetroxide)

8. Propellant tanks

9. Propellant retention reservoir and screen

1.3.1.2.4.1 Test Point Coupling Assemblies. The test point coupling assemblies shall be used for operational checkout and drain of the propellant assembly. The units are coded 111 in Figure 9. Design requirements for the coupling assemblies are presented in Table 25.

1.3.1.2.4.2 Fill, Vent, and Drain Coupling Assemblies (Engine). The coupling assemblies shall be used in servicing the propellant assembly by venting at the engine valve oxidizer and fuel inlets. The couplings are shown schematically in Figure 9, coded 504 and 505. Design requirements for the coupling assemblies are presented in Table 25.

1.3.1.2.4.3 Fill, Vent, and Drain Coupling Assemblies (Tanks). The coupling assemblies shall be used in servicing the propellant assembly as the spacecraft coupling for venting and draining the oxidizer and fuel tanks. The coupling assemblies are shown schematically in Figure 9, coded 112, 113, 202, 205, 301, and 304. Design requirements for the coupling assemblies are presented in Table 25.

1.3.1.2.4.4 Flexible Bellows (UDMH/Hydrazine). Flexible connectors shall be installed in the propellant supply lines upstream of the engine interface. These units shall employ stainless steel bellows assemblies to provide for correction of minor misalignments of the propellant lines at the engine interface. The fuel bellows is shown schematically in Figure 9, coded 306. Design requirements for the bellows are presented in Table 25.

1.3.1.2.4.5 Flexible Bellows (Nitrogen Tetroxide). The oxidizer bellows is shown schematically in Figure 9, coded 207. Design requirements for the bellows are presented in Table 25.

1.3.1.2.4.6 Heat Exchanger (UDMH/Hydrazine - Helium). The fuel-helium heat exchanger shall be mounted in the fuel supply line downstream of the sump tank. The heat exchanger consists of a helium line coiled helically within an enlarged section of the fuel line. The line-mounted, counterflow heat exchanger shall effect transfer of heat from the fuel to the helium at a point in the helium circuit just upstream of where the helium enters the fuel storage tank. By utilizing helium at a temperature near that of the fuel, the heat transfer from the fuel to the helium within the tanks is minimized and resultant tank pressure excursions are avoided. The unit shall provide discharge temperatures as indicated at any combination of the conditions shown.

The heat exchanger is shown schematically in Figure 9, coded 305. The design requirements for the heat exchanger are presented in Table 25.

Fluid	Fluid Inlet Temperature Range	Reference Flow Rate (lb/sec at 60 degrees F)	Inlet Pressure (PSIG)	Discharge Temperature
Helium	Zero to 180 degrees F below propellant temperature*	0.070	178 plus or minus 4	Within 24 degrees F of propellant discharge temperature
Propellant	Plus 40 to 80 degrees F	45.3	173 plus or minus 4	Within 2 degrees F of propellant inlet temperature

* With propellant inlet temperatures of plus 40 and plus 80 degrees F, the minimum helium temperature will be minus 140 and minus 100 degrees F, respectively.

1.3.1.2.4.7 Heat Exchanger (Nitrogen Tetroxide - Helium). The oxidizer-helium heat exchanger shall be mounted in the oxidizer supply line in a similar manner as described for the fuel heat exchanger in paragraph 1.3.1.2.4.6. The oxidizer heat exchanger shall provide discharge temperatures as indicated at any combination of conditions shown:

Fluid	Fluid Inlet Temperature Range	Reference Flow Rate (lb/sec at 60 degrees F)	Inlet Pressure (PSIG)	Discharge Temperature
Helium	Zero to 180 degrees F below propellant temperature*	0.070	178 plus or minus 4	Within 25 degrees F of propellant discharge temperature
Propellant	Plus 40 to plus 80 degrees F	45.3	173 plus or minus 4	Within 2 degrees F of propellant inlet temperature

* With propellant inlet temperatures of plus 40 and plus 80 degrees F, the minimum helium temperature will be minus 140 and minus 100 degrees F, respectively.

The heat exchanger is shown schematically in Figure 9, coded 206. The design requirements for the heat exchanger are presented in Table 25.

1.3.1.2.4.8 Propellant Tanks. The propellant supply shall be contained within four hemispherically domed, cylindrical tanks, two with internal volumes of 160.5 cubic feet each (not pressurized) and two with internal volumes of 127.8 cubic feet each (not pressurized).

Each of the four tanks are fabricated from two rings and two hemispheres, connected by fusion welding. The two tanks for each propellant shall be connected in series by a single transfer line. The larger tanks shall be utilized as sump tanks, and the smaller as storage tanks. The propellant tanks are shown schematically in Figure 9, coded 203 and 204. The design requirements for the propellant tanks are presented in Table 25.

1.3.1.2.4.9 Propellant Retention Reservoir and Screen. A propellant retention reservoir and umbrella screen shall be provided in the fuel and oxidizer sump tanks to minimize RCS propellant settling requirements for restart under space environment conditions. The reservoir and screen are shown schematically in Figure 9, coded 201 and 209, respectively.

1.3.1.3 Propellant Utilization and Gaging Equipment

The propellant utilization and gaging equipment shall consist of oxidizer and fuel tank primary and auxiliary quantity-sensing devices, an electrical control unit, an oxidizer flow control valve assembly, and a crew display panel. An electrical schematic diagram of the subsystem is shown in Figure 10. This system shall monitor the quantities of propellants remaining in order that the desired oxidizer-fuel ratio in the propellant tanks can be manually adjusted during propellant expulsion so that depletion of the oxidizer and fuel occurs simultaneously. The primary quantity sensors shall be cylindrical capacitance probes mounted axially in each tank. The auxiliary gaging system shall utilize impedance-type point sensors, providing a step function impedance change when the liquid level passes their location centerline and shall use a nominal flow integrator between sensor levels. Oxidizer flow control shall be accomplished by manual control of a motor-operated, redundant, double-blade valve assembly, which shall have three positions, for increased, decreased, or normal oxidizer flow rates. The control unit shall provide the circuitry to perform all required computations for the gaging system. The control unit shall: (1) compute total propellant quantities from individual tank quantities; (2) compute propellant unbalance; (3) continuously compare oxidizer-fuel ratio; (4) continuously compare total propellant quantities indicated by primary and auxiliary systems. The crew display panel shall provide all onboard output indications required by the crew and all switches for control functions and onboard testing.

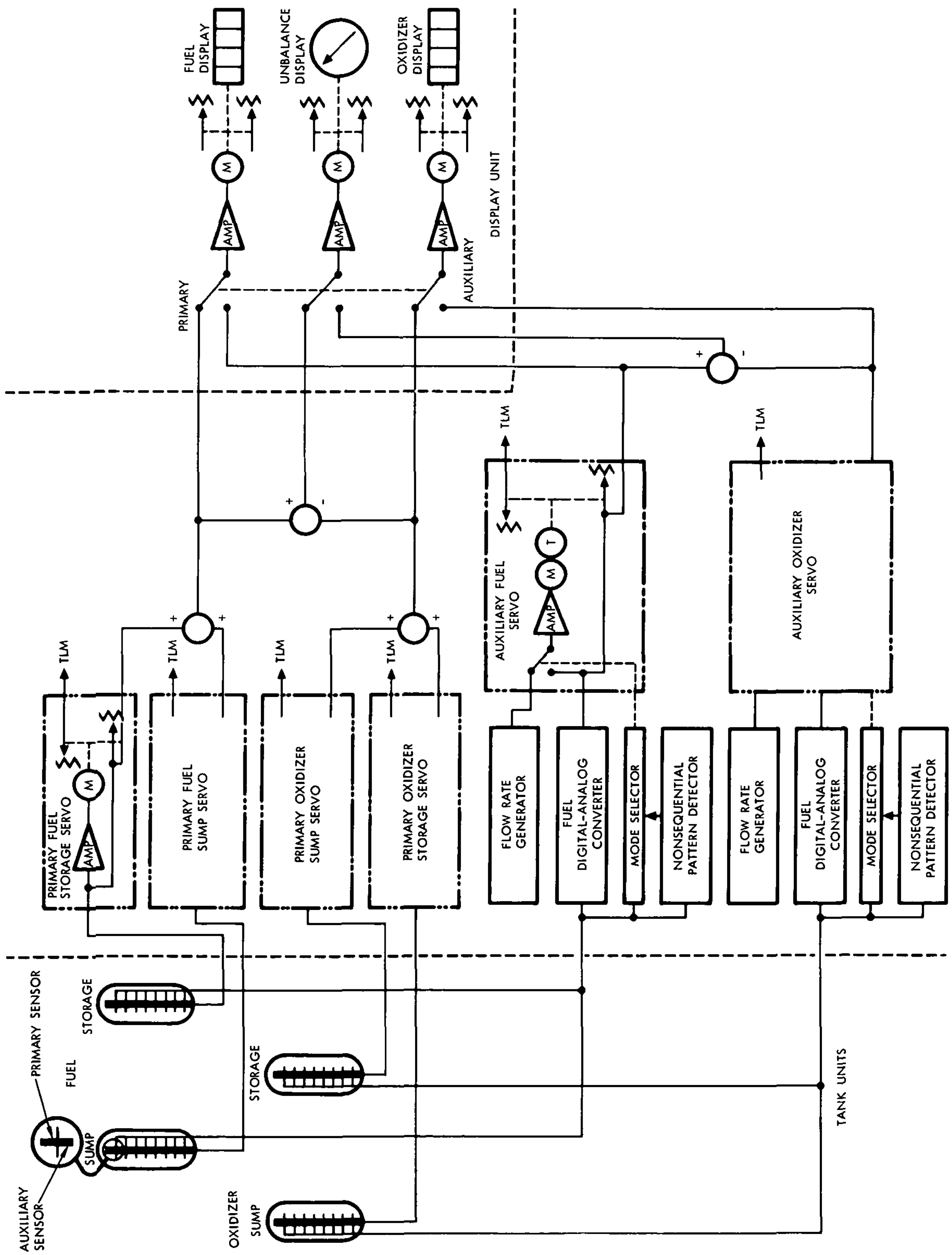


Figure 10. Primary and Auxiliary Gaging Subsystem

1.3.1.3.1 Propellant Utilization and Gaging Subassembly (PUGS). The PUGS shall consist of the following components, identified in Figure 9 by the coding indicated:

1. Oxidizer and fuel tank primary and auxiliary sensing devices 403, 404, 405, and 406
2. Electrical control unit 402
3. Oxidizer flow control valve assembly 407
4. Crew display panel 401.

1.3.1.3.1.1 PUGS Function. The PUGS shall be capable of meeting the performance requirements specified herein.

1. The PUGS shall provide propellant quantity information in percentage by weight remaining to the crew through the use of indicators on a display panel and to the earth through appropriately conditioned signals to the spacecraft telemetry system.
2. The PUGS shall indicate oxidizer/fuel (O/F) unbalance from a 1.6 weight ratio to the crew to permit crew adjustments of the O/F feedout ratio such that simultaneous depletion of the oxidizer and fuel may be achieved.
3. The oxidizer flow control valve will provide O/F mixture ratio control and the operating of the valve shall be reflected to the crew by position indicators.

1.3.1.3.1.2 Liquid Surface Angle. During propellant feedout, the PUGS will meet the requirements herein with any propellant surface angle up to 13 degrees. The planes of the propellant surfaces in all tanks will be commonly oriented.

1.3.1.3.1.3 Propellant Feedout. When the propellants are settled at the aft end of the tanks under at least 0.1 g settling force during either no flow or propellant feedout conditions, the airborne PUGS shall be capable of performing the following functions:

1. Provide to the crew remaining propellant information for total oxidizer and total fuel in percentage by weight and oxidizer unbalance from the desired propellant ratio.
2. Provide to the spacecraft telemetry system a conditioned voltage signal representing propellant quantities remaining in each tank.

1.3.1.3.1.3.1 Time Delay. The PUGS shall incorporate provisions for a 4.5 second time delay between initiation of engine thrust and display of propellant quantity changes to ensure an adequate propellant settling period.

1.3.1.3.1.3.2 Propellant Loading. During propellant loading, the subsystem shall provide a signal indicating propellant quantities in the tanks to the propellant servicing equipment during separate or simultaneous oxidizer and fuel loading.

1.3.1.3.1.4 Quantity Indicators. The difference between the actual total propellant quantity aboard and the total quantity presented by the indicators for each propellant shall not exceed 0.25 percent of full tank plus 0.25 percent of propellant remaining at a tank pressure of 175 psig. This requirement shall apply separately to the total oxidizer and total fuel assembly.

1.3.1.3.1.4.1 Propellant Unbalance Indicators. The difference between the unbalance shown on the quantity indicators and that shown on the unbalance indicators shall not exceed 0.1 percent.

1.3.1.3.1.4.2 Valve Position Indicators. When the valve assembly is actuated from its normal flow position, the valve position indicator shall not be completely visible at the required corresponding position until at least 80 percent of the required valve movement relative to correction capacity is completed. The indicator shall remain visible until the valve has returned to within 20 percent of its normal flow capacity position.

1.3.1.3.1.4.3 Telemetry Signals. The difference between actual propellant quantity in each tank and that represented by the signals to the telemetry equipment shall be within 0.25 percent of full tank quantity plus 0.25 percent of propellant remaining at a pressure of 175 psig. This requirement shall apply separately to each oxidizer and fuel tank.

1.3.1.3.1.4.4 Propellant Loading. With simultaneous or separate filling of the oxidizer and fuel tanks, the PUGS shall provide signals representing propellant quantities to be used in conjunction with tank loading equipment. The propellant quantities, represented by the signals, shall not differ from actual quantities by more than 0.25 percent plus 0.25 of propellant loaded at the propellant levels corresponding to the point sensor stations as corrected for loading pressures. The above requirements shall apply with any fill rate of 12.0 pounds per second or less of oxidizer and 7.5 pounds per second or less of fuel.

1.3.1.3.1.5 Power Consumption. With any combination of PUGS components operating and while any combination of duties are being performed, power consumption shall not exceed 100 volt amperes at 0.75 power factor, or 75 watts power. In addition, any combination ac and dc power consumption shall not exceed 75 watts.

1.3.1.3.1.6 Warmup. The PUGS shall commence operating within 2 seconds after introduction of power.

1.3.1.3.1.7 Oxidizer Flow Control. The oxidizer flow control valve assembly shall be designed such that during operation of the propellant assembly, the redundant standby valve remains idle in its normal flow position with all flow adjustments being made by manual positioning of the primary valve. The valve location in the SPS is shown in Figure 9, coded 407.

1.3.1.3.1.7.1 <u>Control Range</u>. With oxidizer at a temperature of 70 degrees F and at a pressure of 168±4 psig, the valve assembly shall be designed to provide the following conditions in its three operating positions.

Valve Position	Flow Lb/Sec
Flow increased	TBD± ___
Normal flow	39.1±10
Flow decreased	TBD± ___

The standby valve of the redundant valve assembly shall be capable of providing the same correction with the primary valve failed in any position within its operating range.

1.3.1.3.1.7.2 <u>Valve Response</u>. With the flow conditions of paragraph 1.3.1.3.1.7.1 prevailing, as applicable, the valve shall travel from its normal flow position to either correcting position or from either correcting position to its normal flow position within 3 seconds after energization.

1.3.1.4 Engine Assembly

The engine shall be a nonthrottleable, gimballed, pressure-fed rocket engine consisting of an ablation-cooled combustion chamber, a radiation-cooled nozzle extending from approximately 6:1 to 62.5:1 area ratio, a bolt-on aluminum injector, a thrust and gimbal mount, propellant valves, gimbal actuators, electrical harness, and propellant feed lines. The engine shall nominally operate at a chamber pressure of 96.0 pounds per square inch absolute (psia), measured at the injector face, and produce 20,000 pounds vacuum thrust. For mission planning considerations, the average nominal specific impulse shall be 314.9 seconds and the minimum specific impulse (minus 3 sigma) shall be 311.7 seconds after 750 seconds firing duration. The engine shall be capable of a minimum of 36 restarts within the engine design life of 750 seconds. The oxidizer and the fuel shall be injected into the thrust chamber at a nominal mixture ratio of 1.6 to 1 by weight of oxidizer to fuel where ignition occurs by hypergolic reaction. Control of propellants to the thrust chamber assembly shall be provided by a redundant set of series-parallel ball valves actuated by pneumatic pressure controlled by electrically operated solenoid valves. The pneumatic pressure shall be provided through the use of some inert gaseous media compatible with all other components of the subsystem. Engine gimballing on the yaw and pitch axis shall be accomplished by electrical actuators (26 volts dc) with redundant motors, gear train, and clutches operating on a single ball-screw shaft.

1.3.1.4.1 Total Impulse. The total impulse provided by the SPS to satisfy normal design mission and emergency requirements shall be 11,966,200 pound-seconds nominal, minus 4500 pound-seconds for each engine restart.

1.3.1.4.2 Altitudes and Temperatures. The engine, with nozzle extension removed, shall be designed to start, operate, and shut down satisfactorily through pressure altitudes from sea level to an altitude equivalent to 1.0×10^{-6} millimeters of mercury (mm Hg) absolute. With nozzle extension installed, the engine shall not be required to operate at ambient pressures exceeding 0.20 psia.

1.3.1.4.2.1 Static Exposure. The engine shall start, operate, and shut down when supplied with propellant at conditions specified in paragraphs 1.3.1.2.2 through 1.3.1.2.2.2, after environmental exposure as specified in SID 64-1344.

1.3.1.4.2.2 Attitudes. The engine shall start, operate, and shut down satisfactorily in any attitude, provided propellants at the nominal conditions specified in paragraphs 1.3.1.2.2 through 1.3.1.2.2.2 are furnished to the propellant line inlets, under space environment conditions.

1.3.1.4.3 Ratings and Performance Data. The nominal vacuum performance of the engine shall be as specified in Table 26. The ratings and data are based on the use of propellants conforming to the pressures and temperatures specified in paragraphs 1.3.1.2.2 through 1.3.1.2.2.2 and include run-to-run and engine-to-engine variations.

1.3.1.4.4 Abnormal Operating Conditions. The engine shall be designed for operating under the abnormal conditions specified in paragraphs 1.3.1.4.4.1 through 1.3.1.4.4.6. Variation in performance from the limits specified in Table 26 shall be consistent with the off-design conditions stipulated in these paragraphs.

1.3.1.4.4.1 Starting Mode. The engine shall be capable of safely starting under the following conditions:

1. A minimum of 10 starts with propellants supplied to the engine propellant line inlets at a maximum pressure of 240 psia.

2. A minimum of 20 starts with propellants supplied to the engine propellant line inlets at a pressure of 219 psia.

3. A minimum of 5 starts with propellant line oxidizer inlet pressure of 193 psia and propellant line fuel inlet pressure of 183 psia.

1.3.1.4.4.2 Continuous Operation Mode (Constant Pressure). The engine shall be capable of safe continuous operation with fuel supplied to the engine inlets at 173 psia and oxidizer supplied to the engine inlets at 170 psia until 39,900 pounds of propellant are consumed.

Table 26. SPS Ratings and Performance Data in Vacuum
(Steady-State P_c = 96.0 psia)

Parameter	Units	Rating and Data
Thrust	lb	20,000±1% after initial 30 seconds of operation 20,000±10%-1% after 750 seconds of operation
Instantaneous mixture ratio W_o/W_f (static)		1.6±1%
Operating life (minimum)	seconds	750*
Instantaneous specific impulse (minimum)(-3 sigma value)	lb-sec/lb	311.7
Instantaneous specific impulse (average nominal)	lb-sec/lb	314.9
Nozzle exit design pressure, P_e	psia	0.126
Nozzle pressure ratio, P_c/P_e	- - -	755
Nozzle area ratio, A_e/A_t (actual)	- - -	62.5
Total propellant flow rate, W_T (start)	lb/sec	63.5
Fuel flow rate, W_f (start)	lb/sec	24.4
Oxidizer flow rate, W_o (start)	lb/sec	39.1

* A maximum of 30 seconds of the 750-second thrust chamber operating life specified shall be expended by the engine subcontractor during engine acceptance testing.

1.3.1.4.4.3 Continuous Operation Mode (Variable Pressure). In the event of malfunction in the pressurization subsystem, the engine shall be capable of safe continuous operation under the following conditions:

1. Fuel inlet pressure of 225 psia and oxidizer inlet pressure of 220 psia both decaying steadily to 155 psia over a period of 300 seconds.

2. Fuel inlet pressure of 171 psia decaying steadily to 163 psia and oxidizer inlet pressure of 178 psia decaying steadily to 160 psia in approximately 30 seconds.

3. Fuel inlet pressure of 166 psia and oxidizer inlet pressure of 151 psia for 30 seconds.

1.3.1.4.4.4 Reduced Thrust Operation. The engine shall be designed to operate safely with propellant inlet pressures as low as 137-psia fuel pressure and 135-psia oxidizer pressure at a propellant inlet temperature of 80 degrees F.

1.3.1.4.4.5 Propellant Valve Failures. The engine shall be designed to provide safe starting, steady-state operation, and shutdown when operating with either the primary or secondary valve flow paths. Engine operation with either valve flow path shall result in a propellant mixture ratio and specific impulse as specified in Table 26.

1.3.1.4.4.6 Mixture Ratio Excursions. In the event of malfunction in the engine or propellant subsystem, the engine shall be capable of a safe start at mixture ratios between 1.4:1 and 1.8:1. The engine shall be designed for safe continuous operation at a mixture ratio of 1.4:1 for a maximum duration of 570 seconds or at a mixture ratio of 1.8:1 for a maximum duration of 590 seconds.

1.3.1.4.5 Starting and Restarting. The engine shall be capable of a minimum of 36 starts during the operational life of 750 seconds and under conditions specified in paragraphs 1.3.1.4.2 and 1.3.1.4.2.1. Starts shall be accomplished under either sustained or intermittent operation within the tolerances specified in paragraph 1.3.1.4.7.2. Propellant loss during the start cycle due to prime, delayed ignition, and valve leakage shall be minimized. The engine shall be capable of accepting a start signal any time after receipt of a shutdown signal. Conversely, the engine shall be capable of accepting a shutdown signal at any time after receipt of a start signal with a minimum impulse bit equal to or less than 5000 pound-seconds and a run-to-run tolerance of 200 pound-seconds (1 sigma). The post-propellant valve flow passage volume shall be minimized to prevent propellant loss after shutdown. The hypergolic propellants used shall provide for starts consistent with the engine life cited in Table 26. The engine shall be capable of satisfactory operation for a period of 45 days following a nonfiring functional checkout.

1.3.1.4.6 External Power. Electrical power shall be supplied to the engine as described in paragraph 1.3.1.9.

1.3.1.4.7 Controls. The controls shall be such that the engine shall operate within the limits specified in paragraph 1.3.1.4.3.

1.3.1.4.7.1 Chamber Pressure. The engines shall operate at an initial nominal thrust chamber pressure of 96 psia measured at the injector face during steady-state operation.

1.3.1.4.7.2 Thrust. The engine shall develop 90 percent steady-state thrust within 0.4 to 0.6 second after onset of the electrical command signal to the pilot valve. The start transient total impulse from onset of electrical command to 90 percent rated thrust shall be from 400 pound-seconds (minimum) to 1200 pound-seconds (maximum). The run-to-run tolerance on start transient impulse shall be plus or minus 200 pound-seconds (1 sigma).

1.3.1.4.7.2.1 Shutdown Transient. The engine shall accomplish thrust decay to 10 percent rated thrust within 0.4 to 0.6 second after receipt of the command signal. The engine shutdown impulse from onset of electrical command signal to 10 percent rated thrust shall be from 5000 pound-seconds (minimum) to 8500 pound-seconds (maximum). The shutdown impulse from 10 percent to 1 percent rated thrust shall not exceed 500 pound-seconds. The run-to-run tolerance on the shutdown impulse shall be plus or minus 200 pound-seconds (1 sigma).

1.3.1.4.7.3 Starting Stability. The transient starting chamber pressure shall not be greater than 120 percent of the nominal chamber pressure shown in paragraph 1.3.1.4.

1.3.1.4.7.4 Steady-State Stability. The steady-state mean thrust shall not exceed the limits specified in Table 26. The SPS engine shall achieve dynamic combustion stability under all design operating modes. A dynamically stable engine is defined as an engine which, upon experiencing chamber pressure oscillations initiated by any natural or artifically induced triggering pulse, shall decay to the normal steady-state combustion oscillation level (less than ± 5 psi) within 40 milliseconds after termination of the triggering pulse, the resultant oscillation, or the recovery. Recovery to normal steady-state combustion shall occur after application of artifically induced pulses having the maximum magnitude compatible with the structural limits of the injector, or 250 psia ± 30 psia peak, whichever is less.

1.3.1.4.7.5 Thrust Vector Alignment. The engine thrust vector shall pass within 0.125 inch of the intersection of the gimbal axes. The thrust vector shall be perpendicular within ± 0.5 degree to the plane established by the engine mount plane when the gimbal actuators are in the null position. The thrust vector is defined as a line connecting the geometric centroids of the throat and the chamber section at the nozzle attach flange as determined by dimensional measurement. Installed alignment relative to the SC indexes must be plus 2 degrees ± 15 seconds null offset in the X-Z plane and plus 1 degree ± 15 seconds null offset in the X-Y plane.

1.3.1.4.7.6 Engine Gimbal. Thrust vector control of the service propulsion engine shall be achieved by means of two redundant electromechanical servo actuators, capable of positioning the engine during engine operation.

1.3.1.4.7.6.1 Position Limits. The engine deflection shall be plus or minus 4.5 (+0.5, -0) degrees in both the X-Y and X-Z planes. A snubbing device shall be provided to limit engine overtravel to 1 degree beyond the design deflection.

1.3.1.4.7.6.2 Gimbal Actuator. The gimbal actuators shall be used in a closed loop control system as shown in the block diagram of Figure 11.

1.3.1.4.7.6.2.1 Gimbal Actuator Design. The servo actuator shall be a hermetically sealed unit consisting of a ball-screw ram assembly controlled by two sets of counter-rotating, magnetic particle servo clutches, driven by dc motors. Integrally packaged within the actuators shall be two drag-cup velocity transducers and four linear voltage transducers for position indication.

1.3.1.4.7.6.2.2 Gimbal Actuator Requirements. Gimbal actuation system reliability is predicated on the proper utilization of the redundancy provided by the two channels within the actuator. Both the operating and the standby channels shall be composed of a dc motor driving two counter-rotating, dry powder, magnetic particles servo clutches controlling the reciprocating motion of the ball-screw ram assembly. Each channel shall also contain one rate and one position transducer, which shall provide control system feedback. A third position transducer in each actuator shall provide engine position information to the astronaut's control panel in the CM, and information to the TM. A fourth position transducer is provided in each actuator as a spare.

1.3.1.4.7.6.2.3 Dynamic Requirements. With a maximum of 600 milliamperes of differential current applied to either of the two clutches of one channel, the servo actuator shall be capable of producing an engine nozzle acceleration of no less than 3.0 radian/sec^2. This acceleration shall be achieved in the presence of actuator friction, gimbal bearing friction, torques caused by engine plumbing, thrust misalignment torque, maximum accelerations of the gimbal point caused by spacecraft motions, and damping forces caused by engine exhaust gases.

a. Engine Deflection Rate. The servo actuator shall be capable of producing a nozzle deflection rate in either direction of at least 0.23 radian per second while producing a force of 450 pounds.

b. Dynamic Performance. The servo actuator, when coupled to the "spring-mass" system within the control loop, shall perform in a manner compatible with efficient SCS operation.

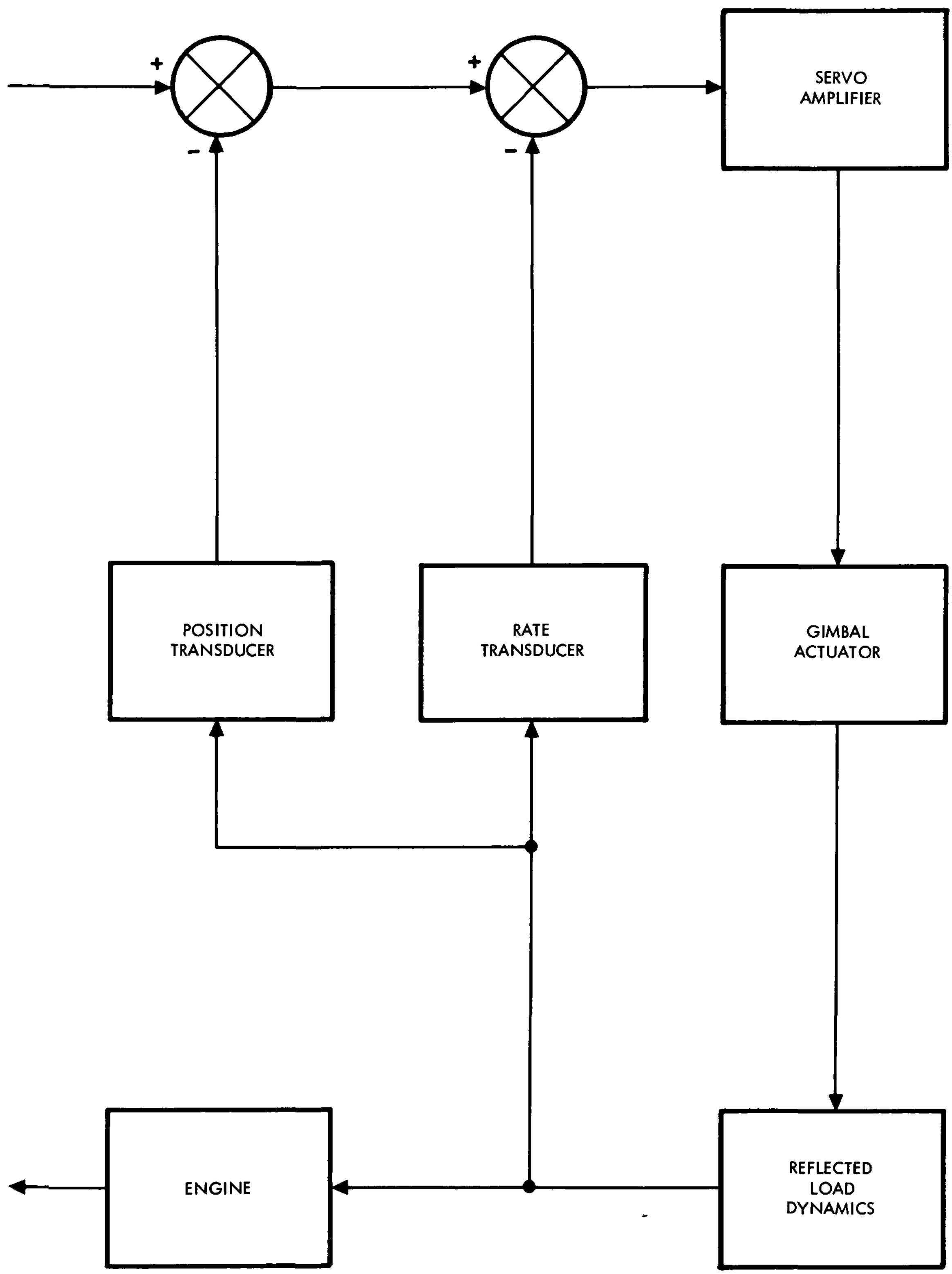


Figure 11. Block Diagram - Gimbal Actuation System

1.3.1.4.7.6.2.4 Construction. The servo actuator assembly shall be designed to withstand the required static and dynamic loads. The actuator shall be mounted by rod end fittings and must provide adjustments of the midposition length with an adjustment locking capability. The actuator shall operate satisfactorily when mounted in any position. The pitch and yaw actuator and attachment fittings shall be designed such that the two actuators cannot be physically interchanged on the engine assembly. The gimbal actuator assembly shall be hermetically sealed and filled with a gas that is compatible with the actuator internal operation.

1.3.1.4.7.6.2.5 Motor. The motor in each actuator channel shall be powered by 26 plus or minus 4 volts dc.

a. Clutches. Each control clutch shall have isolated windings terminating in two leads per clutch coil. The effective response characteristics defined in terms of actuator force output response to coil current inputs shall be compatible with the dictates of efficient SCS operation.

b. Velocity Transducer. The servo actuator assembly shall have, integrally packaged, two rate transducers providing voltage outputs proportional to actuator ram rates. Excitation voltage to the generator shall be 26 plus or minus 0.5 volts single phase at 400 plus or minus 4.0 cps. The nominal voltage output shall be 0.5 volt per inch per second of ram travel.

c. Position Transducer. The servo actuator assembly shall have, integrally packaged, four position transducers providing voltage output proportional to actuator ram displacements. Excitation voltage to the position transducers shall be 26 plus or minus 0.5 volts single phase, 400 plus or minus 4.0 cps. The normal voltage output shall be 1.0 volt per degree of engine deflection.

1.3.1.4.7.6.2.6 Actuator Force Output. The servo actuator shall be capable of producing a force output proportional to the applied differential current. Each channel of the servo actuator shall have the following characteristics.

1.3.1.4.7.6.2.7 Gain. The force gain shall be between 2.55 lb/ma and 3.45 lb/ma.

a. Linearity. The maximum deviation from the best straight line of the force gain curve shall be no more than 5 percent of the nominal value.

b. Peak Output Force. Each channel of the actuator shall attain an output force of 1500 pounds at an input differential current of not more than 600 ma in both directions.

1.3.1.4.7.6.2.8 Input Power Requirements. The maximum peak power input to the motor of either channel of the servo actuator shall not exceed 1800 watts while that channel is operating at the peak output force in accordance with paragraph 1.3.1.4.7.6.2.7.

1.3.1.5 Instrumentation

The SPS shall be provided with the instrumentation listed herein. The instrumentation is shown in the electrical schematic of Figure 12.

1.3.1.5.1 Helium Pressurization Equipment.

1.3.1.5.1.1 Helium Isolation Valve Position Indicator. Each helium isolation valve will incorporate a position indicator switch. The switch will have a minimum power rating of 2.5 watts and a maximum contact resistance of 50 milliohms. The switch will be compatible with a 28-volt dc circuit which includes an electromagnetic valve position indicator.

1.3.1.5.1.2 Helium Temperature Indicator. A temperature transducer will be bonded onto the helium tanks common supply line to the propellant tanks for helium temperature presentation in the CM.

1.3.1.5.2 Propellant Supply and Distribution Equipment.

1.3.1.5.2.1 Engine Inlet-Feed Pressure Transducer.

1.	Range	0-300 psia
2.	Accuracy	4 percent of full scale
3.	Excitation voltage	28 vdc to signal condition 1.5 vdc to transducer 0.5 vdc to display panel

1.3.1.5.2.2 Engine Inlet-Oxidizer Pressure Transducer.

1.	Range	0-300 psia
2.	Accuracy	4 percent of full scale
3.	Excitation voltage	28 vdc to signal condition 1.5 vdc to transducer 0.5 vdc to display panel

1.3.1.5.3 Propellant Utilization and Gaging Equipment.

1.3.1.5.3.1 Propellant Utilization (PU) Valve Indicator. PU valve oxidizer increase and decrease flag indications will be provided when the valve indicator will not actuate to the corresponding position until at least 80 percent of the valve movement is completed. The indicator will remain visible until the valve has returned to within 20 percent of its normal flow position.

1.3.1.5.3.2 Oxidizer and Fuel Quantity Indicator. Propellant quantities will be displayed in percent remaining by digital presentation. The difference between the actual total propellant quantity aboard and the total quantity presented by the indicators for each propellant will not exceed 0.25 percent of full tank plus 0.25 percent of propellant remaining. This requirement is applied separately to the total oxidizer and total fuel system.

1.3.1.5.3.3 Propellant Unbalance Indicator. Oxidizer-fuel quantity unbalance will be displayed as plus or minus oxidizer pounds. Total quantity gage signals will be utilized for the unbalance indication. The difference between the unbalance shown on the quantity indicators and that shown on the unbalance indicators will not exceed 0.1 percent.

1.3.1.5.3.4 Propellant Utilization Warning Light. Circuits will be provided that will energize a 28-volt, 80-ma caution lamp and audible warning when malfunction occurs in the propellant sensing system or when the oxidizer to fuel weight unbalance from 1.6 to 1 reaches 300 pounds or 90 percent of the critical unbalance, whichever is less. The critical unbalance is defined as the oxidizer-to-fuel unbalance point beyond which the quantities cannot be corrected to simultaneous depletion within the remaining flow time.

1.3.1.5.4 Rocket Engine Assembly.

1.3.1.5.4.1 Main Propellant Valve Position Transducer. Redundant position potentiometers or limit switches with the following characteristics are to be supplied on each interlinked fuel-oxidizer propellant valve:

1. Potentiometers

Type:	A potentiometer capable of producing an output signal directly proportional to valve position
Resistance:	Less than 5000 ohms
Accuracy:	Plus or minus 2 percent of full scale (3 sigma)
Excitation:	5 volts dc
Output signal:	0 to 5 volts dc from a fully closed valve to a fully open valve, single ended

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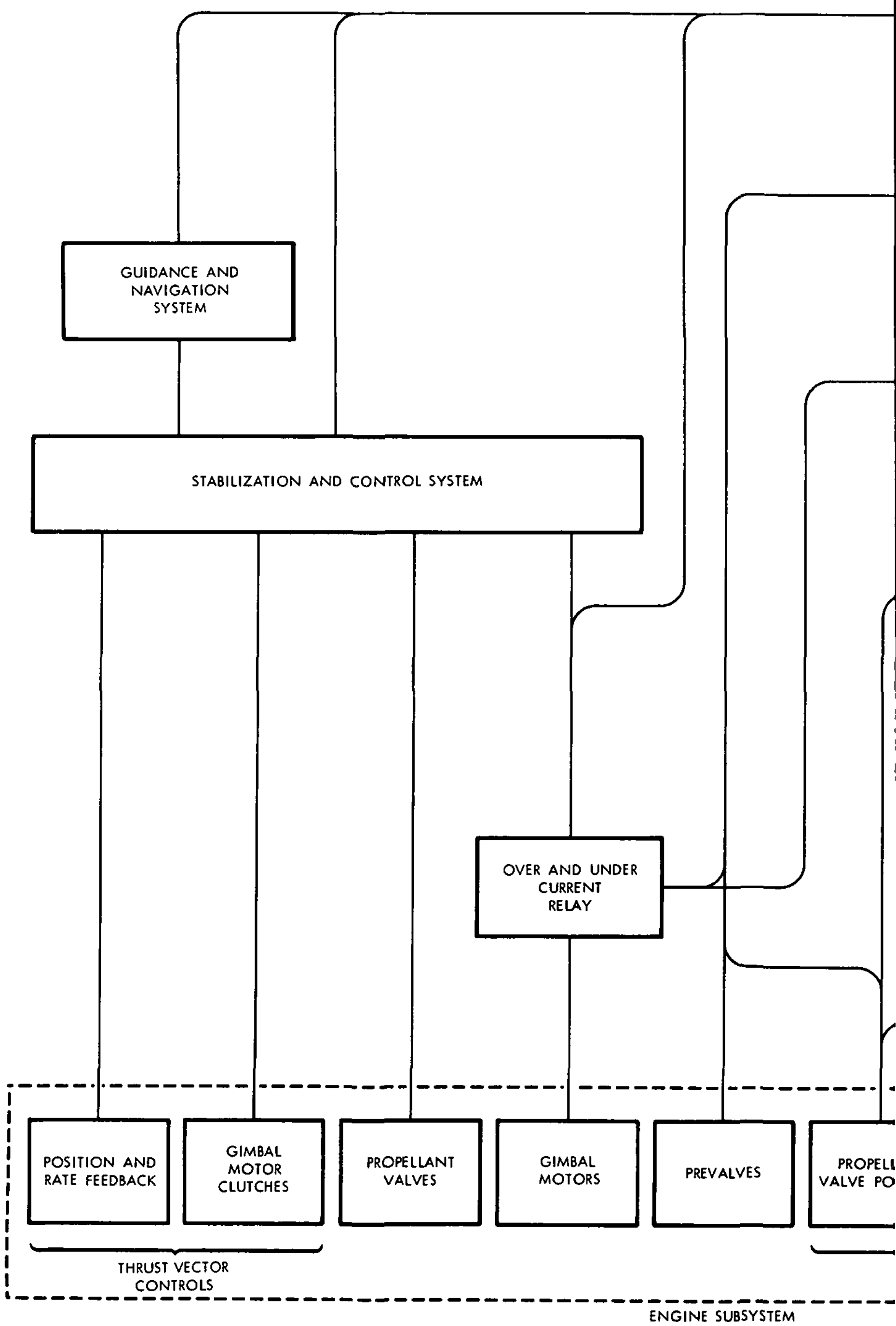

Fig 12

1

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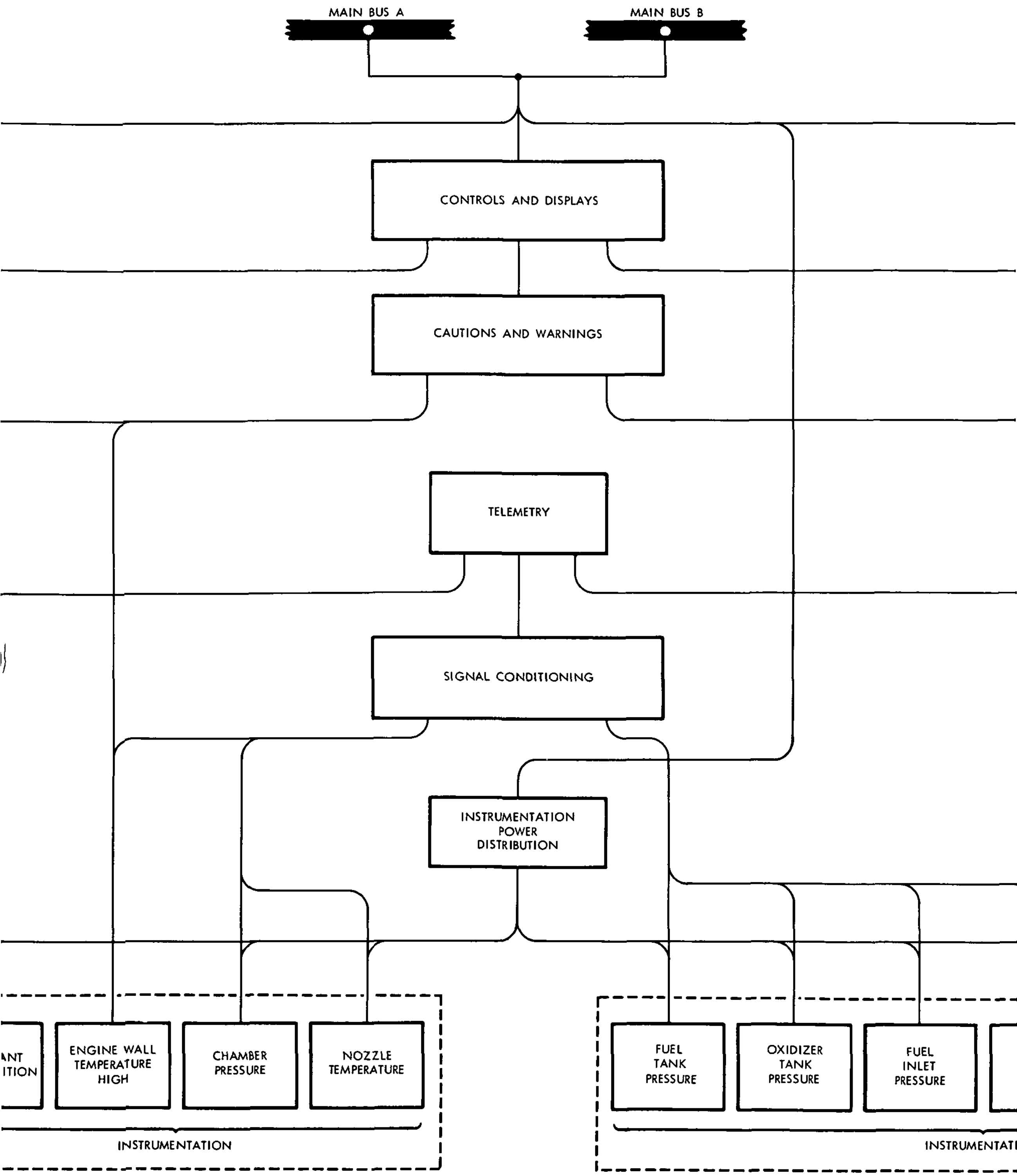

MAIN BUS A
MAIN BUS B
CONTROLS AND DISPLAYS
CAUTIONS AND WARNINGS
TELEMETRY
SIGNAL CONDITIONING
INSTRUMENTATION POWER DISTRIBUTION
ENGINE WALL TEMPERATURE HIGH
CHAMBER PRESSURE
NOZZLE TEMPERATURE
INSTRUMENTATION
FUEL TANK PRESSURE
OXIDIZER TANK PRESSURE
FUEL INLET PRESSURE

Fig 12

2

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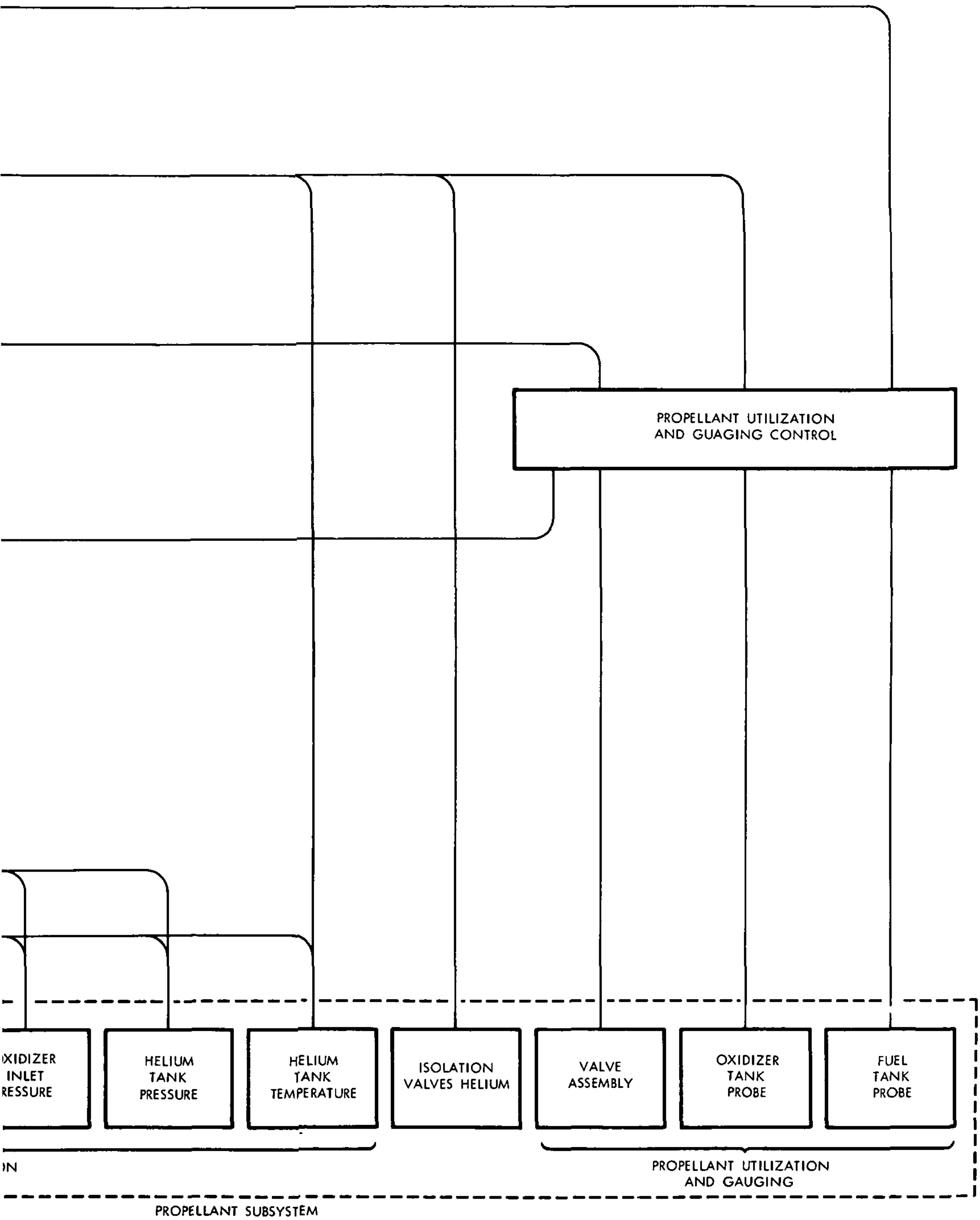

Figure 12. SPS Electrical Block Diagram

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2. Limit switches

Excitation:	5 volts dc
Setting:	Switches shall be adjusted so that they are actuated 5 degrees before the full open or full closed positions

1.3.1.5.4.2 Instrumentation Bosses. Instrumentation bosses for chamber pressure, propellant valve inlet oxidizer pressure, propellant valve inlet fuel pressure, primary engine valve actuation system tank pressure and secondary engine valve actuation system tank pressure will be provided on the engine assembly for installation of pressure transducers as shown schematically in Figure 9. Instrumentation bosses for the helium, oxidizer, and fuel tank pressure transducers are provided in the connecting plumbing of the pressurization and propellant assembly as shown schematically in Figure 9.

1.3.1.5.4.3 Combustion Chamber Skin Temperature Sensors. Temperature sensors with the following characteristics will be provided on the outside surface of the combustion chamber:

1. Range	0 degrees to 500 degrees F
2. Accuracy	3 percent over range (3 sigma)
3. Excitation voltage	5 volts dc

1.3.1.5.4.4 Nozzle Extension Thermocouples. One chromel/Alumel thermocouple shall be provided on the engine nozzle extension at a station location 2.5 inches aft of the nozzle flange.

1.3.1.5.4.5 Thrust Chamber Pressure Transducer. A miniaturized pressure transducer with the following characteristics will be provided in a sealed enclosure, integral with the outer structure of the engine injector:

1. Range	0 to 150 psia
2. Accuracy	3 percent over range (3 sigma)
3. Excitation voltage	28 vdc to signal conditioner 1.5 vdc to transducer 0.5 vdc to display panel

1.3.1.5.5 Engine Gimbal Assembly.

1.3.1.5.5.1 Gimbal Angle Position Transducer. The servo actuator assembly shall have integrally packaged, four position transducers providing voltage output proportional to actuator ram displacements. Excitation voltage to the position transducers will be 26 volts plus or minus 0.5 volts single phase, 400 plus or minus 4 cps ac. The normal voltage output shall be 1.0 volt per degree of engine deflection when loaded with 10,000 ohms. The operating range shall be plus or minus 9.5 degrees engine deflection, minimum for yaw actuator and plus or minus 7.0 for pitch actuator. The total error over the required range of travel, and the specified temperature range will be no more than 2 percent of full scale. The output voltage shall be in phase (plus 5 degrees, minus 0 degrees) with excitation voltage when ram is extending, and 180 degrees (plus 5 degrees, minus 0 degrees) out of phase with excitation voltage when ram is retracting.

1.3.1.5.5.2 Gimbal Angular Rate Transducer. A gimbal angular rate transducer shall provide feedback to the SCS. This assembly shall have integrally packaged, two rate transducers providing voltage outputs proportional to actuator ram rates. Excitation voltage to the generator shall be 26 plus or minus 0.5 volts single phase ac at 400 plus or minus 4 cps. The nominal voltage output shall be 0.5 volts (rms) per inch per second of ram travel with the transducer loaded with a 15,000-ohm resistor. The residual voltage at zero speed shall be 0.020 volts (rms) maximum. Phase shift variation from unit to unit, and within a given unit, over the temperature range, and over the output range from 0.006 to 0.15 radian per second equivalent engine rate, shall not exceed plus 5, minus zero degrees. The output voltage shall be in phase with the excitation voltage when the ram is retracting (plus 5, minus zero degrees), and 180 degrees (plus 5, minus zero degrees) out of phase with the excitation voltage when the ram is extending. The signal-to-noise ratio shall be at least 100 to 1 at full output.

1.3.1.5.5.3 Pitch and Yaw Gimbal Actuator Drive Failure Indication. Electronic sensing of an over or under current condition at the various gimbal drive motors will be accomplished as a means of indicating actuator drive failure.

1.3.1.5.6 Electrical Characteristics. Electrical characteristics and power requirements are outlined in paragraph 1.3.1.9. SPS operational parameter instrumentation for CM display shall be provided as specified in Table 27.

1.3.1.6 Displays and Controls

The following CM control functions and displays shall be provided for the SPS.

1.3.1.6.1 Helium Pressurization Equipment. The helium pressurization equipment shall include the following:

1. Helium Tank - Pressure and Temperature Meter

2. Propellant Tank - Fuel Pressure Meter

3. Propellant Tank - Oxidizer Pressure Meter

Table 27. SPS Operational Parameter Instrumentation Command Module Display

Number	Parameter	Range		Units
		Low	High	
SP0001	Helium Pressurization Tank	0	5,000	PSIA
SP0002	Helium Temp Tank	-100	+200	Deg F
SP0003	Press. Oxid Tanks	0	+250	PSIA
SP0006	Press. Fuel Tanks	0	+250	PSIA
SP0009	Press. Main Valve Eng Oxid	0	+300	PSIA
SP0010	Press. Main Valve Eng Fuel	0	+300	PSIA
PS0011	Total Quantity Oxid	0	100%	%
SP0012	Total Quantity Fuel	0	100%	%
PS0020	Temp. Chamber High	0	+500	Event
SP0026	Position Fuel/Oxid Valve 1	0	+90	Deg
SP0027	Position Fuel/Oxid Valve 2	0	+90	Deg
SP0028	Position Fuel/Oxid Valve 3	0	+90	Deg
SP0029	Position Fuel/Oxid Valve 4	0	+90	Deg
SP0030	Helium Isolation Valve 1	Close	Open	Event
SP0031	Helium Isolation Valve 2	Close	Open	Event
SP0046	PU Valve Increase			Event
SP0047	PU Valve Decrease			Event
SP0640	Propellant Unbalance Oxid	-300	+300	Lb
SP0661	Press. Engine Chamber	0	150	PSIA
SP1000	Pitch Motor Fail			Event
SP1001	Yaw Motor Fail			Event
SP1002	SPS PU Sensor Fail			Event
	Engine Gimbal Pitch	-10	+10	Deg
	Engine Gimbal Yaw	-5	+13	Deg
SP0600	Eng VLV Act Sys Tank Press PRI	0	5,000	
SP0600	Eng VLV Act Sys Tank Press SEC		5,000	
	Flight Combustion Stab Mon			Event

4. Helium Isolation Valve ON-OFF Auto Switch (2)

5. Helium Isolation Valve OPEN-CLOSE Indication (2)

6. SPS Pressure (Tank Overpressure) Light (Caution and Warning)

1.3.1.6.2 <u>Propellant Supply and Distribution Equipment</u>.

1. Engine Inlet - Fuel Pressure Meter

2. Engine Inlet - Oxidizer Pressure Meter

1.3.1.6.3 <u>Propellant Utilization and Gaging Equipment</u>.

1. PU Valve - Oxidizer Flow Increase-Decrease Switch

2. PU Valve - Primary-Secondary Gate Selector Switch

3. PU Sensor - Primary-Auxiliary Switch for change from continuous sensing to point sensing

4. PU Auto-Test Self Test Switch

5. Oxidizer Quantity Meter

6. Fuel Quantity Meter

7. Unbalance Meter

8. Oxidizer Flow Increase Indication

9. Oxidizer Flow Decrease Indication

10. SPS Propellant Critical Unbalance Warning (Caution light and 1000 cps audible warning)

1.3.1.6.4 <u>Rocket Engine Assembly</u>.

1. Pilot Pre-valve ON-OFF Switch (1)

2. Thrust On Switch

3. Engine Valve OPEN-CLOSE Indication (4)

4. Thrust On Indication

5. Engine-Thrust Chamber Pressure Meter

6. SPS Wall Temperature High Light (Caution and Warning)

7. Primary Engine Valve Actuation System Tank Pressure Indication

8. Secondary Engine Valve Actuation System Tank Pressure Indication

1.3.1.6.5 <u>Engine Gimbal Assembly</u>.

1. Gimbal Actuator Motor ON-OFF Switch P1, P2, Y1, Y2

2. Gimbal Actuator Trim Thumb Wheels - Pitch and Yaw

3. Engine Pitch Gimbal Position Indication

4. Engine Yaw Gimbal Position Indication

5. Pitch Gimbal Drive Failure Light (Caution and Warning)

6. Yaw Gimbal Drive Failure Light (Caution and Warning)

1.3.1.7 SPS Operation

1.4.1.7.1 <u>Engine Start and Restart Operations</u>. A positive supply of propellants shall be maintained prior to and throughout the engine start and restart sequence. Operation of the SPS engine requires the following sequence of events when propellants are supplied to the engine as stipulated in paragraphs 1.3.1.2.2 through 1.3.1.2.2.2.

1.3.1.7.1.1 Flight Start Operations. Starting the SPS engine requires the following sequence of events:

<u>Action</u>	<u>Results</u>
Instrumentation energized	Five volts dc power supplied to temperature sensor and valve position indicators.
Gimbal actuators energized by activation of four gimbal motor switches in sequence and activation of control circuits.	Command signal supplied to clutches. Twenty-six volt ac, 400 cps signal supplied to rate and position transducers. Twenty-six volt dc power supplied to energize motors.
Pilot prevalve (two-way solenoid) energized by activation of prevalve switch.	Twenty-six volt dc power supplied to pilot prevalve.
Pilot prevalve (two-way) OPEN.	Pneumatic pressure supplied to pilot valve inlets.

Action	Result
Pilot valves (three-way solenoids) energized.	Twenty-six volt dc power supplied to pilot valves.
Pilot valves (three-way) OPEN.	Pneumatic pressure supplied to propellant valve actuators.
Valve actuator pistons and interconnect linkage moved to OPEN position.	Propellant valves move to OPEN position.
Propellants pass through valves and enter thrust chamber through injector.	Ignition established and thrust builds up to 90 percent of nominal value in +0.4 to +0.6 seconds.

1.3.1.7.1.2 Ground Start Operation. The sequence for ground starting and steady-state operation shall be as specified in paragraph 1.3.1.7.1.1.

1.3.1.7.1.3 Restart. The restarting sequence of operations shall be the same as that specified in paragraph 1.3.1.7.1.1.

1.3.1.7.2 Engine Shutdown Operations.

1.3.1.7.2.1 Flight Operation. Shutdown of the SPS engine requires the following sequence of events:

Action	Result
Pilot valves (three-way solenoids) deenergized.	Twenty-six volt dc power removed from pilot valves.
	Pneumatic pressure in actuators vented through pilot valves (three-way).
Valve actuator pistons and connect linkage move to position.	Propellant valves move to closed position.
Propellant flow to thrust chamber stops.	Thrust decays as the propellant downstream of the propellant valve is exhausted (thrust decays to 10 percent nominal in 0.4 to 0.6 seconds).
Pilot prevalve (two-way solenoid) deenergized by deactivation of pilot prevalve switch.	Twenty-six volt dc power removed.

Action	Results
Pilot prevalve closes.	Pneumatic supply pressure shutoff to pilot valve inlets.
Gimbal actuators deenergized by deactivation of motor switches and control circuits.	All power removed from gimbal actuators.
Instrumentation deenergized.	Power removed from instrumentation.

1.3.1.7.2.2 Ground Operation. The sequence for ground shutdown shall be in accordance with paragraph 1.3.1.7.2.1, except that power control functions shall be accomplished by GSE.

1.3.1.8 Ground Support Equipment

The SPS will interface with GSE in various areas involving the engine assembly such as handling and checkout equipment, the propellant utilization and gaging assembly, the helium fill and drain coupling assemblies, test point coupling assemblies, and gimbal assembly.

1.3.1.9 Electrical Requirements

The SPS lunar mission electric power requirements (except for instrumentation, displays, and controls) as defined for design purposes are shown in Table 28.

1.3.1.9.1 Dielectric Strength. Unless otherwise specified, all components of the subsystem shall withstand a voltage of 1050 volts ac at a frequency of 50 to 60 cps for a period of 1 minute at sea level and 550 volts ac at altitudes of 70,000 feet to an equivalent altitude of 10^{-6}mm Hg, without evidence of insulation breakdown, flashover, current leakage in excess of 2 milliamperes, and other deterioration which results in subsequent component malfunction, when applied for 1 minute across each external termination and the component housing and between all external terminations except those connected internally within the component.

1.3.1.9.2 Combustion Chamber Sensor. The combustion chamber skin temperature sensor and the propellant valve position transducer shall withstand without electrical breakdown a potential of 1000 volts dc (rms) at sea level and 200 volts ac (rms) at 70,000 feet to an equivalent altitude of 10^{-6}mm Hg.

1.3.1.9.3 Gimbal Actuator. Except for deviations noted herein, the gimbal actuator shall not exhibit evidence of arc-over, insulation breakdown, or current flow in excess of 2 ma at an internal pressure of 2 psia maximum when subjected to a potential of 550 volts ac (rms) for 1 minute at a frequency of 50 to 60 cps applied between ungrounded pins and between ungrounded circuits and case. The motor circuits shall meet this requirement with all capacitors disconnected. The motor circuit capacitors, while under a pressure of 2 psia maximum, shall withstand twice the peak voltage to which they

Table 28. SPS Design Values AC and DC Electrical Power Requirements by Equipment

TOTAL AC POWER REQUIREMENTS			
Equipment	Power Input	Hours ON (Prelaunch to Touchdown)	Watt-Hours
PU Gaging Subsystem	69.5 W (incl. valves)	0.174	12.1 (1)
Total AC Power Required - 12.09 Watt-Hours			
TOTAL DC POWER REQUIREMENTS			
Equipment	Power Input	Hours ON (Prelaunch to Touchdown)	Watt-Hours
PU Gaging Subsystem	19.6 W (incl. valves)	0.174	3.4
Helium Solenoid Valves	125.0 W	0.174	21.7
Gimbal Actuator Motors	1348.0 W (2) 1825.0 W (3)	1.22 (2)(4) 1.74	1640.0 (2) 317.0 (3)
Two Prevalves	70.2 W	1.22 (4)	85.5
Pilot Valve	140.0 W (SCS)	0.174	24.3
Actuator Clutches - Activated by Milliampere Signals from SCS			
Total DC Power Required - 2092 Watt-Hours			

Notes:
(1) Static inverter efficiency not included in power factors.
(2) Idle condition (1 minute prior to Delta V) and launch condition.
(3) Delta V condition.
(4) Assumes launch duration of 0.35 hours and 26 starts.

will be subjected during service or 100 volts ac (rms), whichever is greater, for a period of 1 minute. Testing of capacitors and motor circuits may be performed prior to installation into the actuator.

1.3.1.9.4 Semiconductor Dielectric Strength. Component circuitry employing semiconductors shall show no evidence of insulation breakdown, and flashover between any terminal and the component housing, when subjected to 500 volts at 50 to 60 cps.

1.3.1.9.5 Insulation Resistance. Each component of the subsystem shall have an insulation resistance of at least 100 megohms between any nonconnected pair of terminals and between the component housing and any terminal, when subjected to a potential of 500 volts dc for a period of 1 minute.

1.3.1.9.6 Grounding. No components of the subsystem shall be internally grounded.

1.3.1.9.7 Connectors. Where electrical wiring from the propellant sensing devices pierce a tank, electrical connectors shall be designed to provide dual sealing surfaces or a sealing method of equivalent reliability.

1.3.1.9.8 Electrical Connections. All components of the subsystem that utilize a terminal block or stud feed-through shall comply with the following requirements:

1.3.1.9.8.1 Terminal Blocks. Component terminal blocks shall consist of terminal studs, hardware, and a nonconductive, semirigid insulated cover which is impervious to and compatible with the fluids to which the component is normally exposed. Insulating barriers shall be provided between terminals. The terminal blocks shall be permanently and legibly marked. The markings shall be legible with the wiring installed and the terminal cover removed. The electrical circuit shall not depend on the conductivity of the nuts, washers and other hardware, or the stud to complete the current path. The electrical connection shall not depend upon rubber or plastic in compression for tightness. The terminal block stud shall incorporate a pad or base to bear directly against the connecting wire terminals.

1.3.1.9.8.2 Stud Feed-Through. Stud feed-through connections shall be made of materials compatible with sealing and current requirements. Hardware for completing electrical connections shall be provided with the component.

APPENDIX B

TANK SIZING CALCULATION METHOD

In this appendix a typical calculation is presented showing the method used to size the pressurization and propellant systems for the various AES tank configurations considered during this study.

Block II data was used, where applicable, in the preliminary calculations presented.

It is to be noted that the calculation method presented results in approximate but conservative solutions. During the next study, a rigorous analysis will be accomplished for the configurations chosen for the AES missions.

TYPICAL PROPELLANT/PRESSURIZATION SYSTEM CALCULATION

Example: 10,000 lb usable propellant loading
1.6 mixture ratio
75°F loading temperature
Specific gravity @ 75°F A-50 = 0.860, N_2O_4 = 1.436

Tank Volume Required

	A-50		N_2O_4	
	lbs	ft^3	lbs	ft^3
Usable propellant	3850	71.6	6150	68.5

Residuals in Tanks

	A-50	N_2O_4
Estimate of empty tank vapor = $\left(\frac{\text{Vapor Block II}}{\text{propellant Wt Blk II}}\right) \times \left(\text{propellant Wt AES}\right)$ =	1.5	18.5
Retention reservoir inside pull-thru (Blk II)	27	43
Retention reservoir outside pull-thru (Blk II)	29	53
Gaging system tolerance (Blk II)	55.6	55.5
GSE loading tolerance (assumed limit)	50	50

Total Residuals in Tanks	163	3.0	220	2.5
Estimate of minimum ullage required		6.7		6.6
Unpressurized Total Tank Volume		81.3		77.6

Helium Requirement

Ullage pressure (Blk II) A-50 177.5 psi
N_2O_4 175.9 psi

Empty tank temperature (estimate) 80°F

Empty tank ullage volume:

(V usable +V initial ullage) x (1.02)(factor for tank pressure and temperature expansion) = A-50 = 79.8 ft^3

N_2O_4 = 76.6 ft^3

Weight of helium in empty tanks:

He compressibility, Z = 1.01 @ 175 psi

$$\text{Wt He} = \frac{(P_{He} - P_{prop.\ vapor})}{ZRT}$$

A-50 tank = 9.3 lb He

N_2O_4 tank = 7.9 lb He

Total He in empty propellant tanks = 17.2 lbs

Block II Helium bottle volume = 19.6 ft^3

Initial Helium Temperature = 80°F

Final He bottle pressure required for regulation (Blk II) = 395 psia

The basic equation for the helium weight and pressure required is given by:

1) $\frac{P_1 V}{WT_1} = Z_1 RT$, $\frac{P_1}{WT_1} = 79 \frac{PSI}{\text{lbs He loaded}}$

where the subscript 1 indicates initial fill values.

The second equation needed is similar:

2) $$\frac{P_F V}{WT_1 - WT \text{ in empty tanks}} = Z_F R T_F \quad , \quad WT_1 = \frac{2830}{T_F} + 17.2$$

where the subscript F indicates final values.

Thus there are two equations and three unknowns. It is then assumed that the helium will expand in the bottle isentropically.

3) $$T_F = T_1 \left(\frac{P_F}{P_1} \right)^{\frac{\gamma - 1}{\gamma}}$$

In addition, heat is added to the helium during coast from the bottle walls. It was assumed that the only heat to be transferred to the helium was from the bottle walls:

4) $$Q = C_P \text{ WT Bottle } \Delta T \qquad \text{(BTU)}$$

Solving equations (1), (2), and (3) gives the first approximation of the helium total weight, final temperature, and loading pressure.

WT He = 26.6#

T_F = 300°R

P_1 = 2100 psia

Since it was assumed that the helium bottle temperature was 540°F initially, thus maximum possible ΔT is 240°F. The total heat available for transfer to the helium is obtained from equation (4).

Q = 1020 BTU maximum

If it is assumed that the heat stored in the bottle wall is given up uniformly as the helium leaves the bottle, then the final helium temperature will be approximately 25°F higher than the isentropic value without heat transfer from the bottle wall. Several iterations were necessary to converge this solution sufficiently. The final results are:

Initial temperature	80°F
Initial pressure	2300 psi
Helium weight	29 lbs
Final pressure	395 psi
Final temperature	325°R

As can be seen from the above discussion, the information generated by these calculations is approximate, but conservative. During future studies, these calculations will be redone using available computer programs and techniques.

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