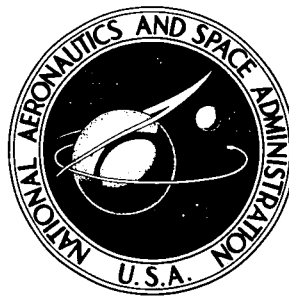


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APOLLO EXPERIENCE REPORT - ASCENT PROPULSION SYSTEM

by Clarence E. Humphries and Reuben E. Taylor

Manned Spacecraft Center

Houston, Texas 77058

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16. Abstract The development of the Apollo lunar module ascent propulsion subsystem is documented. The ascent propulsion subsystem was designed, built, and tested to provide propulsive power for launching the ascent stage of the lunar module from the surface of the moon into lunar orbit for rendezvous with the orbiting command and service module. Because total redundancy was prohibitive in this subsystem, special emphasis had to be placed on reliability in design with adequate demonstration short of an extensive flight-demonstration program. The significant technical problems encountered during all phases of the program are identified and the corrective actions are discussed. Based on this experience, several recommendations are made for any future program with similar subsystem requirements.					
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CONTENTS

Section	Page
SUMMARY	1
INTRODUCTION	2
REQUIREMENTS	2
SYSTEM DESCRIPTION	3
DEVELOPMENT AND QUALIFICATION	5
Components	6
Ascent Engine	10
Test Articles and Test Rigs	16
SYSTEM CHECKOUT	20
FLIGHT PERFORMANCE	21
Apollo 5 Mission	21
Apollo 9 Mission	22
Apollo 10 Mission	23
Apollo 11 Mission	24
CONCLUDING REMARKS	24
Definition of Realistic Requirements	24
Flexible Program Flow Plan	24
Leakage and Contamination Control	24
Inspection and Manufacturing Repeatability	25

TABLES

Table		Page
I	MAJOR TEST ARTICLES AND RIGS	16
II	THE PA-1 WHITE SANDS TEST FACILITY TEST SERIES SUMMARY	19

FIGURES

Figure		Page
1	Ascent propulsion system with an ambient-helium-pressurization system	3
2	Ascent-engine assembly	4
3	Ascent-engine injector and valves	4
4	Major phases of the APS development program	5
5	Squib valve	8
6	Ascent propulsion system propellant-tank torus ring	10
7	Types of injector grid patterns	11
8	Stability devices on the production injector	12
9	Final-design injector	15
10	Propellant-inlet section of the injector manifold	15
11	Injector filter	15
12	Modified engine fuel duct	15
13	Test rig PA-1	19

APOLLO EXPERIENCE REPORT

ASCENT PROPULSION SYSTEM

By Clarence E. Humphries and Reuben E. Taylor
Manned Spacecraft Center

SUMMARY

In June 1962, the Manned Spaceflight Management Council selected the lunar-orbit-rendezvous mode for the Apollo Program. From this concept the requirement evolved for a separate lunar module ascent stage propulsion subsystem to return the two crewmembers to a lunar-orbit rendezvous with the command and service modules. A propulsion system in which hypergolic bipropellants and a pressure-fed ablatively cooled engine were to be used was developed. The system was to provide 3500 pounds of thrust.

During the development program, emphasis was placed on the use of proven manufacturing techniques, design integrity, and ground-based testing. This approach resulted in the discovery and correction of all significant problems in the engine and in the pressurization and feed system before using the equipment on manned flights.

The following principal problems were encountered and resolved. Engine combustion instability was resolved ultimately by developing a new injector under contract to a second manufacturer. Repeated failures and discrepancies on the original pressurization-system regulator were resolved by modifying and adapting the lunar module reaction control system regulator to the ascent propulsion subsystem. Braze and weld-joint design problems on several pressurization and feedline components were resolved. Vehicle system leakage lead to redesign of feed system mechanical joints to provide redundant sealing capability.

Major vehicle-integrated propulsion system testing and certification were accomplished on the prototype PA-1 test rig at the NASA White Sands Test Facility using flight-type engine and component hardware. Early flight-vehicle checkouts, accomplished at Bethpage, N. Y., and at the NASA John F. Kennedy Space Center, involved problems that necessitated frequent component replacements because of failures on the spacecraft caused by component failures during qualification. The lunar module ascent propulsion subsystem flight systems performed within nominal predicted ranges and only on one series, lunar module-3, was a system problem indicated. A failure of the ascent propulsion subsystem helium primary regulator was indicated by a shift in operating pressures to a secondary regulator and was attributed to contamination during replacement of a component in checkout.

A review of the Apollo ascent propulsion subsystem experience is indicative that future system development can be augmented by early definition of realistic requirements (including leakage and contamination control), a more flexible program-flow plan, and improvement in design and certification techniques to ensure adequate braze- and weld-joint inspection and parametric control.

INTRODUCTION

The ascent propulsion subsystem (APS) had to operate satisfactorily to return the crewmen safely from the lunar surface. It was imperative that the system be as simple as possible to increase its reliability. Therefore, the primary considerations in the design and development of the APS were simplicity and reliability, whereas the secondary considerations were performance and weight.

The propellants were selected on the basis of experience with other programs, storage requirements, hypergolicity, performance, and density requirements for packaging. Pressure-fed engines were selected to decrease complexity by eliminating the need for pumps, associated moving parts, and controls. High-pressure gaseous helium (stored at ambient temperature) was selected for the pressurizing fluid. Ablatively cooled thrust chambers (as compared with regeneratively cooled chambers) were used to decrease the possibility of the propellant freezing. Wider operational limits of mixture ratio, propellant temperature, and chamber pressure also were possible with ablatively cooled chambers than with regeneratively cooled chambers. A single-engine concept was chosen to simplify the vehicle control system requirements and thereby increase total system reliability. Subsystem reliability had to be achieved with a single system. Early studies showed that the majority of propulsion-system failures were caused by failures of controls, valves, and solenoids, and were not the result of injector or thrust-chamber failures. Therefore, redundancy of these components was used, where practical, to increase reliability in these failure-sensitive categories.

REQUIREMENTS

To meet guidance requirements, the APS was required to produce 3500 pounds of thrust, to fire for a total duration of 550 seconds, to develop 90 percent of the rated thrust within 0.450 second after the start signal, and to decay to 10 percent of the rated thrust within 0.500 second after the cutoff signal. The maximum allowable combustion-chamber pressure during start transients was 177 percent of the nominal combustion-chamber pressure. This was a vehicle structural limitation. To minimize the heat transfer to the chamber, the magnitude of any periodic or uniformly cyclic chamber-pressure fluctuation or oscillation that occurred at a frequency of 400 hertz or less could not exceed ± 3 psi, and those variations that occurred at a frequency greater than 400 hertz could not exceed ± 6 psi. In addition, the engine was required to be stable dynamically in the presence of all self-induced or artificially induced disturbances, thereby causing fluctuations of 175 percent of nominal chamber pressure in the combustion process. Recovery time to a stable steady-state operation could not exceed 0.020 second.

The required oxidizer-to-fuel mixture ratio was 1.6 to 1, with a 170 psia required pressure at the interface between the propellant-feed section and the engine. The required propellant bulk temperature was $70^{\circ} \pm 20^{\circ}$ F, and the fuel and oxidizer temperatures were to be within 10° F of each other.

The liquid propellants for the APS were nitrogen tetroxide (oxidizer) and an equal mixture of hydrazine and unsymmetrical dimethylhydrazine (fuel). A total of 5213 pounds of propellant was required, of which 196 pounds was residual.

SYSTEM DESCRIPTION

A schematic of the APS is shown in figure 1. The propellants are pressurized by gaseous helium at ambient temperature, supplied from two identical tanks and routed through redundant-flow lines into the pressure regulators. The helium is stored at a nominal pressure of 3150 psia and a nominal temperature of 70° F.

Before initial ascent-engine operation, explosive valves are used to isolate the helium-storage tanks. A filter in each helium-flow path traps debris from the explosive valves. Downstream from the filter, each helium-flow path has a normally open latching solenoid valve and two series-connected pressure regulators.

The primary and secondary helium-flow paths merge downstream from the regulators to form a common helium manifold. The manifold routes the helium into two flow paths: one path leads to the oxidizer tanks and the other path leads to the fuel tank. A quadruple check-valve assembly isolates the upstream components from the corrosive propellant vapors and prevents possible hypergolic action in the common manifold that may result from mixing of propellants or propellant vapors as a result of backflow from the propellant tanks.

Two parallel explosive valves, downstream from each quadruple check-valve assembly, provide a positive seal upstream from the propellant tanks to isolate the fuel and oxidizer (liquid and vapor) before the initial ascent-engine start. This configuration reduces compatibility problems involving helium components and prolongs the life of the pressurization-system components.

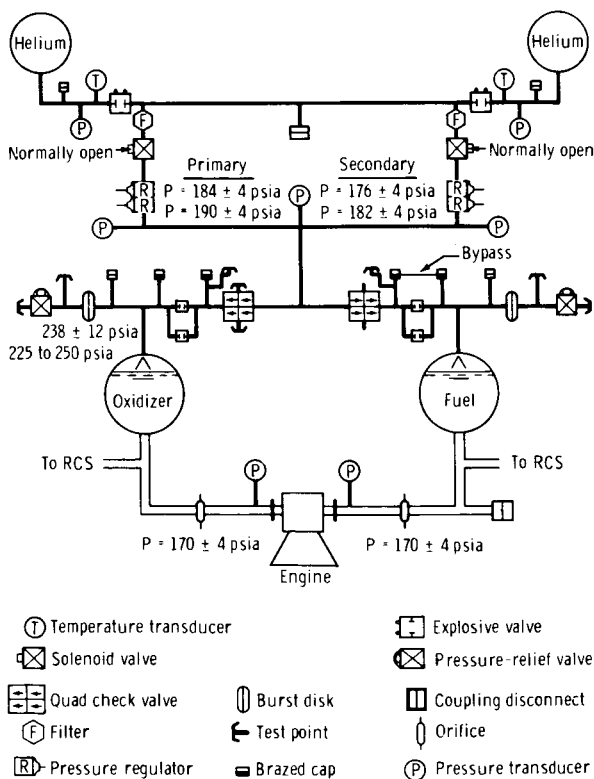


Figure 1. - Ascent propulsion system with an ambient-helium-pressurization system.

Immediately upstream from the fuel and oxidizer tanks, each helium-flow path contains a burst-disk relief-valve assembly. The relief valve on each helium-flow path can pass the entire helium flow from a failed-open pair of regulators without damaging the propellant tanks. A low-level sensor in each propellant tank causes a cabin caution light to turn on when the remaining propellant in either tank is limited to approximately 10 seconds of engine operation.

Each propellant flows through a trim orifice to the propellant filter in the engine assembly, then to the isolation and bipropellant valve assemblies (propellant-shutoff valves). The trim orifice provides an engine-interface pressure of 170 psia for proper propellant utilization by the engine. A secondary distribution line is interconnected to the reaction control system (RCS). A series-parallel arrangement of RCS/APS interconnect solenoid valves (part of the RCS) permits the RCS to burn APS propellants (providing the APS is pressurized and the propellant is settled during the time the interconnect valves are opened).

The ascent engine, shown in figure 2, is a fixed-injector, restartable, bipropellant rocket engine that has an ablatively cooled combustion chamber, throat, and nozzle extension. Propellant flow to the ascent-engine combustion chamber is controlled by a valve-package assembly, trim orifices, and an injector assembly (fig. 3). The valve-package assembly is equipped with dual passages for both the fuel and the oxidizer and has two series-connected ball valves in each flow path.

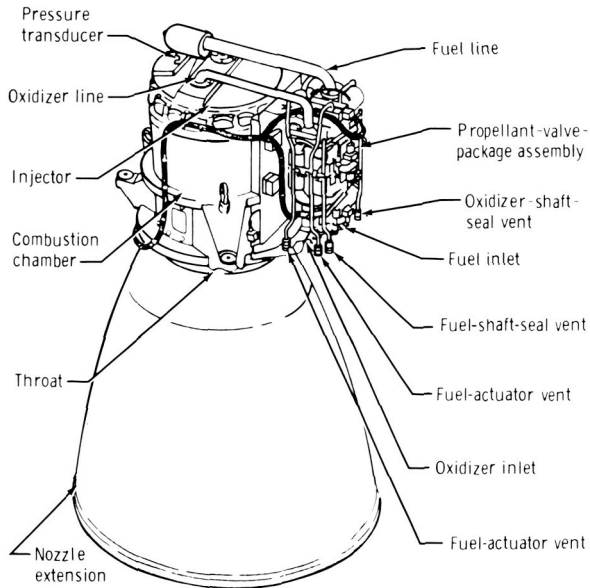


Figure 2. - Ascent-engine assembly.

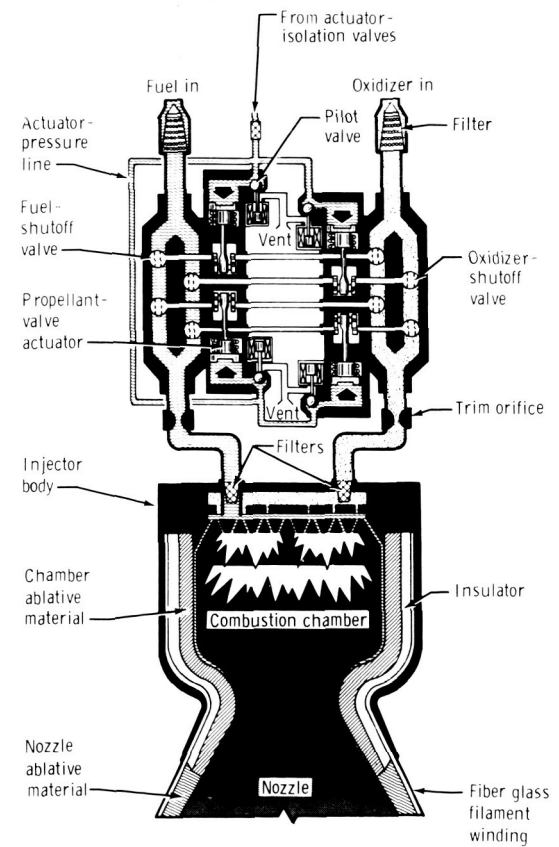


Figure 3. - Ascent-engine injector and valves.

DEVELOPMENT AND QUALIFICATION

Development of the APS through an extensive flight-test program was not feasible economically. Therefore, emphasis was placed on using proven manufacturing techniques, design integrity, and ground-based testing. The program was designed to achieve a fully qualified system in time to support the first manned lunar module (LM) flight. The plan was to test and evaluate materials, components, and assemblies in progressively integrated configurations, using various test rigs and prototype structural simulators. Major phases of the development program to accomplish this plan are shown in figure 4. The development and qualification of the APS started at the component level, progressed through the modular level, and, finally, moved into full-scale system tests.

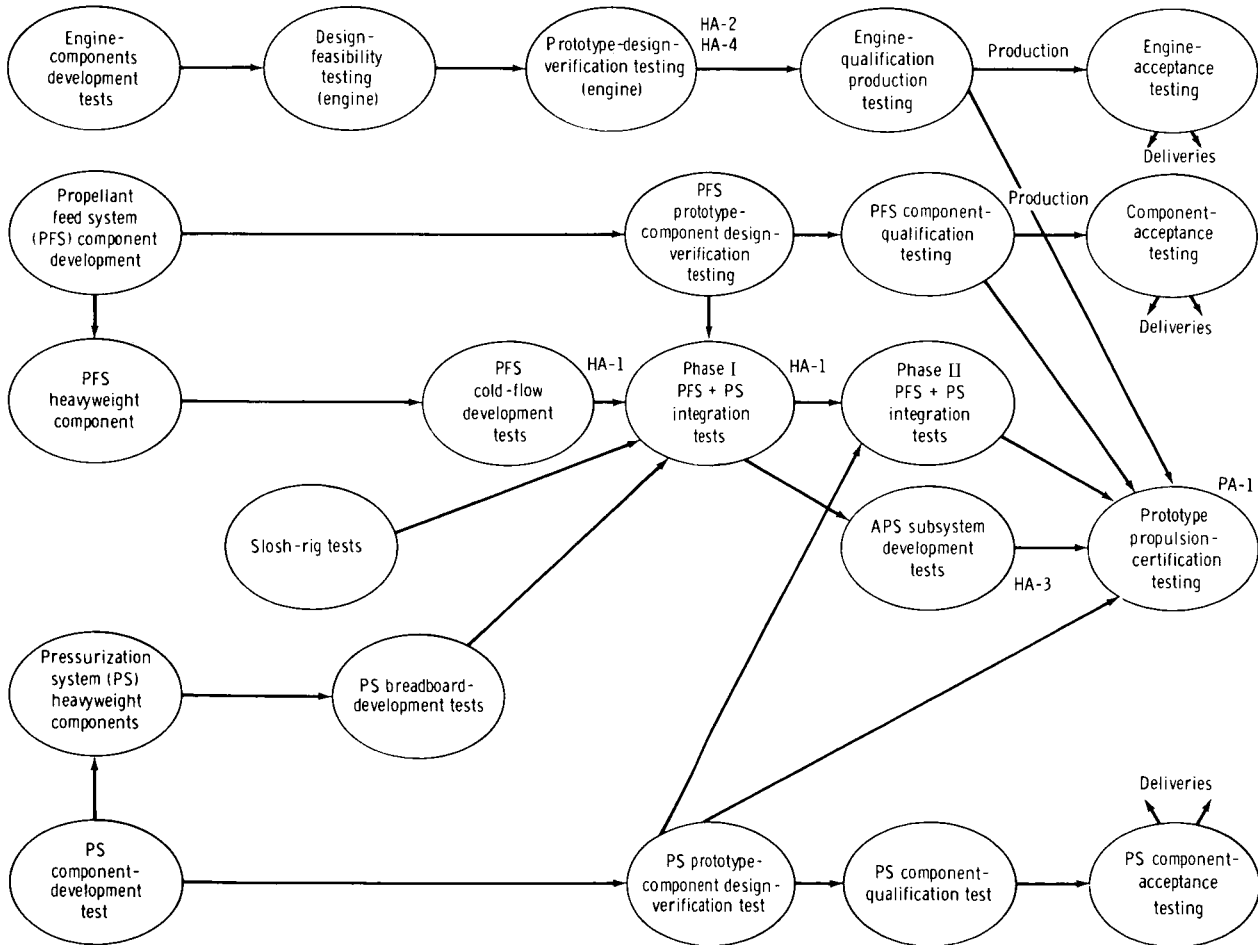


Figure 4. - Major phases of the APS development program.

In addition to the mainstream hardware-test programs, other tests were designed to identify and resolve potential problems and to acquire test data to support the flights. Tests to determine the effects of a vacuum on fuel venting from the actuator during engine-valve shutdown, the effects of a vacuum on the engine-valve fuel actuator when a large leakage occurred, and the restart limitations on the ascent engine in space were among the other tests.

Components

Helium tanks. - The contract that was awarded for the helium-tank pressure vessels was delayed because of a delay in the final propulsion-system-concept definition. Spacecraft-weight problems necessitated detailed trade-off studies on the possible use of a supercritical-helium-storage system. To avoid the complexity of the supercritical-helium-storage system, the ambient-helium-storage system was retained even though there was a potential weight penalty.

Ten ascent helium tanks that were approximately 0.1 inch undersize across the mounting base were fabricated. This fabrication error was caused by the failure to consider weld shrinkage when the assembly drawings were made. The affected tanks were used in the design-feasibility test program, the design-verification test (DVT) program, and some of the ground tests. Qualification of the ascent helium tank was completed successfully; no significant problem was noted.

Helium solenoid latching valve. - The original subcontractor had difficulty meeting the specified differential-pressure and internal-leakage requirements. The subcontractor requested and was granted a release from the contract; therefore, a second source was selected for the development and qualification of the flight solenoid valve.

Originally, the solenoid valves were designed without a requirement for fuel and oxidizer compatibility; and nylon seats, in combination with Butyl O-rings, were used to seal high-pressure cavities within the valve. When it was determined that propellant vapor would migrate because of helium pressurization, the requirement for compatibility was imposed. The subcontractor was directed to carry on parallel efforts that consisted of testing the existing valve design with propellant vapors for 3.5 days to determine whether the solenoid valves (with compatibility squib valves in the system downstream from the check valves) were acceptable and developing a valve to meet requirements for a 44-day period of exposure to propellant vapors. At that time, use of nylon and Butyl O-rings was considered adequate for a 3.5-day exposure to propellant vapors. However, a design-feasibility investigation (DFI) test program was initiated to determine nylon and Butyl compatibility with propellant vapors for 44 days. Before completion of the compatibility testing, the subcontractor was directed to stop all work related to the 44-day propellant-compatibility redesign for the solenoid valves because the decision had been made to retain the compatibility squib valves located downstream of the check valves.

The DFI testing indicated that backup rings and lubricants were necessary to prevent severe chafing of the Butyl O-rings during cycling of the solenoid valves. The compatibility tests showed that nylon disintegrates when it comes in contact with oxidizer vapor, proving the nylon seat was inadequate; therefore, the seat material was

changed to the present material (Teflon). Other design changes brought about during testing were the replacement of Teflon backup rings with Teflon "cap seals" and the incorporation of electromagnetic interference (EMI) suppressors.

Cold-temperature testing during the DVT program showed the need to reduce O-ring friction to permit reasonable closing times. The use of a Teflon cap seal and a compatible lubricant (PR 240AC) solved the problem.

When the solenoid valves were deenergized, back electromotive force in the solenoid coils caused voltage spikes that damaged other electrical equipment. Zener diodes were added between coil leads to redirect this energy until it dissipated. The qualification program was completed successfully; however, the change to the dual-diode EMI-suppression package required a supplemental qualification program.

The solenoid latch valve installed in test rig PD-2 (a descent propulsion test rig at the NASA White Sands Test Facility (WSTF)) demonstrated a problem in its ability to remain in the open-latch position after open-close commands in rapid sequence. The problem was one of residual magnetism in the latch plunger, causing it to exceed the spring-return force when the valve was cycled rapidly. The design changes to eliminate this problem were a higher force latch spring, a modified latch plunger (to house the new spring), and shims to maintain the required latch-spring force on assembly of each valve. Two valves were successfully subjected to a requalification test series.

The solenoid valve on LM-3 leaked externally through the brazed joint at the valve body and an inlet tube. Upon review of the brazed-joint design, it was found that the difference in the thermal expansion of the valve body and the tube eliminated the clearance required for the nickel-brazed material to flow properly. The decision was made to redesign the solenoid valve for LM-5 and subsequent vehicles by using a brazed joint with the proper clearances and gold-nickel alloy as the braze material. A test program was initiated, with the original nickel-brazed configuration, to certify the solenoid valves for LM-3 and LM-4. These tests were completed successfully.

The external leakage reported at the prime-contractor plant for the LM-3 solenoid valve later increased when the valve was checked at the NASA John F. Kennedy Space Center (KSC). The increase was partly attributed to the difference in measuring techniques; however, the decision at that time was to replace the leaking solenoid valve with the new-configuration solenoid valve at the KSC. Subsequently, the LM-4 valves were changed to the gold-nickel-brazed configuration. A delta-qualification test program was successfully conducted to certify the gold-nickel-brazed configuration for flight.

Helium pressure regulator. - The original subcontractor had difficulty meeting the specifications for external-leakage control and control bandwidth. The requirements were determined to be unnecessarily stringent and were modified.

Continued failures of the regulators to meet the lockup and cold-helium requirement led to the search for a suitable backup. The helium regulator used for the LM and the service module reaction control systems was chosen and was modified to meet the flows that were consistent with the APS requirements. A backup subcontractor was engaged to provide the helium regulator. Testing of this regulator configuration showed low-frequency oscillations at nominal APS flow rates. Evaluation of the test

results indicated that the regulator was very sensitive to the APS modular configuration and, possibly, the oscillations were caused by coupling between the check valve and the regulator. Tests were run at the WSTF on test rig PA-1 to determine the effects of the oscillation on the system. It was determined that the oscillations did not propagate to the engine interface; therefore, the oscillations did not affect engine operation. As a result of these tests, the specification was changed to allow oscillations less than 15 psi peak to peak, which were representative of those observed during PA-1 testing.

Both subcontractors submitted regulators to the qualification program. The original subcontractor submittal had many discrepancies and underwent many failures that would require further developmental work. The backup subcontractor submittal underwent one failure; the regulator failed to lock up after a long blowdown. This failure was attributed to freezing of excessive moisture in the helium used during testing. The moisture was verified as the failure source during subsequent tests; therefore, the backup unit was selected.

The prime contractor in-house investigation of the cause and effect of oscillations in regulated pressure was continued, and a muffler to damp the oscillations was developed and added to the pressurization system for LM-3 and subsequent vehicles in the spring of 1968.

Squib valve. - During the DVT program for the squib valve, it was noted that the piston of the valve did not travel to the fully open position. The cause of the failure was determined to be a design deficiency in the piston seal (fig. 5). This design deficiency allowed pyrotechnic blowby. The valve was redesigned and, by means of subsequent testing, it was verified that the system was acceptable.

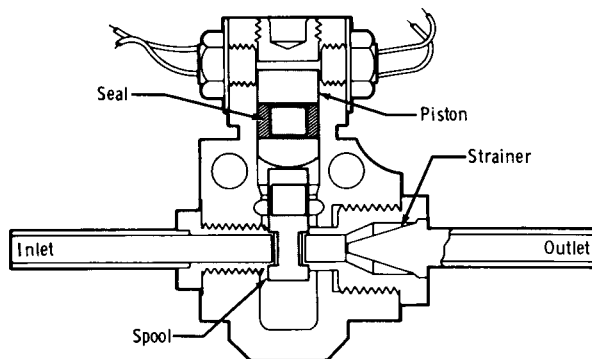


Figure 5. - Squib valve.

In late 1966, a squib valve that had previously met all acceptance-test leakage requirements was found to be leaking at a braze joint while it was installed on one of the WSTF test rigs (PA-1). An investigation of the failure revealed an inadequate braze joint between the inlet tube and the valve body. The faulty braze joint probably resulted from excessive porosity or poor brazing flow. The design of the braze joint did not lend the joint to adequate inspection procedures; therefore, the prime contractor initiated an inline braze change from an internal braze joint to an external braze joint that could be inspected. The modified design was not incorporated at this point in the program because the failure was believed to be an isolated case.

During the Apollo 9 (LM-3) mission in March 1969, the descent-engine pressurization system had a leak that was similar to that observed at the WSTF on test rig PA-1. Later, after the LM-2 drop test in May 1969, another squib valve was found to be leaking after operation. As a result of these additional failures, a decision was made to replace all valves of the early brazed configuration on the lunar modules.

Quadruple check valve. - Only minor problems with the quadruple check valve were experienced in the development program. The qualification-test program was successful, even though some of the valves did not meet the leakage specification of 10 scc/hr. The leakage rates experienced were determined to be acceptable for the LM missions because squib valves were retained for helium-system isolation purposes and operated just before ascent-engine operation.

Propellant tank. - During development testing, a production LM ascent-stage propellant tank was mounted in a fixture, placed in a temperature-controlled environmental chamber, filled with oxidizer (nitrogen tetroxide) to 98 percent of the total tank volume, temperature conditioned at 105° F, pressurized to 245 psi with 100 percent gaseous helium, and maintained at these test conditions for 47 hours. At the end of this period, the test vessel started losing pressure at the rate of 2.5 psi/min, indicating a tank failure. Visual inspection of the tank revealed a 0.5-inch crack in the membrane area of the lower hemisphere. A metallurgical evaluation of the tank material disclosed the presence of stress corrosion similar to that experienced at about the same time period with other titanium oxidizer tanks. The amount of nitric oxide in the oxidizer had been reduced recently in the process specification to procure purer nitrogen tetroxide. The resolution (addition of nitric oxide to the nitrogen tetroxide) was common to all oxidizer tanks in the program. The amount of nitric oxide required in nitrogen tetroxide to eliminate the stress corrosion was 0.45 to 0.95 percent.

In an effort to save weight, all-welded tank configurations were developed in which the manhole closures at the bottom of the tank and on the diffuser at the top of the tank were modified. This alteration saved approximately 8 pounds per tank by eliminating the heavy bolted flanges. Titanium pipe elbows were welded to the closure and the diffuser, and bimetallic joints were used to make the transition from the titanium elbows to the type 304L stainless-steel lines. The zero-gravity can and antivortex baffle were redesigned using titanium rather than aluminum. The modified tanks successfully completed qualification in May 1969.

Propellant feedlines. - Originally, the propellant feedlines were designed to incorporate flex hoses to absorb the movement of the lines. During design-verification testing, a failure of the flex hose was experienced. In June 1966, the flex hose was replaced with a bellows assembly. The bellows assembly was modified further to incorporate restraining bars after misalignment of the lines was noted on LM-1. For LM-3 and subsequent vehicles, ball joints were installed instead of the bellows.

Because of the weight problem of the LM, a weight-saving program was initiated that resulted in reducing the propellant-feedline wall thickness. The new feedlines were installed on LM-6 and subsequent vehicles, and the subcontractor initiated a qualification program to verify the acceptability of the thin-wall line and ball-joint assembly. A leak was detected during vibration qualification testing on the lightweight lines. The leak was located at the bracket that holds the line to a torus ring at the bottom of the tank (fig. 6). The propellant-feedline design required welding of the bracket to the line; during subsequent vibration testing, cracks were noted in the welded area. Examination of the failure revealed that the problem resulted from a concentration of stresses in the welded area during vibration. The solution was to distribute the stresses by incorporating another supporting member in the design.



Figure 6. - Ascent propulsion system propellant-tank torus ring.

During the setup for the tank-qualification test, which incorporated the section of propellant line where the leak had been experienced, cracks were noted in the welded areas. A thorough investigation revealed that the cracks were induced during the welding of the bracket to the line. The following decisions were made as a result of the investigation.

1. The LM-5 (heavyweight configuration) was to be flown with no modifications because the 0.049-inch wall thickness was provided already.

2. The bracket for all lines less than the 0.049-inch wall thickness was to be ground off and, if there were no cracks, a clamshell bracket and saddle were to be installed with epoxy for structural support. This modification was made on the LM-6 oxidizer line and on both oxidizer and fuel lines for LM-7 to LM-9.

3. For LM-10 and subsequent vehicles, a new propellant-feedline section without the weld was to be used.

Ascent Engine

The contract to develop the ascent engine was awarded in July 1963. By early 1967, it was apparent that the injector development probably would constrain the Apollo missions as a result of failures that had occurred during stability testing. Consequently, a backup contractor was selected to develop an injector that would satisfy the requirements of the Apollo missions and that would be compatible with the other engine components developed by the original engine contractor.

The original ascent-engine development was divided into four categories: the injector, the ablatively cooled thrust chamber, the bipropellant valve, and the engine assembly.

Injector. - Three different types of injector patterns were released for fabrication. The types were an alternating-triplet grid pattern, an alternating-triplet grid and radial patterns, and a radial-triplet pattern (fig. 7). Each pattern used a barrier doublet (oxidizer and fuel) for chamber cooling.

In the summer of 1964, an injector that had a radial-triplet pattern was selected for the prototype because of its characteristic compatibility with the ablatively cooled thrust chamber. Performance was still a problem; therefore, modifications to the

injector pattern that increased the film-barrier propellant-mixture ratio to achieve higher performance and still maintain thrust-chamber compatibility were pursued.

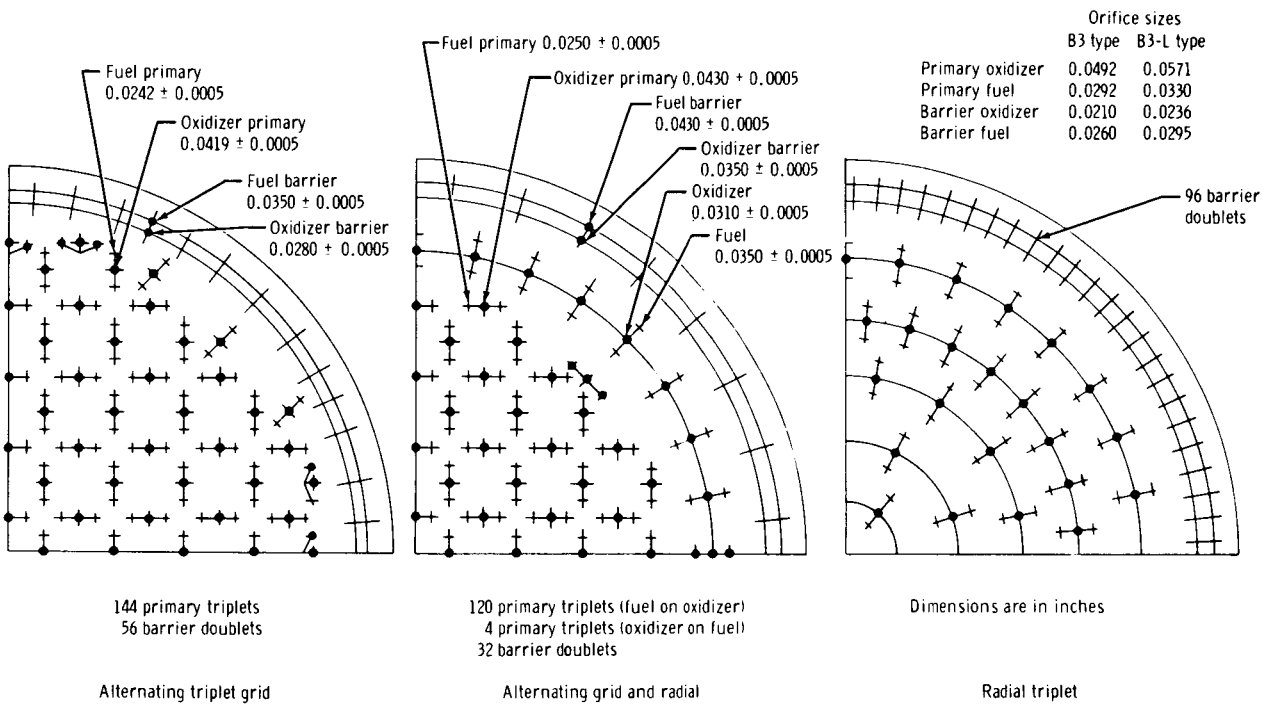


Figure 7. - Types of injector grid patterns.

In the fall of 1964, two changes in requirements were imposed: the NASA Manned Spacecraft Center (MSC) combustion-stability criteria were clarified and the continuous-burn time was increased to 460 seconds because of increased vehicle weights. Combustion-stability tests on the un baffled injector were delayed because of the lack of availability of instrumentation. In late January 1965, the un baffled injector was determined to be unstable under the new criteria, and the design of a baffled configuration was initiated.

The original contractor investigated several baffle configurations. Using experience from the descent-engine program, a flow-through baffle design for cooling was implemented. This design was selected, even though several combustion instabilities occurred with metal chambers used in bomb tests during the development phase of the program. These failures were explained as hardware failure before instability, location of the bomb in the throat of the thrust chamber that was thought to be an unrealistic test condition, or the result of a gap between the baffle and the chamber. This decision was explained further with the hypothesis that an ablatively cooled chamber would help damp combustion oscillations. Two versions of the baffled-injector configuration were designed. Because of fabrication problems, both injectors required 6 to 7 months to fabricate. As a result of the time required to develop the baffled injector and the requirement to meet schedules, the production release was required before either

configuration was hot fired. A decision was made to select the configuration for production that gave the higher performance and ablative compatibility, as determined from the test results of various development injectors.

The initial firing of the production injector exhibited erosion of the ablative chamber material near the injector-chamber interface. To eliminate this problem, modifications were made to the barrier orifices; these modifications reduced the erosion. During the oxidizer- and fuel-calibration runs for the engine qualification with the baffled injector, the engine underwent combustion instability when bombed with an artificially induced disturbance in an ablatively cooled chamber. The engine was inadvertently allowed to continue to fire in this unstable mode for a time sufficient to cause extensive damage to the injector. Because of the damage, it could not be determined whether the hardware was defective before instability or whether the instability caused the damage. It was concluded that the instabilities were the result of hardware failure. Additional testing proved that these conclusions were erroneous.

A plan was devised to evaluate the ability of the engine to withstand combustion disturbances without becoming unstable and to characterize the instabilities if they did occur. The lack of hardware caused this plan to proceed slowly. When combustion instability occurred in an ablatively cooled chamber during bomb testing, it was concluded that the injector was not satisfactory for the Apollo vehicle. Later, a spontaneous instability occurred 290 seconds into an acceptance firing, further substantiating the conclusion that the injector design was unsatisfactory.

Numerous modifications were made to the injector in an attempt to solve the stability problem. A photograph of the final configuration (fig. 8) shows the injector and the devices that were used in an attempt to ensure stable operation of the engines. The final configuration exceeded the performance requirement for the engine.

Ablatively cooled thrust chamber. - The original design for the ascent engine defined the thrust chamber as being ablative, with a radiation-cooled nozzle. The radiation-cooled nozzle had been shown to be technically feasible; however, this concept had never been applied to an engine that was buried in the vehicle or that was required to perform fire-in-the-hole starts from another stage (using the descent stage as a launch platform). Evaluation of the possible problems with the radiation-cooled nozzle led to the selection of an ablatively cooled nozzle for the ascent engine, even though the engine weight would be slightly greater. Some of the reasons for abandoning the radiation-cooled nozzle were high design risk, high cost, lack of development experience with buried installations, and modifications required in the test facilities.

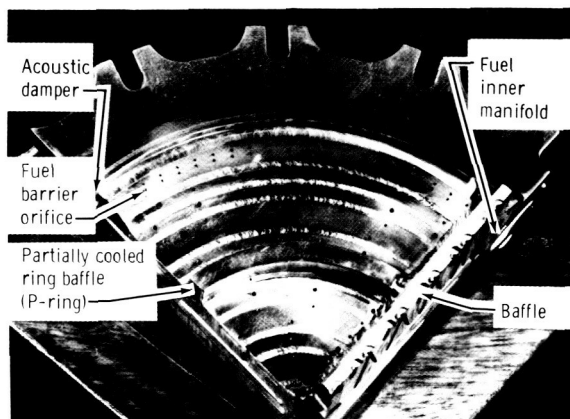


Figure 8. - Stability devices on the production injector.

The total weight of the chamber with an ablatively cooled nozzle was approximately 161 pounds. A weight-saving program caused a change in the material of the thrust-chamber insulation and nozzle ablator and, thereby, the weight was reduced 34 pounds. Test results showed a marked improvement in the ablative characteristics of this chamber and it was certified for use on manned flights.

Bipropellant valve. - The bipropellant-valve concept was a series-parallel ball-valve arrangement that was actuated hydraulically by an electrical solenoid pilot valve. The development testing of the valve was performed at the valve level; however, because of direction to qualify the component at the highest possible subassembly level, qualification testing was performed at the engine level to certify the valve for LM-1. Out-of-specification leakage was observed during qualification testing of the valve actuator. The leakage was acceptable for the unmanned flights, but design changes were incorporated for later vehicles.

As a result of long-term usage of bipropellant valves in oxidizer vapors at the WSTF, it was found that the needle bearings were subject to corrosion. Thus, in mid-1967 the needle-bearing material was changed to stainless steel. In mid-1968, the fuel actuators experienced leakage caused by O-ring twisting during valve operation. Several valves on production engines at the backup-engine contractor plant were rejected. Top-visual-quality O-rings were installed without success. Subsequently, it was determined that the original leakage requirements were too stringent and that the valves were satisfactory under more realistic requirements.

Engine assembly. - Analyses of the heat-shield structural margins during fire-in-the-hole (FITH) testing of the engine indicated a negative margin because of the combustion-chamber-pressure peaks during the start transient of the engine. The contractor investigated several means of reducing the chamber-pressure overshoot without changing the valve design, but all solutions tried were unsuccessful. Therefore, the heat-shield structure on the vehicle was modified to accommodate the higher chamber overshoot during FITH testing. Similar problems were experienced by the backup contractor.

Because of the problems in combustion stability and ablative compatibility associated with the first production injectors, the engine-development program of the original contractor was divided into two phases. The first phase was a test program to certify that the early engines, equipped with early production injectors, were acceptable for flight on the LM-1 and LM-2 unmanned vehicles, and the second phase was a test program to certify that the engine with a stable combustion injector was acceptable for the manned LM-3 and subsequent manned vehicles.

The first phase, certifying the engine for LM-1 and LM-2 vehicles, consisted of one DVT sea-level test engine, five DVT altitude-test engines, and three DVT altitude-qualification-test engines. Two additional problems were noted during this test program. The problems were the ablatively cooled thrust-chamber erosion and leakage in the valve actuator. However, it was concluded from these tests that the engine was satisfactory for the unmanned vehicles.

The second phase of the program consisted of three DVT altitude engines, one DVT sea-level engine, and four DVT altitude-qualification engines. Because the injector-fabrication processes were changed to use electrical-discharge machining

of the orifices, an additional qualification engine and one DVT sea-level engine were added to the test program. Thrust-chamber erosion was observed during these tests.

Backup engine contractor. - The backup contract was initiated and managed by the MSC with close interface with the vehicle contractor. The scope of the contract that was awarded encompassed the design, development, and qualification of a flight injector for the LM ascent engine. This effort also included the assembly, test, and production of complete rocket engines by integrating this qualified injector with thrust chambers and propellant valves furnished by the original engine contractor.

Candidate injectors were selected, tested, and evaluated to evolve a stable, high-performance configuration that was compatible with the ablatively cooled thrust chamber. The basic injector design selected was a baffled, flat-faced, multiring unit with integral propellant-distribution manifolding and acoustic cavities. The three injector-orifice patterns that were evaluated included a triplet, consisting of two fuel streams impinging on each oxidizer stream; an unlike doublet, in which a stream of oxidizer impinged on each fuel stream; and a mixed doublet, in which this mixing of oxidizer and fuel was a secondary result of fuel-fuel and oxidizer-oxidizer impingement.

The unlike doublet was selected as the production injector because of its performance, compatibility, and stability margins. Efforts were directed toward making minor changes in the fuel-film cooling in the basic unlike-doublet injector design to optimize performance and ablative compatibility while maintaining a wide margin of combustion stability. The final configuration exceeded the performance and compatibility requirements.

Because of a combustion instability during bomb testing on an injector that had experienced severe face damage, a "soft center" modification was made to improve stability characteristics. The modification consisted of closing off six of the nine impingement sets at the center of the injector. The change reduced the injection density in the center of the injector and reduced the amount of propellant available for possible enhancement of radial combustion waves.

A complete final-design injector is shown in figure 9, and a section through the propellant inlets is shown in figure 10 to illustrate the flow passages. An early injector inlet screen (filter) is shown in figure 11 installed in the typical manner. As can be seen in figure 11, the filter is a woven-wire-cloth truncated cone with pleats. The filter has a solid plug that effectively serves as a dynamic-head-suppression device, and the filter protects the injector orifices from contamination. The pleated filter, made of all-welded corrosion-resistant steel, with a 120-micron length along two major axes, was designed to stop all particles.

Although the pleated filter successfully completed all the required qualification tests, the contaminant-capacity requirements of the test plan were not consistent with levels of potential contaminant in the vehicle propellant tanks. The potential for pressure drop across the pleated-injector-inlet filters could have a significant effect on the engine mixture ratio and, consequently, on the ascent-stage propellant utilization. For vehicles not landing on the lunar surface, the propellants usually were off-loaded, propellant utilization was not considered critical, and the pleated filters were acceptable for use in engines for these vehicles. A more efficient inlet screen that was to have a

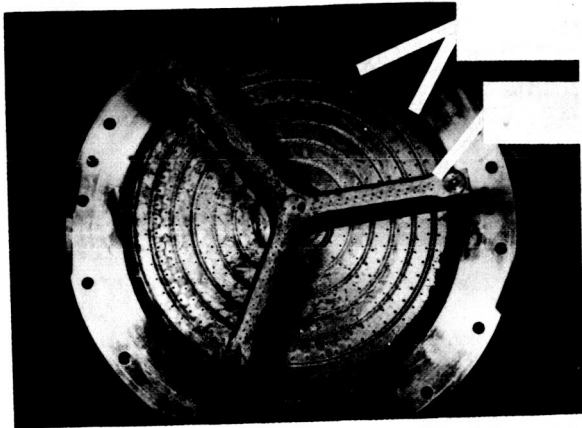


Figure 9. - Final-design injector.

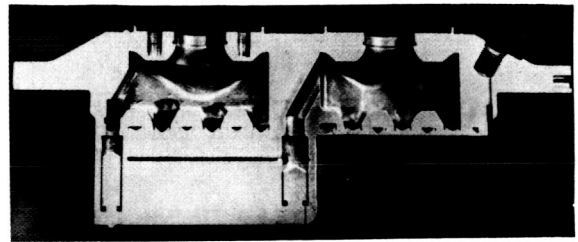


Figure 10. - Propellant-inlet section of the injector manifold.

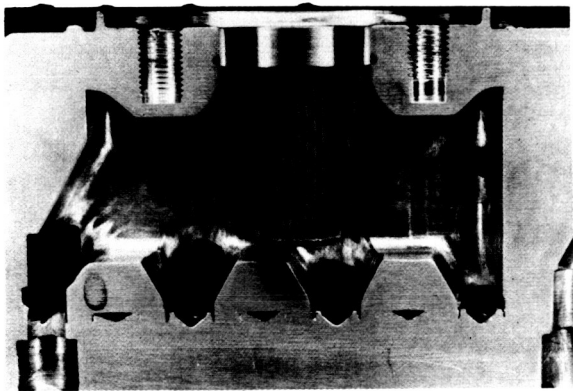


Figure 11. - Injector filter.

lower pressure drop when contaminated was designed for lunar-landing vehicles. The new injector-inlet filter (caged filter) was designed to meet the revised requirements for contaminant capacity and to provide additional structural support for preventing screen deformation.

After qualification tests were complete, a structural failure of a production filter occurred during an engine-acceptance test. The failure was caused by the resistance-weld procedure that was used for attaching the heavy, outer mesh-support basket to the upper collar of the filter. The strength developed by the weld was insufficient to withstand the maximum filter load during service. A modification that eliminated the need for resistance welding was made to the filter design.

The occurrence of preignition-pressure spikes during engine startup required a change in propellant ducting to ensure that the oxidizer would reach the combustion chamber before the fuel. The modified duct is shown in figure 12.

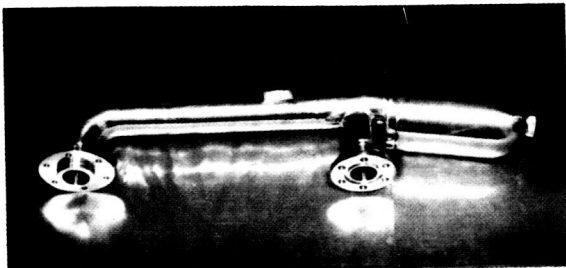


Figure 12. - Modified engine fuel duct.

Formal demonstration of the acceptability of the complete engine for lunar flight was accomplished by testing six rocket-engine assemblies through a qualification series of environmental tests and hot-firing demonstrations. The engines were produced as complete production units and were fully representative of flight engines. Reuse of bipropellant valves during this program was necessitated because of hardware shortage. The acceptance

testing revealed a problem in production fabrication that required a change in the welding procedure to prevent fatigue cracks from developing in the weld attaching the baffle base to the injector face. The change in the welding procedure improved the injector so that it was declared capable of performing the requirements of a manned lunar-landing mission.

The selection of the backup-contractor engine for use on the flight vehicles was made from an evaluation of each contractor production engine. The backup contractor also had complete qualification-testing facilities and had production hardware available for installation on the vehicle when required.

The responsibility for the backup engine was assigned to the vehicle contractor subsequent to the decision to use that engine for the LM. Formal demonstration of the acceptability of the complete engine with the lighter weight chamber, updated valve, and improved engine filters was accomplished by testing four additional rocket-engine assemblies under the direction of the vehicle contractor.

Test Articles and Test Rigs

Several test articles and test rigs were developed and tested to accomplish the APS testing program. The test articles and test rigs contributing to the APS development and qualification are listed in table I.

TABLE I. - MAJOR TEST ARTICLES AND RIGS

Development phase	Test article or rig	Remarks
Breadboard	Pressurization modules	
	BA-1	Used at the factory with test rig HA-1
	BA-2	Used at the original engine-contractor facility with test rig HA-2
	BA-3	Cold-flow tests at the factory
	BA-4	Used at the WSTF with test rig HA-3

TABLE I. - MAJOR TEST ARTICLES AND RIGS - Continued

Development phase	Test article or rig	Remarks
Prototype	Propellant storage and feed system test rigs	
	HA-1	Used at the factory for cold-flow testing
	HA-2	Used at the original engine-contractor facility for engine development tests
	HA-3	Used at the WSTF for FITH and engine development tests
	HA-4	Used for engine development tests
	Pressurization modules	Used mainly on PA-1
	PRA-1	
	PRA-2	
	PRA-3	
	PRA-4 PRA-5 PRA-6	
Installed test articles	Propulsion system test rigs	
	PA-1	Used at the WSTF for propulsion system certification
	PA-2	Canceled
	LM test articles	
	LTA-1	Electrical and EMI tests of the propulsion components
	LTA-2	Configured for flight test, flown on AS-502

TABLE I. - MAJOR TEST ARTICLES AND RIGS - Concluded

Development phase	Test article or rig	Remarks
Full-configuration test article	LTA-3	Used for structural and vibration tests to define component vibration levels
	LTA-4	Environmental tests in the factory vacuum chamber
	LTA-5	Descent propulsion rig at the WSTF
	LTA-6	Canceled
	LTA-7	Canceled
	LTA-8	Thermal tests at the MSC to define thermal limits
	LM-2	Used in drop tests at the MSC. Ascent stage sent to Osaka, Japan, for Expo '70. It was transferred to the Smithsonian following Expo '70.

Breadboard testing. - The breadboard testing was initiated to achieve early system verification of design, interaction of components, and technical feasibility of design. Four heavyweight test rigs were built; these rigs were essentially open structures upon which the various propellant-feed-section components and the pressurization modules could be mounted in the same arrangement and relative locations as on the LM. The rigs were used for cold-flow testing at the prime-contractor facility, for testing at the engine-contractor facility in the early engine-development program, for helium-pressure-blowdown tests in the prime-contractor cold-flow facility, and for system testing with preproduction engines and components at the WSTF.

Prototype testing. - Most of the prototype testing of the APS was conducted on the WSTF test rig PA-1. This test rig was an ascent-stage structure that incorporated the APS and RCS equipment in an approximate flight configuration (fig. 13). Simulation was used except where it would affect the APS performance. The test rig accommodated both the pressure-fed, 3500-pound-thrust ascent engine for firing in a downward position and the 16 RCS engines, using a separate propellant-supply system interconnected with the APS propellant tanks. The PA-1 ascent stage was constructed of aluminum alloy and consisted of three major sections: the forward cabin, the mid-section, and the aft equipment bay. The structure included provisions for installing

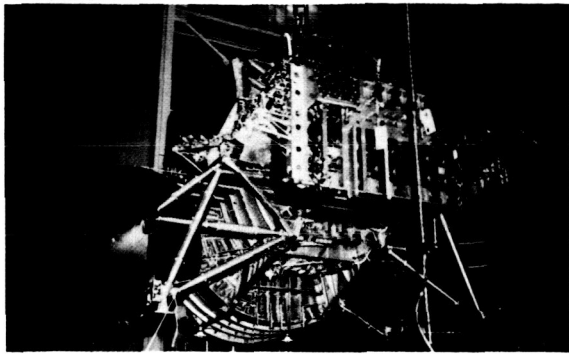


Figure 13. - Test rig PA-1.

heat shields for the FITH testing. A summary of the tests conducted on test rig PA-1 is given in table II. These tests included all normal mission requirements and a number of off-limit tests of possible problem areas such as FITH tests, pressure overshoots, and component-abort tests. Although all of these hot-firing tests were not conducted with a qualified engine injector, the tests proved successfully the integrity of the propellant and pressurization sections.

TABLE II. - THE PA-1 WHITE SANDS TEST FACILITY TEST SERIES SUMMARY

Test series	Number of runs	Series features
1	(a)	Cold-flow (substitute and actual propellants)
2	34	FITH starts, abort, and normal mode starts, prepressure balance and unbalance, compatibility of APS with RCS
4	45	LM-1 mission duty cycle, restarts, engine stability, propellant depletion, off-nominal performance
5	(a)	Cold-flow (substitute and actual propellants) LM-1
6	9	Sea-level firings, LM-1 mission duty cycle (2), pressure regulator evaluation, abort, and normal and off-limit starts
7A	7	FITH starts, abort, and normal mode starts
7C	(a)	Cold-flow (substitute and actual propellants) LM-3
8R	20	Mission duty cycle to depletion, off-nominal starts
8	5	Mission duty cycle with APS/RCS, heat-soak-back temperatures, helium-saturated-propellant evaluation
8A	9	Propellant depletion, pressure regulator malfunction, abort starts, propellant utilization
5A	7	Propellant utilization with off-nominal conditions
9	4	Blowdown mission duty cycle, propellant utilization, cold start
7B	10	FITH start evaluation, adequacy determination of thermal protective blankets and blast deflector
11	2	High-mixture-ratio mission G duty cycle demonstration

^aNo hot firing during cold-flow tests.

With flight-qualified engines, 57 tests were conducted over a total firing time of 3392 seconds. Five engines and six thrust chambers (three heavyweight and three lightweight) were used. The following are the results of these tests.

1. The mixture ratio of the ascent engine could be predicted within 0.6 percent for engine operation at chamber pressures of 112 to 130 psia and temperatures of 40° to 100° F with helium-saturated and unsaturated propellants.
2. Combined with RCS operation, the overall APS and RCS mixture ratio could be predicted within 0.75 percent.
3. The validity of the propellant-feed-system cold-flow calibrations was verified, and the flight-engine characteristics were confirmed.
4. The FITH starts, at simulated lunar-launch conditions, were performed with no structural damage or adverse effect on engine performance.
5. Abort starts were performed safely at ullage pressures of 62 to 215 psia.
6. Engine operation was not affected adversely by operation on redundant regulators.
7. The transition to the adjusted system-pressure levels was smooth and gradual.
8. The engine operated safely in the tank-ullage-decay (blowdown) mode from nominal chamber pressure to 8 psia, and it could be safely shut down by means of propellant depletion.

SYSTEM CHECKOUT

During the functional checks on LM-1 and LM-3, the relief-valve burst disks ruptured prematurely. Investigation revealed that the burst-disk design would not allow any backpressure on the disks without premature rupture. As a result of these failures, the checkout procedures were reviewed and corrected in an attempt to preclude this occurrence on other vehicles. In spite of the precautions taken, premature rupture of relief-valve burst disks remained a checkout problem throughout the program.

Quadruple check-valve leakage was a common checkout problem, and most vehicles required special work and testing to correct the leakage. Either purging to correct the leakage, valve replacement, or waivers to accept the leakage were required on most vehicles. The many debrazing and brazing operations that were required in order to replace the check valves also introduced contamination into the system. As a result, complete helium-module replacement was necessary (for instance, on LM-3). Design changes were considered to improve the checkout problem but were not implemented.

Early vehicle checkout was plagued by component replacement resulting from failures of the components either during checkout or elsewhere in the program (such as in qualification system tests). Particularly, failure was frequent on the helium regulators, the helium solenoid valves, the helium relief valves, and the engine.

FLIGHT PERFORMANCE

The flight program for the LM consisted of one unmanned flight (LM-1). The LM-2 vehicle was planned for an unmanned flight, but this was canceled and LM-2 was used as a ground-based test article. The first manned vehicle was LM-3, which demonstrated successful operation in earth orbit. The LM-4 vehicle was used in lunar-orbit-operation tests, and LM-5 was the first lunar-landing vehicle.

Apollo 5 Mission

Apollo 5 (LM-1) was the first mission with a flight-configuration lunar module. The primary objectives of the Apollo 5 mission included verifying the APS and the abort-staging function for manned flight. An engine from the original engine contractor was installed in the APS.

As a result of the premature shutdown of the descent engine, an alternate mission plan was selected that resulted in two ascent-engine firings. The first firing was of 60 seconds duration. Approximately 1-1/2 hours later, the engine was again commanded on and fired to propellant depletion. The total ascent-engine firing time for the mission was approximately 40 seconds less than predicted. At least 20 seconds of this time can be attributed to higher-than-expected propellant usage by the RCS engines through the propellant interconnect as a result of the control configuration after staging. Another 10 seconds of this time was caused by propellant slosh. High vehicle-attitude rates caused the propellant-tank outlet ports to be uncovered prematurely. An oxidizer-depletion shutdown had been expected. However, when the propellant in the feedlines (sufficient for approximately 1 second of nominal operation) was depleted, as indicated by the engine-interface pressure, helium was ingested into the oxidizer and the fuel lines almost simultaneously, causing thrust decay. As indicated by the abbreviated "on" time between low-level sensor uncovering and thrust decay, an additional 10 seconds of normally usable propellant was in the tanks at thrust decay, but was unavailable because of the sloshing and high vehicle-roll rates.

The start transient and the beginning of steady-state operation showed high-amplitude chamber-pressure oscillations on both the first and the second engine starts. The 400-hertz oscillations, which occurred immediately after the start-transient overshoot, were characteristic of the original-contractor engine during ground-based testing and were expected on this flight. The oscillations appeared to be a form of low-frequency instability, caused by the coupling of the combustion process with the resonant frequency of the engine propellant-feed system. Ground-based tests were indicative that gaseous helium, dissolved or entrained in the fuel, tended to induce coupled instability.

The high-magnitude 400-hertz oscillations resumed approximately 4.5 seconds before thrust decay, an indication of propellant gas entrainment caused by the vortexing or sloshing resulting from the high vehicle-attitude rates that occurred during this portion of the mission. The thrust decay appeared smooth; no spikes or other detrimental effects were apparent after the damping of the 400-hertz oscillations and the initiation of the chamber-pressure decay.

The engine valves should close immediately after fuel depletion because the valves are fuel-actuated. Following depletion, the engine-valve-position indicators showed that the engine valves did not start to close until 85 seconds after the thrust started to decay and were not closed completely when the telemetry signal was lost by the ground station 58 seconds later. System pressures and temperatures indicated that some blockage of fluid flow occurred between the propellant tanks and the engine after the initial thrust decay. The blockage was most likely caused by small amounts of propellant that were frozen by cold helium ingested into the feedlines. Although it was not a normal depletion shutdown, no hazardous or detrimental effects were apparent.

The performance of the ascent engine was not verified to have been within the expected accuracy because of the loss of ΔP (change in pressure from tank outlet to engine interface) flow-rate measurements and because of the high roll rates of the vehicle during propellant depletion; however, the engine-pressure measurements and the vehicle velocities that were obtained indicated that the ascent-engine performance was within the nominal predicted tolerances.

Apollo 9 Mission

The Apollo 9 mission (LM-3) included the first manned flight test of the LM. The mission was the second manned flight test of the Saturn V launch vehicle and the third manned flight of the Block II command and service module (CSM).

The overall mission objective was to evaluate crew operation of the separated LM and to demonstrate docked-vehicle functions in an earth-orbital mission, thereby qualifying the combined spacecraft for lunar flight. The LM operations included an ascent-engine firing to propellant depletion after final separation from the CSM.

The APS was used for two firings: a 3-second firing while the ascent stage was manned and an unmanned firing to propellant depletion. The LM was out of ground-tracking-station range during the first ascent-engine firing; therefore, no data were available. However, when data were first acquired after the firing, system pressures and temperatures were normal. The second ascent-engine firing was initiated successfully and lasted for 362.3 seconds. During the second firing, system pressures were lower than expected for the first 290 seconds, thus indicating a malfunction in the primary regulator. The lower operating pressures produced no undesirable effects in the system. The second ascent-engine firing was terminated by the planned oxidizer depletion. The engine was commanded off approximately 10 seconds later. The depletion shutdown appeared nominal in all respects. The transient characteristics that the engine demonstrated during the oxidizer-depletion shutdown mode were investigated and were found to compare favorably with ground-based test data. All applicable transient-specification requirements were met.

A detailed assessment was made of the lower-than-expected system pressures to determine the cause and magnitude of the malfunction. After extensive testing in the thermochemical test area at the MSC and on test rig PA-1 at the WSTF, it was concluded that the most probable cause of the anomaly was contamination of the regulator during the solenoid-valve replacement. A review of regulator backflow and system-repair procedures during checkout on LM vehicles indicated the possibility of minor backflow on vehicles through LM-7. Cautions were added to the changeout procedures, and special instructions were given to personnel. Increased surveillance of regulator checkout data was maintained by the subsystem office for all subsequent vehicles.

Apollo 10 Mission

The Apollo 10 mission was the first flight test of the LM in a lunar environment and the second manned LM flight. Because the mission objective was to show a fire-to-depletion firing and not a ΔV , the amount of propellant loaded was approximately 50 percent that of the nominal lunar mission.

The overall mission objective was to duplicate, as closely as possible, a lunar-landing mission, with the exception of lunar landing and lift-off. Inherent in this objective was the performance of the APS during the lunar-orbit-insertion maneuver. Also included as mission objectives were verifications of LM operation in a lunar environment and of mission support of all spacecraft at lunar distances.

Docking of the CSM with the LM and separation of the docked vehicles from the Saturn IVB occurred 4 hours after launch. Undocking of the LM from the CSM in lunar orbit occurred 98.5 hours after launch. Approximately 2 hours after completion of the descent propulsion system firing, the descent stage was separated and the APS engine was fired for 15.6 seconds. Upon completion of this insertion maneuver, the ascent stage was docked with the CSM, and the crew and equipment transfer was accomplished. Approximately 6 hours later, the ascent stage was separated, and the engine was ignited for the 248.9-second burn to propellant depletion.

Shortly after the ignition signal for the first APS burn, the ascent-engine-quantity caution light came on, triggering a master caution and warning alarm. The ascent-engine-quantity caution light was controlled by the oxidizer and fuel low-level sensors in the propellant tanks. Approximately 1 second after the ascent engine "on" signal for the manned LM-4 APS burn, the oxidizer low-level sensor was activated for approximately 1 second while the master alarm was on for 2 seconds before being manually reset. Low-level sensors in both oxidizer and fuel tanks operated as expected during the remainder of the first burn and the entire second burn. Based on this performance, it was concluded that the signal received during the first burn was valid.

The cause of the anomaly was believed to be insufficient settling of APS propellants before the burn. The LM-4 propellant tanks were filled to 50 percent, the least on any flown vehicle. Based on information contained in the Spacecraft Operation Data Book, an RCS ullage-propellant-settling maneuver of 3 to 4 seconds in duration would be adequate for a vehicle with the propellant load of LM-4. The actual RCS ullage-propellant-settling maneuver before APS first-burn ignition lasted for a period of 4.1 seconds. The propellant-settling maneuver was reexamined, and the settling maneuver duration was increased for increased ullage (decreased propellant) volume.

Apollo 11 Mission

The Apollo 11 mission (LM-6) was the third manned flight of the LM, the fifth manned flight of the Block II CSM, and the fourth manned flight of a Saturn V launch vehicle. The primary objective of the mission was to perform the first manned lunar landing and return.

Burn time for the APS lunar lift-off maneuver was 434.9 seconds. Upon completion of the insertion maneuver, the ascent stage docked with the CSM, and the crew transfer and equipment transfer were effected. Approximately 3.5 hours later, the ascent stage was jettisoned. All APS performance parameters were well within their respective limits.

CONCLUDING REMARKS

In review of the ascent propulsion system development, several areas that could enhance the development of future subsystems are discussed in the following paragraphs.

Definition of Realistic Requirements

Requirements for the system and its components should be thoroughly defined before the beginning of the design phase. Typical examples of inadequate definitions at the beginning of the design phase were the stability requirement for the engine, the propellant-vapor compatibility requirement for the pressurization components, and the leakage requirements. Late definition of the stability and compatibility requirements caused redesign, which was reflected by cost increase and schedule slippage. Overly stringent leakage requirements caused failures and redesign efforts that were not necessary.

Flexible Program Flow Plan

Early in the testing program, the prime contractor and vendors were working to a cost-incentive flow plan. As a result, vendors directed their efforts to delivery of components rather than to problem solving and hardware development. This policy resulted in failures late in the development phase and in component replacement on the ascent propulsion system. This problem was most apparent in the engine and regulator component area. A more flexible flow plan would have allowed component installation at later opportunities.

Leakage and Contamination Control

Lunar module 1 had many leakage problems caused by overly stringent leakage requirements, the design of the system, and nonstandard measurement techniques. The stringent leakage requirement was initiated by the uncertainties and by the

corrosive nature of the oxidizer vapor. When stringent leakage control was required, design changes (such as installing redundant seals with the capability to measure the integrity of the primary seal) were necessary. In addition, leakage-measurement techniques were different at the vendor facility, at the prime-contractor facility, and at the NASA John F. Kennedy Space Center. Attempts were made to standardize leakage-measurement techniques to ensure the ability to measure changes in leakage. Where stringent leakage control may be required, it is recommended that leakage-measurement techniques be standardized.

Also, contamination was a problem (especially with check valves). This problem was magnified by the many component replacements. Where contamination control is critical, it is recommended that all functional components (such as regulators and check valves) have built-in filters.

Inspection and Manufacturing Repeatability

The squib valve, the helium solenoid valve, the relief valve, and the engine underwent problems of manufacturing repeatability. The main cause of the problem was internal braze joints or welds that could not be inspected. In the engine, the injector, when fabricated, had cracks in the electron-beam welds. These cracks were caused by improper control of the welding parameters. Hardware should be designed to permit inspection of weld- and braze-joints. When this practice is not feasible, the weld parameters should be certified and supported by means of test samples from all qualified welders.

Manned Spacecraft Center

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