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APOLLO EXPERIENCE REPORT - COMMAND AND SERVICE MODULE REACTION CONTROL SYSTEMS

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16. Abstract

The reaction control systems of the Apollo command and service module were developed and modified between July 1961 and July 1969. The successful development of these systems, as part of the Apollo Program, was the result of extensive testing, retesting, and modifications of the hardware to ensure system capability and intrasystem compatibility.

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APOLLO EXPERIENCE REPORT

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SUMMARY

The development of the command module reaction control system and the service module reaction control system resulted in hardware that was capable of meeting lunar mission requirements. The initial system concepts were identified and engineered into preliminary designs, refined, adapted to mathematical models, and subjected to engineering evaluation. Although off-the-shelf components were not available, state-of-the-art technology was applied whenever possible in building prototype items. Also, extensive component development, certification, and qualification tests were conducted. In particular, component development testing resulted in numerous design iterations. Additionally, both vibration and thermal vacuum tests were conducted on components that were assembled to represent flight systems. Successive unmanned flight tests were evaluated, necessary changes were made, and the command module and service module reaction control systems were certified for manned flights. The Apollo 7 to 10 missions resulted in or involved system refinements.

INTRODUCTION

The first statement of work released by the NASA Space Task Group specified that the Apollo spacecraft would be composed of a command module (CM) and a service module (SM). Each module was to be attitude stabilized by a bipropellant reaction control system (RCS). A significant advancement in the state of the art would be required in the areas of RCS cycle life and operational life. Development problems in these as well as other areas were encountered in the component and system development programs. This report describes the evolution of the Apollo CM and SM RCS from the description in the first statement of work to the systems required for the first lunar landing. Particular attention is given to problems encountered, and recommendations are made concerning future space programs.

INITIAL CONCEPTS

In July 1961, the first statement of work for the Apollo CM and SM was issued by the NASA Space Task Group. This statement of work specified that the RCS for the CM and SM would include the following capabilities.

1. The RCS must provide the capability for attitude control, stabilization, propellant settling (ullage maneuver) for a vernier propulsion system, and minor velocity corrections.

2. The RCS for the CM and SM would be pulse modulated and pressure fed, and would use storable hypergolic propellants that were identical with the propellant which was used in the vernier propulsion system.

3. The propellant tanks would be of the positive expulsion type.

By November 1961, the statement of work was expanded to include the following requirements.

1. Both the CM RCS and the SM RCS were each to consist of two independent systems; each independent system was to be capable of meeting the total torque and propellant requirements.

2. The fuel was to be monomethylhydrazine (MMH), and the oxidizer was to be a mixture of nitrogen tetroxide (N_2O_4) and nitrous oxide (N_2O). This propellant choice was based on consideration of the lower allowable storage temperatures that resulted from the lower freezing temperatures (i.e., lower than other propellants that were being considered).

3. The requirement for a vernier propulsion system was deleted.

4. The RCS was to provide the ullage maneuver (propellant settling) thrust for the SM propellant.

On November 28, 1961, NASA selected a prime contractor to design and build the Apollo spacecraft. On December 21, 1961, NASA authorized a letter contract for work to begin on the Apollo spacecraft development program.

The functional requirements for the RCS were defined early in the program. The SM RCS was to provide a three-axis rotational capability for use in orientation and maneuvering, transient damping, and limit cycle attitude holding. A three-axis translational capability was to be provided for use in separation, rendezvous and docking, midcourse correction, and ullage maneuvers. Later, an additional requirement was imposed: the capability to deorbit if a failure occurred in the service propulsion system (SPS). This deorbit capability requirement was applicable only for earth orbital missions. Initially, the CM RCS was intended to provide only a three-axis rotational capability. At approximately the same time that the requirement for SM reaction control deorbit capability was imposed, a CM RCS technique for translation was developed. This CM translation conferred a hybrid deorbit capability that involved the use of both CM RCS and SM RCS for total velocity increment (ΔV) requirements.

Although the functional requirements were well defined early in the program, the environmental conditions to which the equipment would be exposed were not well defined. The environments that lacked adequate early definition included boost vibration levels, thermal exposure ranges, fluid exposure requirements (propellant-flush fluid and moisture compatibility requirements), shock levels, and allowable particulate contamination levels. It was not until January 1965 that a formal, overall set of environmental qualification levels was established. Even these qualification levels were subject to continual changes, particularly with respect to boost vibration levels, thermal exposure ranges, and shock levels. Some qualification levels were not defined until the flight data were analyzed. In other areas where there was control over the environment (for example, fluid exposure and allowable particulate contamination), it became desirable, either for scheduling or economic reasons, to change the environmental requirements. These changes, which occurred late in the program, resulted in frequent configuration changes or in hardware retesting to ensure compliance with revised environmental requirements. In retrospect, meeting an identified, real, environmental requirement was never a major problem. The most frequent problem was the definition of the environments to which hardware would be exposed, so that realistic tests could be configured for use in the demonstration of equipment adequacy.

Initial system configuration requirements were few; thus, the contractor had considerable flexibility in the establishment of system design. However, the November 1961 work statement contained the specification that the CM RCS and the SM RCS would both use two independent systems, each of which had to meet the total torque and propellant specifications. The early specification was implemented in the CM RCS design. Early in the program, the prime contractor proposed that four (rather than two) independent systems supply propellant to the four SM engine clusters. It was suggested that one system supply propellant to each engine cluster. At the outset of the program, NASA technical specialists opposed the concept of four independent systems. However, it is now generally agreed that the four-system concept has been beneficial to the Apollo Program. The major benefit was derived from the capability to modularize completely each of the four SM systems. Other requirements which strongly influenced the choice of configuration were that (1) positive propellant expulsion was to be used, and (2) the propellant pair on both modules was to be MMH (fuel) and nitrogen tetroxide (oxidizer). The SM RCS fuel was changed to Aerozine-50 (A-50) and back to MMH twice during the course of the development program. All manned flights used only MMH as the fuel. Another significant configuration constraint was imposed on the CM RCS in mid-1962. This constraint was that the Gemini 100-pound-thrust engine was to be used on the Apollo CM.

Preliminary Design

The basic design of the CM RCS and SM RCS was not changed appreciably from the original concepts. The SM RCS consisted of four independent assemblies, each of which was composed of a four-engine cluster (a quad), a propellant distribution system, propellant storage tanks, and a helium pressurization system. The SM RCS panel assembly is shown in figure 1, as it existed on spacecraft 104 (Block II), and is shown schematically in figure 2.

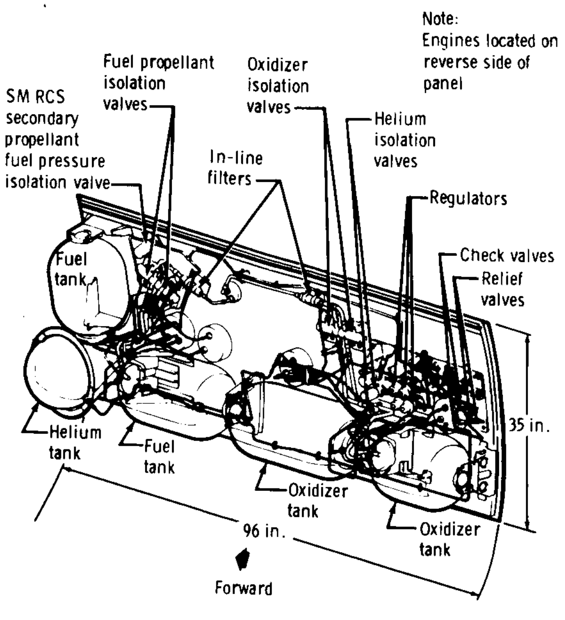


Figure 1. - The SM RCS panel assembly (Block II).

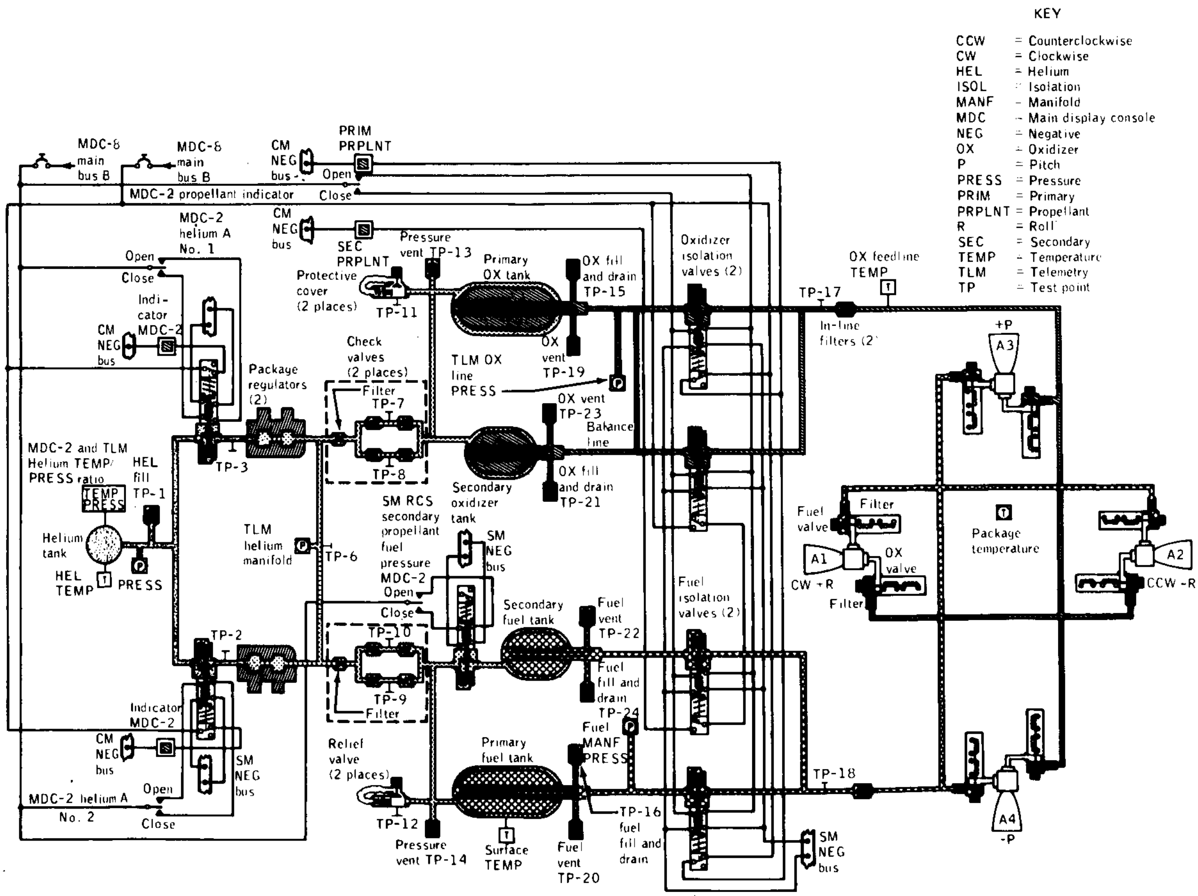


Figure 2. - Schematic of the SM RCS (Block II).

Functional Description of the SM RCS

In each SM RCS assembly, the helium pressurant was stored in a spherical titanium-alloy tank. Initially, the tank was loaded to a pressure of approximately 4100 psia. The helium flowed from the storage tank to the helium isolation valves, which were two-way solenoid valves that were latched open magnetically and that were spring-loaded closed. Each isolation valve contained a position indicator switch that was used to notify the crewmen regarding the valve position. The high-pressure helium was regulated to the desired working pressure (181 psia) by means of parallel regulator units. Each unit contained two nearly identical pressure regulators that operated in series. Either of the parallel regulator units was capable of providing helium for system operation, thereby providing a redundant flow path in case of a failed-closed regulator. Series regulators were used in each unit to preclude overpressurization because of a failed-open regulator. The regulator helium outlet lines were joined at a common point, from which the flow was divided again; thereby, pressurant was supplied to the fuel and oxidizer tanks.

The fuel and the oxidizer were pressurized by means of a common regulator system that maintained the desired oxidizer to fuel (O/F) ratio. Series-parallel check valves were placed between the common point in the line and the fuel and oxidizer tanks. These valves were used to preclude backflow and subsequent mixing of any significant quantities of vaporized hypergolic propellants that might permeate the tank bladders. The series-parallel arrangement was used to guard against the "open" or "closed" failure of any single check valve poppet.

Pressure relief valves were placed between the check valves and the propellant tanks to prevent tank overpressure damage that might be caused by thermal expansion. The series-regulator arrangement precluded overpressures caused by regulator leakage. In retrospect, it would have been wise to have sized the relief valve to handle the flow from both failed "open" series regulators rather than to handle only thermal expansion overpressures; the only penalty involved would have been a slight size increase. The relief valve had two sections. A burst diaphragm was used to provide a positive seal under normal operating conditions. If an overpressure condition occurred, the burst diaphragm would have been ruptured and the pressure applied to the relief poppet. The rupture pressure of the burst diaphragm was between 220 and 236 psid. Control of rupture pressure was maintained by the adjustment of a belleville washer preload on each unit. Relief poppet cracking pressure was between 225 and 248 psid, and minimum reseat pressure was 220 psid. A vent valve was installed between the burst diaphragm and the relief poppet to preclude rendering the burst diaphragm ineffective because of a downstream pressure buildup if a small burst diaphragm leak occurred. The vent valve closed automatically when flow through the valve was sufficient to produce a pressure differential of 30 psid.

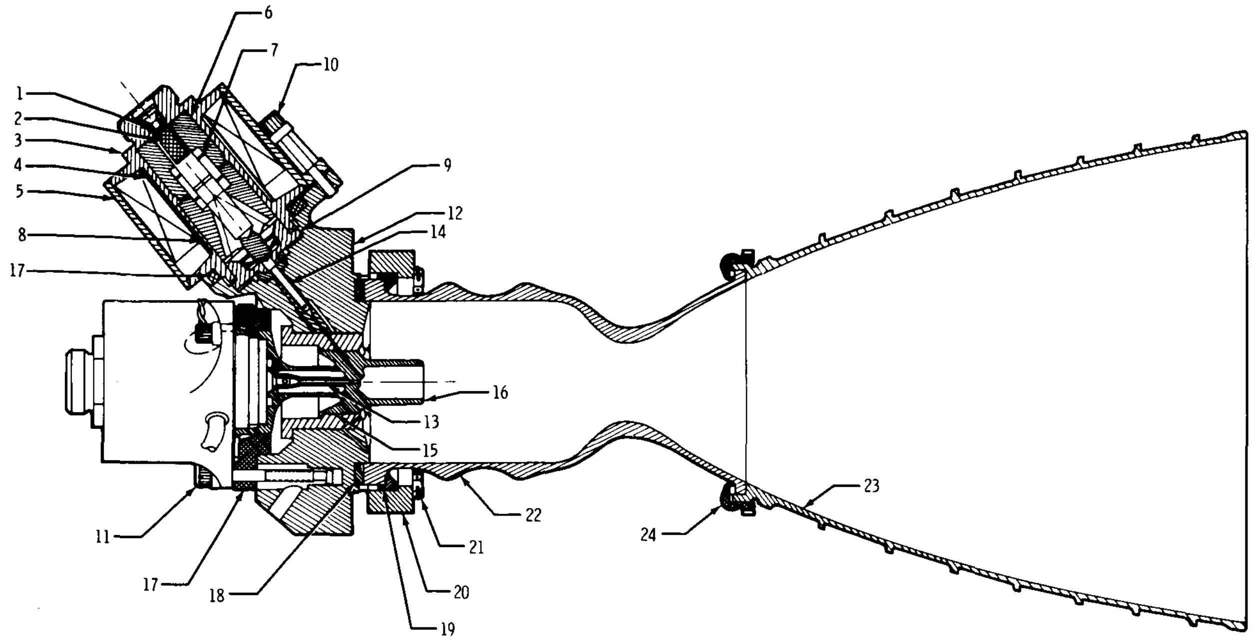
The fuel and oxidizer systems were identical. Each tank consisted of an outer titanium-alloy shell, a Teflon bladder, and an aluminum standpipe. Helium pressurant entered the tank between the metal shell and the Teflon bladder, collapsed the bladder, and forced the propellant out of the bladder through the aluminum standpipe. This arrangement resulted in the capability for positive propellant expulsion in the zero-gravity environment of space. A vent tube was installed inside the standpipe to facilitate servicing of any vertical tank. The vent tube provided a path through which gas trapped inside the bladder could escape when it was being displaced by propellant. Horizontally mounted tanks had to be vacuum serviced.

Initially, each SM RCS quad had two propellant tanks: one fuel tank and one oxidizer tank. Both tanks were mounted vertically. Two additional tanks were installed when propellant requirements were defined more accurately. Spacecraft 101 (Apollo 7, the first Block II configuration) was the first spacecraft equipped with the two additional tanks, which were of the CM type and were smaller than the two existing SM-type tanks.

The additional oxidizer tank was mounted vertically. The added fuel tank, however, had to be mounted horizontally because of space limitations. The horizontally mounted tank had to be vacuum filled. The fill and drain ports that were necessary for servicing the propellants were located in the lines just downstream from the tanks. A propellant isolation valve, which was normally open, was farther downstream from each tank. Initially, propellants were loaded only to the point of the closed isolation valves. Before launch, the isolation valves were opened. These valves were two-way solenoid valves that were latched open magnetically and spring-loaded closed. As was the case for the helium isolation valves, each propellant isolation valve contained a position indicator switch that indicated the valve position to the crewmen. Before spacecraft 104 was built, the only means of propellant management was by opening or closing of the propellant isolation valves. On spacecraft 104, a secondary-fuel-tank helium isolation valve, identical to the existing helium isolation valve, was added just upstream from each smaller fuel tank. Normally, the new valve was closed, and it was intended to be opened only when the fuel in the primary tank was depleted. Fuel depletion was indicated by a drop in the fuel line pressure. This method constituted a one-point check on propellant gaging at the critical switchover point.

In-line filters, added late in the development program, were located in each propellant manifold downstream from the isolation valves and upstream from each engine cluster. The filters were designed to prevent fuel and oxidizer contaminants from entering the engine valves or injectors.

Each of the four engines on each SM quad was a pulse-modulated, radiation-cooled, 100-pound thruster. Major engine components included two solenoid-operated propellant injection valves, an injector, a combustion chamber, and a nozzle skirt. A cross section of the SM RCS engine assembly is shown in figure 3. Each engine valve had two coils, each of which was capable of actuating the armature poppet to the "open" position. The poppet was spring-loaded closed. A coarse filter screen (strainer) and a trim orifice were located in the inlet port of each valve. The filter screen was added late in the engine development program. Each valve was attached to the injector by means of an insulating standoff to prevent overheating of the valve. Valve response was such that the time between the electrical "on" signal to the full "open" position was 9 milliseconds for both valves. The injector that was used on each engine had an 8 doublet main flow pattern; fuel cooling flow of the combustion chamber wall and a preigniter chamber, which was a part of the injector, was provided. The preigniter was incorporated to reduce the magnitude of an ignition overpressure problem; however, the most recent tests indicate that the engines operated satisfactorily without preigniter chambers. The combustion chambers that were used on the engines were made of machined molybdenum forgings which were coated with molybdenum disilicide to prevent molybdenum oxidation at high temperatures. The one-piece molybdenum forgings extended to a nozzle expansion area ratio of 7:1. The nozzle skirt, which was made of a spun cobalt-base alloy, extended to an area ratio of 40:1.



No.	Equipment	Description	No.	Equipment	Description
1	Trim orifice	CRES 304	13	Oxidizer preigniter tube	CRES A286
2	Strainer	CRES 316, 321, 347	14	Fuel preigniter tube	CRES A286
3	Valve body	CRES 446, MIL-E-21562, type MIL-EN6A weld rod	15	Preigniter insert	CRES A286
			16	Preigniter chamber	
4	Automatic coil		17	Thermal insulator	Plastic laminate
5	Direct coil		18	Seal	L605 cobalt-base alloy
6	Plug	CRES 446	19	Split ring	Rene 41
7	Spring	Inconel X	20	Attach ring	Rene 41
8	Armature	CRES 446 with Stellite tip	21	Attach bolts	Rene 41
9	Valve seat assembly	AM 355 with Teflon TFE seal	22	Combustion chamber	Unalloyed molybdenum, disilicide coated
10	Fuel valve attach bolts	6AL-4V titanium			
11	Oxidizer valve attach bolts	CRES A286	23	Nozzle skirt	L605 cobalt-base alloy
12	Injector housing	6061T6 aluminum	24	Nozzle attach nut	WASPALOY

Figure 3. - Cross section of the SM RCS engine assembly (spacecraft 011 and subsequent vehicles).

To ensure that the engines operated within a safe temperature range, the structure to which the engines were attached was heated. To supply heat, two 72-watt, thermostatically controlled strip heater assemblies were bonded to the structure. Although two assemblies were installed, only one was active; the other was available but was held in reserve.

In addition to the items just described, each system had many access ports for use in checkout and servicing. Each access port was provided with redundant closures to prevent any overboard leakage and with inlet filters to protect against externally introduced contamination.

Operational instrumentation was limited to several of the major parameters. The following measurements were displayed on board the SM RCS or were telemetered to the ground.

1. Helium source pressure

2. Helium source temperature

3. Propellant quantity

4. Engine cluster temperature

5. Fuel manifold pressure

6. Oxidizer manifold pressure

7. Helium regulated pressure

The propellant quantity measurement was a temperature-compensated, helium source pressure measurement that was calibrated to read out the number of pounds of remaining propellant. This measurement (known as the onboard pressure/temperature (P/T) sensor) was displayed in the spacecraft and also was telemetered to the ground. A second propellant quantity measurement was computed at ground stations by the use of a helium mass balance equation (pressure volume temperature (PVT) computer program) to calculate the quantity of remaining propellant. This procedure was performed on the ground and was known as the PVT ground computer program.

The P/T sensor propellant values, which supplied propellant quantity data as a function of helium tank pressure and temperature, were displayed in the vehicle in terms of percent full scale of a 0- to 5-volt meter; also, the data were telemetered. The output of the P/T sensor was calibrated to read 100 percent when the helium tank pressure was 4150 psia at 70° F, and to read 0 percent when the pressure was 2250 psia at 70° F. The correct theoretical value of helium tank pressure at propellant depletion was 2450 psia at 65° F. Later, a nomogram was used to correct the P/T sensor readings for this endpoint error, compressibility effects, system temperature variability, and propellant vapor pressure effects.

Pressure, volume, and temperature considerations at an average mission O/F ratio of 1.88 were used in the PVT ground computer program. The quoted accuracy of the PVT ground computer values was only ±6 percent because of instrumentation inaccuracies in the computer input. However, the PVT ground computer program values were assumed to be more nearly correct than were the onboard P/T sensor values. The input involved volumes and propellant loads, O/F shift, and the differential between helium tank and propellant tank ullage temperatures.

In addition to these measurements, the isolation valve positions were displayed to the crewmen.

Functional Description of the CM RCS

The CM RCS was similar to the SM RCS. The CM had two, rather than four, independent assemblies. Each assembly had the total capability that was required to provide entry control. The propellant from both CM assemblies was required only if a hybrid deorbit were needed.

The CM RCS had pyrotechnic helium isolation valves rather than solenoid valves. Normally, the pyrotechnic valves were closed, and were opened just before entry. Once the valves were opened, no provision was made for isolating the helium supply.

Each CM RCS propellant tank had a helium bypass line for use in depleting excess helium before landing. Normally, the flow was shut off by means of a pyrotechnic valve in each bypass line. The line was opened only after the propellant depletion burn was completed.

To provide positive sealing of the system before use, burst-disk-type isolation valves were installed in the propellant feedlines between the tanks and the solenoid-type propellant isolation valves. The burst disks were designed to rupture at a pressure of 241 ± 14 psid.

A major difference between the CM RCS and SM RCS was the type of engines that were used. The SM engines were radiation-cooled, unlimited life engines (from a burn time standpoint). The CM engines were ablatively cooled, limited life engines and were used in a buried application. Although the same major components were used in both the CM and SM engines, some component configurations were significantly different. The CM engine valves were similar to the SM engine valves. Tighter clearances and the lack of armature flutes made the CM engines somewhat more difficult to decontaminate. The injectors on the CM engines were of the 16 doublet splash plate type. The major difference between the CM and SM engines was the combustion chambers. Because the CM engines were buried in the CM skin, ablative chambers were used. The inner section of each engine was a 6° wrap ablative sleeve with a JTA (graphite refractory) throat insert. The sleeve and the throat in the CM engines were 45° wrapped ablative material, which, in turn, was covered by a fiber glass overwrap and an outer stainless steel can. Each engine extended to an expansion area ratio of 9:1. An ablative nozzle extension, which was contoured to the exterior surface of the spacecraft, provided for only limited additional expansion. The 45° wrap and the 9:1 expansion that were used on the Apollo thrusters were significant changes from the 90° wrap and the 40:1 expansion which were used initially on the Gemini 100-pound thrusters. Although these changes represented a departure from the intent of using the same design on the Gemini spacecraft and on the Apollo CM, the improvement in product reliability and entry heating problems warranted the changes.

During the development program, both the Gemini and Apollo engines experienced lamination separation and subsequent outer shell burnthrough. A supporting research contractor (managed by the Manned Spacecraft Center (MSC)) determined that 6° wrap ablative material did not undergo the same burnthrough. Based on this information, the Apollo engine billet was changed to a 45° wrap, and the redesign also included a 6° inner sleeve within the 45° billet. These changes in the Apollo engines resolved the burnthrough problems. It should be noted that, subsequently, the ablative wrap on the Gemini engines was changed to 6°. The Gemini 100-pound engines were not used on the entry vehicle, and heating that was caused by the large exit area was not a problem for Gemini. The change from the Gemini 40:1 expansion ratio to the Apollo 9:1 ratio sacrificed some performance but resulted in a significant reduction in entry heating.

In addition to the ablative burnthrough problems during development, the Gemini and Apollo engines underwent unacceptable chamber streaking. Fortunately, the cause of this problem was found during injector flow investigations that were conducted at the MSC Thermochemical Test Area (TTA) on an Apollo test article. The initial corrective action involved the screening of injectors and the rejection of those that had poor flow patterns. Ultimately, the misalined injector flow passages were alined properly during manufacturing, and streaking was eliminated, which obviated the need for screening. (At the point in the program that these two problems occurred, the program expenditure rates were high. Thus, the supporting development and in-house test evaluation contributions to the Apollo Program cannot be overemphasized.)

To provide redundant flow paths for helium and propellant depletion before landing, interconnect lines and pyrotechnic valves (closed normally) were used between the two CM assemblies. Additionally, an abort propellant and helium dump capability was provided by means of fuel and oxidizer dump valves and helium bypass valves in each CM assembly. The valves were pyrotechnically operated and normally closed. The overboard dump provisions were to be used only in the event of an abort from the launch pad during the first 42 seconds of a mission. As did the SM RCS, the CM RCS had the necessary access ports for the performance of checkout and servicing procedures.

Operational instrumentation was more limited on the CM than on the SM. The parameters measured on the CM included helium source pressure, helium source temperature, and helium regulated pressure.

As on the SM, the propellant isolation valve positions were displayed to the crewmen. Although no formal propellant quantity gaging system was used, propellant quantity could be calculated by the use of a helium mass balance equation.

DEVELOPMENT PROGRAM

From an economic standpoint, it was desirable to use off-the-shelf, readily available components wherever possible. However, because of the unique requirements of most systems, including the command and service module (CSM) RCS, use of these components was not feasible. Although none of the components were off-the-shelf items, most of them were state of the art. For the state-of-the-art components, the development program was rather straightforward and usually consisted of (1) a sufficient number of tests of preprototype hardware to define the design, (2) a design verification test of prototype hardware to verify design adequacy, and (3) qualification tests to demonstrate formally the adequacy of production hardware.

It should be noted that the qualification program was intended to demonstrate formally not only the hardware design adequacy but also the adequancy and consistency of the manufacturing procedure.

In addition to the component tests, a considerable number of system-level tests were conducted. Several of the system-level tests constituted a part of the formal

Certification Test Network (the test demonstration program through which Apollo spacecraft hardware was formally verified as acceptable for manned space flight). The system-level evaluations included system performance demonstration tests, vibration tolerance demonstration tests, and thermal vacuum tests (to verify thermal control system adequacy).

Several CM RCS and SM RCS performance tests were conducted to support the Block I (early spacecraft configuration) vehicles as well as the Block II (spacecraft 101 and subsequent) vehicles. The tests were divided into three major phases for both the CM and the SM systems. These tests began in March 1963 and were completed in March 1968.

Command Module RCS Tests

Phase I. - The Phase I Block I breadboard tests were conducted on the CM from March to September 1963. The objectives of the tests were to evaluate the following circumstances by the use of cold flow techniques.

1. The effect of steady-state and pulse-mode single-engine and multiengine operation on system dynamics

2. Propellant tank bladder efficiency in a system operating environment

Test hardware consisted almost entirely of off-the-shelf components. Results indicated that no major system dynamics problems should be anticipated. Although some component problems occurred, these problems were not considered significant because the hardware was not representative of any flight configuration.

Phase II. - The Phase II Block I breadboard tests were conducted from September 1963 to July 1964. The test objectives were to determine the following by the use of cold flow and live propellant tests.

1. The effect of engine duty cycle on system dynamics and the converse

2. Pad abort dump performance characteristics

3. System operations during all mission activities

The objectives of these tests were met. An overpressure that occurred during system activation resulted in a relief valve burst disk rupture. The corrective action consisted of the installation of 0.070-inch-diameter orifices upstream from the regulators. The effectiveness of this installation was scheduled for evaluation on subsequent breadboard testing. Additionally, system decontamination was performed and results were evaluated.

Phase III. - The Phase III Block I breadboard tests were conducted from September 1965 to November 1966. These tests were the formal performance demonstrations that resulted in certification of the systems for flight. Because of differences in the mission duty cycle or flight hardware (or both), a lengthy series of tests was conducted.

The first test was a spacecraft 009 mission duty cycle. The objective was to certify the spacecraft 009 system by mission simulation. Two significant system shortcomings required corrective action. One shortcoming involved the inadvertent closure of the propellant isolation valves at the time of system activation. The other shortcoming involved a recurrence of relief valve burst disk rupture because of regulator overshoot. A short-term solution of the isolation valve problem was the application of continuous power to the valve during activation. The long-range solution was to redesign the valve flow path to make it insensitive to flow surges. The solution of the regulator overshoot problem was to limit flow surges by reducing the diameter of the orifice from 0.070 to 0.055 inch.

The next tests were in support of spacecraft 012, and the tests included a pad abort simulation and a mission duty cycle demonstration. The objectives of the pad abort simulation were to demonstrate system activation and subsequent oxidizer and helium dumps. The times for oxidizer and helium dumps exceeded the anticipated values. Subsequently, bladder folds were found to cause flow obstruction, which increased the time of the oxidizer dump. Although the dump time was longer than anticipated, the time was still acceptable; thus, no hardware change was required. The excessive helium depletion time was caused by an instrumentation fitting that obstructed the flow path. This instrumentation was not on flight hardware; therefore, a recurrence of this problem was not anticipated.

The objectives of the spacecraft 012 mission duty cycle demonstration included the following specific objectives.

1. To verify the adequacy of the regulator orifice change from a diameter of 0.070 to 0.055 inch

2. To verify the adequacy of the redesigned propellant valves

3. To verify the adequacy of the pressure volume technique for checking propellant load

All objectives were met, except the verification of the adequacy of the regulator orifice change. The tests indicated that the regulator overshoot was still sufficient to rupture the relief valve burst disk. After extensive evaluation of the problem, it was decided not to change the Block I configuration, but rather to wait for an in-line change in the Block II configuration because burst disk rupture did not degrade system performance, and system reliability was compromised only slightly.

One other significant Block I configuration CM RCS test series was conducted from April 1964 to September 1965. The primary objective of the series was to evaluate the operational servicing techniques and checkout procedures at the John F. Kennedy Space Center (KSC). The series was considered valuable because procedures and techniques were developed on other than flight hardware without the pressures of flight schedules.

The final CM RCS ground test was the Block II configuration certification test series that was conducted from January to March 1967. The specific objectives of the series were to demonstrate the Block II mission duty cycle and the pad abort capabilities.

The results of the test were satisfactory. Rerouting the lines to the relief valve eliminated the relief valve burst disk rupture upon system activation. One problem that was revealed during a helium purge was a pressure recovery (to 200 psia) in the manifold 30 seconds after purge. The test proved that the manifold repressurization was caused by a slow release of trapped gas from the propellant tanks. The gas was trapped in the tank by an abrasion pad (over the helium inlet) that acted as a check valve. The problem was resolved on spacecraft 103 by cutting holes in the abrasion pad.

Service Module RCS Tests

Phase I. - The Phase I Block I breadboard tests were conducted on the SM RCS from March to June 1963, and the tests paralleled closely the CM RCS tests. The objective of the tests was to explore system dynamics. No significant problems were noted.

Phase II. - The Phase II Block I tests were conducted from April 1964 to July 1965. The objectives of the tests were as follows.

1. To develop checkout procedures

2. To determine system fill and drain techniques

3. To verify system performance characteristics

Three significant problems were noted during the tests. First, a nuclear gaging device that was under development for this system was very sensitive to temperature variations; additional Phase II Block I component development was required so that the nuclear gaging system could be used. Second, the standoff tubes in the upward-firing engines were damaged during firing. The standoff tubes were strengthened, but with limited success, and it was apparent that sea-level firing of engines with the nozzle pointed upward was unsatisfactory. Residual propellant collected in the standoff tubes, and overpressure damage occurred during subsequent firing. Third, the Block I components in the helium pressurization assembly were degraded seriously because of incompatibility of the assembly with the propellant vapors that moved upstream. The only realistic solution to this problem was to construct the Phase II Block I components from materials that would be compatible with the propellants. When this was accomplished, the system propellant exposure durations were increased satisfactorily. The wisdom of this decision became increasingly evident, because the required system propellant exposure times continually increased throughout the program.

Phase III. - The Phase III Block I breadboard tests were started in November 1965 and were completed in December 1966. As with the CM Phase III tests, the tests were formal and were conducted to certify the system for flight. Both spacecraft 009 and 012 duty cycles were run. In addition to duty cycle simulation, the spacecraft 012 test had propellant gaging system evaluation as a primary objective. The results of earlier tests had indicated that the nuclear gaging system still had major development problems which had to be resolved, and furthermore, a PVT gaging system could be used with greater accuracy. The PVT gaging systems that were used involved ground-computed quantities as the primary gaging technique and an onboard nomogram as a backup gaging method.

The Block II SM RCS was certified in two series of tests. The first series of tests was conducted in support of spacecraft 101 and 103 from December 1966 to March 1967. The objectives of the tests were as follows.

1. To demonstrate a lunar landing mission duty cycle capability

2. To certify the PVT ground computer program and the onboard P/T sensor gaging techniques for flight

3. To certify the servicing procedures for the Block II configuration with the use of additional propellant tanks

The test results indicated that the mission duty cycle objective was met easily. The servicing procedures with the use of the additional tanks also presented no major problem. The onboard P/T sensor technique proved to be considerably more inaccurate than was anticipated. To reduce inaccuracy to a maximum of 10 percent, a nomogram would be needed to correct for temperature variations. The PVT ground computer program was proved to be more accurate than the onboard P/T sensor technique.

One additional problem involved burst disk rupture at the time of system activation. This problem was similar to the difficulty that was encountered on the CM. The proposed solution, similar to that for the CM, was to reroute the helium lines. The change was to be incorporated on spacecraft 102, 104, 106, and subsequent vehicles and was to be certified on the test article for those vehicles. Tests of this hardware were begun in January 1967 and were completed in March 1968. In addition to the requirement for certification of the rerouting of the helium lines, the objectives of this test were as follows.

1. To certify the system operating procedures and the system with the isolation valve upstream from the secondary fuel tank

2. To verify the propellant manifold decontamination procedures

Generally, the results of the tests were satisfactory. However, some anomalies occurred.

Significant engine chamber pressure oscillations occurred during the primary oxidizer tank depletion. The cause of the oscillation was determined to be gas bubbles that passed through the engine. The manner by which the bubbles entered the bladder was never established completely. Agreement was established that, generally, the process involved diffusion of helium through the bladder. A hypothesis (evidenced by all testing) was that diffusion of helium through the bladder was accelerated by a differential pressure which was established across the bladder by the liquid column trapped in the standpipe. The liquid was trapped in the standpipe when the bladder collapsed around the standpipe as the propellant was depleted. Because a differential pressure caused by a liquid head could exist only in a gravity environment, the consensus was that this problem would not exist in space. The consensus was not verified because primary oxidizer tank depletion did not occur in flight.

The inadvertent opening of the secondary propellant isolation valves caused a loss of oxidizer tank ullage as a result of gravity fill from the primary tank. The loss of ullage was critical only if the secondary tank was to be isolated again. Because the separation of the primary and secondary oxidizer tanks served no real purpose, the two tanks were connected by means of a small line so that ullage in either tank would accommodate propellant expansion.

In addition to the CSM RCS performance testing by the contractor, several system tests were conducted at the MSC TTA. A CM RCS test was conducted during May and June 1967 to determine the operational system characteristics at altitude conditions. (The contractor did not conduct this type of altitude testing.) Test firing included a backup deorbit duty cycle that involved the use of the positive- and negative-pitch engines. No significant anomalies were encountered during the tests.

The SM RCS tests were conducted at the MSC TTA during August and September 1965 and during November and December 1966. The objectives of the tests were as follows.

1. To determine system operational characteristics at altitude conditions

2. To demonstrate RCS compatibility with the stabilization and control system (SCS) in a closed-loop dynamic mission simulation

No system problems were encountered during any of these tests. An upward-firing engine failed, but this failure was attributed to low-altitude and low-temperature conditions that retarded vaporization of the residual propellant in the combustion chamber.

System-Level Vibration Tests

System-level vibration tests were conducted on both Block I and Block II SM RCS panels between November 1965 and April 1967. However, no CM RCS vibration tests were conducted above the component level. The logic was that the CM RCS had the same components as the SM, and the CM requirements were significantly less severe than those for the SM, which was subjected to system-level testing.

Block I. - In November 1965, a Block I production-type SM RCS panel was subjected to an acoustic noise level similar to that encountered during spacecraft launch and boosted flight. The complete SM RCS test panel was mounted to simulate spacecraft installation on a 180° SM segment in a reverberant chamber; after it was mounted, the panel was subjected to acoustic noise. The primary test objective was to verify that the system would maintain integrity and function satisfactorily after exposure to severe dynamic conditions. The secondary objective was to measure the dynamic response of RCS components exposed to an acoustic noise environment. After this test, the engine cluster was mounted on another honeycomb panel and was tested separately at a higher level in the plane-wave section of the MSC Vibration and Acoustic Test Facility. The results of the Block I tests indicated that the structural integrity and function of the RCS were not impaired by acoustic testing.

Block II. - In April 1967, a Block II SM RCS panel was subjected to vibroacoustic tests similar to the tests that were performed on the Block I configuration; the test results were similar. However, no tests were performed on the Block II engine cluster alone because of the similarity to the Block I engine cluster.

System Thermal Vacuum Tests

The CSM RCS thermal vacuum tests above the component level were conducted in three phases: spacecraft 008 thermal vacuum tests, test vehicle 2TV-1 thermal vacuum tests, and Block II SM RCS panel tests.

Spacecraft 008 thermal vacuum tests. - Testing of the Block I RCS thermal control system was conducted as part of the spacecraft 008 integrated vehicle thermal vacuum test program in mid-1966. Thermal control of both the SM RCS and CM RCS was evaluated. The two major objectives of the SM RCS evaluation were as follows.

1. To verify the adequacy of the thermal control techniques to maintain the system components within acceptable levels

2. To verify the adequacy of the analytical thermal model

The test results indicated that with functioning heaters, the component temperatures were maintained within acceptable limits. During periods of passive thermal control (heaters off), the temperatures were within the predicted range. However, the rates of temperature change were higher than were anticipated.

One significant anomaly was the failure of the bonding material to hold the heater to the engine cluster on the SM RCS quad D. The heater surface was not rough enough for the bonding material to adhere properly. The solution of the problem for those heaters that were already installed was to clamp the heater in place mechanically; clamping was a secondary holding mechanism for use in case of bond failure. For heaters not yet installed, an improved heater surface preparation technique was used to improve adhesion before bonding.

The major objective of the CM RCS evaluation was to determine the adequacy of the engine heating technique. Test results proved that before activation, the engine valve coil could be used as a heater to cause the injector temperature to rise above a safe minimum limit without overheating the valve.

Test vehicle 2TV-1 thermal vacuum tests. - The test vehicle 2TV-1 thermal vacuum tests were conducted from June to October 1968. All Block II system-level or higher thermal vacuum tests were to be conducted on spacecraft 98 (test vehicle 2TV-1). Because configuration changes were made to the SM RCS late in the program, test vehicle 2TV-1 did not have a representative Block II SM RCS. The test vehicle 2TV-1 SM RCS tanks were of a Block I configuration, and data generated during the test could not be used to verify the entire SM RCS mathematical model. A separate panel test was run in a different facility for this purpose. However, the test vehicle 2TV-1 engine cluster was in the Block II configuration, and the data indicated that the redesigned engine heater system (both primary and secondary) maintained engine components above the minimum redline temperatures. Also, the test established values for (1) the temperature differential between the engine and the nozzle during cold soak and rolling phases, (2) the temperature response of the engine components to a realistic simulation of a rolling mode, and (3) the SM RCS temperature during a full solar hot soak.

The test vehicle 2TV-1 CM RCS had the proper Block II configuration. The test objectives were as follows.

1. To determine the thermal response of the CM RCS to extreme space environment conditions

2. To determine the engine heater warmup response under variable initial conditions

Both objectives were met, and no problems were encountered.

Block II SM RCS panel tests. - Separate Block II SM RCS panel tests were conducted to measure the integrity of the thermal-mathematical model. A production SM RCS panel assembly was used for the tests that were conducted at MSC during February 1968. The mathematical model was necessary because it would be required in real-time flight support to evaluate conditions not included in the tests. The test data indicated the need for several modifications to the model. Final model updating resulted in temperature predictions for hot-case conditions of the propellant feed system to within 7° F of actual values and for cold-case conditions to within 4° F of actual values. All objectives of this test were achieved.

Component Certification and Qualification of the CSM RCS

A certification and qualification test program was conducted for each component in the CSM RCS. These tests included a demonstration of the capability to withstand exposure to temperature, vacuum, vibration, shock, propellants, and acceleration conditions, and demonstrations of operational capability such as functional cycling, proof pressure tests, leakage tests, and pressure drop tests. Tests were also conducted (1) to demonstrate tolerance to particulate contamination and (2) to determine the quantity of contaminants generated. Additionally, selected components were tested under conditions that were more severe than those which were expected during flight, including vibration to 1.5 times the normal qualification levels and pressurization to the component burst point. The test results that were the basis for hardware modifications or operational limitations are given in the following paragraphs.

Test point disconnect coupling. - Only one significant problem occurred with the test point disconnect coupling during qualification. One unit required the use of an engagement torque in excess of that allowed by specification because of a failed thrust washer. The unit was modified by the inclusion of a Rulon bearing to carry the axial loads. The location of the bearing is shown in figure 4.

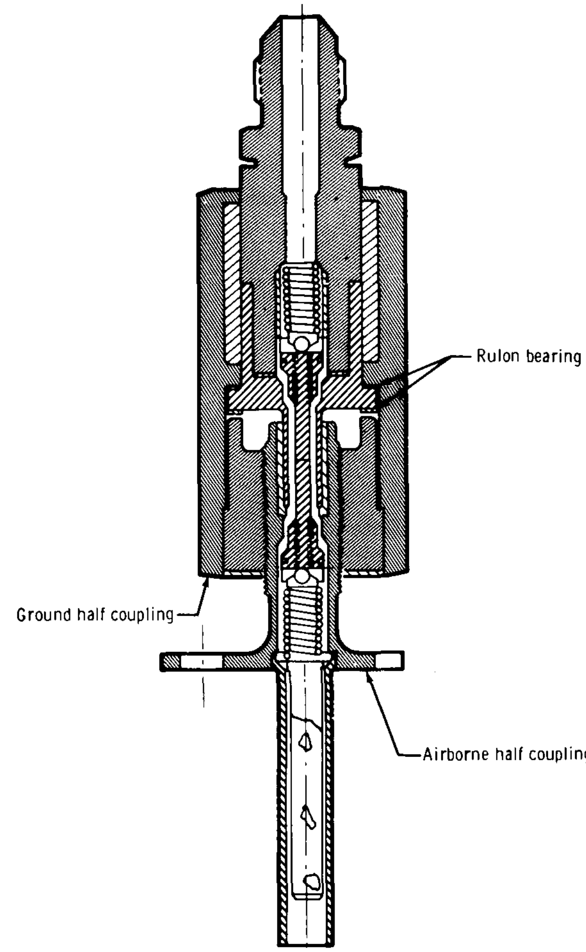


Figure 4. - The CSM RCS test point disconnect coupling.

Helium fill disconnect coupling. - During testing, excessive leakage in the helium fill disconnect coupling occurred at -150° F. Corrective action included a change in the probe seal material from fluorinated ethylene propylene (FEP) Teflon to Kel-F-81, and a design and material change in the probe seal spacers. The probe seal spacer material was changed from aluminum to AISI 303 stainless steel. Also, the configuration was changed so that correct alinement of the ground half probe would be ensured as it entered the airborne half coupling (fig. 5).

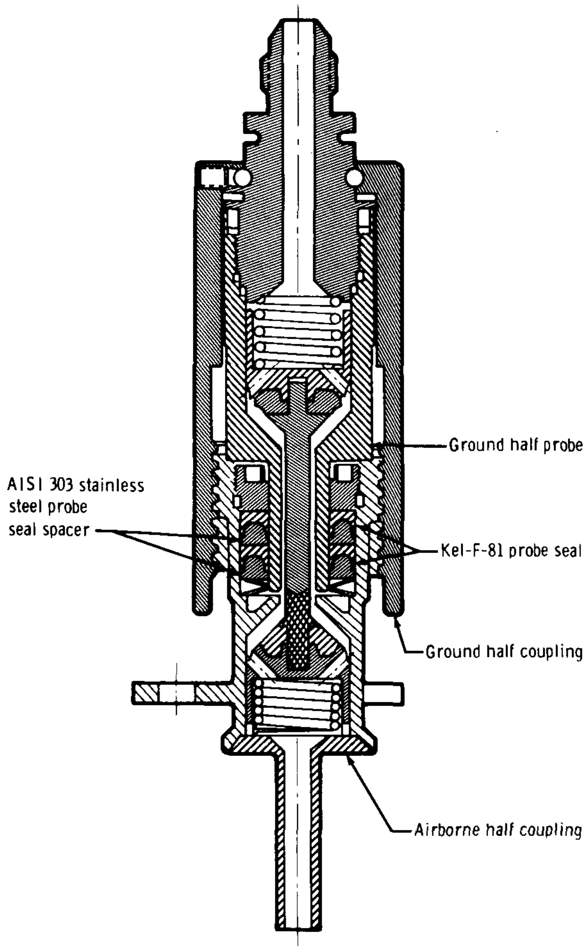


Figure 5. - The CSM RCS helium fill disconnect coupling.

Helium regulator units. - The only significant problem that occurred during the certification testing of the helium regulator units was the tendency of some units to undergo outlet pressure fluctuations at a frequency of approximately 500 hertz. Because these fluctuations did not violate the regulated pressure tolerance bands during certification, no remedial action was taken. Subsequently, oscillations occurred in several units during system- and vehicle-level checkout, and occasionally caused out-of-tolerance regulated pressures. These units were not rejected but waivers were granted because the oscillations occurred at the time that the regulator inlet pressures were below normal.

Helium check valve assemblies. - Several failures occurred in helium check valve assemblies because of out-of-tolerance internal leakage. Most cases were attributed to particulate contamination, and, in some instances, the units were flushed, retested, and found to be acceptable. Because of the nature of these failures, no corrective action was taken at the time of qualification. However, as assembly and checkout of the vehicles progressed, the large number of check valve leakage failures that were attributed to particulate contamination of the seats required corrective action. The correction involved a check valve modification to add filters at the valve inlets. This modification was made on spacecraft 101 for the SM RCS and on spacecraft 104 for the CM RCS (fig. 6).

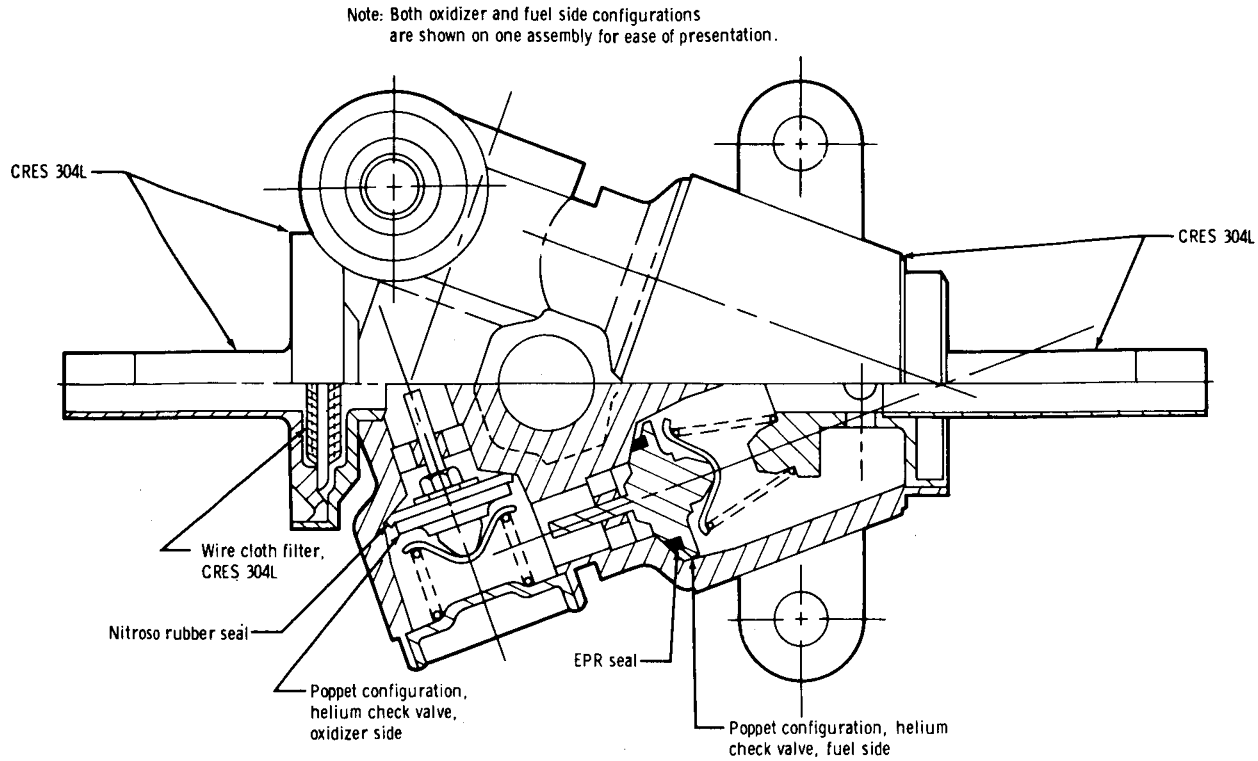


Figure 6. - The CSM RCS helium check valve.

A serious compatibility problem between the soft butyl rubber poppet seal material and the oxidizer was noted during the certification tests. The problem was manifested by internal seal leakage. The seal material for the check valve on the oxidizer side was changed to a nitroso rubber compound on spacecraft 012 and subsequent vehicles. Additional compatibility tests were conducted to certify the new material. At the same time, the valve seal material for the fuel side was changed to ethylene propylene rubber (EPR) to eliminate the difficult process of adding a Teflon coating to the poppet seats, a process that was necessary previously because of the tendency of the original seal material to adhere to the bare metal seats. This change was certified by means of additional compatibility tests. It should be noted that the modifications did not result in complete solution of the compatibility problems. Later, it was noted that these new materials were sensitive to the Freon and isopropyl alcohol (IPA) cleaning fluids. (In future space programs, all materials should be compatible with all fluids to be used in the system.) The use of IPA (in this case) caused the EPR to swell. The swelling could have been controlled by limitation of the exposure time, but the IPA flushing was eliminated. At the same time, it was discovered that Freon flushing fluids that were applied to the nitroso rubber compound caused stickiness and deterioration because the binders were leached by the Freon. A technique for preleaching the nitroso rubber compound was included in the manufacturing process. The use of this technique improved the material significantly, but did not eliminate the problem of stickiness. During the long-term compatibility tests and the qualification verification tests, the nitroso rubber compound was degraded by exposure to nitrogen tetroxide.

Helium pressure relief valves. - Helium pressure relief valve failures that would require hardware modification did not occur during qualification tests. However, on several occasions, the burst disk ruptured at a pressure that was greater than the maximum allowable level. This resulted in a change to the burst pressure level that is set during assembly.

After several cases of corrosion caused by moisture were evidenced on the external side of relief valves, a sealing device was added to exclude atmospheric moisture. The device was an adhesive seal that was pasted over the vent port opening. In case of the relief valve actuation, the device was designed to blow off. The device was reasonably effective, provided it was installed carefully and was used properly. The relief valve changes are indicated in figure 7.

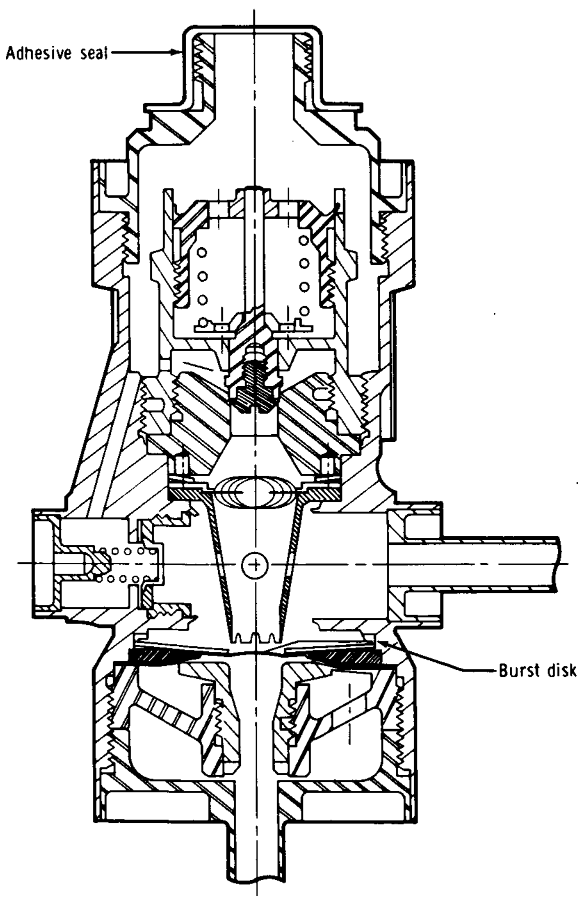


Figure 7.- The SM RCS helium pressure relief valve.

Burst disk assembly. - Five significant problems occurred during certification tests of the burst disk assembly. The internal filter screens were damaged during burst disk actuation. The problem was corrected by adding a stop that was integrated with the closure plug to prevent the burst disk from contacting the screen. Degradation of the burst disk Teflon coating occurred after the nitrogen tetroxide exposure tests. In most cases, the coating bond was destroyed to varying extents by nitrogen tetroxide, which resulted in the deletion of the Teflon coating from the oxidizer burst disks. Initially, the coating was added to increase tolerance to the water in the propellant. After deletion of the Teflon coating, propellant water content in excess of specification could not be tolerated.

Extrusion of the external seal material was a problem that necessitated a redesign of the closure plug to contain completely the exterior O-ring seal. Another problem was exterior leakage of the burst disk seals; this problem required a change of the seal compound from 722-3 to 722-2 nitroso rubber compound. Additionally, low rupture pressures were detected on the oxidizer burst disks after propellant exposure tests were conducted. The cause of these low pressures was not determined; no corrective action was initiated. This problem recurred after the 90-day propellant compatibility testing, but, again, no cause could be determined and no corrective action was taken, because the decrease in rupture pressure was small and no adverse operational effects resulted. Two views of the burst disk assembly are shown in figure 8.

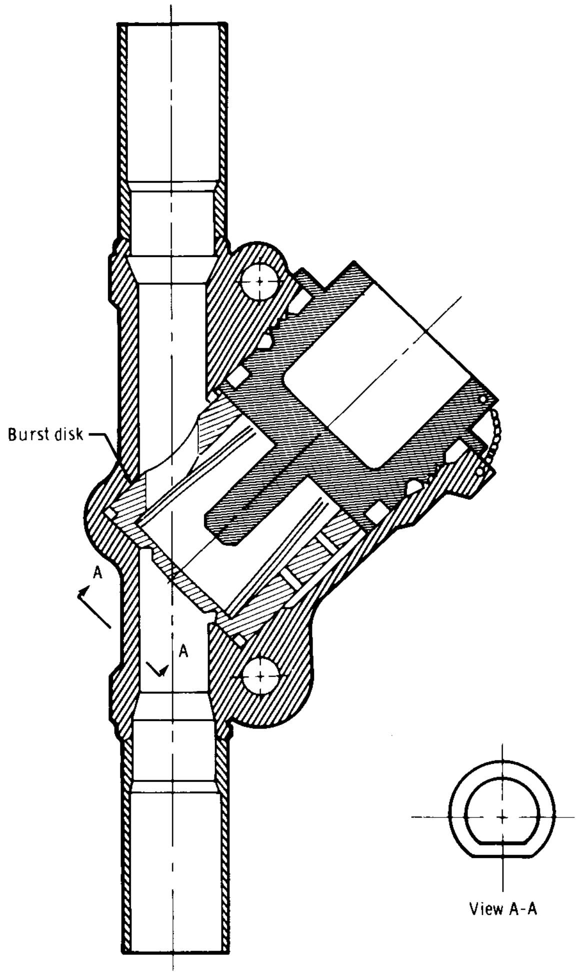


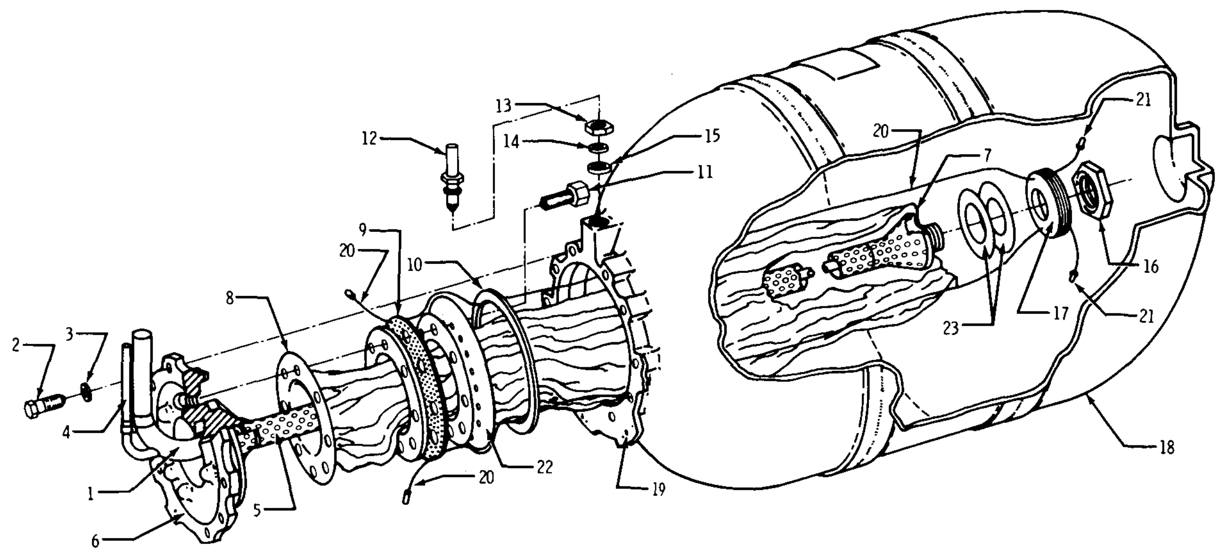
Figure 8.- The CM RCS burst disk assembly.

Propellant tanks. - A limited number of certification tests were completed for the spacecraft 009 propellant tanks (fig. 9). Additional propellant tank tests were completed in support of spacecraft 011, 012, and 103. The tests for the spacecraft 103 propellant tanks were conducted to certify a modification to the bladder flange buffer pad. Holes were punched in these pads to allow for more rapid depressurization during the CM RCS purge operation.

During numerous attempts at propellant tank qualification, significant problems that resulted in hardware changes or changes in operating procedures were encountered. The problems were divided into two categories. One category involved tank shell compatibility; the other category involved mechanical failure of the bladder.

A serious problem was encountered during propellant tank compatibility tests. After a relatively short period of exposure, one of the oxidizer tanks began to leak. An additional group of tanks was subjected to similar exposure tests. The rupture of the tanks that were subjected to pressure began to occur after relatively little exposure. After extensive investigation, it was determined that a lack of nitrous oxide in the nitrogen tetroxide caused rapid stress corrosion of the titanium alloy. The oxidizer that had been used earlier in the program had been manufactured without control of the nitrous oxide content. In 1964, in an effort to improve the uniformity of the propellant, the U.S. Air Force reduced the previously allowable amount of nitrous oxide content to zero. The compatibility test failures occurred while this oxidizer was being used. Therefore, the problem was not evident earlier in the Apollo Program or during the Gemini Program. The corrective action was to specify and control the nitrous oxide quantity in the nitrogen tetroxide between 0.4 to 0.8 percent by weight. Once this corrective action was taken, no further stress corrosion problems with the oxidizer were encountered. However, the investigative effort resulted in the use of the fracture mechanics theory on all Apollo pressure vessels. Briefly stated, this theory is that flaw growth (to failure) in a pressure vessel can be predicted by using empirical data on the vessel material and the vessel pressure history. The application of these techniques to the Apollo pressure vessels is described in reference 1.

The bladders on the Apollo tanks were changed from the independent-ply construction (used on Gemini tanks) to a laminated single-ply construction to preclude using bladders that had inner-ply failures which were not in evidence during checkout. The

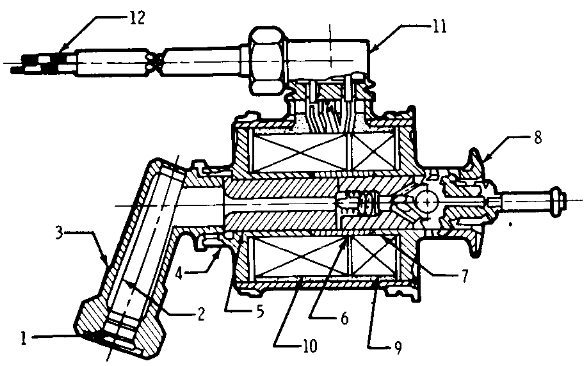


No.	Equipment	Description	No.	Equipment	Description
1	Outlet tube	CRES 347	13	Nut	CRES 347
2	Bolt	Titanium	14	Gasket	Teflon TFE
3	Washer	CRES 347	15	Gasket	Teflon TFE
4	Liquid side vent tube	CRES 347	16	Nut	CRES 347
5	Diffuser tube	6061 aluminum	17	Washer	6061-T6 aluminum
6	Flange	6061 aluminum	18	Shell	Titanium
7	Retainer	6061-T6 aluminum	19	Nut plate	CRES A286
8	Bladder	Teflon TFE/FEP	20	Vent cord	Teflon TFE
9	Ring	6061 aluminum	21	Flanged eyelet	CRES 304
10	Gasket	Teflon TFE	22	Pad	Teflon TFE/FEP
11	Bolt	CRES 347	23	Pad	Teflon TFE
12	Fitting	CRES 304L			

Figure 9. - Typical CSM RCS propellant tank.

inner-ply failures caused the propellant to become trapped between plies. Bladder cycle life was a recognized problem. During propellant servicing, the repositioning of previously twisted bladders caused high stresses on tank bladders, and failures occurred after relatively few cycles. Corrective action included the prepositioning of the empty bladders and the installation of the vent tubes so that the gas could be vented as propellant was added. The vent tubes reduced the cycle life problems in the tanks that were serviced in the vertical position, but were of little value for service to the horizontal tanks. In an effort to solve the bladder stress problem during service of the horizontal propellant tanks, the highly stressed ends of the oxidizer bladders were thickened. All these corrective actions were helpful in meeting cycle life requirements. However, the cycle life requirements eventually had to be reduced significantly. Additionally, the service procedure was modified for the horizontal tanks to ensure that the tanks were never completely filled. High stressing was prevented by eliminating the last phase of bladder repositioning by limiting the oxidizer amount to 96 percent. One further bladder problem involved chafing of the bladder ends during vibration. Corrective action consisted of the addition of a Teflon buffer between the tank and bladder pads at the ends of each bladder.

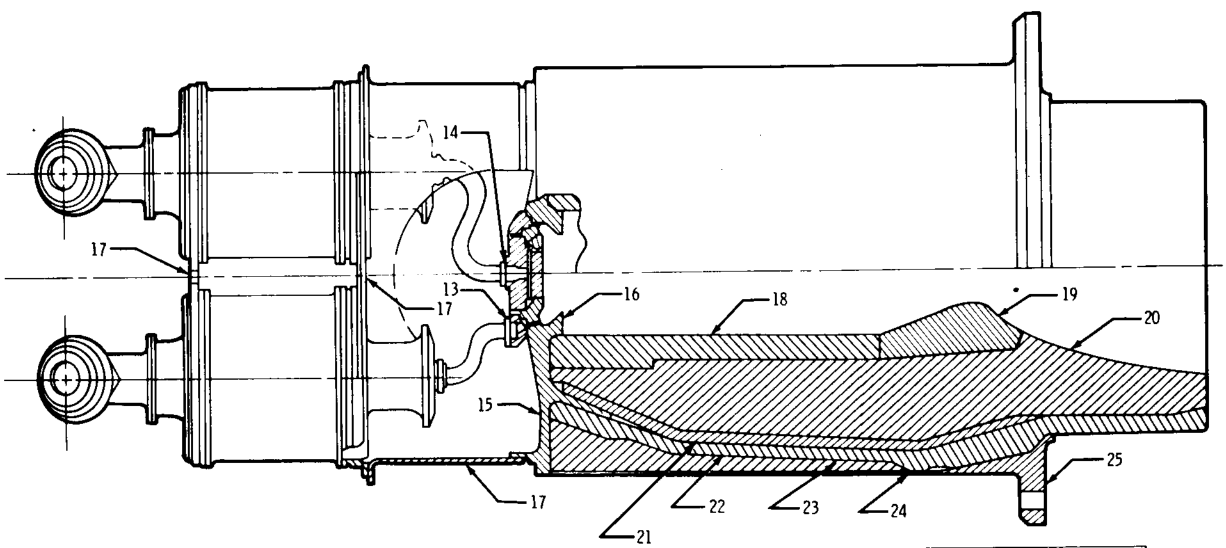
<u>The CM RCS engine</u>. - Several problems occurred in the CM RCS engine (fig. 10) during qualification, and changes were required. Spalling of the JTA throat insert



(a) Valve assembly.

during the humidity and salt fog exposure tests required corrective action that included an increase in the minimum allowable throat insert density, the addition of protective covers to the engine outlets, and the application of epoxy to the engine throats.

Particulate generation occurred during random vibration tests and induced valve seat leakage. It was necessary to coat the engine chambers and throats with epoxy to control the particulate generation that caused valve seat leakage.



No.	Equipment	Description
1	Trim orifice	CRES 17-7PH
2	Filter	CRES 304L, 321, 347
3	Inlet housing	CRES 17-7PH
4	Valve body (bobbin)	CRES 321, 430F
5	Core	CRES 430F
6	Spring	CRES 17-7PH
7	Armature	CRES 430F with Stellite ball
8	Valve seat assembly	CRES 321, 17-7PH with Teflon FEP seal
9	Automatic coil	
10	Direct coil	
11	Leadwire housing	CRES 347
12	Leadwires	AWG 20, MIL-W-16878, type EE, 19-strand, nickel coated
13	Fuel manifold	
14	Oxidizer manifold	
15	Injector assembly	CRES 321

No.	Equipment	Description
16	Splash plate	
17	Valve mounting bracket	CRES 321
18	Thrust chamber sleeve	6° oriented, high-silica, fabric-reinforced, laminated phenolic
19	Throat insert	Refractory graphite, zirconium diboride, silicon composite
20	Nozzle body	45° oriented, high-silica, fabric-reinforced, laminated phenolic
21	Asbestos wrap	
22	Glass wrap	Uncured phenolic, resin-impregnated, glass roving
23	Insulation	Alumina silica fiber
24	Thrust chamber shell	CRES 321
25	Mounting flange	

(b) Engine assembly.

Figure 10. - The CM RCS engine assembly (Block I and Block II).

Poor performance and a changing of the O/F ratio were evident during the pulse performance tests. The corrective action that resulted included the design of a one-piece orifice cartridge to reduce leak paths, the verification of proper orifice installation to prevent bypass flow, and the tightening of controls on the valve seat manufacturing operation. Another problem was that the nozzle extension seal leaked during the hot fire tests; secondary O-ring seals were added.

Supplemental qualification tests were conducted late in the program to certify a 6° oriented uncharred combustion chamber liner as a replacement for the original 6° oriented precharred liner. The change was required because of delamination, blistering, and tackiness problems with the old billets. The apparent cause for this problem was a breakdown of the fabrication control.

Additional supplemental qualification tests were conducted to certify the engine for 200-second deorbit burns that would be required for a hybrid deorbit. Performance degradation was allowed, but engine structural integrity had to be maintained for this type of burn.

Although the problems that developed during the qualification tests were resolved, the rejection rate was very high after the engines were being produced and checked out on the vehicles. The primary problems were internal valve leakage and slow valve response. After the qualification tests, failure analyses showed that the problems were caused by various forms of internal engine contamination: miscellaneous particulate contamination, corrosion products, and residual propellant contamination from the hot fire calibration tests. In all cases, it was determined that the engine valves were not designed for effective cleaning. Although numerous changes were made in the manufacturing and assembly, decontamination, and quality control procedures, the basic problems were never resolved completely. Engine rejection rates caused by this problem remained relatively high. In some cases, discrepant engines were not found until the first checkout after installation on the spacecraft.

Helium solenoid valve. - A basic design deficiency was noted during certification testing of the original 1/4-inch helium solenoid valve. An incompatibility between nitrogen tetroxide and the nickel plating in the plunger bore caused binding of the plunger in the bore and rendered the valve inoperative. A second, equally severe problem was the lack of effective production of the metal-to-ceramic seals that were used to seal the wiring. The severity of these problems resulted in replacement of the valve in spacecraft 017 and subsequent vehicles with one of a newer design.

One of two failures during certification of the new design unit was valve leakage that was caused by particulate contamination, which subsequently required the use of special cleaning procedures before final valve assembly. The other failure was an uninitiated closure of the valve during supplemental qualification vibration tests (random vibration) that necessitated the installation of a new bellows and flow deflector assembly that had a lower mass than did the old unit.

Propellant solenoid-type latching valves. - The original propellant solenoid-type latching valve was used on spacecraft 009 only. Tests revealed that severe propellant incompatibility existed, which caused the valve to fail. Furthermore, the position indicator switch was not linked mechanically to the poppet, but was driven by a separate armature. Thus, a stuck poppet could not be detected by the indicator. The design deficiencies resulted in the replacement of the valve. No significant problems were encountered during the subsequent certification tests of the new valve.

A shortcoming of the qualification tests was observed during the Apollo 9 mission. Some of the valves closed inadvertently at CSM/Saturn IVB (S-IVB) separation. Investigation proved that the shock from separation pyrotechnics, estimated to be up to 260g for 1 to 3 milliseconds, was sufficient to cause the valves to close. The problem occurred on subsequent missions and also involved the helium isolation valve. No design changes were made in either unit because the problem could be handled operationally by cycling the valves, which had inadvertently closed, to the "open" position.

Propellant quantity gaging system.- Design deficiencies were evidenced by numerous failures during the tests of virtually all assemblies within the radioisotope propellant quantity gaging system. Functional failures and out-of-specification problems were a source of constant trouble during the qualification period. Although the equipment was on spacecraft 009, it was not qualified and was not operational. The system was removed from spacecraft 011 and from subsequent vehicles and was replaced with the PVT ground computer program by the use of existing instrumentation. Later, the onboard P/T sensor technique was added to provide gaging information (independent of the PVT ground computer program) in the spacecraft.

The SM RCS engine. - Certification of the SM RCS engine was initiated on a 95-pound thrust configuration unit for spacecraft 009. This unit did not have the thermal insulator on the fuel valve. Basic certification was continued on the spacecraft 011 configuration that had the thrust increased to 100 pounds and incorporated the fuel valve thermal insulator.

During the certification program, two basic problems developed. Injector manifold explosions caused several engine failures that were attributed to a facility problem which was solved by the use of tests conducted at a higher vacuum level. No hardware changes were made on the engine.

Engine explosions that occurred during pulse-firing duty cycles were a severe problem. Three corrective actions were taken to resolve the problem. A preignitor was incorporated in the injector to provide more uniform ignition. Furthermore, the engine bell nut temperature was limited to a minimum of 30° F, and the fuel was changed from A-50 to MMH. The last change was the most significant; it reduced the unstable residues that were formed during pulse-firing modes and that were the basic cause of the explosions.

Supplemental qualification programs were conducted to certify the engine for use with helium saturated MMH fuel, with 0.4 to 0.8 percent nitrous oxide in the nitrogen tetroxide, and with arc suppression in the valve electrical circuitry.

The SM RCS heaters. - Two heater configurations were used on the SM RCS. Early units had an integrally mounted thermostat. On spacecraft 103 and subsequent vehicles, all units had remotely placed thermostats. During the life cycle tests on a unit that had an integrally mounted thermostat, a bonding material failure occurred. The failure was attributed to test equipment; no corrective action was taken. Later, in spacecraft 008 testing at MSC, the design inadequacy in the heater bonding technique was noted again.

As a result of the problem, a mechanical clamp was used to hold the heaters in place until a new bonding technique could be developed and verified.

Additional certification. - A 90-day propellant compatibility program and a component quality verification program were conducted. The primary objective of the 90-day propellant compatibility test was to demonstrate the capability of the Apollo Block II SM RCS to sustain propellant exposure for 90 days. A secondary objective was to determine and demonstrate the minimum launch pad support that would be required during the standby portion of a 90-day period. Another secondary objective was to demonstrate propellant compatibility for extended mission durations.

Three tests were used to support the 90-day test objectives. The first test was to demonstrate 90-day propellant compatibility without requirements for special continuous purges, bleeds, and system exercising, and to establish the simplest method of launch pad operations and support. The second test was to demonstrate SM RCS operation for a 103-day period that included 13 days of prelaunch plus 90 days of postlaunch test activity with the system in operation. The third test was to demonstrate 90-day propellant compatibility by the use of special protective procedures and operations to provide maximum assurance of achieving 90-day propellant compatibility with the existing systems and components. At the conclusion of the exposure tests, all three quads were test fired successfully and were run through a post-test functional check. After the system-level tests, the quads were disassembled and the components were sent to the various contractors for inspection and testing. From the results of these tests, it was established that the system and components had at least a 103-day propellant exposure capability. No component modifications were made as a result of the tests; no failures that required corrective action occurred.

The component quality verification tests were conducted on selected components to verify that the units that were produced were of the same quality as the units that were used to qualify the component. Explosive and solenoid-operated isolation valves, check valves, propellant tanks, relief valves, regulators, and engines were subjected to selected qualification-type tests (including environmental tests). All units passed these tests satisfactorily; no corrective action was necessary.

FLIGHT MISSIONS

Active reaction control systems were not needed or used on the early boilerplate (BP) Apollo flights (BP-15, BP-26, BP-9, and spacecraft 002). However, thermal and vibration data obtained from inactive or simulated hardware on these flights were useful in the design and testing of the first RCS hardware. Data obtained on later flights (beginning with spacecraft 009) included performance evaluation and anomalies within or affecting the CSM RCS. Changes in hardware or procedures (or both) were instituted as a result of the flight experience.

Mission A-102

Boilerplate 15 (BP-15) was launched down the Eastern Test Range from KSC on September 18, 1964. The launch vehicle (SA-7) was an S-I Block II vehicle. The purpose of mission A-102 was to demonstrate the compatibility of the spacecraft with the launch environment. The CM and SM were boilerplates that were used to simulate the actual vehicles in dimensions and mass characteristics.

The SM RCS was simulated by the use of four boilerplate external engine packages, and each boilerplate consisted of the engine package and four simulated engines. Only quad A was instrumented. Sixteen thermocouples were mounted on the positive-pitch, negative-pitch, and counterclockwise-roll engines of quad A. Two accelerometers were mounted in the nozzle of the clockwise-roll engine.

By means of prelaunch thermal analysis, it was predicted that SM RCS engine nozzle temperatures would be in the 1800° to 2200° F range as a result of launch aerodynamic heating. Flight data indicated maximum temperatures of 700° to 800° F, but were invalid for two reasons. First, major differences in geometry, material, and thermal "capacitance" existed between the boilerplate engines that were used and the actual engine hardware. Second, the relatively massive, sheathed thermocouples that were used on the flight were unsatisfactory for the measurement of large temperature transients. The reasons were verified in postflight tests and indicated that further tests should be performed.

The vibration data indicated that the energy was concentrated primarily at a frequency of 190 hertz, at which frequency the power spectral density analyses indicated values from $100g^2/Hz$ to $115g^2/Hz$. These levels, although greater than the design levels, were questionable because of dissimilarities between the boilerplate hardware and the actual hardware.

Missions SA-8 and SA-10

Because of the uncertainties in the SM RCS temperature data from BP-15, two additional tests were planned. These tests were to be added to the existing SA-8 and SA-10 missions, which were launched on May 25 and July 30, 1965, respectively. The purpose of the tests was to obtain sufficient launch heating data for the SM RCS engines to facilitate verification of the capability of the engines to withstand the launch environment. These tests were conducted during the SA-8 mission (BP-26) and the SA-10 mission (BP-9); both test vehicles were launched by the S-IB from KSC.

Each boilerplate SM was fitted with a single RCS quad A engine package that consisted of the engine housing and four engines. The internal components of the quad, which included the propellant storage and helium pressurization systems, were not installed. Twelve fast-response thermocouples were installed on the upward-firing (positive-pitch) engine and on the side-firing (negative-roll) engine.

The BP-26 and BP-9 followed almost identical launch trajectories, and the temperature-time data from the engines were similar. The maximum temperatures measured were 1320° F on the lip of the upward-firing engine nozzle and 1390° F on the upper (windward) surface of the side-firing nozzle. The temperatures peaked at approximately T + 132 seconds. Based on these temperature-time data, heating rates were determined and were correlated with the aerodynamic flow field model. As a result, the worst-case launch heating predictions were revised; the capability of the hardware to withstand the launch environment was verified.

Mission A-004

Mission A-004 (spacecraft 002) was launched with a Little Joe II vehicle on January 20, 1966, at the White Sands Missile Range, New Mexico. The primary objective of the mission was to subject the CM to a power-on tumbling abort. The secondary objective was to determine vibration levels on the SM structure and on the SM RCS.

The results of previous acoustic and vibration tests on the SM structure and the results of the RCS flight data for BP-15 caused concern. Consequently, one quad on spacecraft 002 was instrumented with six vibration sensors (two on the forward engine, two on the counterclockwise engine, one on the panel, and one on the oxidizer tank support). Masses were used instead of tanks and propellants. The results of the flight indicated that the vibration levels were nominal. The concern that was based on the previous acoustic and vibration tests was dispelled.

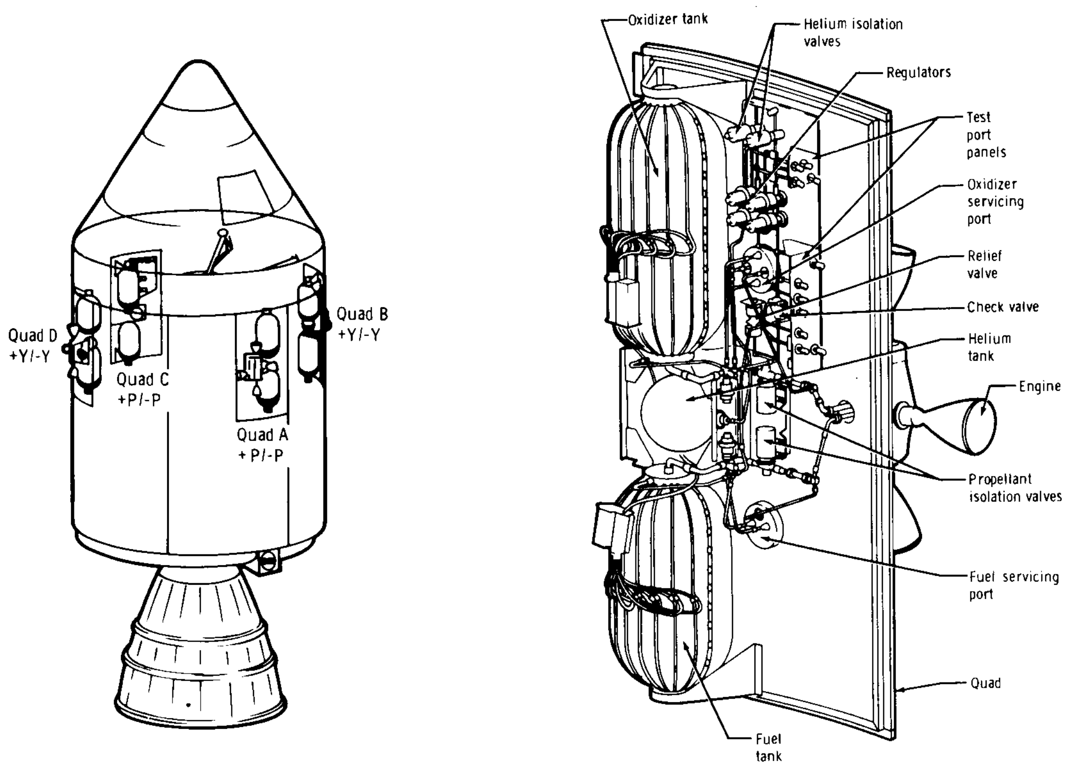
Mission AS-201

Mission AS-201 (spacecraft 009) was the first flight test of a production Apollo Block I type spacecraft in conjunction with the S-IB launch vehicle. The unmanned suborbital flight was launched from Complex 34 at KSC on February 26, 1966. The CM landed safely in the primary landing area near Ascension Island in the Eastern Test Range approximately 37 minutes later and was recovered as planned. The major mission objectives were the demonstration of the compatibility and structural integrity of the spacecraft and the S-IB configuration, and the evaluation of the spacecraft heat shield performance during a high heating rate entry.

Configuration of the CSM RCS. - The SM RCS was the basic Block I system located on the SM (fig. 11); a system schematic is shown in figure 12. The propellants that were used for mission AS-201 were nitrogen tetroxide (oxidizer) and a blend of 50 percent by weight hydrazine (N_2H_4) and 50 percent by weight unsymmetrical dimethylhydrazine (UDMH) fuel. The CM RCS was the basic Block I configuration that is shown in figures 13 and 14; a schematic of the propellant feed systems is shown in figure 15.

The only SM RCS components that were known to be malfunctioning before lift-off were the propellant quantity gaging system, inoperative because of problems encountered during checkout, and a number of primary-stage helium check valves that were leaking. All secondary-stage check valves were functioning normally.

The only CM RCS components that were known to have failed before lift-off were a system B relief valve burst diaphragm and, as was the case with the SM, some of the primary-stage helium check valves. None of the malfunctioned components caused degradation of the mission.



(a) Service module, showing location of quads.

(b) Quad, showing location of components.

Figure 11. - The SM RCS (Block I).

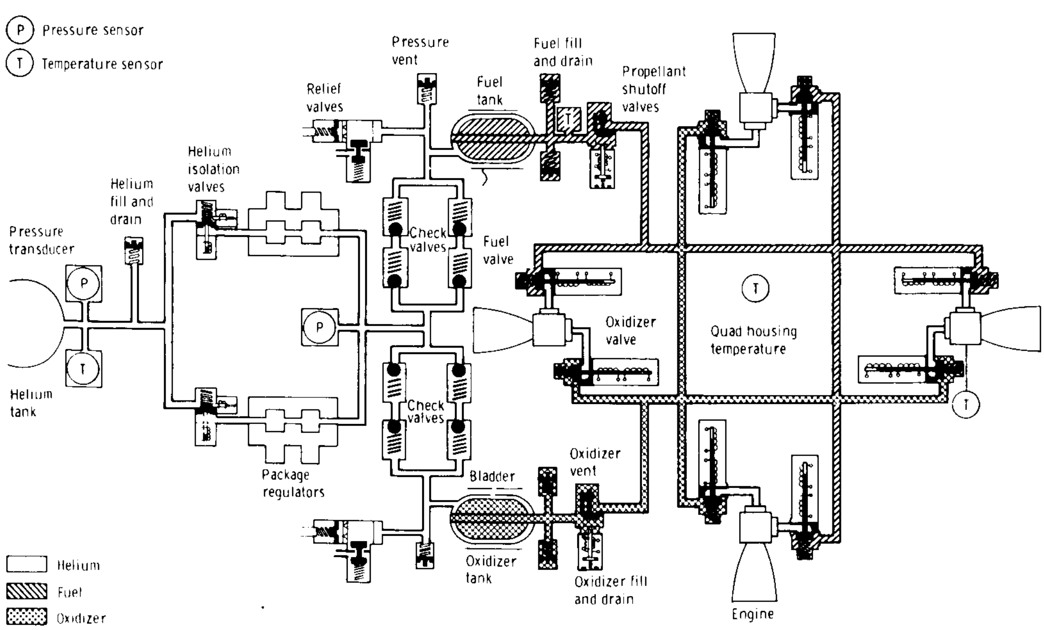


Figure 12. - Schematic of the SM RCS quad (Block I).

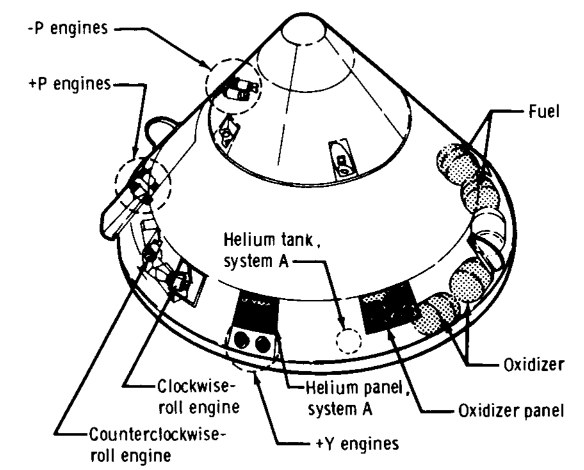


Figure 13. - Component locations on the CM RCS (Block I, +Y axis).

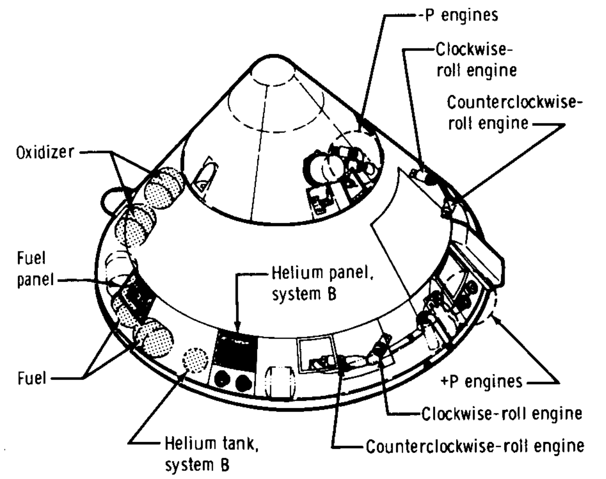


Figure 14. - Component locations on the CM RCS (Block I, -Y axis).

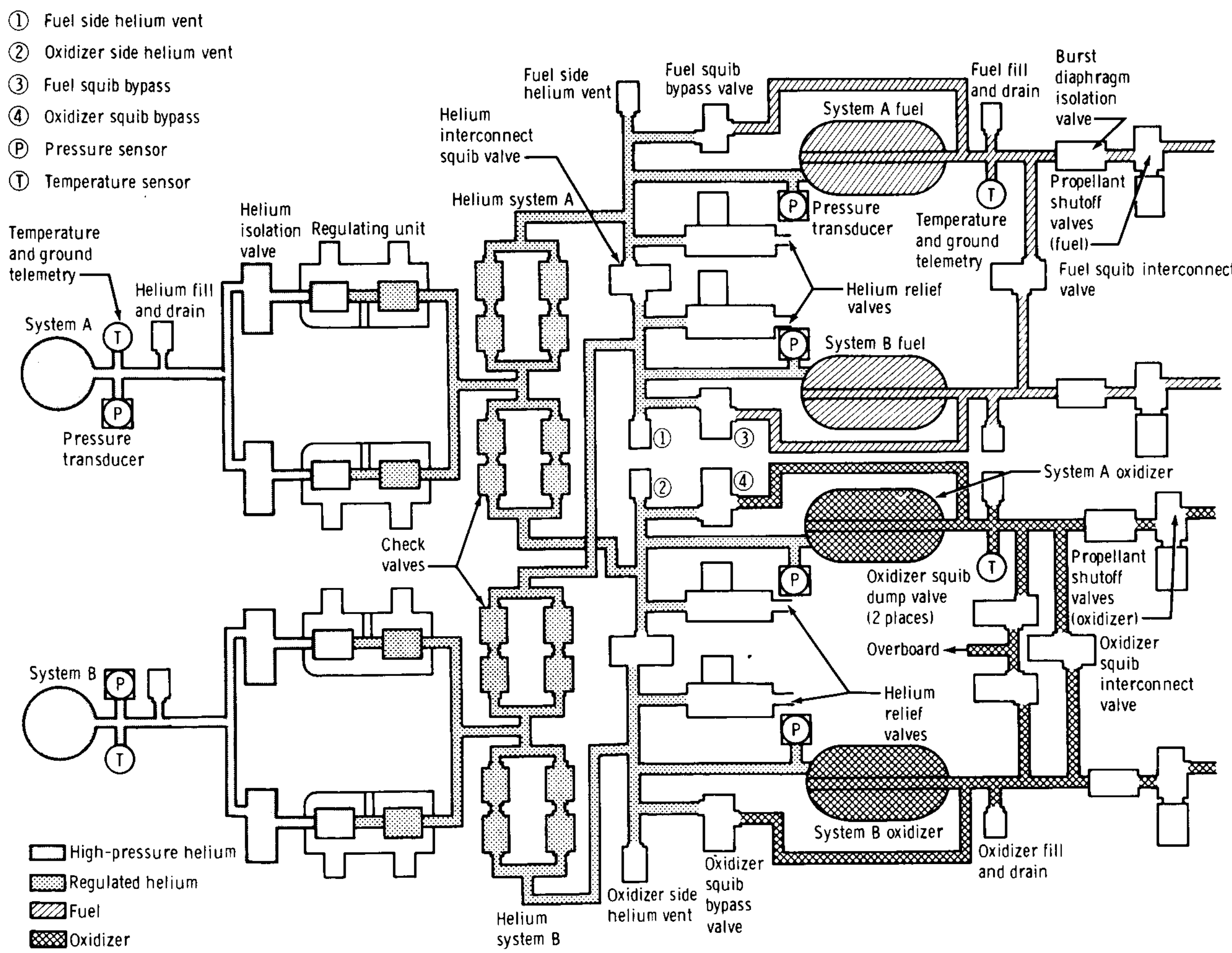


Figure 15. - Schematic of the CM RCS propellant feed systems A and B (Block I).

Performance summary. - The SM RCS successfully performed the pitch maneuver that was required before CM/SM separation. The proper attitude control was maintained, even though quad A was inoperative and one of the negative-yaw (-Y) engines (probably on quad D, although this was never positively established) either was inoperative or produced only partial thrust. However, as a result of the quad A and -Y-engine problems, the positive translation (+X) maneuvers produced less than a nominal velocity change when the spacecraft attitudes and rates were maintained. The performance compared favorably with that which was predicted, considering the effects of the disabled engines, and nominal engine thrusts were produced by the operating engines.

The CM RCS successfully performed all the required maneuvers and maintained proper spacecraft control until electrical problems caused the system B engines to be disabled at T + 1641 seconds and caused the system A engines to be disabled at T + 1649 seconds. Command module control was maintained through the maximum dynamic pressure (max_q) region.

Anomalies. - Several failures were incurred during mission AS-201. The SM quad A became inoperative because of the "closed" position failure of the oxidizer isolation valve. The failure was not detected during system activation because the indicator switch was not linked mechanically to the poppet. A partial, or possibly complete, loss of thrust from one of the -Y engines occurred when the automatic coils were used. The engine that was involved and the cause of the failure could not be determined definitely from the available data. After blackout, both systems A and B of the CM RCS were lost because the RCS control motor switches were transferred from the "CM" to the "SM" position. The system B logic power failure resulted in the loss of system B of the CM RCS for use in the propellant depletion burn. The system B logic power failure caused the loss in the CM RCS of the use of the systems A and B helium interconnect valves, the system A fuel tank helium bypass pyrotechnic valve, and the system B oxidizer tank helium bypass pyrotechnic valve. The failure of the CM RCS oxidizer isolation valves to close during the postflight deactivation occurred because of the incompatibility between the valves and the oxidizer.

Corrective action. - As a result of the failures just described, the following corrective actions were implemented. The incompatible propellant isolation valve was replaced with one of a new design that was compatible with the propellant. Additionally, the position indicator switch within the valve was linked mechanically to the poppet and thereby registered the correct poppet position. The remaining electrical problems were caused by CM/SM umbilical wires that were still energized electrically when severed by the guillotine. When high heating rates were encountered during entry, the insulation burned off and many of the wires were shorted. The shorts resulted in the electrical problems that have been described. To preclude recurrence of this type of problem, all electrical circuits that were to be severed by the guillotine were isolated upstream from the guillotine before CM/SM separation. Furthermore, a redundant transfer switch was added to the control circuitry to ensure that RCS engine control would be transferred from the SM to the CM.

Mission AS-202

Mission AS-202 (spacecraft 011) was the second flight test of a production Apollo Block I spacecraft that used the uprated S-I launch vehicle. This unmanned suborbital flight was launched from Complex 34 at KSC on August 25, 1966. The CM landed as scheduled in the primary landing area in the southwest Pacific Ocean (near Wake Island) approximately 1 hour 33 minutes later and was recovered as planned. The major mission objectives were to prove the structural integrity and compatibility of the combined spacecraft to the uprated S-I configuration, to verify the operational capability, and to evaluate the CM heat shield performance during a high heat load entry.

Configuration of the CSM RCS. - The SM RCS configuration on spacecraft 011 was identical to that on spacecraft 009 with the following exceptions. The SM RCS engine for spacecraft 011 was of the spacecraft 012 configuration; that is, the SM RCS engine produced 100 pounds rather than 95 pounds of thrust and had a fuel valve thermal standoff to increase thermal resistance between the valve and the injector. The propellant isolation valves had been redesigned and had improved performance and propellant compatibility characteristics. The helium isolation valve was of the spacecraft 012 configuration; that is, the helium isolation valve had an improved poppet design. The fuel used for this mission was MMH rather than A-50.

With the following exceptions, the CM RCS configuration on spacecraft 011 was identical to that on spacecraft 009. The propellant isolation valves were of a new design and had improved performance and propellant compatibility. The CM RCS engine for spacecraft 011 was of the new configuration that included an epoxy-coated throat and liner and an improved valve design. The propellant tanks were of the new configuration that had tank bladders which were the same size as the tank shell rather than longer than the shell. Additionally, the ends of the oxidizer tank bladders were 9 mils rather than 6 mils thick.

Performance summary. - The CM RCS and SM RCS inflight performances were nominal throughout the mission. All maneuvers that involved RCS thrusters were completed as planned, and the attitude rates that were attained were as predicted.

Anomalies. - Only one SM RCS component malfunctioned during the mission. Shortly before lift-off, a quad C relief valve burst disk (fig. 7) was ruptured. The burst disk rupture occurred during activation of quad C on the pad approximately 4.5 hours before launch. When the helium isolation valve on quad C was opened to pressurize the propellant tanks, the pressure downstream from the regulators surged to 320 psia during the activation transient and ruptured the relief valve burst disk. Subsequently, the pressure stabilized at 220 psia after activation, which indicated that the overpressure ruptured the burst diaphragm and vented overboard through the relief valve. On the first SM RCS burn, the pressure downstream from the regulator dropped to the nominal regulated pressure, which indicated that the regulator was functioning properly.

Also, one CM RCS anomaly was identified. During postflight inspection, the CM system A oxidizer and the system B fuel relief valve burst diaphragms were noted to have been ruptured. An examination of the data taken at the time of CM RCS activation indicated that a pressure surge occurred which was similar to the pressure surge described for the SM RCS quad C. In the case of the CM, the ullage volumes were smaller than the ullage volumes of the SM, providing less "capacitance" to absorb the regulator activation surge.

Corrective actions. - Although the failures in the SM and CM were similar, the corrective actions were different. The SM problem was resolved by means of prepressurization of the volume downstream from the regulator to control the lockup pressure. Therefore, when the helium isolation valves were opened to activate the system, the regulators were already in the "closed" position. The CM problem could not be resolved by means of a simple procedural change because it was not possible to fly the entire mission with a prepressurized CM RCS. Already, restrictive orifices had been placed in the helium supply lines to limit the maximum helium flow rate. It was decided to reduce the orifice size further to eliminate the activation pressure surge. Because the burst disk rupture on the CM did little to compromise system reliability, it was decided to make the change effective on Block II vehicles.

Apollo 4 Mission

The Apollo 4 spacecraft (spacecraft 017) was launched from KSC on November 9, 1967. Mission objectives included demonstration of (1) the compatibility of the S-V launch vehicle and the spacecraft and (2) the adequacy of the Block II entry systems under simulated high heat rate lunar return conditions. The spacecraft 017/S-IVB combination was placed in an earth parking orbit for two revolutions. Then, the S-IVB stage was reignited to place the spacecraft in a simulated translunar trajectory. The spacecraft was cold soaked for 4.5 hours, and the thickest side of the CM heat shield was positioned away from the solar vector. At the conclusion of the 4.5-hour cold soak, the SPS engine was fired to increase the spacecraft inertial velocity. After the SPS burn, the CM was separated from the SM, and the CM was oriented to the entry attitude. The SM RCS thermal control system was used for the first time on this mission, and it performed satisfactorily throughout the flight.

The heaters on all four quads actuated satisfactorily in a repeatable manner. The quad package temperatures and the temperatures of the engine injector heads that were instrumented were maintained within acceptable temperature limits during CM/SM separation.

The CM RCS engines were maintained passively at acceptable temperatures from launch to system activation. The maximum engine temperatures that were encountered from system activation to landing were within design limits.

Configuration of the CSM RCS. - The SM RCS configuration, except for the addition of a modified heavy engine mount structure and an engine heater system, was similar to that of spacecraft 011. The CM RCS configuration of spacecraft 017 was identical to that of spacecraft 011.

Performance summary. - The SM RCS and CM RCS inflight performance was within the nominal range throughout the mission. All maneuvers were completed as planned during the time the RCS engines were used. Satisfactory maneuver rates, accelerations, and translational velocity changes were attained. Propellant usage was normal for both the CM RCS and the SM RCS. Approximately 16 000 SM engine firings were performed during the course of the mission.

Anomalies. - The only anomaly that was associated with the SM RCS was a prelaunch pressure decay in the quad A helium source pressure. The pressure data indicated a fairly uniform decrease from 4150 psia at servicing to 3910 psia at launch. The leak rate was approximately 5 psi/hr (26 sccm). A decay rate as much as 17 psi/hr and a minimum pressure as low as 3440 psia were acceptable for this unmanned spacecraft launch; hence, the leakage caused no problems relative to the mission.

Two anomalies were noted on the CM during postflight examination. After landing, there was a residual pressure of several hundred psia in the CM RCS after the purge operation. The residual pressure was attributed to the configuration of the propellant tanks and the helium purge systems.

By examination of the CM RCS, it was noted that there were ruptured burst disks in the system A oxidizer relief valve and in the system B fuel relief valve. The anomaly was expected because it had been characteristic of all previous missions and the ground-based test programs.

Corrective action. - The leakage problem on systems of this nature had been a continuous problem. Leak checks before servicing were intended to preclude problems of this type. Because the source of the SM RCS leakage was never identified specifically, no definite corrective action could be taken against any one suspected component. The general action that was taken included added emphasis on leak checks, particularly where mechanical joints (screw fittings) were used.

The residual pressure that was noted in the CM RCS after landing resulted in two corrective actions. First, the Teflon disks that covered the helium vents were perforated to facilitate easier helium outflow from the tanks. Second, the purge bypass line discharge was moved from the liquid side vent line to the tank discharge line. This change made it possible for gas outside the bladder to vent directly into the discharge line without passing through the inside of the bladder. The change negated the capability for purging any residual propellant from inside the tank. Retaining the capability for the purge was not a necessity. The corrective action for the ruptured relief valve burst disk involved the relocation of the two relief valves to provide more volume between the regulators and the relief valves.

Apollo 6 Mission

The Apollo 6 spacecraft (spacecraft 020) was launched on April 4, 1968, from Complex 39A at KSC. The mission objectives were essentially the same as those for the Apollo 4 mission. The launch-phase profile was nominal until two engines in the S-II stage shut down prematurely. The shutdown caused the remaining three S-II engines and the S-IVB engine to fire longer than was planned to obtain the desired velocity. During the S-IVB engine firing, a substantial amount of steering was needed in an attempt to correct the error that the S-II stage had generated in the trajectory plane. At the time of thrust termination, the orbit was 198 by 96 nautical miles instead of the 100-nautical-mile circular orbit that had been planned. The vehicle remained in earth orbit for the next 3 hours. During this period, systems were checked, operational tests (such as S-band evaluation) were conducted, and several attitude maneuvers were performed.

The second S-IVB firing was scheduled to occur during the time of the pass over KSC at the end of the second orbit, but the firing could not be accomplished. Therefore, the CSM was separated from the S-IVB, and an SPS engine firing sequence was initiated. The firing was of a long duration of 442 seconds that resulted in a 12 019.5- by 18-nautical-mile free-return orbit. After SPS engine cut-off, the CSM was maneuvered to a cold-soak attitude, in which the aft end of the spacecraft was oriented sunward. This attitude facilitated the attainment of the desired shading on the CM. The cold-soak attitude was maintained for approximately 6 hours.

Because the SPS was used to insert the spacecraft into the desired high apogee orbit, insufficient propellant remained to gain the high velocity that was desired from the second SPS engine firing. Specifically, the total propellant that remained would have resulted in only 22 percent of the desired velocity increase. The decision was to inhibit the second firing. A complete firing sequence was performed, including all nominal events, except that thrust was inhibited.

After the SPS cut-off signal, the CSM was maneuvered to a separation attitude and the SM was separated. Then, CM entry attitude orientation and coast to an altitude of 400 000 feet occurred. The entry interface was reached at a velocity of 32 830 ft/sec and a flight path angle of -5.85°. The interface conditions were less than planned; as a result, the heating rates and loads during entry were lower than were desired.

Configuration of the CSM RCS. - With two exceptions, the Apollo 6 SM RCS configuration was similar to that used on the Apollo 4 mission. First, some of the engines had integral propellant inlet filters that were mandatory for Block II. Second, heater brackets were used on the Apollo 6 spacecraft to supplement the adhesive bonding because of a failure of the adhesive on the thermal vacuum test vehicle. The Apollo 6 CM configuration was identical to that of the Apollo 4 CM except that the amount of oxidizer which was loaded was reduced by approximately 5 pounds in each system. These quantities were reduced to ensure that the oxidizer would be depleted before the fuel during the depletion burn. The expulsion of raw oxidizer on the parachutes could cause deterioration of the parachute nylon.

Performance summary. - Both the CM RCS and the SM RCS performed nominally throughout the flight except for the thermal control on one quad. All maneuvers during which the RCS was used were completed satisfactorily. Normal maneuver rates, accelerations, and translational velocity changes were attained. Propellant usage by both systems was normal. The thermal control system for the SM RCS maintained the engine mount structure and injector head temperatures at satisfactory levels for quads A, B, and D. Anomalous temperatures were noted for quad C during the early portion of the cold-soak phase of the mission.

Anomalies. - The only nonnominal performance during the flight involved the abnormal thermal performance of quad C. During the early portion of the cold-soak phase of the mission, the quad C engine mount structure cooled excessively, and the quad C clockwise-roll engine injector underwent unexplained temperature excursions.

During postflight examination, two additional RCS anomalies were found. The burst disks in two of the relief valves were ruptured, an anomaly that was anticipated. Corrective action was identified but was not implemented on this vehicle. A second CM RCS anomaly involved the crosswiring of the oxidizer and fuel valves on the four yaw engines. Crosswiring had no effect on flight performance because the valves were wired in parallel and received a common firing signal. However, a postflight decontamination problem was encountered because the valves were energized one at a time so that the fuel and oxidizer systems could be cleaned independently.

Corrective action. - The corrective actions that were taken after the Apollo 6 mission were limited. The cause of the quad C temperature excursion was never identified positively; therefore, no corrective action was taken. The CM relief valve problem was not a new problem on the Apollo 6 flight; corrective action was planned for subsequent vehicles. The CM engine crosswiring problem was not a hardware deficiency, but was rather a manufacturing error that was not detected by existing quality control procedures; therefore, a new test was added to ensure proper wiring. Many times, this type of corrective action is not practical. Most frequently, ensuring adequate or effective quality control is a more reasonable solution.

Apollo 7 Mission

The Apollo 7 mission (spacecraft 101) was the first manned Apollo mission and was launched from KSC with the S-I launch vehicle on October 11, 1968; the mission duration was approximately 260 hours. The spacecraft landed in the Atlantic Ocean on October 22, 1968. The crewmen were Walter Schirra, Walter Cunningham, and Donn Eisele. The purpose of the mission was to determine the capability of the spacecraft, the crewmen, and the Manned Space Flight Network (MSFN) support facilities to conduct an earth orbital mission.

Several detailed test objectives (DTO's) were defined for the CSM RCS. The SM DTO's were to determine the adequacy of the SM RCS during all primary guidance, navigation, and control subsystem (PGNCS) modes, SCS modes, and manual control modes; to operate in both pulsing and steady-state modes; and to operate by the use of both automatic and manual valve coils. Other DTO's were to obtain data on SM RCS propellant consumption during a variety of maneuvers and control modes and to obtain data on CM RCS usage during entry. Additional objectives were (1) to evaluate the accuracy of SM RCS ΔV maneuvers of the PGNCS, (2) to determine the amount of SM RCS propellant that was needed to maintain spacecraft stability during long-term main propellant slosh after SPS or RCS burns, and (3) to develop an optimum procedure for initiating spacecraft attitude control after the SPS and RCS burns.

The determination of the amount of SM RCS or CM RCS propellant consumption was also included in the following DTO's.

1. Attitude control, using PGNCS

2. Entry, using PGNCS

3. Attitude control, using SCS

4. Backup alinement procedure, using SCS

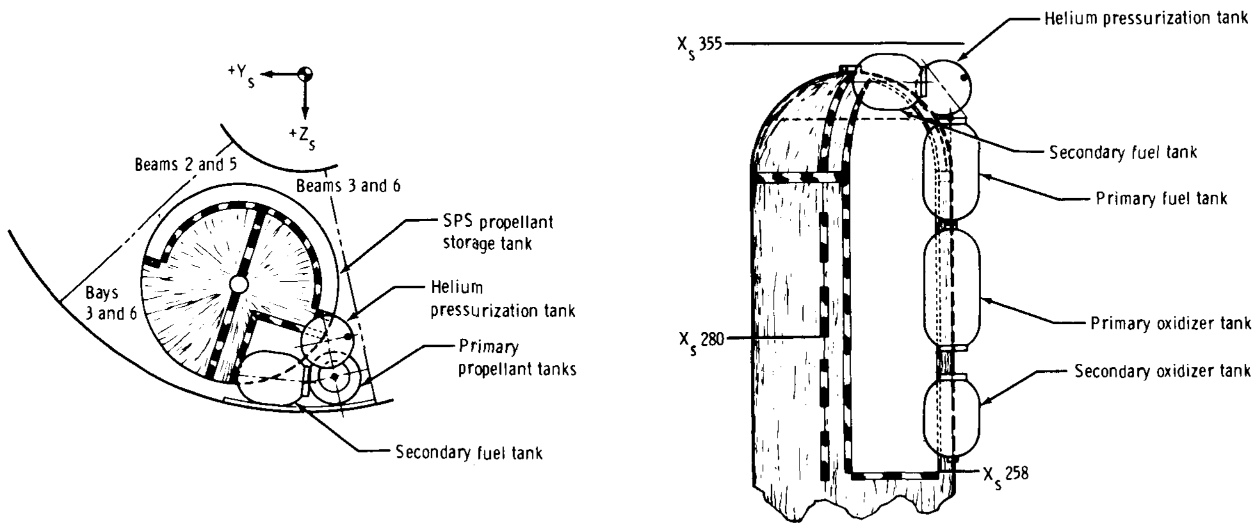
5. Separation, transposition, and simulated docking

6. Active rendezvous of the CSM with another vehicle

Configuration of the CSM RCS. - The Apollo 7 SM RCS had the first Block II configuration. The major change involved the addition of two propellant tanks, which were the size of the CM propellant tanks, on a significantly lengthened SM RCS panel. In addition, several minor changes were incorporated into the Block II configuration. For example, check valves that had integral filters were used at all check locations, and engines that had integral inlet filters were also used. To reduce fire hazards, the SM RCS panel insulation was encapsulated with H-film (polyamide). Propellant isolation valves were added to provide the capability for isolating the new tanks from the engines independently of the old tanks. A larger helium tank was installed to accommodate the increased propellant expulsion requirement. Finally, the onboard P/T and real-time ground-computed gaging systems were used to determine propellant quantity.

A schematic of the Block II SM RCS quad is shown in figure 2, and the quad is shown pictorially in figure 1. The relative locations of the SM RCS and SPS components within the SM are shown in figure 16, the SM RCS quad engine package is shown in figure 17, and an individual SM RCS engine is shown in figure 3.

The spacecraft 101 CM RCS was changed slightly from the Block I configuration. The components were rearranged somewhat on the component panels, and the forward-pitch engines were repositioned from a one-over-the-other to a side-by-side configuration. A schematic of the spacecraft 101 CM RCS is shown in figure 18; a CM RCS engine is shown in figure 10.



(a) Top view.

(b) View looking inboard, bays 3 and 6 (bays 2 and 5 mirror image).

Figure 16. - Location of the RCS and SPS components within the SM (Block II).

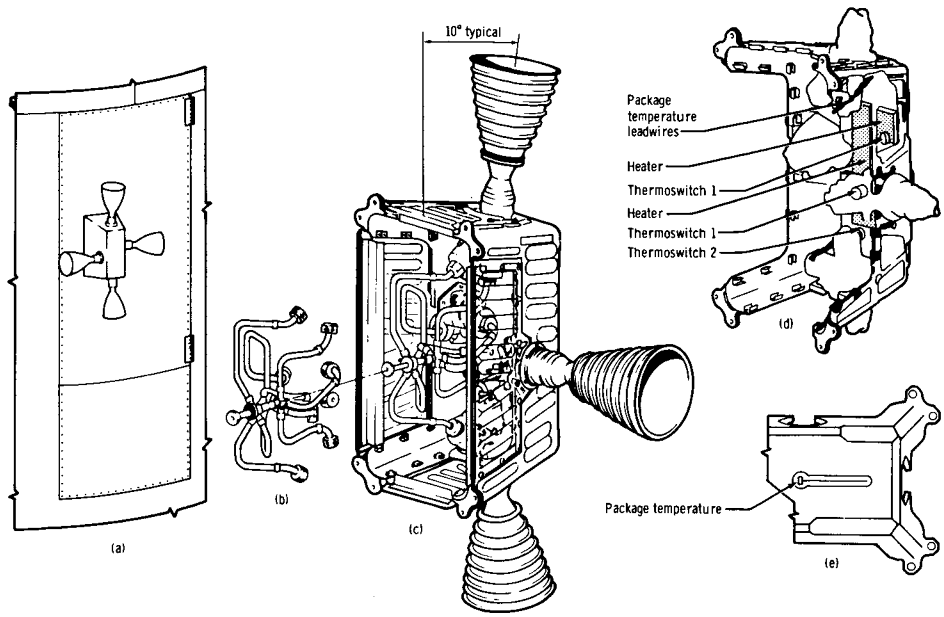


Figure 17. - The SM RCS quad engine package (Block II). (a) Quad; (b) Engine package propellant manifold; (c) Complete engine package assembly; (d) Engine mounting structure, inboard view; (e) Engine mounting structure, outboard view.

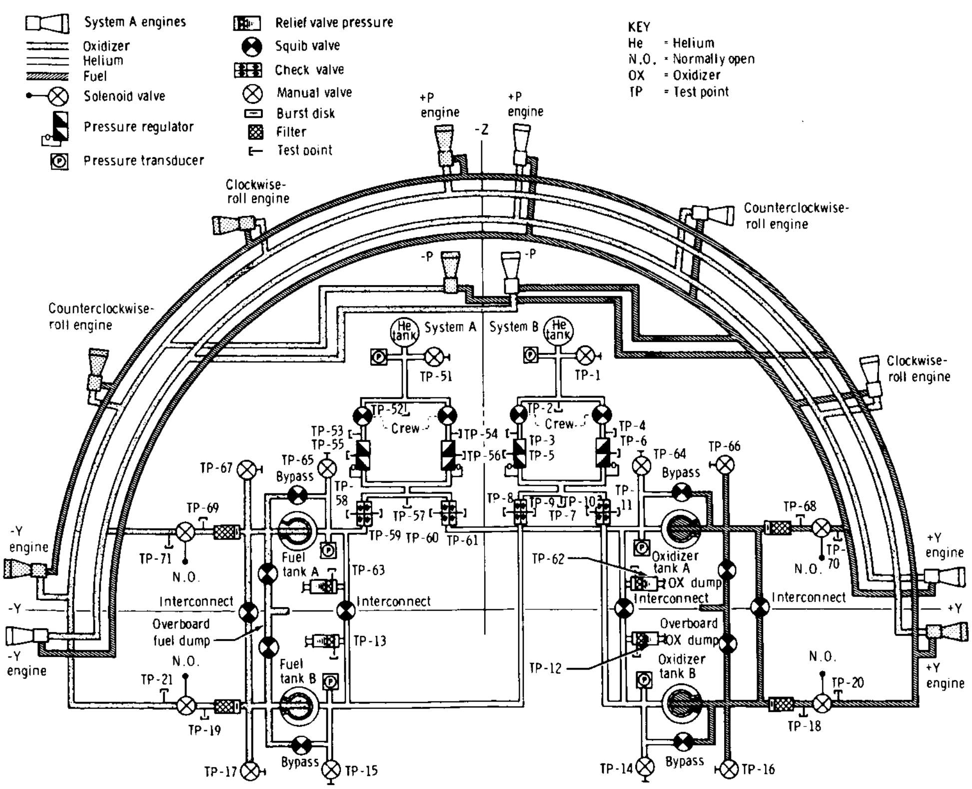


Figure 18. - Schematic of the CM RCS (Block II).

Performance summary. - All system operations were normal, except for the SM RCS quad B onboard P/T sensor, which failed before launch. All test objectives were met. Static firings of the four aft-firing SM RCS engines for 1 second were accomplished satisfactorily approximately 25 minutes before launch. The firings were audible to the crewmen. Evaluation of spacecraft body rates indicated normal performance of the SM RCS throughout the flight.

A total of 875 pounds of SM RCS propellant, of the 1342 pounds that were loaded, was used during the flight. Except for the S-IVB rendezvous, the propellant consumption approximated the predicted usage as adjusted by flight plan changes. The S-IVB rendezvous required approximately 11 percent (37 pounds) more propellant than was predicted.

Thermal control of the SM RCS was satisfactory throughout the flight. The primary heaters on all four quads were activated at earth orbit insertion, and the heaters remained active for the remainder of the mission.

No helium or propellant leakage from the CM RCS was noted before activation for the deorbit maneuver. The system was activated and the propellant isolation valves were opened soon after the deorbit maneuver. Both the manual and the automatic controls were used during entry in combinations of dual- and single-system firings. The systems functioned normally during entry.

A total of 50 pounds of CM RCS propellant was used, 29 pounds from system A and 21 pounds from system B. No propellant depletion burn was performed during the Apollo 7 mission. The CM landed with RCS propellants on board.

Anomalies. - Two anomalies were noted during and after the flight. First, the onboard P/T sensor for the SM RCS quad B failed before launch. The cause of this failure was unknown. Second, during postflight testing of the CM RCS relief valve, an excessive amount of leakage through the closed oxidizer isolation valves was observed. The oxidizer isolation valves opened when voltage was removed, and the position indicator switch so indicated. The valves were spring-loaded closed by means of a bellows preload and should have remained closed when voltage was removed.

Corrective action. - Because the SM was not recovered, the failed SM RCS quad B onboard P/T sensor was not available for investigation, and the cause of the failure remains unknown. Postflight data analysis did not aid in isolating the cause, and no corrective action could be taken. The failed CM RCS oxidizer valves were removed and sent to the contractor for postflight tests and analyses. A 0.05- to 0.06-inch compression of the bellows was noted. Evidently, the permanent compression of the two bellows and the resultant reduction in seat load and opening voltage were caused by a pressure surge when the CM helium system was activated at the time the isolation valves were in the "closed" position. An identical failure mode was noted in the early developmental phase of the valve; the intended procedure was to activate the system with the valves in the "open" position. The fact that the fuel valves did not fail can possibly be explained by the existence of a shorter and less dense fluid column between the isolation valve and the propellant tank. The fuel valves were not tested to determine whether degradation had occurred. The corrective action was to open the isolation valves before the system was activated on subsequent Apollo flights.

Apollo 8 Mission

The Apollo 8 (spacecraft 103) mission was the second manned Apollo mission, the first manned S-V mission, and the first manned lunar orbit mission. Launch was on December 21, 1968, from KSC, and the mission duration was approximately 147 hours. The spacecraft landed in the mid-Pacific recovery area on December 27, 1968. The crewmen were Frank Borman, James Lovell, and William Anders.

The purpose of the mission was to demonstrate the capability of the spacecraft, the crewmen, and the MSFN support facilities to conduct a deep space and lunar orbit mission. Also, the mission was designed to meet development and verification test objectives that were not met on previous missions.

The DTO's that were defined for the CSM RCS are as follows. In the space environment thermal control DTO, an attempt was made to demonstrate that the passive thermal control mode of operation was adequate to maintain spacecraft systems and components (including the CSM RCS) within acceptable thermal limits. In the midcourse correction capability DTO, an attempt was made to evaluate the PGNCS SPS/RCS guidance and control capability to make the required translunar and transearth midcourse corrections. Another DTO was to determine the effect, if any, of SM RCS engine firings on contamination of the CM windows. Also, the determination of SM RCS and CM RCS propellant use was included as a portion of several other test objectives.

Configuration of the CSM RCS. - The Apollo 8 SM RCS configuration was similar to the Apollo 7 configuration except for the following improvements. On the Apollo 8 SM RCS, a capability for electrically isolating individual engines was incorporated. The aluminized Mylar insulation on the SM RCS panels was replaced by multilayer blankets of aluminized H-film; on the Apollo 7 SM RCS, only the outer layer was H-film. The all H-film insulation further improved fire protection. Additional radiation shields were added to the Apollo 8 SM insulation system to prevent the SM RCS tanks from radiating heat to the cold areas that were internal to the SM.

The Apollo 8 CM RCS was similar to the Apollo 7 CM RCS, except that the abrasion pads in the propellant tanks were perforated to facilitate more rapid and complete venting of helium during the purge operation. Also, the amount of oxidizer that was offloaded was increased by 4 pounds per tank to reduce further the possibility of parachute damage during the propellant depletion burn after entry.

Performance summary. - The SM RCS and the CM RCS performed satisfactorily throughout the Apollo 8 mission. All system parameters were within the normal range, and no flight anomalies occurred. All test objectives were met. Static firing of the SM RCS engines on the pad was not performed as it was on the Apollo 7 mission. Evaluation of spacecraft body rates indicated normal performance of the SM RCS throughout the flight.

A total of 634 pounds, of the 1343 pounds of SM RCS propellant that were loaded, was used during the flight. The predicted propellant usage, adjusted for the flight plan changes, was 668 pounds. An estimated 46 000 firings of the SM RCS engines were made.

Thermal control of the SM RCS was satisfactory throughout the flight. The primary heaters on all four quads were activated soon after earth orbit insertion; the heaters remained active for the remainder of the mission.

No CM RCS helium or propellant leakage was noted before activation. The CM RCS systems were activated approximately 1 hour before CM/SM separation, and the systems were checked briefly by the crewmen.

Both manual and automatic controls were used during entry. Approximately 20 seconds after the CM/SM separation, the CM RCS system B was deactivated; the remainder of entry was performed by the use of system A only. An evaluation of the spacecraft body rate data indicated that system A functioned normally. After deployment of the main parachutes, the remaining CM RCS propellant was dumped, and the helium system blowdown and the propellant line purge were initiated. The propellant isolation valves were closed approximately 20 seconds before landing.

A total of 34.7 pounds of CM RCS propellant was used (34.1 pounds from system A and 0.6 pound from system B) before the propellant dump; 825 CM RCS engine firings were made.

Anomalies. - Several anomalies occurred during the Apollo 8 mission. All the anomalies were noted after the flight; these anomalies in no way compromised the mission. During postflight examination of the vehicle rate data, the rates that developed during SM RCS negative-pitch maneuvers were found to be approximately 20 percent less than they should have been. Further investigation indicated that the problem was caused by gas impingement on, and deflection by, the CSM umbilical which was located just forward of the engine.

During CM RCS postflight decontamination and testing, two other problems were noted. At the time of the CM flushing operation, large quantities of fluids leaked past the throat plugs in the engines. Also, the CM RCS system A helium check valve on the oxidizer side was leaking considerably more than was allowed by specifications.

Corrective action. - The quad C negative-pitch rate anomaly did not necessitate corrective action, other than to account for the reduced rate during preflight planning and postflight evaluation. Reexamination of the Apollo 7 data indicated that the reduced rate was present in quad C also but had not been detected.

No single cause was identified for flush fluid leakage around the CM engine throat plug. Conditions that contributed to the leakage included some throat erosion and improperly refurbished throat plugs. Material that dropped into the throat from drilling on the vehicle during postflight recovery activity compounded the problem. Furthermore, possible damage may have resulted from the improper removal of safety plugs that had been installed while the CM was on the recovery ship. The landing safing team director agreed to make notations in the recovery manual to prevent this type of leakage on future missions.

Check valve leakage was caused by deterioration of the nitroso rubber seat material. The nitroso rubber was chosen because of its compatibility with the oxidizer (N_2O_4). The check valves were subjected to an IPA flushing test as a part of the cleanliness verification program for the component. It was noted that an incompatibility existed between nitroso rubber being used and IPA, which resulted in valve seat stickiness, degradation, and leakage. The out-of-tolerance leakage that occurred during postflight testing was located in the spacecraft 103 CM RCS system A oxidizer check valve. The leakage was attributed to the cause just discussed. On subsequent vehicles, only check valves that had not been exposed to IPA were used.

Apollo 9 Mission

The Apollo 9 (spacecraft 104) mission was the third manned Apollo mission, the second manned S-V mission, and the first manned LM mission. Launch was on March 3, 1969, from KSC, and the mission duration was approximately 241 hours. The spacecraft landed in the Atlantic Ocean on March 13, 1969. James McDivitt, David Scott, and Russell Schweickart were the crewmen.

The primary purpose of the mission was to evaluate the LM systems performance and to perform selected CSM and LM operations in earth orbit. Various DTO's were defined for the CSM RCS. For example, the SM RCS propellant consumption was determined during the orbital navigation and landmark tracking. Data were obtained on the effects of the tower jettison motor, the S-II retrofire, and the SM RCS exhaust on the CSM. Performance of the LM passive thermal design that was exposed to natural and propulsion-induced environments was verified. Active docking of the CSM with the LM was conducted, and the SM RCS propellant consumption was determined for this maneuver. The LM/CSM undocking was accomplished by use of the SM RCS, the CSM accelerations were computed, and the SM RCS propellant consumption was determined. A pyrotechnic LM/CSM separation was performed, the CSM acceleration was computed, and the SM RCS propellant consumption was determined.

Configuration of the CSM RCS. - The only difference in the RCS of the Apollo 8 and Apollo 9 CSM was the addition of an isolation valve in the Apollo 9 helium line to the secondary fuel tanks on the SM quads. The purpose of this valve (normally closed) was to provide a means by which depletion of the primary fuel tank could be determined. Depletion would be evidenced by a decay in the fuel manifold pressure. When the fuel manifold pressure decayed, the new isolation valve would be opened, allowing the propellant pressure to return to normal.

Performance summary. - The SM RCS and the CM RCS performed satisfactorily throughout the mission. The only anomaly was the inadvertent closure of some SM RCS isolation valves during CSM/LM/S-IVB separation. The valves were opened later by the crewmen and remained open for the rest of the mission. All system parameters were normal during the mission, and all test objectives were met. The SM RCS engines were not static fired on the pad. An evaluation of spacecraft body rates indicated that RCS performance was normal throughout the flight.

A total of 790 pounds of SM RCS propellant was used during the mission. The predicted usage, corrected for flight plan changes, was 598 pounds. Most of the discrepancy between the actual and the predicted propellant consumption occurred while the quad C propellant isolation valves were closed and during the undocked LM-active period. The secondary fuel tank helium isolation valves on all quads were opened before CM/SM separation, although no primary tank fuel depletion had been indicated. A total of 57 000 firings was estimated for the 16 SM RCS engines. Thermal control of the SM RCS was satisfactory throughout the flight.

Both manual control and automatic control of the CM RCS were used during entry. Approximately 12 seconds after CM/SM separation, system B was deactivated; the remainder of entry was performed by the use of system A only. An evaluation of the spacecraft body rates indicated normal CM RCS performance.

A total of 27.5 pounds of CM RCS propellant was used for entry. The remaining 217.5 pounds were burned during the depletion burn after main parachute deployment.

Anomalies. - Two minor instrumentation anomalies occurred during the Apollo 9 mission. The first anomaly occurred during the preflight phase shortly after servicing, and was determined to be leakage around the seal of the quad C high-pressure helium manifold pressure transducer. The second anomaly was evidenced by the periodic 250-psi increases in the reading of the quad B helium tank pressure transducer. The reading returned to normal after the increase.

Another, more disturbing anomaly was the inadvertent closure of some of the propellant isolation valves. After CSM/S-IVB separation, the crewmen reported a control problem that had lasted for approximately 12 minutes during the transposition period. The crewmen first noticed a lack of capability for translation to the left. The position indicator flags for the quad C primary and secondary propellant isolation valves and the quad D secondary valves were in the "closed" position. The valves were reopened; thereafter, the system performed normally. The valves that had been opened during final checks before launch were verified "open" during orbit insertion checks by the crewmen, and also were verified "open" during a cursory examination of the panel after the commander and the command module pilot exchanged seats before SLA separation.

The propellant isolation valve closure caused triggering of the caution and warning switch for the quad A engine package temperature. The high temperature on quad A occurred because the left translation was being requested, and quad C, with its closed isolation valves, was unable to provide impulse. A counterclockwise roll resulted and caused the PGNCS to correct by firing the clockwise-roll engine. Opposing roll thrusters on quad A fired, which caused the temperature to increase. Although the caution and warning indication was triggered at 206° F, the 210° F redline was not exceeded.

Corrective action. - Of the two minor instrumentation anomalies, the first, which involved leakage at the transducer, was resolved by replacement of the transducer seal. The correction was accomplished during a built-in hold in the countdown and was done without deservicing. The second anomaly, which involved the periodic 250-psi increases in quad B helium tank pressure transducer readings, was never explained, and no corrective action was taken.

Extensive investigative effort was expended in an attempt to explain and correct the isolation valve closure problem. The isolation valves were latched open magnetically and were spring-loaded to the "closed" position. The valves were controlled by means of switches on panel 2 that were spring-loaded to the center "off" position. The four isolation valves in each quad were controlled by one switch.

Propellant usage data indicated that all four quad C valves were closed, rendering the quad inoperative, and that quad D was performing normally before the crewmen reopened the propellant isolation valves. Propellant could have been supplied from both primary tanks and one secondary tank from quad D, because only the valve position indicator for the secondary tank was in the "closed" position. The closure of only one of the secondary valves was sufficient to cause the indication. It was suspected that the valve closure was caused by mechanical shock at CSM/SLA separation.

Shock tests were performed later on several isolation valves and on an assembled quad. These tests were conducted to determine the shock load that was needed to close the valves, and also to determine the effect of the shock loads that were encountered during the CSM/SLA separation sequence. The results of the individual valve tests indicated that 80g (with an onset rate of approximately 11 milliseconds) to 140g (with an onset rate of approximately 1 millisecond) could cause a normal valve to close. The shock to the valve that resulted from the pyrotechnic charges which were used to separate the CSM from the SLA was estimated to be between 180g and 260g (with an onset rate of between 0.2 and 3.0 milliseconds). The Apollo 7 and 8 spacecraft, which had the same configuration, did not have the problem.

The results of the investigations indicated that the shock at CSM/SLA separation could have caused the primary and secondary propellant valves to close and that the valve closures were not detrimental to the valves. Because the hardware was shown not to be affected adversely, flight procedures were modified to verify the isolation valve position after exposure to shock environments. No additional corrective action was taken. Because the caution and warning anomaly was caused by the isolation valve closure, no further action was necessary.

Apollo 10 Mission

The Apollo 10 mission (spacecraft 106) was the fourth manned Apollo mission, the third manned S-V mission, and the second manned LM mission. Launch was on May 18, 1969, at KSC, and the mission duration was approximately 192 hours. The spacecraft landed in the Pacific Ocean on May 26, 1969. The crewmen were Thomas Stafford, John Young, and Eugene Cernan.

The primary purpose of the Apollo 10 mission was to evaluate the crewmen, spacecraft, and mission support facilities performance during a manned LM lunar mission. Also, LM performance was to be evaluated in the lunar environment. The DTO's that were defined for the CSM RCS were as follows.

1. During midcourse navigation, to determine SM RCS propellant consumption

2. During midcourse corrections, to determine the effect of SM RCS performance and midcourse corrections

3. During lunar landmark tracking, to determine the amount of SM RCS propellant required during docked operations

4. During lunar landmark tracking, to determine SM RCS propellant consumption for undocked operations

Configuration of the CSM RCS. - The only difference between the Apollo 10 and Apollo 9 RCS was a change in the Apollo 10 SM RCS secondary-heater thermostat switching limits. The switching temperature was changed from a range of 77° to 104° F to a range of 120° to 129° F. The Apollo 10 CM RCS was not changed from the Apollo 9 configuration.

Performance summary. - The SM RCS and CM RCS performed satisfactorily throughout the Apollo 10 mission. All system parameters were normal during the mission and all objectives were met. An evaluation of spacecraft body rates indicated normal SM RCS performance throughout the mission. A total of 580 pounds of SM RCS propellant was used; the predicted consumption was 850 pounds. A total of 43 000 firings was estimated for the 16 SM RCS engines. Thermal control of the SM RCS was satisfactory throughout the flight. The primary heaters on all quads were activated soon after earth orbit insertion, and the heaters remained active for the rest of the mission.

Both the manual and the automatic controls were used during entry. Approximately 9 seconds after CM/SM separation, system B was deactivated, and the remainder of entry was performed by the use of system A only. An evaluation of the spacecraft body rates indicated normal CM RCS performance.

A total of 32.5 pounds of CM RCS propellant was used for entry. The remaining 212.4 pounds were burned during the depletion burn after main parachute deployment.

Anomalies. - The first anomaly, which occurred approximately 3.5 days before launch, was a decrease of 0.14 psi/hr in the CM system A helium manifold pressure. The pressure distribution indicated that the leak was downstream from the check valves in the fuel leg. The size of the propellant leak which would be needed to produce the established pressure decay and the absence of propellant vapors in the area were evidence that the leak was between the check valves and the fuel manifold of the helium system. A mass spectrometer leak check of the entire system (both preflight and postflight) was unproductive. The system was repressurized to 49 psia approximately 31 hours before launch. The leak rate appeared to diminish during the mission, and no adverse effect on the operation of the system was noted during entry.

The second anomaly occurred when the CM RCS propellant isolation valves were opened approximately 10 hours before launch. The system B helium manifold pressure dropped from 44 to 37 psia. Calculations showed that a pressure drop of this magnitude would be expected if the oxidizer burst disk were ruptured, allowing oxidizer to flow from the tank into the oxidizer manifold when the isolation valves were opened.

Corrective action. - The CM RCS was subjected to extensive preflight and postflight leak checks. Postflight leak checks were conducted at 50 and 285 psia, but no leak could be found. It was recommended that leak tests at 100 psia be conducted approximately 30 days before launch on future missions. This procedural change was designed to ensure that leaks could be found and repaired before the start of the countdown.

After calculations had indicated that the system B pressure drop was of the magnitude which would be expected if the oxidizer burst disk were ruptured, the decision was made to launch without further investigation and to vent the oxidizer from the lines through the engine after earth orbit insertion.

Apollo 11 Mission

The Apollo 11 mission (spacecraft 107) was the fifth manned Apollo mission, the fourth manned S-V mission, the third manned LM mission, and the first manned lunar landing mission. Launch was on July 16, 1969, from KSC, and the mission duration was approximately 195 hours. The spacecraft landed in the Pacific Ocean on July 24, 1969. The crewmen were Neil Armstrong, Michael Collins, and Edwin Aldrin.

The primary purpose of this mission was to perform a manned lunar landing and return to earth safely. No DTO requirements were involved for the CSM RCS on the mission.

Configuration of the CSM RCS. - No changes were made in either the CM or SM RCS configurations for the Apollo 11 mission. The configuration was that which was used on the Apollo 10 mission.

Performance summary. - The SM RCS and CM RCS performed satisfactorily throughout the mission. The two anomalies that occurred were an inadvertent isolation valve closure during CSM/S-IVB/LM separation and the failure of a CM thruster to respond to automatic commands. The isolation valves were reopened later by the crewmen (according to established procedures) and remained open for the rest of the mission. The cause of the closure was determined to be similar to that of the closure on the Apollo 9 mission. (The shock loads were generated during separation.) All system parameters were normal during the mission; all mission requirements were met. An evaluation of spacecraft body rates indicated normal RCS performance throughout the flight. A total of 560 pounds of SM RCS propellant was used during the mission. The predicted propellant usage was 590 pounds.

Thermal control of the SM RCS was nominal throughout the flight. The primary heaters on all quads were activated soon after earth orbit insertion, and the heaters were active for the remainder of the mission.

Both manual and automatic controls of the CM RCS were used during entry. Approximately 65 seconds after CM/SM separation, system B was deactivated; the remainder of entry was performed by the use of system A only. An evaluation of the spacecraft body rates indicated normal CM RCS performance with the exception of the negative-yaw (-Y) engine. The negative-yaw engine did not respond to automatic commands, but the engine performed normally with manual or direct coil commands.

A total of 41 pounds of CM RCS propellant was used during entry. The remaining 205 pounds were burned during the depletion burn after main parachute deployment.

Anomalies. - As was mentioned in the performance summary, the two anomalies that occurred during the mission were the inadvertent closure of the propellant isolation valve and the failure of one of the CM engines to respond to automatic commands.

Corrective action. - The corrective action for the isolation valve problem was established after a similar occurrence on the Apollo 9 mission. Because this action (which involved cycling the valves to the "open" position) was adequate on both the Apollo 9 and Apollo 11 missions, no further hardware or procedural changes were prescribed.

The malfunction of the CM negative-yaw engine in the automatic mode was determined during postflight hardware examination and data review. Electrical continuity through at least one of the parallel automatic coils in the engine was evidenced by the fact that the SCS driver signals were normal. The normal SCS driver signals and the existence of at least some detectable thrust indicated that one of the two valves was working normally.

During checkout at the launch site, another engine failed to respond to commands during the valve signature tests. A faulty terminal board connector was the cause of the problem. The systems operated nominally after the terminal board was replaced.

The postflight tests indicated that two pins in the terminal board were loose and caused intermittent continuity to the automatic coils of the engine valve. This type of failure had been noted on terminal boards that were manufactured before November 1967. The faulty board was manufactured in 1966. A design change that was made to the boards that were manufactured after 1967 precluded failures of this type. Only the new boards were to be used on future spacecraft. (The anomalous circuit boards were not a part of the CSM RCS; therefore, the boards are not discussed in detail in this report.)

CONCLUSIONS AND RECOMMENDATIONS

Development of the command and service module reaction control systems presented some unique and challenging engineering problems. The problems were adequately treated and good solutions developed, as evidenced by the essentially trouble-free operation of the reaction control system during the Apollo missions. However, in retrospect, certain problem areas were common to many of the component development efforts. The recommendations to minimize the impact of the problems on future programs are discussed as follows.

The initial component function design specifications often were more stringent than was necessary because actual requirements were not known. In some cases, the specification requirements were the projected limits of the state of the art at the anticipated times of use. As the requirements were defined more fully, there was hesitancy to relax the specification, which might have resulted in some unnecessary, and perhaps unfruitful, efforts. It is recommended that an intensive effort be expended to define requirements accurately as early as possible. Also, as a relaxation in requirements becomes evident, it is recommended that the specification be relaxed if cost or schedule savings can be realized.

A lack of compatibility of the system and components with the propellants was a recognized problem early in the Apollo Program. The major deterrent to efficient resolution of the problem was the unavailability of elastomeric materials that were compatible with propellants under long-duration exposure. A problem that was not recognized until considerably later in the program involved the incompatibility of the system and components with the flush fluids (or combinations of flush fluids) and propellants. It is recommended that, at such time as material compatibility of the system and components with fluids is established, all fluids and mixtures of fluids that might be introduced into the system be included. Additionally, it is recommended that particular attention be given to the determination of the specific fluids that might be used during manufacturing and checkout of the system and components when the materials are selected. Provisions for adequate drying of systems should be made and verified if fluid mixing cannot be tolerated.

Because of the many small orifices and the close tolerances on moving parts, the problem of cleanliness control increased as the Apollo Program progressed. It was difficult to assemble a clean system, and the need for component removal and replacement further increased the problem. To minimize the problem, filters were added to the system to protect components that had an unusually high failure rate because of contamination. It is recommended that on future programs, all components should be designed to be as insensitive to contamination as possible. Additionally, it is recommended that such components be protected by integral filters. A further recommendation is that, if fluids are reverse flowed through any component during a flushing or filling operation, both inlet and outlet ports on the component should be protected against contamination. If large quantities of contaminants are anticipated, mainstream filters at the fluid source should be considered for better protection in addition to the integral component filters.

A considerable number of unnecessary and costly situations occurred during the development and qualification tests because the production of components was well underway before the test programs were completed, particularly during the system-level tests. Corrective action for problems that existed during these programs almost always involved the retrofit of production units and the modification of completed systems. Some problems were tolerated because of the extensive vehicle rework that would be required for corrective measures. These shortcomings were compensated for by either tolerating higher rejection rates or modifying operating procedures. Only limited changes were made to the systems as a result of these late tests. Consequently, the test results did little for the development of more reliable systems, but rather were useful in instilling confidence in equipment or defining operating constraints. A further recommendation, therefore, is that extensive efforts be made to integrate the test program schedules with the master production schedules. Specifically, the overall schedule should be adjusted to provide time to implement the production hardware changes dictated by the test program.

The experience gained during the Apollo Program identified the significant value (in terms of total resources and schedules) of separately managed, selective supporting development and test evaluation tasks.

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