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TECHNICAL HISTORY OF THE ENVIRONMENTAL CONTROL SYSTEM FOR PROJECT MERCURY

by Frank H. Samonski, Jr.

Manned Spacecraft Center
Houston, Texas

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ABSTRACT

This report presents a technical history of the environmental control system for Project Mercury. Significant system changes and flight experience with the environmental control system are described. Attention is also given to the structure of test programs employed to satisfy the mission objectives.

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TECHNICAL HISTORY OF
THE ENVIRONMENTAL CONTROL SYSTEM
FOR PROJECT MERCURY

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SUMMARY

The primary goal of Project Mercury was to place a man into an earth orbit and to return him safely to earth. Throughout the program, astronaut safety was the prime consideration, and one of the most important design requirements was the maintenance of a habitable environment in space. This report describes the evolution of the environmental control system used in the Project Mercury spacecraft and presents a brief technical record of the significant system changes. Meeting the mission reliability objectives required stringent attention to details of testing, and sections of this report describe the structure of test programs employed to satisfy the mission objectives. The flight experience with the environmental control system is presented for all the missions from Little Joe 5A, successfully accomplished on March 18, 1961, to Mercury-Atlas 9, which was successfully accomplished on May 15 and 16, 1963.

INTRODUCTION

The primary function of the environmental control system (ECS) was to provide and maintain a physiologically acceptable environment to support human life. This task required accurate and reliable control of gas composition, pressure, and temperature. A secondary function was to provide a satisfactory environment for the spacecraft electrical equipment by maintaining cabin pressure and temperature within suitable limits. The ECS design configuration was conceived by the McDonnell Aircraft Corporation (MAC), and the contract for hardware was awarded in February 1959.

The system design is semiclosed, in which carbon dioxide (CO_2) is removed by chemical absorption and in which water vapor is removed by condensation and then stored. The oxygen (O_2) is recirculated by a fan, and makeup is provided from two 7500-psig gaseous-storage supplies. Metabolic and equipment heat loads are rejected by evaporating water at approximately 35° F in heat exchangers which are supplied water (H_2O) from a positive-expulsion water-storage tank. The ECS is described by separating the system into the pressure-suit and cabin subsystems.

A chemical analysis of the MA-9 lithium hydroxide canister was written by Wayland J. Rippstein and is incorporated into this paper as appendix A.

ENVIRONMENTAL CONTROL SYSTEM DESCRIPTION

Pressure-Suit Subsystem

In normal-mode operation of the pressure-suit circuit, the gas leaving the pressure-suit helmet passes through a 34-inch length of flexible hose. This hose has a smooth bore, is silicone-rubber-coated, and has an internal diameter of 1 inch. The pressure-suit exit gas passes through a solids trap (1)[1] made of a stainless-steel screen material which will filter particles larger than 40 microns. A bypass is provided in the event the screen should become obstructed, and it is actuated when the differential pressure across the trap increases to 0.5 inch of water. The gas leaving the solids trap is ducted to the inlet of the suit compressor (2) which recirculates 11.4 ft^3/min of oxygen through the suit circuit. The compressor operates with a differential pressure of 10 inches of water when recirculating oxygen at a density of 0.0272 lb/ft^3. The compressor is of centrifugal design and is driven by a 115-volt, 400-cycle, single-phase electric motor through step-up gearing with a ratio of approximately 1.8, thus producing an impeller speed of 43 000 rpm. The motor heat is absorbed by the gas stream, and the compressor power requirements are 46 watts when operating at the conditions previously described. An identical standby compressor (3) is mounted in parallel with the primary compressor and can be placed in operation automatically or manually in the event of failure of the primary unit. The differential pressure across the primary compressor is continually monitored by a pressure-differential switch (4) which turns off the primary unit and turns on the standby compressor whenever the differential pressure decreases below 3.5 inches of water. This automatic mode of operation occurs only when the suit-fan selector switch is in the normal position. Positions on the selector switch permit manual selection of either compressor. A spring-loaded, double-D, flapper-type check valve (5) is located at the outlet of each compressor; and the check valve opens only when the respective unit is operating, thus preventing recirculation between the parallel compressors. A common discharge duct from the two compressors directs the flow to the inlet of the CO_2 and odor absorber (6) through a flexible rubber bellows designed to facilitate replacement of the absorber and to isolate it from compressor vibrations. The interior of the absorber is divided into identical parallel paths, each containing in order, in the direction of flow, a 0.5-pound charge of activated charcoal and two 1.15-pound charges of 4- to 8-mesh lithium hydroxide (LiOH), or a total of 1.0 pound of activated charcoal and 4.6 pounds of LiOH. The activated charcoal removes odors that would tend to build up in the closed suit circuit. The LiOH absorbs CO_2 according to the reaction $2\,LiOH + CO_2 \rightarrow Li_2CO_3 + H_2O + \text{heat}$. Each of the six charges is contained in a fiber-glass-cloth bag. The gas leaving the absorber passes through an Orlon felt filter. The absorber

[1]Numbers refer to corresponding numbers on the schematic diagram in figure 1.

charge is compressed by springs to prevent separation of the LiOH during launch vibrations, which could cause channeling (or partial bypassing) of the flow through the absorber and, consequently, only partial CO_2 removal.

The gas leaving the absorber next passes into the suit-circuit heat exchanger (7) where it is cooled to approximately 45° F and where metabolic water and water from the $LiOH-CO_2$ reaction are condensed. The heat is rejected into water which evaporates into the vacuum of space at approximately 35° F. The heat of vaporization under these conditions is approximately 1075 Btu per pound of water evaporated. The total heat load in the suit circuit is approximately 700 Btu/hr, which includes the metabolic heat output of the pilot, the heat of reaction of the LiOH, and the compressor motor heat. This heat load would thus correspond to a waterflow rate of 0.7 lb/hr. The heat-exchanger construction is aluminum-plate fin with alternating passages for the gas and for the evaporating water. Heat-transfer configuration is the cross-counterflow type in which the gas makes a single pass while the evaporating water makes two passes. The pilot adjusts the waterflow manually with a needle valve (8). The water for use in both the suit and cabin heat exchangers is stored in a 39-pound-capacity coolant tank (9), which incorporates a bladder to insure positive expulsion in the zero-g environment.

Water condensed in the heat exchanger is collected in the suit water separator (10) which consists of a sponge that is squeezed by a pneumatic piston. The capsule programer energizes the water-separator solenoid valve for 30 seconds every 30 minutes, supplying the piston with 100-psig oxygen to squeeze the sponge. The water-gas mixture trapped in the sponge is discharged through a 5-psig relief check valve into a condensate tank (11). A sintered bronze plug in this tank releases the gas to the cabin and retains the condensate. A normally open check valve of the same type as used at the compressor discharge is provided at the gas outlet of the water separator and is used in the emergency mode of system operation (to be explained in another section of this report).

The recirculating oxygen flow, cleansed of CO_2 and odors, cooled, and dehumidified, is returned to the inlet fitting at the waist of the pressure suit. Four flexible tubes within the pressure suit duct the gas to the hands and feet. The gas flows over the body, collecting heat and water vapor within the suit and collecting CO_2 from the helmet area. The gas then exits from the helmet fitting.

Oxygen makeup for the metabolic process is accomplished by one of two methods. In the first method, the suit pressure or demand regulator (12) supplies oxygen to the suit circuit when the pressure at the suit outlet decreases to 3.0 ± 0.5 inches of water below cabin pressure. A small quantity of oxygen is then admitted to the suit circuit and elevates the pressure above this control point. The demand regulator is connected to the suit circuit just downstream of the solids trap. In addition to the demand diaphragm which controls suit pressure to slightly below cabin pressure, a relief diaphragm is provided which relieves suit pressure to cabin pressure when the suit pressure exceeds the cabin pressure by 2 inches of water. Both the demand and relief diaphragms normally sense cabin pressure, but the regulator will automatically produce an artificial reference pressure of 4.6 ± 0.2 psia should the cabin pressure fall below this value. The artificial reference pressure is developed by either of two redundant aneroids and by a constant-bleed orifice. These aneroids close when the cabin pressure decreases below 4.6 psia and isolate the demand- and relief-diaphragm reference chambers from cabin pressure. Two constant-bleed orifices, which flow a maximum of 30 cc/min each at standard temperature and pressure (STP), maintain pressure in the reference chambers and provide for case leakage of the demand regulator.

The second method of oxygen supply is accomplished by the cabin pressure control valve (13). This valve senses cabin pressure and provides makeup for cabin leakage. The discharge from the valve is ducted through the suit circuit. This arrangement effects an oxygen purge for the pressure suit, supplies metabolic requirements, and tends to maintain the suit above cabin pressure, thus preventing the accumulation of contaminants in the pressure suit. The ECS operates in this mode in the following manner. When the cabin pressure decreases to approximately 5.1 psia, the cabin pressure control valve supplies oxygen at the maximum rate of 0.010 lb/min to compensate for cabin leakage. This flow is supplied to the suit circuit which will increase in pressure until the relief diaphragm in the demand regulator relieves the flow to the cabin (when the differential pressure has increased to 2 inches of water). The operation will continue in this mode until the cabin pressure increase is sufficient to turn off the valve, somewhere within the operating tolerances of $5.1^{+0.2}_{-0.3}$ psia. In the event that the cabin leakage rate exceeds 0.010 lb/min, the cabin pressure will eventually decay to zero. The cabin pressure control valve will continue to supply oxygen at 0.010 lb/min as the suit and cabin pressures decay. The suit pressure will stop decreasing at 4.6 psia. The cabin pressure control valve will automatically cease supplying oxygen when the cabin pressure decreases to $4.0^{+0.2}_{-0.1}$ psia. The cabin pressure control valve has two separate control aneroids and metering orifices for redundancy. The valve also has a cabin-repressurization capability of 0.17 lb/min which is manually operated only and is independent of the cabin pressure level.

The oxygen supply for the spacecraft consists of two 4-pound tanks of gaseous oxygen stored at 7500 psig. One tank is designated as the primary supply (14), and the other is designated as the secondary supply (15). Either tank contains sufficient oxygen for a 28-hour mission, provided the cabin leakage rate is less than 300 cc/min at STP. The oxygen tanks are spheres with an internal volume of 198 cubic inches, and they are constructed of type 4340 carbon steel with electroless nickel plating for corrosion resistance. The structural requirements of the tank are 12 500 psig proof pressure and 16 700 psig burst pressure. The subassembly for each tank includes a high-pressure shutoff valve (16), a filler valve (17) with cap, and a pressure transducer (18) for measuring the quantity of oxygen remaining.

The primary-oxygen-tank pressure is reduced from 7500 psig to 100 ± 10 psig by a single-stage pressure reducer (19). For redundancy, two reducers are provided by the primary system. The secondary oxygen tank has a single pressure reducer (20) which regulates to 80 ± 10 psig. Both tanks are connected by a manifold which is downstream of their respective reducers in such a way that, upon depletion of the primary system, the pressure in the manifold connecting the two systems will decay to 80 psig and permit the flow to automatically initiate from the secondary system. A check valve (21) is provided at the discharge of each reducer to prevent reverse flow from one system in the event of failure of the other system.

The manifold common to both oxygen supplies normally operates at 100 ± 10 psig and supplies oxygen to four valves in the ECS. Three of these have already been discussed, namely the suit pressure or demand regulator, the cabin pressure control valve, and the water-separator solenoid valve. The fourth valve is the emergency oxygen rate valve (22) which provides an emergency mode of operation in the suit circuit.

The emergency mode of operation is actuated automatically whenever the suit pressure decreases to approximately 4.0 psia, or it can be actuated manually by the "Emer O_2" handle on the right-hand console of the spacecraft. An aneroid in the emergency oxygen rate valve continually senses suit pressure and actuates a microswitch within the valve when the suit pressure decreases to $4.0 {}^{+0.1}_{-0.3}$ psia. The microswitch energizes a solenoid in the suit-circuit shutoff valve (23). The shutoff valve, located downstream of the solids trap, consists of a spoon valve which is spring-loaded to close but is held open by a pin that is integral with the core of the solenoid. When the microswitch in the emergency rate valve closes, it energizes the solenoid, retracting the pin. The stored energy in the spring closes the shutoff valve and, by mechanical linkage, rotates a shaft in the rate valve which uncovers an orifice and initiates an emergency oxygen flow rate of 0.049 to 0.075 lb/min. The linkage is extended from the rate valve to the right-hand console, and the "Emer O_2" handle moves to the emergency position. A suit-fan cutoff relay interrupts power to the suit compressor that is operating, and the check valve downstream of the water separator closes, thus isolating the compressors, CO_2 and odor absorber, heat exchanger, and water separator from the pressure suit. The emergency flow rate is relieved to the cabin through the relief port of the suit pressure regulator. When the suit pressure rises to a value greater than 4.0 psia, the handle can be returned to the normal position, which automatically returns the system to the normal mode of operation. With this action, the suit-fan cutoff relay is deenergized; and the suit compressor operates, the system shutoff valve is opened and latched in the open position by the solenoid, and flow from the emergency rate valve is interrupted.

The emergency mode of operation can be initiated manually by the pilot at any time by moving the "Emer O_2" handle to the emergency position. This action, through the mechanical linkage, rotates the shaft of the rate valve. The shaft has an integral helical ramp which, when rotated, lifts a pin that closes the microswitch. The reaction of the system is the same as that which occurs when the aneroid in the rate valve closes the microswitch.

Cabin Subsystem

The ECS for the cabin is comprised of the cabin fan (24) and cabin heat exchanger (25), the cabin pressure relief valve (26), the cabin air inlet valve (27) and cabin air outflow valve (28), and the cabin pressure control valve which has already been described in conjunction with the suit circuit.

The cabin fan delivers gas to the cabin heat exchanger which removes heat generated by onboard electrical equipment. The fan is of vane-axial design and is driven by a 115-volt, 400-cycle, single-phase motor which rotates the fan blade at approximately 24 000 rpm. The fan delivers 22.3 ft^3/min of oxygen at 5.0 psia and 70° F with a pressure rise of 0.5 inch of water. Operating under these conditions, the fan requires 17 watts. The fan inlet is provided with a screen, and the outlet is clamped to the cabin heat-exchanger inlet duct.

The cabin heat exchanger is similar in construction to the suit-circuit heat exchanger already described, and it rejects about 500 Btu/hr of the onboard-equipment heat load with a cabin temperature of approximately 100° F. Operation at these conditions requires a waterflow of about 0.5 lb/hr, and the waterflow is manually controlled by adjusting a needle valve (29).

The cabin pressure relief valve is a differential type which operates at $5.5^{+0.4}_{-0.1}$ psig to relieve cabin pressure during ascent and controls the upper limit of cabin pressure to this value during the orbital phase. The valve begins to repressurize the cabin during descent when the ambient pressure exceeds cabin pressure by approximately 1.0 psi. Construction of the valve incorporates redundancy for both the control element and the relief valve. A manual override permits the pilot to manually decompress the cabin in the event of fire or of toxic-gas buildup. An additional manual control allows the valve to be locked closed after landing, thus preventing water from entering the spacecraft.

The ground ventilation system is actuated during descent of the spacecraft at an altitude of about 17 000 feet. At this time, the inflow and outflow valves are both opened barometrically, allowing ambient air to be drawn through the suit circuit by the suit compressor and to be vented to the cabin through the relief port in the suit pressure regulator. The outflow valve relieves this flow to the ambient environment, thus preventing a pressure increase within the cabin. The inflow valve is equipped with an inlet snorkel (30) to prevent water from being drawn into the suit circuit should the spacecraft submerge upon landing. The inflow valve also incorporates three electrical switches which perform the following functions when the valve opens.

1. Turn off the cabin fan.

2. Place the suit circuit in the emergency mode, that is, open the emergency oxygen rate valve and close the system shutoff valve.

3. Deenergize the suit-fan cutoff relay in such a way that the suit compressor continues to operate.

When the spacecraft lands, it is possible that water trapped in the recovery compartment where the inlet snorkel is located could cause the snorkel ball to seat. The suction from the suit compressor could retain the ball in this position, thus preventing air from being drawn into the suit circuit. To offset this condition, a negative-pressure (vacuum) relief valve (32) was installed in the inflow line between the inflow valve and the suit compressor. The operation of this valve is such that it relieves the vacuum developed in the inflow line by the suit compressor when the differential pressure reaches 15 inches of water below cabin pressure. At this point, the valve opens and allows cabin pressure to vent into the line, thus releasing the ball. The valve reseats when the vacuum has decreased to 2 inches of water.

The inflow and outflow valves are of the spoon type, each spring-loaded to the open position, but held closed by an aneroid assembly. During ascent, the aneroid expands and arms the valve with a detent pin. On descent, the aneroid retracts and releases the spring, thus closing the valve. An override permits the pilot to open both valves manually.

The previous explanation describes the ECS as originally conceived and designed. Several changes were made during the development, qualification, and flight-testing phases, and these changes are discussed later. Additional description of the ECS operating modes, detailed explanations of valves and components, and descriptions of interfaces with the spacecraft electrical and instrumentation systems are given in references 1 to 8.

SYSTEM CHANGES

Oxygen-Supply Filler Valve

A quick-disconnect, spring-loaded-poppet filler valve (subcontractor part number (P/N) 137203-1, MAC P/N 45-83700-97, and vendor P/N 1010-0002) was originally used on the oxygen-supply tanks. Valves that successfully completed preinstallation acceptance testing were found to leak during checkout of assembled systems. Leakage problems were attributed to contamination of the metal-to-metal seal within the valve. A new valve was selected which employed a tapered-shaft seal and required a wrench operation to connect and disconnect the service line and to open and close the filler port. The new valve functioned satisfactorily throughout Project Mercury.

Pressure-Switch Deletion

The original ECS design included a pressure switch (MAC P/N 45-83700-103 and subcontractor P/N 133186) on the primary oxygen supply, upstream of the pressure reducer. The switch measured tank pressure and actuated at 200 ± 25 psig to illuminate the "O_2 Quantity" warning light on the instrument panel. The switch warned the pilot that the primary oxygen supply was nearly depleted and that automatic transfer to the secondary oxygen supply would occur in a few minutes. During vibration testing of the complete ECS, leakage developed in the pressure switch because of extrusion of an internal O-ring. The leak was accompanied by ignition of the O-ring, which vaporized portions of the aluminum housing of the switch and deposited aluminum oxide over other components of the system. This malfunction (and subsequent difficulties with the inherent design problem of sensing 200 ± 25 psig from the 7500-psig operating range for the pressure switch) resulted in the deletion of this component from the ECS. The function served by the pressure switch was replaced by the oxygen flow sensors which are discussed in the following section.

Oxygen Flow Sensor

An oxygen-flow-sensing system (MAC P/N 45-88107-1) was designed to replace the function of the oxygen-pressure switch (MAC P/N 45-83700-103). This sytem used a flow-sensitive thermistor element downstream of each oxygen-supply check valve in such a way that the "O_2 Quantity" warning light would illuminate when flow was interrupted from either supply. A switch on the instrument panel permitted selection of the thermistor on the primary or the secondary supply. Normal operation was to place the selector switch in the primary-supply position; and, since the primary system would normally be supplying oxygen, the "O_2 Quantity" warning light would not be illuminated. Depletion of the primary supply would cause flow to cease and the light to illuminate. The pilot would then place the selector switch in the secondary-supply position; and, since flow would have begun automatically from the secondary supply, the light would go out, thus verifying the automatic transfer function of the oxygen system. The sensor also included screens to prevent malfunction in ECS valves should the thermistor element break off and be carried into critical components by the gas stream.

Difficulties were encountered in the vibration qualification testing because of motion of the air column in the connecting tubing. This motion caused actuation of the flow-sensing circuitry under no-flow test conditions. The sensor was modified to include dampers which prevented motion in this air column. The sensor (MAC P/N 45-88107-1) was flown in the Mercury-Redstone 4 (MR-4) mission and appeared to operate satisfactorily.

The screens and dampers together caused sufficient restriction to make the pressure drop across the flow sensor prohibitive. Under high-flow conditions from the primary supply, the pressure drop was sufficient to cause flow to initiate from the secondary supply simultaneously. The units were modified by removing the dampers and were assigned MAC P/N 45-88107-19.

Further difficulties were encountered with the modified design, and the pressure-drop effect was still observed to occur with the new configuration. The confidence level in the accuracy of the device was questionable, but the device was operative in the Mercury-Atlas 6 (MA-6) mission. The units were also installed for the MA-7 mission, but the electronics were disconnected because of difficulty during preflight systems tests.

A modification was made to the secondary-oxygen-supply-system pressure transducer so that it would actuate the "O_2 Quantity" warning light when the tank pressure had decreased to 6500 psig. This function served essentially the same purpose as the original oxygen-pressure switch, that is, to warn the pilot that flow had begun from the secondary oxygen supply. This transducer (MAC P/N 45-83700-825 and subcontractor P/N 538943) was employed in Spacecraft 20.

Cabin Purge Technique

The first system configuration included a launch purge system which consisted of a tank containing 1 pound of gaseous oxygen stored at 7500 psi, a pressure regulator, and a barometrically actuated launch purge valve. The purge valve would automatically initiate flow from the 1-pound supply when the cabin pressure had decreased to an equivalent altitude of 10 000 feet, and the supply would be depleted when the cabin altitude had decreased to an equivalent of 27 000 feet. Calculations indicated that the effectiveness of the purge would be approximately 65 percent. The launch purge system, although used in the ballistic flight program, was replaced in the manned orbital missions by a prelaunch purge. Removal of the launch-purge-system hardware and of the 1-pound oxygen charge reduced the launch weight of the spacecraft by approximately 4-3/4 pounds. The prelaunch purge was accomplished through a fitting in the spacecraft hatch after the pilot had been inserted and the hatch installed. Approximately 15 pounds of oxygen were used to attain a minimum purity of 98-percent oxygen prior to launch. The purge gas was relieved from the cabin through the manual decompression port of the cabin pressure relief valve.

Suit-Circuit Purge Technique

The purge valve (MAC P/N 45-83700-81, subcontractor P/N PS 137205, and vendor P/N 4510) was originally located on the suit exit duct between the solids trap and the suit-circuit-shutoff valve. Manned altitude testing at MAC demonstrated that purging the suit circuit at this location did not purge the hose between the inflow snorkel and the suit circuit, which represents a considerable volume in relation to the suit circuit. During the ascent from sea level to altitude, the air in this line expanded into the suit circuit and diluted the circuit with nitrogen. This effect was measured, and the result was a 10-percent decrease in the suit-circuit oxygen level. An additional suit-circuit purge valve was therefore added to the inflow line, downstream and adjacent to the ground ventilation inlet valve. Prelaunch purging from this location decreased the decay on launch by 3 to 5 percent. The original purge (ground inlet) valve (34) location was retained and used as a sample extraction point to measure the quality of the suit-circuit purge.

Cabin Pressure Relief Valve

An absolute-pressure-controlled cabin pressure relief valve was designed and developed for the Mercury system. It included two redundant aneroids which controlled the cabin pressure at an upper limit of $5.5 {}^{+0.2}_{-0.1}$ psia and would relieve cabin pressure during ascent, maintaining a maximum differential pressure of 1 psi between cabin pressure and ambient. This valve was identified by MAC P/N 45-83700-77 and subcontractor P/N 102350. Spacecraft structural analysis indicated a desire for a higher differential pressure to insure structural integrity in the event of an abort during the maximum-dynamic-pressure period of flight. The valve was subsequently modified to a differential-pressure relief valve which would both relieve during launch and maintain the upper limit of cabin pressure at $5.5 {}^{+0.4}_{-0.1}$ psia differential. The new valve was identified by MAC P/N 45-83700-725 and subcontractor P/N 102416. The old configuration used a solenoid to close the valve for prelaunch leakage tests at 5 psig. The differential-type valve inherently possessed this leakage-test capability, and deletion of the solenoid resulted in a weight saving of 0.31 pound.

Before the differential-type relief valve was developed and the hardware made available, it was discovered that the original design for negative-pressure relief would admit water, trapped in the recovery compartment after landing, to the pressurized cabin before the scuppers provided could drain the compartment. It was decided to modify the original design as a temporary change for the earlier spacecraft and missions. The modification was to increase the spring constant of the spring associated with the negative-pressure relief function of the valve so that the valve would not relieve until the external-to-internal differential pressure had increased to 15 inches of water, minimum. This modification was identified by MAC P/N 45-83700-733 and subcontractor P/N 102350-1.

The cabin pressure in the Little Joe 5A mission was observed to decay to zero. Postflight inspection revealed a piece of safety wire lodged on the seat of the valve, and this was responsible for the malfunction. The 45-83700-725 configuration of the valve was modified to include debris-protective screens over the cabin relief ports of the valve, and an additional screen (MAC P/N 45-83119) was added to cover the ambient or recovery compartment port of the valve.

Spacecraft 13, 18, and 20 included a manual locking device (MAC P/N 45-83123) which permitted the pilot to positively lock the valve against the entry of water after landing.

Water Separator

Early models of the water separator (MAC P/N 45-83700-59 and subcontractor P/N 175830) used a sponge which was hard and stiff when dry. Since the separator actuation was completely automatic and was controlled by the spacecraft programer with no override capability, several units were damaged when operated inadvertently with a dry sponge. In addition, there was no method of checking the position of the separator piston to verify that the separator was, in fact, in proper working condition and that it had not been damaged by some previous inadvertent actuation. The separator was modified by replacing the sponge with one of polyurethane, which has the property of being flexible when dry. A second modification incorporated a permanent magnet into the piston such that the position of the piston could be determined with the aid of a ground-support-equipment item. This checkout unit used a magnetic switch and a light to sense and indicate the position of the piston. Because of inaccessibility of the water separator in the spacecraft, this method of testing proved unsatisfactory. The part numbers corresponding to these modifications were MAC P/N 45-83700-727, subcontractor P/N 175830-2, and MAC P/N 45-83700-729, subcontractor P/N 175830-3.

In order to provide the pilot with both manual control of the separator and visual indication of separator actuation, two further modifications were made to the water separator. The first modification added a switch on the instrument panel to allow the pilot to actuate the water separator manually. The second added two magnetic switches to the exterior of the separator case and added two lights to the instrument panel. The two lights operated to indicate partial and full travel of the separator piston. A normally functioning water separator in its actuation cycle would illuminate the indicator lights in the following sequence: partial travel, full and partial travel, full travel, full and partial travel, and the final step, partial travel. This last modification was identified by MAC P/N 45-83097-303 and used the MAC P/N 45-83700-729 separator with a bolt-on magnetic switch (subcontractor P/N 177948).

Suit Pressure Regulator

The suit pressure- or demand-regulator configuration originally planned for the Mercury system used a demand diaphragm that actuated at 1-inch-of-water negative-differential pressure with respect to cabin pressure (subcontractor P/N 132190). Early system tests with this configuration indicated that this control level was too sensitive to the operation of the suit-circuit compressor and to motion by the test subject within the pressure suit, which resulted in excessive actuation of the demand valve and, consequently, in excessive and wasteful use of oxygen. The first modification to this valve changed the control level to 3.0 ± 0.5-inches-of-water negative-differential pressure.

During the manned ECS test program at MAC, it was discovered that the relief diaphragm, which normally relieved suit pressure when it exceeded cabin pressure by 2 inches of water, was dynamically unstable when operated in conjunction with the emergency oxygen mode and with decompressed cabin conditions. Subsequent tests and analyses revealed that the relief-diaphragm support required an increase in area to correct the above condition. Radial supports plus an additional annular support ring were added and produced satisfactory operation. The valve was manufactured by the vendor as P/N F-5001000 and was identified before this change by MAC P/N 45-83700-41 and subcontractor P/N 132190-1. The regulator was identified after the change to the relief diaphragm by MAC P/N 45-83700-715 and by subcontractor P/N 132190-4.

The altitude-chamber testing of Spacecraft 11 (Service Engineering Department Report (SEDR) 83-11) at Cape Kennedy (formerly Cape Canaveral) revealed another suit-pressure-regulator problem. Restricting ventilation flow in the suit outlet hose caused the demand valve to stick in the full-open position, releasing the total stored oxygen supply in a short period with no reset capability available. Subsequent investigation revealed that the regulator design did not incorporate a retaining ring on the return spring of the demand-valve stem. The spring, when once dislodged from the grooved seat provided for it (as resulted from full travel of the demand diaphragm), would not return the demand valve to the closed position when the pressure differential was removed from the diaphragm. The design was modified to include a spring-retaining ring, and the modification proved satisfactory. The regulator was reidentified by MAC P/N 45-83700-753.

Further operational experience with this valve indicated a need for debris-protective screens over the relief port openings in the valve case. The part number corresponding to this modification was MAC P/N 45-83700-831.

Coolant-Quantity Indicating System

A coolant-quantity indicating system (CQIS), comprised of a 22-cubic-inch coolant-oxygen-supply cylinder (35), a 500-psi-pressure coolant-quantity transducer (36), and an absolute coolant-oxygen-pressure (reducer) control valve (37), was employed on all orbital spacecraft with the exception of Spacecraft 16 and 20. This system determined the quantity of coolant remaining by measuring the pressure decay of the small oxygen cylinder as the absolute-pressure control valve maintained a constant pressure on the gas side of the coolant-tank bladder. The pressure transducer was calibrated to indicate the percentage of coolant quantity remaining of the 39-pound full-tank capacity. Postflight measurement of coolant remaining demonstrated the inherent inaccuracy of the gaging system which resulted from fluctuating cabin and, therefore, oxygen-cylinder temperature. This system was assigned the following part numbers as the configuration was modified in development: MAC P/N 45-83007-1, -301, -303, -305, -307, -309, and -311.

Emergency Oxygen Rate Valve

The emergency oxygen flow of 0.03 to 0.04 lb/min of the first valve configuration (MAC P/N 45-83700-33 and subcontractor P/N 132186) was demonstrated to be insufficient for cooling the pilot. This determination was made in the first manned system development tests in November 1959. The flow rate was increased to 0.050 ± 0.001 lb/min and this new configuration was assigned MAC P/N 45-83700-483 and subcontractor P/N 132186-1.

The microswitch within the valve, which initiated the emergency mode automatically when the suit pressure decreased to 4.0 psia, was found to stick repeatedly in the closed position. The valve was modified by the addition of a spring, and the subcontractor part number was changed to P/N 132186-2 with no change to the MAC part number.

Recurrent difficulties were encountered with the linkages of the emergency rate valve, system shutoff valve, and emergency rate handle in Spacecraft 5, 7, and 8. The problems resulted from the tedious and complicated rigging procedure required to make the system operate and remain operational. The design employed a Teleflex cable in compression to return the system from emergency to normal mode. The high spring constant of the system shutoff valve required forces greater than 40 pounds to return the system to normal mode. The reset forces had to be transmitted with the Teleflex cable to a bellcrank on the rate valve and then to the system shutoff valve. A multi-strand steel cable connected another arm on the bellcrank to the system shutoff valve. The attachment mechanism of the shutoff valve was a lightly spring-loaded sector arm. This light spring load retained the rate valve in the closed or off position until the emergency handle was actuated to place the system in the emergency mode. In the MA-4 mission, the rate valve opened sufficiently to permit a high oxygen usage rate without the system indicating the emergency mode of operation. Since a crewman simulator was used in this mission, the high oxygen usage rate was not serious. To prevent partial opening without indication in the manned flights, a deadband specification was incorporated into the valve so that the microswitch would actuate between 20° and 25° of travel from the off position and flow would initiate after 25° of travel. The 20° of deadband travel were necessary because of inherent difficulties with the valve-rigging procedure and with the variation of installations in various spacecraft. The modified valve was assigned MAC P/N 45-83700-797 and subcontractor P/N 132620-1.

The flexible cable between the emergency rate valve and the system shutoff valve was replaced with a solid linkage in Spacecraft 16, 18, and 20. Due to the fixed amount of travel required of the system shutoff valve to effect a satisfactory seal in the emergency mode, the travel of the emergency-rate-valve stem with the solid-link configuration varied with installation. During testing (SEDR 83-18) at Cape Kennedy, it was discovered that this condition restricted axial travel of the shuttle valve within the rate valve and prevented the system from being returned to the normal mode after automatic initiation of the emergency mode because of low suit pressure. It was therefore necessary to include another restriction such that the internal shuttle valve would be reseated after 70° of valve travel from the off position. This last modification proved satisfactory and eliminated further installation and rigging problems.

Carbon Dioxide and Odor Absorber

The development program of the original CO_2 and odor absorber (MAC P/N 45-83700-47) revealed a serious LiOH dusting problem at the absorber exit during vibration testing. It was determined that the synthetic felt filter mat did not effectively filter the fine LiOH dust which is extremely irritating to the nose and throat. A polyurethane foam was added, and the two filters in series proved satisfactory. This new configuration was identified by MAC P/N 45-83700-417 and subcontractor P/N 175950. Since the original configuration was of welded construction and was not rechargeable, the design was modified to allow recharging by removal of a bolted flange at the absorber exit. This model was assigned subcontractor P/N 176080-1, and no change was made to the MAC part number.

The quantities of the absorber contents were modified to increase the operational life for the manned 1-day-mission spacecraft from 34.5 to 43.6 hours as follows: the LiOH was increased from 4.6 to 5.4 pounds, and the activated carbon was decreased from 1.0 to 0.2 pound. The increased-useful-life absorber was assigned MAC P/N 45-83700-795 and subcontractor P/N 176080-4.

Postflight chemical analysis of the CO_2 and odor absorber from the MA-9 mission indicated only 73-percent use of the LiOH, although the CO_2 partial pressure was observed to increase after only 32-1/4 hours of flight. Further chemical analyses disclosed definite patterns of channeling within the absorber. The complete report of the chemical analyses and procedures is included in Appendix A. Briefly, this report indicates that the two upper bags of LiOH were 90 percent expended while the lower two bags were but 55 percent expended. The report gives diagrams of the channeling pattern. (See appendix B for results of the analyses on the activated carbon from the flight canisters.)

Outflow Snorkel Valve

The outflow snorkel valve (MAC P/N 45-83700-101 and subcontractor P/N 121048) consisted of an outflow duct fitted with a screen-retained ball, the duct being bolted to the small pressure bulkhead in the unpressurized recovery compartment. The snorkel (31) was intended to prevent water from entering the pressurized compartment through the cabin outflow valve after landing. The design was found to be unsatisfactory in this application since it would not effect a water seal for certain expected flotation positions of the spacecraft. A lightly spring-loaded diaphragm-type check valve (MAC P/N 45-83121) was designed as a replacement and proved satisfactory.

Constant-Bleed Orifice

In order to insure a pure oxygen atmosphere in the MA-6 mission, a constant-oxygen bleed was introduced into the pressure-suit circuit in excess of the metabolic oxygen requirements of the pilot. This orifice (MAC P/N 45-83700-731 and subcontractor P/N 131040) metered oxygen at the rate of 720 cc/min (measured at 14.7 psia and 70° F) to the pressure-suit inlet duct. This open-system technique was employed since more than sufficient oxygen was available for the mission. The modification was deleted after the MA-6 mission.

Ground Ventilation Inlet Valve

Manual closing and resetting of this valve (MAC P/N 45-83700-95 and subcontractor P/N 122294-1) by test personnel was a difficult and time-consuming procedure because of the location of the valve on the small pressure bulkhead. In addition, the valve design was such that it was necessary to release a small, spring-loaded detent pin in order to reset the valve. A problem was also encountered in retaining the O-ring on the periphery of the spoon valve. The O-ring frequently escaped from its groove when the valve was actuated. The valve-latching mechanism and detent pin were modified, and the valve was reidentified by MAC P/N 45-83700-721 and subcontractor P/N 122294-2. The valve automatic-arming and -actuation pressures were determined to be too high and were changed to 20 000 ± 3000 feet. This configuration was assigned MAC P/N 45-83700-723 and subcontractor P/N 122366-2. The sealing O-ring was bonded to the spoon valve, and this model was identified by subcontractor P/N 122366-20 with no change in the MAC part number.

In the MR-2 mission, the cabin pressure decayed to zero during the ballistic flight. Postflight analysis indicated that the ground ventilation inlet valve (MAC P/N 45-83700-721) had vibrated open 57 seconds after lift-off. This malfunction automatically placed the ECS in the postlanding mode, allowing cabin pressure to relieve overboard through the open valve via the negative-pressure relief valve. Postflight inspection of the inlet valve (fig. 2) revealed that a spring-loaded snubber pin and mounting bracket, designed to impose a frictional load on the manual-actuation linkage of the valve, had been removed some time prior to flight because of interference problems with the linkage. The snubber was eliminated from the valve because of the resetting difficulties, and a new latching mechanism was incorporated (fig. 3). This final configuration was identified by MAC P/N 45-83700-785 and subcontractor P/N 122366-3-1. No further difficulties were encountered with this component.

Oxygen Check Valves

The duct-mounted oxygen check valve (MAC P/N 45-83700-53 and subcontractor P/N 123104), employed at the discharge side of both suit compressors, at the water-separator exit, and in the inflow line, was comprised of two semicircular discs hinged at the diameter. Orientation relative to gravity forces was, therefore, essential. Frequent difficulties and delays were encountered during system tests because of improper orientation. This problem was most often experienced when testing flow rates of the suit compressors, since the nonoperating compressor check valve must seal to prevent recirculation between the parallel-mounted compressors. A positioning lug was provided on the check valve, but it was frequently installed improperly.

The time-delay relay associated with the automatic switchover between the two suit compressors often did not allow sufficient time for the check valves to change position, since the compressor flow provided the only actuation force. Consequently, light coil springs were added to the flapper discs, and the springs exerted a force sufficient to prevent the valves from reaching the full-open position under no-gas-flow conditions. This spring-loaded check valve was employed at all locations, with the exception of the water-separator exit.

Negative-Pressure Relief Valve

Early system tests revealed that the ball on the inflow snorkel (MAC P/N 45-83700-99 and subcontractor P/N 121046) would not release from the seal on the snorkel duct because of the vacuum created by the suit compressor. The weight of the ball was increased from 1.3 to 1.55 pounds in an unsuccessful attempt to alleviate this situation. The new snorkel was assigned subcontractor P/N 212046-1 with no changes to the MAC part number.

In order to prevent the resulting interruption of suit ventilation flow in the post-landing mode should the ball not release, a negative-pressure relief valve was designed and installed in the inflow line. The valve (MAC P/N 45-83700-423 and subcontractor P/N 130110-1) actuated when 10 to 15 inches of water vacuum relative to cabin pressure were developed in the inflow line by the suit compressor. At this point, the valve would relieve cabin pressure into the line, thus releasing the ball and permitting pressure-suit ventilation to resume. The valve would reseat when the vacuum had decayed to a 2-inch water maximum.

Difficulties were encountered with valve leakage and sticking due to contamination of the O-ring sealing lubricant and with nonrectilinear motion of the moving member. Cocking of the valve was due to insufficient contact area for the moving shaft within the valve guide. The design was modified and reidentified by MAC P/N 45-83700-703 and subcontractor P/N 130110-2.

Comfort Control Valve

The comfort control valve (CCV) used to manually control the rate of water supplied to the heat exchangers was originally made of aluminum (types 6061-T6 and 2024-T4) and was designated by MAC P/N 45-83700-63 and subcontractor P/N 121034. Corrosion problems which restricted flow necessitated a change of materials to Monel metal. In addition, the flow rate was changed from 1.0 lb/hr maximum to 2.0 lb/hr minimum at $180^\circ \pm 20^\circ$ of valve rotation. An orifice at the valve exit, which limited the maximum flow rate, was deleted. The modified valve was assigned MAC P/N 45-83700-491 and subcontractor P/N 121078-1. It was discovered that the new valve could turn past its stops, permitting rotation angles greater than 360°. The valve was modified by the addition of set screws to prevent turning beyond the stops and was redesignated by MAC P/N 45-83700-711 and subcontractor P/N 121078-2.

Because of the extremely small annular area used to control waterflow under normal conditions (approximately 0.8 lb/hr for the suit-circuit heat exchanger), the valve was subject to clogging and, consequently, to a reduction in the waterflow rate. This type of malfunction occurred in the MA-8 mission. Reference to detail drawings for the valve parts indicates that the distance between the metering stem and valve body is approximately 6.23×10^{-4} inches (16 microns) when the valve is open 90°; this angle of rotation corresponds to the position of the valve for normal operation. The postflight inspection of the valve from Spacecraft 16 revealed clogging of the water passage, resulting from dried-out lubricant flakes. The valve specification was modified to include a thorough cleaning and relubrication with Invelcro 33. Valves so modified were designated by MAC P/N 45-83700-853. No further difficulties were encountered with the CCV. The specification for the water used in the spacecraft cooling system was reviewed and modified to minimize the particle count in the 0- to 50- and 50- to 100-micron ranges. The water-handling procedure for spacecraft servicing was also modified to include filtration through a 0.45-micron Millipore filter.

Freon Check Valve

The most troublesome component in the ECS was the Freon check valve (MAC P/N 45-83700-91, subcontractor P/N PS 132200, and vendor P/N 2303-FP4). Corrosion problems with the first valve configuration resulted in a change of materials from aluminum to stainless steel. The new part was designated by MAC P/N 45-83700-747 and subcontractor P/N 132334-1 and included specifications for improved internal finishes and for 100-percent quality control during manufacturing. The rejection rate on the new valves was extremely high during acceptance tests at both the subcontractor and MAC, the reasons for rejection being contamination and valve sticking due to rough internal surfaces. The valve specification was modified to stipulate that all internal sliding surfaces have a surface finish of better than 16. This second modification was assigned subcontractor P/N 132334-2, and the hardware successfully completed the qualification test program. This latest modification still experienced high rejection rates during acceptance testing because of poor-quality machining and excessive contamination. One of these valves used in the coldplate inverter cooling system failed closed during the MA-6 mission. This malfunction is discussed elsewhere in this report.

A new vendor was retained to manufacture the Freon check valve. This new valve design was similar to the older valves, the primary difference being the change in the manufacturer in an attempt to improve the quality of the hardware. The new valve was assigned MAC P/N 45-83700-799 and subcontractor P/N 132632-1. The specification to the vendor was modified to include a requirement on the internal surface finish to be better than 32. This last modification (MAC P/N 45-83700-849) proved satisfactory, and the Freon check valves operated without malfunction in both the MA-8 and MA-9 missions.

High-Pressure Oxygen Shutoff Valve

The shutoff valve (MAC P/N 45-83700-23, subcontractor P/N 132180, and vendor P/N P8-406) located downstream of the oxygen supply tanks had a history of leakage problems which were both erratic and recurrent. The valve specification required that there be zero leakage both internally and externally. The design incorporated three moving O-ring seals on the valve stem, two to prevent leakage through the bonnet and the third to seal the bottom ambient sensing port. Early valve problems were attributed to scuffing the O-rings. The machining finish of the stem was improved, and the internal valve body was contoured to allow smooth and gradual expansion of the O-rings as the stem was rotated to the open position. In addition, drying-out of the lubricant was believed to be a contributing factor. The O-ring lubricant was changed from molybdenum disulfide to silicone grease and to Invelcro 33. Contamination of the valve was also believed to be causing the erratic leakage, and a filter was added to the ground-servicing equipment used to fill the oxygen-supply tanks.

None of the previously mentioned efforts completely eliminated the recurring leakage problems. A procedure was instituted which required that the valves be fully opened (six turns) and then be closed one full turn. This technique greatly reduced or eliminated leakage problems with most valves. The final solution was to delete the system requirement for a high-pressure (7500 psi) shutoff valve in lieu of a low-pressure (100 psi) valve located downstream of the pressure reducer. This valve (MAC P/N 45-83700-809, subcontractor P/N 132424, and vendor P/N P17-717) and pressure-reducer combination was qualified by a 1000-hour age test at 7500 psi. This new configuration required modifications to the high-pressure reducer, and this problem is discussed in this report in conjunction with the reducer. The low-pressure shutoff valve was flown successfully in the MA-9 mission.

High-Pressure Reducer

The secondary-oxygen-supply pressure exhibited an unexplained decay during the MA-6 mission. Both oxygen supplies had been serviced to approximately 8000 psig prior to launch; and, since the maximum indication of the supply pressure transducers was 7500 psig, the exact time when the secondary supply began to leak cannot be determined accurately. Postflight testing conducted 9 days after the flight (March 1, 1962) revealed no leakage in the secondary supply. Additional tests conducted on March 8, 1962, confirmed zero leakage in the secondary system. On March 9, 1962, a leak was noticed prior to applying pressure to the outlet of the secondary pressure reducer (MAC P/N 45-83700-29, subcontractor P/N PS 132184-1, vendor P/N 1400-3, and serial number 302). The secondary tank pressure decayed from 4955 psig to 4930 psig in approximately 45 minutes. When the downstream side of the regulator was finally pressurized, the relief valve opened at 160 psig (specification is 135 ± 15 psig). Reseating of the valve occurred at 146 psig. During this operation, the secondary tank lost 5 psi. Several more checks of the relief valve were made, and values within tolerance were observed. Subsequent to testing the relief valve, the leak from the secondary system was still in evidence, although there was no demand made on the regulator, that is, the downstream side was "dead-ended." The leak continued as long as there was any pressure on the upstream side of the regulator. For example, the secondary tank lost 128 psi in 25 minutes. The leak was audible, and it was observed to be coming from the ambient sensing port of the secondary pressure reducer. It was speculated at that time that the O-ring on the regulator shaft might be damaged.

On March 10, 1962, the secondary regulator was removed from the spacecraft. On March 13, 1962, the reducer was disassembled at Cape Kennedy by a subcontractor representative, in the presence of MSC Preflight Operations, MSC Crew Systems Division, and McDonnell personnel. Bits of O-ring material were found in the screwdriver slot of the regulator stem (fig. 4) and were also found impacted against the sintered filter which covered the ambient sensing port. The O-ring itself showed a large gouged-out area, and the flourocarbon backup ring also showed damage (figs. 5 and 6). The damaged surfaces of these parts were on the inner peripheries.

The valve was hand-carried to the subcontractor's plant for investigation and analysis.

At the subcontractor's plant, the valve was inspected, and a new O-ring was installed. Cycling at 7500 psig caused no failure, and no leaks could be found. Back pressurizing the valve until the relief valve poppet lifted did not result in any leakage from the stem. The original parts were replaced in the valve, and the valve was later returned to Cape Kennedy for replacement in Spacecraft 13.

In summary, the results of the investigation were as follows.

1. The cause of the secondary pressure decay during MA-6 was not known.

2. The evidence indicated that the secondary pressure reducer was damaged during postflight testing, and explanations for this failure were unsatisfactory insofar as they were unsupported by experiment and they failed to explain the pressure decay observed during flight. A test program investigating oxygen penetration of elastomers at high pressure was conducted by the subcontractor.

The following were the modifications on all high-pressure regulators.

1. The 0.015-inch-thick Teflon backup ring was replaced by a 0.049-inch-thick Kel-F backup ring.

2. Tolerances between the inside diameter of the backup ring and the outside diameter of the poppet stem were controlled to 0.001-inch-maximum diametral gap.

RELIABILITY TEST PROGRAM

The reliability requirements for the ECS were that it have a mean-time-between-component failure of 500 hours. Failure was defined as the malfunction of any single component without regard for any redundancy which the system provides. Mean-time-between-component failure was defined as the total number of equipment-operating hours divided by the total number of failures occurring during the test period.

Two complete systems were tested, one for 36 mission cycles and the other for 18 mission cycles. A mission test cycle was defined as 28 hours. All reliability tests were conducted at 5 psia in an oxygen atmosphere and with simulated flight temperature profiles. During the course of the reliability test program, the subcontractor was required to report any failures and/or malfunctions to MAC, including an analysis as to the cause of the failure. The reliability test program is summarized in table I.

Test Program

Two complete environmental control systems were used in the reliability test program, one for a complete systems test and the second for subsystem and component tests.

The complete systems reliability test consisted of 18 simulated 28-hour missions. The first 10 mission cycles were under the normal operation mode. The remaining eight mission cycles were conducted under various emergency modes with simulated failures. These system tests were conducted with an ECS installed in a Mercury-like capsule which had the same internal geometry and volume as a production capsule. The capsule was installed in an altitude chamber for the tests. A crewman simulator was substituted for the man/pressure-suit combination. The simulator supplied CO_2, heat, and water vapor at the design metabolic load and removed a like quantity of oxygen to load the system (400 cc CO_2/min, 400 Btu/hr, 500 cc O_2/min). A simulated capsule heat load was also provided.

The actual launch operations of purge, pressure-suit-system leakage tests, Freon cooling, et cetera, were followed for each mission cycle test. The altitude chamber was evacuated to simulate the launch operation. The system was then tested in normal orbital operation mode for 28 hours. System data were sampled every 30 minutes throughout the test. At the conclusion of the orbit test, reentry conditions were simulated with both temperature and pressure profiles provided. The cabin pressure relief valve, snorkel valve, and demand regulator functions were carefully evaluated during this phase of the test. Postlanding ventilation was evaluated for 12 hours during the first of the 10 normal mission tests and for 1 hour during the remaining 8 tests.

Emergency Systems Test

The prelaunch and launch procedures were performed as in the normal mission tests. The system was operated for 24 hours in the normal mode of operation, and each of the following emergencies were evaluated.

1. Manual cabin decompression was remotely simulated. The shutoff feature of the cabin pressure regulator was checked, and the maintenance of suit pressure at 4.6 psi was verified. This test was concluded by evaluating the manual repressurization features of the cabin pressure control valve.

2. Simulated failure of the primary suit compressor and of the backup compressor actuation was evaluated.

3. Emergency-oxygen-rate mode was tested for the final 2 hours of one mission cycle test. Automatic actuation of this emergency mode was tested.

4. Postlanding tests were conducted for 1 hour for each of the eight emergency mode tests.

The total number of complete system test hours was as follows.

Normal mode	488 hours
Emergency modes	16 hours
Postlanding	29 hours
Total	533 hours

Subsystem and Component Reliability Tests

The following subsystem or component tests were conducted.

1. The cabin pressure relief valve was subjected to 36 simulated mission cycles. Twenty normal mission cycles were conducted with a simulated cabin volume of 55 cubic feet.

a. Ground pressure tests were simulated at 5 psig, and leakage measurements were made. The valve was cycled through the complete mission profile of ascent and descent.

b. The cabin relief valve was cycled through 16 emergency missions. Eight were conducted with each of the control aneroids failed.

2. The postlanding outflow subsystem was subjected to 36 normal and 8 emergency mission cycles. The automatic actuation was failed, and manual actuation was checked.

3. The postlanding ventilation inflow system (inflow valve, snorkel valve, and negative-pressure relief valve) was tested under 28 normal mission cycles. Emergency actuation of the inflow-valve manual actuation was tested for eight cycles. The subsystem was then tested to show proper functioning of the negative-pressure relief valve for 36 mission cycles.

4. One primary oxygen supply subsystem was tested under 36 simulated mission cycles, 20 normal and 16 emergency. Another primary oxygen subsystem was tested under 10 normal and 8 emergency mission cycles. (The oxygen supply subsystem was comprised of the oxygen tank, filler valve, reducer, transducer, shutoff valve, check valve, emergency oxygen rate valve, cabin pressure control, and demand regulator.) The components were assembled as in the capsule installation. The high-pressure components were immersed in a water tank to allow evaluation of the leakage throughout the tests.

Normal Mission Cycles

The first five of the normal mission cycles were conducted for 28 hours. The oxygen tanks were filled to 7500 psig, and normal prelaunch procedures were followed. A 500 cc/min demand was placed on both the demand and cabin pressure regulators, and this flow was maintained for 28 hours. The oxygen-regulator-control pressure was measured every 30 minutes, as were the control functions of the demand regulator and cabin pressure control valve. Performance of the oxygen-supply pressure transducer was recorded throughout the tests. The remainder of the normal missions were conducted at an accelerated rate; however, the same number of cycles were performed on each component.

Emergency Mission Cycles

Various emergency mission cycles were conducted on the following 10 subsystems.

1. Emergency pressurization and breathing subsystem
2. Water-separator subsystem (36 missions)
3. Normal suit compressor (36 mission cycles, 30 hr/cycle)
4. Standby suit compressor and heat exchanger
5. Coolant water system (36 mission cycles, 28 hours)
6. Emergency shutoff valve (36 cycles)
7. Solids trap (72 mission cycles, simulated failure)
8. CO_2 and odor absorber
9. Pressure-suit relief valve (200 cycles)
10. Cabin fan and heat exchanger (36 mission cycles, 30 hr/cycle)

QUALIFICATION TEST PROGRAM

The ECS qualification requirements to be met by the subcontractor were specified in the McDonnell Aircraft Corporation Specification Control Drawing 45-83700. All testing was conducted in accordance with Military Environmental-Test Specification MIL-E-5272A, except for certain, more stringent requirements. All tests were conducted in an oxygen environment, and component parts were qualified collectively and/or individually as subsystems. The environmental test requirements were summarized as follows.

1. High temperature: 200° F

2. Low temperature: -20° F

3. Humidity: 15 to 100 percent

4. Salt spray: Protected or 50-hours exposure to salt-sea atmosphere

5. Vibration: Mercury spectrum

6. Sand and dust: Protected or 50-hours exposure

7. Acceleration: Mercury spectrum

8. Combined temperature-altitude tests

9. Salt-water ingestion: One pint of salt water introduced at the inlet of the suit-circuit compressor; compressor operated for an additional 12 hours

10. Acoustic noise: 135 dB overall distributed in the 75- to 150-, 150- to 300-, and 300- to 600-cps octaves

11. Mechanical shock: 15g for 11 milliseconds in all axes and directions; 100g shock for 11 milliseconds

12. Disassembly and inspection

13. Proof- and burst-pressure tests

Performance was tested upon completion of qualification tests. Government inspection was conducted by a resident inspector. Test results were submitted for approval to MAC and to NASA.

Five complete systems and an additional 10 sets (30 tanks) of oxygen supplies were used in the qualification test program. Two systems were used for burst-pressure tests; one system for high- and low-temperature and combined high temperature-altitude tests; one system for acoustic noise, vibration, acceleration, mechanical shock, and 100g shock testing; and one system for environmental tests, including humidity, salt spray and dust, and salt-water ingestion. Two of the five qualification systems were employed in systems tests, and the remaining three systems were used in component-level qualification tests.

Component and Systems Tests

Each component in the system was qualified on an individual basis, and testing was conducted in a fixture which used actual spacecraft mounting brackets.

Complete systems tests were conducted for normal and emergency modes, which included vibration, acceleration, and acoustic noise levels for the launch and reentry profiles. A complete system was tested for a mechanical shock of 15g for 11 milliseconds and for an impact shock of 100g for 11 milliseconds in the expected directions of spacecraft impact. The suit compressor was operated throughout the acceleration and shock testing, and the oxygen tanks were charged to 7500 psig. Following these systems tests, component and system performance tests were conducted to verify system operation.

Qualification of 30 Oxygen Tanks

1. Eight oxygen tanks were oxygen-aged at 8500 psig for a period of 150 hours; then, each tank was subjected to 10 000 hydrostatic pressure cycles (50 to 9000 psi in 1 minute) or to failure. Each tank that successfully completed the pressure-cycle test was pressurized hydrostatically to burst (burst pressure, 16 700 psig; proof pressure, 12 500 psig at 70° F).

2. Seven oxygen tanks were oxygen-aged for 1000 hours and then pressurized hydrostatically to burst.

3. Fifteen tanks were oxygen-aged for 1000 hours and then subjected to 10 000 hydrostatic pressure cycles or to failure. Each tank that successfully completed the pressure cycle test was pressurized hydrostatically to burst.

VENDOR MANNED DEVELOPMENT TEST PROGRAM

In November 1959, manned system tests were conducted at the subcontractor's plant. The tests lasted 1 week, and an MAC test pilot using a Mercury-type pressure suit served as the subject. The purpose of the tests was to evaluate a complete ECS under the following modes of operation: prelaunch purge, launch, normal orbit, emergency-rate oxygen flow, decompressed cabin, descent, and postlanding.

The system was subjected to altitude simulation only; launch and reentry heating were not simulated. A capsule mockup with an ECS was installed in an altitude chamber. The ECS was instrumented with several temperature and pressure sensors in addition to the normal system instrumentation. This additional instrumentation provided data on the pressure drop across the components in the pressure-suit system and provided valuable thermal data on the system performance. The subcontractor's medical monitor and a physician from the Lovelace Clinic were in attendance for these tests and participated in the post-test analysis. The tests were established as design evaluations and were not qualification-type tests. The tests produced the following observations.

1. A 5-minute purge produced 98-percent oxygen in the suit circuit.

2. During a 5000-ft/min ascent, the suit pressurized to 0.25 psi because of the exhaust pressure drop across the demand regulator.

3. Normal orbit mode at 27 000 feet simulated altitude showed the following.

a. Compressor flow was within specification requirements.

b. Pressure rise across the compressor was 5.0 inches of water; the differential-pressure switch for the compressor should be increased to 7.5 inches of water.

c. More than a 10-minute lag existed between the heat-exchanger water valve adjustment and the suit comfort change.

d. Suit comfort was satisfactory.

e. Carbon dioxide content in the suit stayed below 0.15 percent.

4. Emergency-rate flow of 0.033 lb/min provided marginal cooling. As a result, the flow was increased to 0.051 lb/min.

These observations are examples of the type of information obtained from the tests. Several significant system changes resulted from the tests. The system was used for approximately 6 days of testing, averaging 4 hours a day, for a total operating length of 24 hours. No component malfunctions occurred, and the system performed satisfactorily.

Additional manned tests were conducted at the subcontractor's laboratory on March 18, 1960. The purpose of these tests was to determine the dynamic interaction of the suit pressure regulator, cabin pressure control valve, and inhabited pressure suit. This test resulted from the system change which ducted the cabin-leakage-makeup flow of the cabin pressure control valve into the suit circuit. The change served two purposes, which were to insure a sufficient oxygen partial pressure within the suit circuit and to increase mobility within the suit by elevating the differential pressure within the suit circuit so that the relief diaphragm of the suit pressure regulator maintained the suit circuit at a minimum of 2 inches of water above cabin pressure whenever the cabin pressure control valve operated to replenish cabin leakage.

A series of 10 tests was conducted in this program, and all results were satisfactory. The change was therefore incorporated in all Mercury spacecraft environmental systems beginning with Spacecraft 5.

PRIME CONTRACTOR MANNED DEVELOPMENT TEST PROGRAM

A test program consisting of 12 manned runs was conducted (Test Request 45-092) by MAC to evaluate the ECS performance under all modes of operation, including realistic time periods in the various modes. Another objective of this program was the evaluation of the ECS and of the biomedical instrumentation system. Actual spacecraft hardware was employed to commutate, mix, discriminate, and decommutate the telemetry signals, the only deviation being simulation of the air-to-ground transmission by the use of hardline.

The test program results are presented in reference 8. Table II outlines the objectives, durations, and dates of tests for the program. The test vessel was installed in an altitude chamber with the necessary support equipment adjacent and external to the chamber. Several equipment problems were encountered during the early test of the McDonnell program, but these were predominantly associated with supporting equipment rather than with the flight-configuration hardware under evaluation. The most significant ECS problem concerned the dynamic instability of the relief diaphragm of the suit pressure regulator while operating in the emergency mode with a depressurized cabin. This serious malfunction was discovered during test run number 6; and the nature of the problem, including corrective measures taken in the design of the regulator, are discussed in the section of this report concerning the suit pressure regulator.

Upon completion of the McDonnell program in October 1960, the equipment was disassembled and shipped to the Air Crew Equipment Laboratory, a facility of the Naval Air Material Center. There, it was installed in an altitude chamber and used by NASA, with the support of the Air Crew Equipment Laboratory personnel, for a program of ECS training for the astronauts.

VENDOR PRESHIPMENT ACCEPTANCE TESTS

Each production component of the ECS was subjected to inspection and acceptance tests prior to shipment to MAC. After visual inspection, each component underwent functional tests to verify that the unit met specification values for acceptable internal and external leakage, control pressures, flow rates, power requirements, and similar characteristics. A copy of the test data sheet was shipped with the component. These acceptance tests were conducted in accordance with test procedures specified in a document approved by MAC and NASA, and the tests were conducted under the surveillance of the Air Force until May 1962. After this date and until program termination, inspection was accomplished by a MAC resident engineer, in addition to the Air Force inspector acting for NASA. The level of testing is illustrated by the following acceptance-test procedure for the suit-circuit compressor, extracted verbatim from the previously mentioned document.

Suit circuit compressor (45-83700-49). - Each production 207970 suit circuit compressor shall be subjected to the tests described below.

"The compressor shall be subjected to a proof pressure test. The compressor shall be pressurized to 10. 5 psig and immersed in water for a period of 2 minutes. There shall be no evidence of leakage (bubbles) or permanent deformation as a result of the applied pressure.

"A break-in test shall be performed by operating the compressor for a period of 15 minutes at laboratory ambient conditions. The electrical power supply to the compressor motor shall be 115-volt, single-phase, 400-cycle power. A 0. 65-microfarad capacitor shall be connected across receptacle pins A and B during this test.

"While the compressor is warm as a result of the break-in test, it shall be subjected to, and withstand, an ac voltage of 1250 volts (rms) at 60 cps applied between the individual windings and between each winding and the frame for a minimum period of

1 minute. For this test, all ground connections and the capacitor shall be disconnected. Any voltage breakdown, arcing, or flashover shall be considered as a compressor failure.

"Finally, the compressor shall be subjected to a performance test. The compressor inlet air density shall be set at 0.0272 lb/ft^3, the pressure rise across the compressor shall be fixed at a minimum value of 10.0 inches of water, and the resulting airflow through the compressor shall be measured and shall be a minimum of 11.44 cubic feet per minute. The compressor input power shall also be measured and shall not exceed 46.0 watts."

PRIME CONTRACTOR ACCEPTANCE TESTS

The ECS components received by MAC at St. Louis were subjected to a visual inspection on arrival. A component log card was prepared at this time, which accompanied the component, and, eventually the spacecraft, to the launching site. The component log card was used to record the number of operating hours accumulated on the component. The card also included a complete history of inspections and rejections by test personnel. The malfunction review record and engineering malfunction report numbers which reflected a test rejection were also listed on the component log card.

After the component was received and satisfactorily inspected, it was subjected to a series of tests in accordance with SEDR 79, "In-Plant Testing, Pre-Installation Acceptance Test for the Environmental Control System." These tests, which varied with the type of component being tested, were similar in nature and detail to the subcontractor's preshipment acceptance tests. A test-data sheet was used to record the results of the preinstallation acceptance testing, and this record was retained at the preinstallation acceptance facility.

SPACECRAFT SYSTEM TESTING

The ECS was assembled on the large pressure bulkhead (figs. 7, 8, and 9) before the bulkhead was spliced to the conical section of the spacecraft. Manufacturing personnel performed a number of functional checks on the ECS to insure both completeness of assembly and pressure integrity before the bulkhead splice was accomplished. After the spacecraft was assembled, the location of the ECS was such that it was not easily accessible for replacement of components.

After all the spacecraft systems had been installed, the spacecraft was released to the capsule systems test team for test and evaluation of the systems prior to final manufacturing changes, government inspection and acceptance, and shipment to the launch site. The ECS portion of the capsule systems test was conducted in two phases, cleanroom ambient tests and altitude-chamber tests.

The cleanroom phase of testing evaluated such items as oxygen-supply leakage rates, ventilation flow rates, and functional tests of manual and automatic controls.

The following list serves to illustrate the level of testing conducted on Spacecraft 7 (MR-3) for this phase.

1. Functional tests of all manual and automatic controls for verification of pull forces, of correct electrical operation, and of automatic actuation

2. Suit-circuit system-shutoff-valve leakage-rate determination

3. Negative-pressure-relief-valve operating characteristics

4. Ventilation-flow-rate test for both suit compressors and emergency-oxygen mode

5. Coolant-quantity-indicating-system leakage test

6. Waterflow-rate test of comfort control valves

7. Heat-exchanger and Freon-lines leakage test

8. Suit-pressure-regulator-actuation test

9. Suit-circuit negative-pressure test

10. Primary-to-secondary oxygen-supply automatic-transfer test

11. Suit-circuit leakage test at 5 psig

12. Suit-circuit relief-valve-actuation pressure test

13. Leakage test of high and intermediate pressure systems for primary and secondary oxygen supplies

The above series of tests was completed in less than 2 days of three-shift testing. In comparison, the corresponding series of tests on Spacecraft 5 and 8 required 20 and 9 days, respectively. In order to clarify, it is further noted that both Spacecraft 5 and 8 preceded Spacecraft 7 through manufacturing and then through capsule-systems-test phases. The increase in the quality of the manufacturing workmanship was readily apparent by comparison, and this increase was believed to be a direct result of the learning process.

After vibration testing on the spacecraft was completed, it was delivered to the altitude chamber where it was subjected to tests which evaluated performance of components which operated as a function of reduced pressures.

A measurement of cabin leakage was the first test conducted on the spacecraft after completion of the vibration testing, and the leakage-rate figure obtained was compared to the leakage rate measured after the spacecraft was first released from manufacturing and then released into capsule systems testing. The altitude-chamber tests of ECS were conducted unmanned, and this phase of testing was usually comprised of two or three separate ascents of the altitude chamber with the spacecraft hatch installed or removed in order to evaluate the various components. The following list serves to illustrate the nature of the altitude-chamber testing conducted on Spacecraft 7 (MR-3) for this phase.

1. Cabin-pressure-relief-valve operating pressures during simulated ascent and descent of the spacecraft

2. Both suit-pressure-regulator operation at reduced pressures and control characteristics under simulated decompressed-cabin conditions

3. Cabin-pressure-control-valve operating characteristics, including points of automatic initiation and cessation of flow

4. Automatic actuation of emergency-oxygen-rate mode under low-suit-pressure conditions

5. Evaluation of prelaunch Freon-cooling system and Freon leak test

The altitude-chamber phase completed the ECS testing conducted by MAC at St. Louis. Certain portions of the cleanroom ambient tests were repeated often to verify system integrity after an ECS change or a modification was made in a manufacturing rework period. The next ECS testing phase in chronological order was conducted at Cape Kennedy.

ENVIRONMENTAL CONTROL SYSTEM TESTING AT CAPE KENNEDY

The detailed ECS subsystem testing conducted at St. Louis was repeated at Cape Kennedy. Here also, the testing was divided into two phases, sea level and altitude. The sea-level tests were accomplished in the ECS test cell, a small cleanroom adjacent to Hangar S. The test procedures for this phase were given in SEDR 80, and the following list of tests, extracted from SEDR 80-18, illustrates the level of testing conducted on Spacecraft 18.

1. CQIS pressure-regulator check

2. CQIS high-pressure leakage test

3. Manual-controls pull-force test

4. Water-tank and Freon-check-valve leakage test

5. Freon-system leakage-rate measurement

6. CQIS low-pressure leakage test

7. Suit-compressor differential-pressure-switch actuation test

8. Suit-circuit leakage-rate determination

9. Suit-pressure-regulator tests

10. Negative-pressure-relief operation of the cabin pressure relief valve

11. Verification of correct rigging for emergency-oxygen-mode components

12. Suit-circuit-shutoff-valve leakage test

13. High-pressure-reducers regulated pressure test

14. Primary-to-secondary oxygen-supply automatic-transfer test

15. Negative-pressure-relief-valve and suit-circuit negative-pressure leakage test

16. Water-separator-operation test

17. ECS controls functional test

18. Suit-compressors and emergency-oxygen-rate-valve flow-rate check

19. CQIS relief-valve-actuation pressure test

20. Comfort-control-valves waterflow-rate check

21. High-pressure-oxygen-supplies leakage test and supply-quantity-transducer calibration

22. CQIS transducer calibration

At a later period during launch preparation, the spacecraft was installed in the altitude chamber located in Hangar S. The reduced-pressure functions of the ECS were verified, associated instrumentation was calibrated, and spacecraft familiarization runs were conducted for the primary pilot of the mission. These operations are described in SEDR 83 which separates the desired tests into four runs as follows.

Run 1: Calibration of instrumentation pressure-sensing monitors (unmanned)

Run 2: Checkout of ECS components at reduced pressure, simulating normal and emergency conditions (unmanned)

Run 3: Manned test simulating actual mission conditions but not simulating mission times

Run 4: Manned test simulating actual mission conditions and time

The altitude testing accomplished in SEDR 83 was the last detailed ECS status evaluation made before flight. However, certain minor tests were conducted after the spacecraft had been mated to the launch vehicle. These were essentially confidence-level tests and included such things as ventilation-flow-rate checks of the suit compressors and functional tests of the ECS controls. The ECS was operated for the launch simulation (SEDR 170) which included all procedures to be employed on the day of launch. Confidence-level tests were also conducted as an integral part of the actual launch countdown (SEDR 103).

During the period between the mating of the spacecraft to the launch vehicle and the actual launch, a precise measurement of cabin leakage rate was made. To accomplish this measurement, the flight hatch was installed and the cabin pressurized with oxygen to 5.00 psig. This pressure was accurately maintained by supplying makeup oxygen through a rotameter until the cabin gas temperature and the rotameter indication had stabilized, which usually required approximately 1 hour. This leakage rate was converted to the rate-of-pressure decay from 3.00 psig for the free-gas volume of the cabin (52 cubic feet). After pilot insertion on the day of launch, the cabin was pressurized to 3.00 psig, and the rate-of-pressure decay was measured for a period of 4 to 8 minutes. If the value obtained was in reasonable agreement with the expected figure, the cabin leakage was assumed to be the same as that measured by the rotameter method prior to launch day. Graphs were prepared in advance to permit an estimate of the leakage rate should it vary appreciably from the expected value. The cabin-leakage-rate figure was recorded and used in conjunction with the decay curve for the oxygen-supply pressure before and during flight to arrive at a figure for the pilot metabolic oxygen consumption.

OPERATION SUPPORT TESTING BY THE MANNED SPACECRAFT CENTER

Upon completion of the pilot-training program at the Air Crew Equipment Laboratory in January 1961, the ECS test vessel was shipped to Langley Field, Virginia, and was employed for sea-level testing since no altitude chamber was available. The vessel was employed first to resolve the problem encountered in the MR-2 mission. The suit-compressor discharge-type check-valve design was modified to be spring-loaded and was installed in the snorkel inflow line, downstream of and adjacent to the ground ventilation inlet valve. The change was incorporated for all subsequent flights.

The test vessel was modified so that the cabin volume could be evacuated directly by a vacuum pump. The test facility was used at Langley Field to investigate the suit-circuit cooling problems encountered during the MA-5 mission and in preparation for the MA-6 mission. The test vessel was moved to Houston, Texas, in February 1962, and operational support testing by the Crew Systems Division at NASA Manned Spacecraft Center was resumed. The manned test time accumulated by Crew Systems Division in support of the Mercury Project Office exceeded 378 hours. A large number of tests were performed both to investigate problems encountered during Project Mercury and to extend the mission capability of the Mercury spacecraft. More important among these tests were the following.

1. The development of the dome temperature as the control parameter for the heat exchanger

2. Evaluation of the effective CO_2 absorption life of the 4.6 and 5.4 LiOH charges for the CO_2 and odor absorber

3. Development of the condensate trap employed in the MA-9 mission

4. Cabin temperature effects on the thermal comfort within the pressure suit

FLIGHT EXPERIENCE

Little Joe 5A (Spacecraft 14)

In the Little Joe 5A mission, the cabin differential pressure was not held on launch, and the pressure decayed to zero psia. A wire was found lodged on the seat of the cabin pressure relief valve. A screen was installed on all subsequent cabin pressure relief valves to prevent foreign objects from entering the valve.

Mercury-Redstone 2 (Spacecraft 5)

During the launch of the Mercury-Redstone 2 mission, the ground ventilation inlet valve vibrated open, causing the cabin to depressurize through the negative-pressure relief valve and causing the suit circuit to command the emergency-rate mode of operation. It was found that the snubber and its mounting bracket, intended to place a frictional load on the mechanical linkage connected to the valve, had been removed prior to flight. To prevent loss of cabin pressure, should this occur in future missions, a second spring-loaded flapper-type check valve was installed in the inflow line adjacent to and downstream of the inlet valve. A bypass switch was installed on the instrument panel which would allow the pilot to reset the emergency-rate valve and to return to the normal mode of system operation by removing the electrical signal from the solenoid on the suit-circuit-shutoff valve.

Upon recovery, water was found in the spacecraft, and the water was believed to have entered through the cabin outflow snorkel. This valve was of the ball-float configuration which was also used on the suit inflow snorkel. As a result, the outflow snorkel was changed to a diaphragm check valve which sealed if there was a hydrostatic head in the recovery compartment. The outflow valve itself was not changed.

Mercury-Redstone 3 (Spacecraft 7)

A postflight chemical analysis of the LiOH canister from the MR-3 mission revealed that a significant quantity of Freon-114 had been adsorbed by the charcoal bed. It was concluded that the Freon had entered the system during the ECS manned altitude-chamber tests. Consequently, procedures were modified to minimize the Freon-114 in the chamber when the ECS was operating in the postlanding mode and when chamber air was being drawn through the CO_2 and odor absorber. No abnormalities occurred in the ECS during the flight.

Mercury-Redstone 4 (Spacecraft 11)

During the manned altitude-chamber ECS test runs for MR-4, the demand-regulator tilt valve opened and caused excessive oxygen usage when the pilot removed the suit outlet hose. The helical spring, which returns the tilt valve to the normal position, had escaped the seat so that the tilt valve hung open. Subsequent valves were made with a retainer on the spring to prevent a hang-open of the tilt valve.

Mercury-Atlas 4 (Spacecraft 8A)

The ECS in MA-4 was connected to a manned simulator. An abnormally high oxygen usage rate was experienced during the mission, which was attributed to a partially opened emergency-rate valve. The emergency-rate handle, held in tension by a spring in Spacecraft 8A, was maintained in place by a detent. The detent was found to be severely worn and required very little force to cause entrance in the emergency-rate mode of operation. The vibration experienced in the cabin during launch could have supplied the necessary actuating force. The emergency-rate handle was redesigned to incorporate a solenoid-locking feature on the handle. This lock was incorporated on the MA-5 and MA-6 missions. The valve was also modified so that actuation of the internal microswitch occurred as the first function of the valve, thus assuring electrical continuity with the suit-circuit-shutoff valve, the suit compressor, and the indicating light before oxygen flow commenced.

Mercury-Atlas 5 (Spacecraft 9)

The comfort control valves on the MA-5 mission, with a primate aboard, were set prior to launch. The primate-couch inlet temperature rose during the mission, which contributed to the decision to reenter after two orbits. The temperature rise was not considered a malfunction of the ECS, but was attributed to the control valves not being set at a high enough flow. It was concluded that the pilot could adjust the valves in the manned missions, and no further action was taken. (See appendix C for ECS performance during missions MA-5 to MA-9.)

Mercury-Atlas 6 (Spacecraft 13)

The oxygen to the suit circuit for the MA-6 mission was delivered by a constant-bleed orifice rather than by the demand regulator as on all other spacecraft. This was done to insure mission completion. The secondary oxygen supply indicated an unexpected decay during the mission. However, postflight testing did not reveal a leak in the system.

The suit-inlet temperature was higher than desirable during the mission. The heat-exchanger exhaust-duct temperature (the cooling-system control parameter) was also high; and the cooling-water flow rate, an average of 1.7 lb/hr, was greater than required to remove the expected heat load of the pilot (approximately 0.8 lb/hr would be required). Postflight inspection did not indicate any cooling-system malfunctions.

The check valve between the Freon-114 and water inlets to the inverter coldplates was inoperative, preventing water from entering the coldplates, and this resulted in increasing inverter temperatures during the flight. Postflight testing indicated this check valve was stuck in the closed position, and a pressure of 18.5 psig was required to open the valve. The valve specification requires flow with a differential pressure of 1.0 psig. A series of check-valve design changes were made, and no further difficulty was encountered during flights although a high rejection rate of this valve continued during systems tests.

Mercury-Atlas 7 (Spacecraft 18)

The CQIS during the MA-7 mission did not contain a spacecraft readout, but the quantity was measured and transmitted by telemetry. The CQIS was removed from subsequent spacecraft. This system was based upon the measurement of the pressure decay from a small oxygen tank used to pressurize the coolant-water tank. However, the effect of the cycling cabin temperature rendered the CQIS useless for accurate monitoring of the coolant quantity remaining.

Suit cooling presented the major problem, and the results were similar to those in the MA-6 mission. The Crew Systems Division of the NASA Manned Spacecraft Center had begun to investigate the cooling system as a result of the MA-6 cooling-system performance, but the results were not ready until after the completion of MA-7. These tests revealed that the heat-exchanger duct temperature was not a responsive control parameter and that the temperature of the steam in the heat-exchanger interpass or dome area was the most desirable monitoring parameter.

The suit-circuit-shutoff valve was redesigned for MA-7 and for subsequent missions to eliminate the tension spring and to incorporate a hard linkage between the system-shutoff valve and the emergency-rate valve. A second 20° deadband was built into the emergency-rate valve so that the actuating shaft would turn 20° at the end of its travel after an internal shuttle valve had reseated. This change was necessitated by the mechanics of the valve, which prevented the necessary balance of pressures required (under the design tolerances on some valves) to end emergency flow manually.

Mercury-Atlas 8 (Spacecraft 16)

The change of the cooling-system control parameter to the dome temperature was employed on the MA-8 mission and proved to be highly successful. The pilot maintained the suit-inlet temperature below 70° F during most of the mission. During the first 2 hours, however, the pilot had to continually increase the coolant flow setting and did not achieve adequate cooling until approximately 2 hours after launch. Postflight inspection revealed that lubricant particles had practically obstructed the CCV, but that the flow at the range of settings finally reached was in agreement with those expected from theoretical calculations and compared favorably with the prelaunch setting of the CCV. A change in lubricating procedures was instituted for subsequent valves.

The cabin heat-exchanger gas-outlet temperature was measured to determine the efficiency of the cooling system and was 40° to 45° F during the flight, indicating maximum-heat-removal operation of the exchanger.

Mercury-Atlas 9 (Spacecraft 20)

The changes in the high-pressure regulator, discussed in the section on the high-pressure reducer, were incorporated on the ECS of Spacecraft 20. Also, the high-pressure-oxygen-supply-shutoff valves were relocated downstream of the pressure reducers.

At approximately 6 hours 22 minutes ground elapsed time, the pilot turned off the cabin coolant water and fan, according to the flight plan, to evaluate the cabin cooling circuit. Concurrently, the electrical load was reduced. A temperature probe was located at the cabin heat-exchanger outlet to help describe the efficiency of the cooling circuit. During periods of maximum use of cooling on Spacecraft 16 (MA-8), this temperature was 40° to 45° F while the average cabin temperature indicated 90° to 105° F. The heat-exchanger outlet temperature served to indicate cabin temperature in the area where the ECS was located during the electrical power-down period of MA-9 with the cooling turned off. This temperature increased to within 10° to 15° F below the indicated cabin temperature. The cabin temperature maintained a range of values which was experienced on previous missions with cabin cooling and electrical power-up. During much of the MA-9 flight, the cabin temperature cycled between 90° and 95° F. It was concluded that cabin cooling in the Project Mercury spacecraft was not required during the electrical powered-down condition. Cabin cooling was resumed at 32 hours 28 minutes ground elapsed time.

Ground testing of the suit cooling system showed that the condensate was not carried by the gas stream to the sponge separator, after being condensed from the gas stream in the heat exchanger. Rather, it was held to the metal heat-transfer surface by surface tension and flowed from the heat-exchanger fins to the heat-exchanger case, and then to the duct walls, thus bypassing the sponge separator. An inline water trap was designed and installed in the MA-9 ECS, downstream of the sponge separator and integral with the suit-inlet hose fitting. The pilot observed condensate flowing from the trap, indicating separation of the condensate which had passed by the sponge and verifying the theory that liquids will adhere to the walls of a container in a weightless environment.

Although the suit-inlet temperatures were the lowest of any Mercury mission (60° to 65° F during most of the flight), the pilot adjusted the CCV excessively to maintain the dome temperature of the suit heat exchanger at the recommended control temperature of 55° F. Postflight inspection did not reveal any system malfunction or abnormalities. It was concluded that the excessive CCV adjustment was due to the fluctuations in heat load to the heat exchanger. These were more pronounced during this mission than in previous missions and were influenced by open visor operation, cabin temperature, and astronaut level of activity.

The CO_2 level at the suit inlet began to increase during the last 2 hours of the mission, although the CO_2 partial pressure never reached a level which would endanger the pilot. It was determined, postflight, that channeling of the CO_2 and odor absorber in the LiOH bed had occurred (appendix A).

CONCLUDING REMARKS

The objectives of Project Mercury were (1) to place a manned spacecraft in orbital flight around the earth, (2) to investigate man's performance capabilities and ability to function in a space environment, and (3) to recover the man and the spacecraft safely. These objectives were successfully accomplished in 5 years, in which millions of people from many major government agencies and from much of the aerospace industry combined their skills, initiative, and experience into a national effort. In this period, six manned space flights were accomplished as part of a 25-flight program.

The successful completion of the project involved the following important guidelines:

1. Existing technology and off-the-shelf equipment should be used when possible.

2. The simplest and most reliable approach to system design would be followed.

3. A progressive and logical test program would be conducted.

The Mercury flight schedule planned early in 1959 included major scheduled flight tests involving rocket-propelled full-scale spacecraft, including boilerplate and production types, and showing 27 major launchings. There were three primary types of tests included, (1) research-and-development tests, (2) flight-qualification of the production spacecraft, and (3) the manned suborbital and orbital flight tests.

The environmental control system development started in January 1959 with the establishment of the system design requirements. The design concepts were changed very little after the initial design, and testing began after the initial design was completed and the first prototype components were built.

Simultaneous programs of preinstallation acceptance testing and system qualification and reliability testing were started. The first complete environmental control systems were scheduled for exhaustive performance and operating — characteristics study, and interface between components was studied. The preinstallation acceptance testing program was continued and conducted on all new components scheduled for spacecraft installation or use as replacement parts.

After unmanned testing had demonstrated the system to be a safe one, manned tests were started. The manned test programs were conducted in three phases to permit each responsible agency to be satisfied that the system and testing programs were valid. First, the vendor ran a series of tests on the environmental system to assure that the system being built was satisfactory. Next, the spacecraft contractor ran a series of manned tests for familiarization with environmental systems operation and, in addition, added the cabin and suit ventilation heat loads to study the total spacecraft. The final series of manned tests were performed on the flight spacecraft for familiarization with system operation and for introduction of the astronauts to the system. The astronauts had observed the previous vendor and spacecraft builder operations, and the final series of tests with the astronauts provided an opportunity to study the prelaunch and flight handling of the environmental system.

The need for careful and continuing attention to quality and engineering detail in all program phases was one of the most significant lessons learned from Project Mercury. The spacecraft was a complex vehicle made up of many individual systems and components, and only through close monitoring of the design and development of each piece of hardware and of the relationship of each piece of hardware to all other associated components could problems be rapidly recognized and corrected before failure occurred. Many performance problems could not be anticipated because of lack of experience or because of inability to simulate adequately realistic conditions in the early test program. Attention to detail during the design phase resulted in the incorporation of system redundancy, where a direct relationship to mission success existed.

The following is a list of some of the major problems encountered in the environmental control system, along with reasons for the problems and the methods of overcoming the problems.

1. The MA-8 mission experienced a serious valve blockage by contamination, and a parallel coolant control valve was added (for the MA-9 mission) for redundancy with the primary valve in the suit cooling-water circuit.

2. During the MA-9 mission, the astronaut was required to make a large number of minor changes to the suit coolant control valve setting in an attempt to keep the heat-exchanger dome temperature (the cooling system control parameter) within the desired range. Changes in metabolic and external suit-circuit heat loads because of changes in the astronaut's level of activity, open visor operation, solar heat on the spacecraft, and internal spacecraft equipment heating are a normal experience that reflected in the coolant requirements for the suit heat exchanger. It is probable that the sensitivity of this small orifice valve and the astronaut's normally varying metabolic heat loads, could have resulted in the need for frequent coolant-flow adjustment. The suit cooling system exhibited a history of undesirable operation, and some of the problems were elevated suit-inlet temperatures, wet undergarments, and a general lack of astronaut comfort. However, metabolic heat loads were removed sufficiently to keep body temperatures well below a physiologically marginal value. The two causes for the cooling system problems were selection of an improper cooling system control parameter during the initial design period and ineffectiveness of the suit cooling-circuit water separator caused by unpredicted behavior of free liquid in a weightless condition.

3. An inline condensate trap, was designed to remove excess water from the suit-inlet hose and was installed near the entrance point on the suit. According to the flight plan, the astronaut opened a hose clamp on the water outlet line from the trap to activate the condensate trap. The astronaut observed condensate water flowing through the water outlet line, indicating that free water had passed around the sponge in the water separator.

Manned Spacecraft Center
National Aeronautics and Space Administration
Houston, Texas, June 1, 1967
914-50-80-10-72

TABLE I. - RELIABILITY-PROGRAM SUMMARY

	Number of mission cycles	Duration (a)
Complete system		
Normal	10	300 hours
Emergency	8	240 hours
Subsystems[b]		
Emergency-breathing subsystem	36	150+ cycles tests
Water-separator subsystem	36	2196 cycles
Cabin-cooling subsystem	36	1080 hours
Normal suit-compressor subsystem	36	1440 hours
Standby suit-compressor subsystem	36	1440 hours
Instrument control box	36	360 cycles
Postlanding-vent-inflow subsystem	36	36 cycles
Coolant-water-tank subsystem	36	1008 hours
Solids traps	36	72 cycles
Emergency shutoff valve	36	36 cycles
Cabin relief valve	36	36 cycles
Pilot-pressure-suit relief valve	36	7200 cycles
Postlanding-outflow subsystem	36	36 cycles
Launch oxygen subsystem	36	
Normal pressurization and breathing subsystem	54	

[a]Subsystems were tested on a mission-hour basis or for a set number of operating cycles.

[b]There were 20 normal and 16 emergency mission cycles.

TABLE II. - SUMMARY OF MANNED ENVIRONMENTAL CONTROL SYSTEM TEST RUNS

Test run	1960 date	Type of run	Duration, hr (a)
1	June 20	Suit normal, cabin normal, no heat loads	4
2	June 23	Suit normal, cabin normal, equipment heat loads	4
3	June 24	Suit normal, cabin normal and decompressed, equipment heat loads	4
4	June 25	Suit normal, cabin normal and decompressed, equipment heat loads	4
5	June 27	Suit normal and emergency, cabin normal, equipment heat loads	4
6	July 6	Suit normal and emergency, cabin normal and decompressed, equipment heat loads	[b]3-1/4
7	July 14	Suit normal and emergency, cabin normal and decompressed, no heat loads	3-1/2
8	July 20	Suit normal and emergency, cabin normal, wall and equipment heat loads	4
9	July 27 to 28	Suit normal, cabin normal with postlanding phase, wall and equipment heat loads	18-1/2
10	August 5	Prelaunch, normal orbit with postlanding phase; wall and equipment heat loads	7
11	August 16	Prelaunch, normal, emergency, and cabin-decompressed operations during orbit, reentry, and postlanding; wall and equipment heat loads	7
12A	August 19 to 20	Prelaunch, normal orbit; wall and equipment heat loads	[c]8
12B	October 8 to 9	Prelaunch, normal orbit, and reentry	28

[a]Test duration included approximately 1/2-hour purge on runs 1 to 8; 1-hour purge on runs 10 and 11; and 2-hour purge on runs 9, 12A, and 12B.

[b]Test halted because of occupant's physiological reactions.

[c]Scheduled for 28-hour duration and 2-hour purge. Run halted because of low oxygen content (75 percent).

(1) Solids trap
(2) Suit fan no. 1
(3) Suit fan no. 2
(4) Pressure differential switch
(5) Check valve
(6) Suit CO_2 and odor absorber
(7) Suit-circuit heat exchanger
(8) Suit temperature manual control
(9) Coolant tank
(10) Suit water separator
(11) Condensate water
(12) Suit demand pressure regulator
(13) Cabin pressure control valve
(14) Primary O_2 supply
(15) Secondary O_2 supply
(16) Primary O_2 shutoff
(17) Quick disconnect filler
(18) Primary O_2 pressure transducer
(19) Primary O_2 pressure reducer
(20) Secondary O_2 pressure reducer
(21) Primary O_2 check valve
(22) Emergency O_2 rate valve
(23) Suit-circuit shutoff
(24) Cabin fan
(25) Cabin heat exchanger
(26) Cabin pressure relief valve
(27) Air inlet valve
(28) Cabin air outflow valve
(29) Cabin temperature manual control
(30) Inlet snorkel
(31) Outlet snorkel
(32) Vacuum relief
(33) Primary O_2 pressure switch
(34) Ground O_2 inlet
(35) Coolant O_2 supply
(36) Coolant-quantity transducer
(37) Coolant O_2 pressure reducer
(38) Coolant O_2 shutoff
(39) Quick disconnect filler
(40) Suit-pressure transducer
(41) Pressure-relief valve
(42) Secondary O_2 pressure transducer

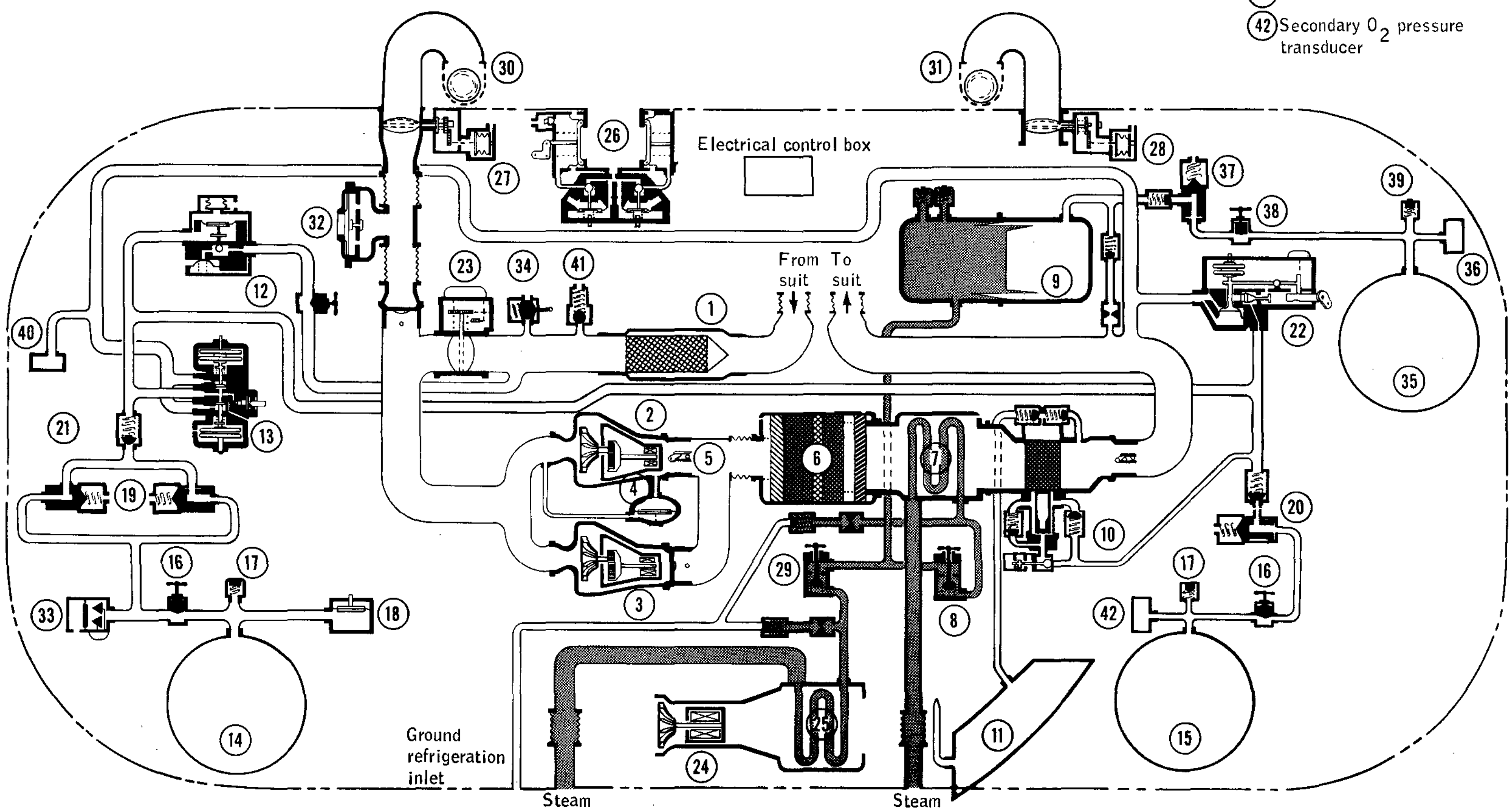

Figure 1. - Schematic diagram of the environmental control system.

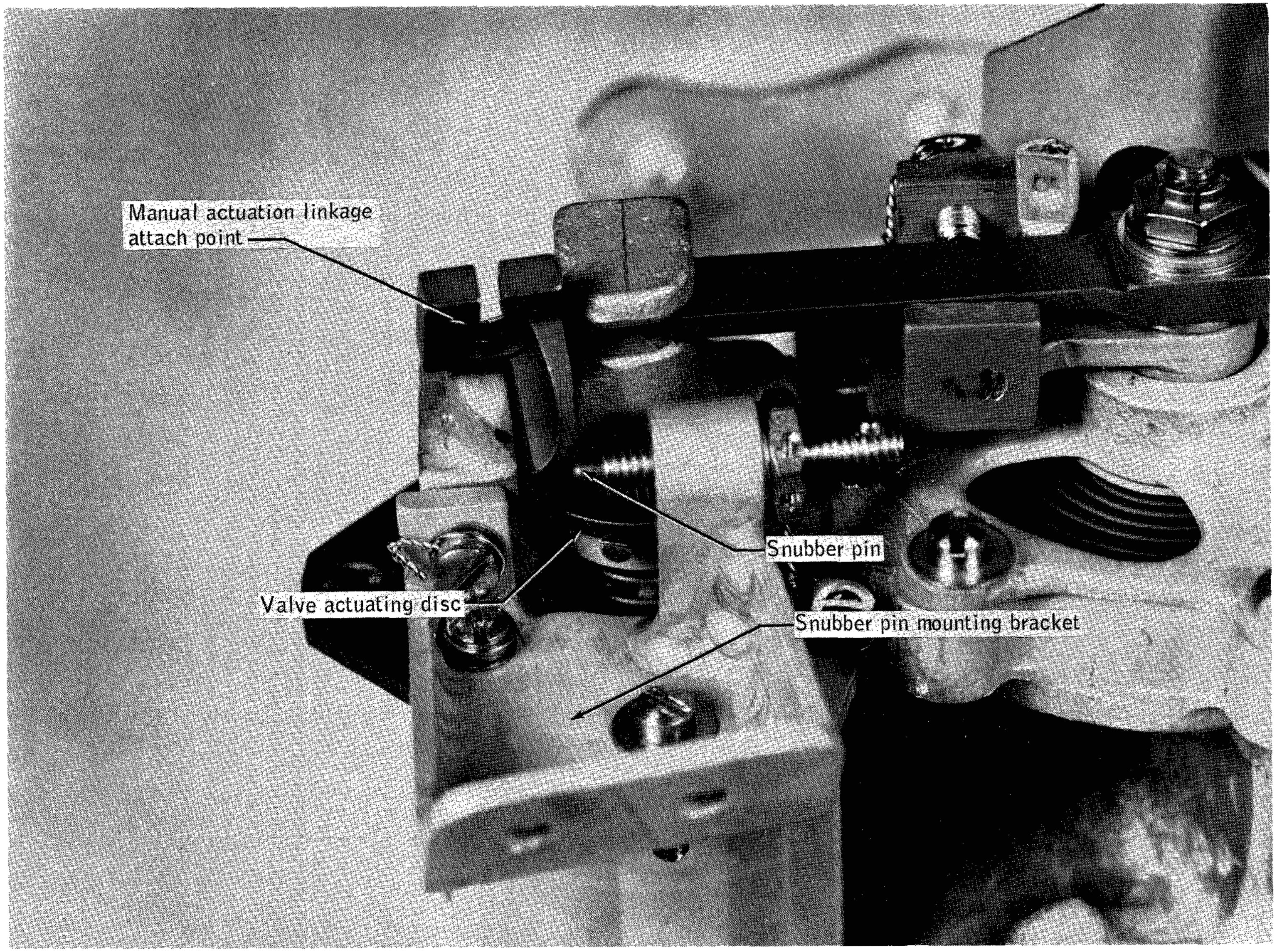

Figure 2. - Ground ventilation inlet valve, MR-2 configuration.

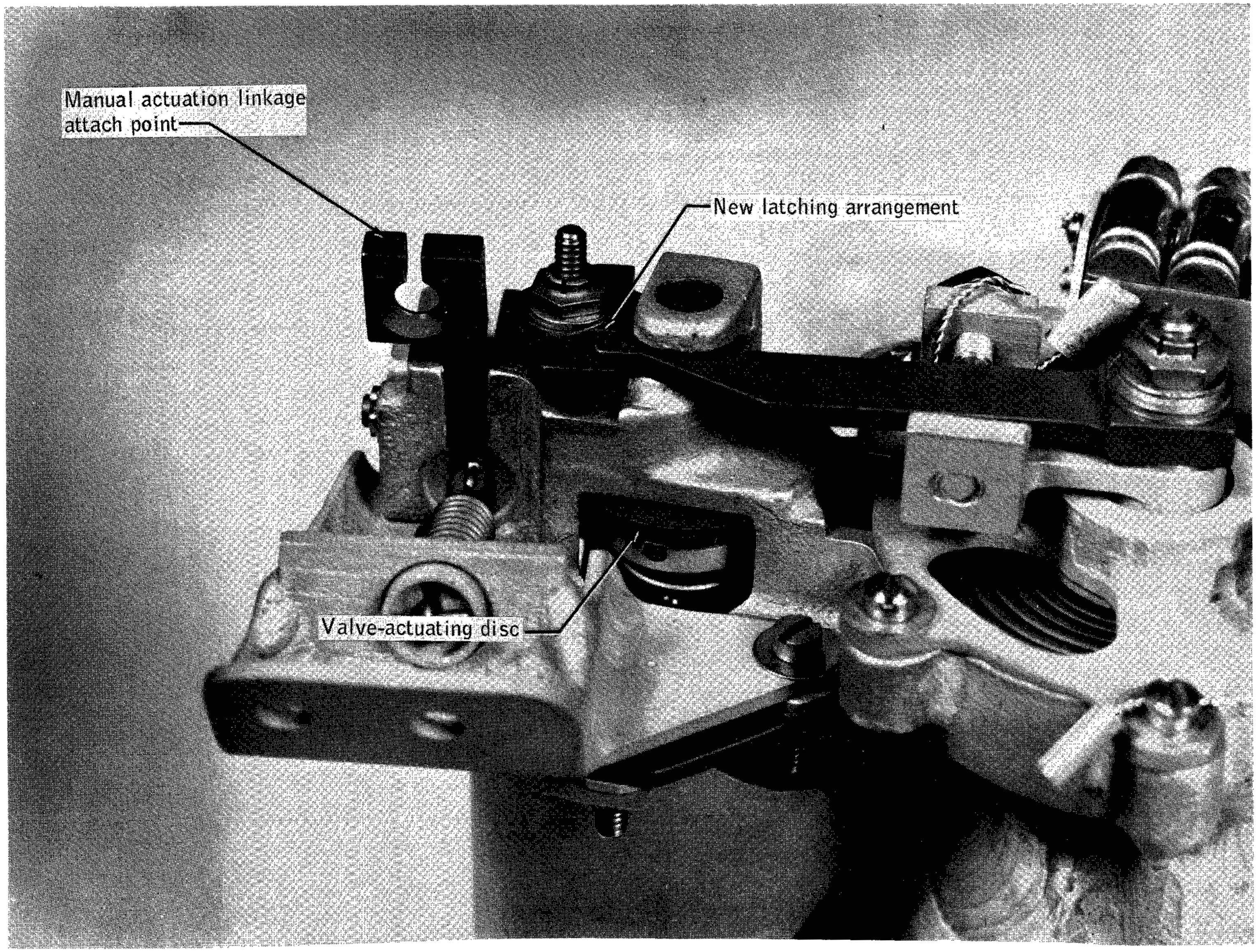

Figure 3. - Ground ventilation inlet valve, modified latching configuration.

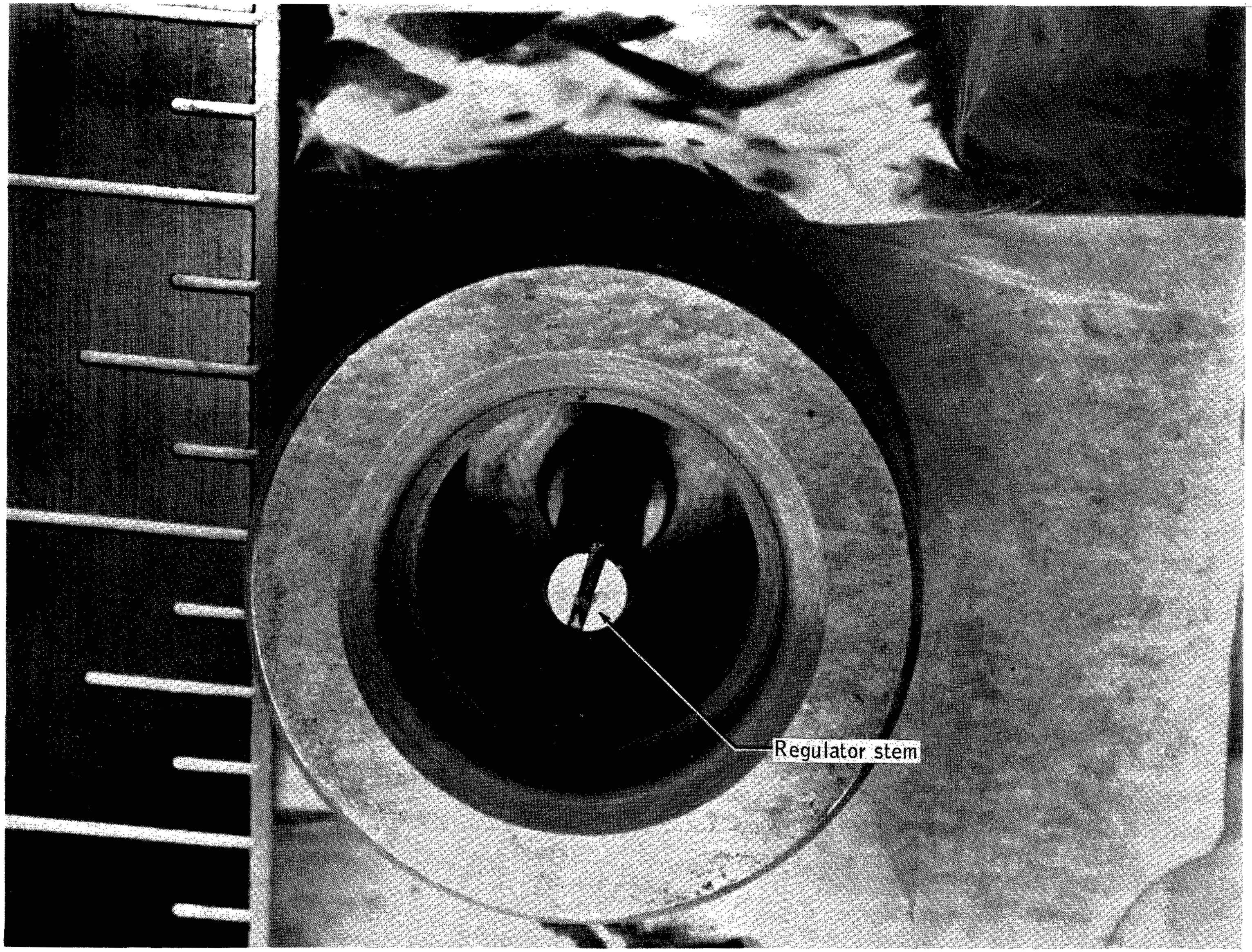

Figure 4. - MA-6 secondary-oxygen-supply high-pressure reducer.

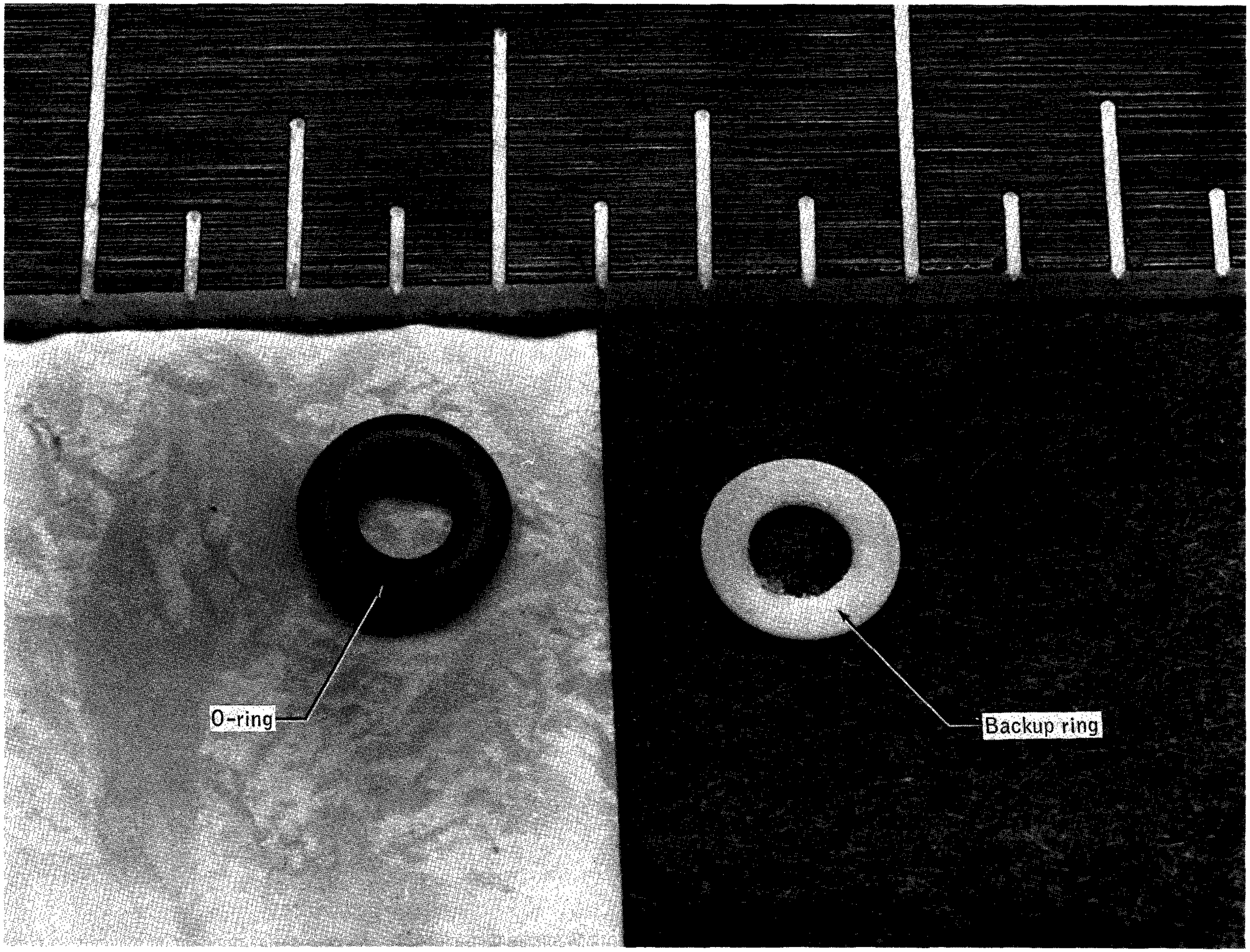

Figure 5. - MA-6 secondary-oxygen-supply high-pressure reducer, outer surfaces of seals.

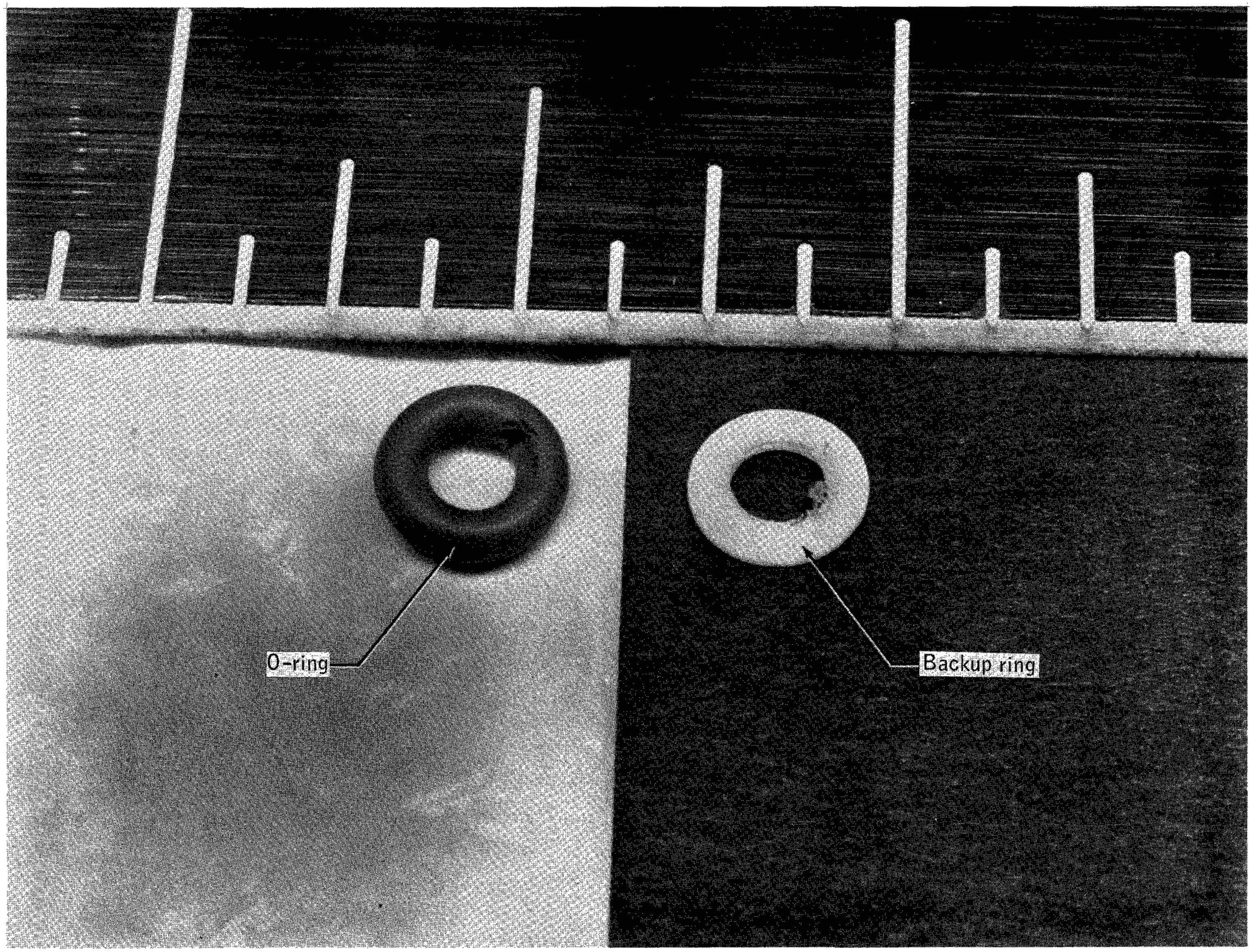

Figure 6. - MA-6 secondary-oxygen-supply high-pressure reducer, inner surfaces of seals.

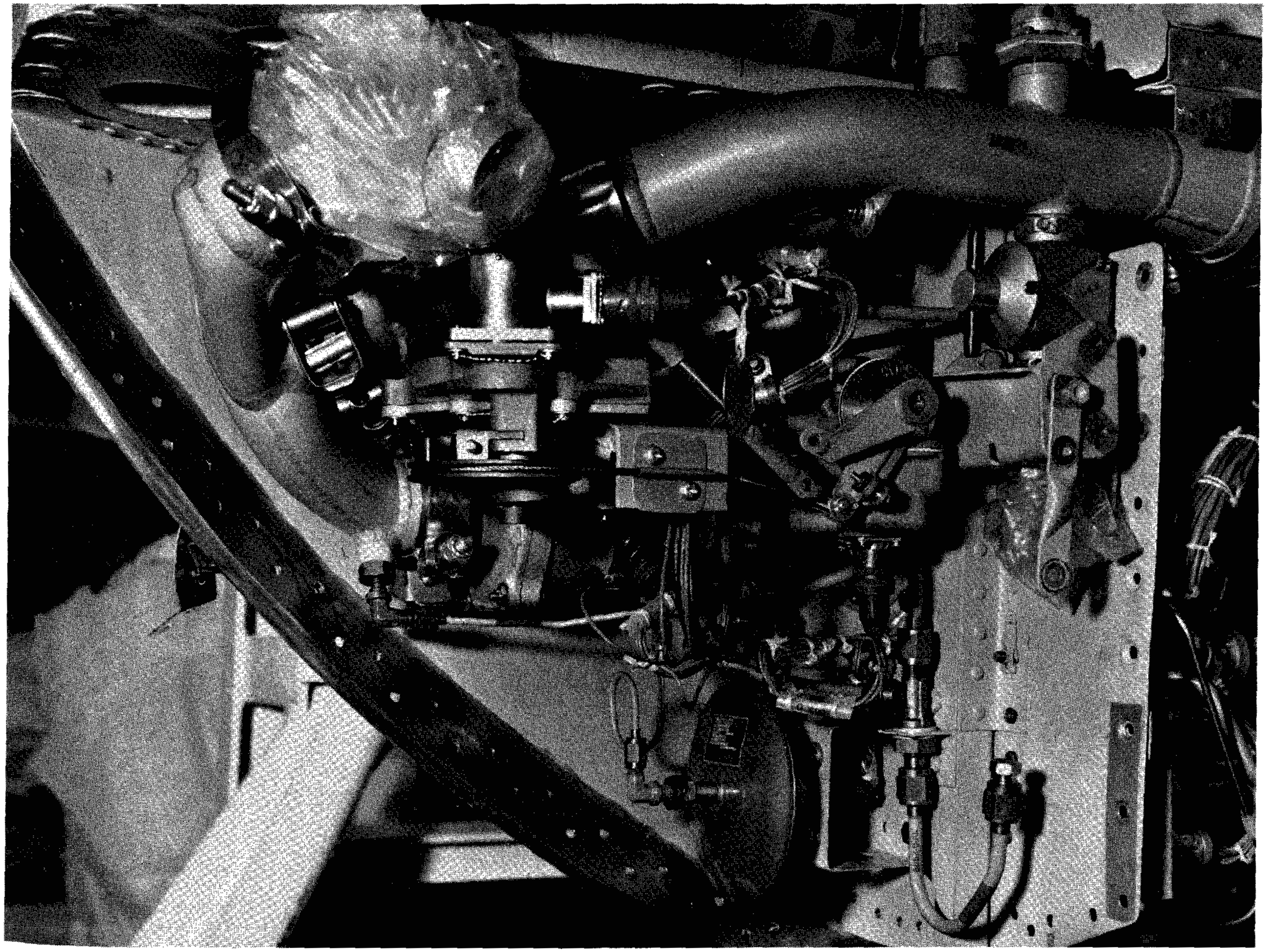

Figure 7. - Spacecraft bulkhead installation of ECS, right side.

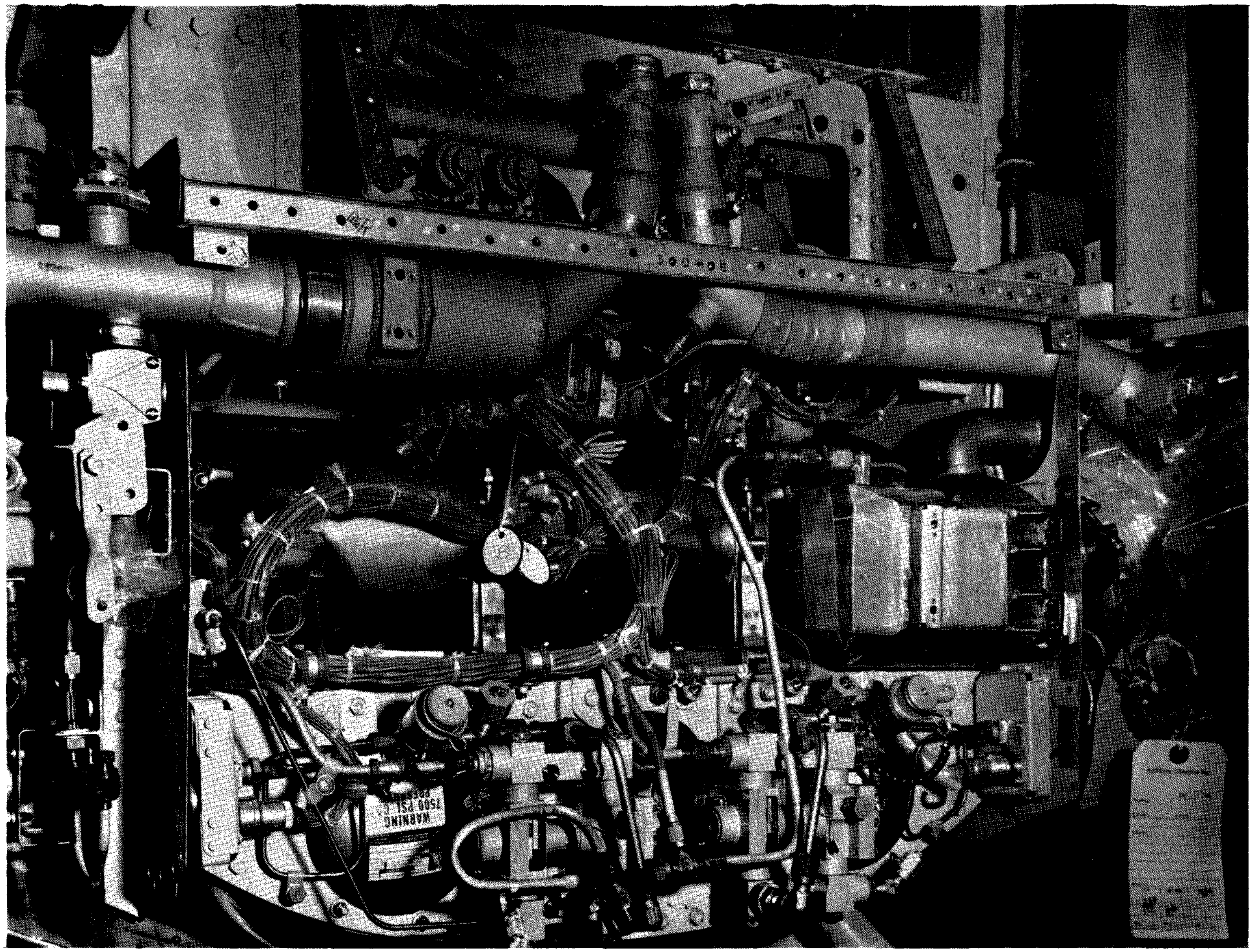

Figure 8. - Spacecraft bulkhead installation of ECS, center.

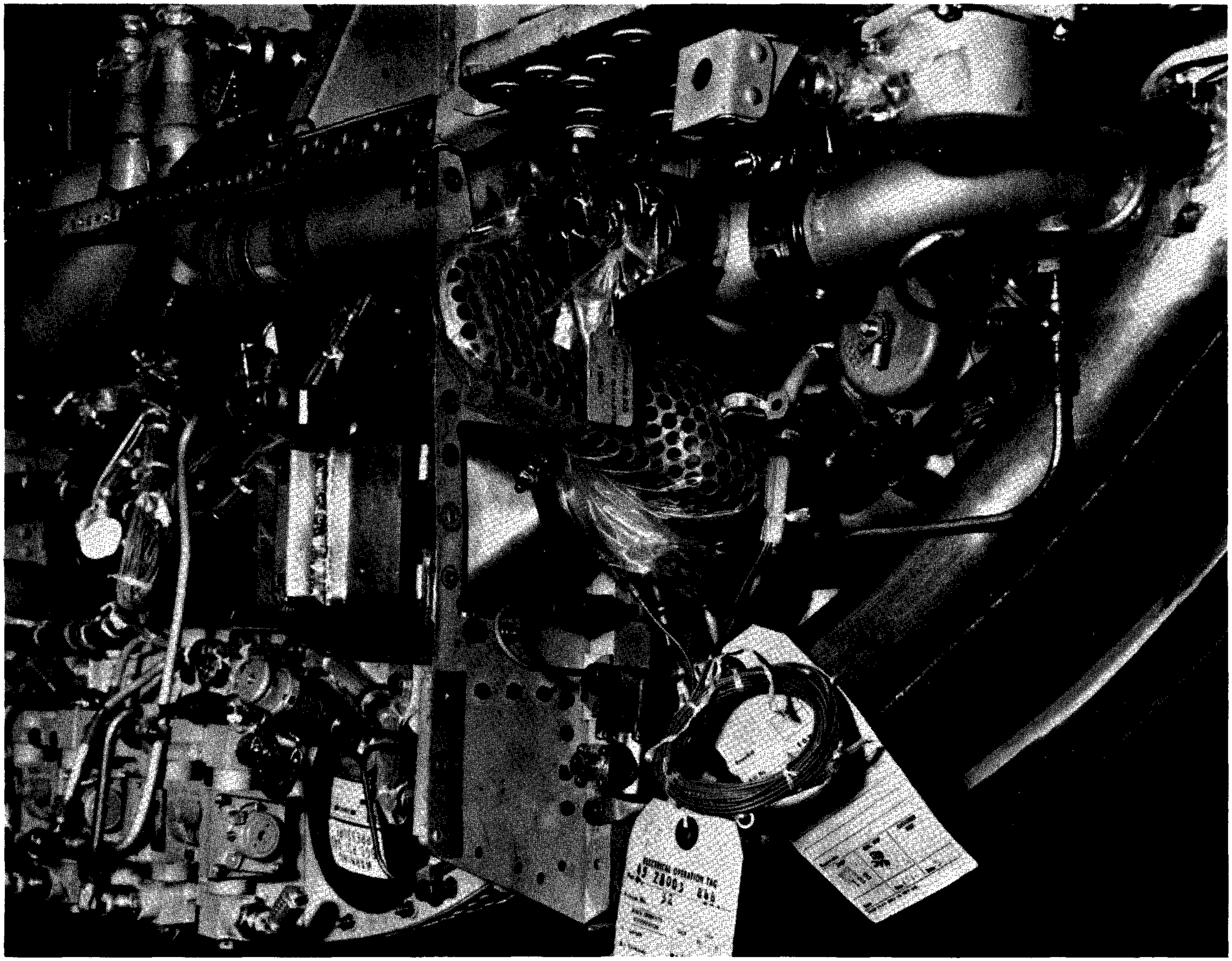

Figure 9. - Spacecraft bulkhead installation of ECS, left side.

APPENDIX A

CHEMICAL ANALYSIS OF MERCURY-ATLAS 9 (MA-9)

LITHIUM HYDROXIDE CANISTER

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Summary

It was determined through the chemical analysis of the MA-9 lithium hydroxide (LiOH) canister that approximately 1000 grams of carbon dioxide $\left(CO_2\right)$ had been absorbed in the upper two LiOH containers (bags 1 and 2). This represented about a 90-percent consumption of LiOH. Approximately 350 grams of CO_2 had been absorbed in the lower two LiOH containers (bags 3 and 4), representing about a 55-percent consumption of LiOH.

It was also determined that there was a definite flow pattern for the CO_2 through the canister. From this determination it was further concluded that this channeling action was the main reason the canister-outlet CO_2 concentration increased during the latter part of the mission.

Test Objectives

The chemical analysis of the MA-9 LiOH canister was made with two main objectives in mind. The first was an analysis of the chemical contents contained in the canister to determine the quantity of CO_2 that reacted with the LiOH to form lithium carbonate $\left(Li_2CO_3\right)$. The second objective was a determination of the flow pattern of the CO_2 through the canister.

Procedures

Sample preparation. - Each bag was carefully removed from the canister, weighed, and sealed immediately in a clean polyethylene bag. Each polyethylene bag was numbered according to the location of the position in the canister (fig. A-1). The sealed polyethylene bags which contained the canister bags were all placed in an airtight container.

In order to determine the flow pattern of the CO_2 through the canister, a new sampling technique was employed. This sampling technique differed from the previous sampling methods primarily in that a greater number of samples were taken from each bag. In order to take representative samples throughout each bag, a sample divider was constructed of aluminum (fig. A-1). This divider made accessible nine separate sections for each layer taken from each bag. Two layers each were taken from bags 1 and 2, resulting in 18 representative samples from each bag. Three layers each were taken from bags 3 and 4, resulting in 27 representative samples from each of these bags (fig. A-2).

One packet at a time was removed from its polyethylene bag. The top of the bag was carefully removed by cutting around the top edge with scissors. Care was taken to prevent the loss of any of the contents. The sample divider was then gently pushed down into the bag to a depth of approximately 1 inch. This allowed approximately 0.25 inch of the divider to remain above the top level of the contents of the bag.

The contents of the nine sections which were made accessible by the divider were individually removed with a spatula. The contents of each section were weighed to an accuracy of ±0.01 gram and then transferred to a clean beaker. Approximately 10 grams were taken from each section to be thoroughly ground in a mortar. Each ground sample was then transferred to a glass vial and sealed.

During the sampling of each bag, 5 grams of each section's contents were transferred to a common container to represent a composite of the packet being sampled. After one bag was completely sampled, the contents of the composite container were thoroughly mixed. Approximately 10 grams were taken from the composite container to be ground in a mortar. The sample was then transferred to a glass vial and sealed. The contents of each bag were treated in the same manner.

Sample analysis. - Four 0.60-gram samples from each vial were weighed into 50-milliliter Griffin beakers. Each 0.60-gram sample was then transferred to a 25-milliliter crucible and weighed accurately to ±0.00001 gram.

The sample in each crucible was transferred to 400-milliliter Griffin beakers. Transfer of the samples from the crucible to the beakers was accomplished by carefully washing the crucibles with distilled water until all chemicals were visually absent. At this point, each crucible was rinsed six times with distilled water so that approximately 250 milliliters of solution were contained in the beaker. The beakers containing the solution were then allowed to stand for a period of approximately 18 hours. This allowed all of the Li_2CO_3 to go into solution. After the samples had completely dissolved, 25 milliliters of 10-percent barium chloride $(BaCl_2)$ were added to each beaker. This permitted the quantitative determination of LiOH in the presence of Li_2CO_3 by precipitating the carbonate ion as barium carbonate $(BaCO_3)$.

All of the titrations of the samples in this analysis were accomplished through the use of an automatic titrator. This titrator recorded the volumes of hydrochloric acid (HCl) required for the titration of each sample. The end-point volumes for both LiOH and Li_2CO_3 were contained in this recording.

Discussion of Results

There were two main objectives in the analysis performed. These were a chemical analysis and an abnormal flow pattern, if such a pattern did occur. The specific objective for the composite analysis was a qualitative as well as a quantitative analysis of the chemicals contained in the canister. The specific objective for the section analysis was the determination of the CO_2 flow pattern through the canister.

It would have been ideal had both types of analyses presented identical results. Even after reducing the sectional data by mathematical procedure to more approximate composite values, the results of the two methods of analysis differed considerably.

By comparing the results of the sectional analysis against the composite analysis, as tabulated in table A-I, the following differences were noted:

1. The sectional value for the weight of LiOH was 6.75 percent less than the composite value.

2. The sectional value for the weight of water was 47.85 percent greater than the composite value.

3. The sectional value for the weight of CO_2 was 2.10 percent less than the composite value.

The discrepancies between these two sets of results are due mainly to the manner in which the test samples were taken. In order to determine which analysis gave the most reliable quantitative results, the application of some basic chemical principles needed to be applied.

By a reverse calculation, based upon the data from the analysis, the quantity of LiOH used in packing the canister can be determined. Since it was known that 5.40 pounds (2431.6 grams) of approximately 98-percent LiOH were used in packing the canister, the calculated value nearest this value would, in all likelihood, have been the most reliable.

The method used to arrive at this result was as follows:

$$2\ \text{LiOH} + \text{CO}_2 \longrightarrow \text{Li}_2\text{CO}_3 + \text{H}_2\text{O} + \text{heat} \qquad \text{(A1)}$$

$$\text{weight of LiOH} = \frac{2(\text{atomic weight of LiOH})\left(\text{weight of Li}_2\text{CO}_3\right)}{\text{atomic weight of Li}_2\text{CO}_3} \qquad \text{(A2)}$$

The calculation of the weight of LiOH packed in the canister prior to usage, based on sectional analysis, was as follows.

Using equation (A2)

$$\text{weight of LiOH} = \frac{2(23.95)(2159.57)}{73.89}$$

$$= 1399.96 \text{ grams of LiOH}$$

This weight plus the weight of LiOH determined by direct titration represented the total weight of LiOH packed in the canister prior to its usage.

This value then was

LiOH used to form Li_2CO_3	1399.96 g
LiOH analyzed for	921.64 g
Total weight of LiOH prior to run	2321.60 g

The following can be obtained by carrying out these same calculations upon the data derived from the composite analysis.

Using equation (A2)

$$\text{weight of LiOH} = \frac{2(23.95)(2205.92)}{73.89}$$

$$= 1430.01 \text{ grams of LiOH}$$

This value then was

LiOH used to form Li_2CO_3	1430.01 g
LiOH analyzed for	988.37 g
Total weight of LiOH prior to run	2418.38 g

Since the weight of the composite analysis was closest to the known packed weight, this was the group of results which was most dependable.

The fact that the sectional analysis did not give the overall quantitative values known to be correct did not mean that they were to be discarded. It should be remembered that the flow pattern was to be established from these values.

The results attained from each section are indicated in figure A-3. The flow pattern of the CO_2 through the canister is indicated in figure A-4. The slant-lined areas represent areas of highest Li_2CO_3 concentration or highest CO_2 flow rate. These areas were derived by correlating the sectional values obtained in the analysis. The front-, left-, and right-side views of the CO_2 flow pattern are shown in figures A-5 and A-6.

Instrumentation

The end points of the titration of LiOH and Li_2CO_3 with HCl were measured on a Sargent Recording Titrator, Model D, serial no. S-29685, and readout on a recorder. The titrator had a range of 2 pH units and an accuracy of 0.25 percent, full scale. The calibration curve was designated no. 1, and the date of the last calibration was June 3, 1963.

Sample Calculations

Method for determining the percent of LiOH. -

$$\text{percent of LiOH} = \frac{(\text{ml HCl to LiOH evaporation point})(\text{LiOH equivalent})}{\text{sample weight}} \tag{A3}$$

Method for determining the percent of Li_2CO_3. -

$$\text{percent of } Li_2CO_3 = \frac{\left(\text{ml HCl to } Li_2CO_3 \text{ evaporation point}\right)\left(Li_2CO_3 \text{ equivalent}\right)}{\text{sample weight}} \tag{A4}$$

Method for determining the percent of water. -

$$\text{percent of } H_2O = 100 \text{ percent} - \left(\text{percent of LiOH} + \text{percent of } Li_2CO_3\right) \tag{A5}$$

Since this equation does not account for the percent of impurities present, it is not precisely correct.

Method for determining the weight of CO_2. -

$$\text{weight of } CO_2 = \frac{\left(\text{molecular weight of } CO_2\right)\left(\text{weight of } Li_2CO_3\right)}{\text{molecular weight of } Li_2CO_3} \tag{A6}$$

Method for determining the volume of CO_2 at standard temperature and pressure (STP). -

$$\text{volume of } CO_2 = \frac{\text{weight of } CO_2}{\text{density of } CO_2 \text{ (STP)}} \tag{A7}$$

Method for determining the flow rate of CO_2. -

$$\text{flow rate} = \frac{\text{volume of } CO_2 \text{ (calculated)}}{\text{length of time canister in use}} \tag{A8}$$

Method for reduction of sectional analysis data for comparison with composite analysis results. -

$$C_i = \frac{\text{(weight of percentage of chemical)(weight of section)}}{100} \tag{A9}$$

where C_i is the weight of LiOH or Li_2CO_3 of each individual section. The percent of chemical composition per section = $\dfrac{\sum_1^i C_i(100)}{\text{total bag contents}}$ where $\sum_1^i C_i$ is the summation of the LiOH or Li_2CO_3 sample weights of each individual section.

$$\text{percent } H_2O = 100 \text{ percent} - \left(\text{percent of LiOH} + \text{percent of } Li_2CO_3\right) \tag{A10}$$

This method for determining the percent of water is in error for the same reason as was the similar calculation for the percent of water in the composite analysis.

Conclusion

The most important conclusion that can be made from the test results is that the LiOH canister had not been completely consumed by CO_2. The second important conclusion that can be made is that the flow of the CO_2 through the canister experienced a channeling effect.

This channeling effect was probably the result of two contributing factors, the compaction and the moisture content of the LiOH bags contained in the canister. Compaction of the LiOH is most likely to occur in the canister during the period in which the gravitational force is increased. Since the canister was positioned on its right side (figs. A-1 and A-4), the LiOH would most likely be packed toward this side. Any moisture contained in the bags would also experience a similar action.

Since the areas of highest compaction and moisture content would offer the highest resistance to gas flow, the remaining areas would naturally offer the least resistance. Those areas which offered least resistance to gas flow would most likely be the areas of highest Li_2CO_3 content. This supposition was borne out in the analytical results as depicted in figure A-4. Gas flow was shown to be in the upper section of the canister. There is some discrepancy with bag 1, and the reason for this is that the sample divider had not been constructed in time for the sampling of this bag.

Diagrams of the flow of the CO_2 through the canister are shown in figures A-5 and A-6.

TABLE A-I. - RESULTS OF CHEMICAL ANALYSIS OF MERCURY-ATLAS 9 LITHIUM HYDROXIDE CANISTER

(a) Sectional analysis[a]

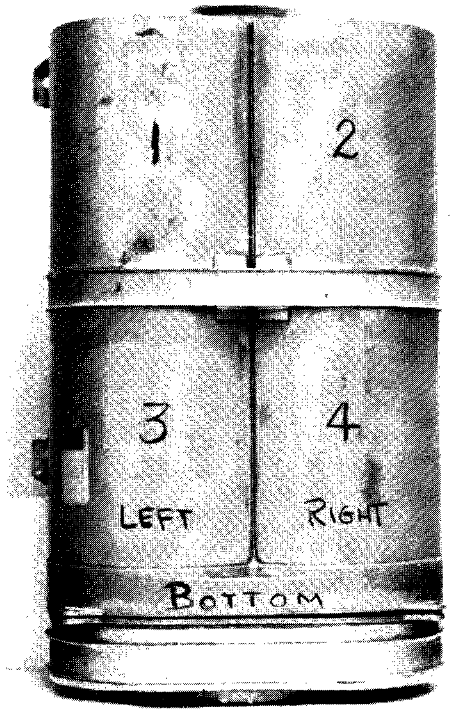
Bag number	Weight of contents, g	LiOH, percent	Li_2CO_3, percent	H_2O, percent	LiOH, g	Li_2CO_3, g	H_2O, g	CO_2, g	CO_2, liters, STP
1	922.0	7.48	83.22	9.30	68.97	767.29	85.75	457.01	231.20
2	910.6	8.78	83.84	7.38	79.95	763.45	67.20	454.72	230.04
3	757.8	49.43	44.01	6.56	374.58	333.51	49.75	198.64	100.49
4	727.2	54.75	40.61	4.64	398.14	295.32	33.74	175.90	88.53
Total	3317.6				921.64	2159.57	236.44	1286.27	650.26

[a]Carbon dioxide flow rate, 255.6 cc/min; canister time, 42.4 hr.

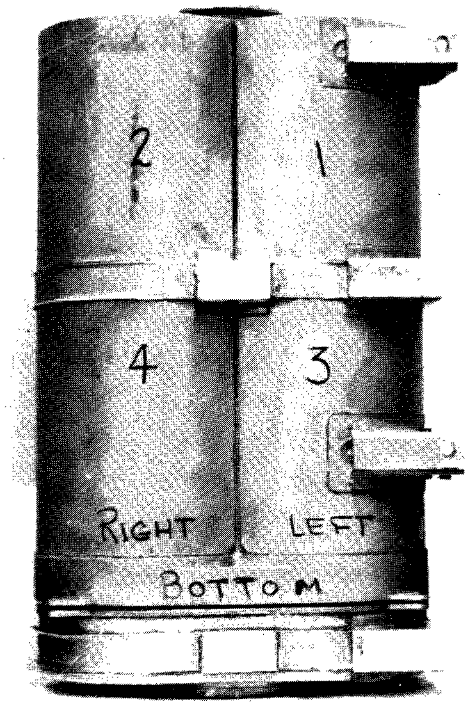
(b) Composite analysis[a]

Bag number	Weight of contents, g	LiOH, percent	Li_2CO_3, percent	H_2O, percent	LiOH, g	Li_2CO_3, g	H_2O, g	CO_2, g	CO_2, liters, STP
1	922.0	7.78	89.02	3.20	71.73	820.76	29.50	488.86	247.31
2	910.6	11.12	86.34	2.54	101.26	786.21	23.13	468.28	236.90
3	757.8	52.85	41.93	5.22	400.50	317.74	39.56	189.25	95.74
4	727.2	57.05	38.67	4.28	414.88	281.21	31.12	167.48	84.73
Total	3317.6				988.37	2205.92	123.31	1313.87	664.68

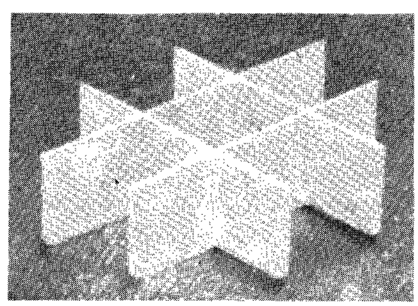
[a]Carbon dioxide flow rate, 261.3 cc/min; canister time, 42.4 hr.



Front view



Rear view



Sample divider

Figure A-1.- Lithium hydroxide canisters and sample dividers.

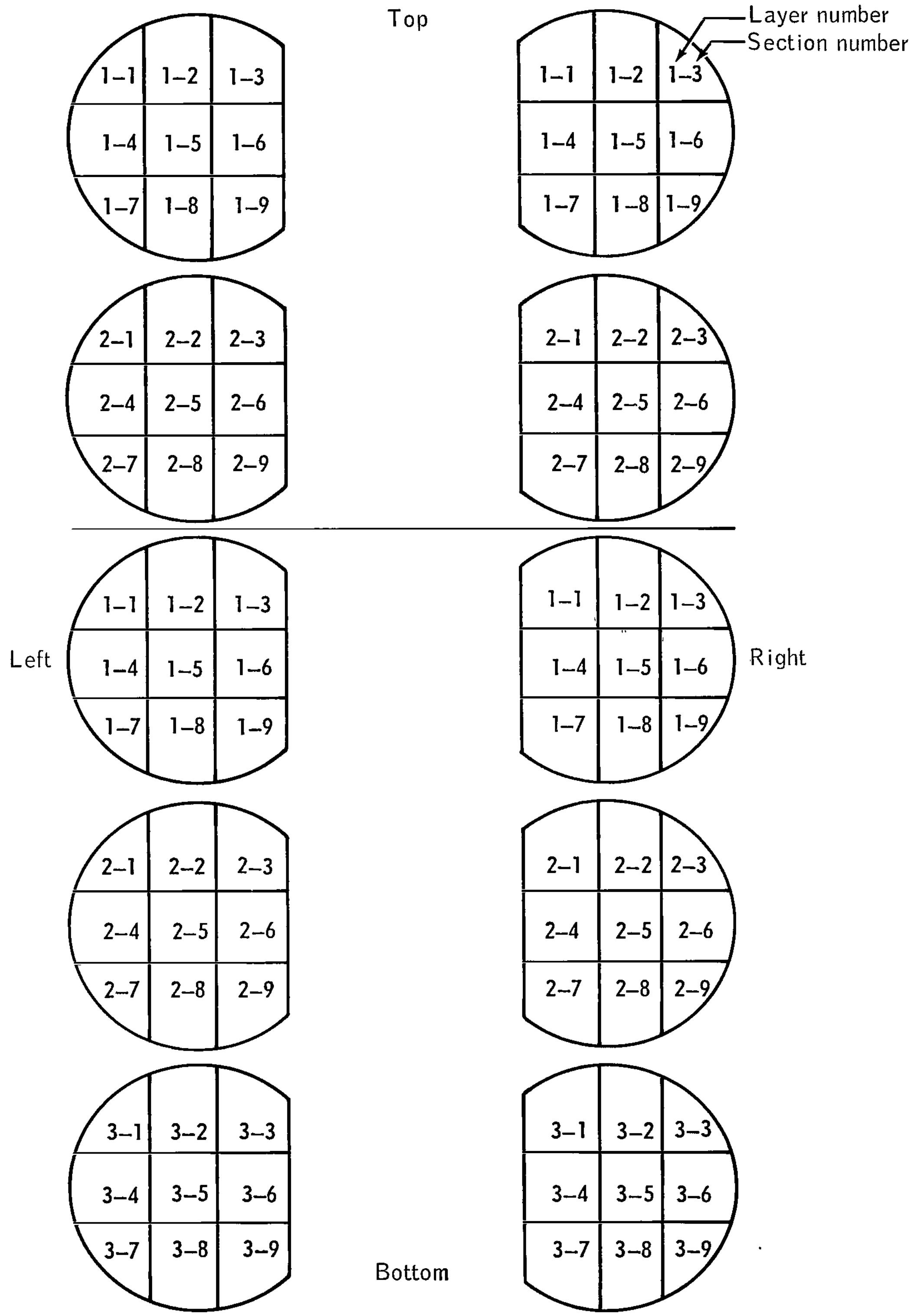


Figure A-2. - Canisters top and bottom showing layers and section numbering.

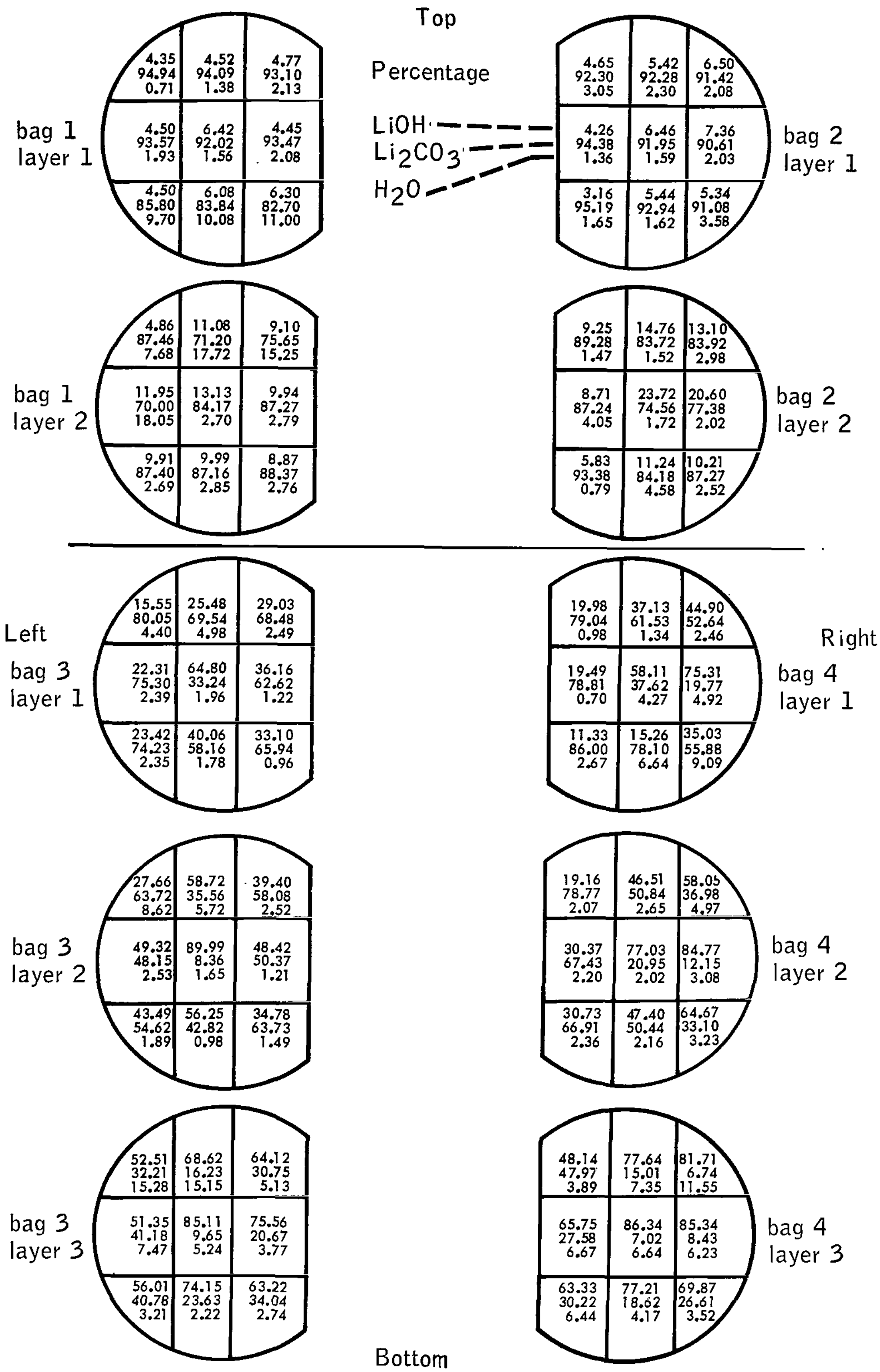


Figure A-3. - Chemical composition in percentages.

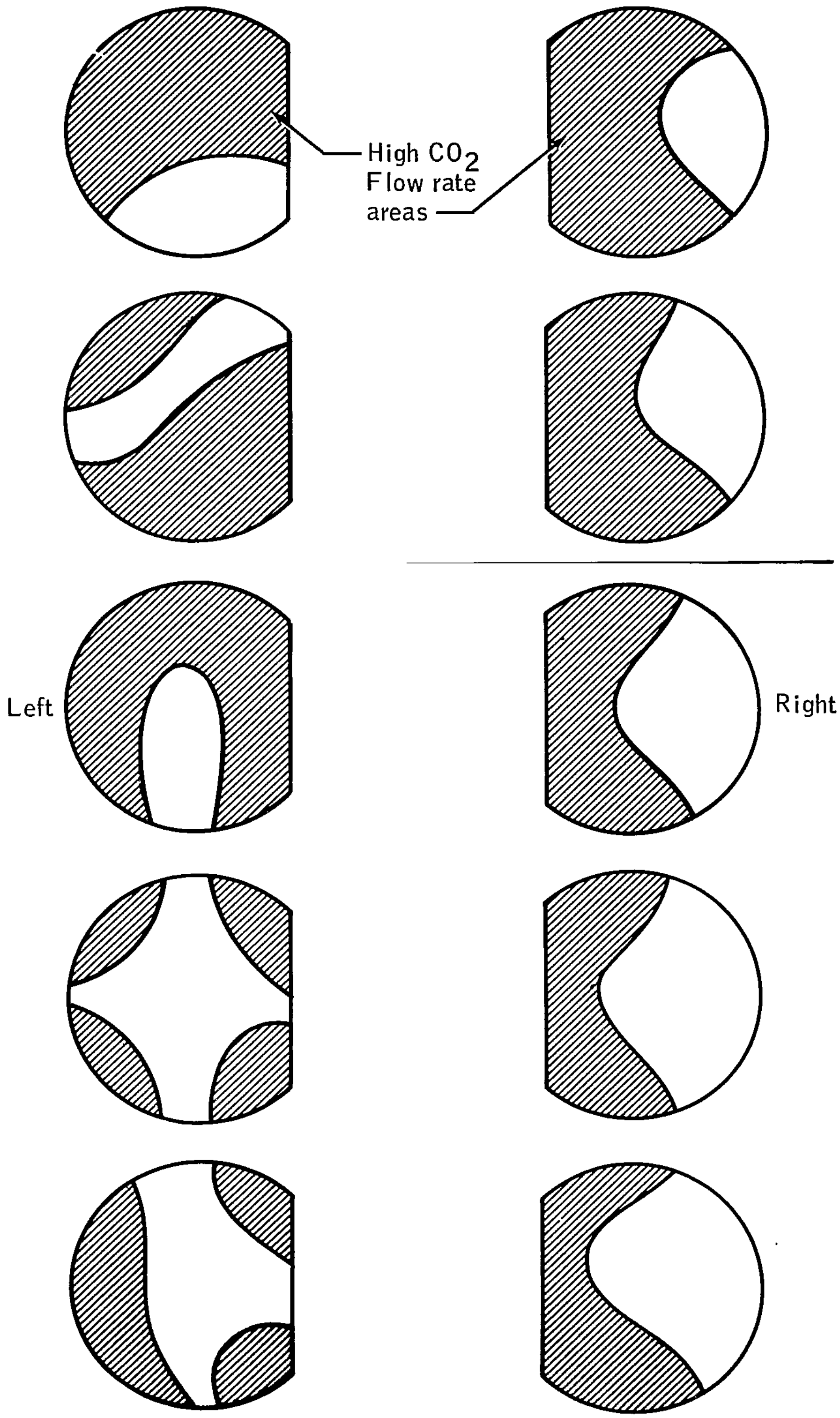


Figure A-4. - Areas of maximum carbon dioxide flow as measured by areas of high lithium carbonate content.

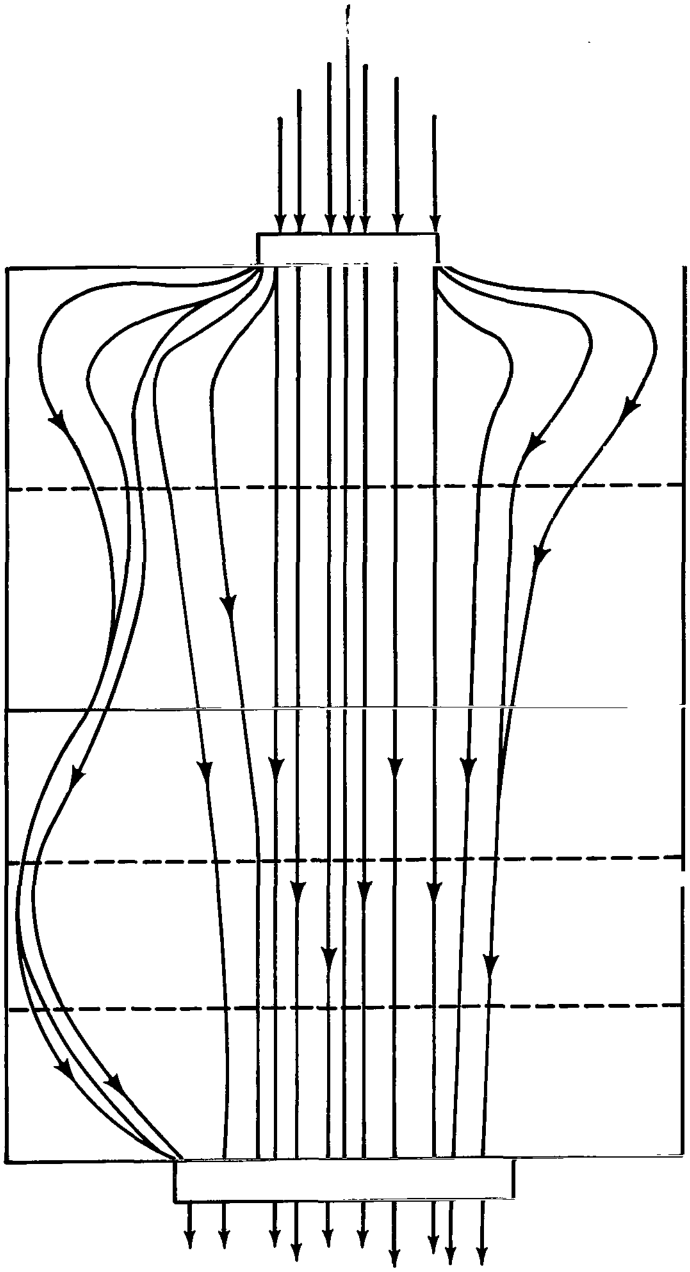


Figure A-5. - Carbon dioxide flow pattern, front view of canisters.

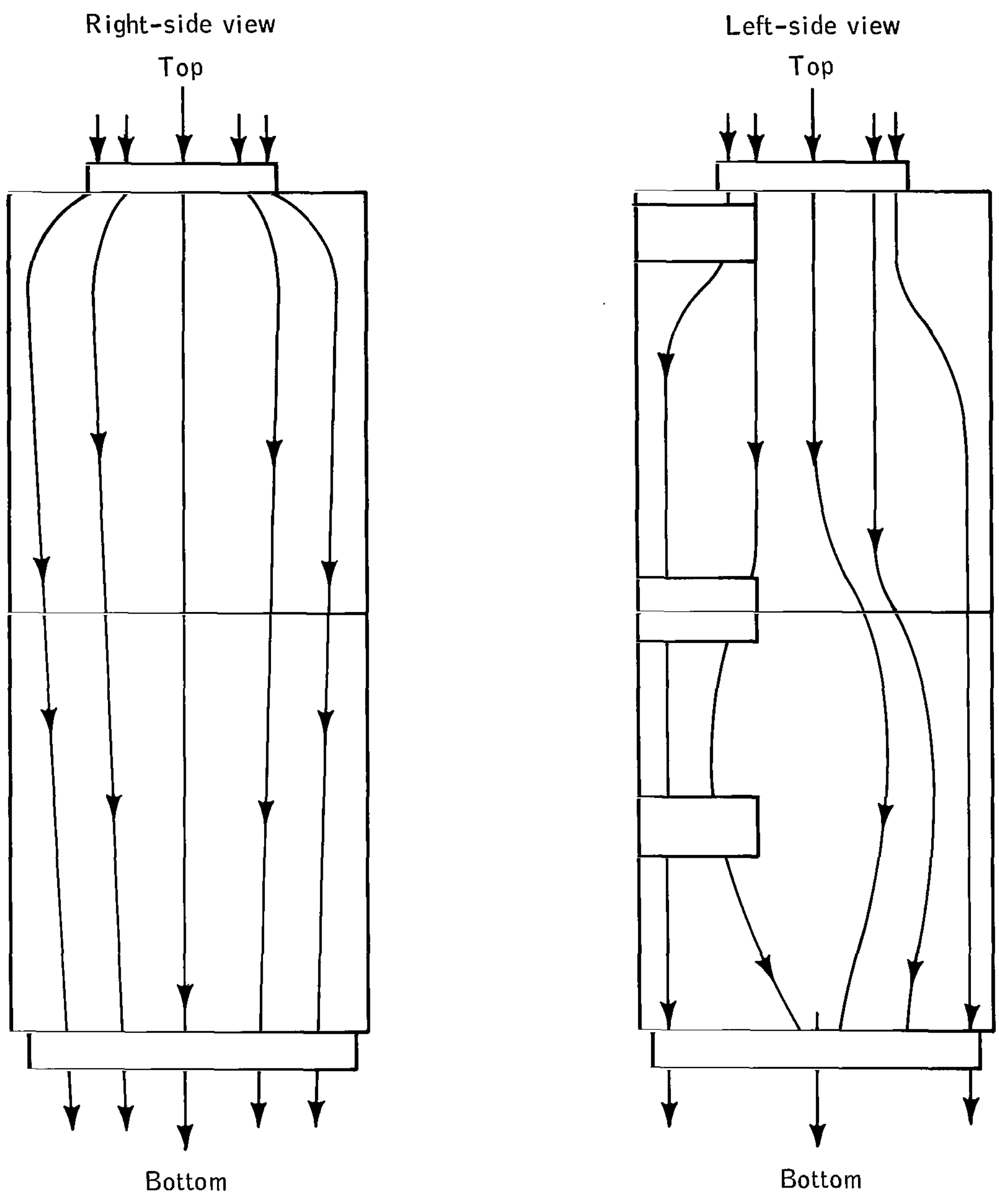


Figure A-6. - Carbon dioxide flow pattern, right- and left-side views.

APPENDIX B

ANALYSES OF CONTAMINANTS IN SPACECRAFT ATMOSPHERE

The following are reports on analyses of contaminants developed in the atmosphere of spacecraft during the orbital flights of Spacecraft 9, which was Mercury-Atlas 5 (MA-5), Spacecraft 13 (MA-6), Spacecraft 18 (MA-7), Spacecraft 16 (MA-8), Spacecraft 20 (MA-9), and capsule 10. Reports were prepared by R. A. Saunders, Physical Chemistry Branch, Naval Research Laboratory (NRL). (Results of chemical analysis of the CO_2 absorber are shown in table B-I.)

Spacecraft 9 (MA-5)

The carbon from canister 60-13, capsule 9, has been analyzed by vacuum desorption at 300° C, followed by chromatographic and spectral examination of the desorbate. Only one bag of carbon (one-half the available supply) was desorbed.

The major portion of the desorbate was Freon-114, 8.7 grams of which were desorbed from 237 grams of carbon. The carbon thus held 3.7-percent Freon by weight, which is about one-fourth the amount necessary for saturation. Since the flow through the two carbon bags was equal, the total Freon removed by the carbon filter was 17.4 grams.

In addition to water (3.0 grams), the desorbate contained carbon dioxide, methylene chloride, ethyl alcohol, methyl alcohol, acetaldehyde, and two unidentified hydrocarbons. The amount of ethyl alcohol recovered from one bag of carbon was 0.008 gram. The total amount of ethyl alcohol removed from the capsule atmosphere by the carbon filter was 0.016 gram. This amount of ethyl alcohol, if dispersed in the 60-cubic-foot volume of the capsule, would have resulted in a concentration of 5 ppm. Dispersed in the smaller volume of the pressure-suit circuit, the concentration would have been proportionately higher. The equivalent concentration of the other organics was less than 1 ppm.

The water desorbed from the carbon had a pH of 2 (to test paper). The cause of this acidity has not yet been ascertained. No hydrochloric, nitric, sulphuric, or acetic acids could be detected.

Spacecraft 13 (MA-6)

A total of 9.35 grams of desorbate was recovered from 461 grams of carbon. A major portion (5.6 grams) of the desorbate was water. The remaining 3.75 grams were gaseous at room temperature. Chromatographic examination of the water indicated the presence of trace quantities of methyl and ethyl alcohol. Infrared spectral analysis of the gas-phase desorbate revealed carbon dioxide and Freon-114 as the major components of this mixture. These two gases accounted for 86.5 and 13.0 percent by weight of the gas-phase desorbate, respectively. The total amount of Freon (0.40 gram) recovered from this carbon was much less than that recovered from previous Mercury carbons. The carbon used for the MA-5 mission, for instance, yielded 28 times as much. The low Freon yield from the MA-6 mission carbon indicated the use of a fresh carbon canister for this flight and a minimum of prelaunch testing involving Freon.

The remaining 0.5 percent of the gaseous desorbate consisted of small quantities of other organic contaminants. Similar contaminant mixtures recovered from previous Mercury mission carbons have been analyzed by using a gas chromatograph to resolve the various components of the mixture. Each component is collected individually, as it is eluted from the chromatograph, with a novel fraction collector developed for this purpose at the laboratory. The pure components are then transferred to infrared gas or to liquid microcells, and their identity is established by means of infrared spectra.

A chromatogram of the desorbate from the MA-6 mission is shown in figure B-1. The contaminants eluted under the three largest peaks (multiplied by one attenuation) were identified from their infrared spectra, as outlined above. Insufficient material was eluted under the remaining peaks, however, to permit identification by this means. Mass-spectral analysis, which is more sensitive than infrared, was used to identify these components. The material eluted under each peak was collected and introduced into the inlet system of a mass spectrometer. The inlet system of this spectrometer was modified by replacing the 3-liter expansion volume with a 300-milliliter expansion volume, thus increasing the sensitivity for microsamples by a factor of 10. Provisions were also provided for efficiently transferring microsamples from the collection tube to the inlet system without diluting the sample air or other foreign gases. In spite of the fact that one or two of the chromatographic peaks were only one division high at maximum sensitivity, ample material was recovered from the effluent stream with the microfraction collector to permit mass-spectral identification. The sensitivity of the analytical techniques described appears sufficient to allow identification of contaminants in the capsule atmosphere at concentrations less than 1 part per billion.

The contaminants which were recovered from the atmosphere of Spacecraft 13 are listed below in order of decreasing concentration. The concentrations given are approximate and represent the value which would have been obtained had the amount of material recovered from the carbon been dispersed in the 60-cubic-foot volume of the cabin.

Contaminant	Minimum concentration, ppm
Vinylidene chloride	3
Benzene	3
Vinyl chloride	2
Methyl chloroform	.5
Methylene chloride	.4
p-dioxane	.3

Contaminant	Minimum concentration, ppm
Unidentified	0.3
Cyclohexane	.2
Toluene	.2
Methyl alcohol	.2
Ethyl alcohol	.05
Trichlorofluoromethane	.05

It is interesting to compare the contaminants from Spacecraft 13 with those found in the atmosphere of capsule 10 following the severe overheating of a pair of stabilizing gyros during simulated flight. Most of the components detected in Spacecraft 13 were also present at much higher concentrations in the cabin atmosphere of capsule 10. These same stabilizing gyros reportedly functioned improperly during the MA-6 mission. This coincidence suggests the question whether any or all of the contaminants found in these two cabin atmospheres originated in the malfunction of the stabilizing gyros, or whether they arose from all of the cabin components, wiring, insulation, et cetera, under normal operating conditions.

The contaminants recovered from the Spacecraft 13 suit-circuit carbon were evolved by the various components of the cabin equipment. There was no carbon filter in the cabin-atmosphere circuit to remove these contaminants. A certain portion of these contaminants was adsorbed on the suit-circuit carbon, however, when the pressure-suit faceplate was open. The open faceplate allowed mixing of the otherwise separated suit-circuit and cabin atmospheres. The pilot had his faceplate open during a major part of the orbital flight. The degree of mixing between the two atmospheres when the faceplate was open is not known, but it is doubtful that all the cabin contamination in Spacecraft 13 was removed by the suit-circuit carbon. For this and other reasons, the concentrations given in the preceding list for the various contaminants should be considered minimum values. The actual concentration of these contaminants in the cabin atmosphere probably was several times higher. The concentration of one or two may have been high enough for olfactory detection.

Infrared or mass spectra of all of the components recovered from the carbon have been obtained in the pure state in spite of the fact that some of them, such as ethyl alcohol and p-dioxane, were hardly apparent on the original chromatogram, and some were of such low concentration as to produce a peak that was only 1 percent of full-scale deflection at maximum sensitivity.

The mass spectrum of one component, as yet unidentified, is shown in figure B-2. This spectrum actually represents only one-half of the sample eluted under the peak shown in figure B-1. To ascertain whether or not the material eluted under this peak consisted of one or more components, the material eluted under the front half and the trailing half of the peak were collected separately. The mass spectrum of each was exactly the same.

A large number of contaminants are released in the cabin atmosphere by capsule equipment during apparently normal orbital flight. Some of these are considered hazardous at higher concentrations, for example, chlorinated and aromatic hydrocarbons. None of these contaminants have yet been detected at unsafe concentration levels. However, overheating of equipment or localized combustion would produce a much higher concentration. A 1-pound activated-carbon filter in the cabin atmosphere circuit would remove impurities. In addition, this filter would provide a more accurate means of determining the concentration of cabin contaminants than is presently provided by the filter in the suit circuit. The latter is partially effective only when the faceplate is open.

The conclusions are as follows:

1. The analytical procedures presently used permit the detection of contaminants in the capsule atmosphere at concentrations approaching 1 part per billion.

2. The sensitivity of these analytical procedures is sufficiently high to permit the possibility of determining or verifying, post facto, the failure or near failure of certain types of capsule equipment by an analysis of the contaminants that have evolved as a result of the malfunction or overload.

Spacecraft 18 (MA-7)

Preliminary report. - All of the contaminants detected in the Spacecraft 18 atmosphere were previously detected in the Spacecraft 13 atmosphere. The Spacecraft 13 atmosphere, however, contained a few contaminants not detected during the MA-7 mission. A comparison of the two atmospheres is given in the list below. The list is arranged in order of decreasing concentration as detected in the atmosphere of Spacecraft 13. An "e" indicates the contaminant was detected in the atmosphere of Spacecraft 18 in approximately equivalent concentration, "h" indicates a much higher concentration, and "l" indicates lower concentration. Contaminants marked "a" were not detected in the atmosphere of Spacecraft 18.

Contaminants in Spacecraft 13	Contaminants in Spacecraft 18
Water	e
Carbon dioxide	e
Freon-114	e
Vinylidene chloride	l
Benzene	h

Contaminants in Spacecraft 13	Contaminants in Spacecraft 18
Vinyl chloride	a
Methyl chloroform	e
Methylene chloride	e
p-dioxane	a
Unidentified	a
Cyclohexane	e
Toluene	h
Methylene	e
Ethyl alcohol	e
Trichlorofluoromethane	e

Relatively large concentrations of certain chlorinated hydrocarbons have been detected twice in capsule atmospheres following partial or complete failure of stabilizing gyros (that is, simulated flight of capsule 10 in St. Louis and orbital flight of Spacecraft 13). These contaminants were either absent or detected at relatively low concentration in the atmosphere of Spacecraft 18. This fact may indicate that the operating temperature of the stabilizing gyros was closer to normal during the flight of Spacecraft 18 than during the flight of Spacecraft 13. The significance of the relatively high concentrations of benzene and toluene in the Spacecraft 18 atmosphere is not presently known.

The total concentration of contaminants from the cabin atmosphere found in the desorbate of the suit-circuit carbon naturally depends upon how long the pilot had his faceplate open. It has been reported that the pilot had his faceplate open during a large portion of the Spacecraft 13 flight. Comparative data for the Spacecraft 18 flight are not at hand.

These results shed no light on the origin of the smoke reported in the cabin of Spacecraft 18 during flight. The increased concentration of benzene and toluene may or may not have been associated with this malfunction. Smoke itself is an aerosol and is composed for the most part of materials difficult or impossible to remove from carbon by the methods presently used. However, contaminants of this type can be completely retained on a micropore fiber-glass filter from which they could be easily removed for analysis. A special Navy gas-mask filter developed at the Naval Research Laboratory (NRL) is well suited to this purpose. This unit, which weighs less than 0.5 pound, has been used at MAC to sample the atmospheres of capsule 10 and might be considered as an added component in the cabin air-circulating system on future missions.

Final report. - It is impossible to determine the absolute concentrations of the various contaminants in the cabin atmosphere from an analysis of the desorbate taken from the suit-circuit carbon for several reasons. For example, the efficiency with which the various contaminants are adsorbed on and desorbed from the carbon are only known approximately, the degree of mixing between the cabin atmosphere and the suit-circuit atmosphere with an open faceplate is not known, and data are not available as to the total time interval during which the faceplate was open. However it is possible to give approximate minimum concentrations. To make this information more meaningful, the concentrations of the contaminants in Spacecraft 18 are compared with the concentrations of those found in Spacecraft 13.

A total of 16.9 grams of desorbate was recovered from 241 grams of carbon (one-half of the available sample). Almost one-half (7.5 grams) of the desorbate was water. Infrared analysis of the remainder indicated Freon-114 and carbon dioxide to be the major components of the mixture. The concentrations of these gases were 40 and 60 percent by weight, respectively. The total quantity of Freon recovered was 7.5 grams. This is 15 times the quantity (0.5 gram) recovered from the atmosphere of Spacecraft 13, but only about one-half that from the MA-5 orbital flight.

A list of contaminants and their approximate minimum concentrations, as found in the atmospheres of Spacecraft 13 and Spacecraft 18, is as follows:

Contaminant	Spacecraft 18	Spacecraft 13
Freon-114	7.5 grams	0.49 gram
Toluene	1.4 ppm	.2 ppm
Trichlorofluoromethane	.2 ppm	.05 ppm
Methyl alcohol	.7 ppm	.2 ppm
Benzene	8.7 ppm	3.0 ppm
Methyl chloroform	.5 ppm	.5 ppm
Methylene chloride	.3 ppm	.4 ppm
Cyclohexane	.05 ppm	.2 ppm
Vinylidene chloride	.4 ppm	3.0 ppm
Ethyl alcohol	$<.05$ ppm	.05 ppm

It should be understood that the actual concentrations of the minor contaminants in the cabin atmosphere could be many times higher than the minimum concentrations reported above, although the relative concentrations would remain about the same. A more accurate determination of contaminant load in the cabin atmosphere could be obtained from an analysis of the desorbate taken from a carbon desorption unit in the cabin atmosphere circuit.

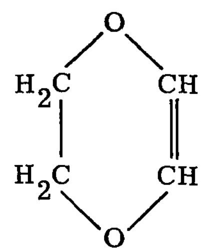
Spacecraft 16 (MA-8)

The carbon was desorbed in the usual manner by being heated under vacuum to a temperature of 350° C and by retaining the desorbate in a liquid nitrogen trap. The desorbate mixture was resolved into its components by means of a vapor-phase chromatograph. The material eluted under each chromatographic peak was recovered with the fraction collector and identified by its infrared and/or mass spectrum. Quantitative data for the major components are based upon chromatographic areas or peak heights. Components for which no quantitative data are given were present at trace concentrations.

One package of carbon (231.9 grams) yielded a desorbate consisting of the following:

Water	4.0 grams
Carbon dioxide	1.37 grams
Freon-114	.85 gram
Toluene	15.6 milligrams
Vinylidene chloride	8.7 milligrams
Methylene chloride	6.4 milligrams
Benzene	5.3 milligrams
Cyclohexane	.6 milligram
Vinyl chloride	
Trichlorofluoromethane	
Methyl alcohol	
p-dioxane	
p-dioxene	
Hexamethylcyclotrisiloxane	
Methyl chloroform	

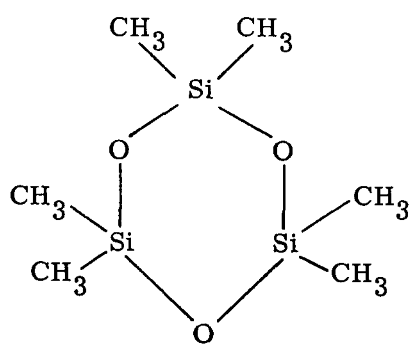
There was a contaminant detected for the first time in the desorbate from the MA-6 carbon (Spacecraft 13) which was not identified at the time in spite of an excellent mass spectrum. This contaminant was not detected in the desorbate from the MA-7 carbon (Spacecraft 18). However, it was present in the MA-8 carbon desorbate in sufficient quantity for both infrared and mass spectra. On the basis of these spectra, it was possible to predict the following structure for this contaminant:



The proposed compound, 1, 2-dihydro-1, 4-dioxin (p-dioxene), was synthesized and proved to have infrared and mass spectra identical to those of the contaminant in question.

The presence of p-dioxene in the capsule atmosphere, if indeed it ever existed there, is hard to believe. It may be that the presence of this unusual compound in the carbon desorbate results from decomposition during the adsorption-desorption process of another contaminant, p-dioxane, which actually does exist in the capsule atmosphere. The presence of p-dioxane in the capsule atmosphere may arise from the use of certain contact cements.

Another unusual contaminant, hexamethylcyclotrisiloxane, was detected for the first time in the atmosphere of Spacecraft 16. This compound



often occurs as a trace component in methyl silicone lubricants. Such lubricants usually comprise a range of molecular weights. The more volatile components, such as the cyclic trimer, generally constitute a very small fraction of the lubricant and are driven off by heat. The cyclic trimer can also be evolved from these lubricants as a result of molecular rearrangement caused by excessive temperatures. The presence of the cyclic trimer in the desorbate from the MA-8 carbon probably indicates that some silicone-lubricated component in the capsule became heated on the MA-8 mission to a greater degree than on previous missions. Siloxane compounds are considered of very low toxicity when inhaled or ingested. However, the vapors of some of these compounds are known to cause minor and temporary eye irritation.

Spacecraft 20 (MA-9)

Activated charcoal from the environmental control system of Spacecraft 20 was removed after orbital flight and sent to NRL for analysis of the contaminants which developed in the spacecraft atmosphere during the flight period. Similar analyses of the atmospheres in all other Project Mercury spacecraft have also been made. The total list of contaminants detected in these various atmospheres now stands at 60, although usually less than 20 are detected in any one atmosphere. These 60 contaminants are as follows:

1. Carbon dioxide
2. Freon-114
3. Ethylene dichloride
4. Toluene
5. n-butyl alcohol
6. Freon-11
7. Vinyl chloride
8. Ethyl alcohol
9. m-xylene
10. Vinylidene chloride
11. Methylene chloride
12. o-xylene
13. Benzene
14. Methylchloroform
15. Trichlorethylene
16. Acetone
17. Methyl ethyl ketone
18. Methyl isopropyl ketone
19. Ethylene
20. n-propyl alcohol
21. Acetaldehyde
22. Ethyl acetate
23. Freon-114, unsymmetric
24. Methyl alcohol
25. 1, 4-dioxane
26. Cyclohexane
27. Formaldehyde
28. Hexamethylcyclotrisiloxane
29. Freon-22
30. Freon-23
31. Freon-12
32. Freon-125
33. Hexene-1
34. Propylene

35. n-butane

36. Butene-1

37. iso-pentane

38. n-pentane

39. Propane

40. n-hexane

41. 2, 2-dimethylbutane

42. trans-butene-2

43. cis-butene-2

44. Acetylene

45. 3-methylpentane

46. p-dioxene

47. Carbon disulfide

48. 1-chloropropane

49. Isobutyl alcohol

50. trans-1, 2-dimethylcyclohexane

51. Tetrachloroethylene

52. p-xylene

53. Isopropyl alcohol

54. n-propylacetate

55. 1, 1, 3-trimethylcyclohexane

56. 1, 1-dimethylcyclohexane

57. trans-1, me-3, ethylcyclohexane

58. Allyl alcohol

59. Sulfur dioxide

60. Tetrafluoroethylene

Charcoal from the MA-9 mission was desorbed in the usual manner by heating it slowly in an evacuated system to a temperature of 300° C and by retaining the desorbate in liquid nitrogen traps. The desorbate mixture was resolved into its components by means of a vapor-phase chromatograph. Individual components were recovered successively from the effluent stream of the chromatograph by a fraction collector and were identified on the basis of their infrared and mass spectra. The quantitative data given for some components are based upon chromatographic peak heights or areas. Small amounts of other contaminants were identified but not determined quantitatively.

The MA-9 mission carried a total of 104. 8 grams of charcoal which yielded 5. 20 grams of desorbate. Over one-half this quantity (2. 72 grams) was water. The major components of the nonaqueous desorbate were Freon-114 (1. 25 grams) and carbon dioxide (0. 75 gram).

A chromatogram of the nonaqueous desorbate mixture is shown in figure B-3. The following components were identified:

Freon-114	$CF_2Cl\text{-}CF_2Cl$	1. 25 grams
Carbon dioxide	CO_2	. 75 gram
Acetone	CH_3COCH_3	. 127 gram

Toluene	$C_6H_5CH_3$	.101 gram
Freon-11	$CFCl_3$	
Tetrafluoroethylene	$CF_2{=}CF_2$	
Methylene chloride	CH_2Cl_2	
Methyl chloroform	CH_3CCl_3	
Benzene	C_6H_6	
Vinylidene chloride	$CH_2{=}CCl_2$	
Isopropyl alcohol	$(CH_3)_2CHOH$	

The numerical values are the quantities recovered from the charcoal. Greater quantities were undoubtedly present in the spacecraft atmosphere. Some idea of the relative concentrations of the minor components of the mixture can be visualized by comparing the size of the various peaks of the chromatogram. Some additional acetone and isopropyl alcohol can be expected in the water desorbate which has not yet been examined. Two of the components of this desorbate mixture, tetrafluoroethylene and isopropyl alcohol, have not been recovered previously from any Mercury spacecraft.

The desorbate from the MA-9 charcoal contained the largest amount of toluene (101 milligrams) ever recovered from a Mercury atmosphere. The second largest quantity, together with a large amount of benzene, was recovered from the atmosphere of Spacecraft 16 (MA-8) and attributed to the preflight use of an adhesive in the cabin interior. The amount of benzene recovered from the Spacecraft 20 atmosphere, however, does not appear to be unusually large. It would be advisable to determine the source of the toluene in the atmosphere of the pilot and, because it is a toxic compound, to minimize or eliminate it on future missions. Naval toxicologists have currently set the maximum allowable concentration of toluene in nuclear submarine atmospheres, for 60- and 90-day exposures, at 10 and 8 μg/liter, respectively. All of the toluene recovered from Spacecraft 20, dispersed in the cabin atmosphere at one time, would have resulted in a total concentration of 64 μg/liter. It is very doubtful that all of the toluene was recovered from the atmosphere, and the actual total concentration in the cabin could have exceeded U.S. Navy standards for long exposures by a factor of 10 or more, not considering any depletion of the toluene by the charcoal adsorbent. The rate at which toluene was evolved in the atmosphere and removed therefrom by the charcoal is not known, but obviously the concentration at any given time was less than the total concentration estimated above. Information is not available on tolerable concentrations of toluene for 50- to 75-hour exposures, but it would be reasonable to assume a value one order of magnitude or higher than that given for 60-day exposures.

Attention is also directed to the high concentration of acetone (127 μg/liter) which existed in the MA-9 atmosphere. Since previous Mercury atmospheres contained considerably less, this may indicate a mixed acetone-toluene solvent system as the source of these materials.

Capsule 10

In order to ascertain the nature and level of the contamination produced in the cabin atmosphere of a Mercury capsule by the deliberately induced malfunctioning of various components, several special gas-mask-type carbon canisters were prepared and sent to McDonnell Aircraft Corporation for exposure in capsule 10. Two of these canisters were exposed in December 1961 and returned to NRL in January 1962.

During the exposure of the first two canisters in this series, a gyro was intentionally burned out. The gyro unit reached a temperature of 320° F. A low-temperature lubricating grease on the bearings reached a temperature of 250° F and volatilized. Electrical-insulating varnish charred and volatilized. Epoxy-resin wedges in this unit also charred and partially volatilized. The analysis of the contaminants introduced into the cabin atmosphere by this malfunctioning follows.

A total of 85.1 grams of carbon from two canisters, which were exposed simultaneously, was vacuum-desorbed to a temperature of 300° C. The total desorbate recovered from this carbon was 3.52 grams, not including water. Assuming 100-percent adsorption and desorption from the carbon and assuming an average molecular weight of 168, this amount of desorbate (3.52 grams), dispersed in the free cabin volume of approximately 60 cubic feet, would result in a contamination level of 275 ppm. It is presently estimated that about one-half of this material was Freon-114. The estimated concentration of the minor contaminants is 5 to 50 ppm for some and considerably less for others.

The efficiency with which some compounds are desorbed from carbon is considerably less than 100 percent, in fact as low as 1 to 5 percent. Some of these compounds, C8-C9 aromatic hydrocarbons, were recovered from this carbon; and the actual concentration of these contaminants in the cabin atmosphere, therefore, may have been many times higher than the 1 to 5 ppm estimated here.

The carbon in these canisters was packed in two approximately equal layers preceded by a micropore fiber-glass filter (on the upstream side of the canister) to remove aerosols from the gas stream. Each of these carbon layers was desorbed and analyzed separately. High-molecular-weight compounds are always retained on the upstream layer of carbon, and lower-molecular-weight compounds are carried deeper into the carbon bed. The high-molecular-weight compounds from the subject canisters, however, were desorbed from what should have been the downstream layer. This fact indicates that the direction of gas flow through the canister was reversed, perhaps as a result of reversed fan-motor leads. Also there was no staining of the micropore filter on the upstream side, as there most certainly would have been had the gas flow been in the proper direction. At any rate, no analysis could be made of the aerosol or particulate matter in the cabin atmosphere. This was lost by adsorption on the carbon bed from which it could not be desorbed. This material would have further raised the contamination level in the cabin atmosphere.

Part of the desorbate remained in the liquid phase at room temperature as a water-immiscible oil. This is the first time that such material has been recovered from a capsule atmosphere. An infrared spectrum of this liquid is quite similar to that of the water-insoluble mixture obtained from submarine atmospheres. However, the mixture of contaminants from submarine atmospheres is exceedingly complex, numbering well over 100 components which are derived from a variety of sources. The main sources of contamination in the capsule atmosphere were considerably less in number, and the resultant contaminant mixture should be less complex. A chromatogram of this mixture shows that this is indeed the case. The presence of approximately 30 components is indicated, although about 10 of these constitute the bulk (more than 90 percent) of the mixture.

Both liquid- and gas-phase fractions of the desorbate were resolved into their various components using a chromatograph as a fractionating device. Infrared spectra were obtained for these components in order to establish their identity. The components identified are listed below. The first dozen are listed in approximate order of abundance:

1.	Water	
2.	Freon-114	$CF_2Cl\text{-}CF_2Cl$
3.	1, 2-dichloroethane	$CH_2Cl\text{-}CH_2Cl$
4.	Carbon dioxide	CO_2
5.	Toluene	$C_6H_5CH_3$
6.	Benzene	C_6H_6
7.	n-butanol	C_4H_9OH
8.	m-xylene	$C_6H_4(CH_3)_2$
9.	o-xylene	$C_6H_4(CH_3)_2$
10.	Ethyl alcohol	C_2H_5OH
11.	Acetone	CH_3COCH_3
12.	Methylene chloride	CH_2Cl_2
13.	Methyl chloroform	CH_3CCl_3
14.	Vinyl chloride	$CH_2{=}CHCl$

15. Monofluorotrichloromethane $CFCl_3$

16. Methyl ethyl ketone $CH_3COC_2H_5$

17. Methyl isopropyl ketone $CH_3COC_3H_7$

All of the major components, and many of the minor ones, have been identified. Those not specifically identified are present at concentrations considerably less than 1 ppm. Amines were detected in the water desorbed from the carbon but were not specifically identified.

In future sampling of the cabin atmosphere, using the special canisters furnished, it is suggested that some care be taken to insure proper direction of gas flow in order that the fiber-glass filter might be used for an analysis of particulate matter.

In view of the rather high level of contamination produced in the cabin atmosphere by the malfunction of components, it is suggested that a carbon filter for the cabin atmosphere be considered as an integral part of the capsule environmental control system.

TABLE B-I. - RESULTS OF CHEMICAL ANALYSIS OF CARBON DIOXIDE ABSORBER[a]

Mission	Packet	Weight of H_2O absorbed, lb	Weight of CO_2 absorbed, lb	Weight of LiOH remaining, lb	Total operating time, hr:min	CO_2 rate, cc/min (b)	Operating time under zero-g conditions, hr:min
MA-6	1	0.109	0.342	0.729	9:19	353	4:38
	2	.185	.380	.687			
	3	.423	.082	.988			
	4	.568	.051	1.068			
	Total	1.285	.855	3.472			
MA-7	1	.274	.672	.390	16:03	369	4:39.5
	2	.285	.658	.409			
	3	.484	.114	.996			
	4	.495	.092	1.017			
	Total	1.538	1.536	2.812			
MA-8	1	.185	.517	.529	14:53	359	8:56
	2	.151	.657	.380			
	3	.366	.159	.927			
	4	.369	.064	1.044			
	Total	1.071	1.397	2.880			
MA-9	1	.065	1.076	.158	42:24	261	34:04
	2	.051	1.030	.223			
	3	.087	.416	.881			
	4	.068	.369	.913			
	Total	.271	2.891	2.175			

[a]Analysis of MA-6 and MA-7 canisters was performed by NRL. The MA-8 and MA-9 canisters were analyzed by Crew Systems Division, Manned Spacecraft Center.

[b]At 14.7 psia and 32° F.

(All peaks multiplied by one attenuation unless otherwise noted.)

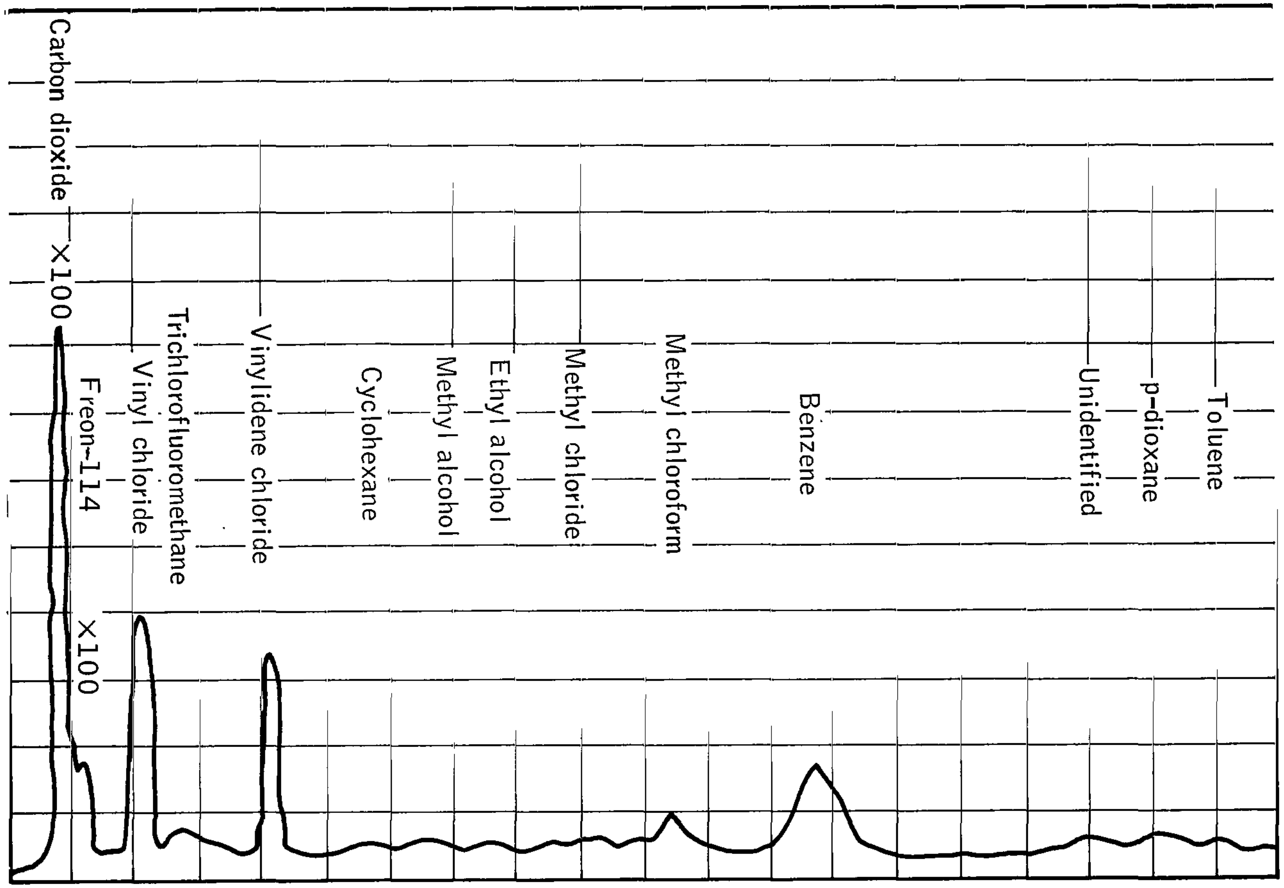


Figure B-1. - Chromatogram of desorbate from Spacecraft 13.

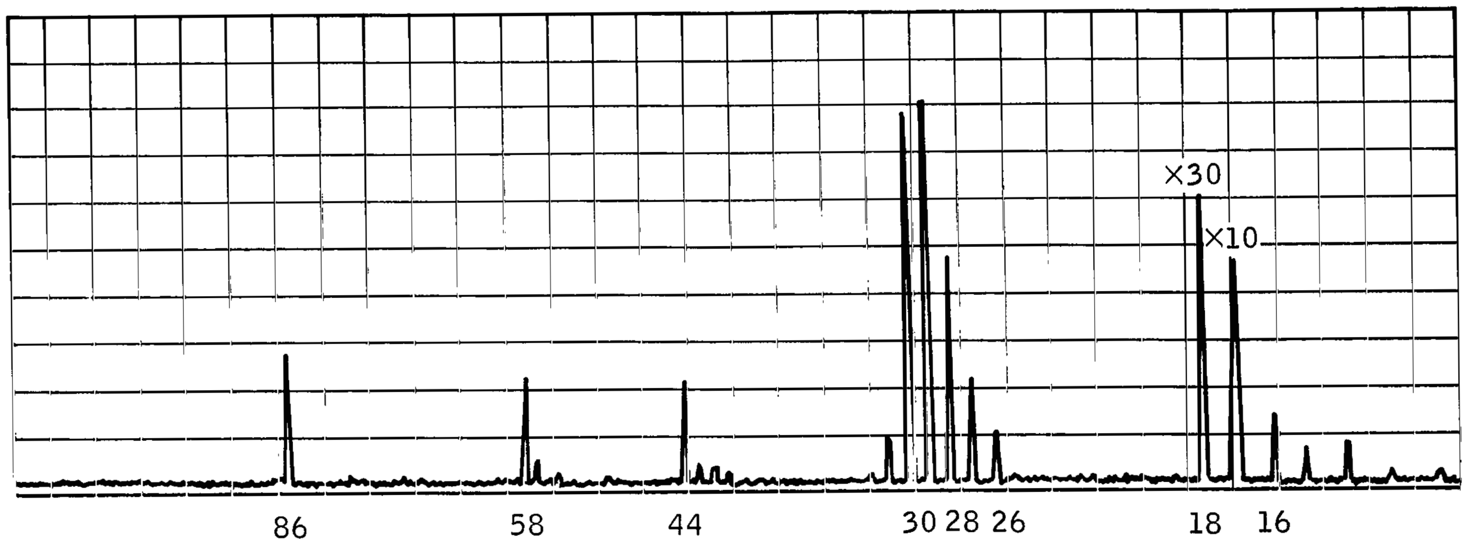


Figure B-2. - Mass spectrum of unidentified component.

(All peaks multiplied by one attenuation unless otherwise noted.)

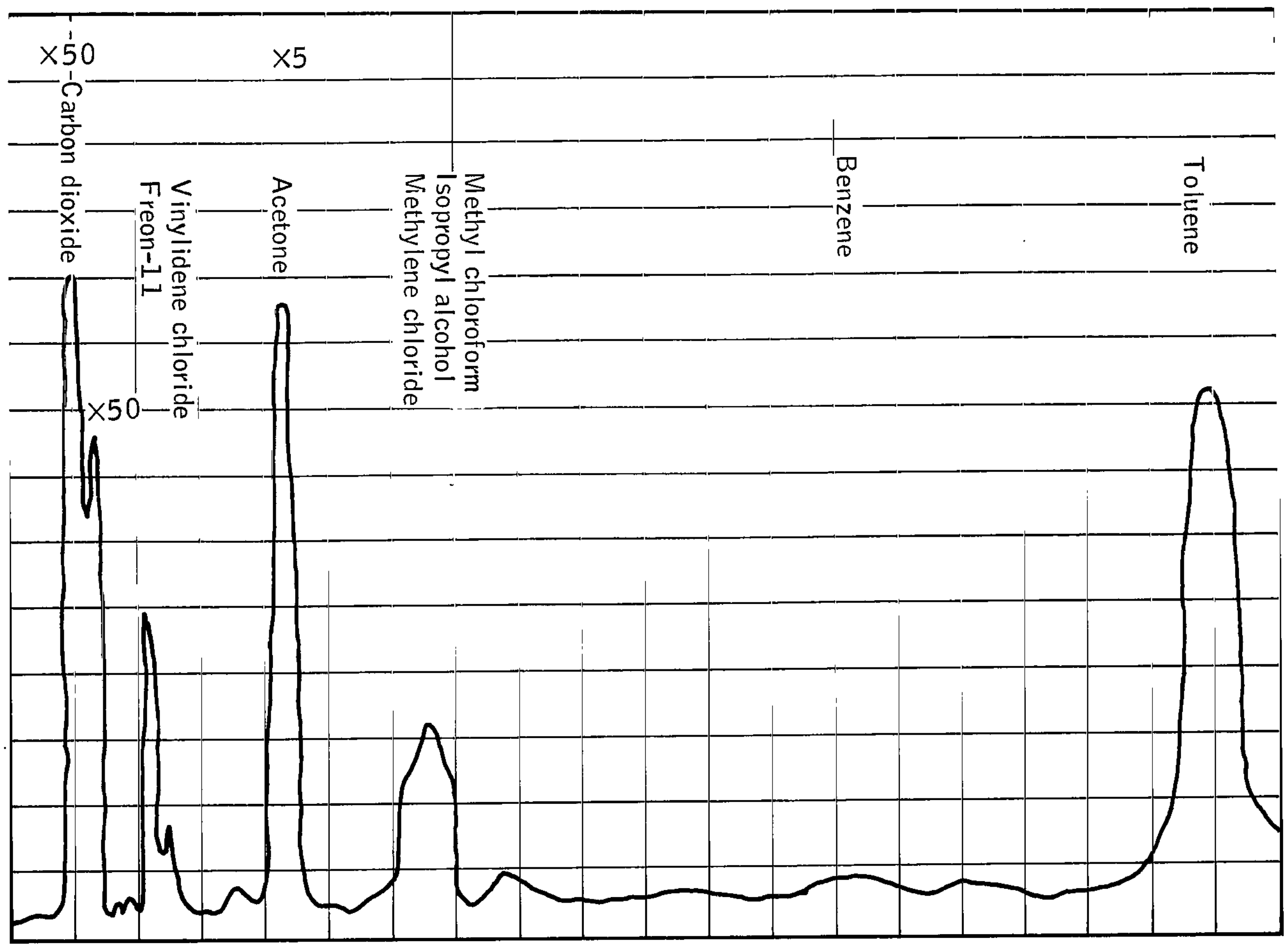


Figure B-3. - Chromatogram of nonaqueous desorbate from Spacecraft 20 charcoal.

APPENDIX C

ENVIRONMENTAL CONTROL SYSTEM PERFORMANCE

Mercury-Atlas 5 (MA-5)

System description. - The environmental control system (ECS), as flown in the MA-5 mission (capsule 9), was the same as that flown in the MA-4 mission (capsule 8A) with the following exceptions:

1. A check valve was installed in the inlet snorkel line between the negative-pressure relief valve and the inflow snorkel valve.

2. An outflow flapper-type check valve was installed in lieu of an outflow snorkel ball.

3. The primary and secondary oxygen bottles were pressurized to the specification pressure of 7500 psi instead of 3000 psi. This was the first flight with the specification pressure in the oxygen bottles.

4. A redesigned, positive-latching-type emergency oxygen rate handle was provided.

5. A functioning water separator was provided for the first time.

6. A chimpanzee was flown in lieu of a mechanical crewman simulator.

7. A 5-psi cabin differential-pressure relief valve was installed.

Suit and cabin pressures. - Suit and cabin pressures (fig. C-1) maintained their nominal values for the entire flight with no significant deviations. The new cabin differential-pressure relief valve functioned properly to maintain the 5-psi differential across the capsule structure until repressurization occurred during reentry. The approximate cabin leakage for the orbital phase of the mission was 1670 cc/min compared to the specification value of 1000 cc/min.

Oxygen partial pressure. - The data from the oxygen partial-pressure transducer used in the flight are questionable in view of an apparent off-scale reading during early powered flight and in view of a probable calibration shift during orbital flight. A meaningful postflight calibration of this conducer could not be made because of the fact that its useful life had been exceeded. The calibration shift found in this mission and in previous missions is believed to be due to the overpressure to which the transducer is subjected during the 5-psig suit-circuit leakage check. This 5-psig pressure overdrives the transducer beyond its maximum range of 15 psia, thereby, apparently producing a permanent calibration shift. No method has been devised to prevent the transducer from being subjected to this overpressure.

Oxygen-supply pressure. - The primary oxygen-supply pressure showed no oxygen usage from lift-off to just prior to the time of inflow-snorkel-valve opening and emergency-mode operation during descent. The expected oxygen usage was apparently offset by cabin-air-temperature effects on the bottles. The secondary oxygen bottle was similarly affected. Raising the bottle temperature 1° F will cause a pressure buildup, in the bottle, of approximately 18 psi. A cabin temperature differential of approximately 20° F (from a lift-off temperature of 81° F to an approximate orbital stabilization temperature of 101° F) would result in a 360-psi increase in bottle pressure. This increase in bottle pressure, along with telemetry accuracy, is sufficient to offset the expected primary bottle-pressure decay of 180 psi per hour.

Cabin air temperature. - The variation of the cabin air temperature with time is shown in figure C-2. The temperature, measured behind the instrument panel, indicated a general rise after launch to a level of around 100° F, which was maintained until the reentry heat pulse occurred. The approximate level of 100° F was higher than the expected range of 60° to 90° F because of poor cabin air circulation, possible marginal heat-exchanger performance based on the presetting of the water valve to a theoretical value, and variations in heat inputs from the cabin equipment. The cycling of cabin temperature showed some correlation with the external environment of exit-flight heating and of orbital flight in sunlight and darkness.

Suit-inlet temperature. - The suit-inlet temperature (fig. C-2) indicated that the suit heat exchanger functioned properly and maintained the temperature at an acceptable level up to approximately 1-1/2 hours from lift-off. At this time an unaccountable temperature rise started. This temperature rise may have been due to water freezing in the felt pad of the heat exchanger or possibly in the overboard-exhaust-duct opening. From approximately 02:45:00 to 02:55:00 ground elapsed time (g.e.t.), the plot indicated that the temperature had tended to level off. This level-off tendency, coupled with a level-off in the chimpanzee body temperature, would have permitted a third orbit insofar as the life-supporting system and the chimpanzee were concerned.

Coolant quantity. - Coolant-quantity measurements indicated no water usage, probably because of temperature effects on the coolant-quantity oxygen-pressurizing bottle.

Mercury-Atlas 6 Mission

System description. - The primary change in the ECS from the Spacecraft 9 (MA-5) configuration was the addition of a constant-bleed orifice to the suit circuit. This orifice provided a continuous oxygen flow greater than the pilot's anticipated metabolic requirement. The excess gas was exhausted into the cabin.

Countdown and launch. - The temperature of the main inverters increased to higher levels than expected during the countdown. This indicated that the Freon flow to the inverter coldplates, though adequate during precount checks, was inadequate during the final count. Temperatures of the 150- and 250-volt-ampere (VA) inverters at lift-off were 162° and 120° F, respectively.

The launch phase was normal. The cabin and suit pressures maintained a 5.5-psi differential above ambient during ascent and held at 5.7 and 5.8 psia, respectively.

Cabin and suit pressures. - Cabin and suit pressures maintained 5.7 and 5.8 psia, respectively, throughout the orbital flight. The decay in these pressures that had been observed in previous missions was absent for at least three possible reasons:

1. Low cabin leakage (less than 500 cc/min)

2. Possible excess oxygen, supplied by the suit-circuit constant-bleed orifice, exhausted into the cabin from the suit circuit

3. Possible leakage from the secondary oxygen supply

Oxygen partial pressure. - The oxygen partial pressure agreed with the suit pressure to within 0.5 psia and was consistently lower. Part of this difference was contributed to water vapor in the suit circuit, which contributed a partial pressure of approximately 0.3 psi. This was not included in the oxygen-partial-pressure measurement. A more careful calibration than those made for previous flights has resulted in a more satisfactory performance of this instrument.

Cabin air temperature. - The cabin air temperature, after the initial heating period, fluctuated as the spacecraft passed through the alternate periods of darkness and sunlight, as expected. The pilot reported that at least five attempts to reduce cabin air temperature, by increasing waterflow to the cabin heat exchanger, resulted in illumination of the excess-water light. This indicated that the cabin heat exchanger was operating near its maximum capacity for the existing conditions. Even so, the mean cabin air temperature was steadily reduced during the mission after the first hour in orbit.

Suit-inlet temperature. - The suit-inlet temperature (fig. C-3) varied between 65° and 75° F during the orbital phase. The pilot reported a coolant flow of 1.7 lb/hr to the suit heat exchanger and a stream exhaust temperature of 65° F. These values are both higher than anticipated and contradict each other, since freezing of the exchanger would be expected at this flow rate. No explanation of this anomaly can be offered at this time.

Inverter temperatures. - The 150- and 250-VA inverter temperatures (fig. C-3) increased steadily from launch values of 162° and 120° F, to 204° and 197° F, respectively, at landing. Postflight testing revealed that the check valve between the coolant supply and the coldplates was stuck in the closed position and would not permit coolant to flow to the coldplates in orbit. The coolant tank was charged with 25 pounds of water before the flight. The coolant-quantity indicating system (CQIS) showed a usage of 7.2 pounds. Postflight weighing indicated a usage of 11.8 pounds. The difference in calibration and final system temperatures can account for about 3.8 pounds of the 4.6-pound discrepancy, and the remaining 0.8 pound is considered to be instrument error.

Reentry and postlanding. - The maximum cabin temperature during this period was 103° F, which was satisfactory. The suit-inlet temperature increased to 86° F during the postlanding phase. This value was reasonable since the air temperature in the landing area was 76° F and the suit compressor raised the temperature in the suit circuit by approximately 10° F.

Concluding remarks. - Examination of the flight data and postflight checks of the ECS revealed several anomalies. As shown in figure C-4, the secondary oxygen supply exhibited an unexpected decay in pressure. This was first noted after approximately 01:50:00 g. e. t.; however, it is not known when this decay began, since the secondary oxygen bottle was serviced to about 8000 psig before flight and the pressure transducer had a maximum indicating value of only 7500 psig. Postflight tests indicated that the secondary system was free of leaks. Also, the postflight checks indicated a usage rate of only 0.13 lb/hr through the suit circuit compared with about 0.18 lb/hr obtained during prelaunch tests. Finally, the pressure-decay rate of the primary supply decreased to much lower than expected values during the last part of the mission; and during the last three-quarters of an hour in orbit, the decay rate of the secondary supply was essentially zero. No explanation for these anomalies can be offered at this time.

Mercury-Atlas 7 Mission

System description. - The ECS as installed in Spacecraft 18 represented the specification system in all respects. It differed from the ECS of Spacecraft 13 (MA-6) in two respects. First, the constant oxygen bleed employed in MA-6 was deleted; and, therefore, oxygen was supplied to the pilot on demand. Secondly, the oxygen partial pressure was measured in the cabin instead of in the suit circuit.

Launch. - The ECS operated properly during the launch phase. The cabin and suit pressures maintained the proper differential of 5.5 to 6.0 psi above ambient pressure during ascent and held at 5.8 and 5.9 psia, respectively.

Cabin and suit pressures. - The cabin and suit pressures decreased slowly during the orbital phase because of a cabin leakage of 1000 cc/min as established before flight. The pressure decay ceased at approximately 03:00:00 g. e. t., at which time the cabin pressure control valve began supplying oxygen to compensate for the cabin leakage. The cabin pressure was then maintained at 4.9 psia.

Oxygen partial pressure. - The only ECS measurement known to be inaccurate was that of the cabin oxygen partial pressure. Difficulty with the oxygen-partial-pressure sensor had been encountered during spacecraft preparations, and the final calibration was known to be only approximate.

Oxygen-supply pressure. - The secondary oxygen-supply pressure increased slightly during the flight. This can be attributed to the increase in supply-bottle temperature as measured during flight. Temperatures were identical for both the primary and secondary supplies, and the temperatures indicated 72° F at launch and 86° F at landing. The decay of the secondary oxygen supply experienced during MA-6 did not recur during this mission.

Cabin air temperature. - Although the cabin air temperature varied between 82° and 108° F, it remained above 100° F for much of the flight. The variation of cabin temperature with time is plotted in figure C-5. These high temperatures, though tolerable, were undesirable. The many changes in the cabin comfort-control-valve (CCV) setting prevented an accurate analysis of the effects of sunlight and darkness on cabin temperatures.

Suit temperature. - The suit temperature started to increase sharply at approximately 04:25:00 g. e. t. (fig. C-6). About 1 hour earlier (03:27:00 g. e. t.), the suit CCV setting had been advanced to give a high coolant-water flow rate. The ensuing temperature decrease resulted in a comfortable suit-temperature level until about 04:15:00 g. e. t., when a fluctuation, followed by a subsequent sharp increase in suit temperature, occurred. It is suspected that freezing of the suit heat exchanger occurred because of a high coolant-water flow rate for this 1-hour period. This resulted in a decrease of cooling efficiency just prior to and during the reentry phase.

Suit-inlet temperature, suit heat-exchanger steam temperature, and suit CCV settings are presented in figure C-6. The pilot reported that he found it difficult to determine the proper CCV setting which would maintain a comfortable level of suit temperature. Figure C-6 shows appreciable suit-temperature fluctuation resulting from changes in coolant-water flow rate (the coolant-water flow rate was controlled by the pilot's manual adjustment of the CCV).

Inverter temperatures. - The 150- and 250-VA inverter temperatures, shown in figure C-7, increased from 112° and 128° F at launch to 175° and 186° F, respectively, by 04:00:00 g. e. t. The temperatures appeared to be stable after this time. The rate of temperature increase appeared to decrease after the inverter coolant control valve was advanced from the no. 4 to the no. 5 position at 03:00:38 g. e. t. The corresponding change in coolant-water flow was from 0. 50 to 0. 64 lb/hr.

Coolant quantity. - The CQIS indicated a coolant-water usage of 10. 0 pounds when corrected for temperature change. Postflight inspection measured a usage of 10. 23 pounds. This agreement represented the most accurate CQIS measurements in flight to date. Coolant usage averaged about 2. 1 lb/hr over a period of 4 hours and 50 minutes, compared with a nominal flow rate of 1. 6 lb/hr.

Metabolic oxygen requirements. - MA-7 was the first orbital flight from which approximate values for the pilot's metabolic oxygen requirements could be calculated. Prelaunch oxygen consumption was determined to be 0. 0457 lb/hr or 261 standard cc/min (measured at 14. 7 psia and 70° F). During orbital flight, the pilot's metabolic oxygen consumption was calculated to be 0. 0722 lb/hr or 408 standard cc/min. These metabolic consumption rates were calculated from the oxygen pressure-decay rates of the primary oxygen tank after accounting for the 60 cc/min constant-bleed orifice of the suit demand regulator. The ECS design criteria for the metabolic rate of the pilot were 500 standard cc/min. This was based upon oxygen-usage data obtained during work of similar difficulty under one g. Pilot activity under weightless conditions demonstrated that weightless oxygen consumption rates were of a similar level as those which occurred under one g.

Reentry. - The performance of the ECS during reentry was normal. The pilot opened the inflow and outflow valves manually at 04:51:18 g. e. t. during descent. This placed the system in the postlanding mode. The emergency oxygen rate commenced at this time.

Concluding remarks. - The oxygen consumption rate obtained from this mission was the first indication of metabolic rate during weightlessness. The inflight value agreed closely with the rate found under one-g conditions for similar work, and the design criteria for oxygen consumption established for the ECS were validated.

The high cabin and suit-inlet temperatures were the only problems encountered during the flight. The inability of the cabin cooling system to reduce the cabin air temperature below 95° F was undesirable and may have been due to the size of the fan which delivered air to the cabin heat exchanger for cooling.

Some difficulty in obtaining the proper valve setting for the suit-inlet temperature control was experienced, primarily because of the lag in suit temperature with control manipulations. However, the high temperatures and humidity, as reported by the pilot, may have partially resulted from obstruction of the heat-exchanger evaporative surfaces by freezing. This partial freezing would slightly increase the evaporation pressure. The design conditions are for evaporation at 0.1 psia and 35° F. An increase of 0.1 psi in this design pressure would raise the corresponding evaporation temperature to 53° F, which in turn would significantly reduce the system capability to condense and collect water in the suit circuit. Flight data show that the suit steam-exhaust temperature was approximately 70° F instead of the expected 50° F, thus indicating that the evaporation temperature was probably near 55° F and that partial freezing may have been experienced.

Mercury-Atlas 8 Mission

System description. - The ECS for Spacecraft 16 (MA-8) was essentially the same as that for Spacecraft 18 (MA-7). The following minor changes were made:

1. The water-sealing device which was incorporated in the cabin pressure relief valve of previous spacecraft was removed.

2. The coolant-quantity-indicating and pressurization system was removed.

3. The coolant-water tank was pressurized from the cabin. The real-time determination of the coolant water remaining during the flight was dependent upon the preflight calibration of the comfort control valves and upon pilot reports of CCV settings.

The most significant change in the ECS was the relocation of the temperature-monitoring point from the steam exhaust to the domes of the suit and cabin heat exchangers. These temperatures were sensed on the exterior surface of the heat exchanger between the first and second pass of the evaporating water. An extensive heat-exchanger testing program indicated that temperature at this position was most representative of heat-exchanger operation and that the highest efficiency of the heat exchanger was obtained when this temperature was 55° ± 5° F. It was further concluded from these tests that a sudden drop in dome temperature below 45° F indicated excessive waterflow.

Prelaunch and launch. - Following pilot insertion, the suit-inlet environment was maintained at a temperature of approximately 60° F, and the cabin environment was maintained at 85° F. Preflight cooling was accomplished through evaporation of the ground-supplied refrigerant flowing through the heat exchangers and through the inverter coldplates at a total rate of about 34 lb/hr. The refrigerant was turned off at T-7 minutes in accordance with normal operating procedures. The pilot's metabolic oxygen consumption rate during the prelaunch phase was 1.17×10^{-3} lb/min (373 cc/min at 32° F and 14.7 psia). In comparison, the metabolic rate during the launch simulation was 0.94×10^{-3} lb/min (300 cc/min at 32° F and 14.7 psia). Both rates were computed from the pressure decay of the oxygen storage tanks. After cabin purge, the cabin oxygen-partial-pressure measurement was 1.0 psi below cabin pressure. This measurement was confirmed as inaccurate by a prelaunch chemical analysis, indicating 98-percent oxygen. The cabin oxygen-partial-pressure measurement was erratic and remained lower than cabin pressure throughout the mission.

During the launch phase of the mission, the cabin pressure relief valve ceased relieving at a differential cabin pressure of 5.9 psi above ambient. These pressures were at the upper limits of design tolerances, but they indicated proper functioning of the relief valve during the launch phase.

Cabin air temperature. - The cabin temperature cycled as a result of the radiation in solar heating at sunrise and sunset, as experienced in previous missions. Electrical equipment power-down and power-up caused the trend of the cabin temperatures to decrease and increase, respectively (fig. C-8). The dome temperature of the cabin heat exchanger was maintained in the range of 45° to 55° F during the mission, and the heat-exchanger outlet gas temperature indicated 40° to 45° F. The cabin CCV was set at position no. 4 at launch, but it was reduced to position no. 3 at 01:03:00 g.e.t. to provide assurance that freezing would not occur.

Suit-inlet temperature. - The suit-inlet temperature increased at a rate of approximately 0.5° F per minute during a portion of the first hour of the mission, but was reasonably stable at 86° F during the second hour, as shown in figure C-9. During this time, the pilot increased the suit CCV setting by one-half-position increments every 10 to 15 minutes from the preflight position of no. 4 to position no. 8 at 01:58:20 g. e. t. The dome temperature of the suit heat exchanger rose from 75° F at launch to 81° F and began a downward trend when the CCV was set to position no. 8. At 01:50:00 g. e. t. the suit CCV was reduced to position no. 3 by the pilot, on request, and a marked increase in the dome temperature of the suit inlet and suit heat exchanger resulted. This increase indicated that the waterflow rate at position no. 3 was inadequate for proper cooling; and, consequently, the valve was reset to position no. 8. At an elapsed time of 2 hours, the dome temperature of the suit heat exchanger dropped to 70° F and remained at this temperature for much of the remainder of the flight. The suit-inlet temperature began a downward trend at 2 hours elapsed time, and indicated 70° F at 3 hours after launch. At 4 hours elapsed time, the suit heat-exchanger dome temperature dropped rapidly to 45° F. The CCV was reset to position no. 7.5, and the dome temperature rose rapidly to the control range of 55° ± 5° F. This performance was in agreement with the heat-exchanger tests to be described, and it was concluded that optimum heat-exchanger operation occurred at a CCV setting between position nos. 7.5 and 8.

The pilot adjusted the comfort control valves in accordance with the preflight briefing and, in so doing, demonstrated that the suit temperature can be adequately controlled inflight.

The preflight calibration data indicated that the suit CCV should have been set at position no. 4 to obtain the required flow rate of 0.72 lb/hr. Postflight testing of this valve as flown revealed that a shift in the valve calibration had occurred so that the flow rate average of four tests at position no. 8 was 0.705 lb/hr, as shown in figure C-10.

Cabin leakage rate. - The cabin leakage rate determined during preflight tests was 570 cc/min at 19.7 psia. The cabin leakage rate during flight was 0.72×10^{-3} lb/min, based on an average pressure of 5.3 psia which would correspond to 630 cc/min at 19.7 psia. This inflight leakage was calculated from the time interval between cabin pressure seal-off (5.9 psia) and cabin pressure regulation (4.8 psia), which commenced at approximately 07:20:00 g.e.t. After this time, the total oxygen usage rate was 1.7×10^{-3} lb/min. Cabin leakage calculated from oxygen-supply pressure decay was equivalent to 475 cc/min at 19.7 psia. The pilot's oxygen usage was calculated to be 1.14×10^{-3} lb/min (364 cc/min at 32° F, 14.7 psia) during the first 7 hours of flight. These oxygen usage and cabin leakage rates were within the acceptable range determined before the flight.

Coolant quantity. - The total water expended for cooling the suit and cabin circuits and the inverter coldplates was 12.82 pounds, as determined in postflight tests. The quantity of condensate collected was 168 cubic centimeters, which was approximately the same amount for this 9-hour flight as on each of the two previous 4.5-hour orbital missions.

Reentry. - The performance of the ECS during reentry was normal. The system was changed to the postlanding mode of operation at 09:07:48 g.e.t. when the pilot manually opened the suit inflow and cabin outflow valves. The oxygen-emergency-rate flow was initiated automatically at this time, which conformed to normal procedures.

Comfort control valve. - In view of the time spent by the MA-8 pilot in initially attaining a low enough temperature in the suit circuit during the first orbit, some discussion of the CCV setting at launch and the plan for manipulating the valve during orbital flight is warranted.

Prior to the MA-8 mission, a series of tests using flight-configuration heat exchangers was conducted by NASA at Houston and by the contractor at St. Louis to develop an effective method of monitoring the heat-exchanger performance and to allow the pilot to use the maximum cooling provided by the heat exchanger. These tests yielded two significant results: (1) monitoring the temperature of the heat-exchanger dome provided a more positive and a rapid-response method of controlling the heat-exchanger performance; and (2) as waterflow rates were increased beyond optimum values, the cooling effectiveness did not increase. These tests provided substance to the hypothesis that partial freezing occurred in the heat exchanger during the MA-6 and MA-7 flights, where an excessive amount of cooling water was used and difficulty was experienced in obtaining satisfactory cooling.

The new monitoring instrumentation, resulting from the tests at Houston and St. Louis, was installed in Spacecraft 16 (MA-8) which had undergone its calibration and altitude-chamber tests and in Spacecraft 19 which was the backup spacecraft for MA-8 and which was to undergo altitude-chamber tests prior to the MA-8 flight. The results of the altitude-chamber tests of Spacecraft 19 were in substantial agreement with the results of the tests at Houston and St. Louis. The results of the altitude-chamber tests of Spacecraft 16 were inconclusive with regard to establishing proper system response, in that one of these tests indicated a normal control-valve setting while the other test required a setting which was considerably higher for equal cooling. The high usage rate experienced during one of the tests of Spacecraft 16 was regarded as another demonstration of nonoptimum flow rates, in view of the substantial amount of evidence from other tests.

As a result of the series of tests, a joint recommendation was made to the pilot that he go into orbit with the suit CCV set at position no. 4 and increase the setting by one-half-position increments at 10-minute intervals if the suit-inlet temperature warranted a change. The position no. 4 setting, using preflight calibration data for the flow rate valve, corresponded approximately to the theoretical flow rate needed and to the valve position established for adequate cooling in one of the two altitude-chamber tests of Spacecraft 16. This setting was used by the pilot during the early part of the mission, and it was found to be too low for adequate cooling. Postflight tests showed that the valve flow passage was restricted by foreign material, thus requiring a higher than anticipated setting of the flow valve to provide adequate cooling.

Concluding remarks. - The use of the heat-exchanger dome temperature as the criterion for adjusting the coolant-water flow rate, tested under one-g conditions, was confirmed under zero-g conditions as a rapidly responding parameter superior to the methods used in previous missions. It was confirmed, from data obtained on this flight, that insufficient coolant-water flow would cause the suit-inlet and suit heat-exchanger dome temperatures to rise sharply, and that excessive coolant-water flow would decrease the dome temperature rapidly. The flow rate at which a rapid decrease in dome temperature was impending represented the flow rate necessary for the maximum efficiency of the heat exchanger.

The difficulty encountered with the elevated suit-inlet temperature was indicated after the flight by a shift in the previously calibrated flow for the suit CCV.

The four postflight calibration tests of the CCV in the flown condition presented a significant envelope of variation for flow rates at a given valve setting, as shown in figure C-10. The valve was disassembled, and an inspection of the valve components revealed flakes of dried lubricant on the valve stem and in the valve seat. These flakes were large enough to cause considerable restriction of the flow through the valve. The valve was ultrasonically cleaned and relubricated on the O-ring and on the male threads of the valve body. Three calibration tests of the cleaned suit CCV showed that the calibration had returned to near the preflight values. It was evident from these data that the flow rates of the cleaned valve were relatively consistent and, therefore, predictable for any setting. Postflight testing of the cabin CCV and of the inverter CCV did not reveal a significant calibration shift.

The MA-8 mission was the first manned orbital mission during which positive control of the suit temperature was demonstrated. The comfort of the pilot during the mission was evidenced by the 168 cubic centimeters of condensate collected, which was a measure of the level of perspiration experienced. During the two previous three-pass missions, which involved a duration of approximately one-half that of MA-8, the pilots experienced high specific humidity and excessive perspiration; and, consequently, the same magnitude of condensate was collected. A low specific humidity at the suit heat-exchanger outlet is indicative of a properly functioning suit cooling system.

A high cabin temperature has been experienced on all orbital missions. The cabin heat exchanger for this flight is known to have been efficient, as evidenced by the 40° F gas-outlet temperature. However, increased heat loads since the design of the cabin cooling circuit have relaxed the initially acceptable temperature limits, and the MA-8 system performed within presently acceptable ranges.

Mercury-Atlas 9 Mission

System description. - The ECS was originally designed for an 18-orbital-pass capability, but three major changes from the ECS configuration used in the MA-8 mission were required for a mission of longer duration. A second bottle containing 4 pounds of oxygen was installed in parallel with the primary oxygen bottle. This addition increased the previous total oxygen supply, including the primary and a secondary source, by 50 percent. The quantity of LiOH was increased from 4.6 to 5.4 pounds, and the amount of charcoal was decreased from 1 to 0.2 pound. Also, an auxiliary LiOH canister was used in the suit-outlet line during manned prelaunch tests to conserve the LiOH in the flight canister. An additional tank with a nominal capacity of 9 pounds of coolant water was installed to complement the original 39-pound-capacity tank. This new tank was serviced before launch with 9.3 pounds of water.

In addition to system changes to accommodate the extended flight, the following modifications from the Spacecraft 16 (MA-8) configuration were required to improve system performance:

1. Because of the partial blockage problem experience in MA-8, a suit bypass CCV was installed, for redundancy, in parallel with the existing CCV.

2. A condensate trap, shown in figure C-11, was installed at the suit-inlet port to extract free condensate water which would adhere to the inner wall of the water separator and thus escape extraction. The inner wall of this trap was of a wicking material which had the property of passing water, but not gas, when the material was wet.

3. The cabin pressure relief valve was equipped with a water-sealing device to enable the pilot to lock this valve and prevent sea water from entering the spacecraft after landing.

4. A sensor for measuring carbon dioxide partial pressure P_{CO_2} was installed in the suit circuit to indicate CO_2 concentration and to actuate a warning tone and light at a P_{CO_2} value of 8 mm Hg. The suit inlet for emergency oxygen flow was repositioned upstream of the P_{CO_2} sensor and thereby permitted a purging of the P_{CO_2} sensor with 100-percent oxygen to verify its operation.

5. The suit circuit, from the water separator to the junction with the suit-inlet flexible hose, was insulated to reduce heat loss, and the LiOH canister was insulated to minimize condensation. The dome temperature, which was monitored by the pilot during the MA-8 mission, was also monitored on telemetry and recorded for the MA-9 mission.

Launch. - The suit-inlet temperature was 55° F at pilot insertion into the spacecraft and gradually increased to 61° F during the Freon cooling period prior to lift-off. The oxygen partial-pressure P_{O_2} readout for the cabin was 0.6 psi below total cabin pressure after the cabin was purged with oxygen at the launch site. A gas analysis at this time indicated 98-percent oxygen at a cabin pressure of 14.9 psia. At lift-off, the P_{O_2} was 0.4 psi below cabin pressure. Measurements during the countdown did not indicate a change in the P_{O_2} amplifier calibration. The telemetry readout (percent full scale) from the P_{CO_2} sensor was negative during the countdown period. This negative value is normal and results from the logarithmic voltage output characteristic of the sensor; that is, the zero reference of the sensor corresponds to a finite level of CO_2.

The ECS operated normally during powered flight, and the cabin pressure held at 5.5 psia. The dome temperatures of both heat exchangers decreased during powered flight. This decrease indicates a rapid response of the cooling circuits upon reaching altitudes at which the water would boil at lower temperatures and pressures.

Suit pressure. - The pilot reported that suit pressure was at times slightly negative with respect to cabin pressure and that it was occasionally necessary to breathe deeply to correct this condition. This action caused the demand regulator to supply oxygen to the suit circuit and, therefore, increase the pressure. This condition could be corrected for operation with the faceplate closed by decreasing the negative pressure required to activate the demand regulator. A configuration using lower negative pressure was tested early in the ECS development program, and it was discovered that operation with the faceplate open caused inadvertent operation of the demand regulator and, therefore, excess oxygen usage.

Oxygen partial pressure. - The P_{O_2} at orbital insertion was 0.3 psi below cabin pressure. The cabin pressure decayed to 5.1 psi at approximately 03:00:00 g.e.t., at which time cabin pressure regulation began. The cabin pressure, which is influenced by cabin temperature, varied between 5.0 and 5.2 psi during the orbital phase. This range of pressure was within the specified tolerance of the cabin pressure regulator.

Calculations of total oxygen consumption showed that 2.18 pounds of oxygen were used from 03:05:00 to 34:05:00 g.e.t. These calculations were based on tabulated data for primary-oxygen-bottle pressure and on estimates of bottle temperature. The time of 03:05:00 g.e.t. was chosen to correspond to the estimated time at which cabin pressure regulation began.

Carbon dioxide. - Postflight analysis of the LiOH canister revealed that 2.90 pounds of CO_2 were absorbed in the canister. This CO_2 was produced during the launch attempt on May 14, the successful launch period of May 15, the mission orbital phase, and (of minor significance) during the major simulated launch and the simulated flight. Based on a total canister-usage time of 42.4 hours and neglecting the small amount of CO_2 produced during the simulated flight and launch, the pilot's average CO_2 production rate was 261 cc/min. Chemical analysis of the two auxiliary canisters used in series with the flight canister during the countdown showed average CO_2 rates of 264 and 285 cc/min. Since these rates were of the same order of magnitude as that calculated for the flight canister, it was concluded that the CO_2 rate of 261 cc/min was believable for the orbital period. Since the faceplate was estimated by the pilot to have been open approximately one-third of this time, an undetermined amount of CO_2 escaped into the cabin. This CO_2 would raise the cabin P_{CO_2} reading and would leak overboard. Since there was no measurement of P_{CO_2} in the cabin, it was impossible to estimate from a systems standpoint the quantity of CO_2 that was lost when the faceplate was open. An estimate of the maximum P_{CO_2} in the cabin based on physiological considerations is contained later in this section. Based on this estimate, a negligible amount of CO_2 was lost through the open faceplate. In any case, the figure of 261 cc/min for the average CO_2 production rate was a minimum value.

Carbon dioxide partial pressure. - At 32:15:00 g. e. t., a marked increase occurred in the P_{CO_2} level, which reached 2.8 mm Hg by 32:45:00 g. e. t. Prior to 32:15:00 g. e. t., the P_{CO_2} was less than 0.1 mm Hg. At 32:44:20 g. e. t., the pilot selected the emergency-rate mode of oxygen flow to purge the P_{CO_2} transducer and to test its readout validity. This mode was in operation for 25.5 seconds. The P_{CO_2} decreased to approximately 1.9 mm Hg at 32:44:56 g. e. t., thereby verifying the validity of the readout. The P_{CO_2} again continued to rise gradually until 33:59:30 g. e. t., which is the exact time of ignition for the first retrorocket. At this time, the P_{CO_2} indication decreased sharply to a negative voltage output. Test experience has shown that the sensor indication will not normally change as rapidly as it did during this brief period. Postflight calibration of the P_{CO_2} sensor did not indicate a significant shift in its calibration.

The LiOH canister had been tested extensively under normal gravity conditions, and the operational life of the 5.4-pound charge of LiOH was well established. Prior to pilot insertion on the day of the launch, the effective life of the LiOH canister had been reduced by approximately 8 hours because of usage during systems tests conducted for the unsuccessful and successful launch attempts of May 14 and 15, respectively. Calculations made after a launch attempt indicated that the canister capability was sufficient to accommodate the mission with at least a 3-hour margin. Consequently, it was known that P_{CO_2} might build up during the latter part of the mission, since the useful life of the canister is defined as the time required to reach a P_{CO_2}

level of 8 mm Hg rather than the time until the first indication of CO_2 buildup. Postflight chemical analysis of the canisters (table C-I) indicated that the pilot's CO_2 production was below the design level of 400 cc/min and that there were 2.18 pounds of LiOH remaining in the canister. However, postflight analysis showed that some channeling of flow occurred in the flight canister. This channeling of flow could explain the indicated CO_2 buildup.

Cabin air temperature. - At approximately 06:22:00 g.e.t., the pilot began the cabin temperature evaluation by turning off the cabin fan and coolant-water flow to the cabin heat exchanger. At this time the cabin temperature was oscillating slowly between 90° and 95° F, as shown in figure C-12. The maximum cabin temperature observed during the period of evaluation was 103° F, which occurred during a powered-up condition; and the minimum temperature was 84° F, which occurred during an extended period while the spacecraft was powered-down. The cabin temperature during the experiment oscillated approximately ±5° F, and the temperature trend was influenced by the cycling of electrical power and sunlight heat loads. The average cabin temperature during the evaluation was 90° F. At 32:05:00 g.e.t., the pilot terminated the evaluation and turned on the cabin cooling system, as planned, in preparation for reentry. The cooling system responded rapidly, as evidenced by a drop in the cabin heat-exchanger outlet temperature.

Suit temperature. - The suit heat-exchanger dome temperature was the measurement used by the pilot to evaluate the operation of the cooling circuit. The pilot made a considerable number of adjustments to the suit CCV, but was unable to attain stabilization of this temperature. The requirement for this frequent adjustment presented the major anomaly in the ECS performance during the MA-9 mission. The dome temperature, which ideally should be maintained between 40° and 55° F, is a function of the coolant-water flow and the heat load on the heat exchanger. The heat load can be affected by the pilot's activity level, the helmet visor position, and the cabin temperature. The dome temperature is, by nature, sensitive to small changes in the coolant-water flow rate, and the resulting dome temperature changes can be rapid. However, the suit-inlet temperature was relatively stable and not appreciably affected by the dome temperature fluctuations. The average suit-inlet temperature for the entire mission was from 6° to 15° F lower than those experienced during any previous flight.

Cabin leakage rate. - The cabin leakage rate determined several days prior to the first launch attempt was 485 cc/min at 19.7 psia and 70° F. This leakage rate was determined by a stabilized flowmeter measurement. Subsequent to this check, the hatch was removed and replaced several times. A brief leakage check on the day of the launch showed no detectable leakage. This check, however, was quite gross and had no particular significance. A gross leakage-rate determination after launch was obtained from the reading of cabin pressure decay from relief-valve seal-off pressure to the point when cabin pressure regulation began. This determination showed a leakage of 510 cc/min corrected to 19.7 psia and 70° F.

Extrapolating the prelaunch leakage determination from the sea-level condition to the orbital condition showed that the equivalent cabin leakage rate during the orbital period would be 0.528×10^{-3} lb/min. This leakage rate was determined by computing the equivalent orifice area required to leak the 485 cc/min at sea level and by then using this orifice area to compute the choked orifice flow during the orbital period.

Based on the average CO_2 production rate and on a respiratory quotient of 0.83, the pilot's oxygen consumption from 03:05:00 to 34:05:00 g.e.t. should have been 1.84 pounds. The leakage for the same period, based on extrapolated precount measurements, should have amounted to 0.98 pound. The sum of the computed pilot oxygen consumption and the computed leakage rate was therefore 2.82 pounds, as compared with the calculated bottle depletion of 2.18 pounds.

A high level of confidence should be placed in the calculations of total oxygen usage and of pilot metabolic consumption. The values for leakage rate are based on prelaunch measurements, and the equivalent orifice areas to produce these leakage rates are very small. It is possible that any given leakage area could change after measurement, which would, of course, alter the leakage rate. Therefore, it is concluded that the leakage rate decreased after the prelaunch measurement. From cabin pressure-decay data, there is evidence that a reduction in leakage occurred soon after the first hour of flight. The average oxygen leakage rate, based on the difference in computed total usage and computed pilot consumption, was 0.183×10^{-3} lb/min. In addition to the oxygen leakage, there probably was a small quantity of CO_2 and water-vapor leakage which would depend on the partial pressures of these parameters in the cabin.

At 1:35:00 g.e.t., the pilot opened the hose clamp on the condensate-trap water-outlet line. He did not observe water flowing through the transparent line and closed the clamp, as planned, after about 30 minutes. At 8:00:00 g.e.t., the pilot again removed the clamp and soon observed the flow of condensate water. The trap remained in operation until approximately 12:22:00 g.e.t., when the clamp was again closed. The clamp was reopened at 26:56:00 g.e.t. for a 6-minute period. Although it had been planned to use the trap more extensively during the flight, failure of the condensate-transfer pump prompted the pilot's decision to discontinue its use. This action was taken to minimize the possibility of water leakage through the condensate tank vent into the cabin. The pilot observed that the water separator cycled throughout the flight. The fact that water flowed from the condensate trap, which is downstream from the water separator, indicated that the water separator was not efficiently removing condensate. Data from prelaunch and postflight tests, as well as results from previous missions, support a belief that water can move freely around the sponge in the water separator under weightless conditions, thereby reducing its water-removal effectiveness.

Coolant quantity. - The cooling water used during the flight was determined by postflight testing to be 23.38 pounds. The average coolant flow rate for both the cabin and suit circuits was approximately 0.6 lb/hr. This rate was determined by using estimates of CCV settings obtained from the onboard voice transcripts and by using postflight measurements of coolant water remaining in the system.

Reentry. - The ECS operation during reentry and landing was normal. The snorkels actuated automatically at 34:15:25 g.e.t. and at an altitude of 18 500 feet. The pilot reported that he was comfortable during the postlanding phase. The flapper diaphragm of the cabin air outflow valve, which is normally open after the water-sealing device in the cabin pressure relief valve has been engaged, would ordinarily be closed by the water pressure. However, a small quantity of sea water, reported to have entered the cabin upon landing, could have passed through this valve prior to the closing of the diaphragm.

Concluding Remarks

The only major anomaly in the ECS was the inability of the pilot to stabilize the suit heat-exchanger dome temperature. A possible solution for this problem is presently under investigation. In all other respects, the operation of the system was satisfactory.

The results of the cabin temperature evaluation show that cabin cooling of the Mercury spacecraft was not required during a powered-down condition.

Since the pilot observed waterflow from the condensate trap, it must be concluded that an unknown percentage of condensate bypasses the water separator in a zero-g environment. Because of the configuration of the condensate trap, it can be further concluded that the condensate water was transported along the walls of the ducting.

Unlike the MA-8 mission, the postflight calibration of the comfort control valves showed no significant shift from the preflight calibration in the flow rate for a given valve setting. Prior to flight, the comfort control valves in the MA-9 spacecraft had been thoroughly cleaned, and their lubrication procedure had been improved. In addition, the coolant water was passed through a 0.15-micron filter before being transferred to the coolant-water tank.

TABLE C-I. - POSTFLIGHT ANALYSIS OF LITHIUM HYDROXIDE CANISTER

Source (a)	LiOH remaining, lb	H_2O present, lb	CO_2 disclosed, lb	CO_2 disclosed, liters	Period used, hr	CO_2 absorption rate, cc/min, STP
Flight canister	2.17	0.27	2.89	665	42.4	261
Auxiliary canister no. 1	1.32	.17	.188	43.5	2.75	264
Auxiliary canister no. 2	1.14	.21	.36	83.4	4.88	285

[a] Auxiliary canister no. 1 was used in the simulated flight no. 3, and canister no. 2 was used in the final launch simulation. The MA-9 astronaut was in the spacecraft during the total time shown for all canisters.

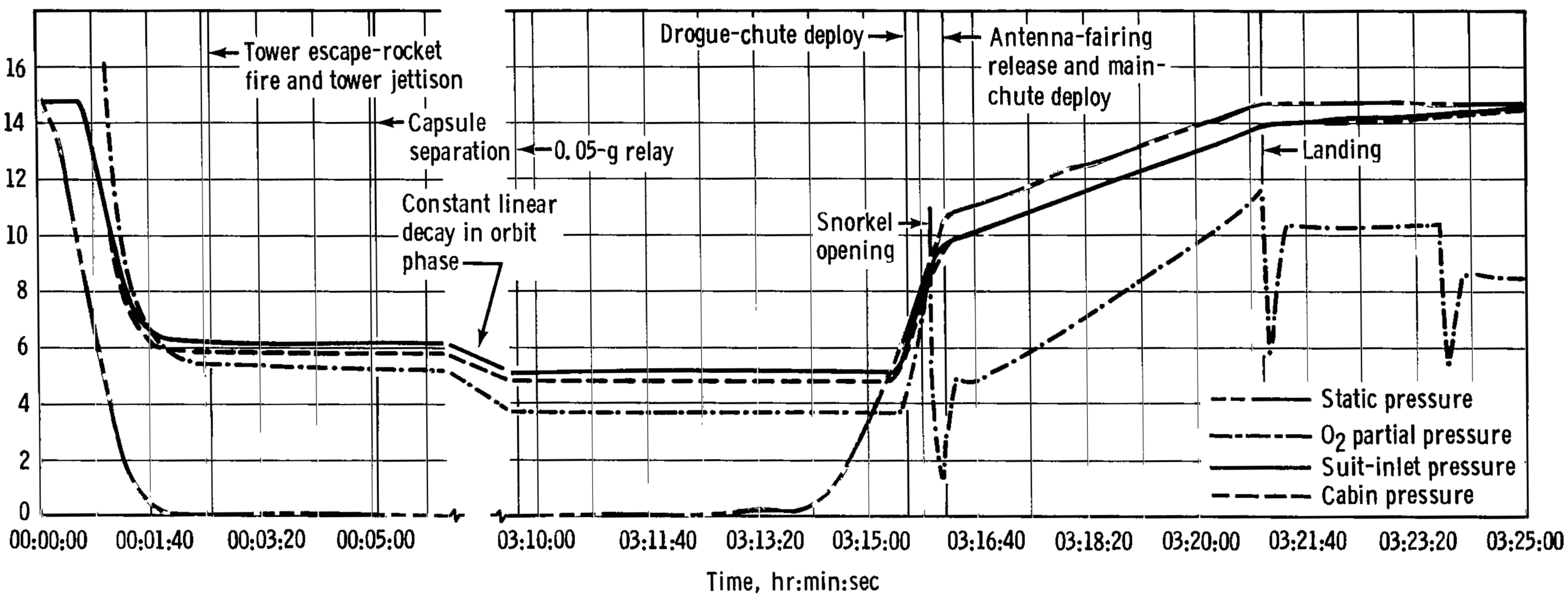


Figure C-1. - Variation of cabin, suit, static, and oxygen partial pressures with time.

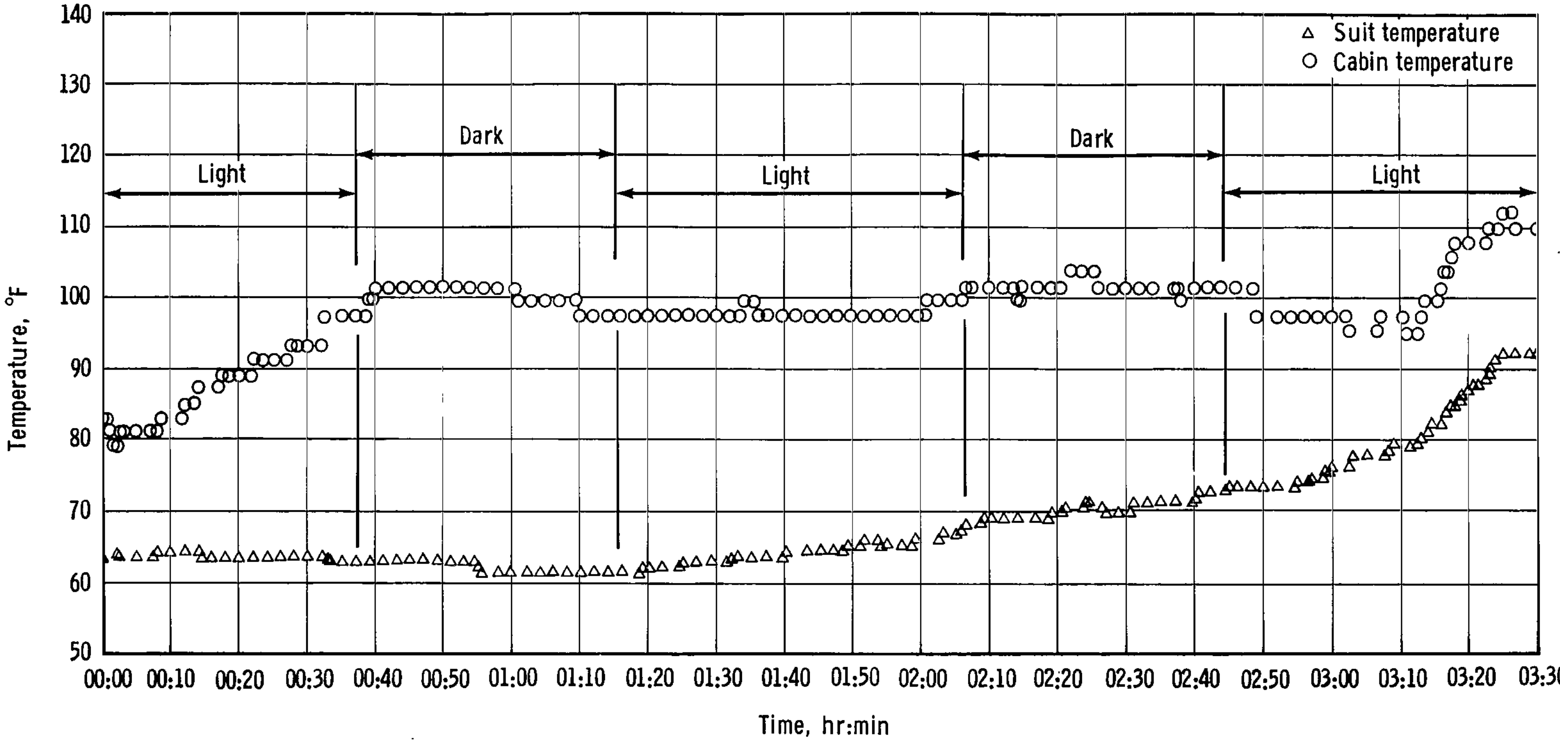


Figure C-2. - Variation of cabin and suit temperatures with time.

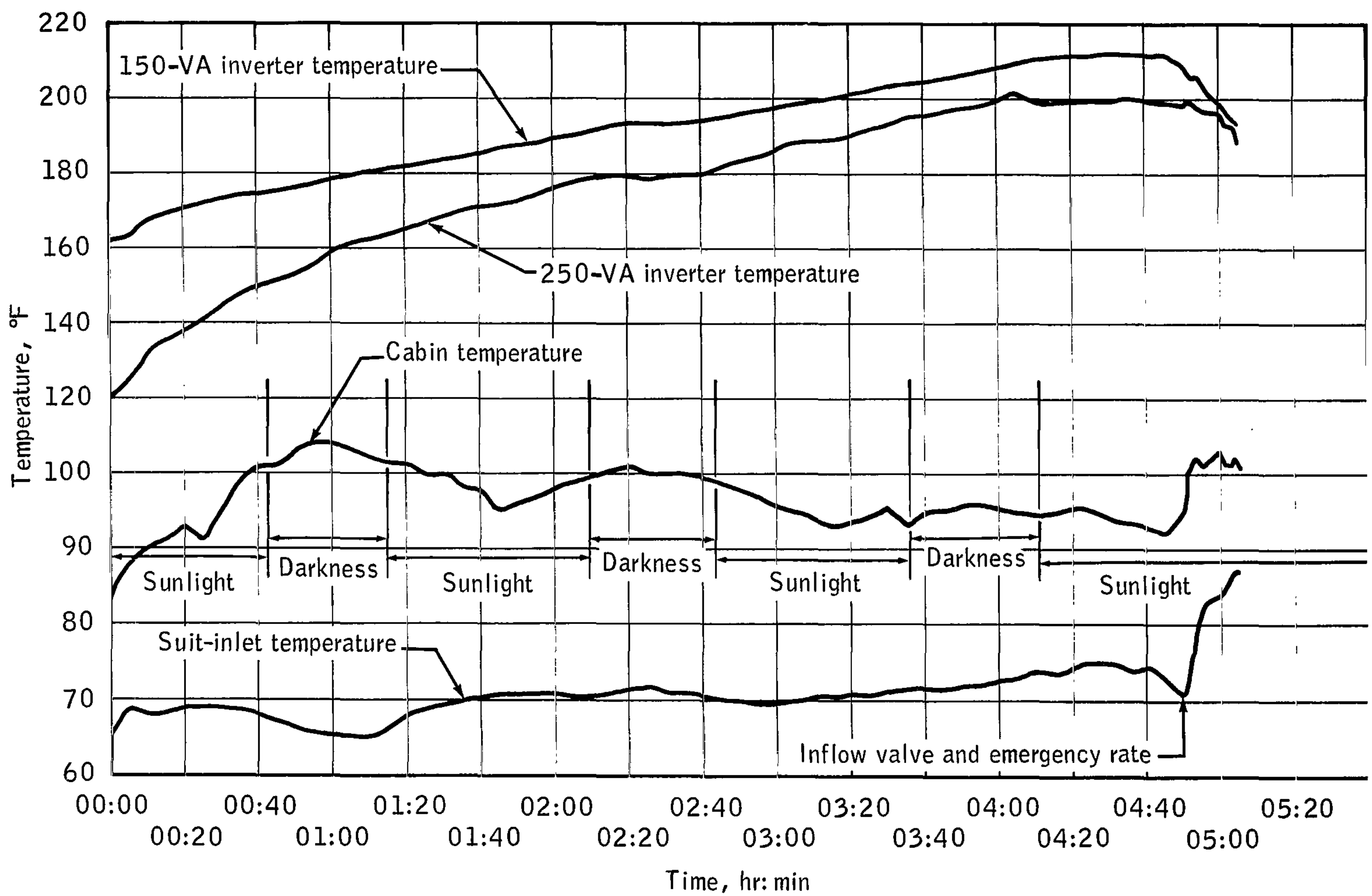
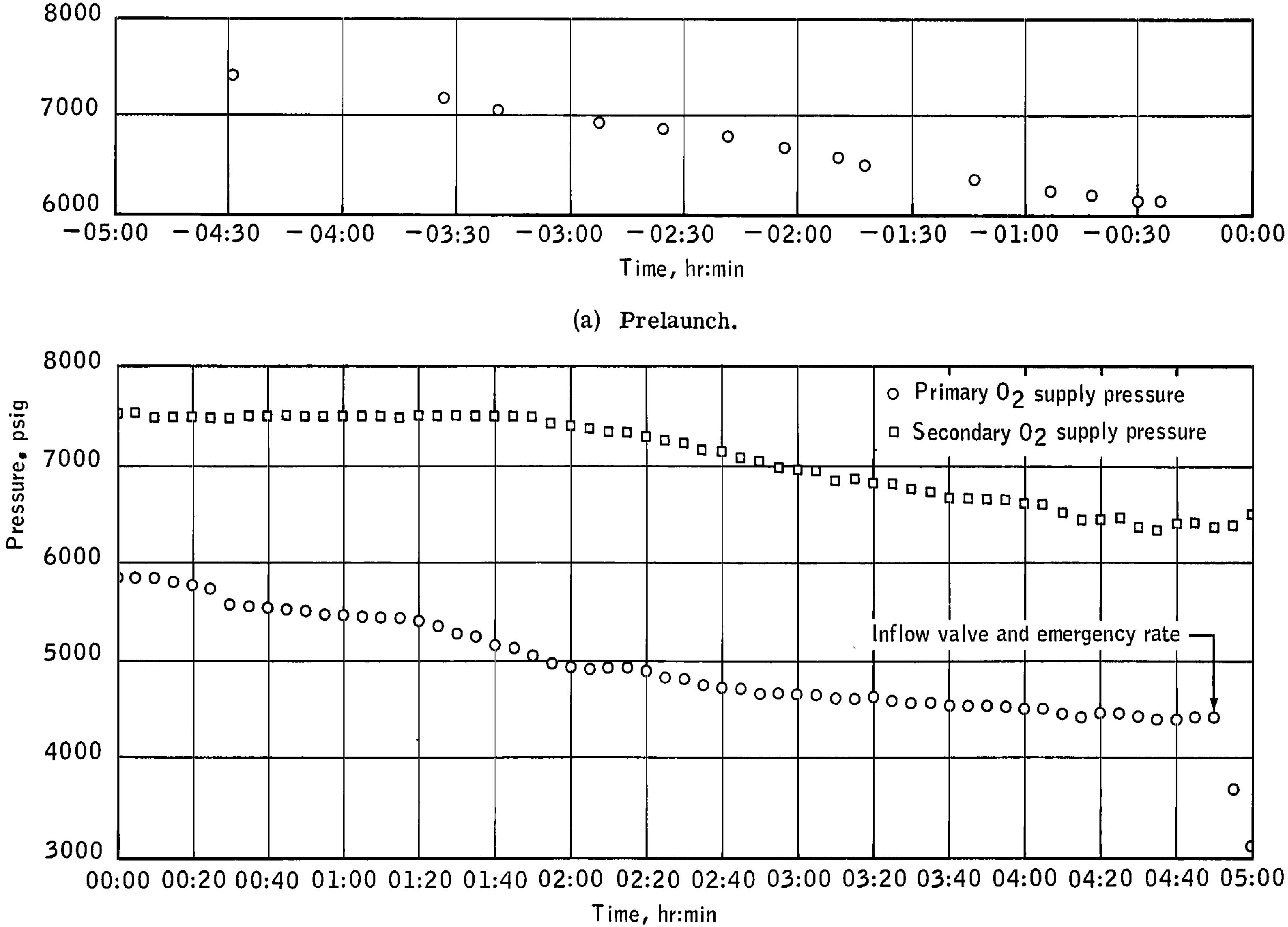


Figure C-3. - Variation of suit, cabin, and inverter temperatures with time.



(a) Prelaunch.

(b) During flight.

Figure C-4. - Variation of primary and secondary oxygen pressures with time.

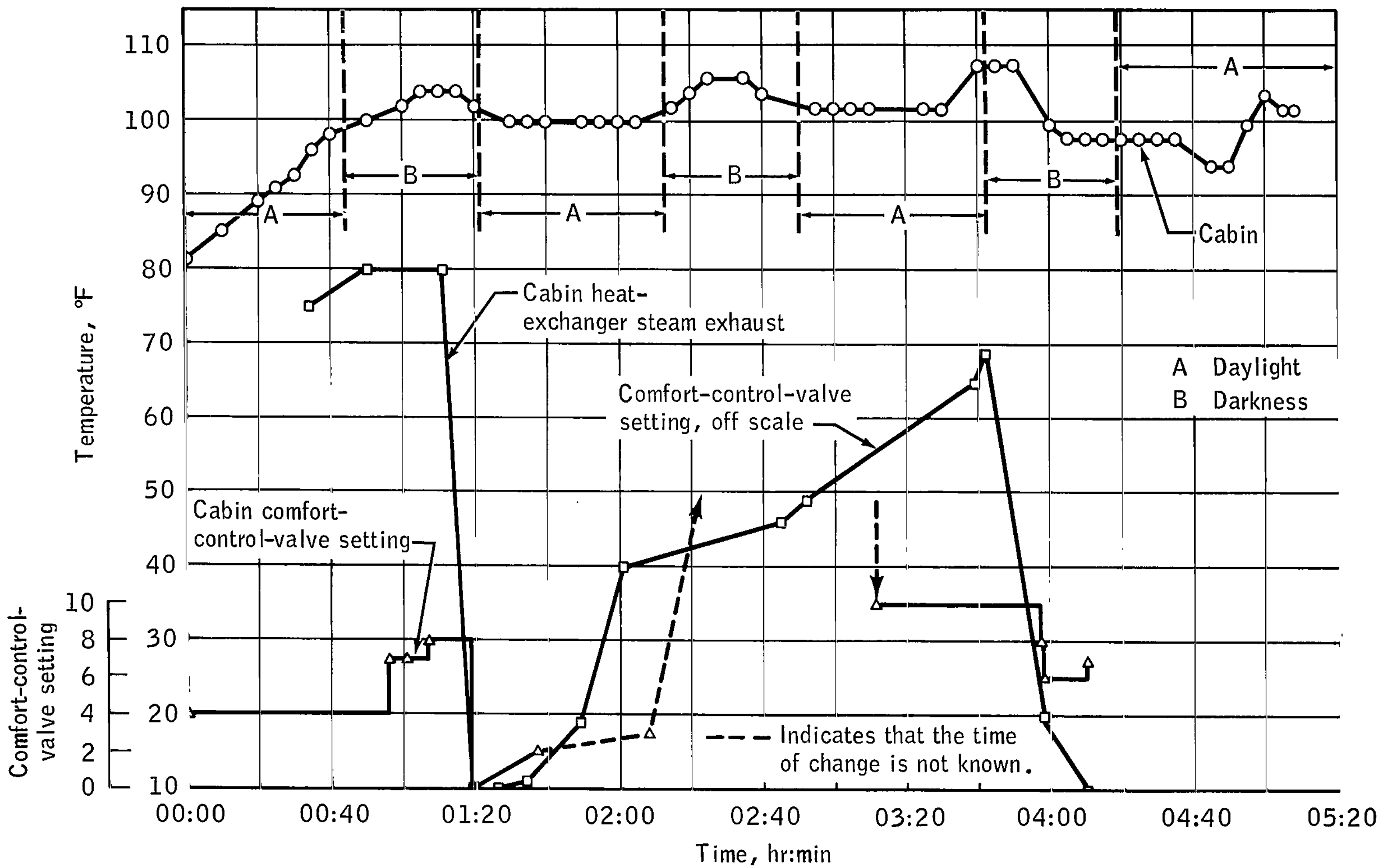


Figure C-5. - Variation of cabin air, cabin heat-exchanger steam-exhaust temperatures, and associated comfort-control-valve settings with time.

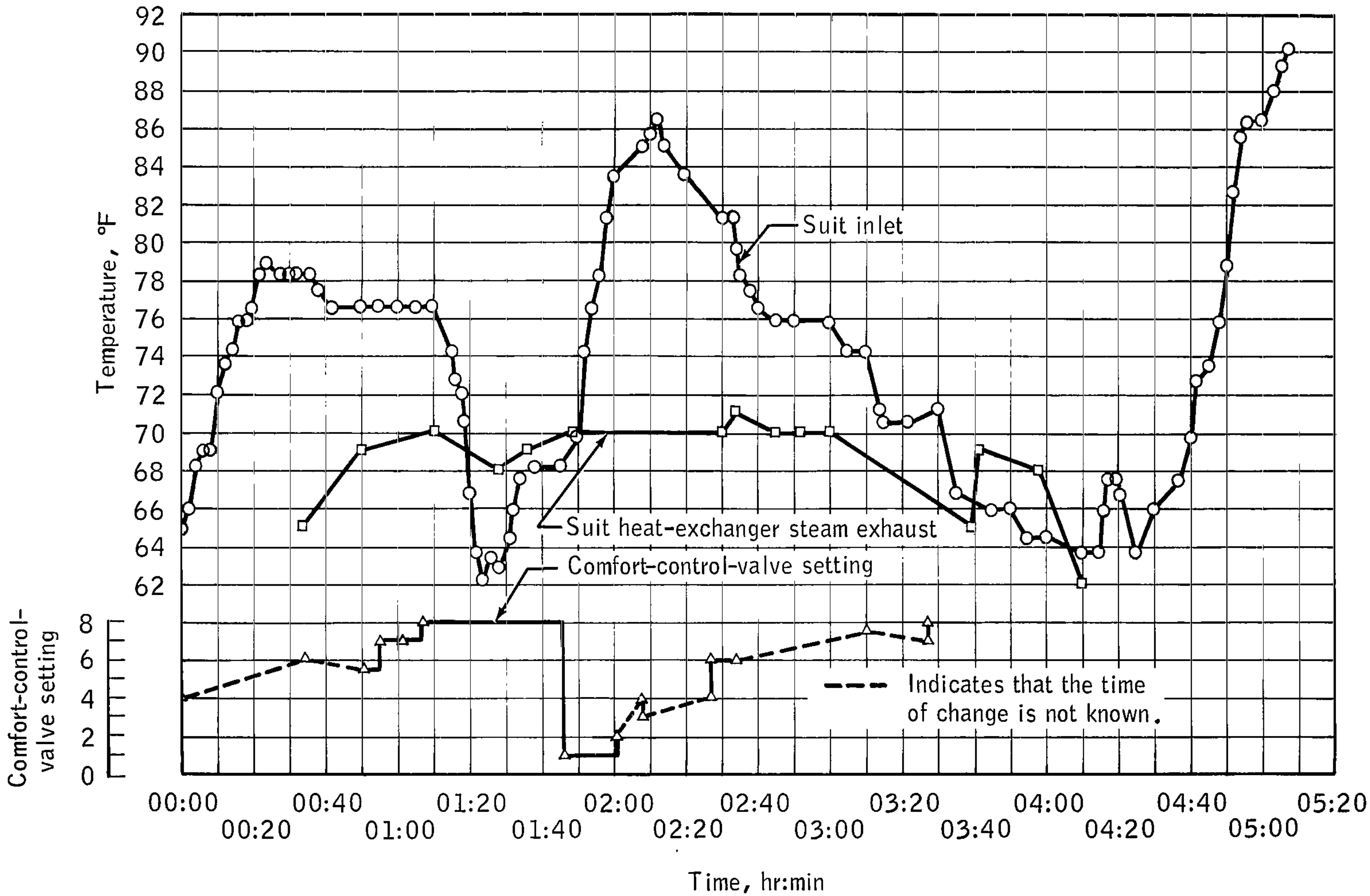


Figure C-6. - Variation of suit inlet, suit heat-exchanger steam-exhaust temperatures, and associated comfort-control-valve settings with time.

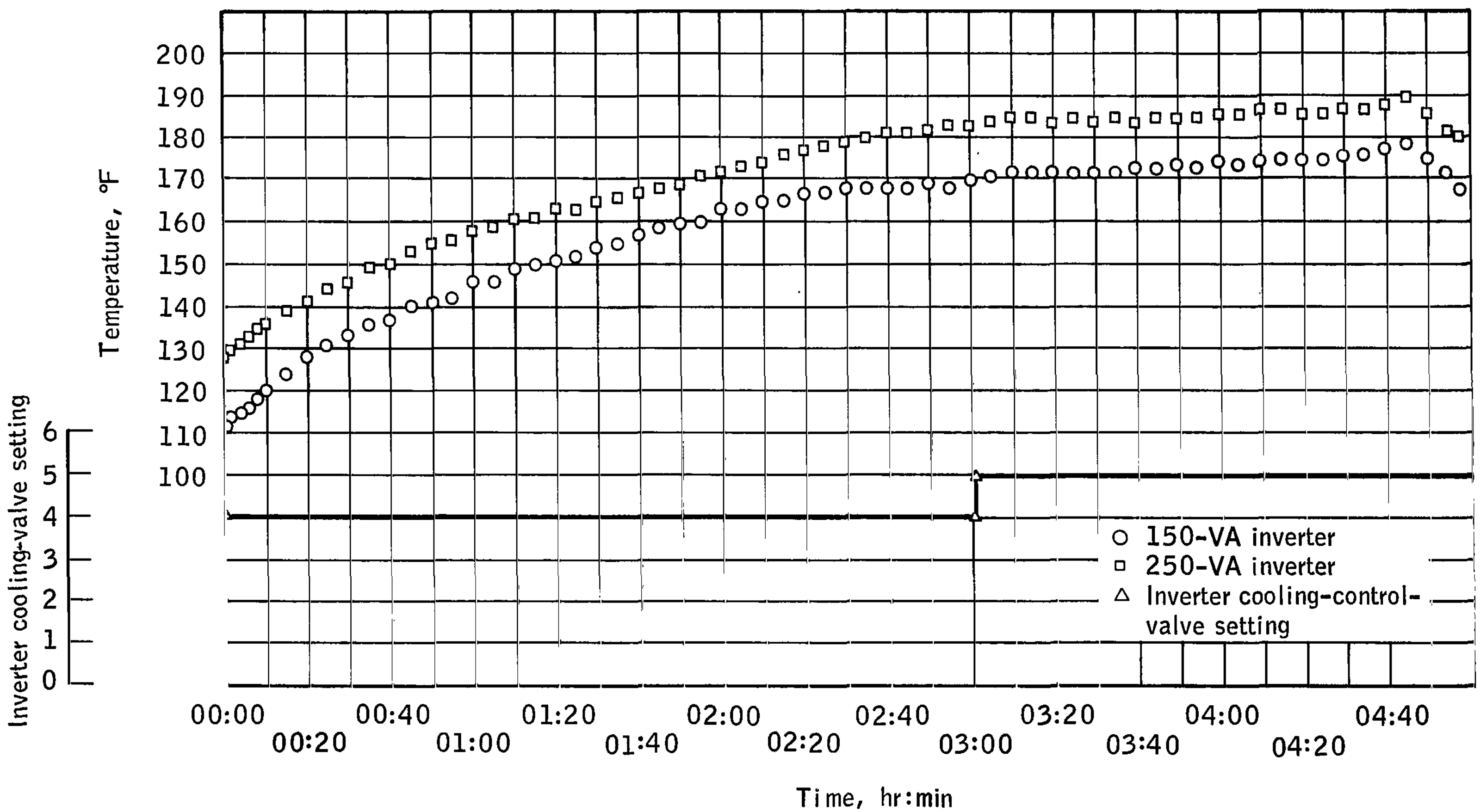
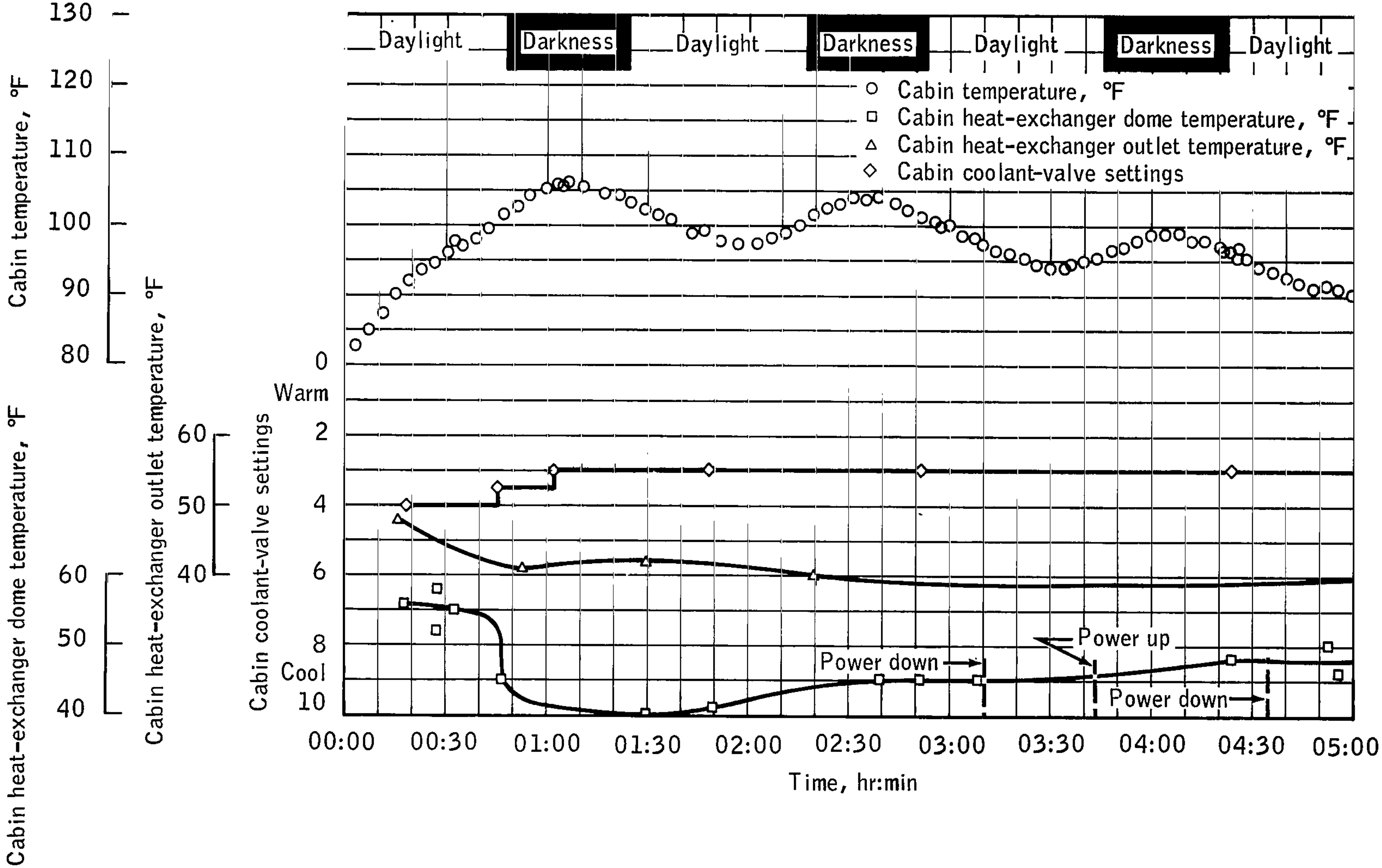
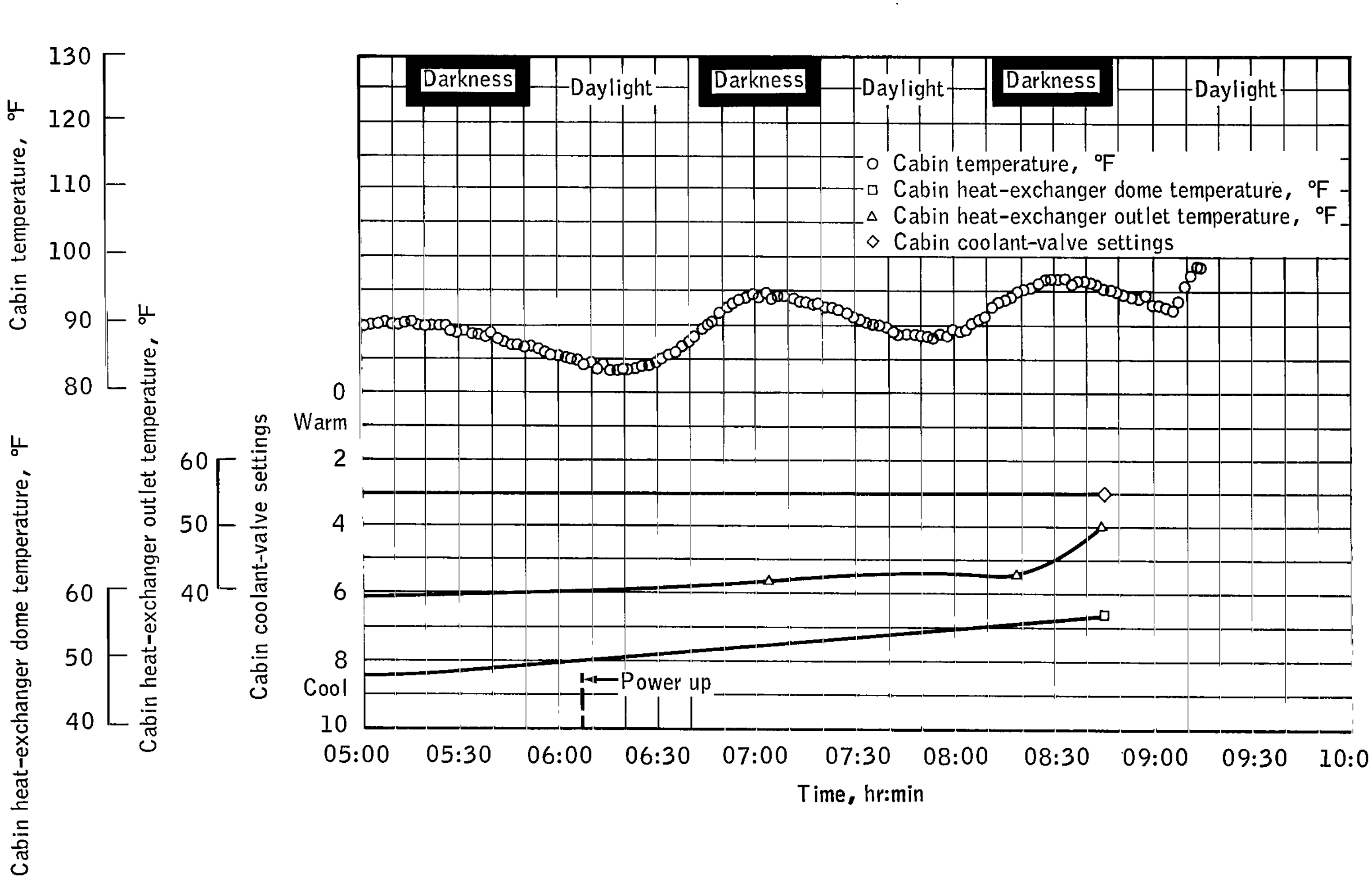


Figure C-7. - Variation of 150- and 250-VA inverter temperatures and associated cooling-control-valve settings with time.



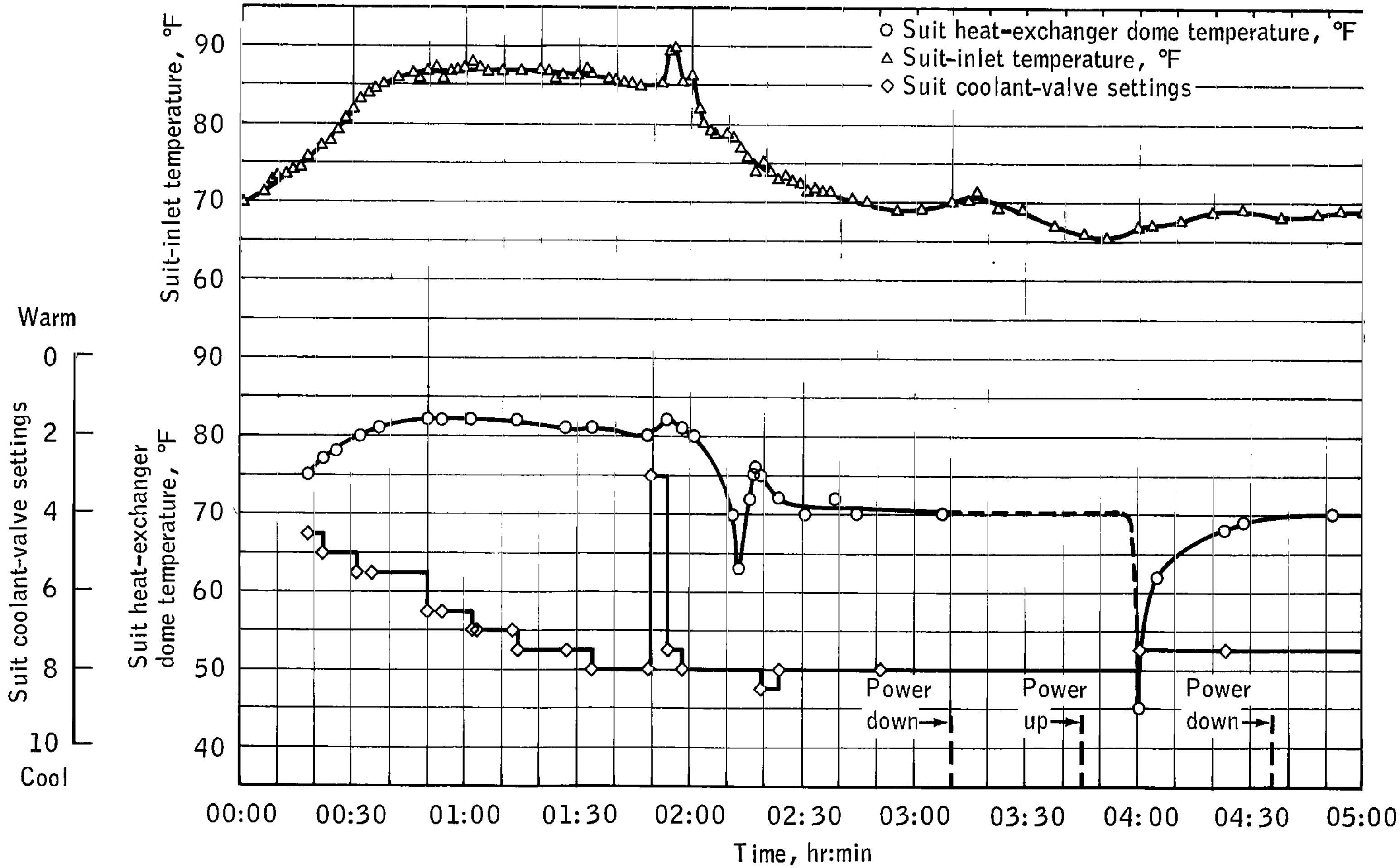
(a) Flight elapsed time, 00:00 to 05:00.

Figure C-8. - Variation of cabin air temperature, cabin heat-exchanger dome temperature, and associated comfort-control-valve settings with time.



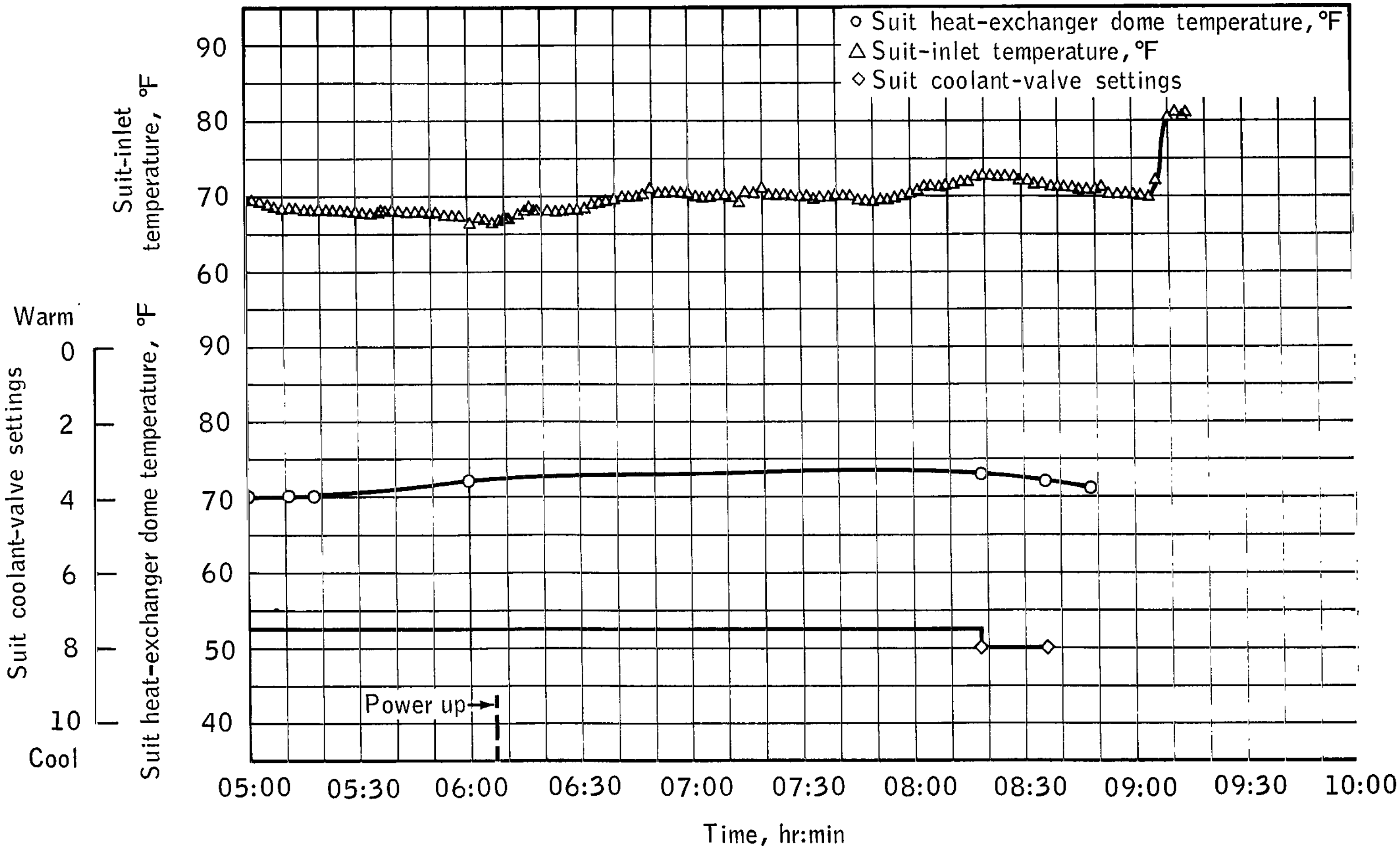
(b) Flight elapsed time, 05:00 to 10:00.

Figure C-8. - Concluded.



(a) Flight elapsed time, 00:00 to 05:00.

Figure C-9. - Variation of suit-inlet temperature, suit heat-exchanger dome temperature, and associated suit coolant-valve settings with time.



(b) Flight elapsed time, 05:00 to 10:00.

Figure C-9.- Concluded.

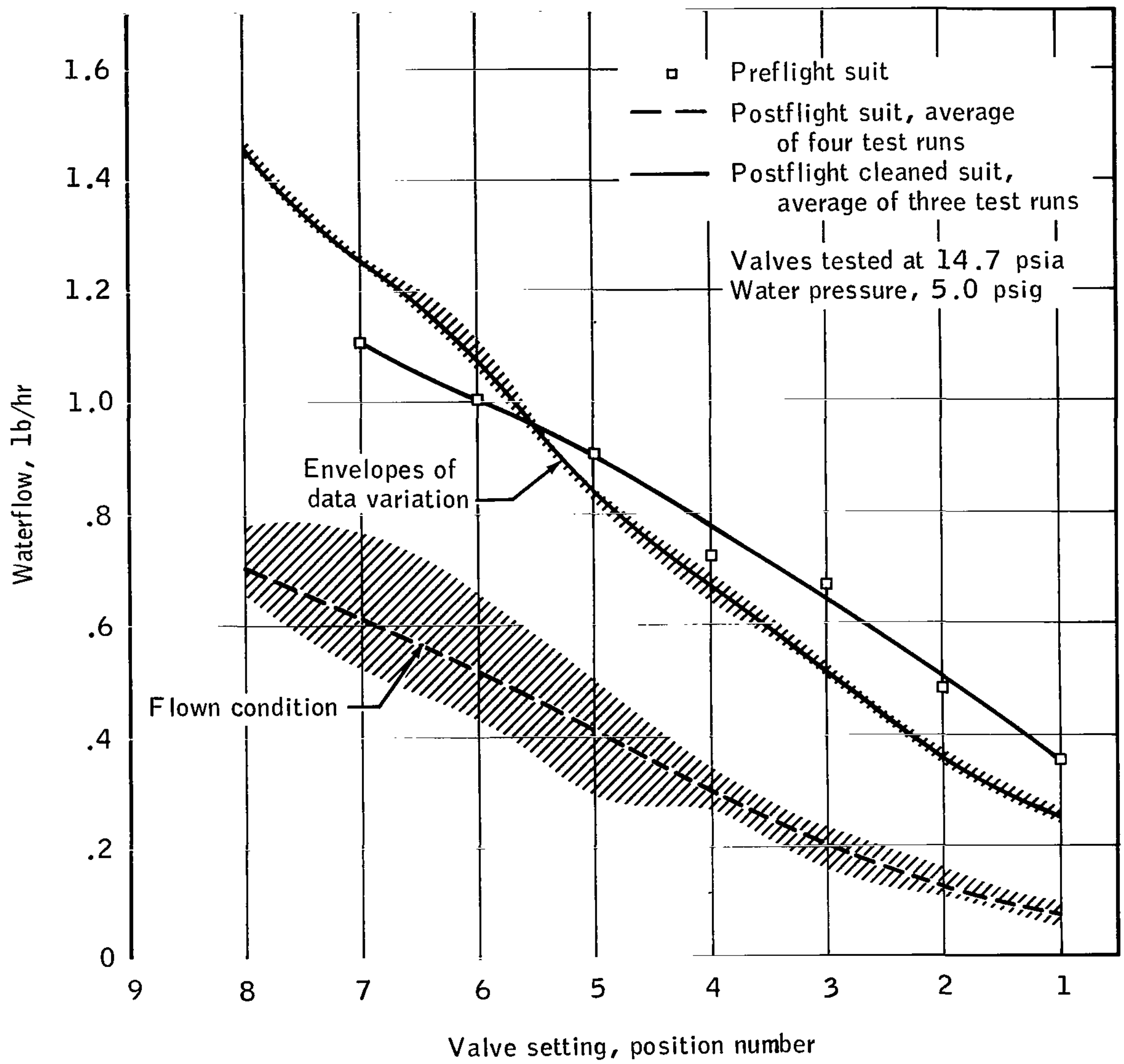
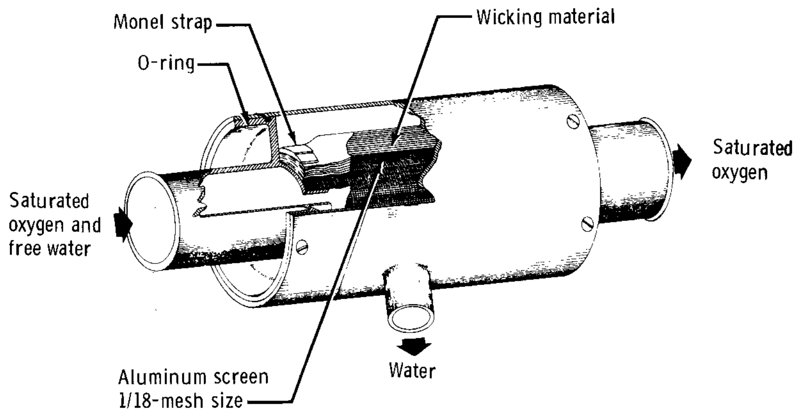
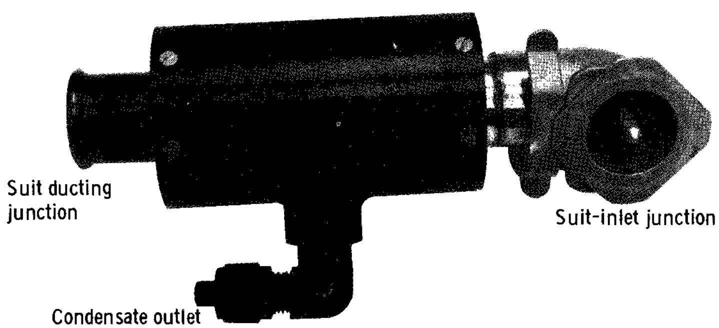


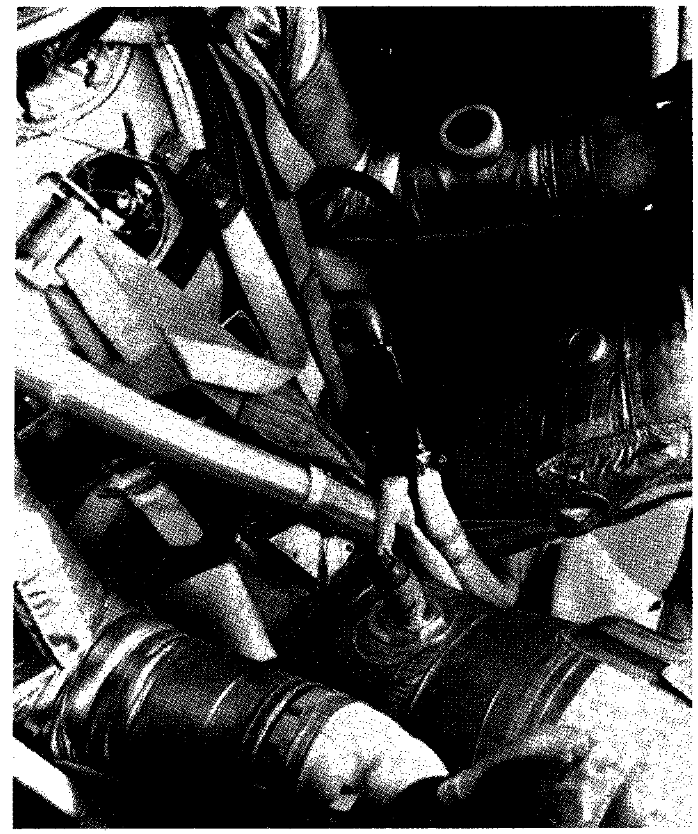
Figure C-10. - Comfort-control-valve calibration curve.



(a) Condensate-trap details.



(b) Condensate trap.



(c) Condensate-trap installation.

Figure C-11. - MA-9 suit-circuit condensate trap.

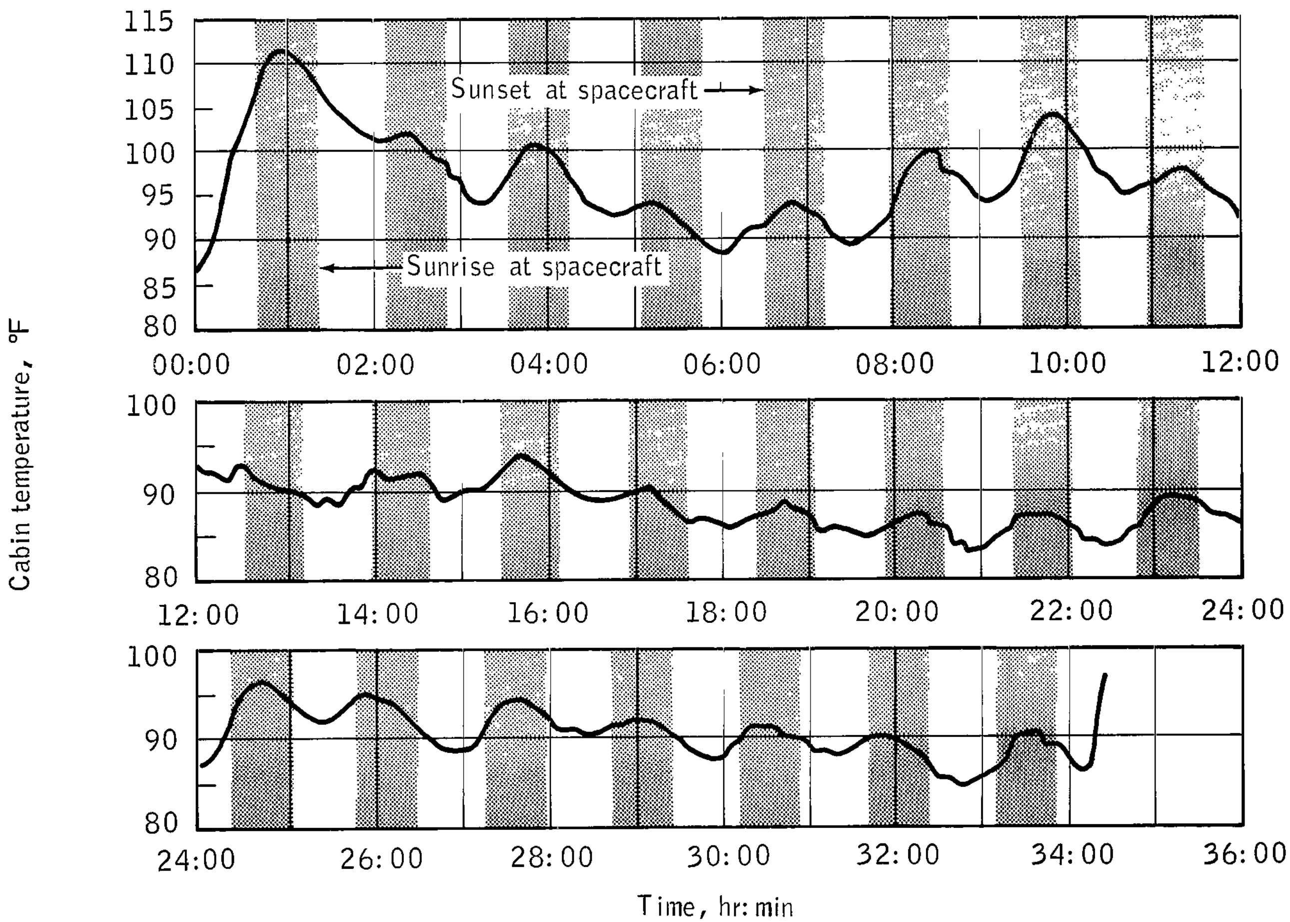


Figure C-12. - Cabin temperature evaluation.

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