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APOLLO EXPERIENCE REPORT - COMMAND AND SERVICE MODULE ENVIRONMENTAL CONTROL SYSTEM

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16. Abstract This paper presents a comprehensive review of the design philosophy of the Apollo environmental control system, and the development history of the total system and of selected components within the system. In particular, discussions are presented relative to the development history and to the problems associated with the equipment cooling coldplates, the evaporator and its electronic control system, and the space radiator system used for rejection of the spacecraft thermal loads. Apollo flight experience and operational difficulties associated with the spacecraft water system and the waste management system are discussed in detail to provide definition of the problem and the corrective action taken when applicable.					
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SUMMARY

The Apollo environmental control system was designed and qualified to support three crewmen for 14 days and to maintain electronic equipment within operating thermal boundaries. The system maintains the pressure atmosphere of 100 percent oxygen and removes trace contaminants and metabolic carbon dioxide by absorption in charcoal and lithium hydroxide beds. Temperature control is provided by heat rejection from radiators and a water evaporator. Oxygen is supplied by the cryogenic storage system, and water is supplied as a byproduct of the fuel cells. The knowledge gained from extensive ground testing and inflight experiments on the behavior of water in zero gravity led to the incorporation of a wick-type porous-plate condensate separator.

The two hardware items requiring the most extensive development were the water evaporator and the radiator. During the Apollo Program, continuous refinements have been required in the construction, material selection, and quality control of the evaporator and its control system. The wide range of the maximum and minimum heat loads led to the use of a selective stagnation radiator designed to employ the viscosity characteristics of the coolant fluid (ethylene glycol and water). The other major problems experienced were in materials selection to reduce corrosion (particularly in the coolant system), materials selection for fabrication of porous plates and heat exchangers, and material and process refinements to eliminate weld crazing.

INTRODUCTION

During an Apollo mission, the environmental control system (ECS) of the command and service module (CSM) provides life support for the flight crew and thermal control for the vehicle electronics systems. During the major portion of the mission, the CSM serves as the living quarters for all three crewmen. The components of the ECS are located in both the command module (CM) and the service module (SM).

This report presents a discussion of the functional and physical ECS design requirements based on the integration of the ECS with the CSM and the mission parameters. A description of systems operations and of some significant problems encountered during development and flight testing of the hardware is also included.

DESIGN CONSIDERATIONS

Requirements

The initial ECS design requirements were in the form of general guidelines, which later were refined for contractual purposes. A 5.0-psia pure oxygen atmosphere was required in the pressurized CM, and a shirtsleeve environment for the crewmembers was established as the normal mode of operation. Pressure suits were to be worn only during critical mission phases, such as launch and entry. In the event of an emergency cabin depressurization, the system should maintain the cabin-pressure level within acceptable limits for a time period sufficient for the crewmembers to don their pressure suits. To provide this capability, statistical data were used to establish a design criterion for maintenance of cabin pressure above 3.5 psia for 5 minutes after a 1/2-inch-diameter puncture in the pressure vessel. Should loss of cabin pressurization occur, life support in the pressure-suit mode would be provided for a sufficient duration to permit the safe return of the crewmembers.

The design requirements limited the carbon dioxide partial pressure to 7.6 torr and specified that the carbon dioxide removal technique was to be chemical absorption by lithium hydroxide. In the early design stages, a regenerable molecular sieve had been considered but was rejected for the more reliable lithium hydroxide absorption process. The CM gas temperature was to be maintained at $75^{\circ} \pm 5^{\circ}$ F, except during entry when 100° F maximum was permissible. The cabin relative humidity was limited to 40 to 70 percent. In addition to the atmospheric requirements, the ECS was to provide thermal control for electrical equipment, and critical equipment was not to be dependent upon the cabin atmosphere for cooling.

The conversion of these design requirements into operational hardware involved several additional factors associated with the interface definition of the system with the spacecraft. The more significant considerations were weight and volume limitations, power requirements, and reliability. In addition, the design approach was frequently dictated by the environments in which the equipment would operate.

Weight and Volume

The free gas volume of the CM is approximately 320 cubic feet as compared with the 80-cubic-foot volume of the Gemini spacecraft. Judicious management of system volumes in the design phase of the CM was necessary to provide the free volume required for the crew functions unique to the Apollo missions, such as crew transfer to the lunar module (LM) in a pressurized environment. The volume occupied by the ECS, including stored expendables such as the lithium hydroxide canisters, is approximately 20 cubic feet. Negligible system volume growth has occurred during the program.

Equipment weights have proved to be more difficult to predict than the equipment volume. Experience with the earlier manned spacecraft has demonstrated an increase in weight with time, as estimated numbers were replaced by weights calculated from detail drawings which, in turn, were revised as actual hardware weights became available. Three points in the weight history of the CSM ECS are presented in table I.

TABLE I. - ENVIRONMENTAL CONTROL SYSTEM WEIGHT SUMMARY

Date	Weight of ECS		
	CM, lb	SM, lb	Total, lb
January 1965	442	105	547
July 1966	528	159	687
July 1969	553	221	774

A significant portion of the weight-increase trend is associated with the design changes, which usually resulted from difficulties encountered during the development of the ECS. Another source of weight increase is the modification of ECS requirements as a result of experience with the system, the spacecraft, and the mission itself. Also, system requirements may shift as more refined definitions of the interfaces with other systems become available. When configuration control to the component level is established, then the weight increases are minimized and generally occur, if at all, as the result of failures during the qualification test program.

Power

The requirement for electrical power is one basic element which is optimized in the trade-off studies conducted during the preliminary design of the spacecraft. The Apollo spacecraft uses three fuel cells as the primary sources of power, with batteries providing a supplemental source during peak-load periods. A summary of the electrical power requirements of the ECS is given in table II. This load is constant and independent of the mission phase.

TABLE II. - ENVIRONMENTAL CONTROL SYSTEM POWER SUMMARY

Components	Power (ac), W (a)	Power (dc), W
Suit compressor	85	--
Pump	52	--
Cabin fan (2)	38	--
Controls	22	33
Instrumentation	—	<u>22</u>
Total	197	55

^aThe values for ac power are as supplied to the rotating machinery. All ac power (115/200 volts, 3 phases, 400-cycle) for the spacecraft is supplied from three central static inverters which energize two independent buses.

Reliability

Reliability was another prime consideration in the design of the ECS. Therefore, the keynote of the ECS design is redundancy. The system must operate continuously throughout the mission, and although inflight maintenance was considered, it did not appear practical. Instead, redundant features were used whenever possible, and in other situations, completely independent manual-override capability was provided. Test experience in all phases of the program, from design-feasibility testing through flight performance evaluation, has been used to increase the confidence in the reliability of the system. In addition, reliability data collected from comparable hardware and design techniques used in aircraft and other spacecraft systems contribute to the confidence level. These data are used in conjunction with reliability logic diagrams and failure modes and effects analyses to gain further confidence that the system design meets its reliability goals.

Environment

The environment in which the system must operate frequently dictates the design approach. The 5-psia pure oxygen atmosphere eliminates consideration of many standard parts and materials because of the hazard of combustion. Other versatile materials are rejected because they outgas at reduced pressures. This outgassing results in toxic-contaminant buildup in the sealed environment of the CM cabin. (Except for the 0.2-lb/hr maximum external leakage, the CM is completely sealed.) The use of the ethylene glycol and water coolant presented a toxicological problem. Although the effects of acute ingestion were known, the toxicity resulting from continuous inhalation of gaseous or aerosolized glycol resulting from a leak was unknown. To establish acceptable levels and a means for detection of glycol in the atmosphere, a series of tests was performed on eight different mammalian species, ranging from the mouse to the chimpanzee. Based upon the results of the animal testing, human volunteers were exposed to glycol aerosols and vapors for periods up to 14 days. It was found that levels which did not produce subjective responses of a sweet odor or pharyngeal irritation would not produce physiological changes in humans in 14 days. Once the sense threshold was passed, however, prolonged exposure was irritating and subjectively intolerable. In the event the threshold were to be exceeded, it would be necessary to isolate the crewmembers from the atmosphere by use of the emergency oxygen masks or the intravehicular space suits.

The vibration and acceleration loads encountered during launch and entry contribute to the complexity of the design. By themselves, these dynamic forces usually are not difficult to overcome in the design of the individual components, but frequently, the forces are amplified by the manner of equipment packaging or by secondary supporting structures in the spacecraft.

ENVIRONMENTAL CONTROL SYSTEM DESCRIPTION

The following description of the ECS is presented to provide orientation for the more specific developmental problems to be discussed later. The schematic diagram of the ECS (fig. 1) may be conveniently divided into the oxygen, water, coolant, pressure-suit, and cabin circuits. For orientation within the spacecraft, refer to figures 2, 3, and 4 which show the spacecraft installation of ECS equipment.

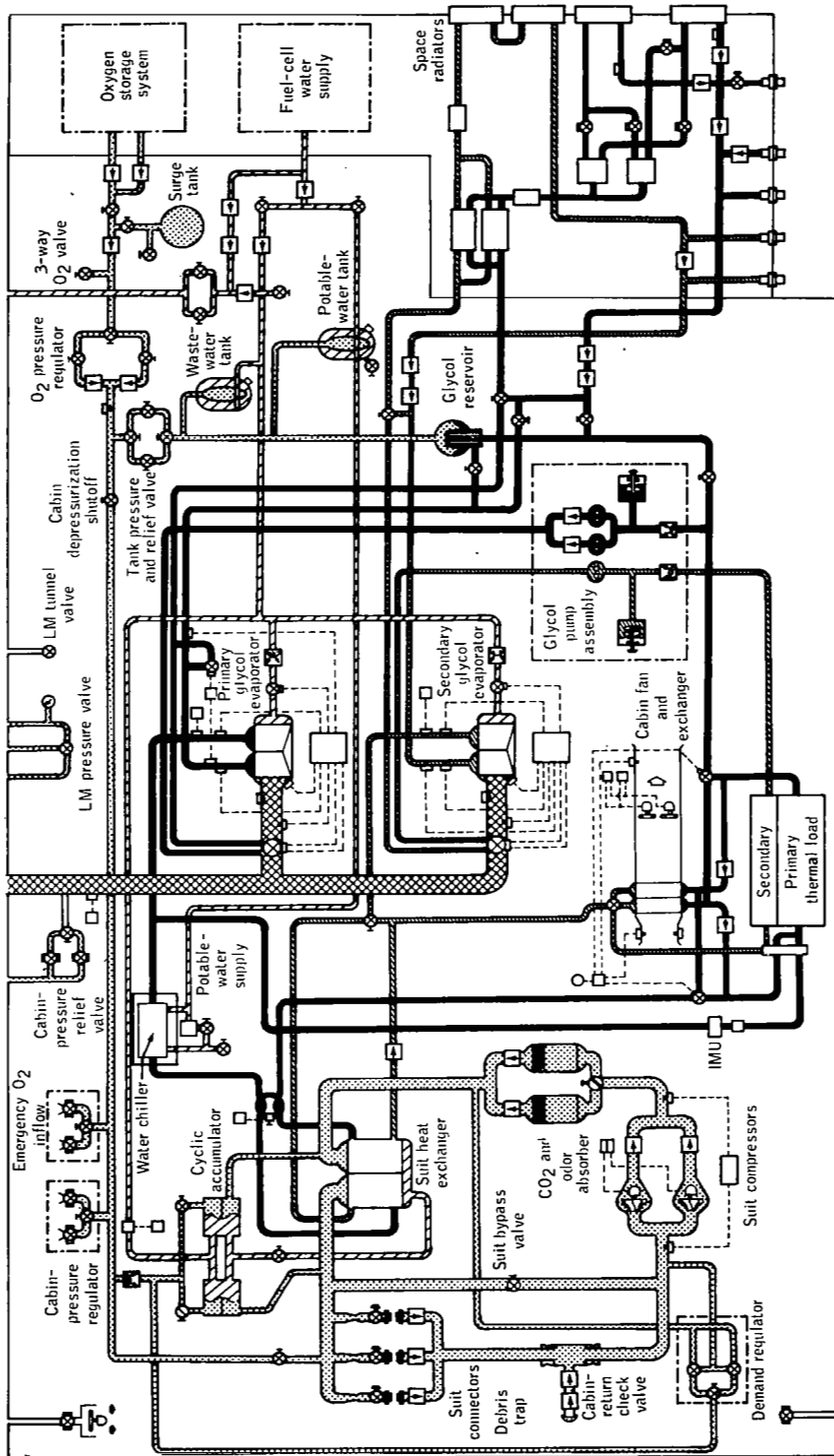


Figure 1. - Apollo environmental control system.

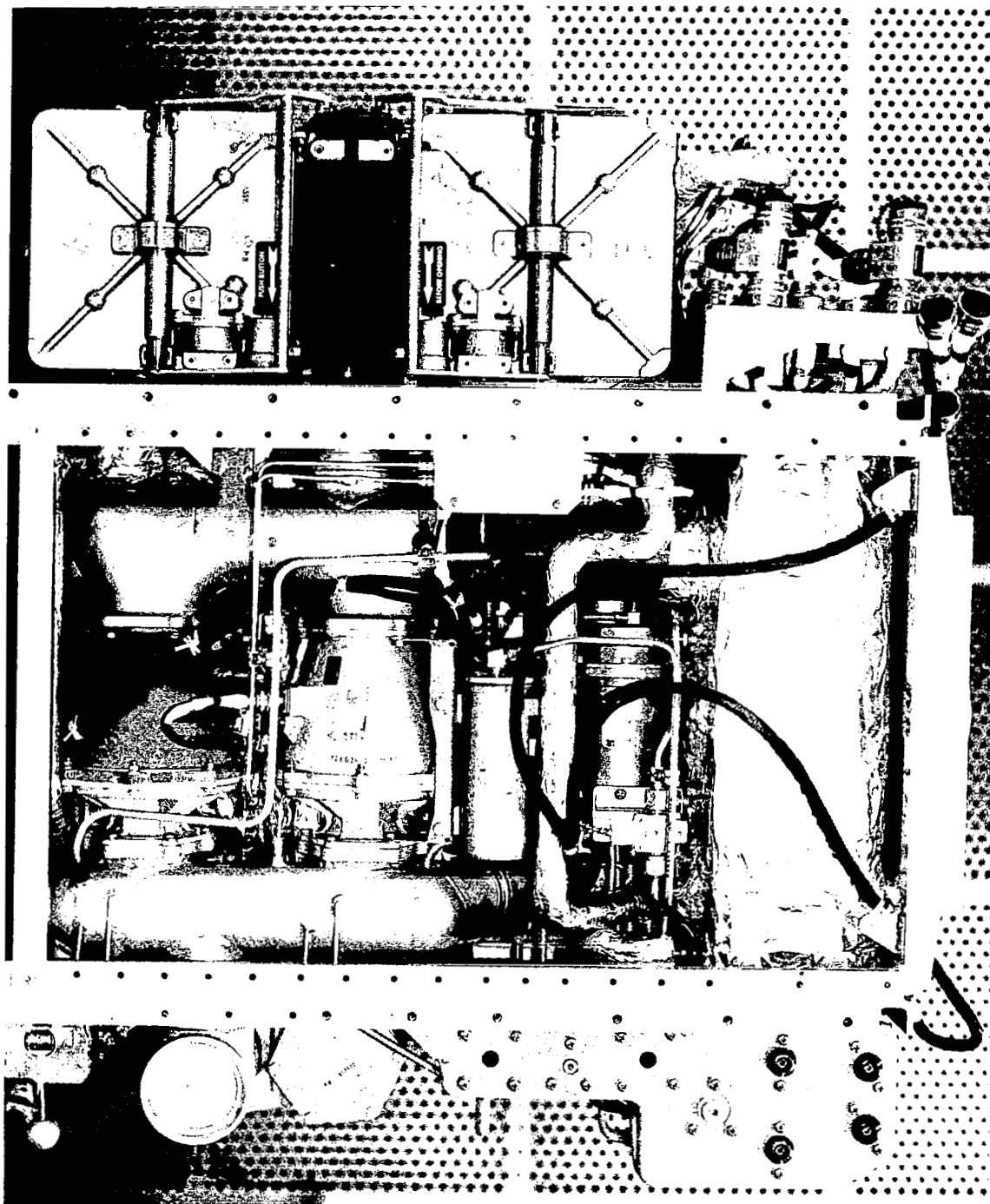


Figure 2. - Cooling system configuration.

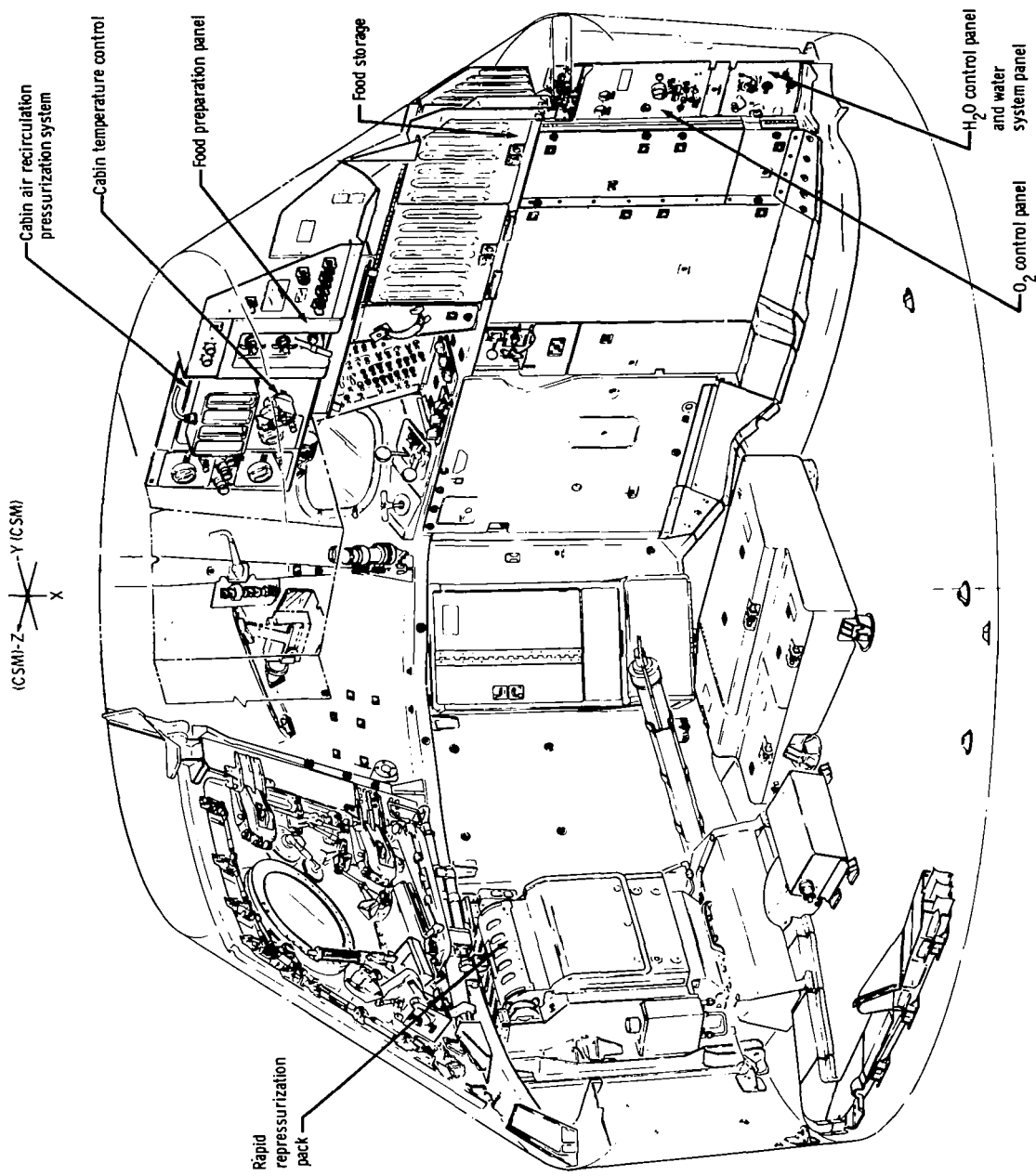


Figure 3. - Environmental control system equipment installation in the left-hand equipment bay and crew hatch area of the central crew compartment.

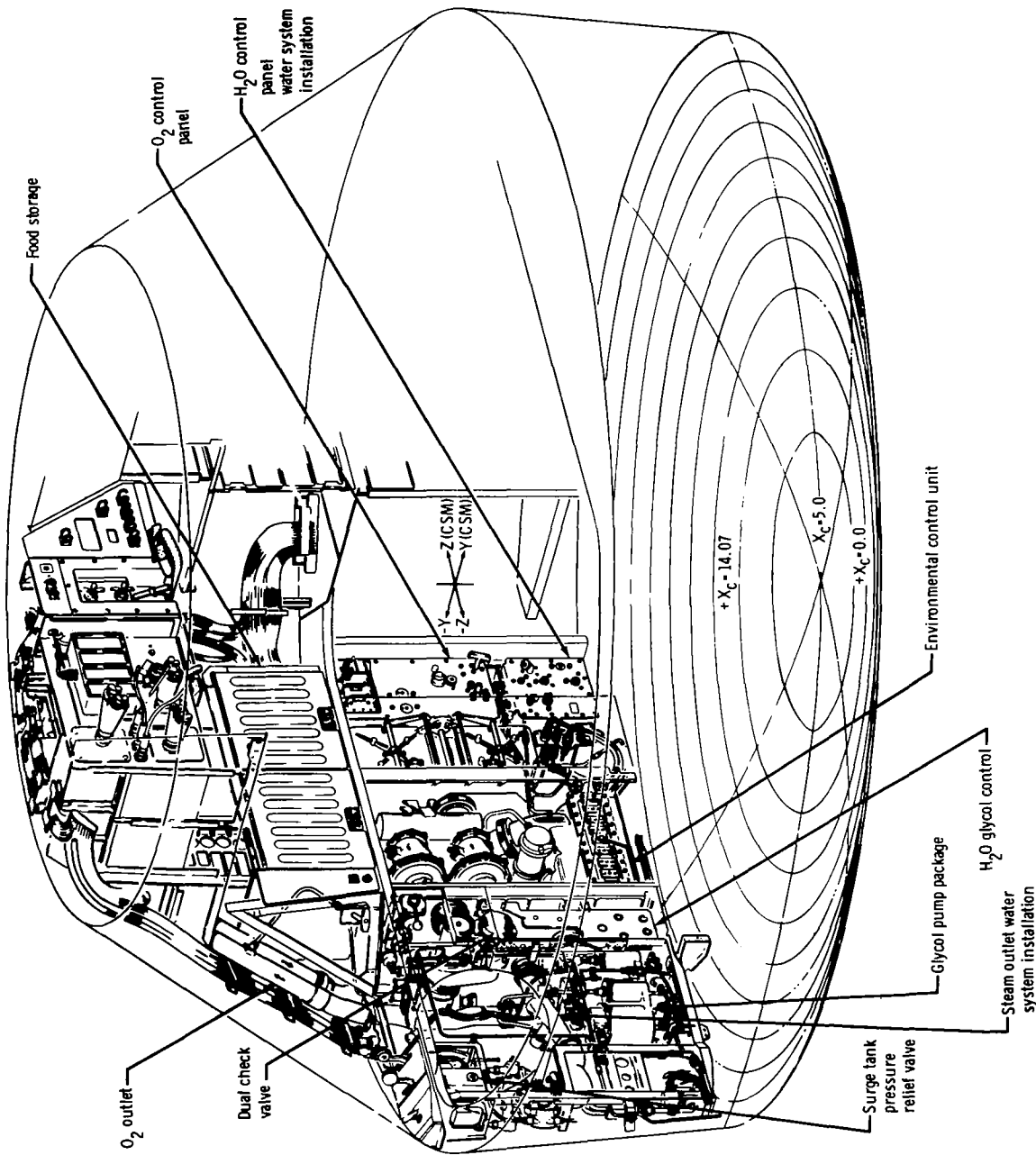


Figure 4. - Environmental control system equipment installation in the left-hand equipment bay of the central crew compartment. The close-out panels are omitted for clarity.

The primary oxygen source consists of two supercritical cryogenic tanks located in the SM. These tanks also supply the oxygen requirements of the fuel cells and are generally considered as part of the electrical power system. The two tanks contain a total of 640 pounds of oxygen, and the design specification allocates 172.6 pounds of this amount to the ECS (table III). In comparison, the actual oxygen allocation to the ECS for the Apollo 11 mission was 72.4 pounds for planned use, 10.4 pounds for LM support, and 15.6 pounds for contingency use. The reduced consumption during the Apollo 11 mission resulted because the mission duration (196 hours) was less than that of the specification mission (336 hours), and the cabin leakage and crew metabolic requirement values were lower than design specification requirements (table IV).

TABLE III. - ENVIRONMENTAL CONTROL SYSTEM
OXYGEN — SPECIFICATION REQUIREMENTS

Allocation	Oxygen, lb
Crew consumption (1.8 lb/man-day for 14 days)	75.6
Cabin leakage (0.2 lb/hr for 336 hours)	67.2
Cabin repressurizations	11.7
One CM puncture	3.6
LM support	<u>14.5</u>
Total	172.6

TABLE IV. - ACTUAL ENVIRONMENTAL CONTROL SYSTEM
OXYGEN CONSUMPTION

Apollo mission	Duration, hr	Quantity of oxygen consumed, lb
7	259.7	102
8	146.5	51
9	240.5	99
10	190.0	71
11	196.0	82

Oxygen delivery to the ECS at 900 ± 35 psia is limited to 4.5 lb/hr from each tank by the combination flow restrictors and heat exchangers, which ensure a minimum temperature of 0° F at the maximum flow rate. Short-duration demands in excess of this capability are satisfied by a surge tank (3.7 pounds at 900 psia) and a repressurization pack (3 pounds at 900 psia). Either or both supplies may be isolated to preserve the oxygen for entry or other use. The main oxygen pressure regulator, which supplies the components and functions as indicated in table V, reduces the circuit pressure to 100 ± 10 psig.

TABLE V. - ENVIRONMENTAL CONTROL SYSTEM OXYGEN-COMPONENT SUMMARY

Component	Functions	Control range	Features		Override
			Redundancy	Selectable	
Cabin-pressure regulator	Automatic supply makeup for cabin leakage or metabolic consumption	5.0 ± 0.2 psia, flow rates to 0.7 lb/hr each	2	No	No
	Manual repressurization	6.0 lb/hr	No	Yes	Yes
Emergency inflow valve	Automatically initiate high flow mode	0.67 lb/min at cabin pressure of 4.4 ± 0.2 psia	Yes	Yes	Yes
Demand pressure regulator	Supplies crew metabolic requirements in suited mode	Supplies when suit circuit is 3.0 ± 0.5 inches of water below cabin pressure; relieves when 2 inches of water above cabin pressure	2	Yes	Yes
	Maintains suit-circuit pressure during cabin repressurization	Controls to 3.75 ± 0.25 psia when cabin pressure is below 3.5 psia; flow rates to 0.67 lb/min	2	Yes	Yes
	Manual suit-circuit integrity-check capability	Pressurizes suit-circuit to >4.0 psig and depressurizes at controlled rate	No	Yes	Yes
Cyclic accumulator	Removes metabolic water from water separator in suit circuit heat exchanger and transfers to waste-water system	Actuates automatically every 10 minutes; capacity of 130 cc water per actuation	2	Yes	Yes
Tank-pressure control and relief valve	Maintains pressure on bladders in potable and waste-water tanks and in glycol reservoir	Controls to 20 ± 2 psig relative to cabin pressure	2	Yes	Yes
	Relieves pressure on increasing quantity	Relieves when fluid pressure increases to 25 ± 2 psig	2	Yes	Yes

The primary source of water for the ECS is the fuel cells, which produce approximately 0.77 lb/kWh as a byproduct of fuel-cell operation. The water storage provisions consist of a 36-pound-capacity potable water tank (fig. 5), and a 56-pound-capacity waste-water tank. Excess moisture in the cabin or suit circuit gas is removed by the water separator in the suit heat exchanger and is transferred by the cyclic accumulator to the waste-water tank for subsequent use as an expendable coolant. The effluent from the fuel cells is directed to the potable-water tank and is used for drinking and food reconstitution. Periodic injection of chlorine by the crewmembers maintains bacteria control in the potable-water system. When the potable-water tank is full, the water circuit automatically diverts the fuel-cell output to the waste tank by elevating the water-system pressure from 25 to 30 psia. When both tanks are full, the water-system pressure is increased to 40 psia, and the fuel-cell effluent is dumped directly overboard. Excess water may also be dumped manually, and this capability has been used in all missions. This manual operation was chosen to preclude interference with photography, sightings with the guidance and navigation equipment, and trajectory determination.

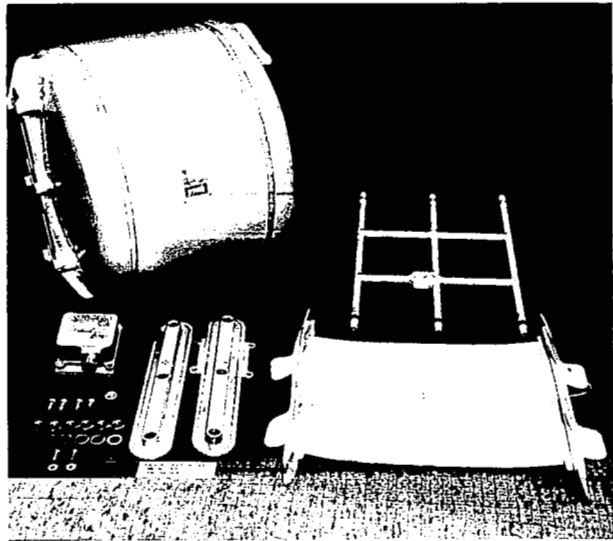


Figure 5. - Potable water tank disassembled with the bladder, support frame, and quantity transducer.

The coolant system consists of a primary loop, which is operated continuously, and a secondary loop, which serves as a backup system. The primary loop uses a centrifugal pump to circulate 200 lb/hr of coolant (ethylene glycol and water) through the heat-absorption and heat-rejection equipment in the CSM. If the coolant returning from the space radiator is less than 45° F, it is mixed with fluid from the CM thermal load, which has bypassed the radiator, to obtain a mixed-coolant temperature of 45° F. Under mission conditions when the space radiator cannot reject the total load, no bypass occurs; instead, the glycol evaporator cools the 200-lb/hr flow to 41.5° F by evaporating water at a controlled pressure of approximately 0.1 psia. The coolant flow leaving the evaporator is divided into a 35-lb/hr flow directed to the inertial measurement unit (IMU) of the guidance and navigation equipment, and a 165-lb/hr flow is routed to the suit heat exchanger through the drinking-water chiller shown in figure 6. The suit heat exchanger, shown in figure 7, provides the humidity control for the CM. The coolant leaving the suit heat exchanger enters the cabin heat exchanger and absorbs heat from the CM lighting, the electronic equipment not mounted on coldplates, the environmental loads, and the crewmembers in the shirtsleeve mode. The effluent coolant from the guidance and navigation equipment mixes with that from the cabin heat exchanger, and the 200-lb/hr flow is directed through a series-parallel arrangement of 22 coldplates, which absorb the major portion of the thermal load. The heat from the coldplate network may be diverted to the cabin heat exchanger through the cabin-temperature control valve for heating the cabin, when required. The fluid leaving the cabin-temperature control valve enters the pump, and the flow is directed

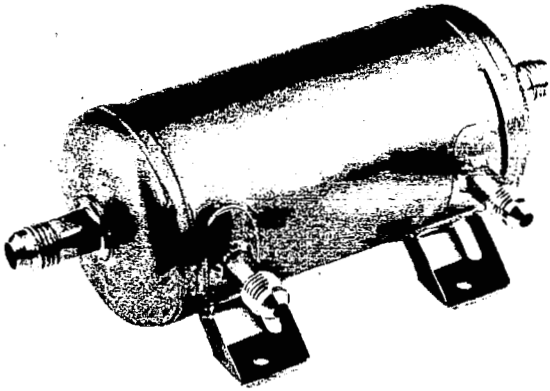


Figure 6. - Drinking-water chiller.

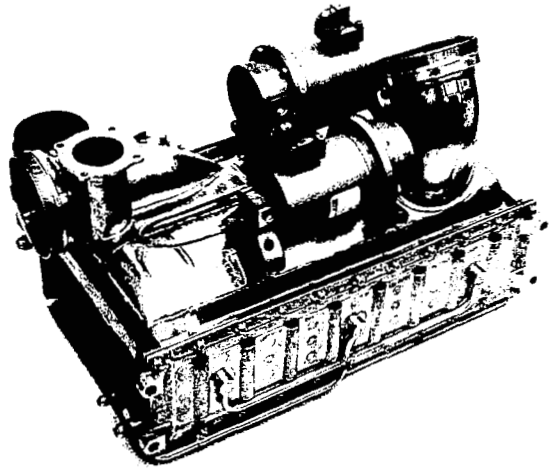


Figure 7. - Suit heat exchanger.

to the space radiator. A secondary coolant loop is provided as a backup for the primary loop and may be operated at the discretion of the crewmembers. Both loops provide cooling for the suit and cabin atmospheres and for the electronic equipment. The secondary loop does not have cabin-heating capability, nor does it provide cooling to the guidance and navigation equipment.

The pressure-suit circuit controls the levels of carbon dioxide, odor, and humidity and can provide a habitable environment for the crewmembers if cabin pressurization is lost. When the crewmembers are in the pressure-suit mode, they are isolated from the cabin. The ventilating gas flow leaving the pressure suits passes through a debris trap, shown in figure 8, which removes particles larger than 0.04 inch. Suit circuit flow is accomplished by one of two centrifugal-flow compressors which deliver 55 lb/hr of suit-circuit gas (35 cu ft/min) at a pressure rise of 10.0 inches of water with an inlet density of 0.0266 lb/cu ft.

As the ventilation gas passes through two parallel elements of lithium hydroxide and activated carbon, the carbon dioxide and odor control for the CM is accomplished. Each element is sized for

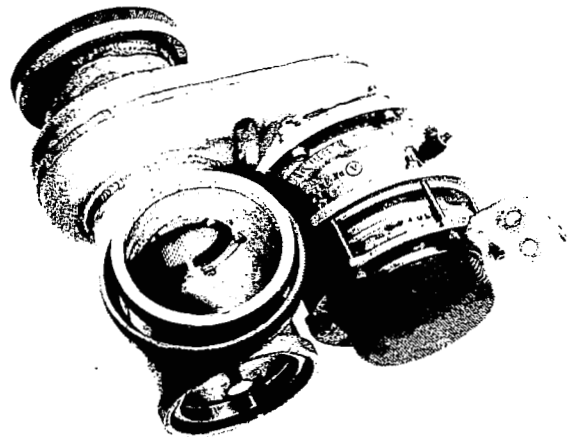


Figure 8. - Apollo solids trap.

1.5 man-days of operation at the design metabolic loads, and the elements are changed by the crew alternately every 12 hours. Twenty elements are carried for 8- to 10-day missions. The element holder, or canister, incorporates the necessary check valves, diverter valve, and interlock mechanisms which permit the changing of elements in a depressurized cabin. The canister is also designed to preclude inadvertent depressurization of the suit circuit.

The gas leaving the carbon dioxide canister enters the suit-circuit heat exchanger, where suit-circuit heat loads are absorbed by the water and glycol. At the heat exchanger, the moisture is condensed, removed by the wicking, and transferred to the waste-water circuit by pneumatically actuated accumulators (fig. 9) which are cycled every 10 minutes by a timing device. The normal gas exit temperature from the heat exchanger is 50° F.

The cool gas is distributed to the three suit-hose-connector units, which incorporate a flow-control adjustment lever and a flow-limiting Venturi tube. When the crewmembers are in the shirtsleeve mode, their portion of the suit-circuit flow is delivered to the cabin through an orifice in the connector unit which approximates the pressure drop of the suit. This flow is returned to the suit circuit for carbon dioxide and humidity removal by the cabin-air-return valve located upstream of the suit compressors. During manned ground testing and during launch, the cabin atmosphere is a mixture of 60 percent oxygen and 40 percent nitrogen. This is the minimum oxygen concentration which will provide a viable atmosphere with a reduction to 5.0 psia in the cabin pressure. Subsequent to orbit insertion, a bleed flow overboard establishes a demand on the cabin-pressure regulator (fig. 10) and enriches the mixture to sea-level equivalent (an oxygen partial pressure of 3.1 psia). The nitrogen content



Figure 9. - Apollo cyclic accumulator.

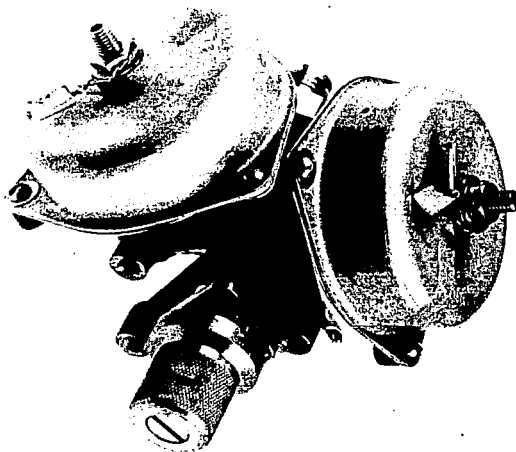


Figure 10. - Cabin pressure regulator.

is reduced further by leakage or LM pressurizations. Technical considerations associated with the selection of the launch environment are presented in the following section.

The crewmembers undergo a period of oxygen prebreathing prior to insertion into the CM suit circuit, which has been purged to an oxygen level greater than 95 percent. This oxygen prebreathing minimizes the possibility of aeroembolism during the boost phase when cabin pressure is reduced from 14.7 to 6 psia. To prevent nitrogen leaking into the suit circuit, a positive pressure relative to the cabin pressure is maintained by a 0.5-lb/hr excess flow.

Finally, the cabin circuit consists of two axial-flow fans. Each fan has a capacity of 86 cu ft/min at 5 psia, which circulates the CM gas through the cabin heat exchanger. A cabin-pressure relief valve relieves the cabin pressure at a differential of 6.0 psi during ascent of the spacecraft and repressurizes the cabin during descent, when the ambient pressure exceeds cabin pressure by approximately 1 psid. After splashdown, a postlanding ventilation system, consisting of an inlet valve and fan and an outlet valve, is activated by the crewmembers to ventilate the cabin until recovery.

In the event of smoke, a toxic gas, or another harmful atmosphere in the cabin during the shirtsleeve environment, three oxygen masks are provided. The mask is a modified commercial full-face-type assembly with headstraps to hold it on. The oxygen is supplied at 100 psi through a flexible hose from the emergency oxygen and repressurization unit. The mask has an integral regulator that supplies oxygen on demand when the crewman inhales.

DEVELOPMENT AND FLIGHT TESTING DIFFICULTIES

Ground Test and Launch Environment

As a result of the criticality of the gas environment in the spacecraft during launch, detailed technical considerations involved in the selection of a two-gas atmosphere for the Apollo CM during ground checkout and launch are presented. The design analysis included a comprehensive engineering trade-off study which resulted in several general conclusions concerning atmospheric selection. The analysis provided the necessary data required for the final selection of a 60-percent oxygen, 40-percent nitrogen launch atmosphere.

The purpose of the evaluation was to determine the feasibility of using a two-gas atmosphere in the CM during ground checkout and launch. Selection of this atmosphere required the consideration of several factors directly affecting crew safety: (1) the physiological acceptability of the cabin atmosphere, (2) the capability of reducing the fire hazard, (3) the operational characteristics of the spacecraft hardware when subjected to the atmosphere, and (4) the additional crew procedures required.

The crewmembers undergo a lengthy oxygen prebreathing period prior to launch, and subjecting them to even a small amount of diluent for a short time could destroy the benefits of this prebreathing period. Therefore, a 100-percent pure oxygen environment during prelaunch checkout and launch was physiologically desirable because it eliminated any potential dysbarism. The pure oxygen environment was least desirable from a

fire-hazard standpoint, since material flammability tends to be proportional to atmospheric oxygen partial pressure. Thus, from a fire-hazard viewpoint alone, pure diluent in the cabin atmosphere seemed most desirable.

Of course, a pure diluent cabin atmosphere would surround the suited crewman with a nonviable atmosphere during the entire ground checkout and launch period. Even though the crew is not dependent on the cabin atmosphere during this time, it was deemed desirable to maintain a livable atmosphere to accommodate any unforeseen emergencies. These considerations dictated the requirement for an analysis of the optimum mixture of oxygen and diluent to be used in the cabin during prelaunch checkout and launch.

During the trade-off study, the following requirements basic to all considerations were made:

1. The fire protection capability of the launch atmosphere must be enhanced by a two-gas mixture, through the use of an inert diluent.
2. The two-gas atmosphere considered must provide a livable cabin atmosphere.
3. If, after orbital insertion, the diluent must be removed to meet physiological requirements, crew operational procedures must be minimized.

In accordance with these requirements, the atmosphere-selection study was conducted, using ECS hardware compatibility testing, systems performance analysis, weight trade-off studies, and materials compatibility studies. An extensive program was organized for analyzing and testing flammability characteristics of components and entire systems within a CM test vehicle.

The trade-off study demonstrated that an air cabin environment would provide a viable atmosphere surrounding the crew during the relatively long prelaunch period, but in order to maintain a viable atmosphere during the entire launch phase and to avoid high purge rates during launch, an oxygen partial pressure higher than that present in air must exist in the cabin atmosphere prior to launch. The oxygen partial pressure in the cabin could be continuously maintained through launch in a physiologically acceptable range with a 50- to 60-percent oxygen concentration. This concentration provides an atmosphere with flammability characteristics near those of a 6-psia pure oxygen environment.

Use of the two-gas cabin atmosphere required certain precautions because the suit-circuit pressure upstream of the suit-compressor inlet is normally less than the cabin pressure. Suit-circuit leaks in that vicinity prior to cabin diluent removal would therefore cause the diluent-rich cabin atmosphere to enter the circuit. Since the diluent is not consumed in metabolism, it would continue to build up in the suit circuit. The oxygen partial pressure in the suit circuit would correspondingly diminish, resulting in potential hypoxia of the suited crew or in aeroembolism upon launch. A bleed rate of approximately 0.80 pound of oxygen per hour into the suit circuit through the manual oxygen-metering valve and vented through the demand pressure regulator was used to provide positive suit pressure to prevent diluent buildup.

The trade-off study showed that, in view of the definition of the cabin oxygen partial pressure required to support the crew during launch and abort modes and in

view of the requirements associated with the fire hazard of various oxygen diluent mixtures, a two-gas mixture for Apollo ground checkout and launch is both acceptable and desirable. Furthermore, the analysis established the feasibility of using a two-gas mixture with current spacecraft hardware for prelaunch and launch. The most desirable composition was found to be a mixture of 50 to 60 percent oxygen in nitrogen. Air and mixtures of helium and oxygen were also considered, but these combinations proved to be undesirable. Several methods of removing the diluent, such as cabin leakage, cabin purging, and depressurization techniques, were investigated. The use of cabin purge through existing overboard dump nozzles was found to be most desirable and has been adopted for each manned mission.

Coldplates

The original design of the Apollo coldplates used parallel passages milled into a 3/8-inch sheet of 6061 aluminum on approximately 1-inch centers. Excess metal was milled chemically from between the strips to lighten the coldplate.

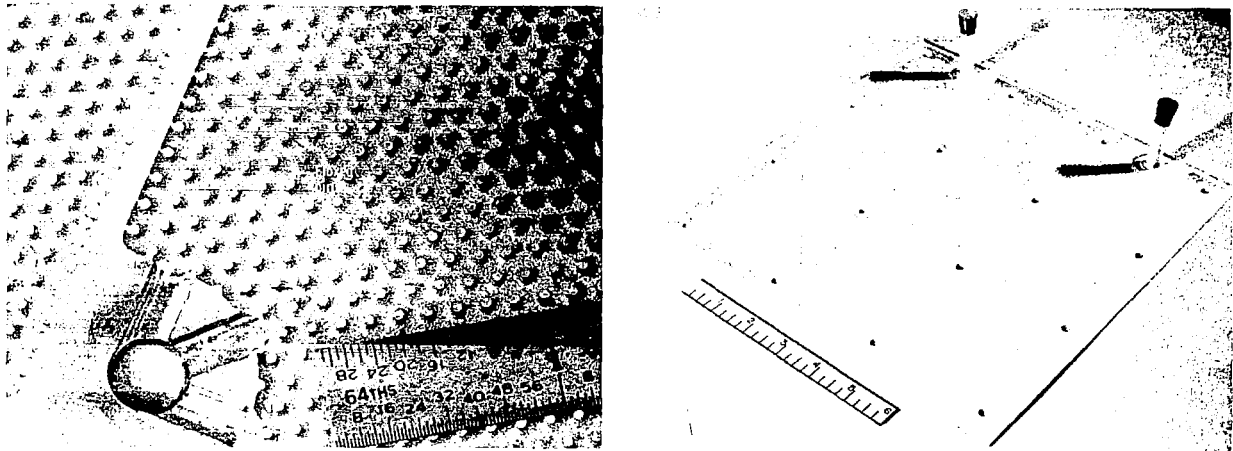
In the original design, the passage through which the fluid travels was small, approximately 0.044 inch wide and 0.100 inch deep. Distribution and collection passages were provided at the inlet and outlet, giving the core the appearance of a ladder-type structure. Base and face sheets of 0.030-inch 5052 aluminum were then silver-eutectic bonded to the core or ladder. One side of the face sheet was electroplated with silver and then bonded to the core by the application of heat and pressure. This process resulted in an assembly which had silver and aluminum eutectic as a bond between the core and face sheet, and this bond was exposed directly to the water and glycol coolant.

The ethylene glycol and water solution is a moderately good electrolyte, and the silver and aluminum couple in the coldplates had a high electromotive potential. Therefore, a galvanic corrosion problem developed in the coldplate, causing concern about potential leaks and flow blockage. Also, the ratio of surface area to coolant volume was large because of the small passage flow area in the coldplates. This condition tended to cause local depletion of the inhibitors at points where the coolant fluid was not circulating, permitting corrosion to occur at these points. Also, the small passage cross section made the coldplate sensitive to obstruction from the corrosion products. Parallel efforts were initiated both to cope with the problem for the first few spacecraft and to eliminate the problem for the later vehicles.

Special ground support equipment was designed to circulate the ethylene glycol and water coolant during periods of spacecraft activity when the coolant circuit was not normally in use. This technique prevented local inhibitor depletion. Also, samples were withdrawn periodically to be analyzed for inhibitor concentration. When this check indicated a decrease in the inhibitor level, the ethylene glycol and water coolant circuit was exchanged and filled with fresh solution.

A redesign of the coldplate was initiated not only to resolve the corrosion problems but also to improve the heat-transfer and fluid-distribution characteristics. The core is eloxed from a sheet of 6061 aluminum to obtain a staggered pattern of 0.05-inch-diameter pins on 0.125-inch centers, with a 60° angle between the center lines. The pin height varies from 0.062 to 0.172 inch with the coldplate application.

The face sheet has a 0.003-inch coating of a fluxless braze alloy consisting of silicon and aluminum, thus eliminating the silver from the bonding process. The new coldplate, referred to as the pin fin configuration (fig. 11), represents an improvement over the ladder-type structure because of a lower surface-to-volume ratio and because of larger passages. The heat-transfer capability was increased from 1.0 to 3.0 W/sq in. for the average dissipation rate and from 2.0 to 6.0 W/sq in. for the maximum local rate.



(a) Pin and inlet manifold details.

(b) Coldplate assembly.

Figure 11. - Pin fin coldplate.

Space Radiator

The ECS space radiator for the Apollo spacecraft was designed as an integral part of the SM structure and originally consisted of two panels located on opposite sides of the SM. Each panel had 30 square feet of radiator area. A four-tube circuit and a six-tube circuit were provided in each panel, with parallel flow through the tubes. The two panels were linked in parallel so that the flow path within the radiator was through 20 parallel tubes on four separate circuits, each circuit having an isolation capability. This configuration resulted in a radiator having a minimum pressure drop and was consistent with optimizing the pumping power requirements of the coolant circuit.

Three factors made the selected configuration inadequate to reject the spacecraft thermal loads. First, gradual growth of power requirements by other spacecraft systems resulted in thermal loads that exceeded the heat-rejection capability of the 60 square feet of radiator area (3700-Btu/hr capability as compared to the 4850-Btu/hr requirement for an average earth-orbital environment). The difference between the capability and the requirement had to be rejected by water evaporation and consequently curtailed the mission-duration capability. Because the radiator was an integral part of the SM structure, it was not practical to increase the area without a major impact

on the program. Second, a problem associated with unequal fluid flow rates through the tubes was caused by the outer tubes on each panel being longer than the other tubes. The resulting effect was magnified by the disproportionately larger fin areas associated with the longer tubes. Third, no provision was made to control the flow between the two panels, which are usually exposed to different radiant environments.

When the radiator was exposed to a cold environment, the combination of the longer outer tubes and the lack of thermal isolation from the SM, coupled with the viscosity characteristics of the ethylene glycol and water coolant at low temperatures, acted to induce flow stagnation in the outer tubes. A slight plumbing change to the outer tubes reduced the "sensitivity to flow imbalance," but it was still necessary to add a supplementary water supply of 112 pounds in the SM to provide the required heat-rejection capability for a 14-day earth-orbital mission.

The redesign of the CSM to a configuration which implemented the lunar-mission capability provided an appropriate change point to correct the ECS radiator deficiencies. The experience gained was used to eliminate all three of the undesirable features in the earlier radiator. The resulting design consists of a primary and a secondary radiator system. The primary radiator circuit consists basically of two radiator panels, each with an area of 50 square feet, located on opposite sides of the SM (fig. 12). With this arrangement, one panel may be exposed to deep space at the time the opposite panel is exposed to a heat source such as the sun, earth, or moon. These extremes in environments will produce large differences in the effectiveness of each panel and, consequently, in the fluid outlet temperatures. The panel exposed to deep space can reject more heat than the panel receiving external radiation; therefore, the overall heat rejection of the system can be increased by increasing the flow to the cold panel.

The flow through the radiators is controlled by a dual-flow-proportioning valve assembly. During operation, if a difference in radiator-panel outlet temperature occurs, the flow-proportioning valve will be positioned to increase the coolant flow to the colder radiator panel. At a temperature differential of 10° F, the flow-proportioning valve will divert approximately 95 percent of the flow to the cold radiator. A redundant flow-proportioning system is provided; the system contains a logic network to initiate and indicate an automatic switchover if improper operation occurs.

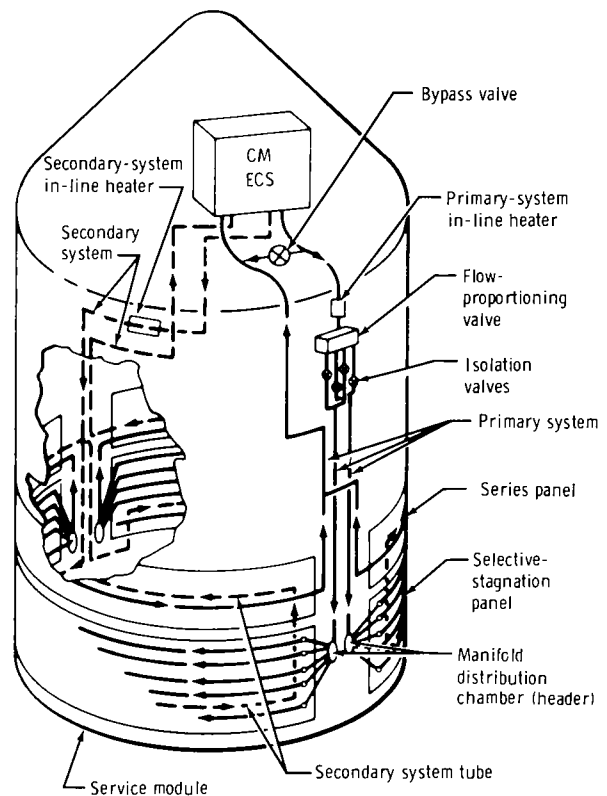


Figure 12. - Selective stagnation/flow proportioning radiator system.

In situations when the radiator inlet temperature is low and the panels have a favorable environment for heat rejection, the radiator outlet temperature starts to decrease, and the hot-glycol bypass flow within the CM is initiated. As more flow is bypassed, the radiator outlet temperature decreases. An in-line heater upstream of the radiator is automatically turned on when the outlet temperature of the radiator mixed coolant drops to -15° F.

In addition to flow proportioning and heater control, a passive system of effective radiator area control called selective stagnation is incorporated. The two radiator panels are identical, with five tubes in parallel and one tube in series downstream of the other five. The five tubes have manifolds sized to provide graduated flow through the individual tubes. Thus, for equal fin areas, the tube with the lower flow rate will have the lowest outlet temperature. As outlet temperature decreases, the flow resistance in the minimum flow tube increases, thus further reducing the flow rate and outlet temperature. As the fin area around the tube gets colder, it draws heat from the adjacent tube, and the same process occurs with each successive tube. In a fully stagnated condition, there will be essentially no flow in three tubes. The primary (minimum resistance) flow tube will carry most of the flow with a small flow through the adjacent tube.

As the heat load increases, the radiator inlet temperature increases; and more heat is transferred into the stagnated tubes, resulting in successive increases in flow rate and ultimately full panel operation. Therefore, at high heat loads the panels automatically provide a high effectiveness (completely thawed panels operating at a high average surface temperature); and at low heat loads, the panels provide a low effectiveness (stagnated panels operating at a low average surface temperature).

The secondary radiator is provided as a backup in the event of failure of the primary system. This radiator consists of four tubes placed close to the hottest primary circuit tubes so that the ethylene glycol and water coolant in the secondary tubes will not freeze when the secondary circuit is inoperative. The selective stagnation principle is not used in the secondary radiator because of the smaller range of heat-load requirements. This lack of a passive-control mechanism causes the secondary coolant circuit to be dependent on the heater control system at low heat loads and on the evaporator at high heat loads for control of the ethylene glycol and water coolant temperature.

Evaporator

The glycol evaporator (fig. 13) uses a plate-fin sandwich construction in a cross-counterflow arrangement. The core is comprised of brazed modules of finned glycol passages manifolded together and of steam passage fins brazed to each side of the exterior surface. Nickel felt-metal wicks of 15-percent density are sandwiched between the glycol modules, and the assembly is brazed to form the evaporator core. The water inlet and distribution plate is composed of a solid plate of stainless steel in which channels are milled for waterflow passages. To this plate is brazed a sintered stainless-steel porous plate which has a pore rating of 5 microns. This assembly distributes the water uniformly over the entire surface of the plate and is bolted to a flange around the base of the core. A cellulose sponge pad is compressed between the distribution plate and the base of the wicks to ensure contact and uniform water distribution

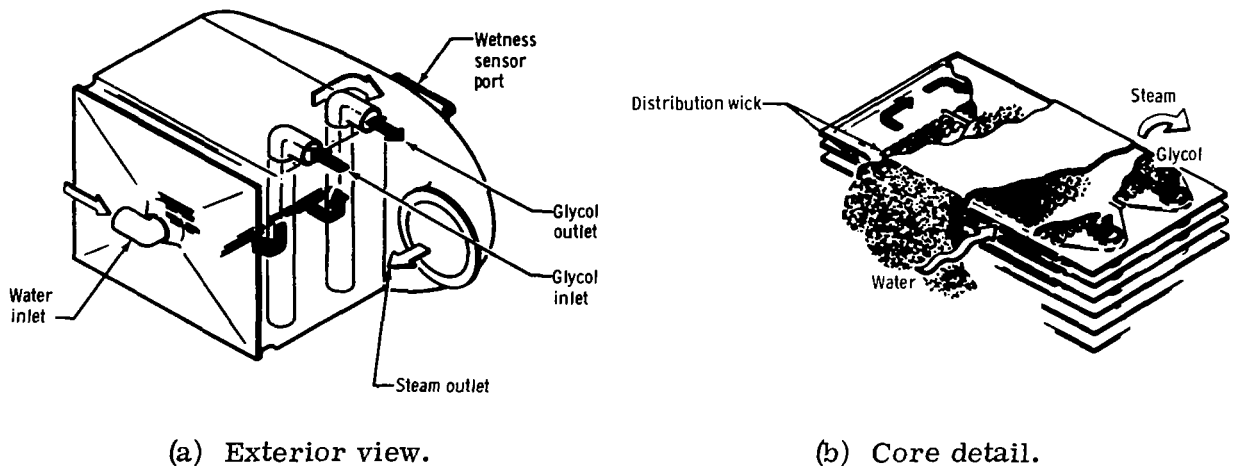


Figure 13. - Original evaporator design.

to all the wicks. To complete the component, the steam exhaust pan, with an integral flange to accept the backpressure valve, is brazed to the top of the core.

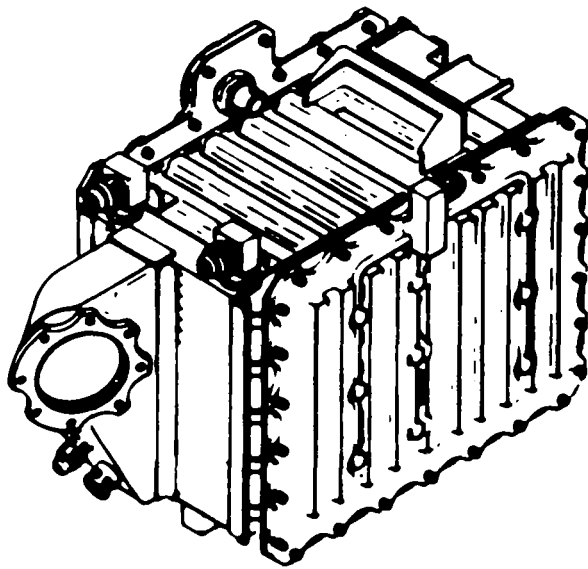
The evaporator, which is automatically actuated when the fluid temperature leaving the space radiator exceeds 49° F, presented a difficult developmental problem because of three considerations involving systems integration. The first factor involved the need for water management in the overall spacecraft mass and energy balance. The glycol evaporator is required to operate frequently and at high evaporation rates because of the marginal heat-rejection capability of the radiator when operating in the relatively warm environments of earth orbit and lunar orbit. This factor dictated an evaporator design requirement of high efficiency, that is, exit steam with a quality approaching 100 percent. The second factor involved the requirement for accurate and constant temperature control of the coolant supplied to the spacecraft guidance and navigation equipment, particularly the IMU. The design that was selected controlled the coolant temperature by controlling the pressure at which evaporation occurred. A backpressure valve, close-coupled to and linking the evaporator with the steam exit duct, allowed the evaporator to be maintained in a wet condition, ready to function immediately upon demand should the radiator exit temperature begin to rise. The third factor involved the steam duct leading from the evaporator to space vacuum. The final design configuration was a 2-inch-diameter duct approximately 8 feet in length and having three 90° bends because of trade-offs on equipment locations within the crew compartment and the spacecraft-attitude requirements during entry. The length and shape of the steam duct emphasized the requirement for free water control, because any free water leaving the evaporator would tend to freeze, accumulate, and eventually obstruct the passage.

Early developmental testing of a control system, which metered water to the evaporator as a function of the coolant outlet temperature error, proved unsuccessful because of its slow response. The backpressure control method was then developed, and its performance is satisfactory for rapid startup and accurate temperature control. However, problems were encountered in the control function which supplied the makeup water to the evaporator, and these problems merit further discussion.

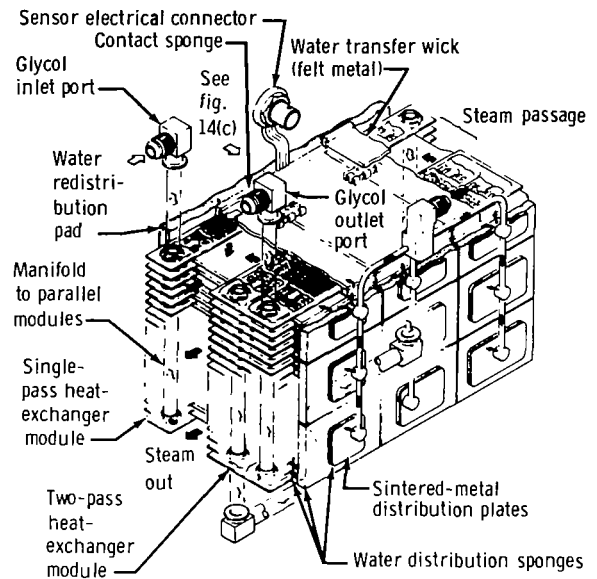
The backpressure-controlled evaporator configuration used a limit switch within the backpressure valve to serve as a permissive function. The switch enabled the water or wetness control to be activated when the backpressure valve moved off the fully closed position. The wetness control function was isolated from the backpressure (or temperature) control function, except for the activation switch in the backpressure valve. The major components that provided wetness control included a solenoid-actuated water-inflow valve, an electronics package or controller, an evaporator inlet glycol-temperature sensor which served as a variable-reference temperature for the controller, and a wetness sensor. The wetness sensor consisted of a glass-sealed thermistor enclosed by a coil-type heater. The assembly was encapsulated in a metal cylinder. The sensor was located on the steam exhaust pan of the evaporator above the core and upstream of the backpressure valve so that the sensor was directly in the path of the steam flow. The concept involved the cooling effect of wet steam on the heated sensor and subsequent turning off of the water inflow to the evaporator. Developmental testing demonstrated that the wetness sensor was not sensitive to the quality of the steam and that droplets of water impinging on the sensor were required to produce the cooling effect necessary to deenergize the water inflow valve. Furthermore, it was discovered that the evaporator was not actually wick-fed but was a pool boiler. When operated with the water inlet port oriented at the bottom (the attitude in which it is installed in the spacecraft), the level of free water in the core would vary as a function of the heat load being rejected. At low loads, 500 Btu/hr, the core would be nearly filled with water and boiling would occur at the top of the core. In this mode, droplets of water needed to travel only a short distance before they reached the wetness sensor. Conversely, at the high loads of 7600 Btu/hr, no free water could be observed.

A control system which permitted free water to accumulate in the core was not acceptable for two significant reasons. The first reason involved the characteristic which the evaporator exhibited when exposed to an increasing heat load. Under conditions of increasing inlet glycol temperature, tests indicated that free water was ejected from the evaporator. The mechanism for the liquid carryover appeared to be bubbles of vapor which passed rapidly from the bottom to the top of the liquid mass, carrying liquid over as the bubbles broke the surface. Also, it was demonstrated that this quantity of water was sufficient to obstruct the steam duct. Testing in this attitude would not be representative of flight conditions, because gravity forces were acting to retain the liquid in the core and, in the absence of gravity, a larger quantity would be expelled under similar conditions. This attention to predicted performance in the weightless environment was the other factor which caused concern over the operating characteristics of the wetness control function.

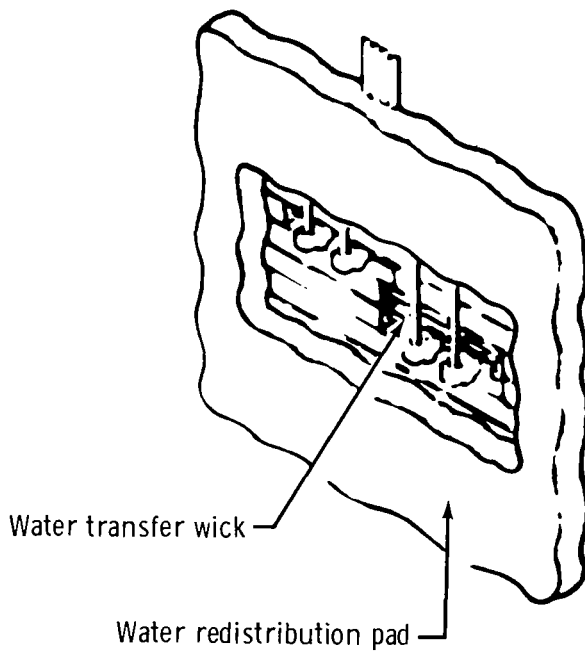
A redesign was initiated (fig. 14) which required that two basic changes to the evaporator be incorporated. First, the heater was removed from the wetness sensor, and the thermistor was relocated so that it was embedded in a wick. This improved the response characteristic of the wetness sensor because conduction, rather than convection, became the mode by which the sensor measured the degree of wetness in the evaporator. Second, the core was redesigned so that a sponge pad connected the top edges of all wicks, thus permitting redistribution of water between the wicks to compensate for nonuniformity in wicking rates. This redistribution capability resulted in a uniform wick wetness throughout the core and permitted the wetness sensor location in one wick to be representative of all wicks.



(a) Exterior view.



(b) Cutaway view.



(c) View of sensor detail.

Figure 14. - Modified evaporator design.

Because of the problems encountered with the attitude sensitivity of the evaporator during testing and because of the difficulty in extrapolating these test results to a prediction of performance in a weightless environment, a requirement was placed on the evaporator that it perform in all attitudes without water carryover. Although this requirement was stringent, it was deemed necessary to demonstrate that capillary forces in the wicks dominate the effects of gravity. As part of the redesign effort to meet the all-attitude test criteria, the one-piece water distribution plate was replaced by nine small plates, each separately fed by a water distribution manifold, for improved distribution. In addition, the material was changed from sintered stainless-steel porous plate (5-micron rating) to a sintered laminate of nickel screens, each having a 12-micron effective pore size. This produced an increased pressure drop, permitting a more uniform distribution of the water over the the plate to the wicks.

Qualification testing revealed that the small passages within the sintered nickel plates became obstructed because of the formation of nickel oxidation products, which built up with exposure to water. Also, this small pore size was very sensitive to particle contamination from the upstream plumbing. The nickel plates were subsequently replaced by a stack of five stainless-steel plates in which 0.005-inch-diameter holes were drilled, thus producing a series-orifice effect which adequately met the uniform distribution requirement and was much less sensitive to contamination. The final configuration of the evaporator, with these changes incorporated, was flown on the Apollo 11 mission, and the performance was entirely satisfactory. However, to avoid perturbations of the spacecraft attitude, the evaporator was used only during launch and entry.

The interaction of the various control functions of the coolant circuit is explained in figure 15 which illustrates the system performance of the Apollo 8 spacecraft during lunar orbit. During the eighth revolution, the radiator outlet temperature decreased as the CSM entered the dark side of the moon. At approximately 49° F, the evaporator operation was terminated, and the CM thermal load bypass was initiated to maintain the evaporator outlet temperature at a nominal temperature of 45° F. Radiator outlet temperature decreased at 18° F minimum as the CMS left the lunar night and then increased to a maximum of 65° F under the temperature increase as a result of solar and lunar radiation. The evaporator operation was initiated automatically as the radiator outlet temperature exceeded a 49° F nominal temperature. The coolant-loop heat loads were essentially constant during this period, as indicated by a stable radiator inlet temperature of 73° to 75° F.

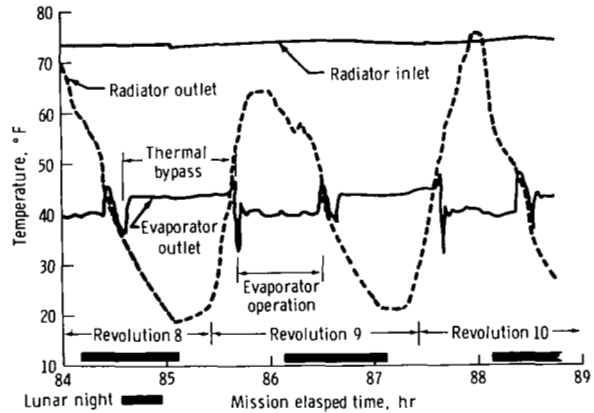


Figure 15. - Coolant system performance during the Apollo 8 mission.

Water System

The Apollo CSM water processing and distribution system is the product of initial design concepts modified by extensive developmental and flight testing experience. The design of the system for Apollo 11 and subsequent missions is adequate for the intended use but could not be considered optimized from a systems-engineering standpoint.

All potable water available in the CM, except for that initially loaded preflight, is supplied by the spacecraft fuel cells located in the SM. Medical requirements dictate the repeated injection of chlorine into the potable water, for bacteria control. The system distribution plumbing is such that all fuel-cell water is transferred to the CM and is either stored in the waste- or potable-water tanks, taken directly to the use ports, or vented directly overboard. System design is such that available water will be preferentially directed to the potable-water tank at any time the tank is not full. When the potable-water tank is filled, product water will be automatically diverted into the waste-water tank.

During the system development and qualification testing and during spacecraft flight testing, several problems were identified and required either operational changes or hardware modifications. The problem areas included plumbing corrosion, undesirable chlorine taste, and excessive quantities of free gas.

After numerous development tests and evaluation of the water system and components, the amount of plumbing corrosion was determined to be undesirable but represented no hazard to mission success for 14-day missions. In an effort to minimize system corrosion, operational constraints were made to preclude exposure of the system to chlorine until countdown operations began for launch. These procedures have proved adequate to preclude inflight failures.

During the early Apollo flights, the crewmembers reported on several occasions that the water had a strong chlorine taste. In most instances, the difficulty was traced to a procedure error occurring during the injection of the chlorine and buffer ampules. When clear and concise procedures were developed and used, the crews had no objection to the taste of the water.

The potable water generated by the fuel cells is saturated with hydrogen at the fuel-cell operating pressure and temperature. When the pressure on the water is reduced from the fuel-cell pressure (60 psia) to the cabin-use pressure (5.0 psia), large quantities of free gas evolve in the water. The free gas consumed by the crew during eating and drinking caused discomfort to the crewmembers. Water/gas separators were developed by NASA and installed to provide gas-free water for crew consumption.

The water/gas separators (fig. 16) were designed to use the unique surface-tension characteristics available from hydrophobic (gas-permeable Teflon) and hydrophilic (stainless-steel filter mesh) membranes. In use, the two membranes are placed in close proximity to each other, and the water/gas mixture is forced to flow between them (fig. 17). The hydrophobic membrane selectively passes the gas from the mixture while rejecting the liquid. The gas flowing through the membrane is dumped to the spacecraft cabin. The hydrophilic membrane, when wetted with water, allows water flow, while acting as a barrier to the gas in the mixture. The water is collected downstream of the hydrophilic membrane and delivered as gas-free water for crew consumption. The water/gas separator was designed to permit inflight installation on the use ports (water gun or food preparation unit) as shown in figure 18.

Operationally, the CSM water system has proved to be very satisfactory. Particularly, the flight crews have noted that the availability of hot water for food and drink preparation was desirable and should be a part of future spacecraft systems. The chilled-water port was not used often because the water gun was more convenient and also provided chilled water.

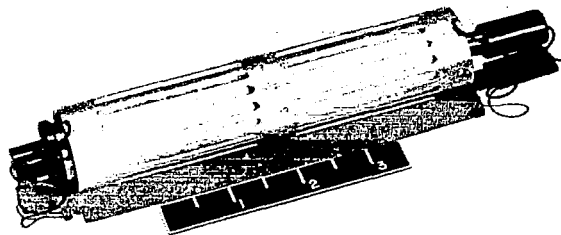


Figure 16. - Water/gas separator.

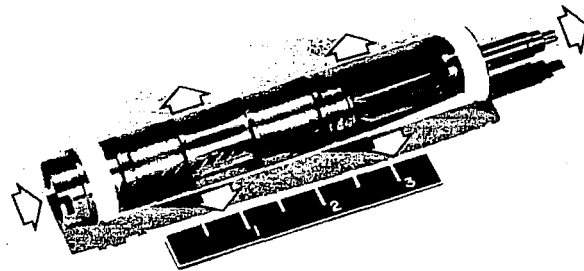


Figure 17. - Cutaway view of water/gas separator.

Waste Management System

The waste management system is required to dispose of crew waste solids and liquids. The equipment is designed to meet requirements for collection and direct overboard venting of urine and for collection and stowage of feces.

The urine collection and transfer system consists of a receiver and collection bag, a transfer hose, miscellaneous valves and plumbing, and an overboard dump nozzle with an electric heater to prevent nozzle freezing. The urine collection assembly consists of a pliable reservoir bag interfaced to the spacecraft plumbing with a quick disconnect fitting and to the crewman with a roll-on cuff (rubber tube) that serves as an external catheter between the penis and the collector inlet valve. During urination, the liquid is vented directly overboard. When the micturition rate exceeds the flow-rate capability of the overboard dump nozzle, the reservoir bag serves as a collector to

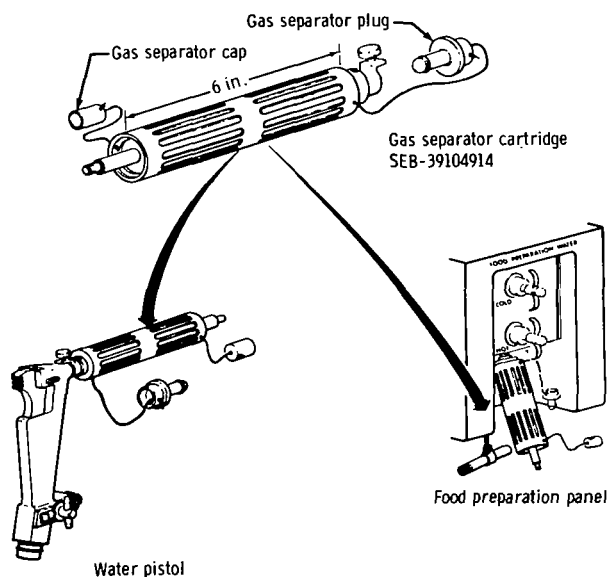


Figure 18. - Inflight installation of water/gas separator.

be vented overboard after termination of urine flow. Also, the volume of the reservoir bag is adequate to store one complete urination volume when it may be undesirable to vent directly overboard. The roll-on cuffs tend to deteriorate during usage, and replacement cuffs are provided for inflight replacement.

Equipment for fecal collection and stowage includes the fecal collection assembly, cleansing tissue, and waste stowage compartment. The fecal collection assembly

consists of a fecal collector and container bag open at one end. The open end of the bag has a flange with a tape surface for adhering the flange to the buttocks. Each bag is packaged with a wet cleansing cloth and a germicide. After use, the germicide is placed in the bag and the bag is sealed. After the feces is permeated with the germicide, the bag is stowed inside the waste storage compartment. The capability is provided to purge the waste storage compartment of objectionable odors if necessary.

Both the urine collection and transfer system and the fecal collection and stowage system have been the subject of extensive criticism by most of the crewmembers. After the Apollo 11 mission, the urine collection and transfer assembly was replaced with a urine receiver assembly (figs. 19 and 20) which, unlike the prior system, does not require crew contact with the unit to prevent urine spillage during use.

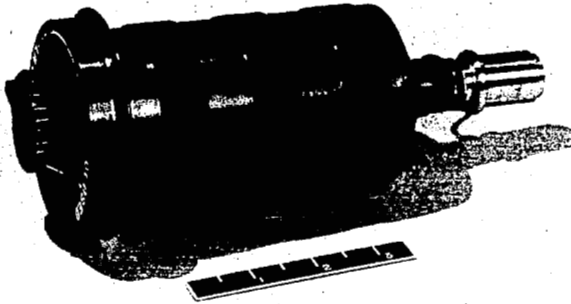


Figure 19. - Urine receiver assembly

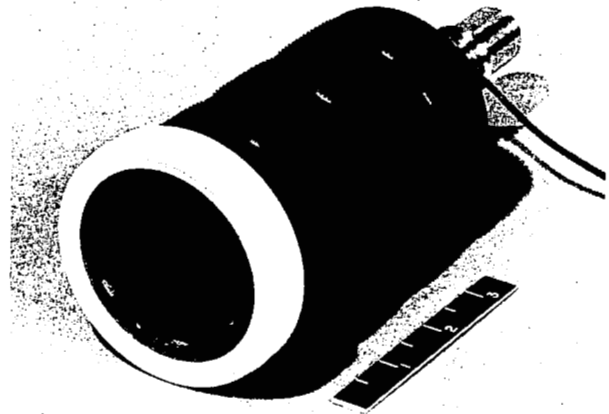


Figure 20. - Urine receiver assembly with cap removed.

Cabin Noise

During the early Apollo missions, the crews registered numerous complaints about the excessive noise level created when the cabin fans were operated. Methods of acoustically isolating the fans were identified but never implemented on flight vehicles. It was determined from additional flight experience that the fans were not mandatory for cabin thermal control for the types of missions which were then planned, and the expense of the modification was not justifiable.

Both in Apollo missions and in long-term ground-based tests, it has been concluded that added emphasis must be placed on reducing or controlling the noise output from spacecraft components and fluid flow systems.

CONCLUDING REMARKS

During the course of the development-qualification and flight-test programs, the environmental control system has performed in an outstanding manner. Within the flight system, there have been no environmental control system failures that have placed the crew or the mission in jeopardy. There have, however, been malfunctions which caused changes to planned operational procedures.

During the period of qualification and flight testing, considerable difficulty was experienced with the coldplates used for electronic equipment cooling, the evaporators for supplemental heat rejection, the radiators for primary heat rejection, the potable-water system, and the waste management system. The difficulties associated with these items are of a basic design nature and should receive prime consideration on future programs.

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National Aeronautics and Space Administration
Houston, Texas, May 24, 1971
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