SPACE SIMULATION AND FULL-SCALE TESTING IN A CONVERTED FACILITY

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FACILITY

The Space Simulation Chamber described in this paper was converted from a former wind tunnel complex (fig. 1). Conversion consisted of removal of the turning vanes, cooling coils, and circulation fan and the addition of three bulkheads. The large chamber is approximately 60 feet in diameter with a volume of 250 000 cubic feet and is utilized for tests to altitudes of 100 000 feet. The second, smaller chamber, and the one of interest, is approximately 30 feet in diameter and 100 feet long and has a volume of 70 000 cubic feet. A $22\frac{1}{2}$-foot extension was added to provide a total height of 45 feet in order to accommodate the Centaur test vehicle. Present vacuum pumping capability is 2×10^{-6} millimeter of mercury, and with the Centaur test vehicle installed an ultimate of 2×10^{-5} millimeter of mercury is attained. Figures 2 to 5 show various views of the Space Simulation Chamber (SSC) and the installation of the Centaur vehicle.

The vacuum system consists of ten 32-inch, 50 000-liter-per-second oil diffusion pumps, one 30 000-cubic-feet-per-minute blower and two 7500 cubic-feet-per-minute roughing pumps. Pumpdown time to reach ultimate vacuum is 24 to 30 hours. The vacuum in the chamber is broken by using dry air (-20° F dewpoint). Breaking the vacuum normally takes 2 hours. A continuous purge with dry air is used when the chamber is open for repairs.

Figure 6 is a pictorial view of the modified chamber used for space simulation with the Centaur test vehicle installed. A liquid-nitrogen cold baffle, tailored to the Centaur test vehicle and consisting of an assembly of copper finned tubes has been installed. It is approximately 20 feet in diameter and 42 feet high. The thermal syphon system is utilized to maintain a full baffle, with three 7000-gallon Dewars supplying liquid nitrogen to the system.

A solar simulator system tailored to the Centaur test vehicle consists of six separately controlled zones of quartz-iodine (500 W) lamps. There are four zones in the forward array and two in the aft. These arrays provide approximately 60 percent open area between the vehicle and the cold baffle and are designed to approximate a collimated light source.

In order to ensure good vacuum capability, the 15-foot access door was provided with a dual O-ring seal. The dome which seals off the extension was provided with a 1/4-inch-plate lip seal, which when welded shut makes the chamber vacuum tight. During modification it was found that the original welds, made some 20 years ago, were structurally sound but contained many porous areas. As a result, all old welds were removed and replaced with high-quality welds. The interior surfaces were sandblasted and given two coats of aluminum paint for sealing.

All chamber penetrations and joints within the chamber were leak checked with helium using a helium leak detector to determine leak rate. All penetrations and joints were required to have less than 1×10^{-8}-atmospheric-cubic-centimeter-per-second leakage.

CENTAUR INVESTIGATION

The Centaur vehicle under test consists primarily of a 10-foot-diameter-pressure-stabilized tank made of 0.010-inch-thick type 301 stainless steel. Most of the electronic and control systems and components are mounted on the forward end, and most of the mechanical and propulsion systems are mounted on the back end. The RL 10 engines, which will not be fired for these tests, are of early vintage. All remaining components have been updated to match current flight configurations.

The purpose of the investigation on the Centaur vehicle is to evaluate the performance of the various vehicle subsystems under simulated thermal conditions encountered during the coast phase of flight. All systems will be operated to determine if there are system interferences as well as to provide operating histories of each component. Parameters being monitored are

(a) Package (component) temperature

(b) Package (component) pressure

(c) Operating time

(d) Power consumption

(e) Package or system output

In addition to the normally supplied facility systems that are required to accommodate the Centaur test vehicle, it was necessary to add several others. One such addition was a pneumatic system with automatic controls for maintaining a nearly constant pressure differential across the vehicle tank structure during, before, and after a pumpdown. This was necessitated by the fact that the structure was not designed to withstand repeated pressure (and resulting stress) cycles inasmuch as such cycling would cause fatigue of the tank structure. The system as now operated maintains the fuel tank pressure between 4 and 6 pounds per square inch above the chamber pressure, and the oxidant pressure 8 to 10 pounds per square inch above fuel tank pressure. In addition to these pressure systems, a pneumatic stretch system was added. This system, which will support the vehicle in the event of an inadvertent loss of internal pressure, makes use of three pneumatic cylinders and mechanical linkage for applying a tensile load to the tank structure.

Inasmuch as the main (RL 10) engines, which provide the power to drive the main hydraulic pumps, are not to be fired during these tests, it was necessary to provide an external hydraulic power supply to the engine gimbal system so that it could be actuated during a simulated flight. In a normal preflight ground checkout, the vehicle pneumatic and fluid systems are connected to ground services through quick-disconnect devices. These devices have a stringent leak rate specification, which is dictated by the flight requirement, but even leakage within the specification will degrade the vacuum in the chamber. Thus, the quick disconnects had to be replaced with AN fittings. Further modifications to the vehicle systems consisted of ducting all vents and relief valves to a point outside the space chamber.

Auxiliary heat was added to some of the components in order to maintain normal flight temperatures during pumpdown. This additional heat was necessary because some of the airborne components on Centaur are normally exposed to an abnormally low temperature environment for only a short period during flight, whereas in this test setup, a long period under extreme cold is required to reach simulated space conditions. Four separate areas are conditioned in this manner: two with radiant heat lamps (aft instrument box and destruct), and two with wraparound heaters (hydraulic and hydrogen peroxide supply lines). A possible consequence of the long pumpdown time to reach space simulated pressure environment is some gas leakage of the pressurized airborne electrical packages so that they would not be at normal internal pressures. Therefore, a separate system was installed to provide the capability of monitoring and adjusting pressures where required. This system functions for 16 of the packages and can supply either of two pressurant gases.

In order to maintain a reasonable heat balance on the overall vehicle during pumpdown and activation of the liquid nitrogen baffles, the solar simulator is operated in an automatic mode that utilizes slug-type calorimeters with a "feedback" thermocouple connected to the controller unit. In addition, water-cooled calorimeters are used to measure the amount of heat in each of the six heat zones on a nominal target plane to check the operation of the automatic system.

At the start of each test the airborne packs must be conditioned to the temperatures that they would have at that point in flight time. In general, raising the temperature is accomplished either by increasing the heat produced by the solar simulator panels or by operating the individual packs and generating internal heat. Where packages require cooling, the solar simulator in the appropriate zone is operated at reduced or zero power, or cold-gas cooling coils installed on some of the larger packages are utilized. (These coils are installed so as not to impair normal solar or space radiation.)

For redundancy during a test, the vehicle electric power supply is arranged so as to provide two parallel systems, airborne and ground, for both 400-cycle and direct-current power. Normal testing is done using the airborne static inverter and batteries, but if troubles develop either or both systems can be switched to the ground power supplies.

The wiring to interconnect the vehicle and control room for both vehicle functions and instrumentation (excluding facility) requires

(a) Controls and airborne instrumentation - 1000 conductors

(b) Hardline instrumentation (including thermocouple) - 900 conductors

Instrumentation for parameter monitoring consists of

(a) 350 channels for temperature

(b) 36 channels for power

(c) 120 channels of events recording

(d) 40 channels of elapsed-time recording

(e) 350 channels of telemetry data

Some of the data are radiated via telemetry and recorded on FM tape at a ground receiving station. Similar data are also recorded via landlines as a check. The data transmitted via telemetry can be used to aid in interpretation of flight data.

Currently the testing program on the Centaur vehicle is just getting underway. Some preliminary runs have been made during which the various facility and vehicle systems have been operated and checked out. One of the most troublesome problems is cross talk or noise pickup in the extensive temperature instrumentation that is installed on the flight packages. This problem has been solved by using an extensive filtering system. Another problem is that of obtaining a reasonable solar heat simulation on the systems and packages within the limitations of the existing solar simulator. It is being overcome by extensive calibrations and some mechanical equipment which will help to approximate a collimated solar source.

Generally, the program requires the investigation of the performance of the systems and components of a parking-orbit Centaur vehicle during that period of time starting with separation from the Atlas booster. Of particular interest are the extremes in environment that could be encountered. Figure 8 shows some extreme (temperaturewise) trajectories that could be encountered during the next flight of a Centaur two-burn vehicle depending on which launch window is met. If the vehicle is launched during the window for November 16, 1965, the trajectory would be as shown and the forward equipment would receive practically no solar heating. If it is launched during the November 4, 1965, window, the forward equipment would receive maximum solar heating. Similar trajectories exist with respect to the propulsion and mechanical systems mounted at the rear of the vehicle.

The trajectory that was actually attempted on the Atlas-Centaur 4 vehicle is shown in figure 9. Unfortunately the vehicle started tumbling about 10 minutes after it was separated from the Atlas and therefore the thermal data after that point in time are not too meaningful. A comparison of the flight and vacuum (SSC) rate-gyro-cover temperatures for the first 10 minutes of Centaur flight is shown in figure 10. Although the starting temperatures were different, the similarity in trends indicates that a reasonable simulation was obtained. Although only preliminary tests have been made to date, a variety of functional difficulties have been uncovered. These are listed in table I. Thus full-scale environmental test chambers appear to be a valuable tool in the development of flight hardware.

CENTAUR NOSE FAIRING SEPARATION TESTS

In addition to space thermal environment testing of full-scale vehicles, the Lewis Space Simulation Chamber was also extremely useful in investigating dynamic problems such as the jettisoning of the Centaur nose fairing. This investigation occurred primarily as a result of the Atlas-Centaur 3 flight, wherein a shock and an interruption of the guidance computer were experienced during nose fairing jettison. Although extensive sea-level jettison tests had previously been conducted without any indication of nose fairing malfunction, it was felt that the jettison events and computer malfunction were related in some manner. A decision was made to investigate the jettisoning of the nose fairing under space simulated pressure conditions (in the chamber) to learn what had actually happened to Atlas-Centaur 3. The prime objective of this investigation was to determine which, if any, of the structural components of the nose fairing were underdesigned and to flight qualify the redesigned nose fairing if necessary. Additional objectives were to determine the nose fairing trajectory, possible interferences, system function and redundancy hinge dynamic loads, and pressure impingement effects on the payload.

A standard flight type fiberglass-honeycomb nose fairing was used for all tests. A photograph of it installed in the facility is shown in figure 11. The salient features of the nose fairing are that it consists of two halves held together by explosive bolts, which are fired just prior to separation, and that the separation force is provided by two high-pressure-gas thrustor bottles installed in the upper end. It was felt that the pressure forces resulting from discharge of the bottles and the resulting nose fairing dynamics could have contributed to the anomaly detected during the Atlas-Centaur 3 flight.

The installation of the full-scale Centaur nose fairing for the Atlas-Centaur 4 flight qualification tests conducted in SSC is illustrated in figure 12, which presents a diagrammatic sketch showing the relation between the nose fairing and the stopper on the left side and the catcher net on the right. Briefly, the stopper on the left in conjunction with a facility-type hinge permitted about 20^{o} rotation of the left fairing, whereas the net on the right permitted about 50^{o} rotation and allowed the fairing to jettison freely in a manner similar to that which occurs in flight (fig. 13). Space limitations (the proximity of the Centaur vehicle) prevented the net from being installed and thus permitting a greater degree of freedom.

The test procedure employed in the nose fairing jettison tests was relatively simple and straightforward. It consisted primarily of pumping down the chamber to the desired pressure and then actuating the nose fairing jettison system in a manner similar to that used in flight. Instrumentation was installed on the fairing to measure pressures, strains, accelerations, and fairing trajectory and motion-picture cameras were installed inside and outside of the fairing to record all events to check the trajectory. Special techniques included the installation of the cameras in hermetically sealed enclosures and the selection of transducers (or the manner of their installation) that would not overheat or malfunction during their long exposure to the vacuum of the chamber.

The chief uncertainty concerning the validity of the test was the change in chamber pressure that occurs during the test. Although thrustor bottle firing is initiated at the correct pressure altitude (100 miles) by the time they are finished discharging the chamber pressure is up to an equivalent altitude of about 100 000 feet.

The answer to this problem was eventually obtained by comparing the chamber data with flight data; however, prior to flight the best that could be done was to compare the sea-level test with the vacuum-chamber test and observe that a considerable difference existed. This difference is illustrated in figure 14, which shows the hardware after the first firing in a vacuum; this compares with no damage for the sea-level firings. A comparison of the nose fairing trajectories obtained at sea level and in a vacuum (Atlas-Centaur 3) is shown in figure 15. The trajectory in a vacuum is much faster than at sea level and accounts for the damage experienced.

Inasmuch as no serious damage occurred when the fairing followed the trajectory labeled "AC-3 Sea Level," it was decided to adjust the bottle pressure and throat size in order to obtain a similar trajectory for flight. When this was done, the trajectory labeled "AC-4 vacuum" was obtained in the chamber. A comparison of this trajectory with Atlas-Centaur 4 flight data indicates good agreement.

A further comparison of flight and vacuum chamber dynamic measurements is presented in figure 16, which shows nose fairing hinge loads during jettison as measured in the vacuum chamber and as measured during Atlas-Centaur 4 flight. Again, relatively good agreement was obtained. Comparison of pressures measured in the fairing thrustor bottle compartment during the Atlas-Centaur 4 flight and in the vacuum chamber also agreed well with a peak pressure of 9.2 pounds per square inch absolute measured in flight and 8 pounds per square inch absolute in the vacuum chamber.

In addition to obtaining good agreement between flight and vacuum chamber data, the chamber proved to be a most valuable tool in developing the nose fairing flight hardware for Atlas-Centaur 4. All of the changes and modifications made between the flights of Atlas-Centaur 3 and 4 were first checked out in the vacuum chamber, and as a result the first completely successful Centaur nose fairing jettison was accomplished on Atlas-Centaur 4.

TABLE I. - CENTAUR VEHICLE EQUIPMENT FAILURES

Unit	Failure mode
1. Main power changeover switch	While in a simulated flight environment, switch failed to respond to command signals to go to airborne position
2. Telemetry system	During an environmental test, subsystems 1, 2, and 4 failed to radiate properly after a primary power source transient
3. Airborne inverter	Unit failed to turn on after long exposure to simulated environmental conditions, even though it was well above the lower red line temperature limit
4. Range safety system	Electrical arming device, after long exposure to liquid nitrogen temperatures, failed to respond to commands

LEWIS ALTITUDE WIND TUNNEL

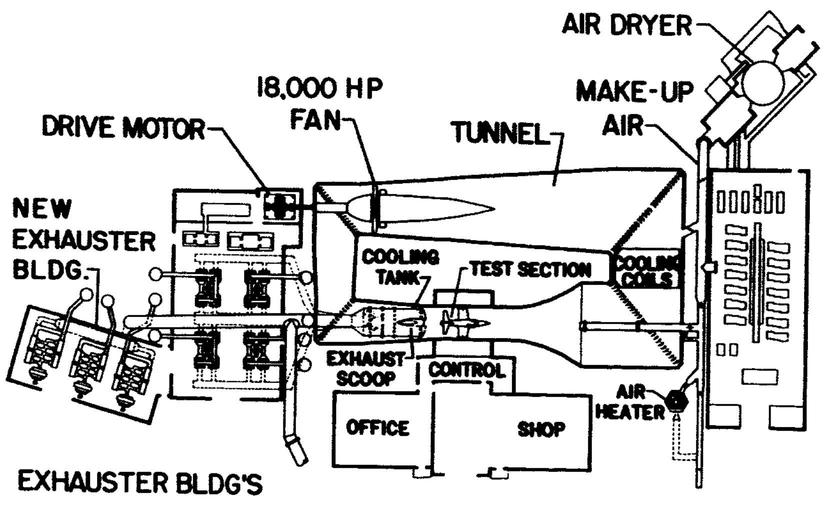

Figure 1.

OVERALL VIEW OF SSC

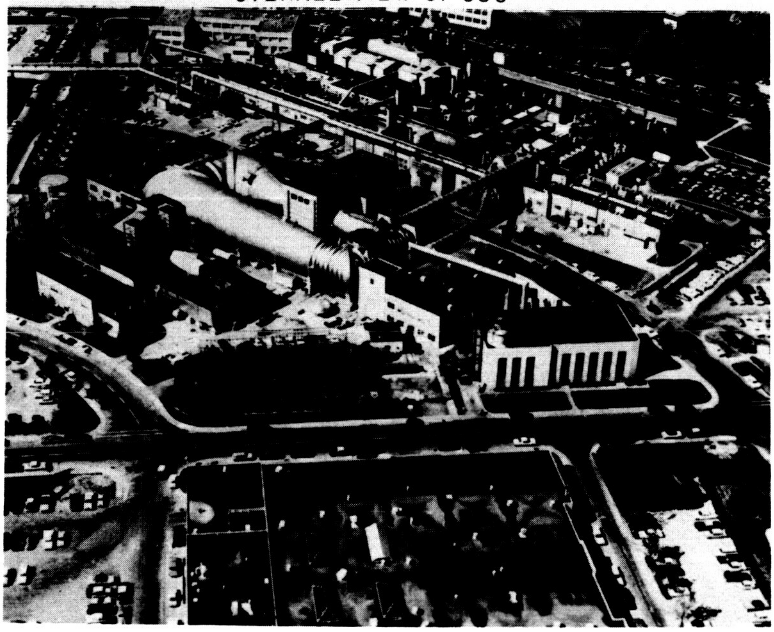

Figure 2

SSC EQUIPMENT ENTRY HATCH

Figure 3

LOWERING OF CENTAUR INTO SSC

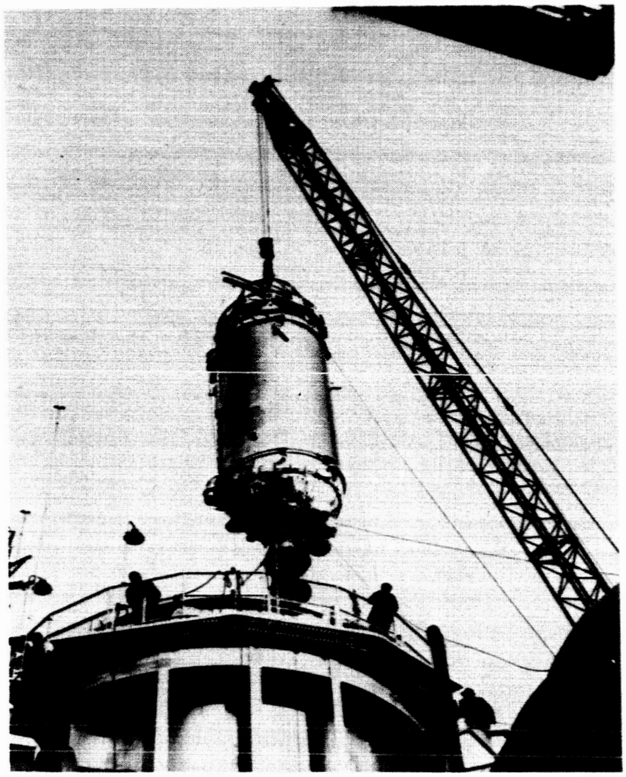

Figure 4

VIEW FROM BOTTOM AS CENTAUR IS LOWERED INTO SSC

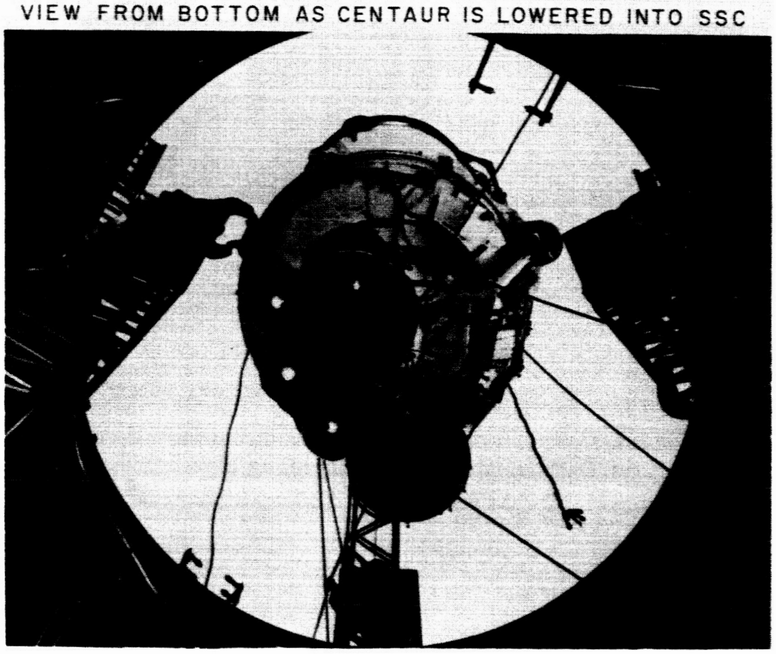

Figure 5

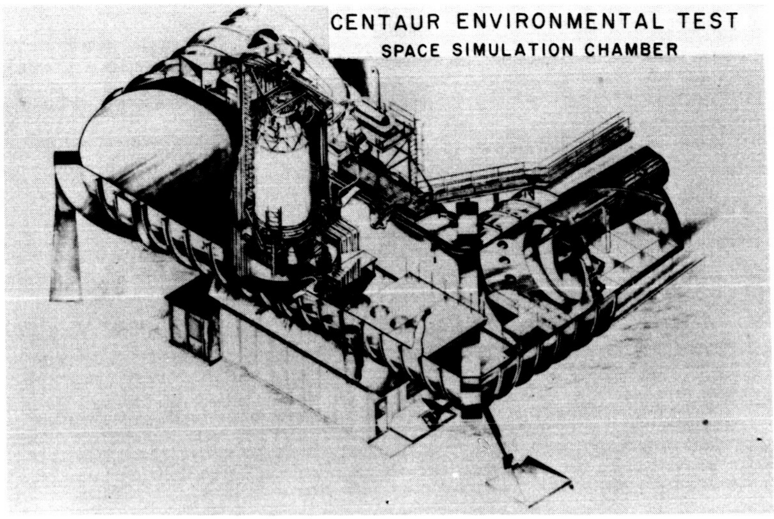

Figure 6

CENTAUR VEHICLE

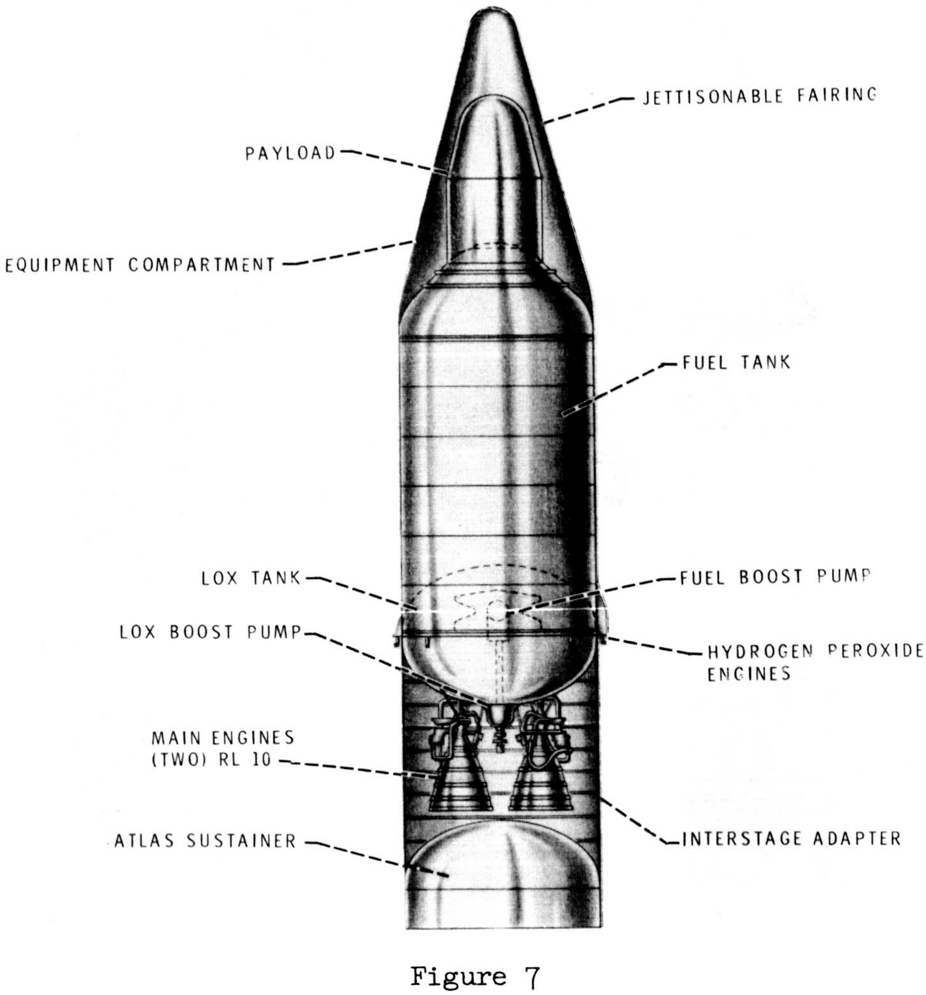

Figure 7

POSSIBLE CENTAUR COAST PHASE PROFILES

25 MINUTE COAST PERIOD

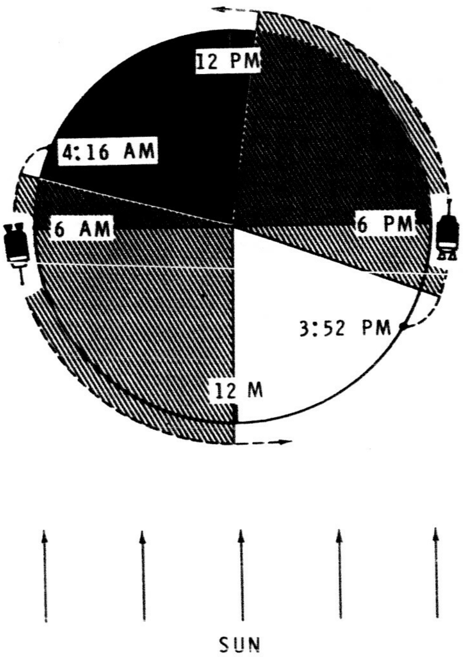

Figure 8

AC-4 FLIGHT PROFILE

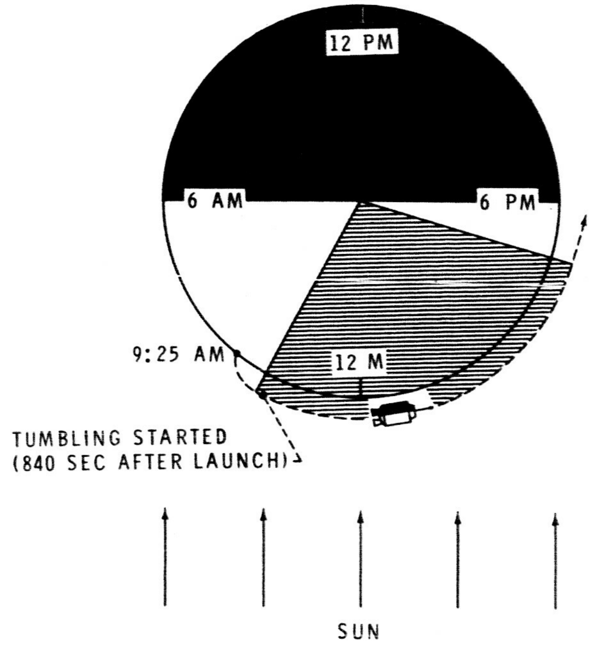

Figure 9

TEMPERATURE COMPARISON, RATE-GYRO-UNIT COVER TEMPERATURE

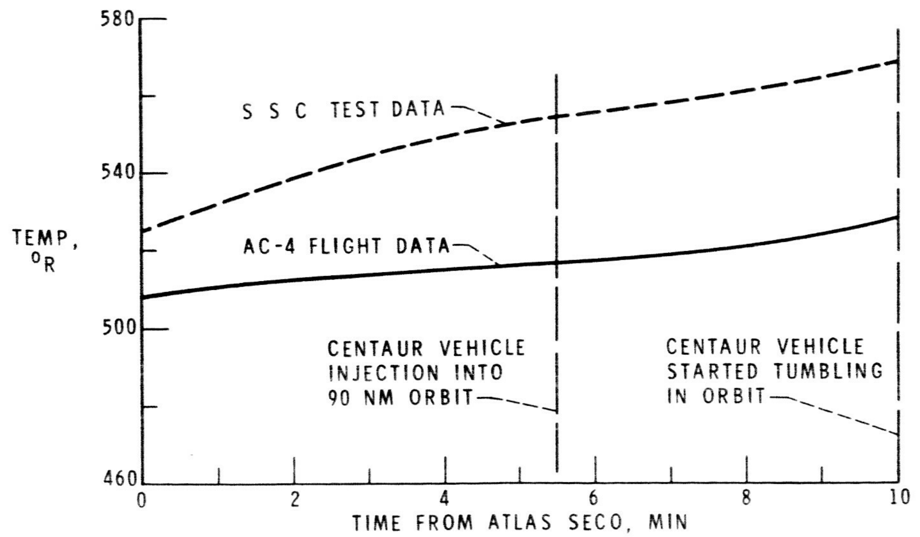

Figure 10

NOSE FAIRING INSTALLED IN SSC

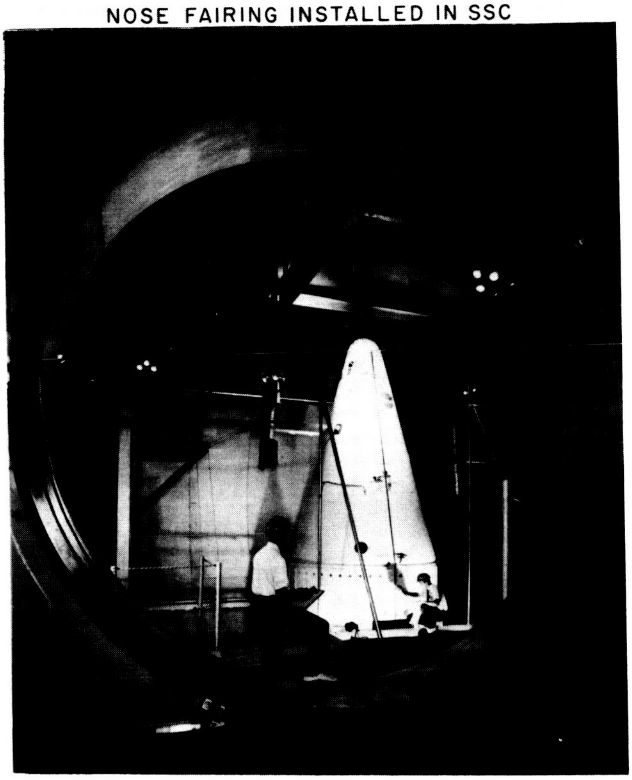

Figure 11

NOSE FAIRING JETTISON SETUP IN S S C

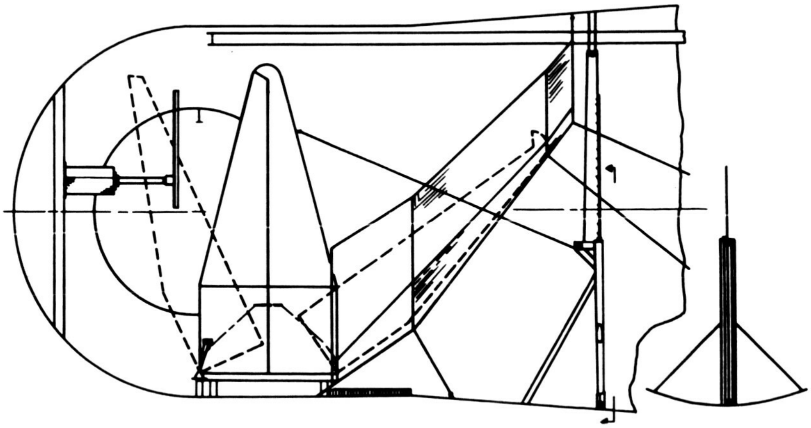

Figure 12

NOSE FAIRING AFTER JETTISON IN S S C

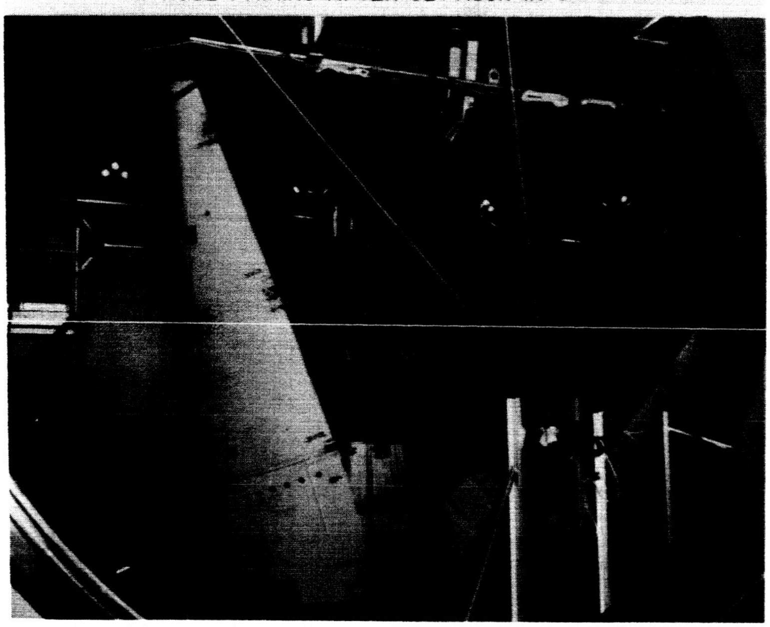

Figure 13

DAMAGED NOSE FAIRING AFTER FIRING IN VACUUM

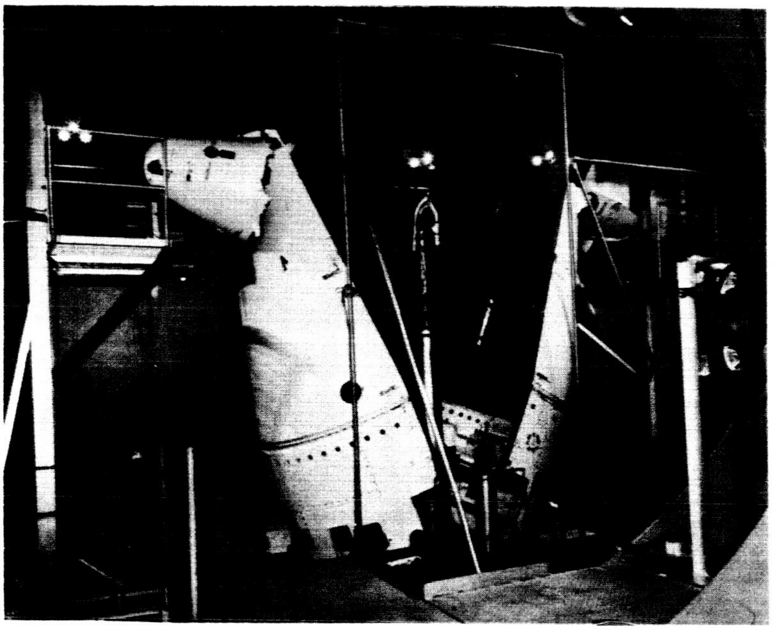

Figure 14

COMPARISON OF CENTAUR-SURVEYOR NOSE FAIRING TRAJECTORIES

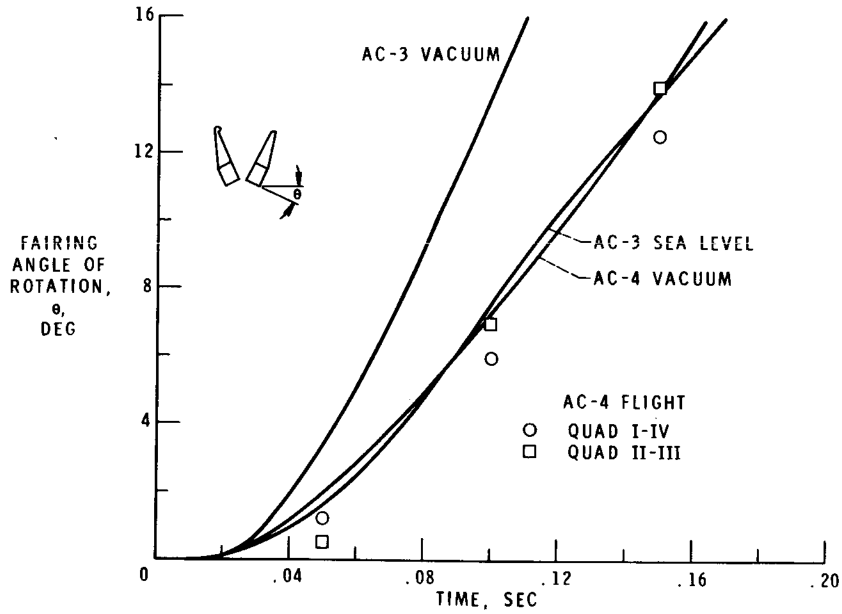

Figure 15

AC-4 NOSE-FAIRING SEPARATION, VERTICLE HINGE LOADS

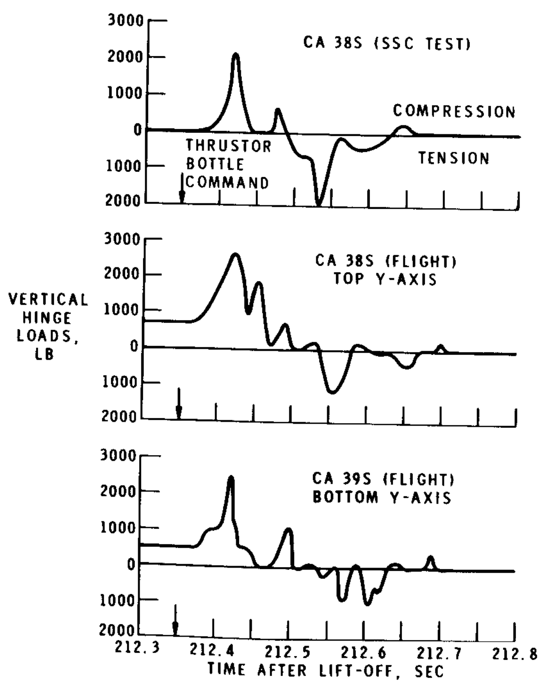

Figure 16