NASA TECHNICAL NOTE

NASA TN D-7375

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APOLLO EXPERIENCE REPORT - SERVICE PROPULSION SUBSYSTEM

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • AUGUST 1973

1. Report No. NASA TN D-7375	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle APOLLO EXPERIENCE REPORT SERVICE PROPULSION SUBSYSTEM		5. Report Date August 1973 6. Performing Organization Code
7. Author(s) Cecil R. Gibson and James A. Wood, JSC		8. Performing Organization Report No. JSC S-378 10. Work Unit No. 914-13-00-00-72
9. Performing Organization Name and Address Lyndon B. Johnson Space Center Houston, Texas 77058		11. Contract or Grant No. 13. Type of Report and Period Covered Technical Note
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

The JSC Director waived the use of the International System of Units (SI) for this Apollo Experience Report because, in his judgment, the use of SI Units would impair the usefulness of the report or result in excessive cost.

16. Abstract

The significant service propulsion subsystem development, qualification, and flight experience from the early portion of the Apollo Program through the first lunar-landing mission is presented. Particular emphasis is given to problems encountered and solutions used to eliminate the problems.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
*Apollo Project *Command Service Modules *Spacecraft Propulsion *Reliability Control *Propellants	

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
None	None	25	$3.00

* For sale by the National Technical Information Service, Springfield, Virginia 22151

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APOLLO EXPERIENCE REPORT

SERVICE PROPULSION SUBSYSTEM

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SUMMARY

A review of the 9-year development program of the Apollo service propulsion subsystem from the initial concept to the Apollo 11 lunar landing is given in this report. During late 1960, separate contractors, under the supervision of the National Aeronautics and Space Administration, prepared feasibility studies of an advanced manned spacecraft program. In 1961, the lunar-orbit-rendezvous mode was selected to replace the direct-lunar-landing mode. By 1967, development and qualification of the service propulsion subsystem were completed. The continued refinements and the evolution of the service propulsion subsystem and its related assemblies were prime factors in the successful manned flights of the Apollo Program.

INTRODUCTION

This review of the Apollo service propulsion subsystem (SPS) covers the 9-year period from the conception of the subsystem to the Apollo 11 lunar landing. During this period, the program progressed from the definition phase to the hardware design phase, to the subsystem development and qualification phases, and, ultimately, to the flight phase. Because of the size of the subsystem, it was not feasible to provide total redundancy. For this reason, the SPS had to be extremely reliable. Simplicity of design and extensive ground testing were required in order to achieve the desired confidence level. Several problems were encountered during the development and qualification phases. These problems, and the actions taken to eliminate them, are discussed in the description of this phase of the program.

In October 1960, the National Aeronautics and Space Administration (NASA) selected three contractors to prepare feasibility studies of an advanced manned spacecraft as a part of the Apollo Program. Several types of propulsion systems, cryogenic propellants, storable propellants, and solid-propellant rocket motors were studied to support the various proposed Apollo configurations. By early 1962, the basic Apollo configuration and the engine subcontractor had been selected.

The evolution of the subsystem began with the definition of design criteria that would be consistent with mission requirements, reliability goals, and spacecraft-design concepts. The development and qualification of the subsystem were accomplished by

developing and testing components, by testing major assemblies, and by full-scale system testing on propulsion test fixtures. Detailed test objectives to verify inflight system performance also were incorporated in the early Apollo missions.

Analyses that encompassed conceptual studies, weight trade-offs, configuration choice, size, and establishment of performance criteria resulted in a final design definition for the subsystem. Materials and processes were investigated to support subsystem development. In connection with the analytical studies, laboratory research was conducted to verify analytical techniques, to improve component design, and to resolve problems. Material-properties research was conducted to determine the emissivities of nozzle and nozzle-coating materials. In addition, nozzle-material welding techniques were investigated. Tube brazing and weld techniques were improved by means of propellant-metal compatibility studies and brazing-welding metallurgical investigations. Thrust-chamber ablative materials were selected after the performance of laboratory tests that limited the materials list before any thrust-chamber testing. Laboratory studies were conducted on 42 potential thrust-chamber-material samples; the studies included high-temperature vacuum tests and thermal- and structural-properties investigations.

Seal materials for propellant equipment were selected after investigation of elastomer and pseudoelastomer compounds to screen for propellant compatibility, swell, creep, resilience, and other required seal properties. Zero-gravity propellant-motion problems were investigated by means of theoretical and experimental research in fluid mechanics. The goals of this research were new modeling and scaling techniques for earth simulation of zero-gravity effects and an improvement in the understanding of fundamental phenomena. The simulation techniques and facilities that were used were the prime contractor drop tower (low gravity), scaled transparent tanks (one g) with slow-motion picture simulation of inflight real time, and the U.S. Air Force KC-135 airplane flying laboratory.

The command and service module (CSM) flight program was the final phase of the SPS development. Inflight testing of the SPS was accomplished in sequence with the vehicles to produce a high-confidence, proven subsystem for the lunar-orbital-rendezvous mission. Because the SPS played a major role in the achievement of crew safety and mission success, an attempt to qualify the subsystem under all space-operational conditions was made during the ground-test program. However, limitations in the ground-test facilities; limited zero-gravity periods in earthbound test vehicles; the impracticability of simulating the combined pressure, temperature, and gravity environment; and environmental unknowns prevented complete demonstration of subsystem performance conditions before and during the early unmanned flights. Thus, the SPS was used conservatively early in the flight program, but the complexity of operating modes and of subsystem demands was increased with each flight as experience and confidence were obtained.

DESIGN PHASE

Design reliability was a product of simplicity. Propellants were selected on the basis of experience with other programs and because of propellant earth-storability,

hypergolicity, performance, and high-density properties. A pressure-fed engine, which eliminated the need for the parts and controls that are required for pump-fed engines, was selected to decrease the operating complexity. High-pressure (4400 psia) helium, stored at ambient temperature and regulated to 180 psia, was selected as the propellant-tank pressurizing agent. An ablative thrust chamber was selected rather than a regeneratively cooled chamber to decrease the possibility of propellant freezing. Also, an increase in the operational limits for the propellant ratio, the propellant temperatures, and the chamber pressure was more compatible with ablative chambers. In addition, studies were indicative that most propulsion-system failures were caused by failures of controls, valves, and solenoids rather than by failures of injectors or thrust chambers. Therefore, redundancy was used, where practical, to increase reliability in these areas.

Approximately halfway through the SPS development program, data generated from an in-house-managed research and development contract were indicative that the SPS could supply a specific impulse of 3 to 5 seconds higher if the oxidizer-to-fuel weight ratio was 1.6:1 rather than 2:1. The higher performance level was desirable to provide for the lunar module (LM) weight growth and for additional flexibility in mission planning. Therefore, the SPS was redesigned for the new ratio. The 2:1 ratio was used in the Block I vehicles. The vehicles using the 1.6:1 ratio were designated Block II.

Block I Configuration

The SPS Block I mission and operational requirements dictated that the subsystem consist of a helium-pressurization assembly, a propellant-supply and propellant-distribution assembly, a propellant-utilization and propellant-gaging assembly, a rocket-engine assembly, instrumentation, and displays and controls. The SPS arrangement is shown in figure 1.

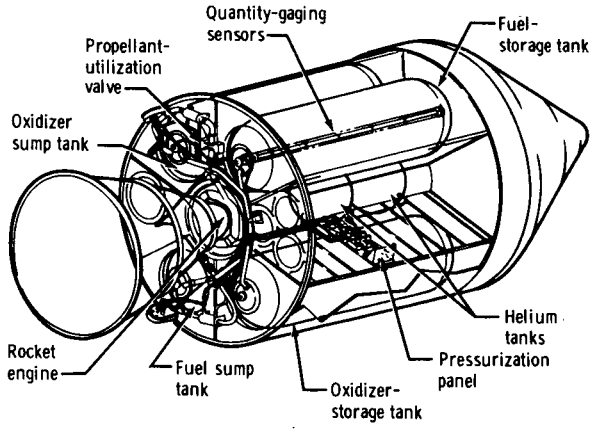

Figure 1.- Service propulsion subsystem.

Helium-pressurization assembly.- The SPS helium-pressurization assembly (fig. 2) consisted of a high-pressure helium supply; two spherical helium tanks; and the associated pressure regulators, isolation valves, check valves, and pressure-relief valves. The tanks were connected to a common helium-distribution line. Two parallel solenoid-operated poppet valves, installed downstream from the helium tanks, isolated the helium supply when the propulsion system was inoperative. Pressure was regulated by two pressure-regulator assemblies. Each regulator assembly incorporated a primary and a secondary regulator in series. The primary regulator reduced the high upstream pressure to the downstream operating pressure. The secondary

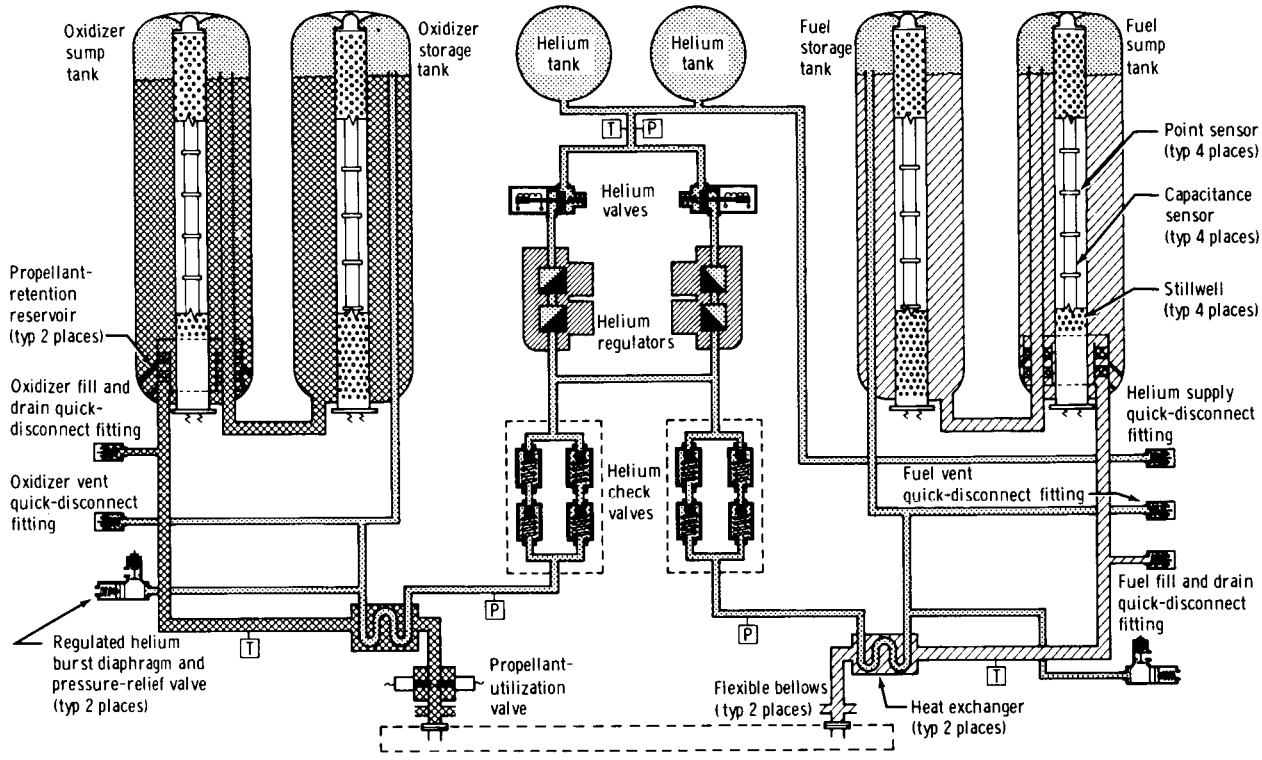

Figure 2.- Service propulsion subsystem propellant feed assembly.

regulator was calibrated to regulate at a higher pressure; thus, the secondary regulator remained open when the primary regulator functioned properly. The installation of parallel-series check-valve assemblies downstream from the helium-pressure regulators prevented backflow of propellants into the pressurization system. An oxidizer- and a fuel-pressure-relief-valve assembly (each consisting of a relief valve, a burst diaphragm, and a filter screen) were provided for the propellant-tank systems. The burst diaphragms were used to isolate the valve seats from the propellant vapors.

Propellant supply and distribution assembly.- The propellant supply and distribution assembly (fig. 2) contained and distributed oxidizer and fuel. The oxidizer supply of the assembly was contained within two cylindrical tanks that had hemispherical domes. The two tanks were connected in series by means of a propellant-transfer line. The upstream tank was used as the storage tank; the downstream tank was used as the sump tank. The oxidizer tanks contained 30 000 pounds of usable propellant, and the fuel tanks contained 15 000 pounds of usable propellant after allowances were made for the loading tolerances, the residuals, and the required ullage. A propellant/helium heat exchanger was incorporated in each propellant line for thermal conditioning of the helium supply.

Titanium was determined to be the best pressure-tank material because of its high strength-to-density ratio and its compatibility with the propellants. Initially, the gross weight of the tanks was 49 850 pounds; the gross weight was reduced to 39 500 pounds in July 1962 because of the change to the lunar-orbit-rendezvous mode,

which required less propellant and tankage. The maximum tank pressure was 240 psig, and the operating-temperature limits were 104° F maximum and 44° F minimum. Zero-gravity expulsion techniques that were studied included the use of mechanical bellows, an umbrella spring-loaded bladder, full bladders, and reaction-control-engine ullage maneuvers. The reaction-control-engine ullage maneuver was selected to settle the propellants for expulsion. However, this technique necessitated the design and development of the propellant-retention reservoir in the sump tanks as a backup to the reaction control system (RCS).

Propellant utilization and gaging system. - The propellant utilization and gaging system (PUGS) consisted of primary and auxiliary propellant-quantity-sensing devices, an electrical control unit, an oxidizer-flow-control-valve assembly, and a crew display panel. The system was used to monitor the quantities of usable propellant that remained in the propellant tanks so that the desired oxidizer-to-fuel ratio could be adjusted manually during propellant expulsion for simultaneous depletion of the oxidizer and fuel. The primary quantity sensors were cylindrical capacitance probes that were mounted axially in each tank. The auxiliary gaging system had impedance-type point sensors coupled with a nominal-flow integrator between sensor levels. Oxidizer flow was controlled by means of a motor-operated, redundant, double-blade valve assembly that was used to provide increased, decreased, or normal oxidizer flow rates. The control unit was used to compute total propellant quantities from individual tank quantities, the propellant imbalance, and the oxidizer-to-fuel ratio; also, the control unit continuously compared the total propellant quantities that were indicated by the primary and auxiliary systems. The crew display panel provided the onboard output indications that were required and provided the switches for use in control functions and onboard testing.

Engine assembly. - The SPS engine (fig. 3) was a nonthrottleable, gimbaled, pressure-fed rocket engine that consisted of an ablative-cooled thrust chamber, a radiation-cooled nozzle (extending from an area ratio of approximately 6:1 to 62.5:1), a bolt-on aluminum injector, a thrust and gimbal mount, a bipropellant valve, gimbal actuators, an electrical harness, and propellant feedlines. The engine operated at a chamber pressure of 102 psia and produced a vacuum thrust of 21 500 pounds. The average specific impulse was 309 seconds. The engine was capable of at least 36 starts and had an engine firing life of 500 seconds.

Ignition occurred by means of hypergolic reaction in the thrust chamber. Propellant flow to the thrust-chamber assembly was controlled by a redundant set of series-parallel ball valves that were actuated by pneumatic pressure that was controlled by electrically operated solenoid valves. Gaseous nitrogen, stored in redundant tanks, provided the pneumatic pressure.

A flight combustion-stability monitor (FCSM) system was designed to monitor SPS engine vibrations and automatically commanded engine shutdown when unacceptable combustion-excited vibration conditions occurred. This system contained engine-mounted accelerometers and associated monitor circuits and an electronic summing circuit, which acted to shut down the engine. An indicator lamp in the command module (CM) was illuminated when a shutdown was commanded. Manual control allowed overriding the automatic system to restart the engine. A combustion-stability verification-test program was conducted to show the inherent stability characteristics

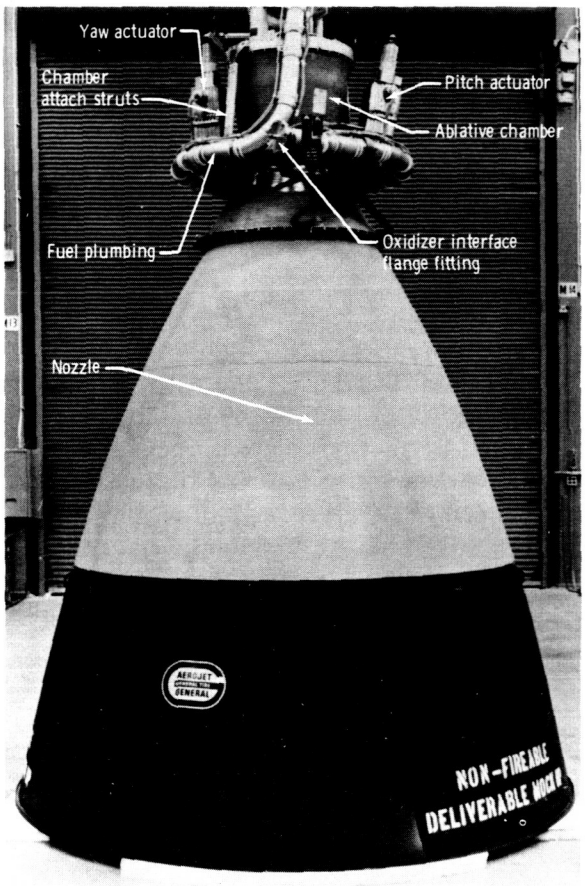

Figure 3.- Service propulsion subsystem engine assembly.

of the Block II engine configuration. Successful completion of this test program allowed deletion of the FCSM.

Block II Configuration

The two major changes in the Block II SPS were the establishment of the operating propellant ratio as 1.6 pounds of oxidizer per pound of fuel, to provide higher engine specific impulse, and the reduction of the total onboard propellant quantity as a result of vehicle-trajectory changes combined with the improvement in engine performance. The changes facilitated making the propellant tanks smaller; and, because the propellant-density ratio was 1.6:1 (oxidizer-to-fuel), the oxidizer and fuel tanks were made identical in size. The cylindrical section of all four tanks was shortened by approximately 11 inches. The Block I diameter and the hemispherical-head design were maintained. An additional advantage of the redesigned tanks was realized: the sump and storage tanks were installed in adjacent bays in the service module (SM), rather than on opposite sides (as in Block I), for a more desirable location of the center of gravity. The Block II propellant tanks also were designed to have a limit pressure of 225 psia, which was a reduction from the Block I value of 240 psia. This change facilitated further reduction of the tank wall thicknesses. To reduce the tank limit pressure to 225 psia and yet maintain the Block I permissible ullage-pressure-rise design limit of 213 psia, a narrow-range relief device was developed for use on Block II spacecraft.

Because less propellant was expelled by the Block II pressurization equipment, less helium was required. The design loading pressure was reduced to a value that corresponded to the new helium quantity, and the wall thickness was reduced to that thickness which was required for the lower loading pressure; thus, an additional weight saving was achieved.

DEVELOPMENT AND QUALIFICATION PHASE

The time-sequenced planning and results that were used in the testing of each major component and assembly of the subsystem are discussed in this section.

Component Development and Qualification

Component development was conducted at the plant of each subcontractor. Component compliance with design criteria and the determination of optimum component configurations were the purposes of the effort.

For most components, the flight phase started before the development and qualification phase was complete. This fact made it necessary to qualify hardware specifically for the first flight and to delay the more general overstress-type tests until final hardware designs were completed. A chronological description of the development and qualification of each component in the engine assembly, propellant-supply assembly, and pressurization assembly is given in the following sections.

Engine assembly. - The SPS engine (fig. 3) design and development effort was started by the subcontractor April 9, 1962. The effort that was contracted originally covered the design and development of the SPS engine, preliminary flight-rating tests, and the delivery of two mockups, five prequalified engines, seven qualified engines, spare parts to support the delivered engines, and some ground-support equipment (GSE).

The initial design effort was concentrated on layouts of the overall engine concept and on the interfaces with the spacecraft. By August 1962, the design definition had progressed sufficiently to permit the start of fabrication of the hard mockup engine components. The design review of this mockup was completed, and the mockup was delivered in November 1962.

Throughout the engine development and qualification phase, many configuration changes occurred in both the Block I and Block II engines as a result of knowledge gained in the test programs. Engine-development testing was conducted at the contractor sea-level test facility at Sacramento, California, and at the simulated-altitude test facility at the Arnold Engineering and Development Center (AEDC), Tullahoma, Tennessee. One of the more significant changes resulted in the incorporation of a baffled injector to reduce the risk of combustion instability. Both baffled and unbaffled injectors were used in the development program. The baffled injector is shown in figure 4. The requirements for the engine to damp pressure oscillations and the poor test experience with unbaffled injectors necessitated the use of a baffle to damp pressure oscillations in the combustion chamber. Many tests included the use of explosive charges in the combustion chamber during engine firings to verify the damping capability of the engine.

The Block I test program, performed at the contractor facility, consisted of extensive firing of the injector as a component and firing in conjunction with other engine components. The sea-level qualification-test program consisted of firing the engine assembly 56 times. The altitude testing at the AEDC consisted of three test phases: development, prequalification, and qualification testing.

The first fireable test engine (SN 003) was shipped to the AEDC in May 1963 for initiation of the phase I program. Two additional SPS engines (SN 004 and SN 008) were shipped to the AEDC in September and November 1963. Simulated-altitude testing began with engine SN 003 on June 26, 1963. An engine firing of approximately 5 seconds resulted in shutdown because of the collapse of the titanium nozzle extension, which was

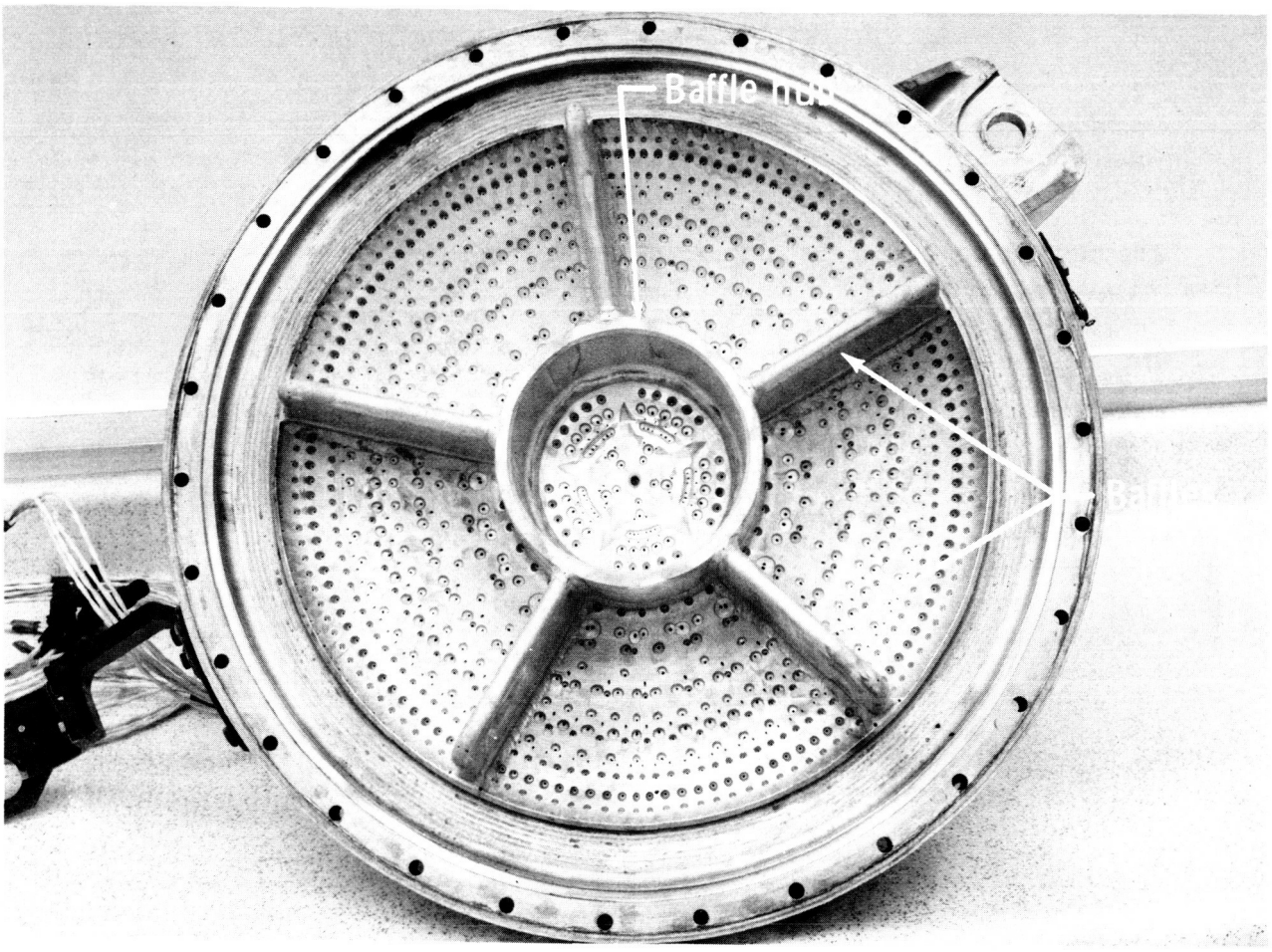

Figure 4. - Service propulsion subsystem engine baffled injector.

caused by excessive back pressure in the test cell. High-frequency chamber-pressure oscillations, which occurred during the acceptance testing of engine SN 004 at the subcontractor plant, were attributed to incomplete air removal from the engine oxidizer circuit. To ensure removal of all gas from the oxidizer circuit, the engine-air-bleed system on the engines was increased in size from 0.25 to 0.50 inch, and the required air-bleed times were lengthened for the phase I testing.

Two engines (SN 009 and SN 011) were used for the phase II prequalification testing at the AEDC. The first altitude-test-cell firing involved engine SN 009; the test was completed in December 1964. In January 1965, because of inadvertent operation of the test stand during stand maintenance, the engine sustained nozzle and chamber damage. No corrective action regarding engine design or operation was required because the damage was attributed to test-stand problems. The second engine (SN 011) was equipped with a baffled injector that was designed to minimize combustion instability. By June 1965, the two engines had completed 101 tests, for a total firing time of 2581 seconds. The tests were used to evaluate engine operation and performance over a wide range of chamber pressures and propellant mixtures. Simulated-altitude gimbaling of the engine was accomplished in March 1965 on engine SN 009 and was repeated successfully on engine SN 011 in April 1965.

Engine SN 009, retrofitted with the first pneumatically actuated valve, was tested at simulated altitudes; the engine completed 27 tests (408 seconds of firing time) in June 1965. The chamber forward flange failed during the last test, which resulted in the loss of the chamber and nozzle extension. The chamber-flange failure occurred again on the second engine (SN 011) in July 1965. The failure occurred on the 27th test after 333 seconds of firing time. Both engines sustained extensive damage and were returned to the subcontractor. The failures were caused by shrinkage of the chamber ablative liner; shrinkage occurred during cooldown after exposure to hot firing conditions. This shrinkage caused the opening of a gap at the chamber-to-injector flange joint, allowing circulation of the combustion gasses in the joint, subsequent charring of the chamber, and overheating of the chamber-to-injector flange. Several firing cycles with the associated heating and cooldown of the chamber and flange joint caused degradation of the joint to the point of failure by means of separation. The chamber was redesigned by step machining the outer liner surface and by adding a mechanical lockring. This eliminated the susceptibility to thermal cycling. Burn-time and coast-time limitations were imposed on flight engines that were not retrofitted with the new chamber.

Three engines completed the phase III simulated-altitude qualification testing; 130 firings were made for an accumulated duration of 3599 seconds. During the first test series of the third engine, severe chamber damage occurred that was attributed to vacuum-grease contamination of the fuel. This contamination caused overheating and warpage of the injector flange, which resulted in burn through at the injector-flange-to-chamber joint on the second test series. The injector was replaced, and the last two test series were completed satisfactorily. The contamination problem was eliminated through cleanliness controls during subsequent testing.

Concurrent with the phase I to III AEDC test program, extensive development testing was accomplished at the subcontractor sea-level test facility to establish compatibility of engine components and for selection of the individual components to be used. Engine SN 023 completed acceptance testing in May 1965, and engine SN 021 completed acceptance testing in July 1965. As a result of the chamber burn-through problem at the AEDC, the engines were retrofitted with the redesigned chamber. Qualification testing was resumed, and the first sea-level qualification-test series was completed on engine SN 022 in November 1965. Two additional test series were completed by December 28, 1965; this marked the satisfactory completion of the sea-level qualification testing. A total of 56 tests was performed, representing 1518 seconds of firing time.

Block I flight engines, which were of the same configuration as the qualification engines, were processed through a standard test cycle that consisted of component acceptance tests, engine assembly acceptance firing, and postfire testing.

The Block II engine-test program consisted of 392 development-test firings at the subcontractor sea-level facility and three test phases performed under simulated-altitude conditions (a total of 663 firings) at the AEDC. In May 1966, the first Block II development testing under simulated altitude was conducted (phase IV). Forty-nine tests were performed, and 810 seconds of firing time were accumulated. Phase IV testing involved two engines; 265 tests were performed, and 6704 seconds of firing time were accumulated.

Phase V, which was simulated-altitude qualification testing that consisted of six test series, began November 18, 1966, and was completed in February 1967. A second engine was used for the last two series; 108 tests were performed and 4521 seconds of firing time were accumulated without unscheduled interruptions. Engine operation was evaluated over the extreme range of thrust-chamber pressures, propellant ratios, firing durations, and propellant temperatures. One significant problem was encountered: leakage of the ball-valve seals was noted after testing. Subsequently, the seals were redesigned to provide a second seal for each ball, and the seal material was changed from TFE Teflon to a glass-filled Teflon (BF-1 Blue Teflon) seal.

In addition to the ball-valve seal-leakage problem that was encountered during phase V testing, overboard leakage of gaseous nitrogen through the actuator piston was possible because of shrinkage of the Delrin pistons at temperatures below 30° F. To eliminate this problem, the piston material was changed to aluminum. The phase VI testing, which consisted of 222 firings, was designed to qualify the design changes that were made to eliminate these problems and to evaluate the magnitude of chamber-pressure overshoots by the use of high-response instrumentation.

The ball-valve seal-leakage problem was not eliminated completely by the redesign, but the resulting total leakage rate was determined to be acceptable for flight use. The Delrin piston had been replaced with an aluminum piston and a Delrin sliding surface; the replacement was satisfactory. The chamber-pressure-overshoot evaluation was indicative that excessive pressure spikes may result from igniting the engine by opening both the redundant flow paths simultaneously. A revised operating requirement, the use of only one flow path to start the engine, was incorporated for flight use. The primary basis for qualification of the engine was the simulated-altitude testing conducted at the AEDC. The tests accomplished on the final (Block II) configuration are summarized in table I. The maturity and reliability of the engine design were established through the number of firings conducted and the conditions simulated.

Propellant utilization and gaging system. - During the development and qualification testing of the PUGS, several design changes were needed to eliminate discrepancies. However, other problems associated with the system also resulted from interactions between the PUGS and the SPS tanking arrangement.

In the first generation of control units, the connector panel was integrated with the control-unit housing. Thus, technicians had to solder approximately 400 wires in a limited space. Solder inspection was difficult because only one row of solder cups per connector could be seen at any one time and some solder cups had more than one wire. To eliminate this problem, the control unit was redesigned and the connector panel was separated from the housing. Therefore, the connectors were wired in a separate subassembly, allowing adequate inspection.

The original design for overload protection involved the use of fuses packaged in the modules. This method was costly because the module had to be unwired, depotted, repotted, and reinstalled when a fuse was blown. As a result, a design change was made so that all fuses were relocated on an accessible terminal board. This reduced the time and cost required to repair units with blown fuses.

Several other operating problems were noted during subsystem development testing at the NASA White Sands Test Facility (WSTF) and during the flights. A propellant

TABLE I. - RESULTS OF BLOCK II SPS ENGINE ALTITUDE TEST PROGRAMS AT AEDC

Date	Objectives	Numbers of tests	Firing time, sec	Test conditions			Remarks
				Mixture ratio	Temperature, °F	Chamber pressure, P_C, psia	
Apr. to Oct. 1966	Engine development Evaluate SPS engine performance at nominal and off-nominal conditions	261	6695	1.4 -1.8	70 -110	77 120	Objectives successfully met; no hardware problems
Nov. 1966 to Feb. 1967	Engine qualification Qualify engine at nominal and off-nominal conditions	180	4523	1.4 -1.8	35 -110	79 -115	Engine qualified to specification requirements except for the bipropellant valve
Feb. to Apr. 1968	Delta qualification Model 1-C bipropellant-valve configuration	116	863	1.6	30 -115	96 -117	Model 1-C valve failed because of excessive ball-seal and shaft-seal leaks
May 1968	Model 1-D bipropellant-valve configuration	20	139	1.6	25 -35	100	Valve certified for earth-orbital missions; slow operation at low temperature because of actuator leakage
Sept. 1968	Model 1-E bipropellant-valve configuration	31	782	1.6	30 -115	96 -117	Valve qualified to specification requirements
June 1968	Research and development Define magnitude and duration of engine-start transient overshoot	54	62	1.6	30 -70	96 -117	Overshoot magnitude for dual-bore starts was 25 to 40 percent of steady-state P_C; for single-bore starts, overshoot magnitude was 5 to 25 percent
May to Sept. 1968	Define engine-restart limits	62	25.2	1.6	30 -65	99	Minimum coast time = 60 sec, minimum firing time = 0.5 sec, some hand restarts at 0.3 sec; no damage

bias that resulted in off-nominal fuel flow was detected by means of the PUGS during the early flights. Initially, it was believed to be a gaging problem, but later it was established as an increased fuel-flow rate over the rate that was predicted based on ground tests. This phenomenon could not be reproduced in helium-saturation ground tests. The propellant-utilization (PU) valve was used in the increased oxidizer flow position to eliminate most of the flow imbalance until engine reorificing eliminated the bias.

Another bias in both fuel and oxidizer flow was observed on the spacecraft 001 (SC 001) tests at the WSTF and on early flights. The propellant level in the cylindrical tube, which houses the gaging probes in the sump tanks, was lower under flow conditions than was the level of the bulk propellant in the tanks. This inequality resulted from a Bernoulli effect that was caused by propellant flowing out of the bottom of the tank. Flow dividers were added in the retention reservoir to eliminate this problem.

Two other errors associated with the PUGS have been identified and are accounted for in preflight predictions of propellant usage. One error resulted from absorption of helium from the ullage of the sump tank, which allowed the propellant level to rise above the top of the gage. This error was compensated for operationally. The other error resulted from an offset calibration of the oxidizer-storage-tank probe, which was necessary to eliminate a residual signal from the empty storage tank. The probe offset calibration is incorporated during gaging system checkout prior to servicing. This error is also compensated for operationally.

Flight experience also was indicative that approximately 25 seconds of engine operation are required to settle propellants in the gaging probe housings. This circumstance caused activation of the caution and warning system, indicating a critically unbalanced condition between the remaining fuel and oxidizer quantities during the Apollo 9 mission. The design estimate of settling time required was approximately 4 seconds. The caution and warning system was designed to be activated approximately 5 seconds after ignition, and the system compared the quantity of fuel and oxidizer that remained. The caution and warning activation system was revised to delete the PUGS comparison and thus eliminate erroneous activation of the caution and warning system.

The auxiliary gaging system also failed on several flight vehicles during ground checkout. Leakage of conductive fuel vapor into the sealed electronic package of the fuel probe resulted in shorting of the point-sensor electronic circuits. This problem was not eliminated because of the significant cost and schedule impact and the lack of criticality of the auxiliary portion of the gaging system. The auxiliary system was used only if the primary system failed. Also, other methods were available to establish onboard quantities such as acceleration, helium usage versus burn time, and predicted propellant flow rate times burn time.

Propellant-distribution assembly. - The Block I propellant-tank qualification test began in October 1963. Four fuel tanks and four oxidizer tanks were used to qualify the propellant tanks. The major problems that were encountered during qualification testing are described as follows.

In October 1963, after proof-pressure testing at 320 psig for 30 minutes, the number 1 fuel tank had a local meridional crack in the lower dome section just below the weld joint. The crack in the number 1 fuel tank was caused by stress corrosion

that resulted from a localized contaminant. Manufacturing processes were revised to eliminate the use of materials containing halogens from processes at temperatures greater than 500° F. The cracked lower dome and cylinder were replaced; subsequently, the tank was tested successfully.

In March 1964, a partial vacuum that was applied inadvertently to a fuel tank caused the tank to buckle. The tank returned to acceptable dimensions after the vacuum was released. The tank was returned to the qualification program; no damage was noted after a thorough inspection. The test stand and procedures were modified to prevent a recurrence of the problem. The Block I qualification testing was completed in January 1965.

Subsequent stress corrosion failures in RCS oxidizer tanks and in one of the spacecraft 017 oxidizer tanks required additional SPS tank-compatibility tests. These failures were caused by stress corrosion resulting from a process change in the manufacturing of nitrogen tetroxide (N_2O_4). It was determined that the use of additives to increase the free nitrous oxide content of the oxidizer resulted in satisfactory compatibility with the tank material. All of the N_2O_4 that was used in the Apollo vehicles had to meet the nitrous oxide content requirement. A Block I SPS propellant tank was subjected to an 80-day compatibility test to demonstrate satisfactory results.

During the flight of spacecraft 009 (SC 009) February 26, 1966, the transfer line in the reservoir failed, causing helium to be ingested into the SPS engine, which resulted in loss of thrust. The reservoir was modified by strengthening the standpipe support brackets and weld joints and was recertified by means of vibration testing during the Block II propellant-tank qualification. After completion of the Block II qualification test, the propellant-retention reservoir had cracks in each of six welds that connect the outlet ports to the main body of the reservoir. Truck transportation of the tanks with the retention reservoir installed was done only on the qualification-test tanks because the reservoirs were installed at the prime contractor facility. The manufacturer indicated that the cracks could have occurred during transportation of the unit; however, the retention reservoir had not been disassembled for inspection between intermediate stages of testing, and the cracks could not be definitely attributed to transportation. Therefore, a Block I and II retention-reservoir qualification retest was conducted to verify the integrity of the crossover-tube welds in the environmental conditions that the retention reservoir would undergo during boost and space flight. By means of these tests, the structural integrity of the retention reservoir was shown.

Helium-pressurization assembly.- Qualification testing of the first helium tank began in October 1963 and was completed in April 1965. A major problem was encountered when qualification-tank units three and four burst at less than the design burst pressure. By means of a design review, it was determined that the cause of failure was excessive stress concentration in a heavy girth-weld bead. Removal of the weld bead allowed the weld joint to work in unison with the membrane during cycling. Two tanks from which the weld bead was removed were added to the qualification-test program; therefore, the program was completed successfully.

Qualification testing for Block I regulators was performed from July 1 to November 3, 1965. Although the two units met the qualification-test requirements, the

primary stage of the class IV regulator leaked excessively after endurance-cycle tests. A higher internal-leakage-specification limit was accepted because the leakage through the redundant secondary stage was less than the required limit. Block II regulators completed the qualification-test program satisfactorily.

The helium check valves were redesigned after the Block I test program to reduce high leakage characteristics. The primary change was the deletion of Teflon poppet seats and the incorporation of Resistazine 88 as the poppet material on the oxidizer valve to comply with leakage requirements of 1.08 scc/hr. Qualification testing was completed on the helium-isolation valves without major redesigns.

Subsystem Tests

During the development of the SPS, a comprehensive ground-test program was conducted at the WSTF. A high level of confidence in the reliability of the basic, simple configuration of the SPS resulted from these tests. Various portions of the test program are summarized according to the test vehicle.

From September 1964 to September 1965, tests were conducted at the WSTF using a Block I test fixture. The test rig consisted of a boilerplate configuration that simulated the spacecraft propellant-line sizes and routing and that had the necessary instrumentation and safety provisions that were required for static-test operations.

The initial test series was conducted using a preprototype engine and off-the-shelf hardware in the propellant and helium-pressurization systems to establish operational procedures for fluid servicing and to evaluate system-operating characteristics until flight components were available. Later, the helium-pressurization system was updated to the flight configuration. The engine was updated continually throughout the program; a Block I qualification engine was installed for the final test series.

Two injector failures occurred during the fourth series of tests. In both cases, posttest inspections were indicative that the hub of the injector baffle had separated from the baffle. This problem was caused by afterburning of propellants in the injector. The injector was purged with nitrogen in all subsequent SPS ground tests to reduce the thermal stress caused by burning of residual propellants in the baffle.

In preparation for Block II testing, the Block I test fixture was modified extensively to a Block II configuration. The major differences between the flight configuration and the test configuration are listed as follows.

1. Propellant tanks: Test-article propellant-storage and sump tanks were boilerplate, but the wetted surfaces were flight configuration.

2. Helium-storage tanks: The helium supply for phases I and II and for series I and II of phase III was provided by nonflight-configuration external GSE. Series III and IV of phase III had flight-configured Block I helium-storage tanks.

The Block II SPS test program that was conducted at the WSTF began in November 1966 and was completed in April 1969. The program consisted of sea-level tests

that were conducted in three phases. Phase I consisted of eight test series; the objectives were system verification and performance demonstration under nominal, off-nominal, and malfunction conditions. Phase II consisted of seven test series; the objectives were to show lunar-mission performance at normal and various abnormal conditions. Another objective was to evaluate the improved (double seal) SPS engine bipropellant configuration. The final test phase, phase III, consisted of four test series; the objectives were to investigate SPS performance under extreme off-limit conditions, to investigate flight anomalies, and to perform an additional evaluation of the engine bipropellant-valve double-seal configuration. The Block II SPS test program consisted of 650 test firings that had a total firing time of 20 478 seconds. A summary of this program is contained in table II.

TABLE II.- BLOCK II WSTF TEST SUMMARY

Test phase and date	Objectives	Test series	Number of test firings	Firing time, sec	Test conditions	
					Temperature, °F	Oxidizer and fuel ullage pressure, psia
Phase I	System evaluation and characterization at nominal, off-limit, and malfunction conditions	I	36	825.51	70	175
Nov. 15, 1966 to June 7, 1967		II	47	917.84	40 to 70	175
	Demonstrate system operation for CSM 101 (Apollo 7) mission duty cycle	III	30	805.47	70	175
		IV	56	905.50	50 to 100	175 to 205
		V	45	869.63	50 to 70	179 to 187
		VI	36	896.82	70 to 100	179
		VII	54	1 016.09	45 to 70	100 to 179
		VIII	28	975.19	70 to 100	179
			[a]332	[a]7 212.05		
Phase II	Demonstrate lunar-mission performance at nominal, off-limit, and malfunction conditions	I	15	1 193.69	70	179
July 13, 1967 to Apr. 3, 1968		II	17	1 210.41	70	173
	Evaluate ball valve double-seal (1-C) configuration	III	18	1 016.90	70	197
		IV	17	1 207.75	70 to 100	170 to 190
	Evaluate PUGS performance at nominal and off-nominal conditions	V	17	1 210.62	45 to 70	179
	Demonstrate system operation for CSM 101 (Apollo 7)	VI	56	947.45	70	70 to 179
		VII	26	553.03	70	142 to 179
			[a]166	[a]7 339.85		
Phase III	Investigate SPS operation under extreme off-limit conditions and multiple malfunctions	I	17	1 497.60	70	68 to 179
May 2, 1968 to Apr. 30, 1969		II	50	909.07	70 to 105	179
	Perform additional lunar and SC 101 mission simulations	III	51	1 569.35	35 to 70	165 to 179
		IV	34	1 950.72	40 to 70	179
	Evaluate ball valve (1-E) configuration		[a]152	[a]5 926.74		
	Investigate PUGS flight anomalies		[b]650	[b]20 478.64		

[a]Test-phase total.

[b]Test-program total.

Integrated Systems Tests

The SPS subsystem demonstration tests were conducted on SC 001, which included a flight-type SM, at the WSTF from February 5, 1965, to September 7, 1966. The SPS that was installed in SC 001 was identical functionally to the Block I flight systems except for minor modifications that were required for ground testing and a more detailed instrumentation system. The SC 001 SPS was updated as required to maintain the Block I configuration.

A special test was initiated to investigate the flight anomaly that was observed during the SC 009 flight. Before this test, the oxidizer sump-tank standpipe was modified to simulate the failed transfer line in SC 009. The results of this test led to a structural improvement in the propellant-transfer lines. A spacecraft 011 (SC 011) mission duty cycle was conducted to evaluate system characteristics that resulted from the reworked transfer lines.

System operational characteristics of the SPS during nominal and off-limit conditions were demonstrated successfully by SC 001 tests. These tests included firings that ranged in duration from 23 to 600 seconds; dual- and single-engine bipropellant-valve operation; engine firings at minimum expected propellant and hardware temperatures; propellant-depletion firings; engine operation with a high engine-valve actuation pressure (190 psig); engine starts with propellant only in the retention reservoirs; rapid restart firings; chamber-pressure-decay firings at different propellant loads; engine operation with and without postfire injector purges; system operation with tank ullage pressures of 225 psia; simulated failures of individual engine-valve banks during steady-state operation; PU valve cycling during steady-state operation; firings during which "zero" propellant imbalance was maintained by means of cycling the PU valve; system operation using fuel-cell power only; and sinusoidal and step gimbaling during engine-start transients and steady-state operation.

Before the flight of AS-201 (SC 009), the SM was static fired at the NASA John F. Kennedy Space Center (KSC) launch pad 16 in November 1965. Before the firing, the oxidizer sump-tank standpipe leaked propellant into the transfer line and storage tank. It was concluded that a maximum of 230 pounds of propellant would leak into the storage tank. A shift of this magnitude would not cause a significant change in center-of-gravity location. The condition was waived because the engine operated satisfactorily during the test. This was the only Block I flight vehicle that was subjected to a static firing (see discussion of SC 009 flight).

A static firing of the first Block II SPS (spacecraft 101) had been planned. To expedite the launch schedule of spacecraft 101 (SC 101) by approximately 30 days, it was decided to static fire spacecraft 102 (SC 102) instead, decontaminate it, and return it to the production cycle. This was considered acceptable and desirable because the two spacecraft were identical, and SC 101 would not be degraded because of propellant exposure. The general objective of the SC 102 test program was to verify that the Block II SPS was ready for flight. To accomplish this, certain specific objectives were mandatory and were completed successfully.

Flight-Simulation Tests

Under simulated mission conditions, the spacecraft 008 (SC 008) SPS was evaluated during the thermal-vacuum-test program. The program contained provisions for verification that the subsystem could withstand the environments to which it would be exposed during the flight phase. The program was conducted in the Space Environment Simulation Laboratory at the NASA Lyndon B. Johnson Space Center (JSC), formerly the Manned Spacecraft Center (MSC), with simulated propellants. Data were gathered to determine the heat balance and the equilibrium-temperature distributions and to evaluate the effects of cyclic heating and cooling on the operation of the subsystem.

In August 1966, the first two tests were conducted but were invalid for the SPS. Flight-type heaters on the engine bipropellant valve were not connected, and the heaters on the engine-gimbal-ring brackets were connected so that the primary and redundant heaters were operated simultaneously. These discrepancies were corrected for the third test, and additional changes were made to bring the SPS configuration up to the spacecraft 012 (SC 012) configuration (Apollo 1). These changes consisted of insulating the system feedlines, adding flight-type heaters to the system feedlines, and placing temperature sensors in the locations that were planned for use on SC 012.

Similarly, in June 1968, spacecraft 2TV-1 was used to evaluate operation of the Block II CSM in a simulated thermal-vacuum environment. Two test objectives were applicable to the SPS. The first objective was to determine the SPS engine-temperature-decline rate during a cold soak. The second objective was to establish the ability of the SPS engine thermal control subsystem heaters to maintain temperatures satisfactorily when exposed to a simulated side sun. The data from these tests were used to update the computer thermal model, which was used to predict SPS engine temperatures for the Block II missions. Subsequent flights resulted in verification that the SPS thermal control system satisfactorily maintained critical points within the established temperature limits.

SUBSYSTEM FLIGHT RESULTS

Up to and including the lunar landing (Apollo 11), nine flights had involved the use of the SPS. Four of the systems were of the Block I configuration; five systems were of the Block II configuration. All Block I flights were unmanned, and all Block II flights were manned.

Block I Flights

Mission AS-201 (SC 009). - Two SPS firings were planned. Performance was acceptable for the first 70 seconds of the 180 seconds of firing that were scheduled for the first firing. At that time, engine-chamber pressure began a gradual decay and was approximately 70 percent at SPS shutdown. The shutdown was initiated by the control-programer backup command based on elapsed firing time. As a result of the degraded performance after the first 70 seconds, approximately 20 percent of the planned delta velocity was not achieved. At shutdown, oxidizer-tank pressure unexpectedly dropped

approximately 17 psi. Erratic chamber pressure was noted during the start of the second firing; stabilized burning was not achieved during the planned 20-second burn. The engine did appear to be recovering from the effects of helium ingestion before shutdown.

Because the fuel-systems performance was either normal or was as would be expected as a result of the change in chamber pressure, the initial malfunction analysis was indicative that the probable malfunction modes were confined to the oxidizer system. Also, it was concluded that helium ingestion into the engine and two-phase flow (gas and liquid) between the sump and storage tanks had occurred. Scale-model testing of the oxidizer tank and retention reservoir at the JSC and a full-scale test at the WSTF (SC 001) were indicative that the probable failure mode was a leak in the oxidizer-transfer line inside the retention reservoir as discussed previously.

Mission AS-202 (SC 011). - Mission AS-202 was the second flight test of the SPS. The primary test objectives were to verify the SPS standpipe fix by means of a minimum firing of 198 seconds and to demonstrate multiple SPS restarts (at least three firings of at least 3-second duration at 10-second intervals). Secondary test objectives were to determine long-firing (approximately 200 seconds) SPS performance (including shutdown transient characteristics) and to obtain data on SPS engine-firing stability. All objectives were met.

Apollo 4 (SC 017). - The Apollo 4 mission included the third flight test of the SPS. The primary SPS test objectives were to demonstrate a satisfactory start without an RCS settling maneuver and to determine SPS performance during a long-duration burn. Both objectives were met. The SPS operated nominally during both firings. During propellant crossover, the engine-inlet pressure and chamber pressure increased as was expected; the pressures were steady throughout the firing. The general effects of propellant crossover were as expected.

Apollo 6 (SC 020). - Essentially, the SPS mission objectives were the same as for the Apollo 4 mission. Because of the inability to restart the S-IVB stage, the SPS was used to transfer the CM from an earth-parking orbit to the highly elliptical earth-intersecting orbit that was needed to satisfy entry conditions for the CM heat shield entry test. The objectives of satisfactory SPS operation and a no-ullage start were accomplished. In addition to these objectives, this firing was the first in which the SPS demonstrated the firing-duration capability that was needed to insert the CSM into a lunar orbit.

Engine performance was satisfactory except for an overshoot in chamber pressure during engine start. All other engine-transient criteria were met. For the Apollo 4 and 6 missions, the chamber-pressure transducer had been mounted on a 2-inch adapter to reduce the thermal effects that had caused an erroneous indication of chamber-pressure drift. The overshoot noted with the new adapter was significantly higher than with previous adapters. A special series of tests involving the use of high-resolution instrumentation was scheduled at the AEDC to determine if the overshoot was caused partially by instrumentation error. It was determined that thrust-chamber-pressure overshoots were reduced significantly if the engine firings were initiated using a single bank of ball valves. It became standard operating procedure to start each firing in the single-bank mode. If the burn was scheduled to be longer than 6 seconds, the redundant bank was opened approximately 3 seconds after ignition.

Block II Flights

Five manned Apollo Block II missions were accomplished up to and including the Apollo 11 lunar-landing mission. The SPS performance on these missions is discussed as follows.

Apollo 7 mission. - The Apollo 7 mission was the first mission on which the Block II SPS was flight tested. As was planned, there were eight firings of the SPS engines. The four primary objectives related to the SPS were minimum-impulse burn, performance, primary/auxiliary propellant-gaging system, and thermal control. All objectives were met.

Apollo 8 mission. - The Apollo 8 mission was in jeopardy after the first SPS burn, which was a midcourse correction. A momentary drop in chamber pressure was observed at the start of the first burn. This drop was attributed to the presence of a gas bubble in the oxidizer feedline. The gas was determined to be trapped helium that resulted from an inadequate engine-oxidizer bleed during preflight servicing. After extensive discussions and analysis by ground-based personnel, the decision was made to continue the mission. Two key items were important in the real-time decision to continue this mission, which was the first attempt to leave earth and orbit the moon with a manned spacecraft. Telemetry data from the AS-201 mission, in which the engine also had sustained some helium ingestion, and data from ground tests, in which helium ingestion occurred, were compared with telemetry data to establish the signature trace in chamber pressure of a small amount of helium ingestion. Recordings of voice-track tapes from the Apollo 8 loading operations at the KSC were indicative that the engine feedlines had not been bled properly to remove the helium in the high point of the line. With this evidence, the cause of the chamber-pressure anomaly was established, and the required confidence that the engine was not damaged was provided.

During the early portion of translunar coast, a drop of approximately 7 psi was noted in the SPS oxidizer-tank pressure. It is believed that the pressure decrease was caused by helium absorption; the decrease stopped when the oxidizer apparently became saturated.

The engine was started on all maneuvers by the use of only one of two redundant sets of valves in the engine bipropellant-valve assembly. This procedure was instituted to decrease the initial chamber-pressure and thrust-level overshoot, which are characteristics of a start with both valve sets open. A noticeable decrease in the overshoot magnitude was achieved. During the lunar-orbit-insertion (LOI) and transearth-injection maneuvers, the redundant-valve set was opened approximately 3 seconds after ignition to increase the operational reliability for the remainder of the firing in case one of the valve sets should close.

Apollo 9 mission. - The Apollo 9 mission involved both the CSM and LM. The fifth SPS firing was made after a docked LM descent-engine firing of approximately 372 seconds duration. Preflight analyses indicated that, when a descent-engine firing was performed with the spacecraft docked, a negative acceleration greater than $0.1\ ft/sec^2$ would result. This acceleration could cause depletion of the propellant captured by the SPS sump-tank retaining screens. Although the retention reservoir would still remain full, some helium could be trapped and ingested into the engine

during a subsequent SPS firing. However, after the docked descent-engine firing, all SPS firings were normal and smooth, indicating that no significant quantity of helium had been ingested.

Apollo 10 mission.- One of the most significant changes in the Apollo 10 SPS was the addition of strip heaters in the propellant-distribution lines that ran from the tank outlets to the bipropellant valves. The strip heaters provided a method of maintaining propellant temperatures above the 30° F minimum allowance in a deep space environment to prevent freezing of propellants in the feedlines. The SPS performance was satisfactory during each of the five maneuvers, and the total firing time was 545 seconds. The longest engine firing was the 356-second burn on the lunar-orbit-insertion maneuver. The fourth and fifth SPS maneuvers, which occurred after depletion of the storage tanks, were preceded by a plus-X RCS translation to settle the propellants. All firings were conducted under automatic control.

Apollo 11 mission.- The SPS performance was satisfactory during each of the five maneuvers; the total firing time was 531.9 seconds. The longest engine firing was 357.5 seconds during the lunar-orbit-insertion maneuver. The fourth and fifth SPS firings were preceded by a plus-X RCS translation to settle propellant, and all firings were conducted under automatic control. The steady-state performance during all firings was satisfactory. The steady-state pressure data were indicative of essentially nominal performance; however, gaging-system data were indicative of a propellant flow ratio of 1.55:1, rather than the expected propellant-ratio range of 1.60:1 to 1.61:1. The lower than expected mixture ratio decreased the amount of propellant available for velocity changes. One SPS anomaly occurred during the LOI burn. An abnormal pressure decay was noted in the secondary gaseous nitrogen (GN_2) supply, indicating a leak. The decay ceased at engine shutdown, and no additional decay was noted. The leakage was attributed to contamination on the seat of the GN_2 actuator solenoid valve. Filters were incorporated to prevent recurrence of this problem on subsequent spacecraft.

CONCLUDING REMARKS

The program that was developed to meet the service propulsion subsystem requirements lasted 92 months, from the time that the prime contractor was given contract go-ahead until the Apollo 11 manned lunar landing. During this period, nine flights were completed successfully, and all of the original requirements were met. The program necessitated the use of a prime contractor, 13 major subcontractors, approximately 500 vendors and suppliers, and two major Government-owned test facilities.

Although several significant events occurred during the development phase, the point at which the program began to reach maturity was on November 21, 1964, when it was decided to initiate a Block II (lunar landing) service propulsion subsystem configuration. By this time, most of the technical problems had been identified or at least were understood to the point that designs were available for incorporation into the subsystem. However, schedules for a research program in which the problems are unknown usually have to be modified as the program proceeds. The service propulsion

system program was no exception. During 1965, the most critical period of development occurred when component-fabrication changes reached a peak. Hardware modifications, which are made in order to meet reliability requirements, not only delayed the completion of the qualification of each component, but necessitated a substantial amount of retesting with the heavyweight subsystem test rigs to determine if interactions would occur between major assemblies. The schedule also was delayed by assembly and checkout of the first flight service propulsion subsystem and its subsequent failure because of the structural collapse of the propellant-retention reservoir. By 1966, the subsystem configuration was completed and the program proceeded into the flight phase.

The most significant lesson that was learned from the service propulsion subsystem program was the need to develop basic technology for propulsion systems before initiating full-scale hardware designs. Besides the anticipated technical problems, such as engine performance and combustion instability, schedule delays were experienced during hardware development, and these delays generally were associated with the high reliability requirements of the Apollo Program and the lack of experience with the propellants and their effects on materials. Typical of these problems were propellant-tank stress corrosion, deterioration of seals in the tank doors and pressurization components, limited engine bipropellant-valve-seal cycle life, and nitrogen tetroxide corrosion of aluminum parts. Although some studies had been conducted in selected areas, there was a lack of knowledge on the behavior of propellants in zero gravity, on the mechanism of propellant ignition, on the effects of freezing and thermal decomposition of propellants, on ablative materials suitable for use in reliable thrust chambers, and on data concerning the generation and effects of contamination.

Lyndon B. Johnson Space Center
National Aeronautics and Space Administration
Houston, Texas, March 27, 1973
914-13-00-00-72