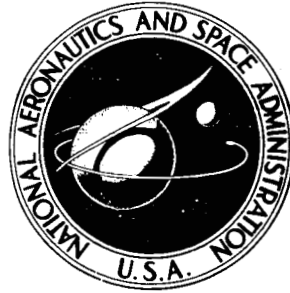


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APOLLO EXPERIENCE REPORT - DESCENT PROPULSION SYSTEM

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16. Abstract The propulsion system for the descent stage of the lunar module was designed to provide thrust to transfer the fully loaded lunar module with two crewmen from the lunar parking orbit to the lunar surface. A history of the development of this system is presented. Development was accomplished primarily by ground testing of individual components and by testing the integrated system. Unique features of the descent propulsion system were the deep throttling capability and the use of a lightweight cryogenic helium pressurization system.			
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APOLLO EXPERIENCE REPORT

DESCENT PROPULSION SYSTEM

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SUMMARY

The propulsion system for the descent stage of the lunar module was designed to transfer the fully loaded lunar module containing two crewmen from the lunar parking orbit to the lunar surface and to hover above the lunar surface for exact landing-site selection. To accomplish this maneuver, a propulsion system was developed that used hypergolic propellants and a gimbaled pressure-fed ablatively cooled engine that was capable of being throttled. A lightweight cryogenic helium pressurization system also was used. During the developmental and operational phases of the program, significant problems were encountered with long-term propellant compatibility, seal leakage of the propellant system, the supercritical helium system, and interfaces with the guidance and navigation system.

INTRODUCTION

The mission requirements for the Apollo descent propulsion system (DPS) were to provide the necessary change in velocity (ΔV) for transfer of the lunar module (LM) from lunar orbit to descent orbit and for the powered descent from descent orbit to lunar-surface landing. A secondary mission requirement was to provide contingency abort capability for return to earth if the service propulsion system (SPS) in the command-service module failed during translunar and prelanding lunar-orbit phases of the mission. The history of the DPS design and development covers a period of approximately 7 years from conception to the first manned lunar-landing mission (Apollo 11). The technical problems encountered during the development of the DPS and the solutions developed for the problems are presented.

MISSION REQUIREMENTS

Initially, the DPS requirements were defined by a 26 000-pound LM separation weight and a ΔV requirement of approximately 6000 fps. Later, as the vehicle design was defined better, the requirements were increased to approximately a 33 000-pound LM separation weight and a 7000-fps ΔV requirement for landing.

The nominal mission requirements included a passive standby of the DPS from launch through translunar coast and lunar-orbit coast (approximately 100 hours), a burn to accomplish a Hohmann transfer from a 60-nautical-mile circular lunar orbit to a 60-nautical-mile by 50 000-foot elliptical descent orbit, and a powered-descent burn from 50 000 feet to lunar-surface landing. The powered-descent burn consisted of a variable-thrust profile (fig. 1) that dictated the requirement for an engine capable of being throttled. The hover phase included the last 1000 feet of descent to the lunar surface. Because the mission was to be manned, very high system reliability was

another design requirement. Also, the DPS was required to provide contingency mission-abort capability if loss of the SPS occurred during translunar coast, lunar-orbit insertion, or early lunar-orbit coast.

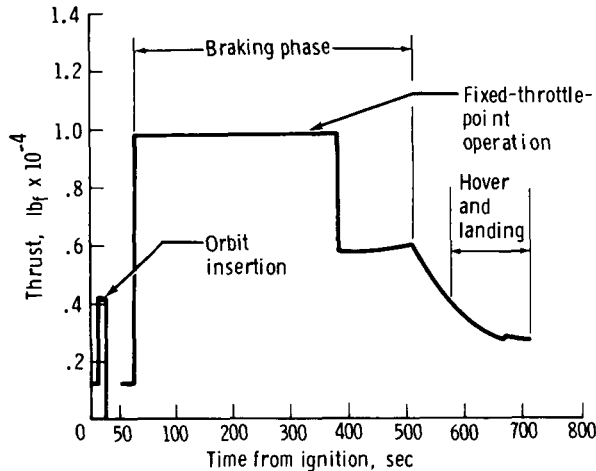


Figure 1. - Powered-descent variable-thrust profile.

To achieve the optimum design for the mission requirements (reliability, performance, and weight), a pressure-fed storable-propellant propulsion system using an ablatively cooled engine was selected for the DPS. In addition to the throttling requirement for the terminal phase of descent, the DPS engine was required to be gimballed to provide thrust-vector control. The propellants selected were nitrogen tetroxide (N_2O_4) (oxidizer) and 50 percent unsymmetrical dimethyl hydrazine (UDMH) and 50 percent hydrazine (fuel).

SYSTEM DESCRIPTION

The initial DPS design was defined in 1963 (fig. 2). Later, as the system reliability, weight, and component designs were refined, the DPS design changed (fig. 3). An explanation of these changes is discussed in this paper.

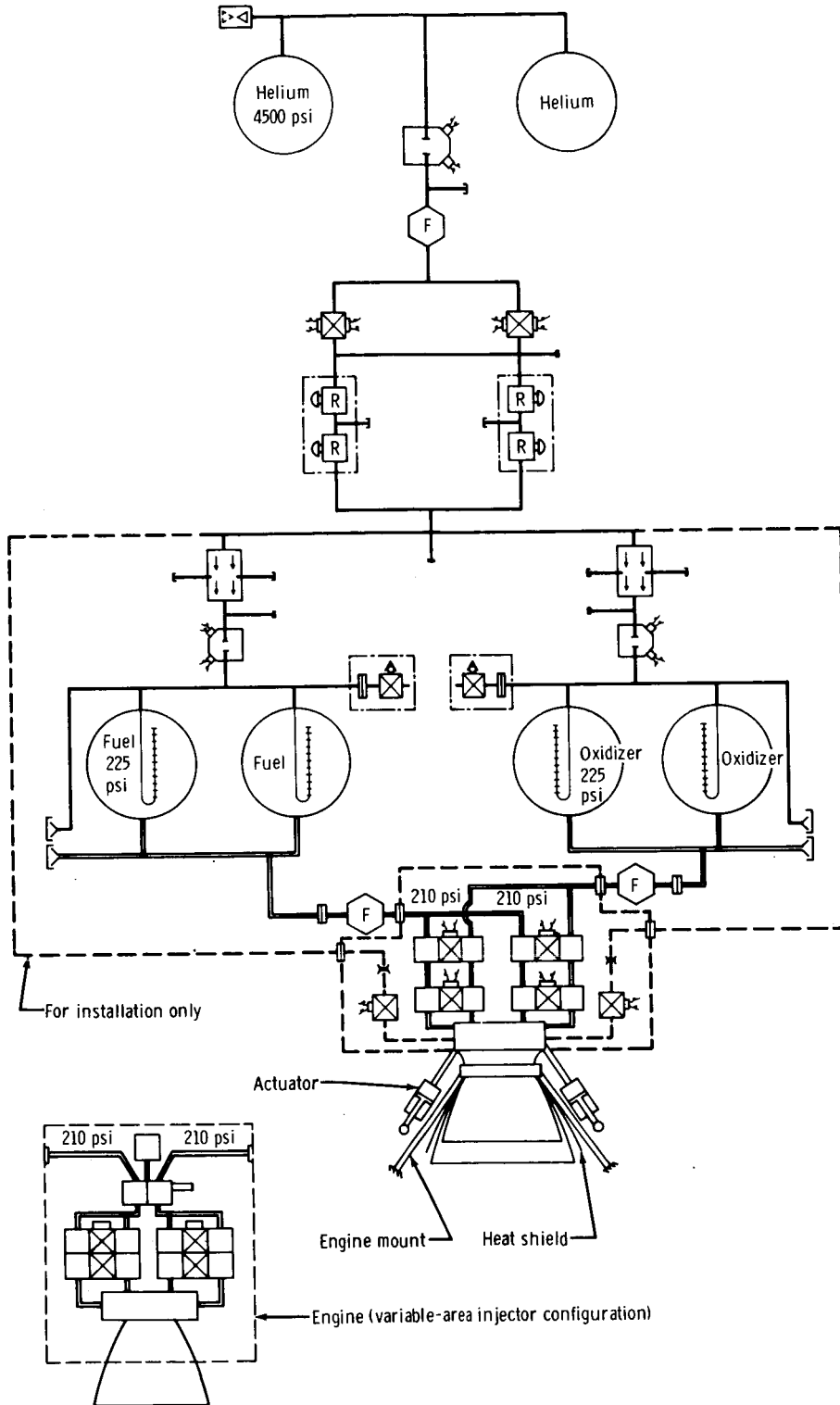


Figure 2. - The initial DPS design.

Pressurization System

The initial concept of the DPS pressurization system used helium stored in two high-pressure tanks with pressure control and supply components (fig. 2). Helium was to be stored at 4500 psia but later was changed to 3500 psia to optimize system weight.

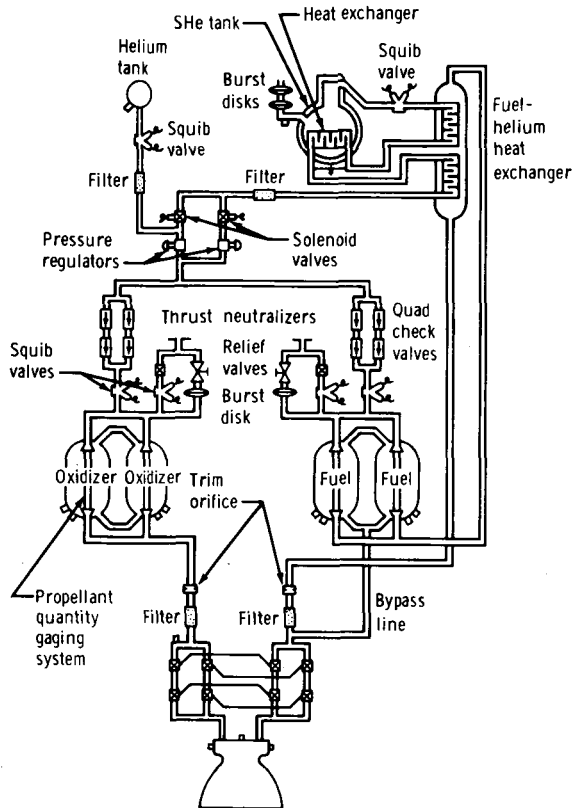


Figure 3. - The final DPS design.

the helium was routed through a second pass in the fuel-to-helium heat exchanger to raise the helium temperature to nearly ambient conditions (approximately 40° F) before it was supplied to the pressure regulators in the pressurization system.

By late 1964, the analysis and feasibility testing of the SHe system indicated that it was operationally feasible and that a weight savings of 280 pounds could be realized by using the SHe system rather than the ambient-storage system. In February 1965, a decision was made to redesign the DPS pressurization system to incorporate an SHe storage tank (fig. 3). Because the pressure in the SHe storage tank increased with time, a minimum-required standby time was defined as 131.5 hours from prelaunch toff until first usage in the nominal lunar landing.

As a result of developmental problems and reassessment of reliability requirements, a decision was made in October 1966 to use only parallel redundancy in the pressure regulation system (single regulators in each leg). The rationale for this decision was that if a wide-open regulator failure occurred, the propellant tanks would be protected from overpressurization by the relief valves, and the crew, using the latching solenoid valve, could eventually isolate the failed regulator. This factor would save the system, and the mission could be continued on the second regulator or could be aborted with the ascent propulsion system. To achieve required engine performance, another design change was made to increase the regulated pressure level from 210 psia to 246 psia.

During design and development of the SHe pressurization system, freezing of the fuel in the fuel-to-helium heat exchanger was found to occur during the start sequence under certain start conditions. The freezing condition was caused by the flow of a substantial amount of helium needed to bring the propellant tanks from prepressurization levels to regulator lockup pressure conditions, during which time no fuel was flowing through the fuel passages of the heat exchanger. A study was made of various systems to alleviate the flowing of cold helium with no fuel flow. An ambient-helium prepressurization start bottle and an electrical heater system were the two main methods considered. The use of an ambient-helium prepressurization start bottle was selected for overall simplicity and reliability.

Propellant Storage and Feed System

The LM vehicle configuration used four propellant-storage (two oxidizer and two fuel) tanks for optimum center of gravity (c.g.) and weight control. The identical propellant tanks were fed in parallel to minimize c.g. excursion during propellant usage. Balance lines were provided at the top and bottom of the two identical tanks to ensure balanced propellant flow from each tank. For overpressurization protection, a burst-disk assembly and relief-valve assembly were provided in the pressurization line leading to each pair of tanks. The burst-disk assembly was in series with the relief-valve assembly to provide a positive seal against leakage until actuation of the relief system was needed. In the initial design of the burst disk and relief valve, an integral assembly was used that incorporated a frangible disk and cutter arrangement (fig. 4). Because the burst pressure for this design was not repeatable, the maximum tank pressure was 308 psia. The burst-pressure rating of this design disk was 288 ± 20 psia, which was based on lot-sample testing and qualification testing. The safety margin of the propellant tank (1.3 safety factor to burst) was limited severely with this design. Also, the fracture-flaw growth potential caused by pressure cycling (fracture mechanics) was more critical than normal, thereby limiting the allowable pressure cycles on the tanks during checkout, servicing, and flight.

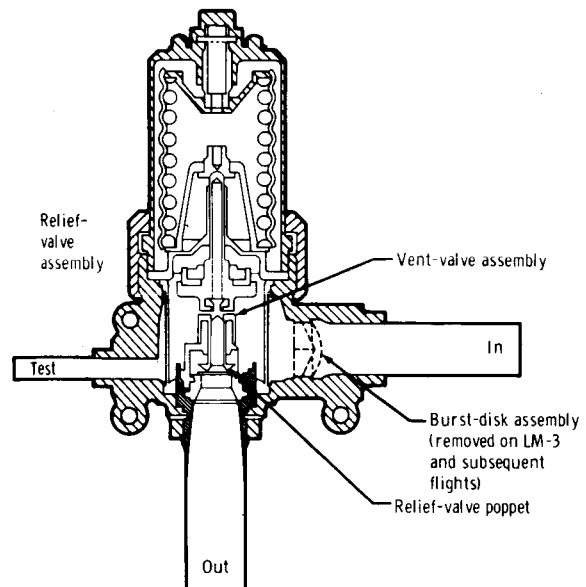


Figure 4. - Burst-disk and relief-valve assembly.

A newly designed burst disk (fig. 5) with a burst-pressure setting of 267.5 ± 7.5 psia was selected to improve the design safety margins and pressure-cycling limitations on the propellant tanks. Another desirable feature of the newly designed burst disk was the ability to withstand

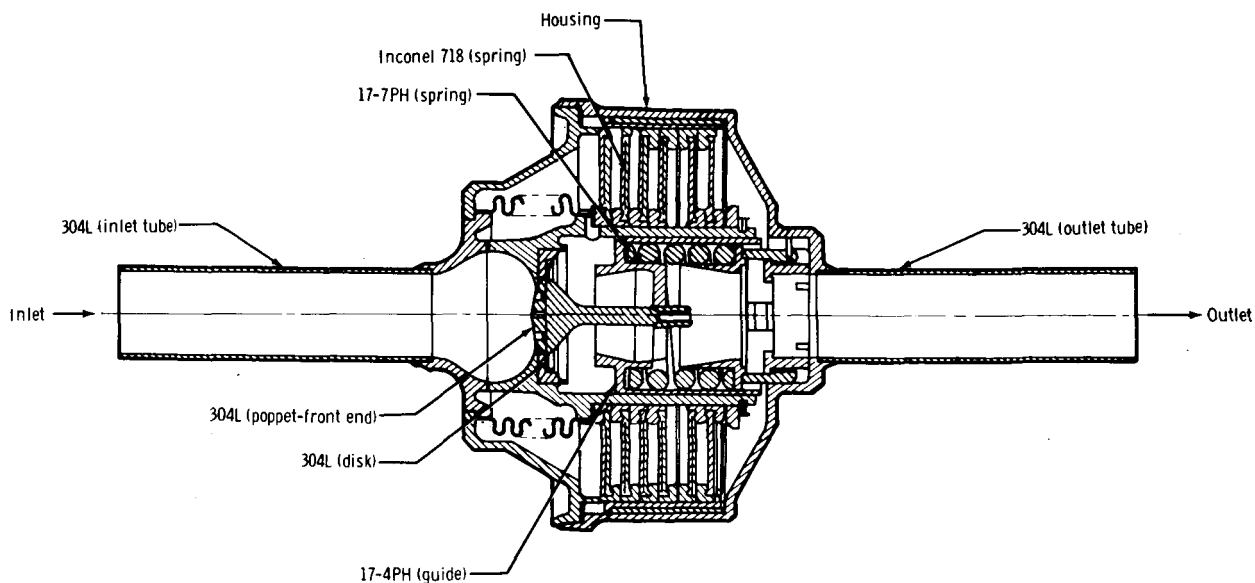


Figure 5. - The new burst-disk assembly.

reverse-pressure differentials (approximately 150 psid) without incurring damage. Before selection of the new design, many failures of the old burst disk occurred in checkout as a result of inadvertent reverse pressure.

During the design of the DPS and the evaluation of the operation of the DPS during a lunar mission, a determination was made that heat soakback from the engine after lunar landing could cause the propellant tanks surrounding the engine compartment to warm, thus causing the pressure to rise to burst-disk and relief-valve pressures. Heat soakback could have caused venting of propellant vapors through the relief valves during astronaut extravehicular activity on the lunar surface, thus spraying the astronauts with propellant vapors. As a result of this possibility, a propellant-tank lunar-dump system was added to each propellant-storage system. This system consisted of an explosive valve and latching solenoid valve in series in a bypass line around the burst-disk and relief-valve assemblies. Actuation of the explosive valve and latching solenoid valves was controlled from the LM crew station.

In each of the four propellant tanks, a propellant-quantity gaging probe provided continuous monitoring from 0 to 95 percent of the tank capacity. Integral to the propellant-quantity probes was a propellant low-level detector that provided a crew warning of impending propellant depletion as the propellant quantity reached a discrete value of 5.6 percent in any tank.

Descent Engine

The basic design requirements for the descent engine were as follows.

1. Throttling ratio: 10:1
2. Maximum-rated thrust: 10 500 lb
3. Gimbaling: $\pm 6^\circ$ in the Y and Z axes
4. Duty-cycle life: 1000 sec
5. Pressure fed: 210 psia (later changed to 246 psia)
6. Propellants: N_2O_4 (oxidizer); 50 percent UDMH and 50 percent hydrazine (fuel)
7. Specific impulse (end of duty cycle): $305 \text{ lb}_f\text{-sec}/\text{lb}_m$

The descent engine was located in the central bay of the descent stage and was partially buried (up to 16:1 expansion ratio). Because of this location, an engine design with sufficient cooling was needed to prevent overheating of the surrounding propellant tanks during engine operation. In the initial phases of engine-design definition and development, two different throttling concepts were considered.

One throttling concept used a fixed-area injector with helium injection at reduced thrust to maintain adequate combustion efficiency. Propellant-flow variation was controlled by noncavitating throttling valves that used system fuel pressures (fig. 6) to actuate the hydraulic servocontrol valves.

The other throttling concept used a variable-area injector combined with cavitating-venturi propellant-flow-control valves. This injector was a coaxial center-element design. A single movable sleeve was used to modulate the injector-fuel and oxidizer-flow area. This injector sleeve was linked mechanically to the two cavitating-flow-control valves and the electrically driven throttle-actuator assembly.

After approximately 18 months of parallel engine development by two different contractors using these two throttling concepts, the variable-area-injector throttling concept was selected (January 1965). The final design configuration of this injector is shown in figure 7. The basic reasons for this selection were that the fixed-area injector with helium injection experienced significant problems with combustion instability during development, and reduced-thrust performance was lower than that with the variable-area injector.

Maximum-operational thrust occurred at a fixed throttle point (FTP) that was calibrated for optimum-mixture ratio and injector pressure difference in settings. The FTP was optimized at 92.5 percent of maximum-rated thrust (10 500 lb). The initial design attempted to provide variable thrust near the maximum-rated thrust; however, excessive problems with mixture-ratio control and throat erosion caused the selection of an FTP maximum-thrust setting, which was compatible with mission requirements.

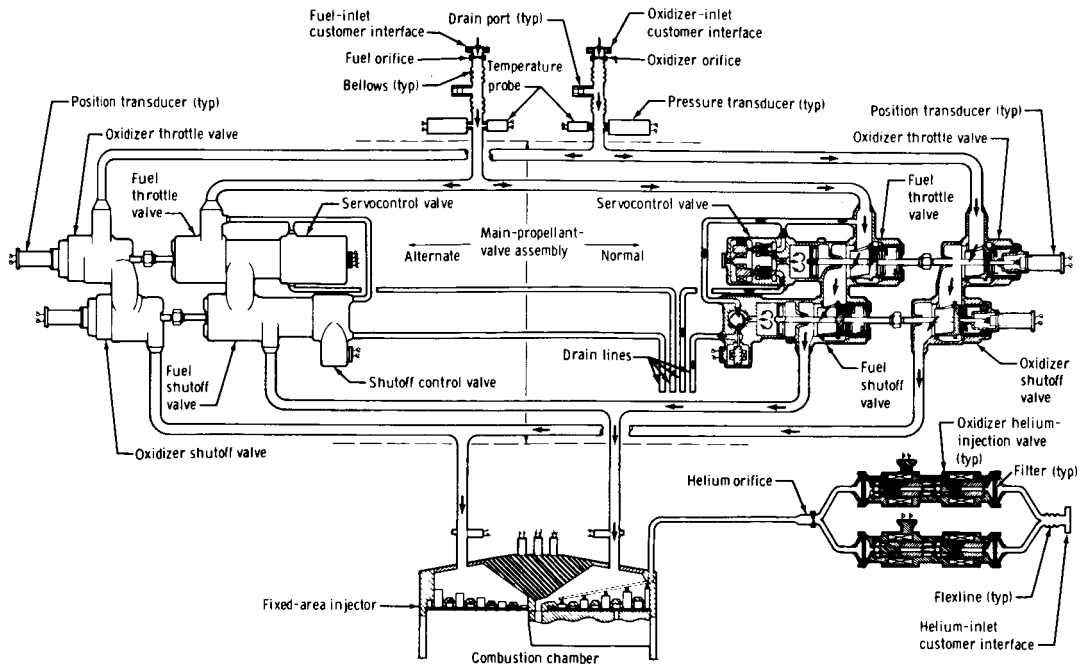


Figure 6. - Descent-engine schematic, fixed-area helium injection.

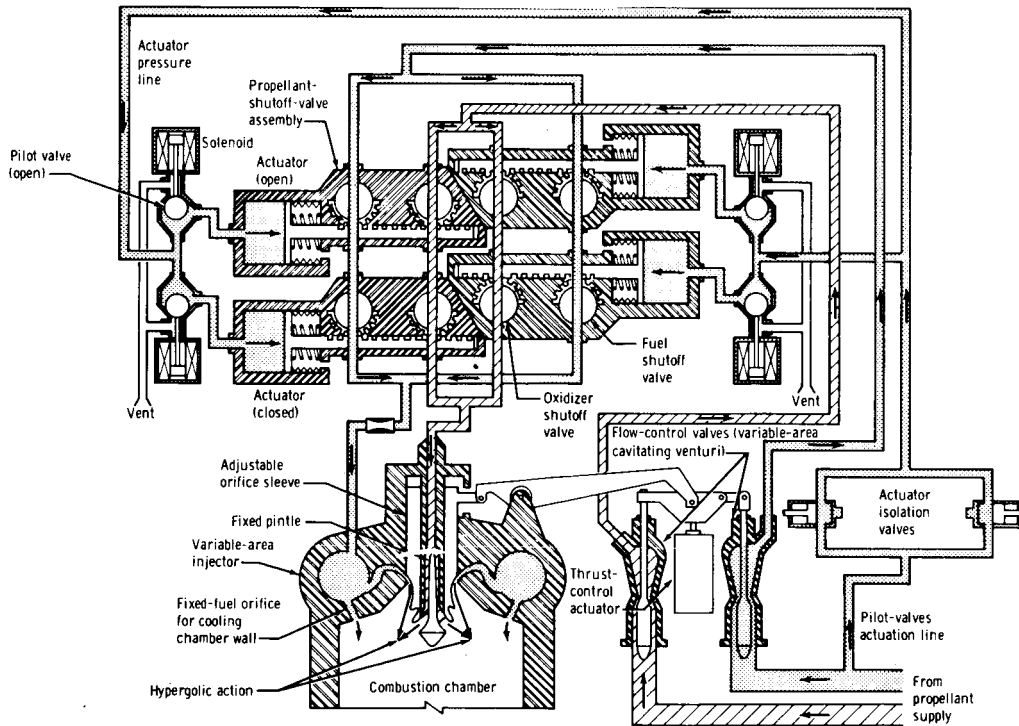


Figure 7. - Variable area of the descent engine.

Thrust between 65 and 92.5 percent was considered a nonoperating region, although the engine can be physically throttled to any point in that region. The minimum-throttle point was 10 percent of the rated thrust.

The fuel-actuated shutoff valves were parallel-series redundant ball valves to provide fail-safe capability. The fuel and oxidizer injector manifold was sized to achieve an oxidizer lead during the start transient.

The thrust chamber was ablatively cooled to an area ratio of 16:1. The nozzle extension was a sheet-metal columbium-alloy skirt extending from 16:1 to 47.5:1 area ratio. Cooling of the nozzle extension was by radiation.

The throttle actuator was a triple-redundant, electrically driven device. Accomplished by an electrical-null-balance method, direct-current command signals were compared to throttle-position indication and throttle-position control. A more detailed description of the throttle actuator is provided in reference 1.

DEVELOPMENT AND QUALIFICATION

The development and qualification of the DPS in support of the first lunar-landing mission covered a period of approximately 6 years from August 1963 to April 1969. This period included component-level and system-level development and qualification. In many cases, preproduction configurations of components were used in early system-level developmental testing. In the developmental and qualification testing of components and systems, extensive design-limits tests, off-limits tests, and malfunction tests were used to determine potential design deficiencies and to document operational limits of the system.

Pressurization System

Developmental tests of the SHe system were initiated with an early design-feasibility-test rig for evaluation of the flight-system and the ground-support-equipment (GSE) design. Caused by the unique SHe tank-loading requirements, development of the SHe servicing system was an integral part of the overall development of the SHe system.

Cryogenic-helium storage vessel. - The design and development of the SHe flight tank was initiated by the vendor in May 1965. In the early developmental phase, a preproduction tank configuration was used that had a slightly larger internal volume and that incorporated an unsupported (other than inlet and outlet tubes) coil-type internal heat exchanger.

The principal design deficiency in the preproduction tank was the internal heat-exchanger configuration. Because of the relatively unsupported configuration of the heat exchanger, it was very sensitive to the flight-vibration environment and caused a large amplification of vibrations at certain frequencies. Also, the thermal performance (heat leak) of the SHe tank was affected significantly by large-pressure oscillations in the circuit of the heat exchanger caused by a thermoacoustical phenomenon. The phenomenon resulted in an oscillating pressure wave passing through the internal heat exchanger. To correct this problem, a small interconnecting line was added between the heat-exchanger inlet and outlet lines just outside the tank, thus damping the oscillations. This interconnection presented another problem in that the interconnecting line provided a path that permitted continuous circulation through the internal heat exchanger and an attendant degradation in the thermal performance of the tank.

Developmental tests at White Sands Test Facility (WSTF) on preproduction tanks proved that this circulation phenomenon was a function of the vertical heat-exchanger orientation and the associated head difference between inlet and outlet lines. As a result of circulation, pressure-rise rates as high as 200 psi/hr were detected. In the production tank, the internal heat exchanger was reconfigured to a fully supported (from inner-vessel wall) coil in the lower half of the tank with the inlet- and outlet-tube penetrations going through the bottom of the inner vessel. The interconnecting line was still required to prevent the pressure oscillations in the heat-exchanger circuit.

In the first production-tank configuration, the copper pinch-off tube for the vacuum annulus had an excessive pressure rating that would rupture the outer shell in the event of a gross helium leak into the annulus from the inner vessel. The tube was redesigned and tested to provide a lower relieving pressure (within outer-shell limits), and all production tanks were retrofitted or fitted with the new pinch-off tube.

During the handling and acceptance tests, the outer shell collapsed several times in local areas. The most prevalent location of local collapsing was around the penetration of the heat-exchanger tubes through the outer shell. No definite conclusions were reached concerning the cause of the collapse other than the fact that some tanks were marginal in outer-shell strength. As a result of the collapse, the external pressure-proof test on the outer shell during acceptance testing was reduced from 19.5 psia to 18.5 psia. However, this reduced pressure did not completely eliminate occurrences of collapsing. Collapsed tanks were refurbished by installing new outer shells.

The dual burst-disk assembly on the SHe tank was designed with stainless-steel belleville washer springs (17-7 PH) that controlled the burst-pressure setting of the burst disk. During certain vehicle checks on an SHe tank, the primary disk ruptured at approximately 300 psi below the set pressure of 1881 to 1967 psia. This rupture was traced to stress corrosion of the belleville springs caused by long-term exposure to moisture. One spring of the four-spring stack was cracked completely through; thus, the preload on the burst disk (fig. 8) was lowered. If moisture was detected by a vacuum-dryness check, a pressure-proof test was performed on all production burst disks. Also, desiccant bags were maintained on the outlet tube of all burst disks after the dryness check and the pressure-proof test. Subsequent to LM-5 (Apollo 11), on some vehicles the burst disk was modified to incorporate an Inconel belleville spring, which is resistant to stress corrosion.

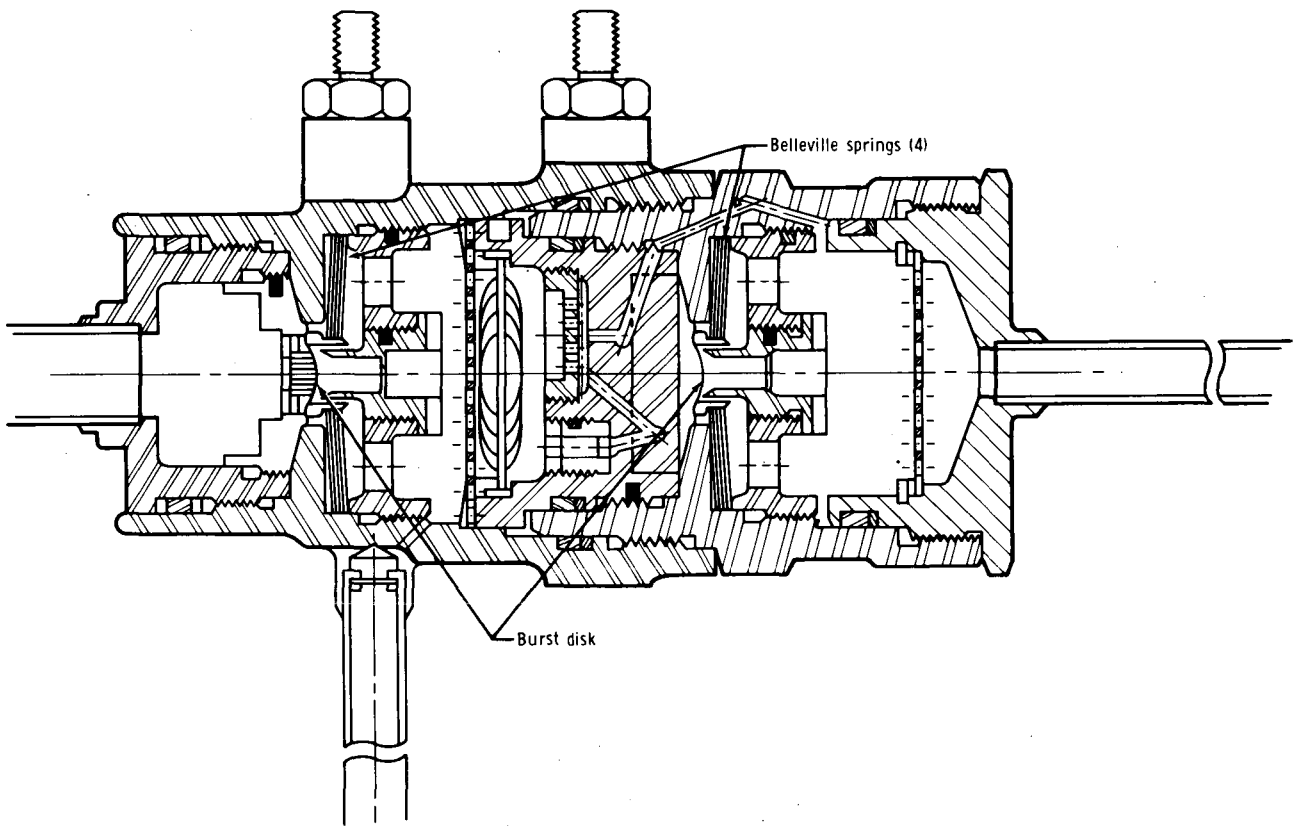


Figure 8. - The SHe tank burst disk.

Supercritical helium GSE. - The principal items that were developed for servicing the SHe system were an SHe conditioning unit and a liquid-helium dewar. Among the design requirements of this equipment were that the automatic checkout equipment (ACE) could remotely control and monitor the equipment and that the equipment could meet the size and weight limitations required for placement close to the LM SHe tank (on the launch-pad mobile-service structure). These two requirements were significant constraints in the development of this equipment.

In the initial design of the liquid-helium dewar, the structural support of the inner vessel was marginal. During trucking of one dewar with a full load of liquid helium, the inner vessel vibrated so badly that the fill and discharge tube at the bottom of the tank was cracked, and it released liquid helium into the annulus, thus imploding the inner vessel to release the expanded helium (the outer vessel relief was not adequate). Subsequently, the dewar was modified to increase the stiffness of the bumper pads and to improve the suspension-cable arrangement in the annulus.

Fuel-to-helium heat exchanger. - The development of a two-pass fuel-to-helium heat exchanger was initiated in June 1965. Primary goals were high thermal performance, high reliability, and good structural integrity.

The initial heat-exchanger configuration incorporated a nickel-chromium braze alloy in bonding the side panels to the core. During the testing of lunar test article number five (LTA-5) at WSTF, one of the side panels separated from the core and ruptured. This rupture resulted in a gross fuel leak and subsequent fire. The cause of this failure was traced to the factory test of the rig; the heat exchanger had been subjected to cryogenic temperatures with water in the fuel passages. Cryogenic temperatures caused freezing of the water that resulted in structural failure of the nickel-chromium braze material. Subsequent exposure to system-operating pressure at WSTF resulted in a rupture of the side panels. Two items were implemented to avoid this problem on subsequent vehicles. The nickel-chromium braze was changed to a gold-alloy braze to increase the bonding strength of the side panels, and water was eliminated from cold-flow testing of vehicles when cryogenic helium was used.

Latching solenoid valve. - During the initial design of the latching solenoid valve, use of nylon and butyl O-rings was considered adequate for a 3.5-day exposure to propellant vapor. By mid-1966, tests had proved that backup rings and lubricants were necessary to prevent severe chewing of the butyl O-rings during cycling. Conducted at a private research institute, parallel compatibility tests proved the nylon seat was inadequate. Late in April 1966, the seat material was changed to the present Teflon material. Other design changes resulting from the tests were the replacement of Teflon backup rings with Teflon cap seals and the incorporation of electromagnetic-interference suppressors.

In November 1967, the latching solenoid valve, installed in a prototype descent stage number two (PD-2) at WSTF, failed to remain in the latched-open position after a rapid open-close command. The problem resulted because of residual magnetism in the latch plunger that exceeded the spring-return force when the valve was cycled

rapidly. The design changes to eliminate this problem were a high-rate 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 subjected successfully to qualification tests that were completed in the spring of 1968.

In March 1968, the latching solenoid valve on the LM-3 vehicle was leaking externally through the valve body-tube nickel-chromium brazed joint. The solenoid valve was redesigned for LM-5 and subsequent vehicles with a brazed joint that had the proper clearances and that used gold-nickel alloy as the braze material. To certify the solenoid valves for LM-3 and LM-4, a special test program was successfully completed with the nickel-braze configuration. Because the valves were not to be flown on lunar-landing missions, a program decision was made to accept the potential occurrence of a leakage failure on LM-3 and LM-4. A delta qualification test program was conducted successfully to certify the gold-braze configuration for flight.

Helium pressure regulator. - The development of a pressure regulator for the pressurization system was initiated in June 1964. The development of this regulator was troubled by many problems, especially internal and external leakage. The regulator failed to meet the slam-start requirements (lockup with high pressure upstream and initially with vacuum downstream coupled with minimum downstream volume). Cracking of the main poppet and high lockup pressure were the two major problems during slam starts. Unavailable hardware was a problem that restrained the developmental program.

In October 1966, an alternate vendor was selected for development of a pressure regulator. In mid-1967, the alternate vendor was selected for continued development, and the first vendor was terminated. The newer regulator exhibited a pressure-spiking characteristic during slam starts although it locked within the maximum limit of 251 psia. The spike was as high as 266 psia in the vicinity of the burst disk and relief valve. No corrective action was made to the regulator; however, this phenomenon was specified as a design requirement for the burst disk.

Two incidents of contamination in the reference sensor port resulted in high lockup pressures. One incident of wide-open failure on the primary stage was caused by contamination in the pilot poppet seat. The failures resulted from inadequate contamination control; thus, special precautions were implemented to prevent their occurrence on flight parts.

Quad check valves. - The quad-check-valve design was used in the ascent propulsion system and DPS. Conducted on two valves, the qualification test program was completed successfully in November 1966. Some of the valves did not meet the initial specification of 10 scc/hr; however, the actual leakage rates were acceptable for the LM missions. The maximum-allowable leakage was increased to 100 scc/hr for each element and to 100 scc/hr for the overall valve. Excessive leakage later occurred on several flight vehicles. In August 1968, failure analysis conducted on internally leaking check valves indicated that most of the internal leakage problems experienced on the LM vehicles were caused by contamination on the seat. Investigation of these failures led to modification of the assembly, installation, and cleaning procedures at the facility of the vehicle contractor and at the facility of the component vendor.

Propellant Storage and Feed System

The development and qualification of the propellant storage and feed system were conducted on a component and a system-level basis. Early evaluation tests were conducted on the feedline and parallel-tank balance-line configuration by use of a heavy-weight test rig. The major objectives of these tests were to determine the effectiveness of the propellant utilization and the propellant management control under system-operating flow conditions. The major concern was evidenced at maximum thrust because differences in system pressures would cause maximum propellant unbalance. At reduced thrust, the engine-cavitating venturis provided the major propellant-utilization control. No major problems were discovered during these tests.

Propellant-storage tanks. - Considerable weld porosity was experienced early in the Apollo Program. This porosity required some rewelding; however, as experience was gained, porosity was gradually reduced by adjusting weld speeds and amperages. Also, some membrane distortion was experienced in the heat-affected zones; however, this problem was not major.

During the proof-pressure test of tank S/N P-009 (March 8, 1965), rupture occurred at a pressure of 267 psi. The required proof pressure was 360 psi. The premature failure was traced to a massive alpha inclusion in the upper dome near the helium-inlet boss. The source of the inclusion was not determined; however, the vendor speculated that it probably occurred during smelting operations when a small piece of oxidized titanium dropped off the side of the vat into the slurry at the time of solidification. The metals company was advised of this fact so that the possibility of recurrence could be minimized.

During the helium-leakage testing of tank S/N G-029 (August 23, 1966), rupture occurred at a pressure of 180 psi. This rupture happened after the tank had been proof tested. Failure was traced to stress corrosion in the cover (access door). Early in the Apollo Program, the covers were made of titanium alloy (6Al-4V). However, a careful analysis indicated that weight could be saved by changing the material to aluminum alloy because of the large amount of low-stress material in the bosses. Previously, these covers had been proof tested with uninhibited tapwater as the fluid. Initiation of the stress corrosion that caused the tank failure probably occurred during the uninhibited water proof test because aluminum alloy (2014-T651) is susceptible to stress corrosion under these conditions. Therefore, the decision was made to return to the use of titanium-alloy covers.

In the feasibility tests, no major problems were experienced with the tank. However, numerous cracks were found in the contractor-furnished support skirt and deck. These cracks were found after the vibration tests were conducted in the X-X axis. Also, some cracks occurred in the plumbing support brackets, and minor cracks occurred in the baffles. In some cases, the deficiencies were corrected by increasing the material thickness; in other cases, they were corrected by elimination of stress risers.

Early in the Apollo Program, the seals between the tank and the cover (access door) were to be of butyl rubber SR 634-70. However, this material tends to deteriorate above 90° F after exposure to nitrogen tetroxide. The seals were replaced with a different design composed of a V-shaped stainless-steel ring coated with Teflon.

The weight-improvement program involved changes to the stub support skirt, to the flanges at the cover attachment, and to the cover itself. The stub skirt was shortened, and the spaces between the rivet holes were scalloped to remove useless material. The elimination of one of the seals, which caused considerable difficulty in subsequent helium-leak checks, allowed material to be removed from the tank and cover flanges.

Stress corrosion from nitrogen tetroxide was a major problem; thus, several solutions were considered, including coating the walls with Teflon, shot peening the wall surfaces, changing the tank material, and changing the propellant nitrogen-tetroxide specification. The nitrogen-oxide content in the nitrogen tetroxide was increased to inhibit the stress corrosion by nitrogen tetroxide. In the Apollo Program, this problem was universal in systems using nitrogen tetroxide.

Propellant quantity gaging system. - The development of a propellant quantity gaging system (PQGS) was initiated in January 1966. The initial requirement for calibration accuracy of the continuous-quantity probes was ± 0.25 percent throughout the full range. Because of the configuration of internal wiring, standoffs, et cetera in the space between the inner and outer electrode, this accuracy requirement could not be met; thus, the accuracy requirements were changed. No associated weight penalty was present in the budgeted propellant; however, the propellant uncertainty in the indicated quantity at the mission redline was greater. Low-level sensor accuracy was not affected. The revised accuracies are presented in the following table.

Range, percent	Accuracy, percent
0 to 8	± 0.5
8 to 25	± 0.25
25 to 95	± 0.5

During vibration qualification tests (Y axis) of the PQGS probe, the outer electrode incurred a crack adjacent to a rivet hole. The vibration qualification requirements were more severe than those required for flight. The vibration levels were redefined to more realistic (representative of flight) values. Subsequently, the PQGS probes passed the vibration qualification test. During tests at WSTF and on the flight vehicles, the fuel-quantity indicators exhibited an initially low value (90 to 95 percent for full tanks) when the PQGS was first started. In approximately 15 minutes, the indications achieved the correct or nearly correct full reading of 95 percent (high off-scale). Although this time-lag anomaly was not fully understood, it was attributed to a conductivity gradient in the fuel caused by trace amounts of contaminants. Heat from the sensor electronics on the bottom of the probe tended to disperse this conductivity gradient.

Excessive leakage occurred at the flange-to-tank door-joint seal on LM-1. This excessive leakage was attributed to inadequate flatness of the flange when the bolts

were torqued, putting uneven compression on the single Teflon O-ring. A 3/8-inch stainless-steel backup ring was added to the flange to improve the flatness when the bolts were torqued. Also, a secondary injectant-sealant groove was cut in the tank-door flange to provide a backup seal. This groove was injected with a viscous sealant (Nitroso for the oxidizer and Vistanex for the fuel) before bolt torquing and primary seal-leak checks.

Descent Engine

A parallel developmental effort was conducted in the initial part of the descent-engine program. The engine throttling concepts and the selection of the final throttling concept are discussed in the following paragraphs.

Helium-injection throttling. - The development of an engine with helium-injection throttling that used a fixed-area injector was initiated in February 1963. The testing of developmental hardware was started in August 1963. For the first 6 months, the developmental test program was paced by the limited availability of injectors. Early in the developmental test program of the injector, weld problems caused interpropellant leaks that resulted in catastrophic injector failures. As a result of the failures, a major redesign of the injector was made to allow easier fabrication and better inspection of the injector.

Combustion-stability problems occurred in early injector developmental tests. The self-induced instability was predominantly first-tangential mode resonance (high frequency). A Y-baffle arrangement was incorporated on the injector face to damp the instabilities. From limited tests, the high-frequency self-induced and bomb-induced instabilities appeared to be damped by the baffles; however, a low-frequency buzz (200 to 500 hertz) and intermittent high-amplitude disturbances (popping) were two other forms of combustion instability that remained. Severe baffle erosion and undercutting of the ablative chamber near the baffles resulted during the buzz instabilities. Mainly, the buzz instability was a problem at reduced throttle, and it was present with and without helium injection.

Other developmental problems with the engine included performance degradation (as the engine was throttled to reduced thrust) and excessive ablative chamber-throat erosion at midrange thrust.

Variable-area-injector throttling. - The development of an engine using a variable-area injector (fig. 9) was initiated in July 1963. Testing of developmental hardware for this engine began in January 1964.

No major combustion-stability problems were encountered with this injector configuration. The lack of stability problems was attributed to the inherently stable design of the center-element injector, which minimized energy release near the chamber walls where most high-frequency instability modes are sustained. However,

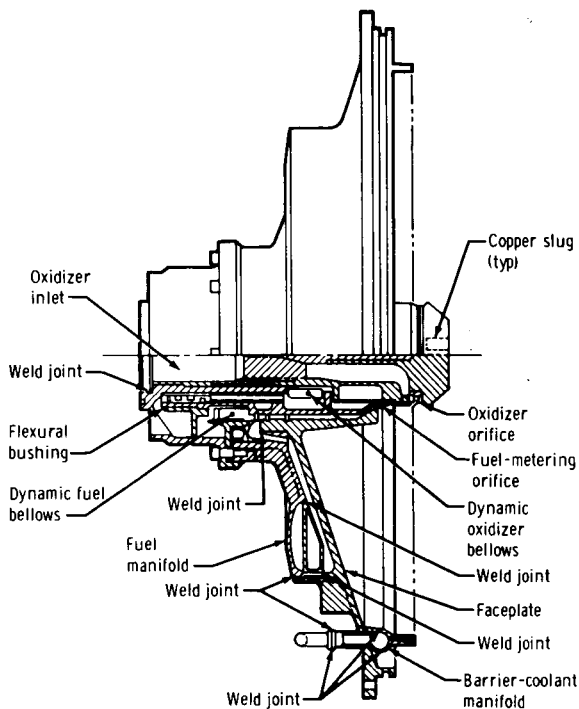


Figure 9. - Cross section of LM descent-engine injector.

combustion roughness was experienced, which was a random-type pressure oscillation with no cyclic resonant frequency. The source of this combustion roughness appeared to be primarily the coarse injector design. The relatively few oxidizer streams (36 primary and 36 secondary) provided a source of uneven combustion. Bomb tests were performed at various positions in the chamber, and no incidence of combustion instability occurred. The bomb tests were performed in a steel-wall combustion chamber to minimize the natural damping effect of the ablative chamber.

Throat erosion of the ablative chamber was significantly higher than that usually associated with fixed-thrust engines. The greatest rate of erosion occurred at full thrust. Some occurrences of excessive heating and bulging of the titanium case of the throat section were observed during heat soakback following a long full-thrust burn (approximately 400 seconds). This effect occurred after engine shutdown and was not indicative of a potential burnthrough.

Engine selection. - In January 1965, a decision was made to continue development of the variable-area-injector engine and to terminate development of the fixed-area-injector helium-injection engine. This decision was based upon an evaluation of the developmental problems and the progress that occurred from initiation of each of the engine programs until the selection time. The variable-area-injector engine was selected because of better combustion stability and better combustion performance throughout the range of thrust. Because of better combustion stability and better performance profile, fewer problems were anticipated in the development of the engine. The increased mechanical complexity of the throttling mechanism gave the variable-area-injector engine a lower reliability than the fixed-area-injector helium-injection engine.

Engine developmental problems. - Because the engine qualification configuration had to be selected before all developmental problems were resolved completely, the engine qualification was conducted in two phases. The phase-A qualification was conducted simultaneously with continued engine development to meet the early delivery requirements for the vehicle engine. The phase-B engine qualification was initiated in January 1967 and was completed in August 1967.

The original injector pintle tip (figs. 9 and 10) was made of stainless steel, and it incurred severe erosion during engine operation because of inadequate cooling. As a result of the erosion, copper slugs were incorporated in the pintle tip to promote better

heat transfer. This configuration was used in the first engine qualification design (phase A). In the phase-B qualification engine, a columbium-alloy pintle tip was used because the copper-slug configuration did not eliminate completely the erosion. No erosion occurred with the columbium tip.

The phase-A injector exhibited very uneven throat-erosion patterns with severe gouging in local areas. This gouging was caused by poor injector-inlet hydraulics on the oxidizer side; therefore, a small step (a turbulator) was added to the oxidizer-inlet tube of the phase-B injector to provide an even velocity distribution in the oxidizer fans. As a result of this addition, uneven throat erosion was reduced greatly and gouging was eliminated.

The ablative thrust chamber of the phase-A engine had a slotted turbulence ring approximately 5 inches downstream from the injector face. The turbulence ring was eliminated on the phase-B engine to allow the flow of fuel-barrier coolant to extend farther down the wall of the chamber and to reduce throat erosion.

During the phase-B qualification test, gouging did occur on one engine. The gouging was traced to one of the fuel-barrier cooling orifices being plugged by room-temperature vulcanized (RTV) rubber, which was used as a seal around the tubes of the cooling orifice at the ablative faceplate. A process change was made to prevent plugging of the fuel-barrier cooling orifices during assembly. Also, during the phase-B qualification test, one engine incurred a failure of the electron-beam (EB) weld on the guide bar that positions the oxidizer sealing ring (fig. 10). This failure also occurred on several WSTF engines, and in some cases, the guide bar came completely out from under the oxidizer sleeve. Subsequently, an EB tack weld in the center of the guide bar was changed to a structural EB weld around the periphery of the guide bar.

During systems tests at WSTF, 500-hertz oscillations in pressure were observed in the oxidizer-injector manifold at approximately 40-percent thrust on several engines. The worst case of these oscillations was as high as 300 psi peak-to-peak. Initially, the oscillations could not be duplicated on engine-test stands at the facility of the vendor. A special investigation was conducted at WSTF and at the facility of the engine vendor to determine the cause of the oscillations and to determine whether they had any detrimental effect on the engine. This investigation included a special instrumented engine test to evaluate the effects of gas saturation, instrumentation installations, and structural oscillation; tests and studies to evaluate the effects of special flow-control-valve cavitation; and tests to evaluate the effects of special instrumentation dynamic response. From these investigations, the oxidizer-manifold oscillations were determined to be the result of small-amplitude (5 psi peak-to-peak) oscillations from the flow-control-valve cavitation process (at approximately 40-percent thrust); these oscillations were being amplified only in the pressure transducer line. The 500-hertz oscillations were observed to a lesser degree at the engine feedline interface. The chamber pressure was insensitive to the 500-hertz oscillations. A more detailed summary of this investigation is found in reference 2.

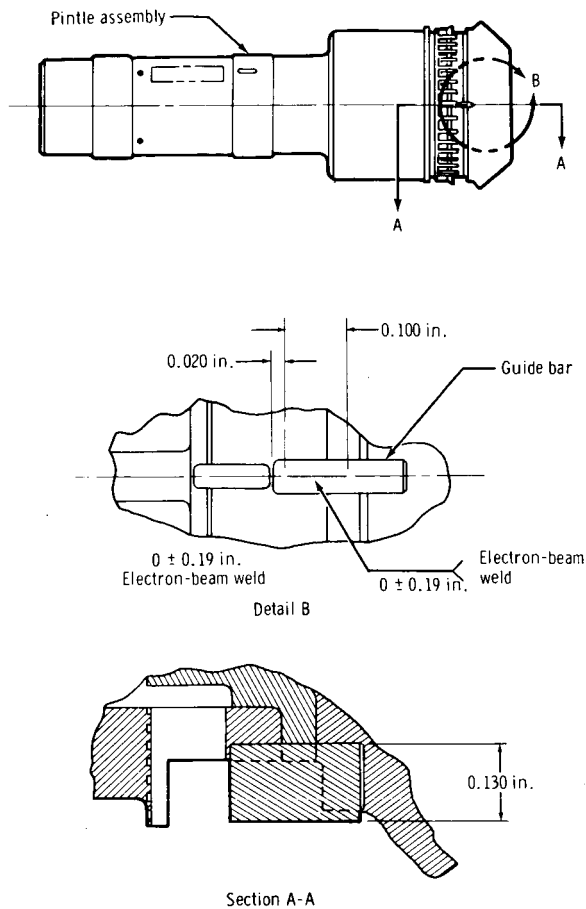


Figure 10. - Pintle-assembly diagram showing guide-bar installation.

Several engines exhibited resonant 100-hertz oscillations in chamber pressure at approximately 25-percent thrust. These oscillations were in the order of 10 psi peak-to-peak. From the 500-hertz test, the oscillations were found to be damped by the presence of helium saturation in the propellant. Previous engine testing had been done without helium-saturated propellants. Helium saturation is representative of the propellant conditions in flight.

During the development and qualification of the engine shutoff valve (SOV), several minor problems occurred. A chronic problem was SOV rotor-seal leakage. Generally, the seals met the 10-scc/hr leakage requirement for gaseous nitrogen, but they were sensitive to contamination and scoring. The major causes of leakage were cycling of the rotors in the dry condition and long-term propellant exposure, which generated residue contaminants. The allowable number of dry cycles before launch was limited to 250. Also, special precautions were taken to reduce the valve-actuation time during dry cycling (approximately 2 to 3 seconds) to prevent seal damage caused by very rapid actuation. Contamination leakage was corrected by replacing leaky seals before engine delivery.

In the testing of developmental and WSTF engines, several instances of rack jamming or sluggish valve actuation occurred. This problem was caused by oxidizer leakage past the shaft seals (fig. 11) into the gear cavities and by subsequent reaction of the oxidizer with the gear lubricant. No cases occurred where the valve completely jammed, but all cases resulted in sluggish valve actuation on the affected rack. The gear cavities on all flight engines were checked for leaks and acidity (pH factor) before delivery. Also, a force-pressure check to detect sluggish valves was performed at the plant of the contractor and at the NASA John F. Kennedy Space Center to measure the pressure for cracking and full-open actuation of each of the four actuators.

The valve position for each actuator was indicated by reed switches. During development and qualification, the switches were unreliable and gave false indications. The cause of these failures was not fully understood, but it was thought to be caused by the engine-vibration environment and delicately constructed reed switches. After the flight of LM-1, on which a false indication was noted, a reevaluation of the need for valve-position indication was made. Because no mission decision required the use of valve position and because of the unreliability of the reed switches, the switches were eliminated with a resultant savings in weight of approximately 1.5 pounds.

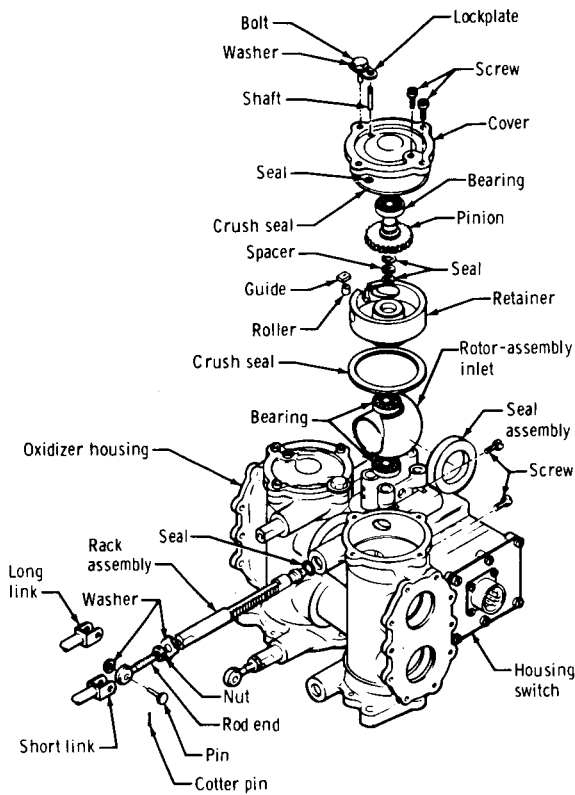
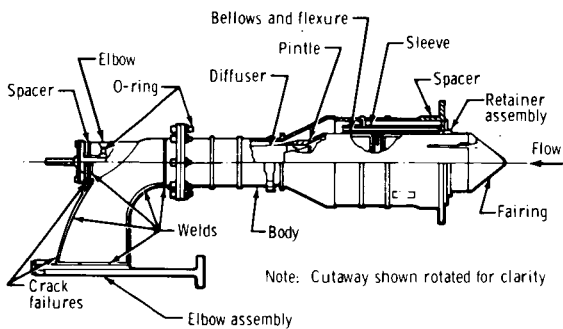
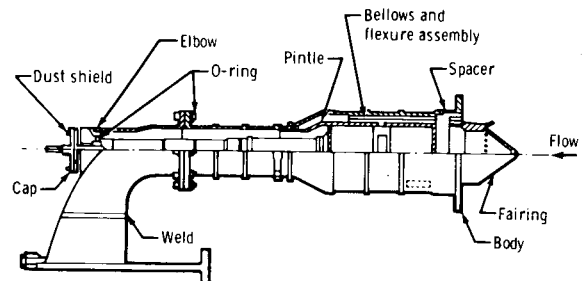


Figure 11. - Exploded view of oxidizer section of engine shutoff valve.

The original elbow of the flow-control valve (FCV) was made of stamped 6061 aluminum sheet metal and machined 6061 aluminum flanges (fig. 12(a)). The elbow was welded together as shown in figure 12(a). During qualification tests, cracks occurred in a weld near the flange; these cracks were caused by the closeness of the weld to the flange, which resulted in a stress concentration. Because of the design, the weld was very difficult to inspect. The elbow was redesigned to relocate the weld farther from the flange to improve the strength and inspectability. The redesigned elbow passed qualification; however, during production of elbows for flight engines, several elbows cracked during the proof-pressure test (540 psig). These cracks occurred in a weld adjacent to the throttle-linkage boss. An X-ray of the weld in this area showed that some elbows had a void in the weld caused by inadequate penetration from the welds made on the inner and outer surfaces of the elbow. Subsequently, all elbows were inspected by X-ray for voids in the welds. Because this inspection screened the inadequate welds, it resulted in a high rate of scrapage. A new two-piece elbow (fig. 12(b)) was designed and qualified for



(a) Sheet-metal elbow.



(b) Two-piece elbow.

Figure 12. - Flow-control valve.

use in later flight engines. The new elbow consisted of two pieces machined from solid stock (6061 aluminum) and joined by a single weld in an easily inspectable area. This elbow configuration was incorporated in one of the phase-B qualification engines.

During production of phase-B engines, an FCV retaining nut was found to have a fatigue crack in the root of a thread. Further investigation revealed that the fatigue crack occurred during water-flow calibration of the FCV. At certain thrust levels (approximately 10 percent), the water flow produced excessive pressure oscillations in the cavitating venturi that resulted in fatigue of the retaining nut. This phenomenon was associated with the water-flow test but was not associated with propellant-flow conditions. Because the retaining nuts on all flight engines were susceptible to the problem, corrective action had to be taken. Provided the upper part of the nut was not free to move, structural failure of the nut would not result in any adverse effect on the engine. Therefore, all engines were modified (in the field for delivered engines) by applying epoxy beneath the retaining nut, thus bonding it in place if it cracked completely through.

During development and qualification, the engine-pressure transducers (chamber pressure and injector pressures) acquired a calibration shift after exposure to propellant and atmospheric environment. This calibration shift was traced to etching of the transducer diaphragm by acid formed from residual propellant and moisture. To prevent this problem, the flight-engine pressure transducers on all delivered engines were installed after acceptance firing and after cleaning the engine for delivery.

TEST SUMMARY

At WSTF a comprehensive ground-test program was conducted to develop and certify the DPS for lunar landing. This test program encompassed approximately 3-1/2 years. Testing started early in 1966 and continued through mid-1969. The test program used three propulsion-test rigs.

Descent-Propulsion Developmental Vehicle

The PD-2 propulsion rig was used for high-altitude firings. The PD-2 employed a flightweight primary structure and a secondary structure designed for small titanium propellant tanks and prototype propellant components. Tests performed on the PD-2 included substitute- and live-propellant cold-flow tests, off-nominal start and operating characteristics, component malfunctions, vehicle mission-duty cycle firings, and 500-hertz oxidizer-manifold oscillations.

Lunar Test Article 5

The LTA-5 was a full-scale LM descent stage with a complete production DPS. On July 3, 1967, following completion of the first system test (LM-1 mission-duty cycle (MDC)) and while still at altitude, the fuel-helium heat exchanger ruptured. During facility-securing operations, fuel that leaked from the heat exchanger was ignited by an unknown source. As a result of the ignition and consequent fire, LTA-5 was damaged extensively. Subsequently, the LTA-5 was refurbished to approximately the LM-3 and LM-4 configuration, and it was redesignated LTA-5D. Two tests, one in support of mission B (LM-1) and one in support of mission D (LM-3), were conducted before the start of mission-G testing.

Five MDC tests were accomplished to support the lunar-landing mission (mission G). The tests were performed to demonstrate the capability of the DPS to perform the mission-G MDC under nominal conditions, off-nominal conditions, and malfunction conditions. All of the tests were made with propellants that were saturated with helium after gaseous nitrogen had been removed. In addition to the MDC tests, one SPS abort-duty cycle was conducted in which two long DPS firings were performed that had a long coast in between.

FLIGHT EXPERIENCE

Apollo 5 (LM-1)

The LM-1, the first flight LM, was launched on January 22, 1968, by a Saturn 1B. The LM-1 was flown in an unmanned earth-orbital mission to evaluate subsystem flight performance and to verify manrating.

System configuration. - The DPS was basically the same as the lunar-landing configuration (LM-5) except for the following differences.

1. The pressurization system did not include an ambient-helium start bottle.
2. The fuel-to-helium heat exchanger was an early configuration that utilized a nickel-chromium brazing alloy for structural bonding.
3. The propellant-tank manual-venting system (lunar-dump system) for lunar-surface venting was not incorporated on this vehicle.
4. The propellant-tank door seals and various other tank and inline flange seals were different from the final LM-5 configuration.
5. The propellant-tank burst disk was an early design that had a higher nominal setting and a larger tolerance band on the burst pressure.

Flight problems. - During the flight of LM-1, two anomalies were associated with the DPS. The first anomaly was the premature shutdown of the first descent-engine burn. The second anomaly occurred in the second and third descent-engine burns in which an engine-propellant SOV out-of-phase indication occurred.

Premature shutdown of first descent-engine burn: The first descent-propulsion maneuver, controlled by the primary guidance, navigation, and control system, was scheduled to last approximately 38 seconds. The descent-engine thrust monitor was programmed to stop the engine if any three consecutive 2-second-accelerometer samples (taken after the commanded "engine on") indicated an accumulated velocity of less than 45 cm/sec. This criterion was based on a nominal engine start with the propellant tanks initially at full-operating pressure and with the helium supply on line.

The LM-1 did not have an ambient-start helium-storage tank, and the SHE tank was isolated by the three explosive valves that were fired automatically by the pyrotechnic system 1.3 ± 0.3 seconds after the first "engine on" command. Therefore,

the system pressures during the first DPS start, which were normal for this particular system configuration, did not rise fast enough to meet the thrust-time criteria programmed into the guidance computer. All logic and circuits that could command any engine cutoff or inhibit any engine start were reevaluated to prevent unnecessary engine shutdown on subsequent flights. The thrust-monitor logic was the only circuit found to produce a premature engine shutdown.

Out-of-phase indication from descent-engine propellant shutoff valves: During the transition from 10-percent throttle to full throttle on the second and third descent-engine firings, an out-of-phase indication was received from one pair of the two-pair position switches that indicate the position of the eight propellant shutoff valves. The out-of-phase indication remained until the end of both firings.

An investigation was conducted to determine the cause of the phasing anomaly. The primary areas investigated as possible causes for the out-of-phase indication were as follows.

1. Reed-switch failure (false indication)
2. Pilot-valve or actuator-hydraulic malfunction or leakage (valve failed full or partially closed)
3. Electrical-circuit failure (valve failed full or partially closed)

This investigation could not establish conclusively a single cause or combination of causes as the basis for the observed phasing anomaly. However, because of the past unreliability of the reed switches, the most likely cause of the anomaly was concluded to be a false indication by the reed switch. A decision was made that the reed switches should be eliminated from LM-5 and subsequent vehicles.

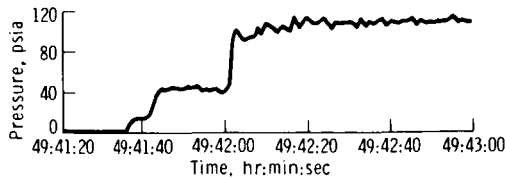
Apollo 9 (LM-3)

The LM-3 was the second LM to fly. The LM-3 was flown in a manned earth-orbital mission launched on March 3, 1969. The LM-3 mission was a 10-day flight to qualify the LM for future flights and to demonstrate certain combined spacecraft functions for manned lunar flight.

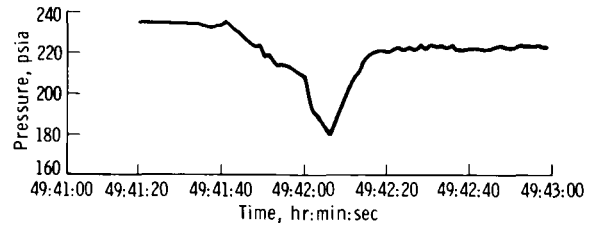
System configuration. - Two major configuration differences between the LM-3 and the LM-1 DPS were evidenced. The LM-3 incorporated the ambient-helium pressurization-start system to allow prepressurization of the propellant tanks before the first burn. The LM-3 also incorporated a lunar dump-valve venting system on each set of the propellant tanks. The LM-3 guidance computer program was modified to prevent the inadvertent ΔV cutoff phenomenon experienced on the LM-1 flight.

Flight problems. - During the flight of LM-3, three anomalies associated with the DPS occurred.

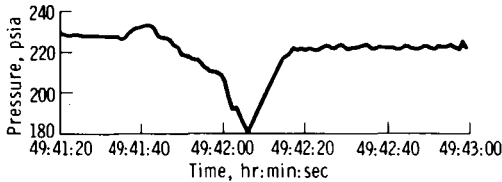
The SHe tank internal-heat-exchanger freezing: During the first 35 seconds of the first descent-engine firing, the regulator outlet-manifold pressure decreased from 235 to 188 psia. The regulator should have regulated at 247 psia (fig. 13). The



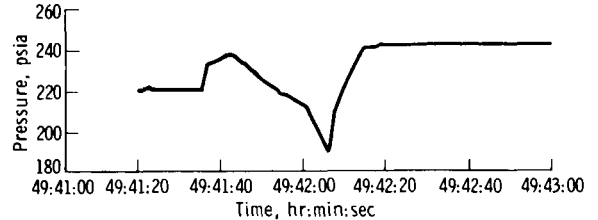
(a) Chamber.



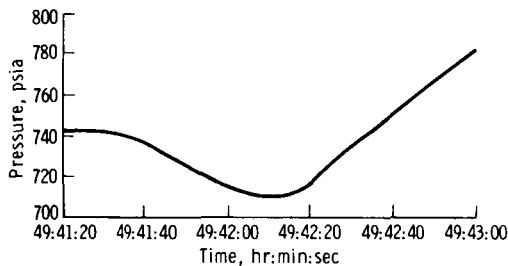
(b) Fuel interface.



(c) Oxidizer interface.



(d) Regulator outlet.



(e) Supercritical helium tank.

Figure 13. - Apollo 9 DPS pressures during start of first firing.

temperature data indicated that the internal heat exchanger initially was blocked. At approximately 35 seconds after engine ignition, the blockage cleared and allowed the regulator outlet-manifold pressure to rise to the proper operating level.

During final servicing of the SHe tank for flight at the launch complex, an event occurred that caused concern. After the SHe tank had been topped-off and the helium heat-exchanger manifold vented to approximately 25 psia, an inadvertent venting of the helium heat-exchanger manifold to approximately atmospheric pressure occurred while the vehicle quick-disconnects were being warmed by a helium purge to

allow their disconnection. During the quick-disconnect purge, the supply pressure probably was allowed to drop below atmospheric pressure, and outside air probably was cryopumped into the helium manifold by the cold heat exchanger in the SHe tank. Subsequently, the air was condensed and frozen in the near-liquid-helium temperature environment of the internal heat exchanger. To isolate the purge system from the helium-manifold pressure-control system, the GSE was modified for subsequent flights. In addition, continuous pressure recording with proper scaling was employed on the helium manifold.

The SHe system leak: The pressure in the SHe tank for the DPS began decaying at the rate of 2.9 psi/hr immediately after termination of the first descent-engine firing. The pressure decay continued until staging. Because of heat transfer into the

tank, the pressure normally increased under no-flow conditions. Calculations showed that a leak of about 0.1 lb/hr would have caused the 2.9-psi/hr pressure-decay rate.

A failure of an internally brazed squib valve was found during drop tests of LM-2 at the NASA Manned Spacecraft Center. The failure was caused by a crack in the brazing material, which was thin in the failed area. The time of the failure could not be ascertained; however, it probably was caused by the shock of the squib firing to pressurize the ascent propulsion system.

The LM-3 SHe squib valve was similar to the LM-2 squib valve in that the valve fittings were internally brazed, which prevented inspection of the joint. The leak experienced during Apollo 9 probably was caused by a defective braze that was internal to the squib valve and could not be inspected.

The mission criticality of external leakage was evaluated for each of the squib valves on the Apollo 10 vehicle. On the DPS, the oxidizer-compatibility squib valve was determined to be of the internal-braze configuration; thus, if external leakage developed, it could have been hazardous to the mission. Therefore, this valve was modified to incorporate RTV potting reinforced with an external clamp (fig. 14) to contain any leakage if the brazes cracked. This configuration was thoroughly tested before the flight. The valves on all vehicles subsequent to Apollo 10 were of the external-braze configuration.

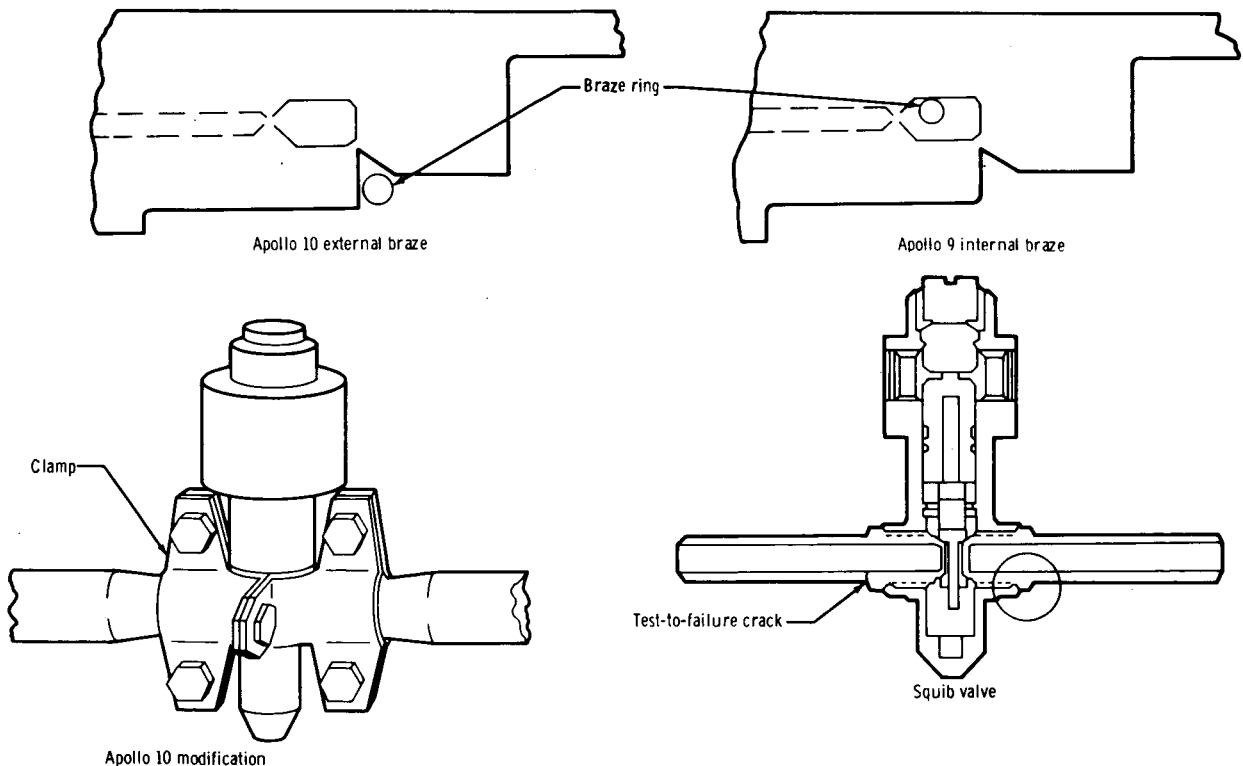
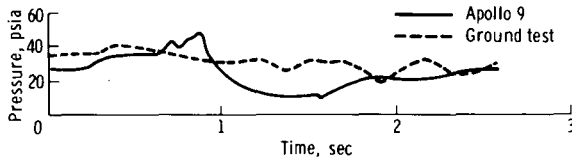
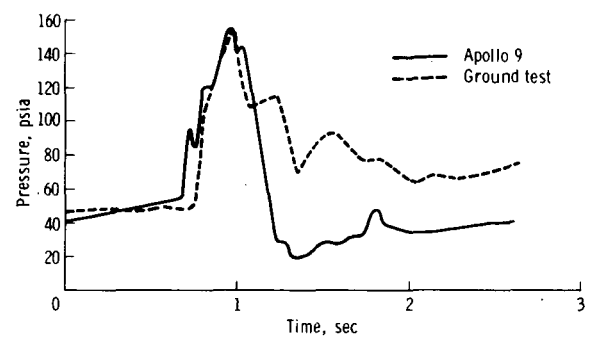


Figure 14. - The SHe squib-valve configuration.

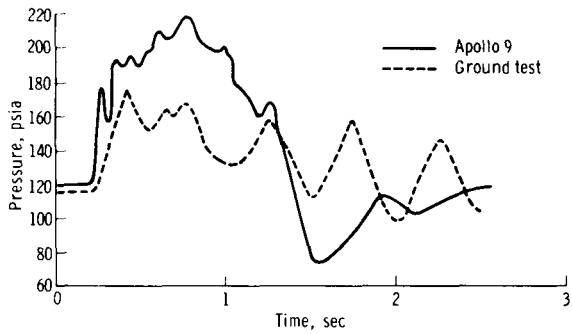
Engine roughness: During the second descent-engine firing, rough combustion was encountered at approximately 27-percent thrust as the engine was being manually throttled from 12 to 40 percent. This roughness lasted for approximately 2.5 seconds, during which time the crew delayed increasing the throttle. The resultant transients in propellant-injector pressures and chamber pressure are shown in figure 15.



(a) Thrust chamber.



(b) Fuel injector.



(c) Oxidizer injector.

Figure 15.- Comparison of roughness during second descent-engine firing and ground test.

Ground-test data of engine pressures as helium was deliberately ingested into the engine are superimposed on figure 15. The similarity of data indicates that helium ingestion was the probable cause of the roughness. A possibility existed for helium bubbles to migrate into the propellant feedlines from the propellant tanks, which had 60-percent ullage volume after the first burn. This migration would result from lateral or rotational vehicle acceleration (or both) combined with the fact

that the zero-g-can retention screen was not capable of holding propellants at the 114-inch head between identical propellant tanks (fig. 16).

Helium ingestion requires that the zero-g can be uncovered and that lateral or rotational accelerations be present. This condition was highly probable with the 60-percent ullage volume present in Apollo 9 propellant tanks before the second descent-engine firing. Tests have demonstrated that ingestion of helium into the engine in this manner has no detrimental effect on the system.

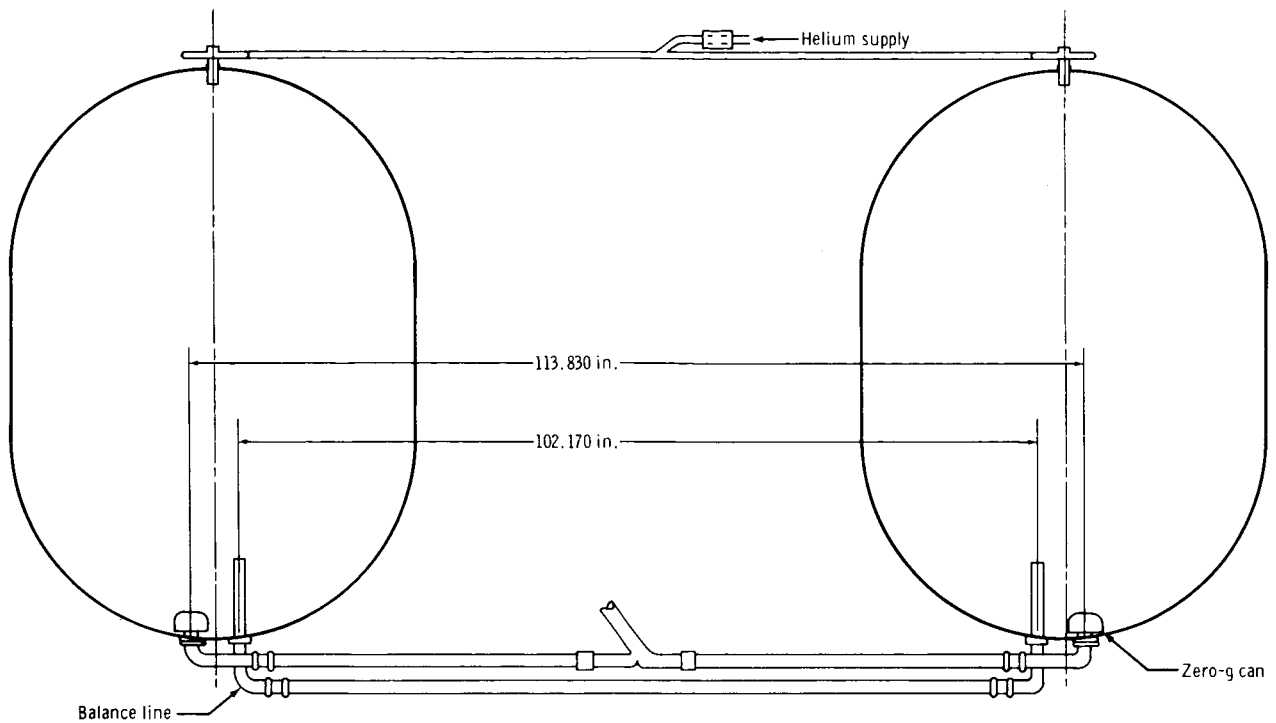


Figure 16. - Tanks and plumbing of the DPS.

Apollo 10 (LM-4)

The LM-4, the third LM to fly, was the second manned LM flight. Launched on May 18, 1969, the LM-4 mission was an 8-day translunar and lunar-orbital mission designed to confirm all aspects of the lunar-landing mission exactly as it would be performed, except for the actual descent, landing, lunar stay, and ascent from the lunar surface.

During the phasing burn, the crew reported two occurrences of a master alarm caused by a propellant-quantity low-level indication. The first alarm was coincident with engine "on" command, and the second alarm was noted immediately after throttling to FTP. The crew reset the master alarm after the first low-level alarm, and the low-level indication was reported to have ceased working at that time.

The low-level indication probably was caused by a gas bubble in the unsettled propellant uncovering the low-level sensor before the burn but after turning on the PQGS. Once the low-level sensor was uncovered, the indication was latched on by a latching relay and remained on until the PQGS was turned off again after completion of the burn. The crew display of low quantity was inhibited until the engine "on" command was given, thus explaining the master alarm at the time of engine ignition. This

fact was verified by the telemetry recording of the low-level indication during the phasing burn (fig. 17). The second alarm (immediately after throttle up) could not be verified by telemetry data nor explained by normal system operation. On subsequent missions, the low-level indication was removed from the master alarm because a visual indication by the low-level warning light was considered adequate warning for the crew. Also, if the low-level indication occurred at engine ignition, a contingency crew procedure was instituted to reset the low-level indication by cycling the PQGS off and on.

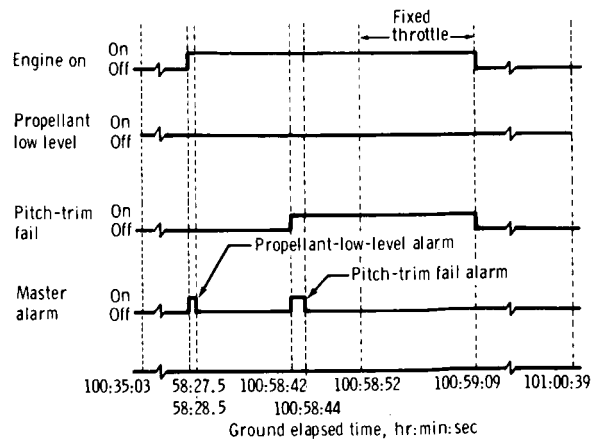


Figure 17. - Master alarms during Apollo 10 phasing maneuver.

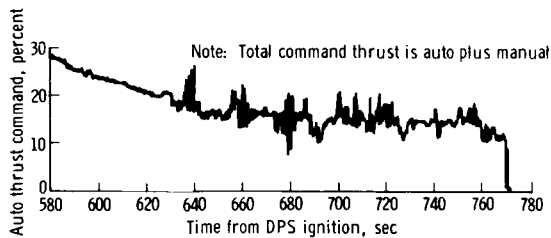
Apollo 11 (LM-5)

The LM-5, the first lunar-landing vehicle, was the fourth production LM to fly. The primary objective of the Apollo 11 mission was to perform a manned lunar landing and to return safely to earth. At landing, the quantity of propellants remaining in each of the four tanks was as follows.

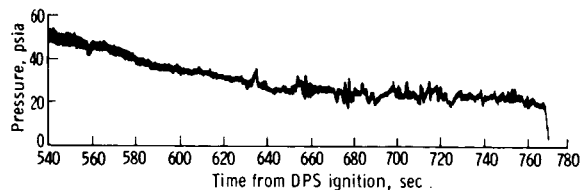
1. Oxidizer tank 1: 5.7 percent
2. Oxidizer tank 2: 2.7 percent
3. Fuel tank 1: 3.9 percent
4. Fuel tank 2: 3.9 percent

The minimum-usable quantities, based on the indication of oxidizer tank 2, allowed a remaining hover time of 63.5 seconds. Several minor anomalies occurred in flight. During the power-descent-initiation (PDI) burn, the oxidizer-engine-inlet pressures indicated low-frequency oscillations that varied from 24 psi peak-to-peak during the FTP operation to 67 psi peak-to-peak during throttle recovery (approximately 57-percent thrust). The oscillations observed were characteristic of those observed in oxidizer-injector manifold and engine-inlet pressures in several ground-test engines, which were verified to be harmless to engine or system operation.

During the last 140 seconds of the PDI burn (hover phase), a series of large oscillating-throttle changes occurred (fig. 18). These changes were approximately 15 percent peak-to-peak about a nominal throttle setting of approximately 26 percent. Large and rapid changes in vehicle attitude (as much as 15° at $10^\circ/\text{sec}$) in pitch during the final phases of landing caused centrifugal accelerations on the inertial-measurement-unit accelerometers located at the top of the ascent stage high above the vehicle c.g., thereby giving a false indication of vertical acceleration. This false indication caused the lunar guidance computer to command a throttle change to compensate for an unreal change in vertical acceleration. Later, this problem was investigated under the analysis of the guidance and control system.



(a) Measured automatic-command voltage.



(b) Measured chamber pressure.

Figure 18. - Apollo 11 hover-phase throttle profile.

During the PDI burn, the oxidizer quantity in tank 1 and tank 2 diverged, as observed by the PQGS readings, until, at 360 seconds into the burn oxidizer, tank 2 was approximately 3.4 percent less than oxidizer tank 1. This divergence in remaining oxidizer quantity was attributed to propellant transfer from one oxidizer tank to the other through the propellant balanceline during the burn. The offset in the vehicle c.g. from the centerline between the oxidizer tanks (on Z axis) resulted in the tilting of the vehicle in that axis to put the thrust vector through the vehicle c.g. (fig. 19). This tilt caused preferential feeding from the tank that was highest with respect to vehicle c.g. Although the observed magnitude was less than the calculated magnitude, calculation of propellant transfer between the two tanks, based on the known geometric head difference, confirmed the transfer of propellant from oxidizer tank 2 to oxidizer tank 1. Adjustments were made to the propellant margin to account for this difference.

At 685 seconds into the PDI burn, the propellant-low-quantity warning light was triggered by the sensor in one of the four propellant tanks. Based on remaining calculated quantities and corrected PQGS indications, the occurrence of the propellant-low-quantity warning was discovered to be premature by 36 seconds. The early warning was the result of propellant slosh waves created by sudden vehicle maneuvers and by attitude changes. These slosh waves caused the propellant level in the PQGS probe to oscillate about a negatively biased level. At the lowest point of the oscillation, the low-level sensor was momentarily uncovered, thus latching a relay to the warning light.

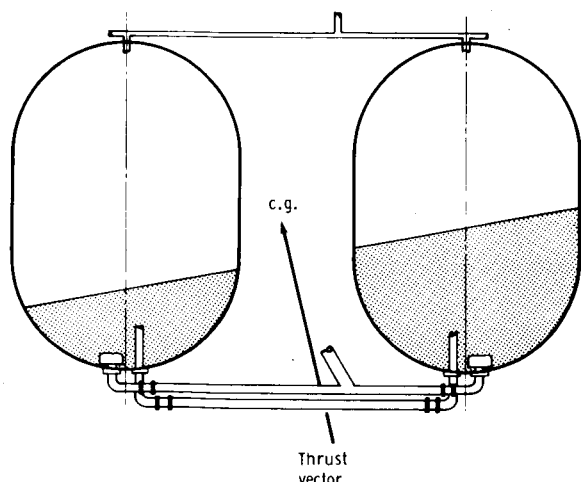


Figure 19. - Schematic of propellant unbalance.

Before the flights of Apollo 11 and 12 (during which the anomaly was observed) a ground-test program was conducted in which the critical slosh modes were identified and potential modifications to eliminate or reduce the error were evaluated. From this test program, the lateral-mode and ring-mode slosh were found to produce the most errors in PQGS readings. Slosh baffles were incorporated on Apollo 14 and subsequent vehicles to minimize the slosh in the tanks.

Venting of the propellant and the SHe tanks after lunar landing resulted in the freezing of the fuel-to-helium heat exchanger. Fuel and oxidizer interface pressures as compared to time during the postlanding venting are presented in figure 20. Venting of the SHe tank through the lunar-dump valves resulted in the flow of residual cold helium into the SHe tank through the heat exchanger, thereby freezing the fuel in the fuel passages and blocking flow of fuel through the heat exchanger. During the subsequent hour, the engine experienced a peak in heat soakback from the ablative chamber, which caused the fuel trapped in the line between the heat exchanger and engine SOV to expand thermally; therefore, the pressure rose rapidly. After 30 minutes, the fuel pressure was relieved by thawing of the heat exchanger, failure of the line-bellows linkage, or failure of the seals in the prevalue. The exact cause of relief was not determined. On subsequent flights, the venting procedure was modified to isolate the SHe tank by the latching solenoid valves during venting of the propellant tanks and to delay SHe venting until immediately before launch from the lunar surface. Based on autoignition tests of the fuel at various pressure levels, the final-vent fuel-pressure level was revised from 20 to 40 psia to 8 psia to prevent any autoignition hazards in the aerazine-50 fuel during engine peak periods of heat soakback. On LM-7 and subsequent vehicles, a bypass line around the heat exchanger was incorporated as an added safety feature to relieve the trapped fuel pressure if freezing of the heat exchanger should occur.

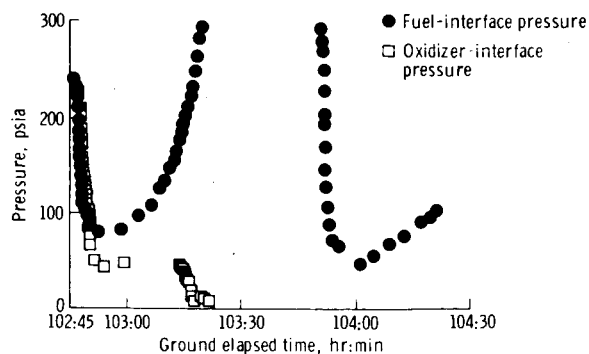


Figure 20. - The Apollo 11 descent-propellant and helium venting.

CONCLUDING REMARKS

Because of the complexity caused by the throttling and gimbaling requirements and the use of an advanced pressurization-system concept, the DPS was one of the last Apollo propulsion systems to reach developmental maturity. Because the DPS was also a pressure-fed earth-storable hypergolic-propellant design, as were the other Apollo spacecraft propulsion systems, DPS development did benefit by some of the knowledge gained in the earlier development of other systems.

Testing of the DPS with substitute propellants, in the development and flight-vehicle-acceptance phases of the program, successfully achieved the proper pressure-drop balance in the subsystem, thus minimizing propellant residuals required for a mission. Pressure-drop balance was critical to the DPS because of the use of parallel feed tanks in the design and the criticality of propellant depletion to mission success.

Early in the Apollo Program, a decision was made to incorporate a SHe pressurization system into the DPS design for a vehicle-weight savings. The design of the pressurization system was tailored to meet the time-profile requirements of the lunar-landing mission. The capability of the pressurization system to perform other mission

time profiles, such as earth-orbital multiburn missions and many lunar-abort missions, was limited severely. Also, this lack of capability constrained launch-delay turnaround time. Even though a DPS weight savings was realized, a significant penalty occurred in launch and mission flexibility.

During development and qualification of the DPS, several unanticipated hardware and system problems occurred. Long-term propellant compatibility resulted in seal degradation and corrosion of materials supposedly compatible with the propellants. Contamination, which resulted in component leakage or malfunction, was another significant problem. For future systems of this type, steps should be taken to isolate critical components from excessive contamination by filters or other means. This isolation should include periods of component replacement in the system and normal system exposure. Also, the component design should be made less sensitive to contamination, if possible; a certain amount of contamination in the system is unavoidable.

Many cases of excessive seal-leakage problems occurred, particularly in the propellant storage and feed system. Many of these problems resulted from minimizing the weight of the seal configuration, not providing adequate sealing-surface stiffness, and other seal-design features.

In the design of the PQGS, two significant problem areas were encountered that should be avoided. In the design of the fuel probes, the accuracy of the quantity measurement was affected greatly by fuel-conductivity gradients within the fuel tanks. Trace amounts of contamination in local areas could change the local conductivity. This was a design weakness in the fuel probe. The other problem area with the PQGS was propellant slosh near low-level quantities that caused cycling and biased readings at low level. A more thorough evaluation of vehicle motions near the lunar-landing point in the mission may have prevented encountering this problem in flight.

Based on the experience with the DPS, system-to-system interface requirements are very critical, and a periodic review of system interface requirements should be conducted. For example, the guidance and navigation system on the LM-1 flight had a ΔV monitor that terminated the engine burn if a certain ΔV threshold was not met. This burn termination might have been avoided if the LM-1 engine-start characteristics had been compared with the guidance and navigation program ΔV monitor.

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