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FLIGHT SIMULATION TESTS OF A CENTAUR VEHICLE IN A SPACE CHAMBER

by Ralph F. Schmiedlin, Robert J. Turek, B. J. Dastoli, Glen M. Hotz, and Lawrence J. Ross

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16. Abstract

As part of an extensive ground test program, the Centaur, a hydrogen-oxygen second stage rocket vehicle, was installed in a Lewis Research Center environmental space chamber and operated electrically in a series of simulated flight tests. Twenty-five 1-hour low orbital missions that included 25-minute coast periods were simulated at various flight conditions. Secondary tests were performed to investigate effects on system operation of (1) leakage in sealed subsystem canisters, (2) canister internal heat transfer at simulated zero gravity, and (3) variation of radiant heat input with time. This report describes the test procedures, simulation techniques, and typical results obtained during these flight simulation tests. Included is a unique method of determining test values of radiant thermal input by use of a one-quarter-scale model.

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SUMMARY

A full-scale Centaur vehicle was operated electrically in a series of simulated two-burn flights in a space chamber at the Lewis Research Center. Twenty-five flights, each with a 25-minute coast period, were simulated under various radiant heat inputs.

The test results showed that the design of the Centaur vehicle's electrical system was satisfactory for the thermal and pressure environment of two-burn missions with coast periods up to 25 minutes. Relatively few problems were uncovered during the tests; these were subsequently corrected prior to the seven successful launches of Surveyor spacecraft.

Secondary testing was also performed to investigate effects of leakage of sealed subsystem canisters on subsystem operation, internal heat transfer under simulated zero gravity conditions, and parametric variation of radiant heat input with time.

In general, from the many tests in which radiant heat was applied in a variety of time-varying modes, it was determined that exact simulation of thermal inputs of the flight mission was not important. The use of maximum-average-heat and no-heat tests provided worst-case results. However, this type of constant-rate-input-thermal testing would not be expected to determine if thermal and operational cycling causes premature component failure.

The results of the tests emphasize the importance of designing space vehicle electrical equipment for a minimum power level consistent with satisfactory operation, to avoid overheating. Also, the use of pressure-sealed canisters should be avoided, but if required, should be extensively tested and some provision made to verify the integrity of the seal near the time of flight.

INTRODUCTION

The Centaur was developed as a second stage for the modified Atlas D launch vehicle. The Centaur stage, a high performance stage using liquid hydrogen and liquid oxygen propellants, was first used to boost Surveyor spacecraft for soft landings on the Moon.

The unique features of the Centaur vehicle associated with the use of liquid hydrogen fuel and the mission requirement of orbital coast-phase flight necessitated an extensive development program. In addition to the requirement of propellant management in a zero-gravity environment, the Centaur's electrical/electronic systems were required to perform satisfactorily while exposed to the environment of space for the duration imposed by the assigned missions.

A significant portion of Centaur development was concentrated in an extensive ground test program at the component, subsystem, system, and vehicle levels. Environmental qualification tests of components, subsystems, and systems were performed at conditions exceeding in severity the levels expected during an actual flight. Tests reported herein exposed an entire Centaur vehicle to a simulated space environment. The principle objective of this test program was to verify that all airborne systems would perform properly under the pressure and temperature conditions of prolonged space flight. The tests were conducted in the Space Power Chamber at the NASA-Lewis Research Center. An illustration of the Centaur vehicle installed in the chamber is presented in figure 1.

The installation of a full-scale launch vehicle in a space chamber allows the extensive testing of the total system under a wide variation of test conditions and system configurations. A large quantity of data can be acquired which would be impractical to obtain on a research flight. Any failed system is available, after the test, for analysis. For this series of tests, the various systems on the Centaur were updated to the configuration planned for the fourth flight test of Atlas-Centaur (AC-4). Later in the program, the vehicle was updated to the configuration planned for AC-8 by installation of a new flight control programmer and C-band radar beacon. Both AC-4 and AC-8 were two-burn research and development (R&D) flights with coast-phase experiments. A coast phase is the period between main engine firings.

This report describes the test planning, test procedures, simulation techniques, and some test results of a series of Centaur flight simulation tests completed in 1966. A brief description of these tests is included in reference 1. The analysis of an inverter failure which occurred during these tests is discussed in reference 2.

The work was done in cooperation with personnel from General Dynamics (Astronautics) Convair and Minneapolis-Honeywell, Aeronautical Division, Inertial Guidance Center.

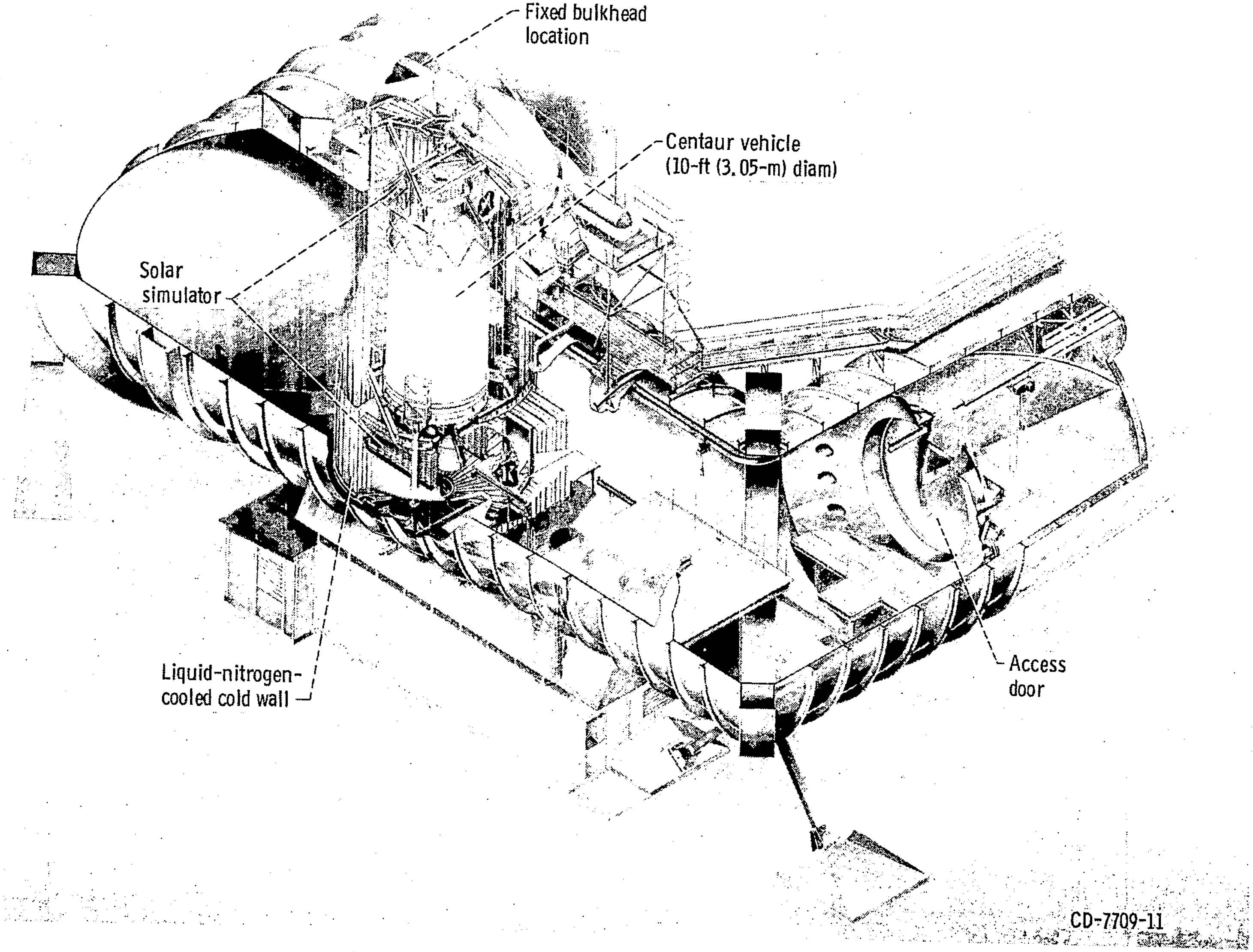


Figure 1. - Centaur vehicle in Space Power Chamber.

SCOPE

A complete flight-type Centaur vehicle was erected vertically in the Space Power Chamber as the test specimen for simulated space environmental testing (fig. 1). The vehicle systems were operated to the fullest extent possible. The pressure and thermal environments of a space flight having a 90-nautical-mile-altitude (167-km) orbital coast phase were simulated as closely as practical. Acoustical and mechanical vibration conditions associated with booster and Centaur engine firings were not simulated. The main engines were not fired, although they were gimbaled. Umbilical and auxiliary cabling was used to connect the test vehicle to monitor and control equipment located in a control room near the space chamber. The telemetry system onboard the test vehicle was connected to an antenna mounted outside the chamber for radiation to a receiving ground station. The effect of propellants in the vehicle was simulated by partially filling the fuel and oxidizer tanks with liquid nitrogen. Periodically, as newer hardware became available, the test vehicle systems were updated to reflect changes in the flight configuration.

TEST OBJECTIVES

The principal objective of the space environmental (flight simulation) test series conducted on the Centaur stage was to verify that all airborne systems would perform properly together under the pressure and temperature conditions of prolonged space flight. Specifically, the tests were planned to support the following objectives:

(1) To determine the effects, if any, of interaction between the various systems and subsystems

(2) To examine the degree and duration of any anomalous system performance due to the following factors: vacuum environment, thermal environment, operating time, out-of-tolerance inputs, out-of-tolerance or defective components, and output overloads and underloads

(3) To determine possible system improvements or simplifications which would result in greater reliability or which would reduce vehicle weight

(4) To obtain operational characteristics of systems such as time constants and drifts from calibration points

(5) To obtain time-temperature data on systems, subsystems, and components under various simulated environments

(6) To obtain sufficient thermal data to predict the system temperature profile accurately for an actual flight situation and to verify the original design assumptions and predictions

(7) To measure the power consumption of systems and subsystems

(8) To obtain satisfactory operating times and failure history data on all systems for reliability studies

(9) To examine the thermal flow characteristics of the Centaur forward equipment area with respect to heat source and sink properties

TEST EQUIPMENT

Centaur Vehicle

General vehicle description. - The Centaur was designed to be used with a modified series D Atlas booster to form the Atlas-Centaur launch vehicle. It is a two-engine upper stage vehicle using liquid-hydrogen and liquid-oxygen propellants. The general arrangement of the Atlas-Centaur launch vehicle with a payload (Surveyor) is shown in figure 2. Structurally, Atlas and Centaur are similar; both stages are of a constant 10-foot (3.048-m) diameter and use propellant tanks constructed of thin wall stainless steel which maintain their shape and load carrying capacity through internal pressurization.

The Centaur stage, which is 28.5 feet (8.69 m) long (from the tip of the fuel tank to the aft portion of the main engines), is powered by two Pratt & Whitney, 15 000-pound (66.7-kN) thrust each, RL-10 engines (main engines). Directional control of Centaur during powered flight is accomplished by gimbaling the main engines. Hydrogen peroxide engines are mounted in the aft end to provide low thrust for propellant settling as well as attitude control during coast periods.

Most of the Centaur electrical and electronic equipment (fig. 3) is mounted on or above the forward bulkhead of the liquid-hydrogen tank (forward equipment compartment, fig. 2). This equipment, as well as the payload, is protected during the ascent portion of flight by a fiberglass nose fairing, which is jettisoned after the vehicle leaves the atmosphere. The Centaur propellant tanks are enveloped by insulation panels to reduce cryogenic boiloff due to heat transfer into the tank during ground operations and ascent through the atmosphere. The insulation panels are jettisoned shortly before the nose fairing. A description of the various systems of the Centaur vehicle is given in reference 3.

The Centaur missions simulated in these tests represent those which inject a Surveyor spacecraft on lunar impact trajectories. The Surveyor was designed to soft land on the Moon and provide data in support of the Apollo program.

Test vehicle description. - The test vehicle used in the simulated space environmental test series was one of the early configuration Centaurs assembled on tank 6A. This vehicle was initially used for engine firing tests conducted at a static firing test site at Sycamore Canyon, California. This same vehicle was also used to verify

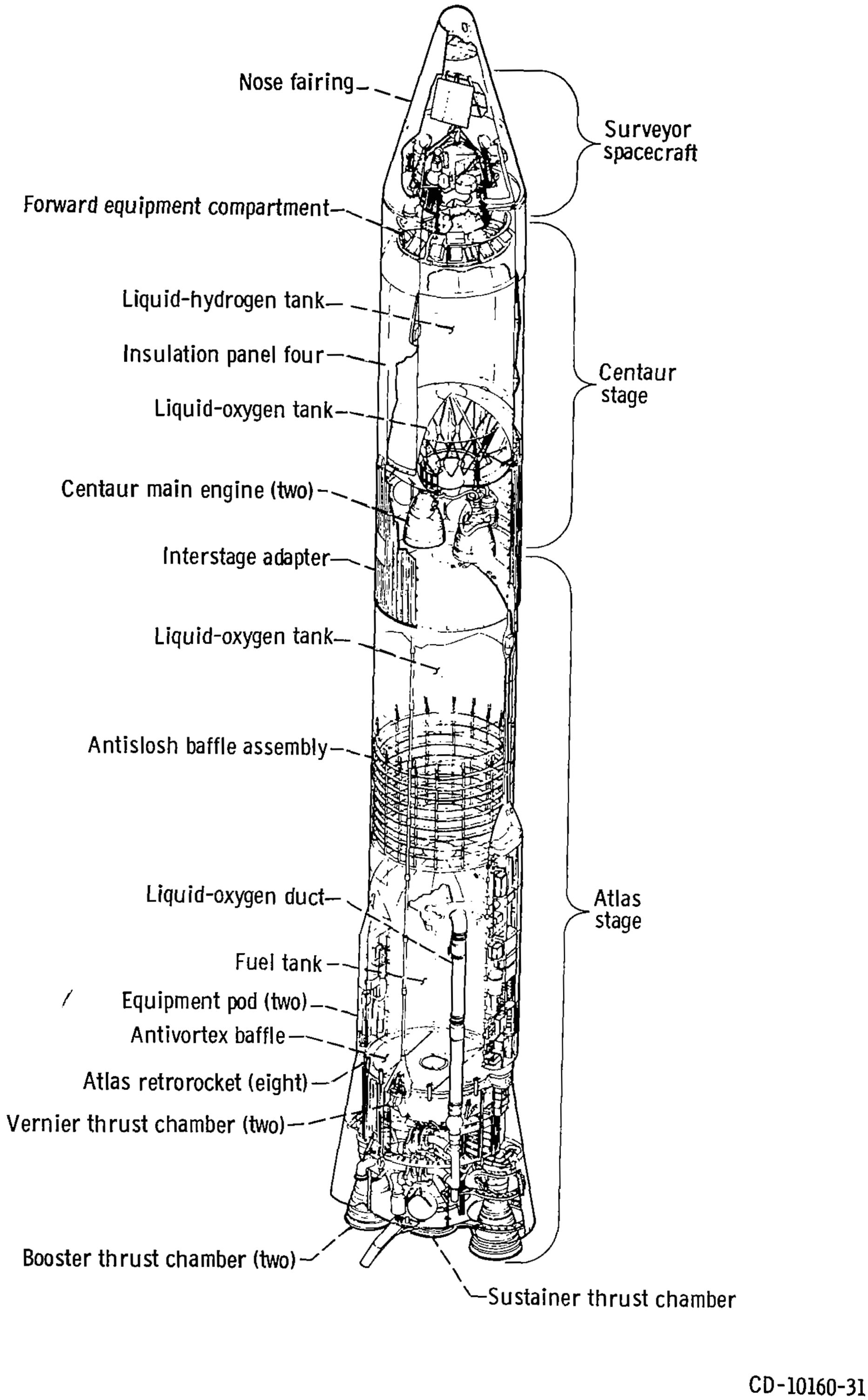
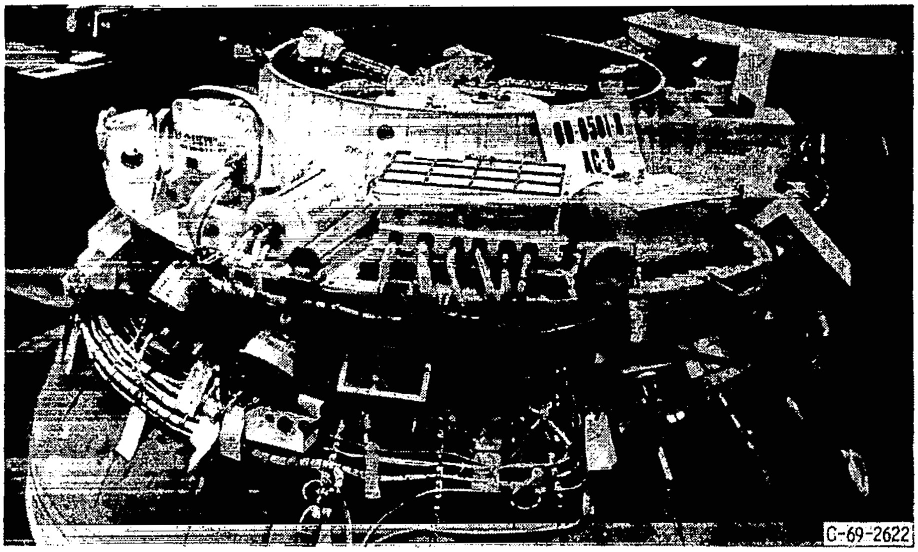
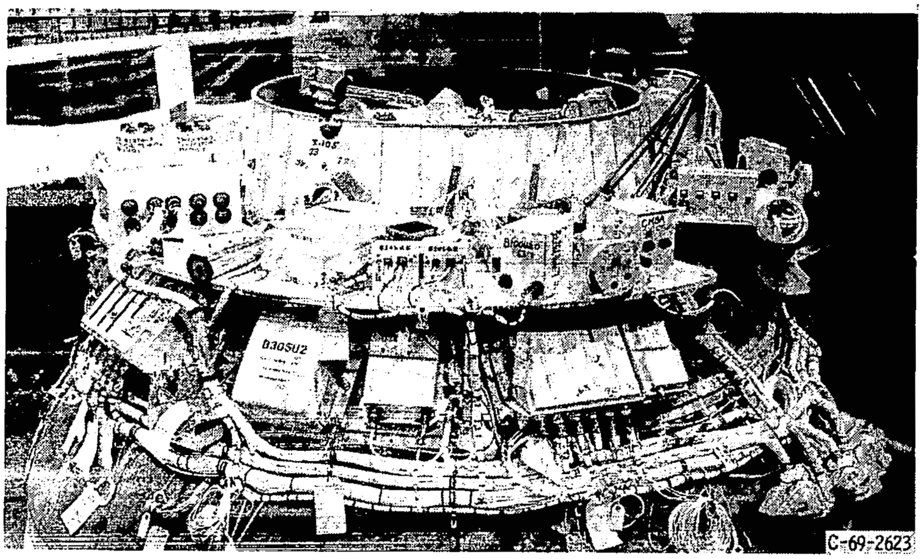


Figure 2. - Atlas-Centaur launch vehicle with Surveyor spacecraft.



(a) Guidance side.



(b) Autopilot and power side.

Figure 3. - Factory mockup of AC-8 configuration of forward area.

vehicle-to-launch-site compatibility at launch complex 36A, Cape Kennedy, Florida. Its fuel tank wall thickness was 0.010 inch (0.254 mm), compared with 0.016 inch (0.406 mm) in later vehicles. In consideration of the thin walls and the previous history of the test tank, tank pressures were maintained at levels which would not cause cracks leading to explosive fragmentation. Maximum tank wall pressure differentials of 5 to 6 psi (3.45 to 4.17 N/cm^2) in the fuel tank (even when containing liquid nitrogen) and 15 psi (10.3 N/cm^2) in the oxygen tank were maintained.

The description of the specific systems of the test vehicle and the deviations to current Centaur vehicles is presented in appendix A.

Facility

The space simulation chamber utilized for the tests is a section of the converted altitude wind tunnel complex at Lewis Research Center, Cleveland, Ohio. Two test chambers were constructed in the existing facility, utilizing the shell as the chamber walls. The equipment incident to its use as a wind tunnel was removed. Three bulkheads were installed to isolate the test chambers from the rest of the facility. The space simulation chamber contains one fixed bulkhead and one having a 15-foot- (4.57-m-) diameter access door. This chamber has an average diameter of 30 feet (9.14 m), is approximately 100 feet (30.48 m) long, and has a volume of 70 000 cubic feet (1982 cu m). In order to accommodate the Centaur vehicle, a $22\frac{1}{2}$-foot- (6.86-m-) diameter vertical cylindrical section and a removable dome were added to increase the maximum chamber height to 45 feet (13.72 m). Figure 1 shows the modified chamber used for space simulation with the Centaur test vehicle installed. Figures 4 and 5 show the Centaur vehicle being installed in the chamber.

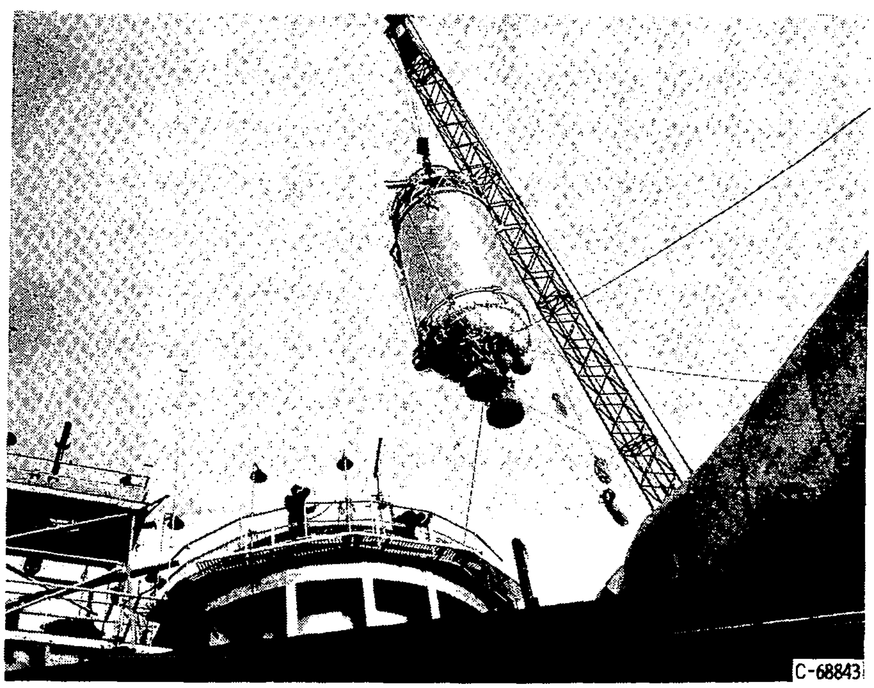


Figure 4. - Centaur vehicle being positioned for installation in test chamber.

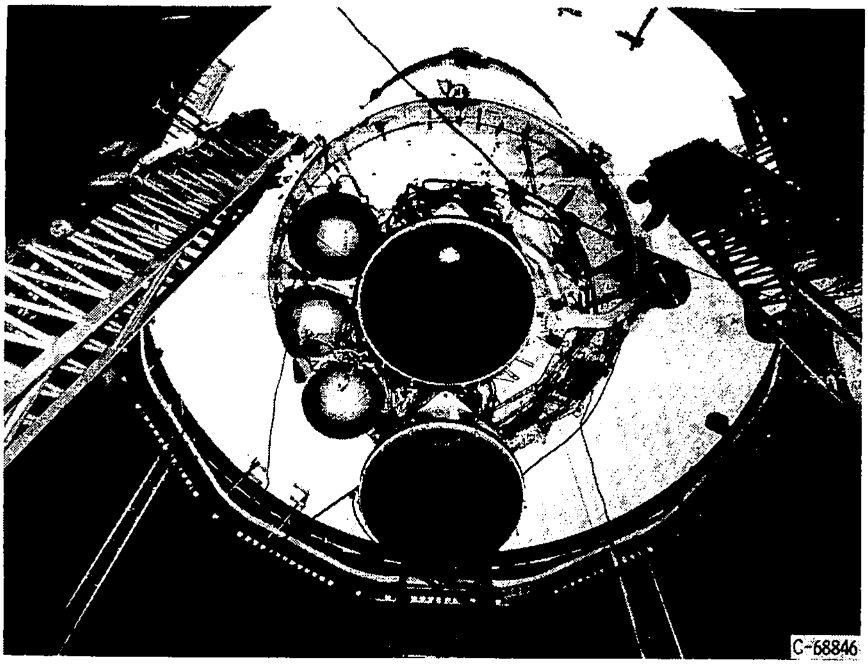


Figure 5. - Centaur vehicle being lowered into test chamber.

The access door integral with one bulkhead is equipped with dual O-ring seals. Instrument cables and other equipment lines pass through a series of ports located around the dome and in the bulkhead on both sides of the access door. The removable dome, used for installation of the vehicle, has two 1/4-inch- (6.35-mm-) thick rings welded together on the outside diameter with the inside diameter of one welded to the dome and the other to the facility. A leak-tight joint was formed but allows ease in cutting when removal of the dome is desired.

All interior surfaces of the chamber were sandblasted to remove scale, rust, and engine exhaust byproducts from previous tests. The chamber was painted with an aluminum paint.

All chamber penetrations and weld joints were leak checked using a helium leak detector. Acceptable joints had less than 1×10^{-8} standard cubic centimeters per second leakage. Individual joints or penetrations were checked by a "bagging" technique. Typically, a polyethylene bag was secured to a small area of the chamber outer wall and helium was injected into the bag. A vacuum was maintained within the chamber. If any leaks were present, the helium would leak into the chamber and the rate of the leak was determined by the leak detector.

A detailed description of the various systems of the test facility and the ground support equipment (GSE) used to control and monitor the test vehicle is given in appendix B.

Two auxiliary test areas were established to provide for inspection, calibration, and failure analysis of the flight components. An electronics and radiofrequency test area was provided near the test chamber. Another area was provided at the 8- by 6-foot supersonic wind tunnel service area for the guidance system and for the inverter. The guidance laboratory is shown in figure 6.

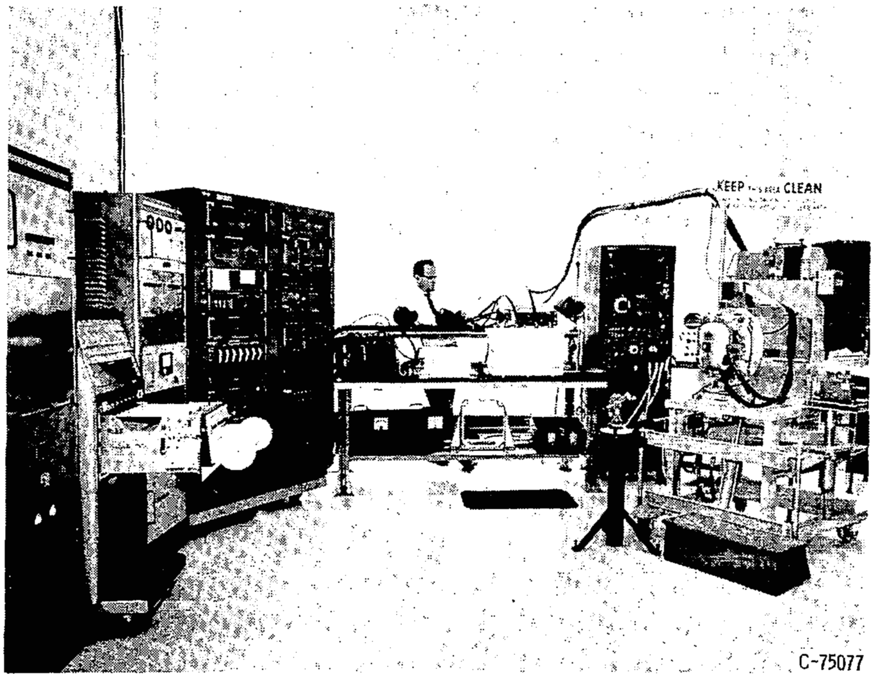


Figure 6. - Guidance laboratory for Space Power Chamber tests.

TEST PROCEDURES

Test Conditions

Altitude (pressure). - Ambient pressure at the "parking" orbit altitude (90 n mi; 167 km) of Centaur coast phase missions is approximately 4.4×10^{-6} torr. Due to vacuum equipment limitations, the lowest pressure obtainable with the test vehicle in the space chamber was 2.4×10^{-5} torr (71 n mi; 131 km).

Radiant heat sources. - The effect of solar and Earth radiant heat on the test vehicle was simulated by the use of tungsten-iodine lamps. The lamps were arranged in six variable intensity zones about the vehicle. Four of the zones (fig. 7) enveloped the forward equipment compartment of the vehicle. Two zones were located at the aft end

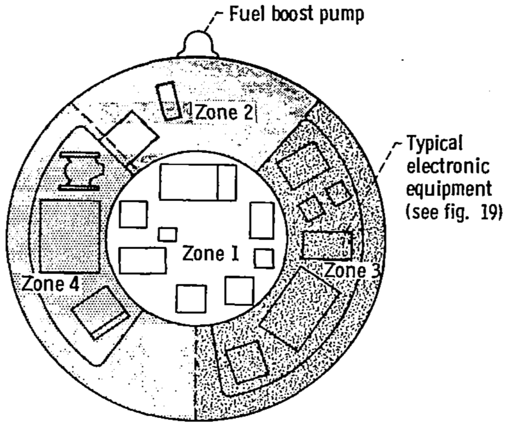


Figure 7. - Radiant heater forward zone areas (view looking aft).

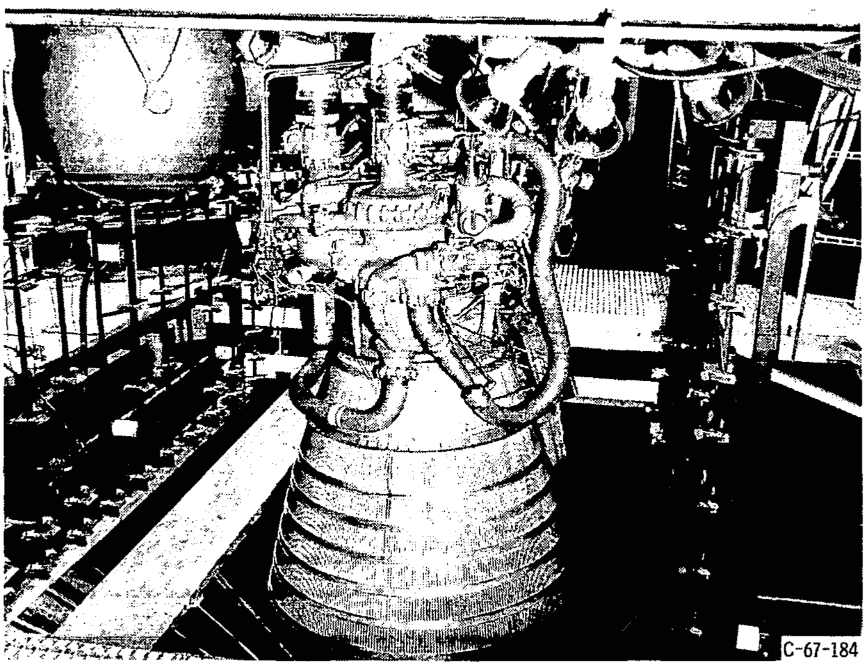


Figure 8. - Radiant heaters for RL-10 engine area (aft end).

of the vehicle (fig. 8) to direct heat at the engines, fuel boost pump, and other equipment, such as the destructor unit (fig. 9).

Special studies on solar and Earth heat simulation, discussed in appendix C were conducted to establish the intensity of the lamps required for proper thermal simulation.

Radiant heat absorption. - The radiant heat sink characteristic of the space environment was simulated by surrounding the entire vehicle with a copper tube-in-strip baffle cooled with liquid nitrogen. The cold wall was coated with a flat black paint on the side

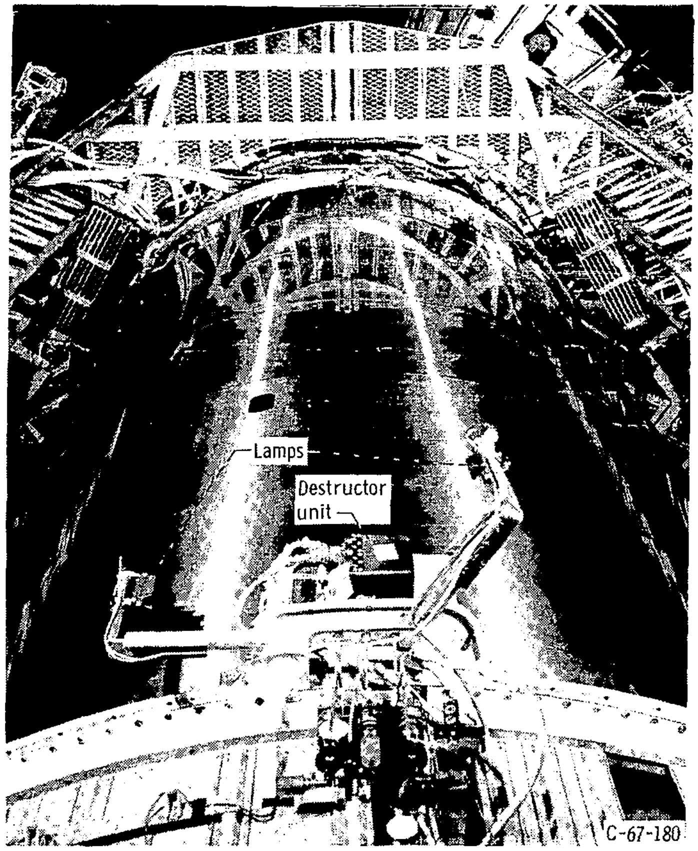


Figure 9. - Two lamps for heating destructor unit.

toward the vehicle and was unpainted on the side toward the chamber. Figure 10 shows the radiant heat absorption system.

Test vehicle. - The liquid-hydrogen tank could be filled with liquid nitrogen only to 10 percent of its volume because of the high density of nitrogen compared with hydrogen and pressure limitations of the tanks. Test runs were made with one, both, or neither propellant tank containing liquid nitrogen in order to evaluate the thermal effect on the equipment.

The vehicle's helium storage sphere was pressurized with helium to between 2000 and 3360 psig (1389 and 2327 N/cm^2) during tests. Vehicle hydraulic power was supplied for all engine gimbaling actuations from a facility source.

Flight-type 28-volt batteries were used on the vehicle for certain test runs; however, the batteries were simulated by facility sources for most tests. During the actual flight simulations, the airborne 115-volt, 400-hertz, three-phase power was normally supplied by the vehicle inverter. Provision was made to supply the vehicle ac loads from a facility inverter if the airborne inverter failed during a test run.

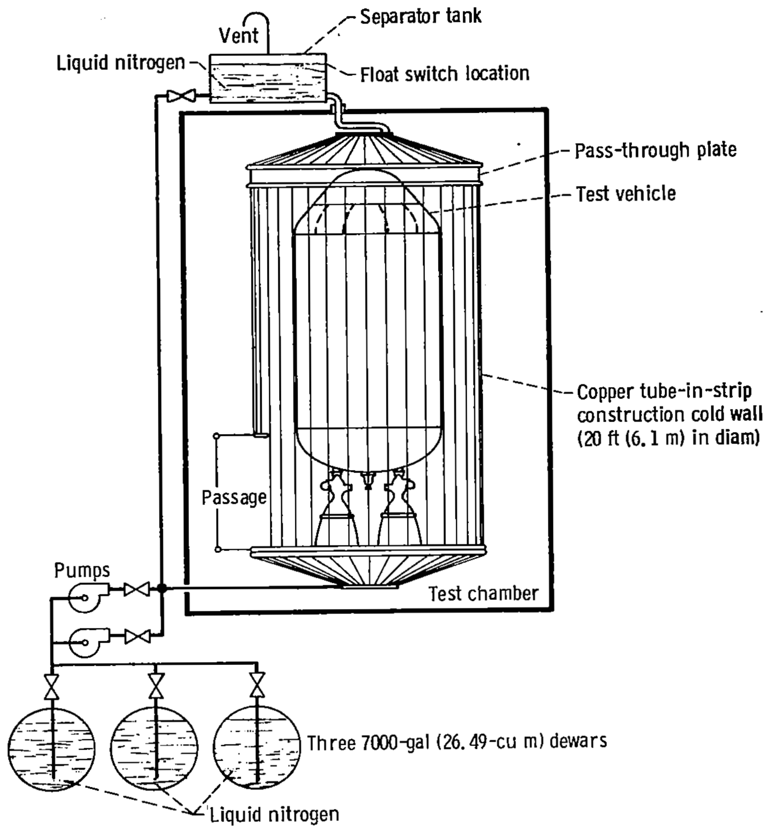


Figure 10. - Radiant heat absorption system (cold wall).

Test Method

General. - The Space Power Chamber environmental test was conducted as a flight simulation; that is, the vehicle systems were exercised, as closely as possible, in the same mode and sequence as the systems of a flight vehicle during an actual mission. Although it was not possible to simulate the ascent environment by rapidly decreasing the test chamber pressure, it was possible to simulate the heat input corresponding to all phases of flight by regulating the intensity of zone controlled radiant heaters.

Prior to the start of a simulated flight in the space chamber, the vehicle system packages were conditioned to the approximate skin temperature they would experience at lift-off of an actual flight. This thermal conditioning was accomplished by varying the intensity of the radiant heaters to offset the heat sink effects of the facility cold wall and vehicle cryogenic temperature.

Thermal programming. - The solar radiant and Earth radiant heat input to the various areas of Centaur during flight is a function of the following variables:

(1) Time of launch from Eastern Test Range

(2) Time in the flight

(3) Duration of coast (mission requirements)

(4) Vehicle roll and pitch profile

(5) Size and shape of payload (shadow considerations)

The desired variation in heat input with time for each package was determined by measuring projected (lighted) areas from photographs of a one-quarter-scale model of the forward equipment compartment which was properly positioned in relation to a light source. The model method is discussed in appendix D. Initially, this information was employed to program the radiant heater zones such that the input to equipment was equivalent both in time and intensity of application, as well as total heat input, to the predicted flight profile. Later tests were run with the total heat input duplicated by heating at a uniform rate.

Curves were prepared for the solar lamp filament voltage setting required (in each zone) to provide thermal input to the packages equal to the computed absorbed solar and/or Earth reflected heat flux values. The individual lamps were arranged over the packages to provide the required thermal heating corresponding to the package color. Before a test was run, a prerun ''sea-level'' survey was conducted using a portable water-cooled black calorimeter, corrected to altitude, to ascertain that the heat flux to each of the packages coincided with that desired. For these prerun checks, a measuring location on each package was used which represented the average flux as determined from an extensive survey that had been made on each package. The calorimeter measured the variation in total radiant flux from the simulator lamps with lamp voltage. The product of this flux and the total lamp absorptance for the color of the package (as determined before) provided the amount of radiant flux being absorbed by the package.

Typical simulated flight. - A simulated flight-test proceeded in the following order:

(1) Simulated flight at sea-level pressure, to verify test vehicle systems proper operation and to obtain baseline data

(2) Sealing and evacuation of test chamber

(3) Filling cold-wall with liquid nitrogen

(4) Tanking test vehicle (when required) with liquid nitrogen

(5) Conditioning vehicle equipment to time-of-launch temperatures

(6) Switching to vehicle (internal) electrical power

(7) Starting test

The test corresponded to a mission of an Atlas-Centaur vehicle. The first 3 minutes, approximately, of the test corresponded to the Atlas powered portion of the flight. The planned radiant heater program was begun at time-of-launch. After this time, the Centaur was ''turned on'' when the autopilot programmer was started by a guidance discrete. From this point, on an actual mission, the sequence of events is as follows:

separate Atlas from Centaur; prestart operations for Centaur's main engines; first firing of main engines; cutoff main engines; coast phase; second firing of main engines; cutoff main engines; Centaur separation from the spacecraft; Centaur turnaround; and retromaneuver (to prevent Surveyor spacecraft star seekers from fixing on the Centaur). The test ends when Centaur shuts itself off (the programmer shuts off electrical power by energizing a power changeover switch to the ground supply position).

In flight, the guidance computer calculates steering vectors and provides appropriate excitation through the platform resolver chain to the autopilot. For these simulation tests, the guidance computer was programmed to provide sawtooth-shaped steering signals. These signals commanded the autopilot to gimbal the engines, alternately between maximum yaw positions and maximum pitch positions. During the coast phase of the simulated flight, the guidance computer again issued sawtooth steering signals to verify that the autopilot would command hydrogen peroxide Centaur vernier engines at the proper time to provide attitude control. The autopilot permitted guidance signals to be accepted, provided these signals were in proper sequence and at the proper time. All the discretes were time-monitored by observing indicator lamps. Exact times were later determined from event recorders.

Special secondary tests. - During several flight simulations, equipment and procedural modifications were made to determine the following effects: (1) simulated leakage of sealed subsystem canisters on operation of the subsystem, (2)selected operational variables on heat transfer, (3) variation in radiant heat input with time on canister temperatures, and (4) simulated zero gravity on internal canister heat transfer.

Canister pressurization leakage: Several Centaur electronic packages (canisters) have built-in vent systems. These packages operate at pressures near the environmental pressures which surround the test vehicle. Other packages, which are sealed to maintain finite pressures, may leak during flight with possible degrading effects in package performance. Therefore, several of the packages were connected to a canister purge system which enabled the test conductor to measure internal pressures, to vary package pressures, and to change gas mixtures at any time during a simulated flight test (see table I).

The effect of leakage on sealed canisters was determined by decreasing the internal pressure at a slow rate so as to pass through the range of critical electrical discharge pressure (Paschen's law). Reference 4 presents a brief discussion of Paschen's law and identifies some new problems with radiofrequency voltage.

Heat transfer variations: A series of tests was conducted to obtain thermodynamic data in support of a parametric heat-transfer study. The first test conducted was a simulated two-burn operation at sea-level conditions without cryogenics and without simulated solar heat. The second series of tests was a duplicate of the first tests with the exception that it was performed at altitude. Comparison of results of the latter test

TABLE I. - ELECTRONIC PACKAGES, INTERNAL PRESSURES, AND ALLOWABLE LEAK RATES

Part number	Part name	Internal gas	Operating pressure		Leak test				
			psi	N/cm^2	Initial pressure		Allowable leak rate		
					psig	N/cm^2	Pressure drop		Time, hr
							psi	N/cm^2	
55-01117-3	Telemetry transmitter SS 1	Nitrogen	[a]29.7	20.5	15	20.5	0.2	0.14	1
55-13503-905	Telemetry transmitter SS 2		[a]29.7	20.5			.08	.006	1
55-01105-3	Radiofrequency amplifier SS 2		[a]29.7	20.5			.08	.006	1
55-13505-831	Telemetry transponder SS 4		[a]24.7	17.0			.5	.345	6
55-13594-801	Radiofrequency amplifier SS 4		[a]17.7	12.2	12	18.4	1.25	.86	
26-12203	Azusa transmitter		[a]30.7	21.1	15	20.5	5.0	3.45	
27-36014	Range safety receivers		[a]17.7	12.2	12	18.4	1.25	.86	
55-06256-3	Multicoupler		[a]17.7	12.2	15	20.5	.5	.345	
55-04028-5	Inertial guidance platform	Helium (92%)-nitrogen (8%)[b]	[a]18.0	12.4	18	22.5	.25	.17	
55-04030-7	Platform electronics	Helium (92%)-nitrogen (8%)	[a]18.0	12.4	18	22.5			
55-04031-5	Signal conditioner		[a]18.0	12.4	18	22.5			
55-04033-5	Pulserebalance (coupler)		[c]2.5	11.8 to 1.7	2.5	11.8			
55-40001-819	Rate-gyro pack	Nitrogen	[c]2.0	11.5 to 1.4	5	13.6			
55-40002-837	Servoamplifier	Nitrogen	[c]2.0	11.5 to 1.4	5	13.6			
55-40003-893	Programmer, electronic	Nitrogen	[c]2.0	11.5 to 1.4	5	13.6			

[a]Absolute.

[b]Now, helium (30%)-nitrogen (70%); guidance computer was sealed with nitrogen; the Agena timers were not sealed; C-band transponder was sealed with nitrogen at 5 psig (13.6 N/cm^2).

[c]Above ambient.

with the previous test provided an indication of the amount of heat loss by convection from the various electronic packages. The third series of tests was conducted at simulated altitude with the facility liquid-nitrogen baffle filled and the vehicle fuel tank first unloaded and then loaded to 10 percent of its capacity with liquid nitrogen. The results from the above series of tests provided a measure of the thermal conduction to the vehicle propellants from the various electronic packages.

Variation in radiant heat input: To determine the degree of sensitivity to radiant heating, a series of tests was run in which the solar simulator was operated at constant input heating rates of 50, 75, 100, 130, 150, and 175 percent of the anticipated average solar heat input. For the higher heating rates the test was ended when an item reached red-line (maximum designed for) temperature.

Another series of tests was conducted in which the fuel tank was 10 percent filled, the liquid nitrogen baffle was filled, and the chamber was at altitude, but the simulated solar heat input was varied during each test in a "stepped" manner. During one series of tests, the heat was applied at a high rate and then stepped to a lower rate as the test continued, while in a second series of tests the heat was applied at a low rate and then increased to a higher rate during the second half of the test.

When a vehicle is in a near-Earth coast-phase orbit, it is subjected to a continually changing Sun and Earth radiation characteristic. Where thermal data are required to determine only maximum temperatures at the end of a flight, it would be desirable to simplify the testing by use of average heat input. Therefore, further tests were run with constant heat equal to the average total for the stepped cases in order to determine the validity of this simplified testing technique.

Simulation of internal heat transfer at zero gravity: One limitation in studying vehicle thermal behavior in a simulated space environment is the absence of an orbital (zero-gravity) effect. The primary heat-transfer mode that is impaired by zero gravity is free convection. As gravity approaches zero, the coefficient of free convective heat transfer approaches zero. Since this coefficient is a function of the convective medium density, the zero-gravity effect can be simulated by reducing this density by pressure reduction. A limiting factor in the reduction of pressure is the effect on thermal conductivity. If a pressure equivalent to approximately 100 000 feet (30.5 km) of altitude (8.45 torr), is established inside a package, the coefficient of free convective heat transfer would decrease from its sea-level value by about 85 percent while the thermal conductivity would decrease by only 8 percent (approximately). Both of the previously mentioned effects are shown in figure 11, obtained from reference 5.

To investigate the zero-gravity effect on the thermal behavior, the autopilot canisters were operated at the normal 2 psia (1.38 N/cm^2) internal pressure (nitrogen) and also at approximately 8 torr internal pressure. Full solar heating was provided for the first 6 minutes followed by 6 minutes without solar heating. The facility cold wall was filled and the vehicle propellant tanks contained liquid nitrogen.

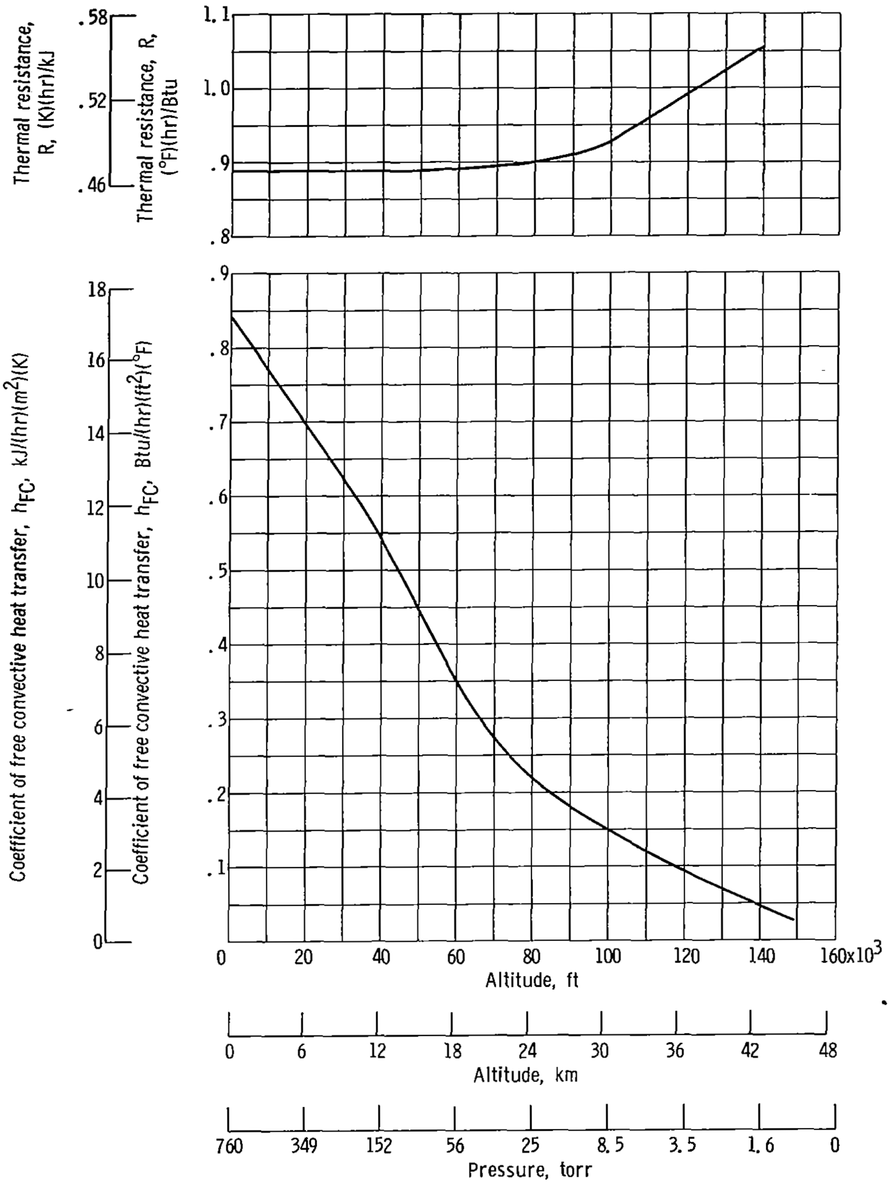


Figure 11. - Free convection heat-transfer coefficient and conductive thermal resistance as function of canister internal pressure expressed as altitude. (Data from ref. 5.)

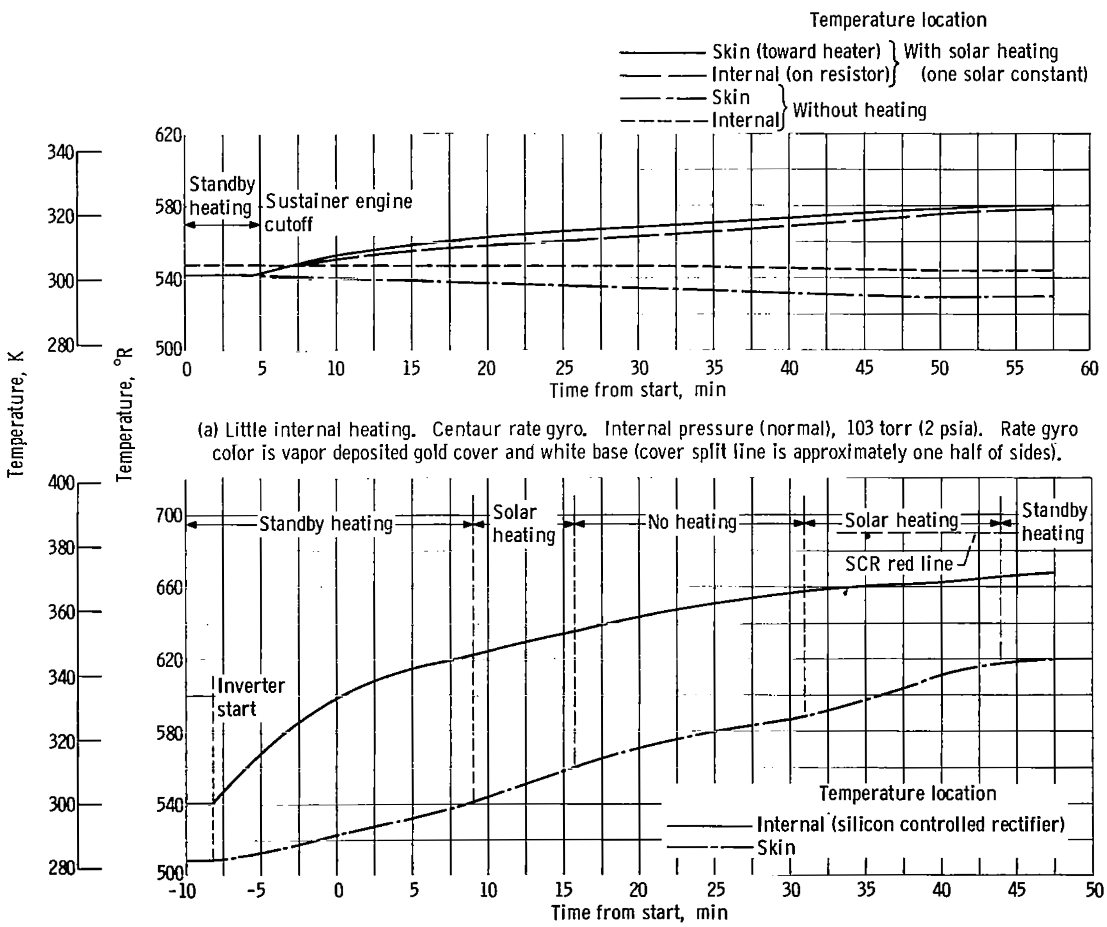
TEST RESULTS

General

The quantitative results and general observations reported herein, for the most part, are related to the length of time the equipment tested was exposed to space environmental conditions. The tests were restricted to a simulated space coast time of 25 minutes (approximately 1 hr total flight time); longer coast periods would require additional testing.

The types of electrical-electronic problems expected to be encountered in a space environment include overheating, overcooling, and arc-over failures due to package leakage. Some of each type were experienced but, in general, only under extreme thermal conditions or simulated leakage.

Typical time related curves of internal component and package skin temperatures are shown in figure 12. The effects of solar heating on the temperature of a typical package having little internal heat generation (Centaur rate gyro) and one having high internal heat generation (inverter) are also shown in figure 12. As would be expected, the skin temperature of a package having high internal heat generation is not a good indicator of internal component temperatures.



(b) Internal heat generated. Centaur inverter. Color is all white.

Figure 12. - Typical heating and no heating as function of time.

In such packages having internal parts that are critically temperature limited, an accurate correlation survey between package monitoring skin temperature measurements and internal temperatures is required.

A comparison of the test data with flight thermal data for the rate gyro skin temperature at various heating conditions is shown in figure 13. Comparable temperature trends are indicated. References 3, 6, and 7 present postflight evaluations of AC-4 and AC-8 flights.

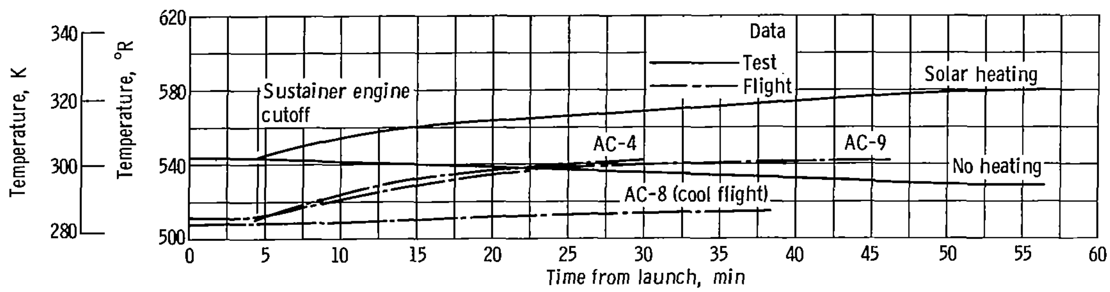


Figure 13. - Comparison of test thermal data with flight data. Centaur rate gyro skin temperature.

Canister Internal Pressure Effects (Leakage)

Telemetry systems. - The transmitters and radiofrequency amplifiers used in the test vehicle telemetry system were designed as pressurized packages. On several occasions, when the package pressure was reduced (simulation of a seal failure) serious degradation of performance or total failure resulted.

C-band beacon tracking system. - The C-band tracking system, used on Centaur flight vehicles to extend the tracking range, was included in the test program. Initial testing on this system began in July 1965, with an airborne system configuration consisting of a AN/DPN-66 (SST-102A) transponder set. Validation of the C-band ground test equipment was accomplished using this transponder for several sea-level pressure tests. These tests also verified that the transponder set performed nominally. On July 27, 1965, the Space Power Chamber was evacuated for a simulated flight test of the AC-8 mission. During the test, the C-band transponder was energized and no beacon response could be detected. It was noted that the input current to the transponder increased during the test and exceeded the normal value of 1.5 amperes. Subsequent sea-level bench tests performed on the transponder verified that the unit failed to reply to interrogations and drew excessive current, the magnitude of which increased with time. Examination of the unit disclosed that the cover seal was faulty. Although the transponder is not pressurized, it must be sealed to maintain internal sea-level pressure for operation at reduced ambient pressure.

After the launch of AC-4 (Dec. 1964), Centaur flight vehicles used a newer, lightweight, C-band transponder SST-131. In September 1965, the new transponder and related hardware were installed on the test vehicle for inclusion in the continuing AC-8 flight profile tests. The transponder was operated during the test program for approximately 20 hours (30 on-off cycles). Including the time and cycles on the unit when received, the total time on the transponder at the conclusion of testing (Dec. 1965) was 44.8 hours (85 on-off cycles). No failures or malfunctions were observed during the tests.

During the flight of AC-6, the Centaur C-band tracking system performance was seriously compromised when the transponder malfunctioned. The mode of this failure was successfully duplicated on the ground by reducing the internal pressure of a similar transponder package thus simulating an ineffective canister pressure seal.

Based on the transponder failure experienced during the Space Power Chamber test program, and on subsequent tests performed by the Centaur prime contractor, it was apparent that to assure successful transponder flight performance, prelaunch preparations must include a positive check of the transponder canister pressure-seal. In January 1966, the Centaur prime contractor was directed to conduct transponder seal tests (independent of those performed by the transponder vendor prior to contractor acceptance) as part of the overall test plan for Centaur vehicle launch readiness checks.

Thermal Effects

General. - Many simulated flights were made with AC-8 flight profiles representing various launch time thermal conditions since launch time had not yet been set. An early daytime launch would result in maximum solar heating whereas an early evening launch would result in a small degree of solar heating (see fig. 14). For all the AC-8 mission simulations (all 25-min coast), all packages tested remained within operational limits.

Heat transfer variables. - When a sea-level test was compared with a similar altitude test, it was determined that the convective cooling at sea level was not appreciable. Forced convection was required to cool the high heat generating subsystems for operation at sea level for over an hour. The conductivity to the vehicle from the electronic packages had little effect on the package temperature. In other words, the insulation between the packages and the vehicle propellant tank was satisfactory. Because only 10 percent of the hydrogen tank was filled, and because liquid nitrogen was used in place of liquid hydrogen, the thermal conduction observed in these tests was below those to be expected in the flight vehicle.

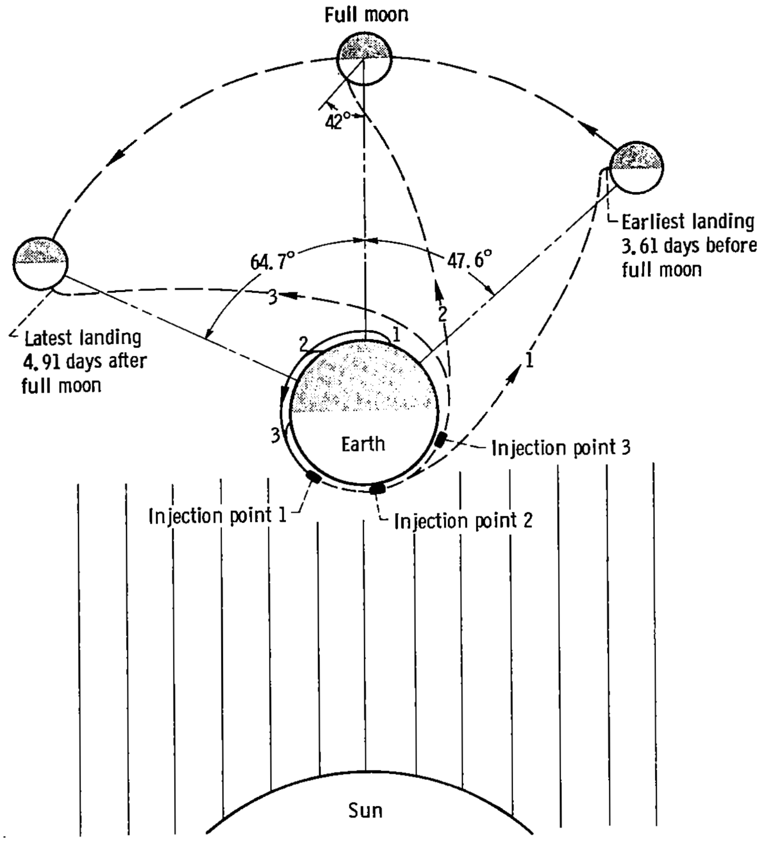


Figure 14. - Extremes of Centaur-Surveyor lunar trajectories with constraint of 150 hours lighting after landing. (30-min coast period 90 n mi (166.7-km) orbit.)

Variation in radiant heat input. - The effect of various radiant heating rates on the skin temperature of the Centaur rate gyro package under simulated space conditions is shown in figure 15. Since this package had little internal heating after the gyros were up to operating temperature, the skin temperature, in the absence of radiant heating, actually cooled with time. The temperature rise with one solar constant and with 130 percent of a solar constant are also shown. In all cases, after the 1-hour run, the package had nearly reached a new thermal equilibrium.

The effect, on skin temperature, of several methods of stepping the radiant heating rates to produce nearly equal total heat is shown in figure 16. The final temperature in each case was approximately the same and therefore correlated with the total heat input. The data at 27 minutes indicates that for quite different total heat inputs, the higher input has the lower indicated skin temperature. This can be explained if a careful analysis is made of all sources of heat input (solar radiation and internal) as well as the heat sinks (cold wall, vehicle, package intervals) available to this skin temperature location. Critical internal temperatures would vary much less. Therefore, unless an

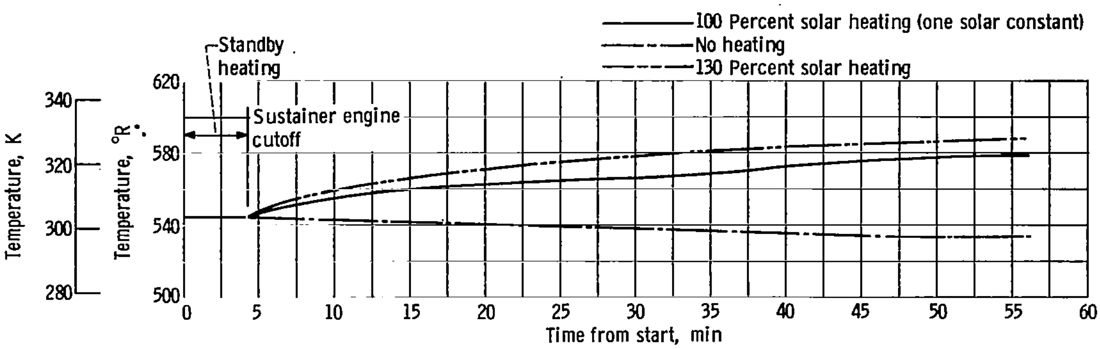


Figure 15. - Effect of various radiant heating rates on skin temperature of low interval heating package, Centaur rate gyro.

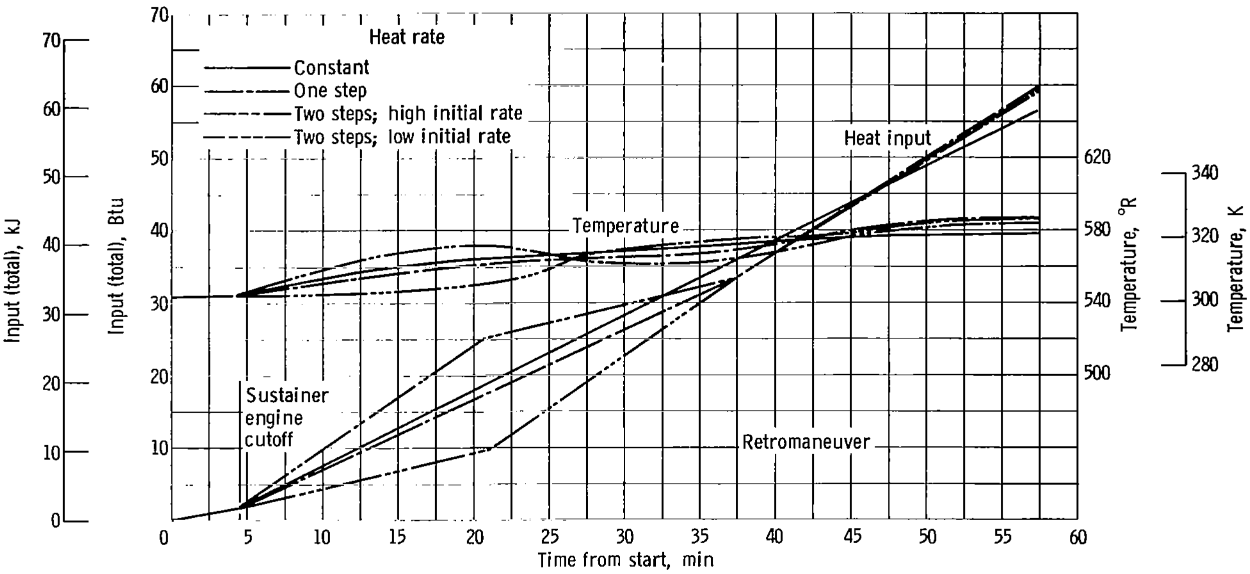


Figure 16. - Effect of several stepped radiant heating rates against average heating rate with nearly equal total heat. Centaur rate gyro skin temperature.

actual temperature against time must be known, the observed anomaly should not preclude the use of average heat input to determine typical operating temperature data.

Telemetry. - Two of the three telemetry subsystems tested incorporated radiofrequency amplifiers in the output of the tube-type transmitters to boost output powers from 5 to 50 and 65 watts. It was found that when these subsystems were operated during a simulated flight, the radiofrequency amplifiers and, to a lesser degree, the transmitter package (which generated the 115-V, 400-Hz power for the amplifier) showed a tendency to rapidly exceed their allowable upper limit of skin temperature.

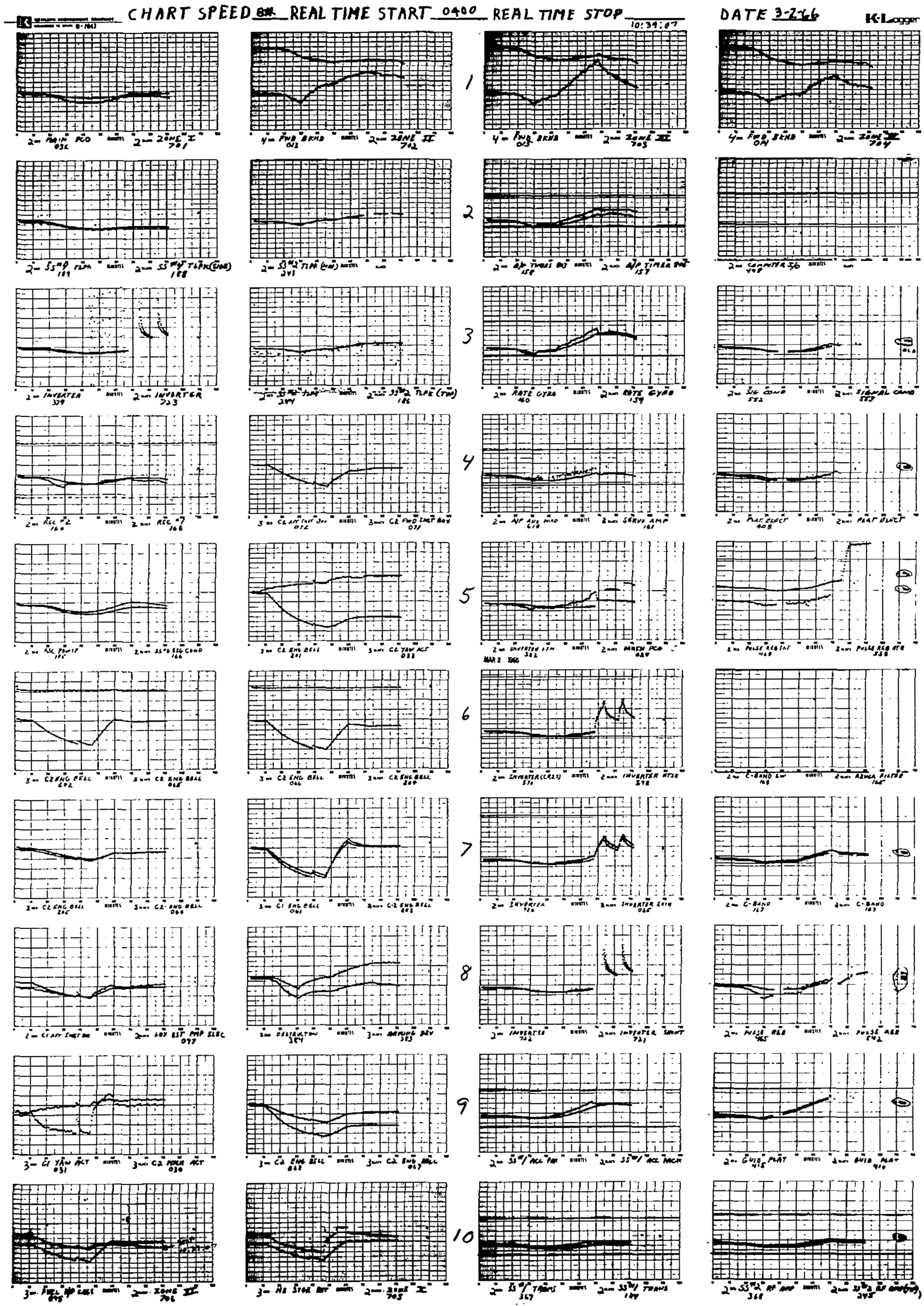


Figure 17. - Typical K-logger record (greatly reduced, see fig. 27).

These radiofrequency amplifiers were not used in actual flights as the 5-watt transmitters were concluded to be adequate. From AC-9 and on the transmitters on the flight vehicles were replaced by 4-watt (minimum) solid-state units.

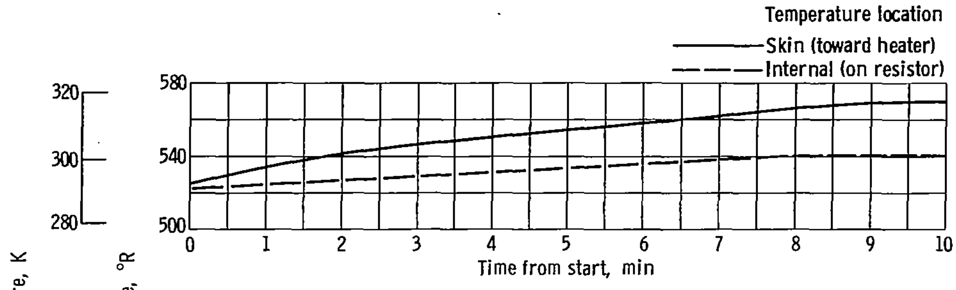
Inverter. - The inverter was a 650-volt-ampere unit with an efficiency of approximately 64 percent. Without solar heating, the operating temperature equalized below the upper operating temperature limit. With solar heating, temperatures nearly reached the upper limit set by internal solid-state rectifiers by the end of a simulated flight.

Recognizing the invertor's sensitivity to high temperatures, the unit as installed on the Centaur vehicles at the launch site is air-conditioned to maintain a maximum prelaunch temperature of 105° F (313.7 K). In the Space Power Chamber tests, even with a few auxiliary cooling coils, it was necessary to avoid using the flight inverter until near time of simulated flight test. The unit also had to be cycled between tests to keep it above 30° F (272 K) or it would not "start." Included in figure 17 are several curves of inverter temperatures.

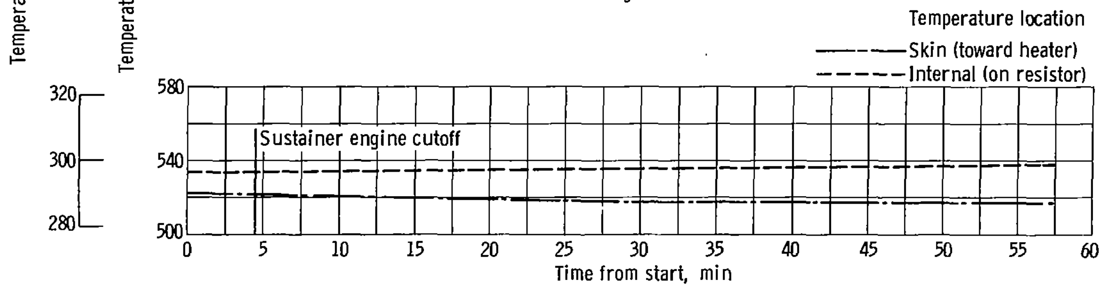
Because of the possible overtemperature problems, and because of design deficiencies (refs. 2 and 8), the Centaur inverter was redesigned. Flights AC-9 and on used the improved inverter. While only 3 percent higher in efficiency (67 percent), the new inverter cooled during prelaunch operations to 90° F (305.4 K) (without increased air-conditioning) compared with 105° F (313.7 K) with the older inverter. With a lower heating rate, this 15° F (8.3° C) difference extends considerably the operating time to reach the upper temperature limit.

Internal Heat Transfer Under Simulated Zero-Gravity Conditions

A small amount of the data for the zero-gravity heat-transfer simulation is presented in figure 18. The data using the normal internal pressure is given in figure 12(a). At zero gravity, less heating or cooling should result between the skin (container) and nonconductively mounted internal equipment because of lack of convective heat transfer. This is shown to be so by comparing figures 12(a) and 18. For the simulated zero-gravity case, the skin temperature increased 45° F (25° C) in 10 minutes when one solar constant of radiant heat was applied: whereas, for the normal case, an increase of only 18° F (10° C) resulted in the same period of time. Also, for the normal case, the internal temperature followed the skin temperature more closely both in heating and cooling (fig. 12(a)). In fact, for the simulated zero-gravity case with no solar heating, the internal temperature actually increased as the skin temperature decreased (fig. 18(b)).



(a) Solar heating.



(b) No heating.

Figure 18. - Simulation of zero-gravity heat transfer by reduction of internal pressure in package. Centaur rate gyro. Internal pressure, 8 torr.

CONCLUSIONS

The following conclusions were made based on the results of the flight simulation tests on the Centaur vehicle:

1. The small number of problems uncovered in a number of flight simulations of the Centaur electrical-electronic system indicates that the systems were adequately designed for their intended coast-phase environment.

2. Results observed with the high power subsystems emphasize the desirability of design for the lowest possible power consistent with satisfactory requirements. Solid-state devices in general would be expected to develop the minimum heat.

3. The serious effects that resulted from loss of pressure in the sealed telemetry transmitters and C-band transponder suggest that the design of these components should be such that their operation would be independent of internal package pressure. Since these components normally contain both high voltage, low and very high frequency circuits, a design that proves impervious to low pressure internal to the package would be difficult, if not impossible, to achieve successfully. Therefore, steps must be taken to assure that pressurized packages are properly sealed prior to a launch.

4. The results of the series of stepped radiant heat input tests and constant radiant heat input tests indicated no appreciable difference in final temperature. The net tem-

perature rise in any individual electronic package was approximately the same. Therefore, where final temperature is the limiting factor, it is not necessary to simulate actual heat input rates for all the possible flight missions but only the average heating. A range of steady radiation heat input tests could map the allowable operating time for various missions.

5. Great care and extensive thermal surveys are required to locate single flight-temperature sensors on packages with internal heat generation and temperature critical parts. This is especially true if a package skin temperature measurement is used for monitoring.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, June 24, 1969,
491-05.

APPENDIX A

TEST VEHICLE DESCRIPTION

Electrical Power Systems

The Centaur electrical system provides airborne 28-volt dc power from a battery composed of silver-oxide - zinc cells. Alternating-current power is supplied by a 115-volt ac, three-phase, 400-hertz solid-state static inverter powered from the battery. In addition to the main missile battery, for this vehicle, telemetry and range safety batteries are used. A 28-volt rectifier power supply, external to the test chamber, was used in place of the flight batteries for most of the tests.

The forward equipment compartment arrangement of the electrical systems for the second updating of the test vehicle (AC-8 configuration) is shown in figure 19.

Autopilot System

The autopilot system consists of a programmer, rate-gyro package, servoamplifier, auxiliary electronics unit, and steering feedback transducers. This system during an actual flight

(1) Stabilizes the vehicle during powered flight

(2) Accepts commands from the inertial guidance system to correct for deviations of the vehicle attitude during powered flight

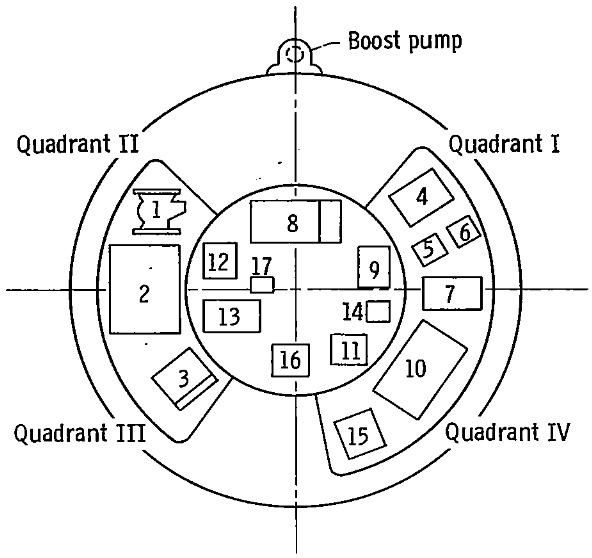
(3) Controls the vehicle in pitch, yaw, and roll-rate during coast periods

(4) Provides logic circuits for control of the ullage and attitude-control hydrogen peroxide engines

(5) Maintains the vehicle parallel to the surface of the Earth during coast periods

(6) Provides numerous switching functions (i.e., start main engines, cutoff main engines, control vent valves, etc.)

The AC-4 configuration used an all electronic programmer. This unit was subject to possible timing upsets caused from stray voltage spikes. Starting with AC-8 configurations, mechanical timers were used; two were required for missions involving two main engine firings and one for missions involving but one main engine firing.



(a) Shelf and inside payload adapter.

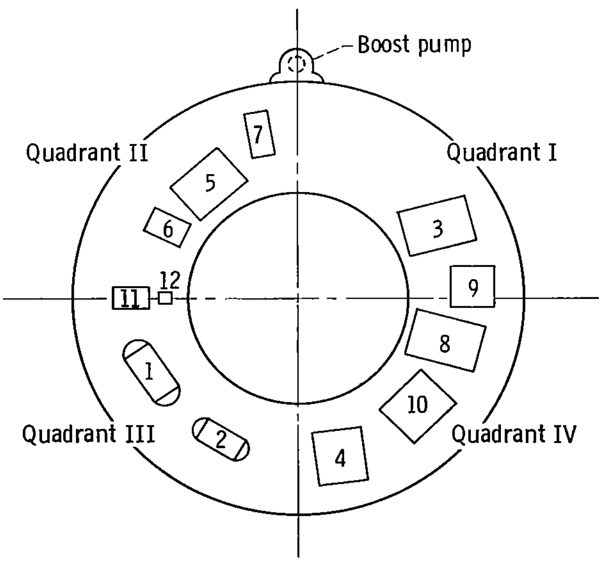
Guidance system:
- 1, Inertial platform, Dgg 8024 A2
- 2, Pulse rebalance, DDG 8014 A1
- 3, Computer, DBG 8016 Al

Autopilot system:
- 4, Auxiliary module, 55-44000-829
- 5, Timer A, 55-04376-803
- 6, Timer B, 55-04376-805
- 7, Rate gyro, 55-40001-819

Telemetry system:
- 8, Telemetry package SS 4, 55-13505-831
- 9, Radiofrequency amplifier SS 4, 55-13594-801
- 10, Accessory pack SS1, 55-13500-843

Range safety system:
- 11, Range safety command receiver 1, AD319600MK1
- 12, Range safety command receiver 2, AD319600MK1
- 13, Power control unit, 55-36060-3
- 14, Ring coupler, 7-36044-801

Electrical system:
- 15, Main battery
- 16, Telemetry battery
- 17, Telemetry power changeover switch, 55-61369-1



(b) Bulkhead mountings.

Guidance system:
- 1, Platform electronics, DEG 8039 A2
- 2, Signal conditioner, DDG 8015 Al

Autopilot system:
- 3, Servoamplifier, 55-40002-837

Telemetry system:
- 4, Transmitter SS 1, 55-01117-3
- 5, Telemetry package, 55-13503-905
- 6, Radiofrequency amplifier, 55-13504-3
- 7, Multicoupler

Electrical system:
- 8, Inverter, 55-06267-SM
- 9, Main power changeover switch, 55-06265-1
- 10, Instrumentation junction box 55-11329-01-04

C-band system:
- 11, C-band transponder, 55-01040-1
- 12, C-band duplexer, 82-74551-001

Figure 19. - Forward end electrical systems arrangement for AC-8 simulation.

Guidance System

The vehicleborne inertial guidance system consists of five components: inertial platform, platform electronics, pulse rebalance with power supply, navigational computer, and signal conditioner. The inertial platform equipment provides the base guidance information. This platform is stabilized by the platform electronic package. The pulse rebalance electronic and power supply package serves two functions; it processes signals going to and from the inertial platform accelerometers, and provides regulated power for guidance equipment. The navigational computer accepts the output of the pulse rebalance package, solves the necessary equations, and issues flight commands. The signal conditioner unit normalizes selected guidance signals for delivery to the telemetry system.

A special program was added to the computer for this test series to provide, at selected times, a saw-tooth steering signal to cause the main engines to gimbal or the attitude engines to receive firing commands. The computer was also commanded to issue the following discretes; booster engine cutoff (BECO), sustainer engine cutoff (SECO), vernier engine cutoff (VECO), and start the timers.

Main Propulsion System

The main engine system consists of two Pratt & Whitney RL10A-3 liquid propellant rocket engines. The engines are fixed thrust, gimbal mounted, and capable of multiple starts in space. The propellants, liquid hydrogen and liquid oxygen, are each fed by hydrogen peroxide turbine driven boost pumps through insulated ducts to pumps on each main engine. On each engine, both propellant pumps and a hydraulic pump are driven by one hydrogen gas-driven turbine. The drive gas is heated by regeneratively cooling the engine nozzle.

In these tests, the boost pump rotors were locked and the propellant lines blocked. The main engines were gimbaled from an auxiliary hydraulic system during the test.

Hydraulic System

The hydraulic system nulls (alines) the engines prior to main engine start and gimbals them during powered flight. Each engine is gimbaled by its own separate turbine-driven hydraulic system consisting primarily of a constant displacement vane-type pump, two double-acting actuators, an accumulator, and associated controls and interconnecting lines. In addition, an electric motor-driven pump is activated either by thermostats to prevent thermal stagnation of the fluid, or by the autopilot programmer to null the engines prior to main engine start.

Only the electric motor-driven pumps were operated during these tests. A ground hydraulic supply was used to drive the gimbal actuators during periods that simulated main engine firing.

Hydrogen Peroxide Supply System

The hydrogen peroxide (H_2O_2) system on the Centaur test vehicle was updated to the AC-4 configuration. The system consisted of a positive-expulsion propellant tank (pressurized from the airborne helium supply), valves, and tubing to supply the attitude control system and both propellant boost-pump turbodrives. Hydrogen peroxide was not tanked during these tests, although the facility provided this capability.

Boost-Pump System

The boost-pump turbodrives are gear-coupled in housings integral with the pumps. Hydrogen peroxide is supplied to the turbines throughout main engine firings in response to commands from the autopilot programmer. In the tests, the boost-pump control valves were operated electrically.

Attitude Control System

The attitude control system provides thrust for yaw, pitch, and roll-rate control during coast periods and reorientation maneuvers. It consists of six individually solenoid-controlled hydrogen peroxide engines in two clusters. These engines were not fired but were actuated electrically during the tests.

Telemetry System

The Centaur telemetry system for the test vehicle originally consisted of six separate subsystems. Subsystem 1 has high data-handling capacity programmed to transmit at high power levels during powered flight. Subsystem 2 has lower capacity and power, and transmitted continuously until after the last attitude-control engine cutoff. Subsystem 3 was a tracking beacon and low-data capacity transmitter. Subsystems 4 and 5 were primarily for the purpose of studying propellant behavior under zero-gravity conditions. Subsystem 4 transmitted data and subsystem 5 was for the signal from a slow-scan television camera looking into the hydrogen tank. Subsystem 6 was for extra instrumentation for structural evaluations. Subsystems 3, 5, and 6 were removed from the test vehicle because they were not used on the flights being simulated. The telemetry subsystem outputs were radiated to a telemetry receiving station.

Tracking Systems

Originally, the Centaur vehicle carried equipment which facilitated the use of two ground tracking systems. The equipment consisted of Azusa and AN/FPN-16 (C-band) radar transponder subsystems. Both subsystems require vehicleborne transponders. The Azusa type C transponder received a continuous signal from the Azusa Mark II ground station and retransmitted a phase-locked signal, offset in frequency from the received signal, to the ground station. The ground station, using receiving antennas on two intersecting baselines, determined the direction and range of the transponder. From this information, the position coordinates of the vehicle can be obtained along with vehicle velocity.

The C-band transponder system, while less accurate than the Azusa system, has the advantage of being able to utilize several ground stations for long range tracking possibilities.

The test vehicle was equipped with both systems, which were connected through hard lines to ground test equipment. After the Azusa transponder failure during the AC-3 flight, the Centaurs were no longer provided with the Azusa system. The C-band transponder was retained.

Range Safety System

The range safety command (RSC) system provides a method for destructively terminating the flight of the vehicle from coded ground command if its continued flight would endanger life or property. The RSC system consists of two command receivers and antennas, a power control unit, an electrical arming and disarming device (since modified), a destructor unit, and a ring coupler which passively routes signals from both antennas to both receivers.

Redundant circuitry is incorporated in the power control unit and electrical arming device. The two command receivers provide separate and independent receiving and decoding of the command signals for ordering destruction of the vehicle.

For these tests, the power-control unit was slightly modified to provide a measurement of the voltage to the relay that actuates the destructor unit. The magnitude of this voltage determined the degree toward firing at any time. A passive destructor unit was used.

Pneumatic System

The helium pneumatic system controls propellant tank pressures and furnishes continuous regulated pressure for engine controls and the hydrogen peroxide bottle. The two helium storage spheres on the vehicle were pressurized during these tests.

APPENDIX B

TEST FACILITY AND GROUND SUPPORT EQUIPMENT DESCRIPTION

Vacuum System

The primary pump system of the test chamber vacuum train consists of 10 oil diffusion pumps, each of which has a pumping speed of 50 000 liters per second (50.0 cu m/sec). These pumps are equipped with water-cooled chevron baffles and oil level gages. The manifolded discharge flows into a roughing vacuum train consisting of one lobe-type blower feeding two parallel eccentric-piston pumps. The blower has a capacity of 30 000 cubic feet per minute (849.5 cu m/min) and the eccentric-piston pumps each have a capacity of 750 cubic feet per minute (21.34 cu m/min). The vacuum system is shown in figure 20.

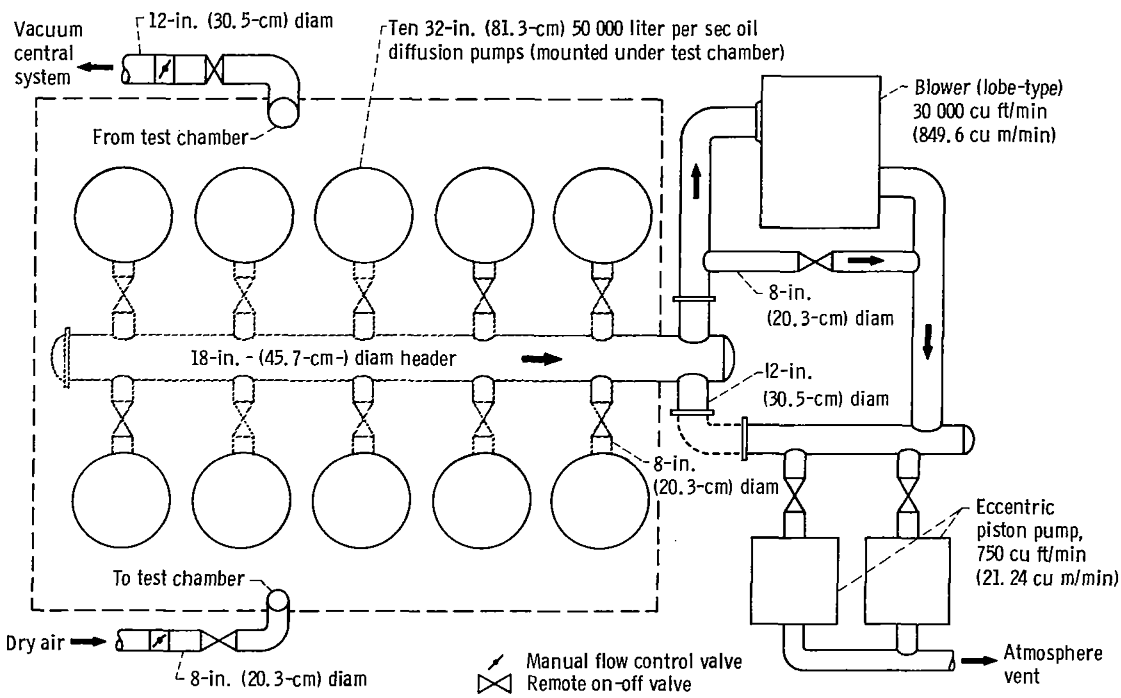


Figure 20. - Test chamber vacuum system and dry air bleed system.

The test chamber was connected to the laboratory's central exhauster system which enabled the chamber to be evacuated to 100 000-foot- (30.48-km-) altitude-equivalent pressure in approximately 15 minutes.

Vacuum gages having various ranges were provided for measurement of chamber vacuum. During the beginning of the pump-down cycle, chamber pressure was read on an aneroid pressure gage. Lower chamber pressures were monitored on thermocouple and ionization gages located at several positions in the chamber and vacuum train piping.

Radiant Heat Absorption System (Cold Wall)

The primary element of the radiant heat absorption system (fig. 10), specifically configured for the Centaur vehicle, was a cold wall constructed of copper tube-in-strip material (fig. 21). It consisted of a vertical cylinder approximately 20 feet (6.1 m) in

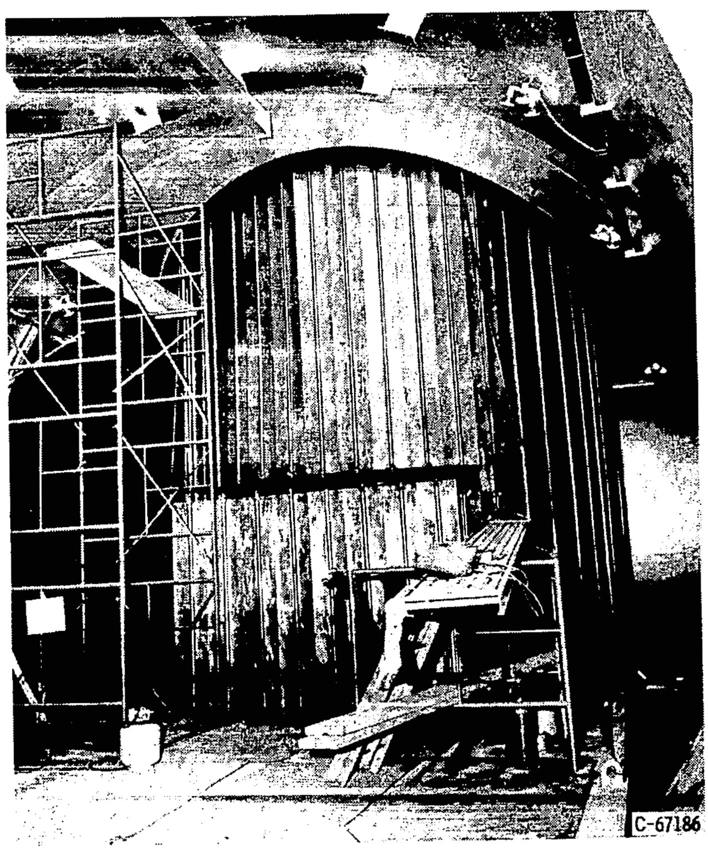


Figure 21. - Installation of copper tube-in-strip cold wall into test chamber.

diameter and 42 feet (12.8 m) high, with ends in the form of shallow truncated cones. The inside wall was painted nonreflective black. The system (fig. 10) operated on the thermal-syphon principle using liquid nitrogen in the tubes as the cooling medium. The liquid nitrogen was supplied from three storage Dewars, 7000 gallons (26.49 cu m) each, with pneumatic remote-reading liquid-level indicators. Remotely controlled self-pressurizing systems were provided for each of the storage Dewars.

During filling of the cold wall, the Dewar pressure maintained a positive pressure to two parallel, electrically driven pumps. Thermocouple outputs at various vertical locations on the cold wall were recorded on a multichannel strip chart which provided an excellent indication of filling progress. A liquid-gas separator tank external to the test chamber contained a float switch to maintain a completely filled cold wall automatically.

Radiant Heater System

Heat source. - A radiant heater system designed for the Centaur vehicle was installed in the Space Power Chamber. The system consisted of six individually, automatic or manual, controlled zones of lamps. The four zones on the forward end of the vehicle are shown in figure 7. Zone 1, a circular array 5 feet (1.52 m) in diameter, consisted of 23 lamps and was suspended over the payload adapter area. A conical array consisting of three zones with a total of 91 lamps surrounded the balance of the Centaur forward equipment compartment electronic equipment. A photograph of a portion of the zone 3 heater is shown in figure 22.

The aft end of the vehicle was heated by two zones having a total of 145 lamps. These aft panels, shown in figure 8, were located parallel to a plane through the axes of both engines.

The lamps were 500-watt tungsten-iodine type and emitted a radiation with wavelengths between 0.4 to 2.2 micrometers. Heating flux could be varied from zero to 200 watts per square foot (2.15 kW/m^2) (approximately 1.5 solar constants).

The radiant heaters at the forward end of the vehicle were designed to be raised to a position that provided sufficient head room for accessibility to the electronic packages. A jig was used to reposition the heaters over the packages before testing.

Facility controls, including those for the radiant heater, vacuum system, and cold-wall, are shown in figure 23.

Calorimeters. - The prime requirement of solar simulator evaluation was the capability of measuring the total irradiance. The capability to perform this task depends on the ability of the detector to function in a simulated space environment and the availability of reference standards against which the detector may be calibrated. The water-cooled calorimeters used were thermoelectric sensors measuring temperature differences

Figure 22. - Portion of zone 3 radiant heater.

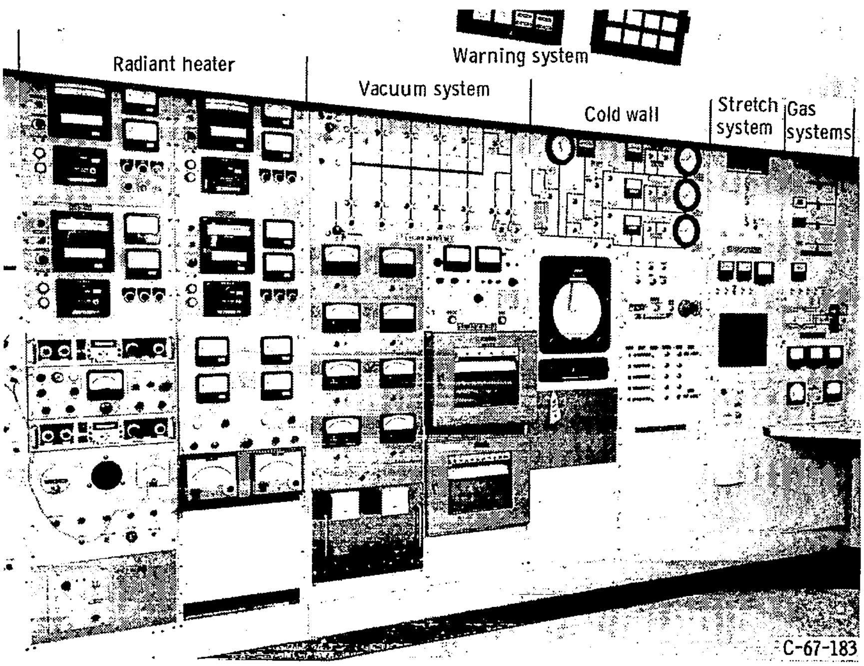


Figure 23. - Facility controls for space simulation chamber.

across a heat conducting path. A calorimeter was placed in each of the four heating zones in the forward equipment compartment of the vehicle.

The detector millivolt output was found to be a function of the pressure of the operating medium. The output from the detector at 1×10^{-3} torr or less is approximately 2.7 times that obtained at sea-level conditions. This characteristic required the detector to be calibrated in a vacuum. In lieu of a nonavailable National Bureau of Standards standard, a calorimeter was used for calibration which had been established by the Lewis Research Center Instrumentation Division as the Lewis "standard."

Fluid Systems

General. - In order to carry out a complete series of environmental tests on the Centaur vehicle, it was necessary that extensive ground support equipment be installed. The necessary gaseous helium, gaseous nitrogen, liquid nitrogen, hydrogen peroxide, and hydraulic systems were installed at the test chamber. Whenever possible, ground support control valves were utilized in series with vehicle valves. For safety reasons all ground support equipment was remotely operated from the control room.

Vehicle propellants. - During actual flight simulation the vehicle was tanked with liquid nitrogen instead of liquid oxygen and hydrogen to eliminate the explosion hazard. To supply liquid nitrogen, a 7000-gallon (26.49-cu m) storage Dewar was installed and was pressurized from an auxiliary gas source. The tanking process was monitored on the vehicle propellant level indicating system.

Gaseous helium. - Two complete gaseous helium systems were installed at the test chamber. One system enabled the filling of the helium storage bottles on the vehicle and the other system supplied helium to the vehicle stretch system. The stretch system was used to support the vehicle in the event of an inadvertent loss of internal tank stabilization pressure. It consisted of three pneumatic cylinders and mechanical linkage that allowed application of a tensile load to the tank structure. The stretch system was controlled from the control room.

Hydraulic system. - A ground hydraulic system was constructed to provide pressure to the vehicle. The Centaur vehicle turbine-driven hydraulic system was inoperative because the main engines were not fired. The vehicle quick disconnects were not employed because previous tests indicated that they leaked at a rate which was acceptable for normal use but not for service in a vacuum chamber. The hydraulic feed lines were heated to maintain the hydraulic fluid within allowable temperature limits when exposed to the cooling effect of the radiant heat absorption system.

During actual test operations, the hydraulic system was operated at a reduced pressure to slow the engine gimbal rate and thereby reduce the possibility of damage to the aft bulkhead. This was necessary because the liquid-oxygen tank was maintained at a reduced internal pressure and did not possess the rigidity of flight vehicles.

Gaseous nitrogen. - Several individual gaseous nitrogen systems were provided. One was an automatic pressurization control system which was used to regulate the Centaur propellant tank pressures during flight simulation. This system incorporated individual subsystems for the fuel and liquid-oxygen-tank pressure control. During chamber pumpdown, the vehicle fuel tank pressure was decreased below atmospheric pressure by the use of two parallel mechanical vacuum pumps, each with a capacity of 140 cubic feet per minute (0.39 cu m/min). When the chamber was returned to sea-level conditions, the vehicle propellant tank pressures were returned to standby conditions by admitting gaseous nitrogen.

Whenever desired, gaseous nitrogen could be admitted into the test chamber to return its pressure to sea level. However, the normal method employed was to admit dry air (dewpoint 10° to 15° F (260.9 to 263.7 K) into the chamber through a manually controlled valve (fig. 20).

Data Systems

General. - The test vehicle as received had extensive instrumentation for use as the second Atlas-Centaur (F2) developmental flight. Eighteen of the landline items and approximately 200 of the 450 recorded telemetry transducers were used for these tests. Vehicle guidance system voltages and 80 event voltages were land-lined. At Lewis, approximately 400 (total) thermocouples were added externally and internally to most of the subsystem packages and the vehicle structure. Instrumentation was also installed for the operation of the facility. A block diagram (fig. 24) presents the overall perspective of data systems.

A computer tabulation of all the instrumentation was made before each test. Listed was an identification number, word description, recorder, channel, scale factors, and other related information.

Telemetry systems. - The telemetry system, composed of three independent subsystems, was used on the Centaur test vehicle for the environmental test program. Since it was desired to obtain general operating data on the telemetry design as a whole, no updating of the existing system was effected to duplicate the exact hardware configuration of the current Centaur launch vehicles. However, it should be noted that there was a considerable degree of similarity both in design and hardware between the system used on the test vehicle and the then current flight configurations.

The radiofrequency outputs of the subsystem were connected to an antenna mounted outside the test chamber and radiated to a telemetry receiving station located approxi-

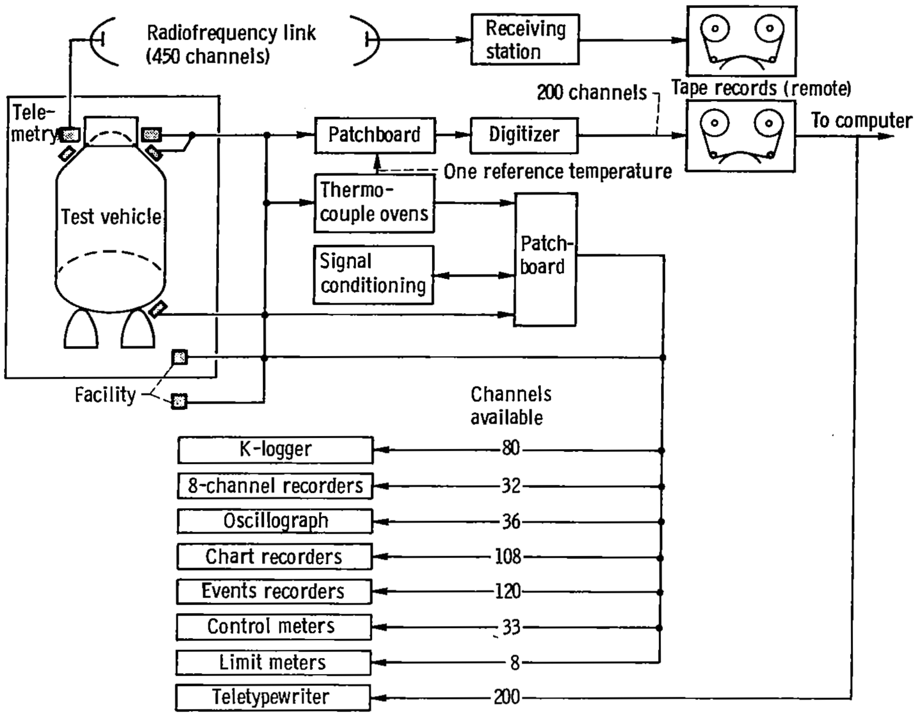


Figure 24. - Block diagram of data system.

mately 800 feet (243.8 m) away. At the telemetry station, all data contained on the telemetry signals was tape-recorded for subsequent selective reduction. The input to the telemetry system consisted of approximately 450 measurements, most of which were multiplexed. Radiofrequency-power and frequency-measuring test equipment was installed in the control room to observe system operation during testing.

Digitized central system. - Some 200 measurements were digitized at the test facility and transmitted by wire to the Lewis central data processing area. These data were tape recorded and preselected data sent back to the control room teletypewriter for test progress evaluation. Later, the recorded data were computed and typed in tabular form in engineering units; and selected data were machine plotted against time with up to six related parameters on each plot. A typical plot is shown in figure 25. A timer was used to start data recording at fixed periods to enable the machine to plot against time.

Thermocouples. - Since the test was a controlled environment test, considerable effort was expended to extract temperature data. Thermocouples were used extensively to monitor vehicle component temperatures. Temperature variations were expected to range from -320° to 300° F (77.6 to 422.0 K). Alloy combinations were selected which were most compatible with the range to be read.

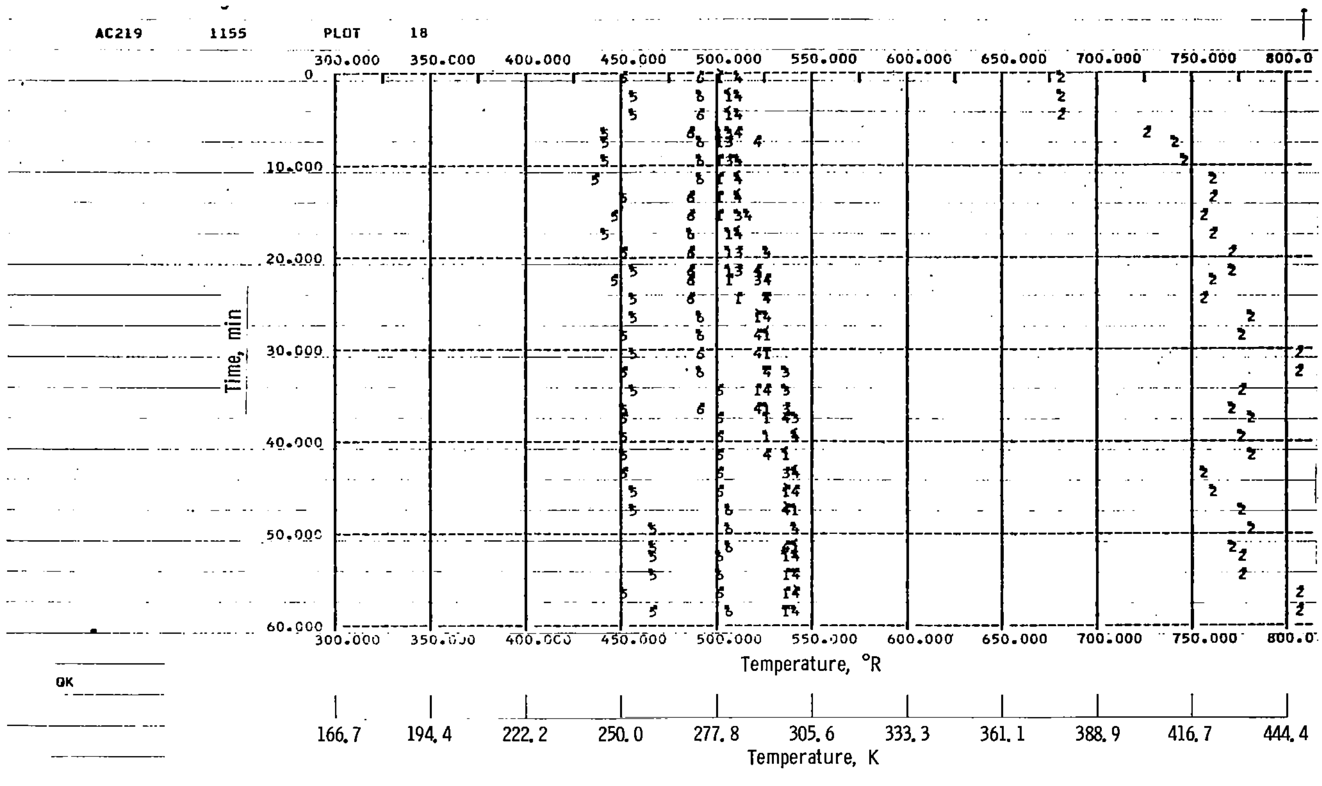


Figure 25. - Typical machine plot of thermal data as function of time.

Thermocouples were mounted on the top, bottom, and cylindrical sections of the cold wall baffle and on the test chamber wall. The test-chamber thermocouples were monitored to ensure that the chamber wall temperature did not fall below -20° F (244 K) because at that temperature the alloy carbon steel structure becomes less ductile and susceptible to fracture.

Initially, the thermocouple system consisted of 216 oven-referenced alloy pairs. An expanded thermocouple system, 200 pairs, was installed when it became apparent that the surface temperature data should be supplemented with internal temperature data for many of the electronic packages. A simplified schematic of the expanded thermocouple system is shown in figure 26. This system required no additional reference ovens. Also, less alloy wire, total conductors, and feed-throughs were required than for a conventional system.

The expanded thermocouple system was connected to a "null" seeking amplifier through a patchboard and a 20-sample-per-second digital scan device. A common bias voltage, referenced to a 150° F (338.7 K) oven, and electrically insulated thermocouple junctions were required throughout the system. The digital system is sensitive to noise (stray voltages) which could cause an erroneous "null" signal to be recorded. Careful

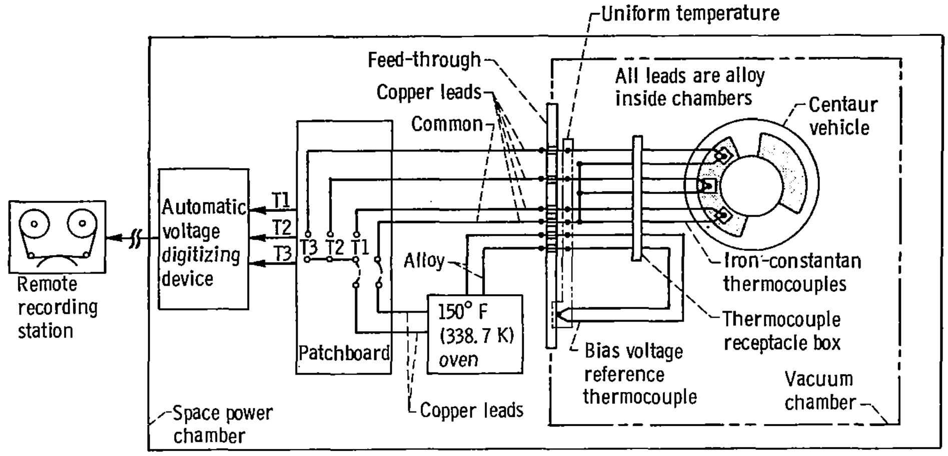


Figure 26. - Simplified schematic of expanded thermocouple system (200 pairs).

shielding, grounding, and elimination of erratic thermocouples provided a satisfactory system.

It was also found that the use of iron-constantan alloy thermocouples presents a problem when the test operations extend over several months. Oxidation of the iron at most of the junctions resulted in inaccurate signals and a need for frequent servicing.

Very few problems occurred in the use of the original 216, individually oven-compensated, thermocouples. Standard grounding and shielding techniques maintained the noise at a satisfactory level. These measurements were reserved for parameters requiring more exact data.

Control room recorders. - One of the most useful test tools used for comparing trends and knowing current temperatures of test components was a recorder called a "K-logger." The unit has 40 charts each with two parameters (a black and a red) recorded against time. Chart recording times of 8 hours and 2 hours were used; the longer period for standby times and in general the shorter period during a simulated flight (1-hr test). The device is a self-balancing potentiometer which prints one point at a time and scans the 80 plots in 1.5 minutes. This unit is shown in figure 27. A typical "K-logger" record, greatly reduced in size, is shown in figure 17. Each small chart is actually 3 by 6 inches (7.6 by 15.2 cm).

Four eight-channel recorders were used to observe transient data. One oscillograph was used for a few high transient measurements.

Pressures. - Airborne instrumentation transducers were connected to the main control room by means of umbilical landlines, or were recorded by the telemetry systems.

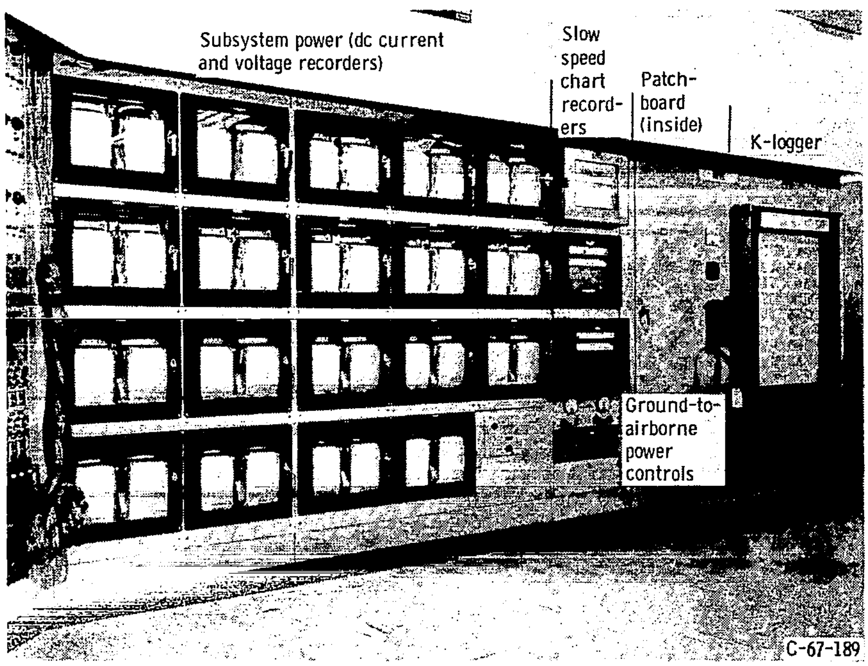


Figure 27. - Recording chart wall of control room.

The pressures in the fuel tank, oxidizer tank, and test chamber were recorded on a multipoint potentiometer-null strip-chart recorder. The recorder was positioned so as to be in view of the test conductor, who was concerned with critical differential pressure relations during test operations.

Direct current power. - To evaluate the electrical power system performance, dc shunts were inserted in the input power leads to all major airborne components. A controlled temperature environment near the forward end of the vehicle was provided for the calibrated manganin shunts. Eighteen dual drive D'arsonval movement strip chart recorders were used. Related current and voltage measurements were displayed on the same dual unit. The currents were an excellent indication of subsystem status.

A photograph of the recording charts in the control room is shown in figure 27. The 18 dual recorders are included in the 19 units shown on the left. An observer was stationed to monitor these charts for out-of-tolerance parameters during all tests.

Alternating current power. - A search for a suitable transducer that would convert accurately 115-volt, 400-hertz power to a high-level recording signal led to the selection of a system which employed the semiconductor "Hall effect." Six special Hall-effect transducers (compatible with 400 Hz) and three voltage phase-shifters were obtained to allow recording the three-phase wye-connected output of the inverter (ac power source) as three single-phase units. Both real power (watts) and reactive power (Vars) were measured. A schematic of this power measuring system is shown in figure 28.

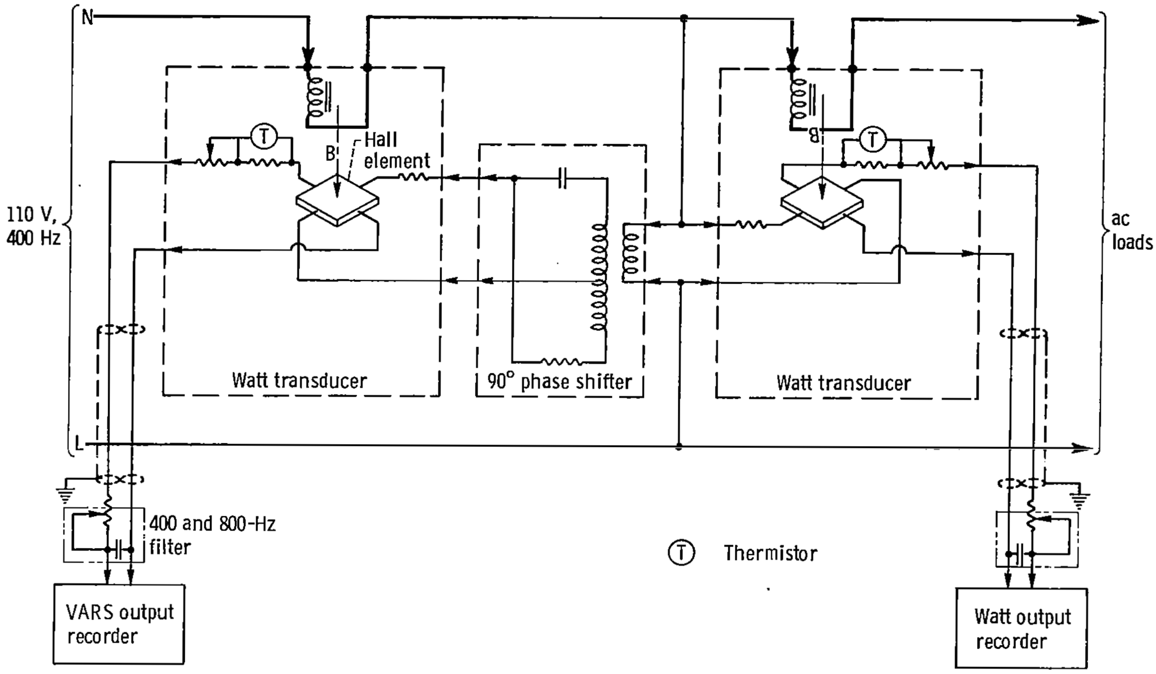


Figure 28. - Schematic of three-phase, 400-hertz power-measuring system. (Single-phase system shown: typical for each phase.)

A 200-watt load would produce a signal of 50 millivolts. The transducers proved to be satisfactorily linear in the range used. By connecting the 90° phase-shifting network to these transducers, it was possible to obtain a good quadrature signal which was calibrated with reasonable accuracy to record load reactive power. The only problem encountered with this system was an inherent double-frequency transducer output superimposed on the desired dc signal. The problem was solved by using a resistance-capacitance filter network in the signal line.

Time code. - In order to correlate all strip-chart recorded data in real time, an elaborate system for time-coding each chart was used. The Lewis central receiving station receives the U.S. Government time signal broadcast by radio station WWV. The signal is processed by a time-code generator and sent as a time-coded signal to the various test sites. The time signal is pulse-amplitude and width-modulated on a 10-kilohertz carrier.

In the control room, the signal was detected by a time-code translator which powered the "event" pens of each of the analog recorders. By means of logic circuits in the time-code translator, several printout rates were used to be compatible with the variation of recorder chart speeds.

Events. - The Centaur programmer issues "discretes" which activate various sequential functions. These discrete signals were connected from the test vehicle to the control room by means of the landlines. Event recorders in the control room received the signals and related the functions in real time.

Elapsed time indication. - Performance degradation with total time of operation of all the vehicle electronic packages is of interest in evaluating component reliability. To record all package accumulated operating times, each unit was connected to an elapsed-time indicator. Each indicator was activated whenever power was applied to its associated package. The time indicators read out in digital units indicating tenths of a minute.

Before and after each test run, the indicator readings were recorded to preserve a running time record of each component. The photograph (fig. 29) of one area of the control room shows these elapsed-time indicators as well as the event recorders, time code control, and canister pressure controls.

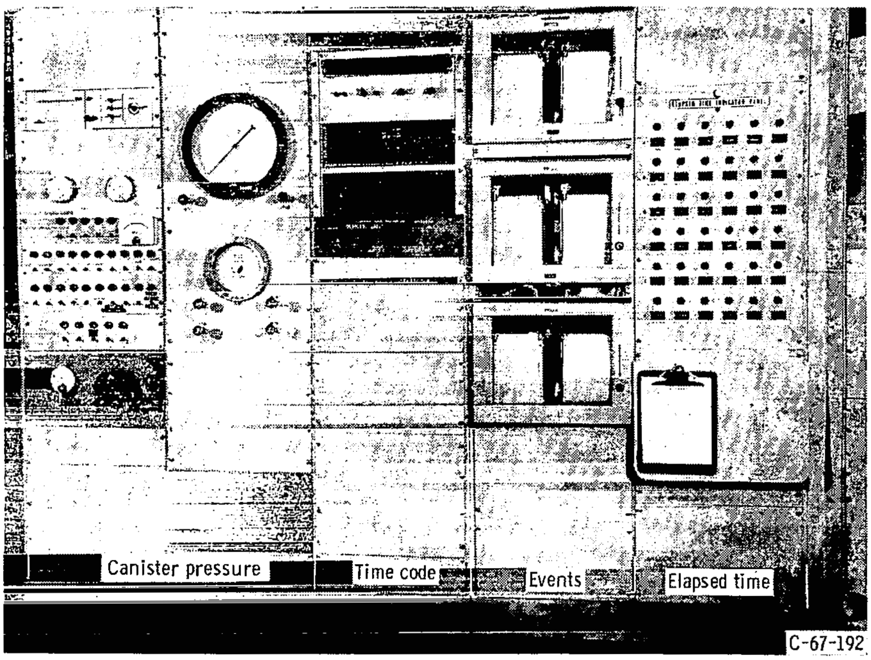


Figure 29. - Canister pressure control with time-code, events, and elapsed-time recorders.

Tracking systems. - Ground support test panels were connected by low loss coaxial cables to the airborne C-band and Azusa Beacon Tracking Systems for verification of proper operation.

Electrical Systems

General. - Facility installations provide dc and 400 hertz power to supplement the vehicle airborne systems. Also, 60 hertz power was required to operate control room ground support equipment (GSE) and instrumentation.

Direct current system. - The ground source direct current (dc) for the vehicle main power was supplied from a 300-ampere, 20- to 32-volt dc variable rectifier supply set for 28 volts at the vehicle. The power supply converts three-phase, 440-volt ac, 60-hertz line power to an essentially ripple free dc potential for use by the airborne electronics. Direct current excitation power to the various Centaur components was controlled by the GSE located in the main control room. To minimize long cable runs and reduce heating losses, a relay transfer station was located between the vehicle and the control room GSE. This is similar to that used at the Centaur launch complex 36 at Cape Kennedy.

Two modes of vehicle operation were available to the test conductor. The vehicle systems could be operated in either an internal or an external mode. This was accomplished by an airborne motor-driven, multipole, double-throw switch known as the main power changeover (MPCO) switch. When the switch was in the external position, vehicle power to each component was controlled by relays energized through GSE function switches.

When the MPCO switch was in the internal position, the vehicle was isolated from ground power and all vehicle components were supplied from the vehicle airborne batteries. To avoid the large cost of batteries, the battery cables were connected to the ground supply through special cables. The test vehicle had two main vehicle batteries (one for telemetry) so two sets of simulator cables were installed as shown in figure 30. Further, high-current switches were installed so that either an airborne battery or battery-simulate mode could be selected while still on internal power. This redundancy was desirable because several test runs were scheduled for the same pumpdown (vacuum environment), and a depleted battery, which could not be recharged or removed, could be "replaced."

During the time of these tests, the dc ground supply at launch complex 36 had batteries "floated" across it. To be similar to the launch complex, several lead-acid batteries were installed across the Space Power Chamber dc supply. One of these batteries shorted internally during a simulated flight and these batteries were removed.

Alternating current system. - The Centaur airborne electronics require precise three-phase, 115-volt power chiefly to the guidance and autopilot gyros. To provide backup to the airborne system, a second flight-type inverter was installed external to the test chamber and forced-air cooled. Switches were used similar in function to those of the battery simulation system. The capability was thereby provided to allow selection

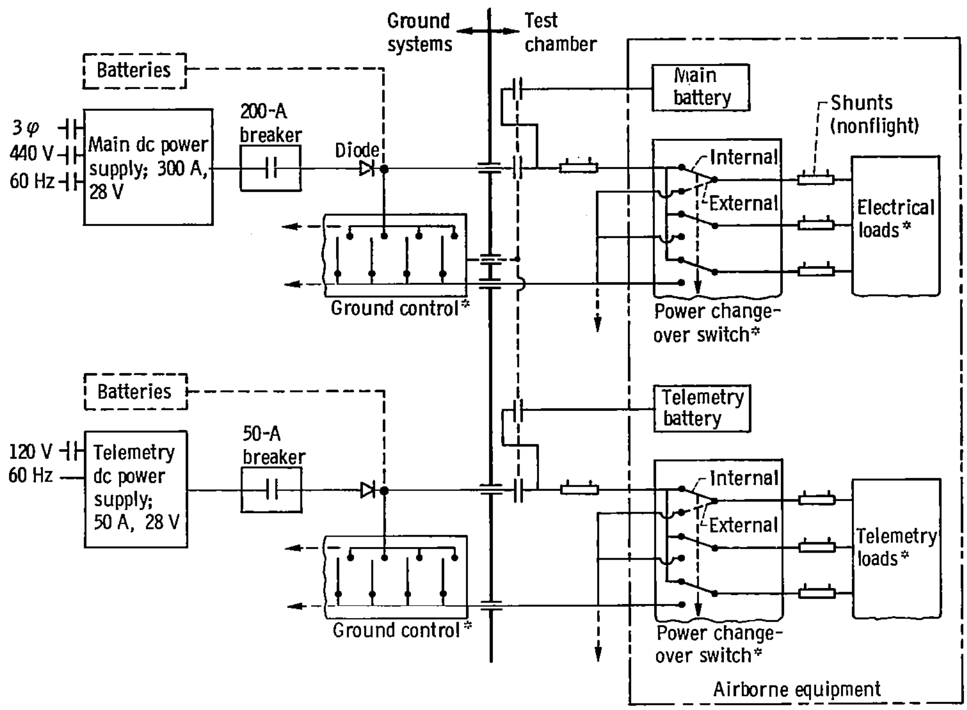


Figure 30. - Direct current power system. (Asterisk denotes components that are only partially shown.)

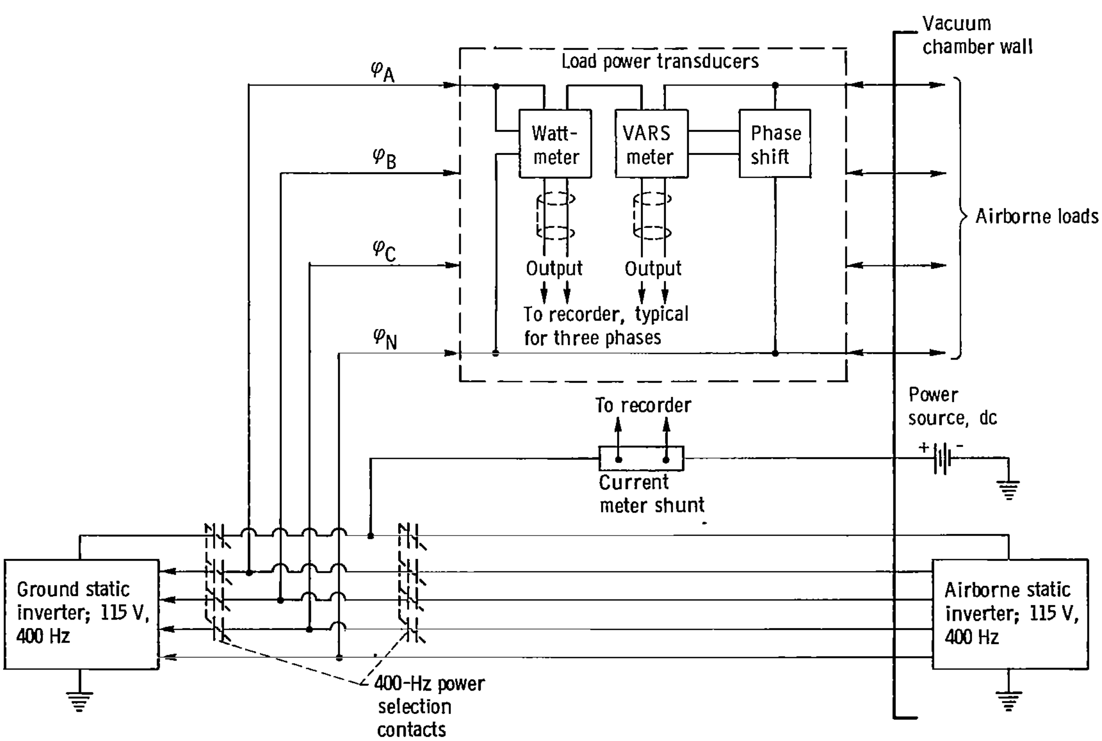


Figure 31. - Alternating-current power system.

of a "ground inverter" mode of operation, when desired, regardless of the position of the MPCO switch. The ground 400 hertz power source was a flight-type inverter to preclude the possibility of spurious power waveform.

The ac power system distribution and control apparatus are shown in figure 31.

Other Facility Systems

Annunciator system. - The annunciator alarm system called to the attention of the control room personnel conditions which were potentially or immediately dangerous. In the event that a facility, GSE, or vehicle system, under the annunciator's surveillance, exceeded a set limit, an audible alarm would sound and a lamp would light on the monitor panel to indicate the affected system. The audible alarm would continue to sound until the condition was corrected or an acknowledge switch was actuated.

Particular parameters could also be monitored by variable-limits meters. These meters were connected to one indicator of the annunciator.

Safety systems. - A stainless-steel basin was built under the cold wall (and therefore vehicle) to catch any cryogenic fluid if a major spill occurred. The test chamber wall would likely fail due to thermal shock if it were exposed to cryogenic temperatures.

In case of electrical power failure or loss of test chamber vacuum, a quick transfer of the liquid nitrogen from the cold wall could be made into a disposal tank. Manual control valves were also provided for control of vehicle and stretch pressures with standby pressure supply systems.

A system was installed which provided a means for detecting a fire and flooding the vacuum pump house with carbon dioxide. In the event of a fire, a warning sounded, and the heating and ventilating system would automatically shut down and the pump house would be flooded with carbon dioxide.

Television system. - A closed-circuit television camera with remote pan-and-tilt was installed in the vacuum chamber. This camera was located at the aft end of the Centaur vehicle and was used to observe engine gimbaling action during flight simulation tests. The monitor and the camera controls are located in the control room.

During long periods between flight simulations, it was necessary to maintain power to the television camera in order that the camera remain warm enough to operate. It was also undesirable to operate the vidicon during part of the pumpdown period because of high voltage arc-over (Paschen's law) at the critical pressure around 80 000 to 90 000 feet (24.38 to 27.48 km) altitude.

Vehicle Ground Control Systems

Flight control (autopilot). - The ground support equipment (GSE) for the flight control system is shown in figure 32. These panels allowed manual control as well as armed and disarmed modes of automatic operation. In the automatic modes, the vehicle timer could be started, stopped, and reset. Lamp monitors were provided for all timer functions.

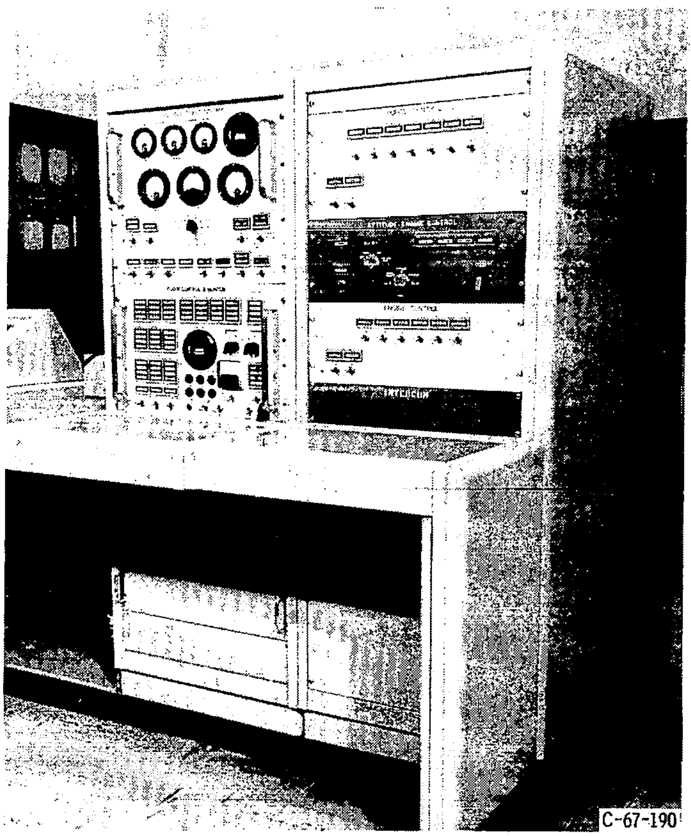


Figure 32. - Vehicle flight control panel.

In addition to the autopilot functions, this GSE provided for manual control of the hydrogen peroxide engines and monitoring of vehicle dc voltage, and (by switching) the voltage and frequency of the three phases of the 400-hertz inverter output.

Propellant tanks. - The GSE for control of propellant loading (and unloading) and the pressurization and vent control are shown in figure 33.

Inertial guidance system. - GSE was used to check out, calibrate, and aline the missile guidance set (MGS) after installation on the test vehicle.

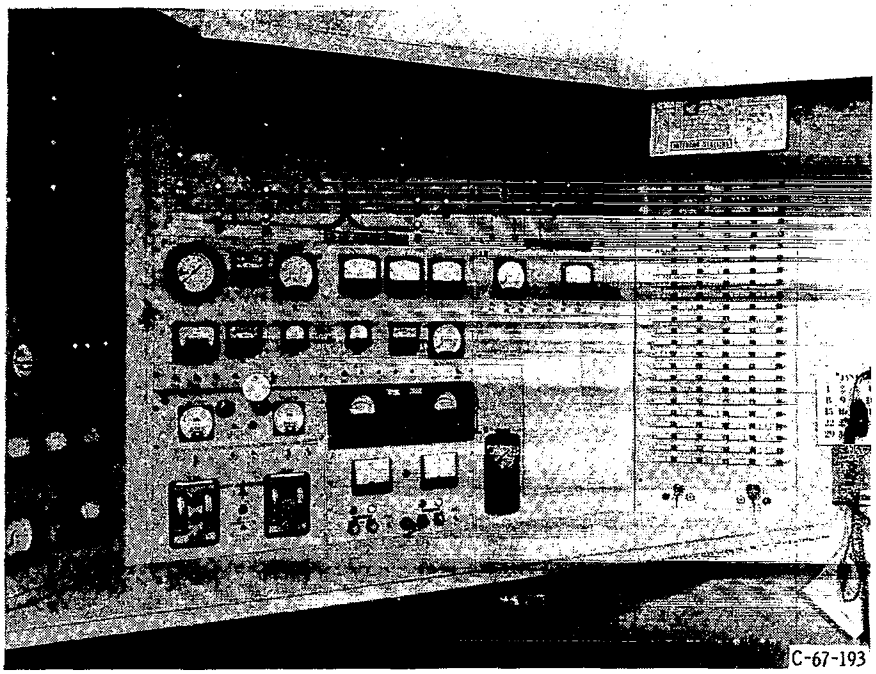


Figure 33. - Vehicle propellant and pressurization control panel.

Canister purge system. - A canister purge system, for monitoring and adjusting package pressure, was installed in the test chamber. The controls are shown in figure 29. By use of the canister purge system, internal pressure in 16 individual packages could be varied to desired levels. Specifically for the four guidance packages, the proportions of helium and nitrogen in the internal gas could be altered.

APPENDIX C

SOLAR AND EARTH RADIANT HEAT SIMULATION

General

The requirement of a solar simulator is to produce irradiance equivalent to that of the Sun at the top of the Earth's atmosphere (one Earth solar constant). The value of 442 Btu per square foot per hour (1393 W/m^2) was used for these tests. An effort was made to obtain a light source standard; however, it was not possible to obtain a National Bureau of Standards standard with capacity to calibrate meters used to measure the flux from the solar heat simulator. Reference 9 indicates that NBS has now developed a high-intensity standard irradiance.

The Earth reflects an average of 40 percent of the incident solar radiation. Earth-thermal radiation absorbed was taken to be a constant value of 66 Btu per square foot-per hour (208 W/m^2).

Solar Constant (Background)

A number of determinations of the Earth solar constant have been made during the last half century. The classical high-altitude surface measurements were made by Dr. Abbot at the Astrophysical Observatory of the Smithsonian Institute over a period of 40 years.

The irradiance figure which has been generally accepted in this new field of interest, space solar simulation, is credited to Francis S. Johnson who revised Dr. Abbot's data. The value is the accepted solar constant, although there remain some questions as to the accuracy. A 1965 survey is discussed in reference 10.

Spectral Emission

Solar and Earth. - The thermal radiation from the Sun is confined essentially to the wavelength range between 0.2 and 4.0 micrometers in which more than 99 percent of the total radiation is emitted. The Earth albedo radiation is in the 0.3- to 4.0-micrometer wavelength region (ref. 11). The Earth-atmosphere thermal radiation consists of infrared energy nearly all at wavelengths longer than 4.0 micrometers.

Simulator lamp. - The 500-watt tungsten-iodine lamps used in the simulator emit a radiance with a wavelength of 0.4 to greater than 2.2 micrometers.

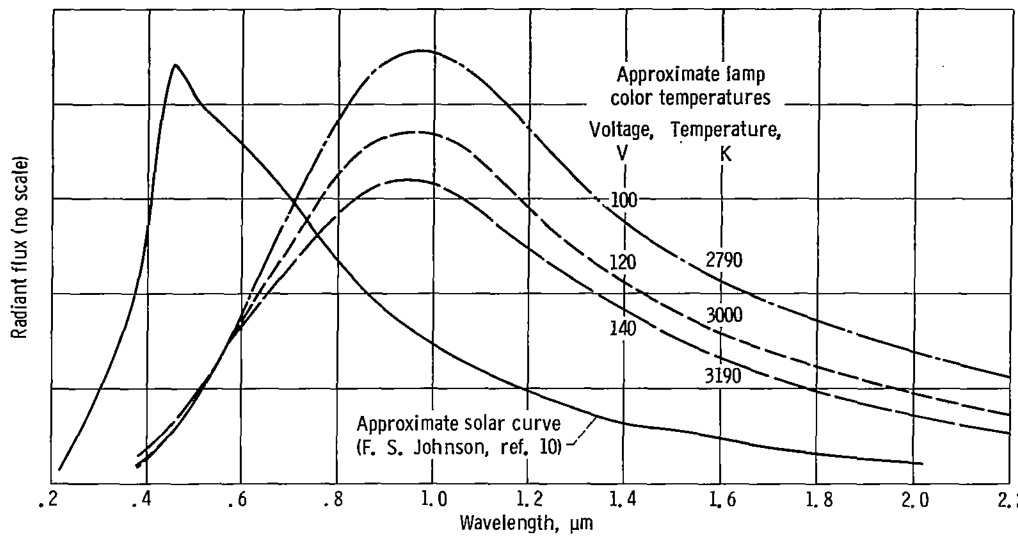


Figure 34. - Spectral emission of Sun and simulator lamp at various voltages.

Comparison. - The spectral emission of the Sun (F. S. Johnson curve) and of the simulator lamp are shown in figure 34. As shown in the figure, the spectral energy distribution of the tungsten-iodine lamps used in the solar simulator does not duplicate the energy distribution received from the Sun. Although they are similar, the Sun's peak energy is at a wavelength of about 0.45 micrometer while the peak energy of the simulator lamp is at about 0.95 micrometer, which is higher in the infrared.

Spectral Absorption - Centaur Package Coatings

The total normal spectral characteristics of Centaur paint coatings were determined by Convair-Astronautics in 1961 to provide the required thermal radiation control for the various parts of the Centaur vehicle (ref. 12). An absorption between 0.05 to 0.95 of the total solar heat flux, and an emittance of 0.05 to 0.95 is possible by selective color coating. Many coatings were investigated. The three most used, for Centaur application, are: flat white acrylic, aluminum enamel, and gold plate. Table II presents data for total solar absorption and emittance (400 K) for these three coatings and flat black enamel.

The curves of spectral absorption for the same three colors are shown in figure 35. These data (ref. 12) were obtained by measuring reflectance of samples as a function of wavelength. For samples which are opaque, the sum of the reflectance and absorptance at the same wavelength is 1.

TABLE II. - ABSORPTION AND EMITTANCE FOR FOUR PAINTS

[Data from ref. 12.]

Coating	Total solar absorptance, α_s	Emittance (400 K), ϵ	Ratio, α_s/ϵ
Flat white acrylic	0.411	0.901	0.456
Aluminum enamel	.369	.243	1.52
Gold paint	.507	.360	1.41
Flat black enamel	.928	.800	1.16

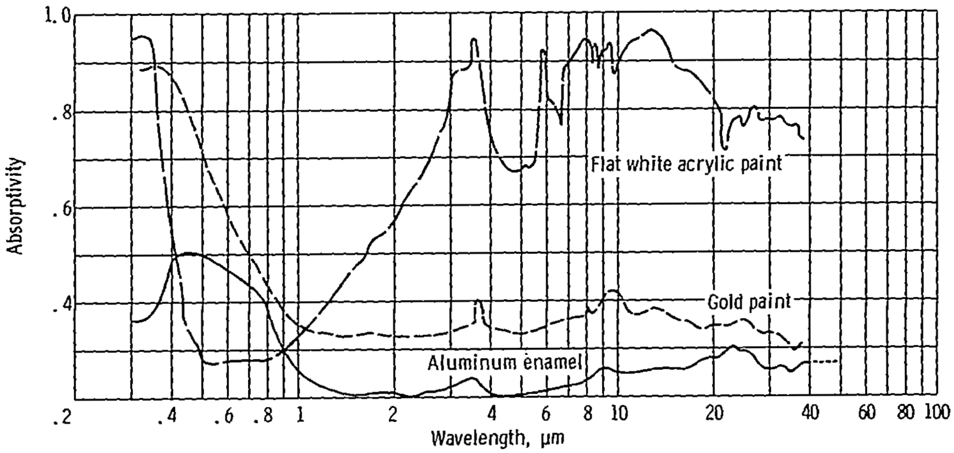


Figure 35. - Spectral absorptivity for three Centaur points.

Correction for Spectral Energy Distribution

Requirement for correction. - Once the thermal input desired has been determined (discussed in appendix D), a knowledge of the effectiveness of the simulator to provide the desired thermal input is required. Due to the difference in spectral absorption for various coatings, each color package must be treated separately. Because the spectral emission from the simulator lamps is different from that of the Sun, it was required that the total lamp absorptances be determined.

Method for obtaining corrections for spectral energy distribution. - The absorption of the radiant energy by any given color package on the test vehicle was calculated by the weighted-ordinate method. Such a method is described in detail in reference 13.

The product of the source energy (at a specified lamp voltage) and the absorptance value of the surface being heated at identical wavelengths provide a weighted value of flux that will be absorbed from that source at that wavelength. The ratio of the areas under plots of these weighted numbers and of the source flux, both against the full spectrum of wavelengths, gives a number signifying the total absorptance of the specimen for radiant flux of the specified radiant source.

APPENDIX D

THERMAL INPUT MODEL STUDY

General

In order to formulate a thermal input program, a method was needed which would quickly provide information on Sun and Earth-albedo radiation incident to the Centaur electronics packages during various vehicle orbital attitudes. A one-quarter-scale model was used to obtain these data.

Flight Profile

The extremes of possible Centaur-Surveyor launch trajectories are shown in figure 14. Approximately 200 seconds after launch, the nose fairings are jettisoned and space radiant thermal effects are present. Orbit is achieved in approximately 9 to 10 minutes. The coast phase may vary from zero to 25 minutes. The Centaur electronic equipment is turned off approximately 30 minutes after termination of coast phase. Therefore, for the environmental tests, a maximum exposure time of approximately 1 hour was assumed. From figure 14, it can be seen that the forward equipment compartment receives either full Sun early followed by nearly no Sun, or the reverse. Full Sun for the full time would be an extreme worst heating case, and zero solar and Earth heating would be a worst cooling case.

The Centaur vehicle is controlled so that during coast the axis of the vehicle is always parallel to the surface of the Earth. The allowable roll rate is such that less than one rotation would be expected to occur during the coast period.

Layouts were made of the Sun, Earth, and vehicle relations against flight time for early and late launch times. Actual flight roll, velocity, and distance-covered data were used. From these layouts, the direction, in relation to the vehicle, of the Earth and Sun against time could be determined.

Model of Centaur

Using expanded polystyrene material, a one-quarter-scale model (fig. 36) of the Centaur forward equipment compartment was constructed. The model consisted of the forward bulkhead, the equipment platform, payload adapter, and all major electronic packages. These elements were mounted to correspond with the actual AC-8 configuration

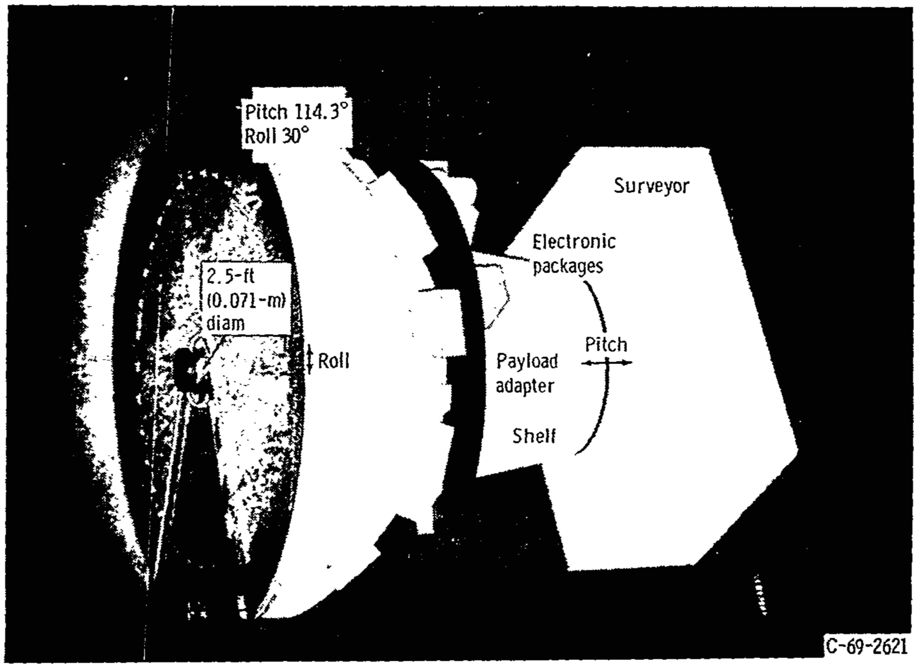


Figure 36. - One-quarter-scale model of Centaur forward area for use in radiant thermal input program.

as shown in figure 3. In addition, a simulated Surveyor payload was mounted on the payload adapter of the model. The payload was not a facsimile reproduction, but rather a composite representing the operational Surveyor envelope. The entire unit was mounted so as to be capable of rotating about its roll and pitch axes. Degree markings were scribed on the periphery of the model's base in order that the different roll attitudes could be accurately set. The model itself was mounted on a swivel base enabling the setting of desired pitch attitudes. The unit was designed so that all its elements could be easily disassembled to make up new vehicle configurations. The electronics packages along with the equipment platform were painted different colors to provide contrasting backgrounds.

Procedure

The model was erected in a photographic dark room and a moderately collimated light source was provided to simulate the Sun's rays striking the Centaur forward area. This collimated source was provided by means of a parabolic reflector passing light rays through a converging lens system. The need for some degree of collimation was brought about by the requirement for the same sharply delineated light and shadow regions as encountered by objects in space. A camera was located adjacent to the light source.

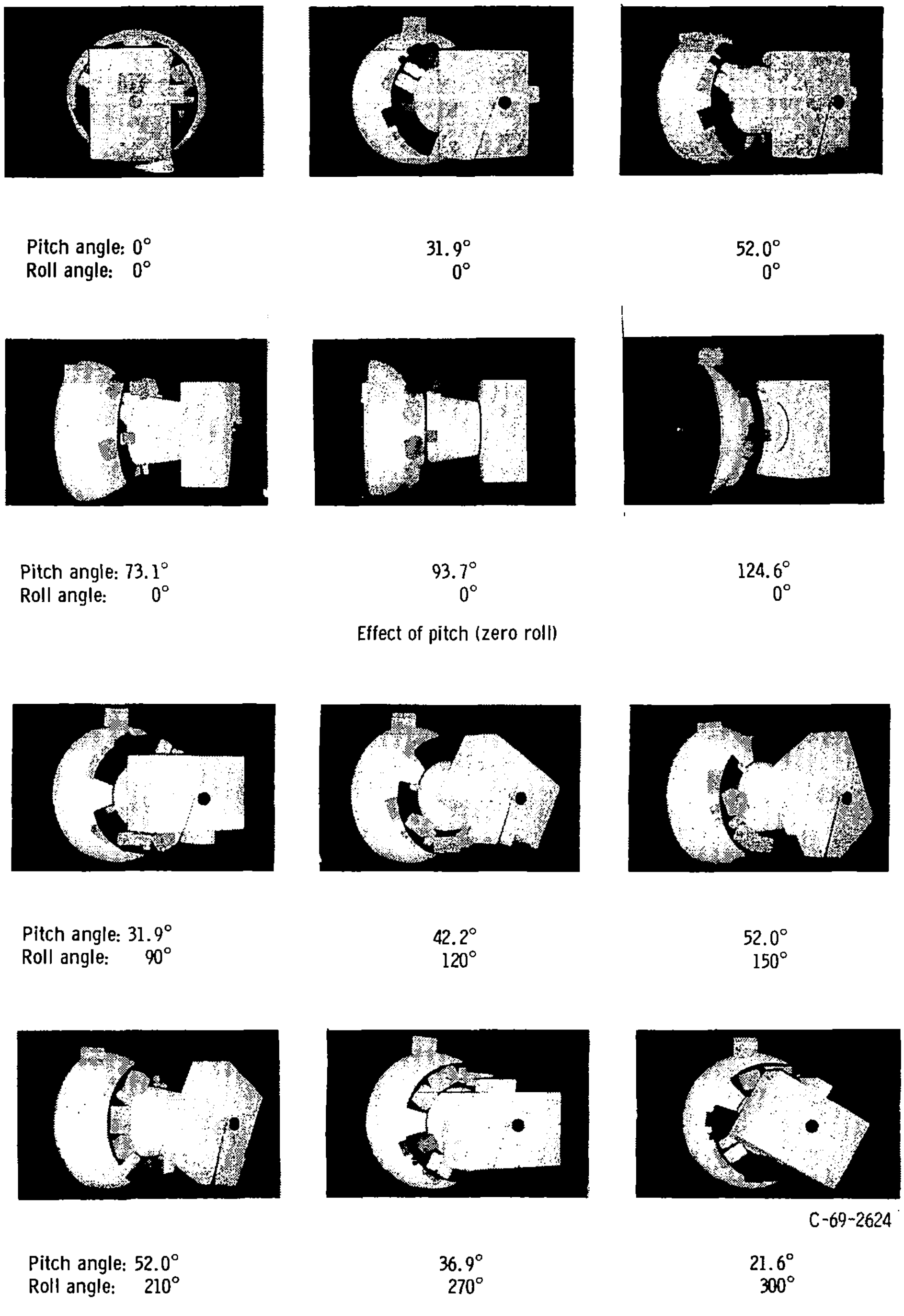


Figure 37. - Typical series of photographs of one-quarter scale model for varying pitch and roll.

Photographs were made of the model as it was positioned in different simulated attitudes. Part of a typical series of photographs is shown in figure 37. This process was continued to include the whole spectrum of clockwise and counterclockwise roll as well as the different pitch attitudes at each roll setting. A reference surface of known area was included in each photograph to aid in data reduction.

Enlarged prints were made using a microfilm printer and the projected area of each visible electronic package was measured using planimeters. This reading was adjusted by a multiplication factor, derived from the planimetering of the known reference area in each photograph and by a scale factor, to give the equivalent projected area on the full size Centaur, as seen by the Sun, in the simulated attitude.

Using the solar constant of 442 Btu per square foot per hour (139 W/m^2) and correcting for total solar absorption of the specific package, then by multiplying by the projected area values the solar heat input to each package was obtained for discrete increments of mission time.

A similar approach was used to determine heat inputs based on Earth albedo and Earth thermal radiation. However, the heat from Earth albedo is a variable. The thermal flux from the Earth albedo with geometric relation between vehicle, Earth, and Sun is shown in figure 38. The Earth albedo was computed from an alined Earth reflectance of 0.4 as given in reference 14. Also given in reference 14 is the constant value of 66.36 Btu per square foot per hour (209.2 W/m^2) for Earth thermal radiation.

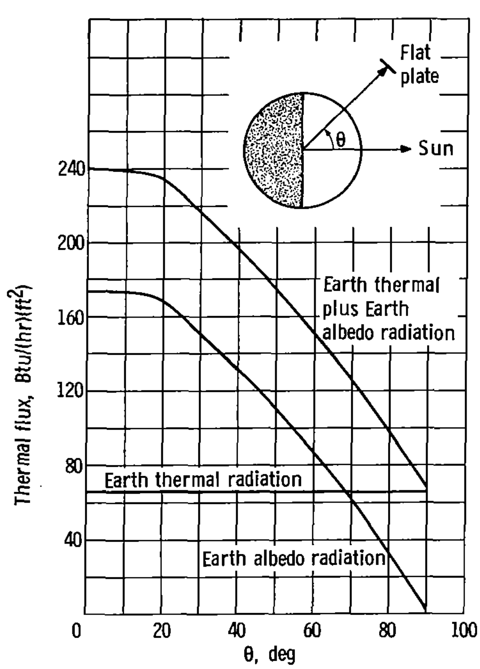


Figure 38. - Earth thermal and Earth albedo radiation as function of θ for a 1-square-foot (0.305-sq m) flat plate normal to Earth at altitude of 100 nautical miles (185 km).

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