NASA CR 71205

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report No. 32-885

Comments on the Operation of the JPL 25-ft Space Simulator

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January 25, 1965

Prepared Under Contract No. NAS 7-100
National Aeronautics & Space Administration

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ABSTRACT

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The JPL 25-ft space simulator has been used continuously during 1963 and 1964 for tests of *Ranger, Mariner,* and *Surveyor* spacecraft. Comments are made concerning the importance of combined solar and space simulation to the JPL flight projects. Problems, which are peculiar to the operation of a large space simulator, are discussed relative to operating experience at JPL.

Author

I. INTRODUCTION

The difficulty of simulating space flight environments is now well established. On the other hand, many of the problems and deficiencies inherent in present space simulators have been reduced or circumvented by the experimenters and facility operators. Some of the significant problems and their solutions for the existing JPL 25-ft space simulator are commented on in this report.[1]

II. DESCRIPTION OF THE JPL 25-FT SPACE SIMULATOR

The JPL 25-ft space simulator tests spacecraft under the interplanetary conditions of extreme cold, high vacuum, and intense solar radiation. The principal uses of the facility are to determine spacecraft equilibrium temperatures and the capability of spacecraft systems to perform satisfactorily in simulated space environments. The physical arrangement of the facility is shown in Fig. 1.

The vacuum test chamber (Fig. 2) is a right circular cylinder 27 ft in diameter and 52 ft high. The top head of the chamber contains a 25-ft-diameter parabolic mirror. The bottom head, extending 5 ft below the floor level, contains numerous *feedthrough* or instrumentation ports for making electrical and mechanical connections to spacecraft and test equipment inside the vacuum chamber. The bottom of the chamber is also provided with a separately supported platform for mounting a vibration driver. Construction and mechanical details are described in Ref. 1.

A cylindrical solar dome caps the vacuum chamber, increasing the overall height of the simulator to 80 ft. Simulated solar radiation, originating in the solar dome, passes into the vacuum chamber through a 36-in. quartz

[1] Material in this report was presented at the International Symposium on Solar Radiation Simulation in Los Angeles, California, January 18–20, 1965, and is included in the proceedings for that meeting.

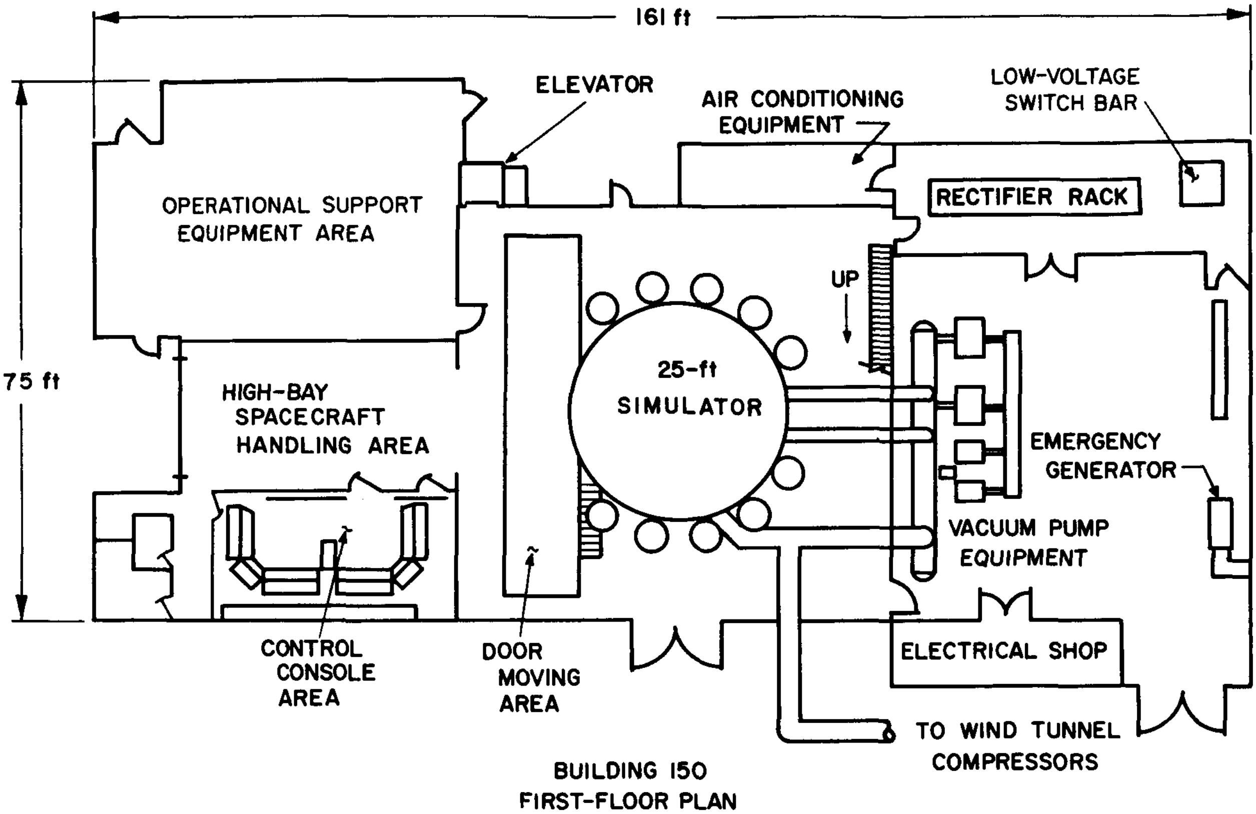


Fig. 1. Arrangement of the 25-ft space simulator area

lens mounted in a cylindrical well in the top of the chamber. The light reflects from a multifaceted reflector (virtual source) onto a cooled parabolic mirror which directs the radiation as an "off-axis" collimated beam into the test area. A servo-controlled iris is used to maintain a constant level of solar intensity and to cut off the solar beam for simulation of Earth shadow without turning off the compact arc lamps.

The heat sink of deep space is simulated by liquid nitrogen-filled black panels (shrouds) which line the walls and bottom of the chamber.

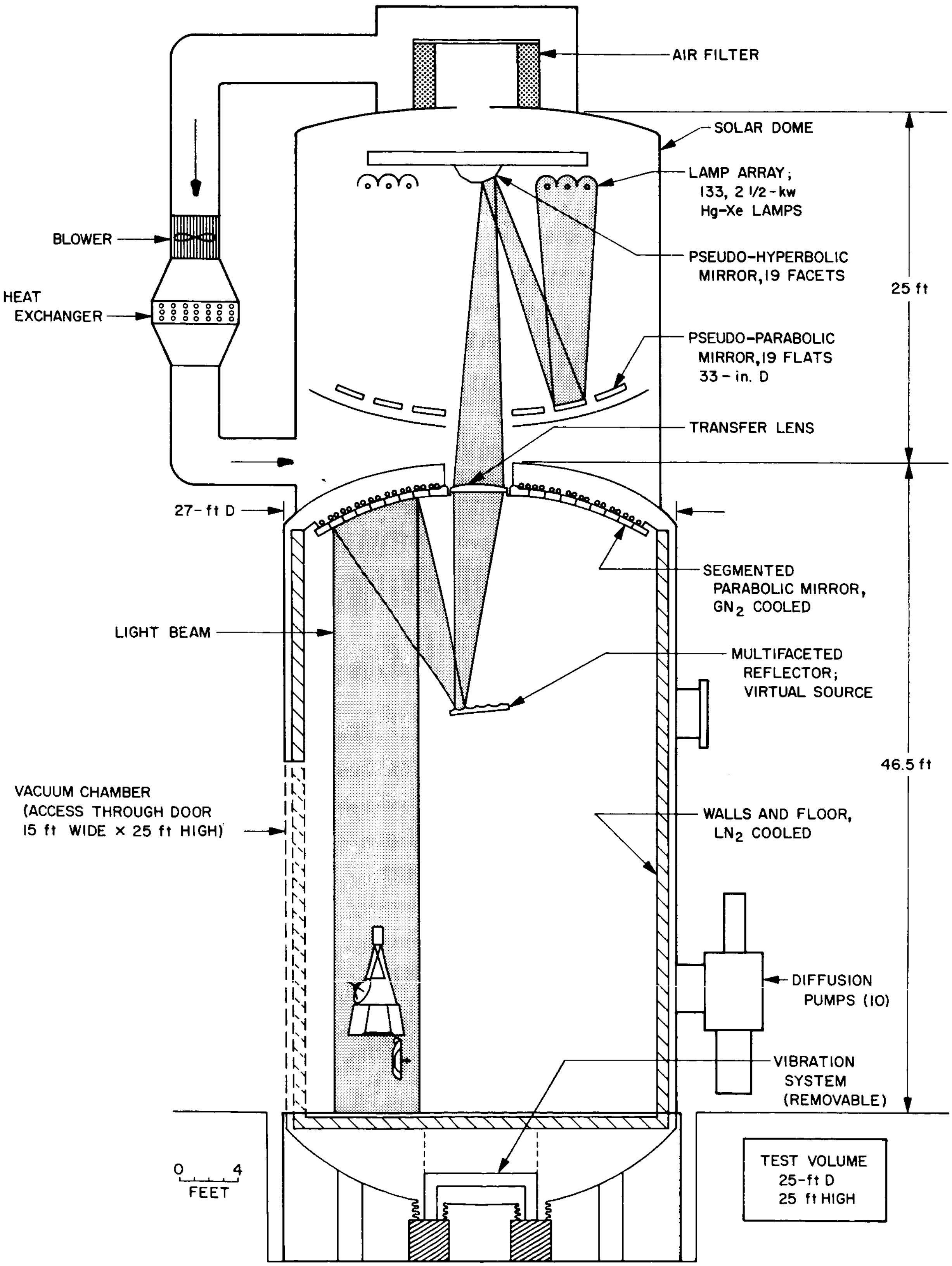


Fig. 2. Cross-section of the 25-ft space simulator

III. SPACECRAFT TESTING

A wide range of tests has been completed in the simulator for the *Ranger, Mariner,* and *Surveyor* flight projects during 1963 and 1964. A typical assortment for a given project usually includes tests of a solar panel, a temperature control model (TCM), a proof test model (PTM), and finally the flight spacecraft. Tests such as firing the midcourse motor on the *Mariner* Mars PTM and shaking the *Ranger* PTM during the launch pressure change have also been run and will be relatively common in the future. Some typical test installations are shown in Figs. 3 through 8.

To some extent JPL spacecraft have been designed to operate in the 25-ft space simulator as well as in space, since it is illogical to design and build a spacecraft which cannot be tested in available ground test facilities prior to flight. The high mission cost and limited launching opportunities make flight tests impractical for probes of the *Ranger* and *Mariner* type. In the case of *Surveyor* (where some "engineering" shots are planned prior to the "science" shots), it is still absolutely necessary to perform a thorough space simulation test program on the ground.

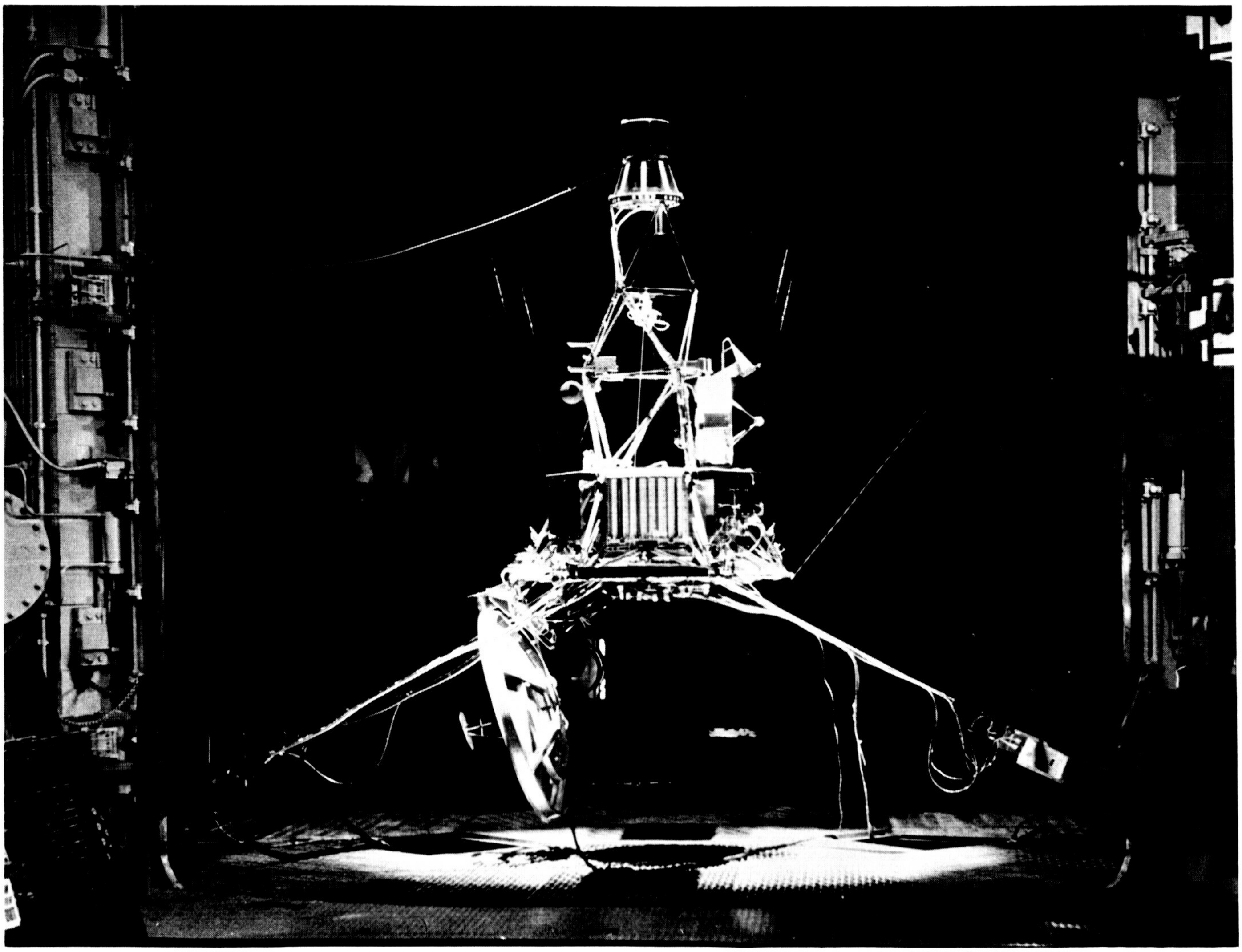


Fig. 3. *Mariner* Venus in the 25-ft space simulator

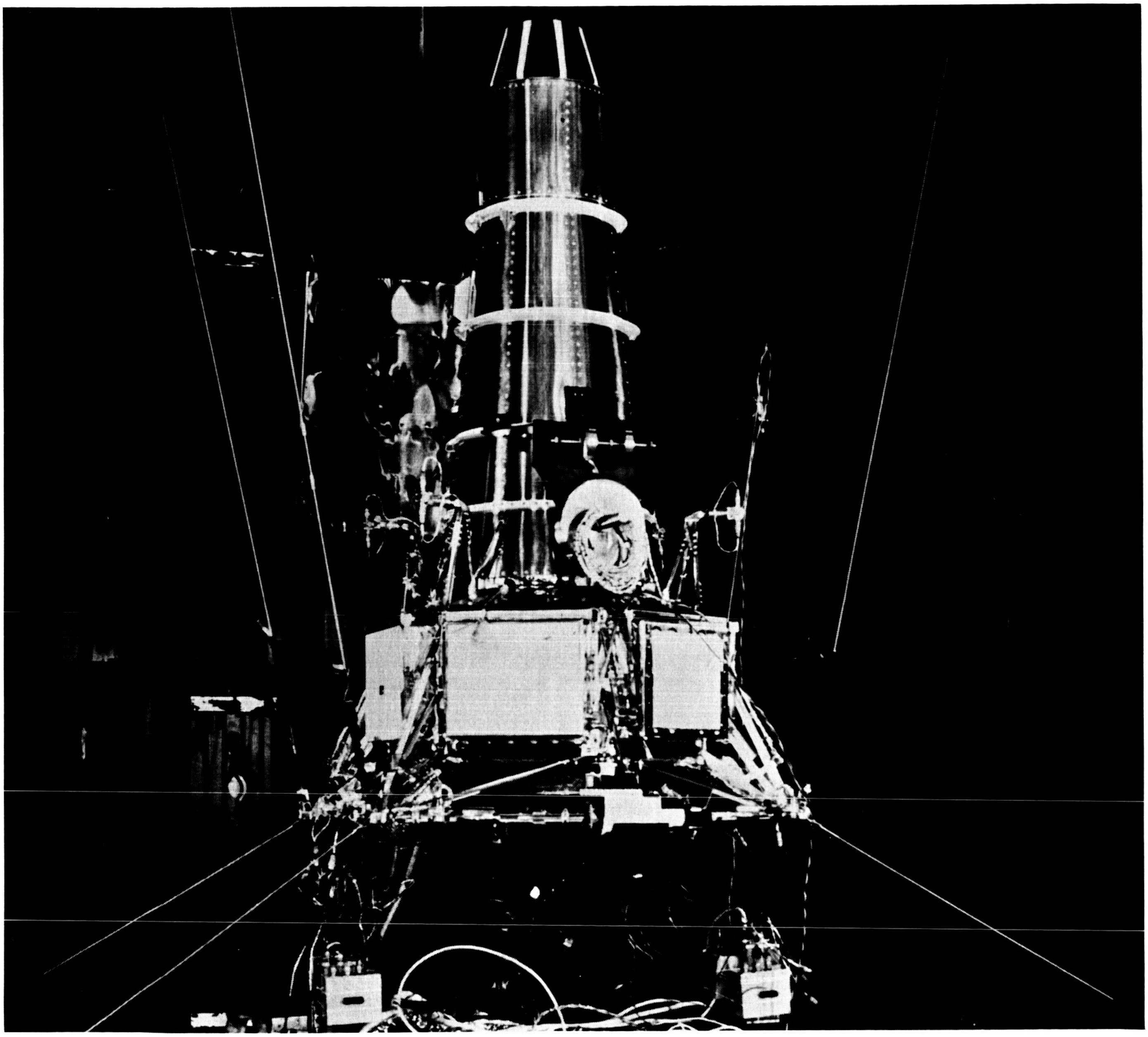


Fig. 4. *Ranger VII* in the 25-ft space simulator

Fig. 5. Solar mapping above *Mariner* Mars-4 in the 25-ft space simulator

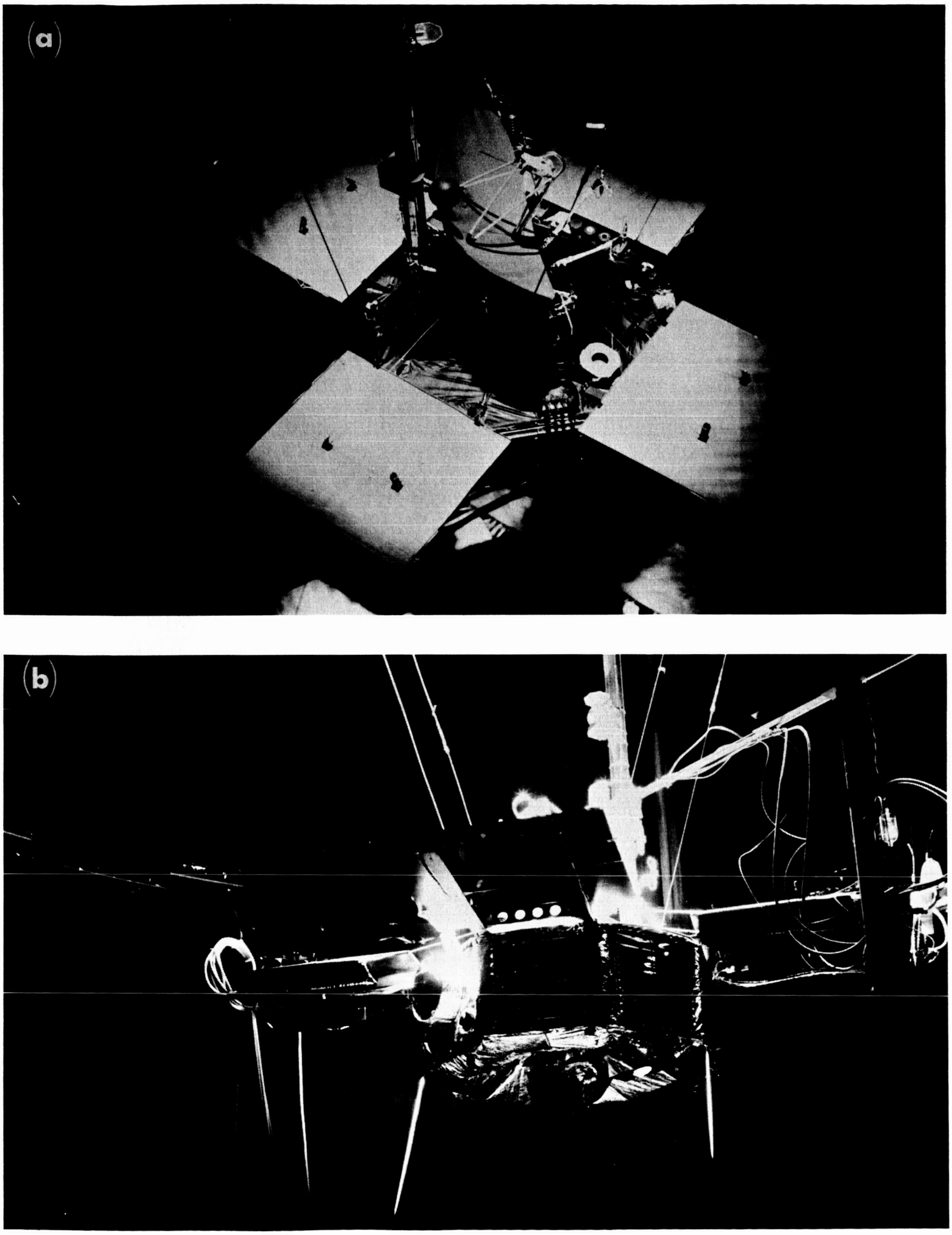


Fig. 6. *Mariner* **Mars-4 irradiated in the 25-ft space simulator: (a) view from the top and (b) view from below and to the side**

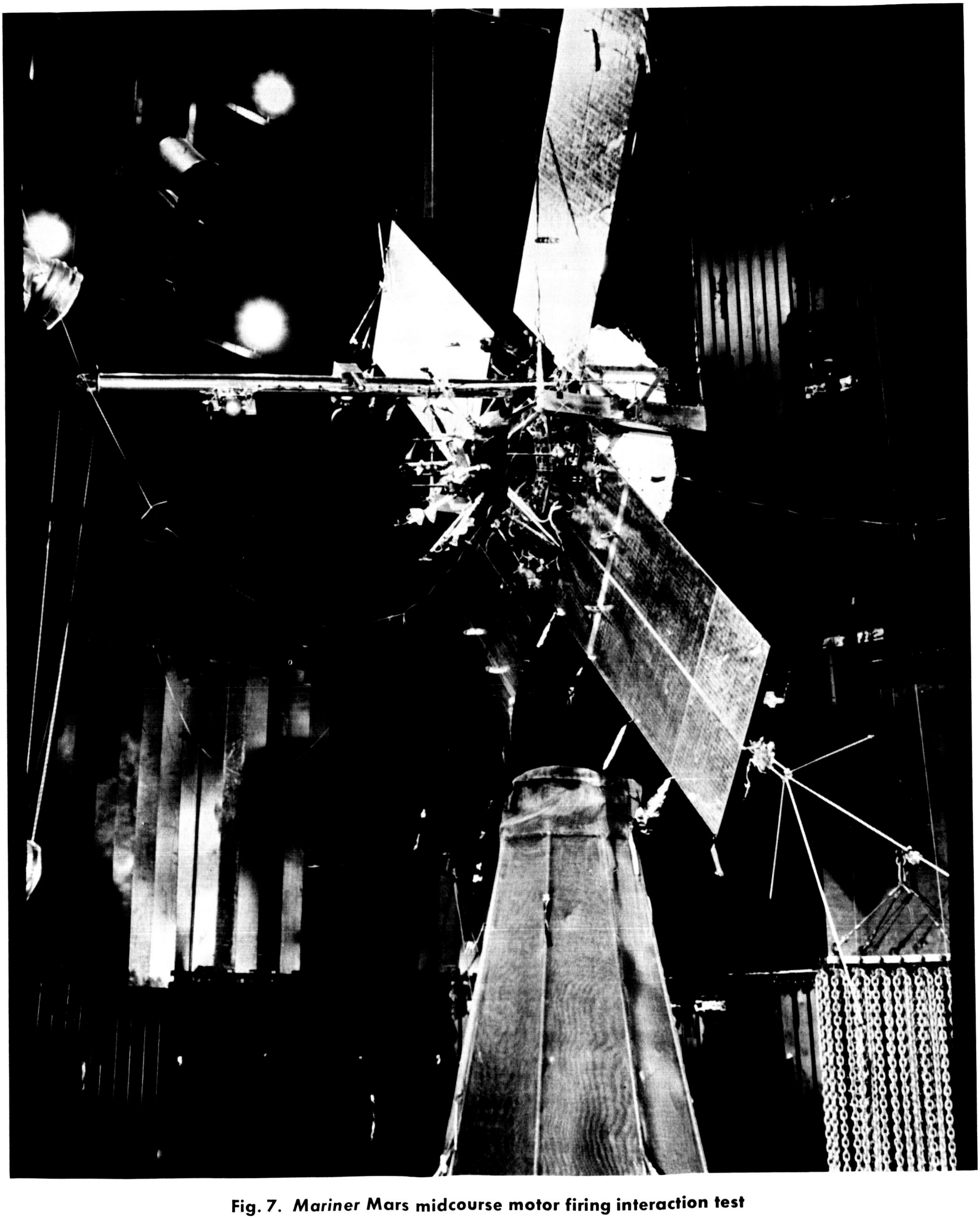
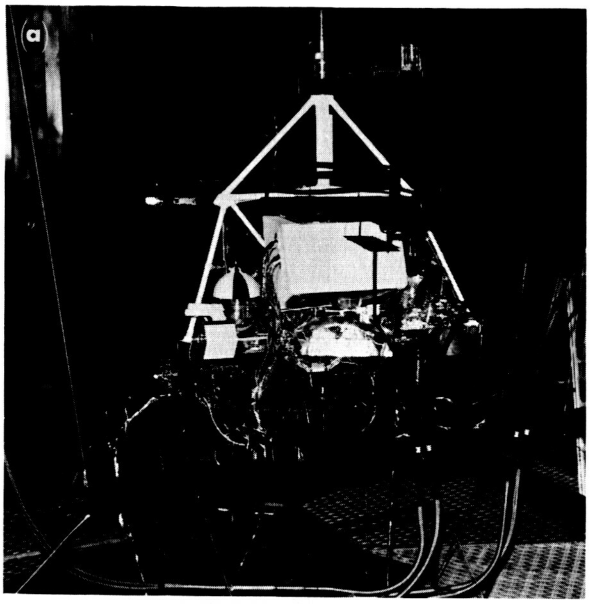


Fig. 7. *Mariner* Mars midcourse motor firing interaction test



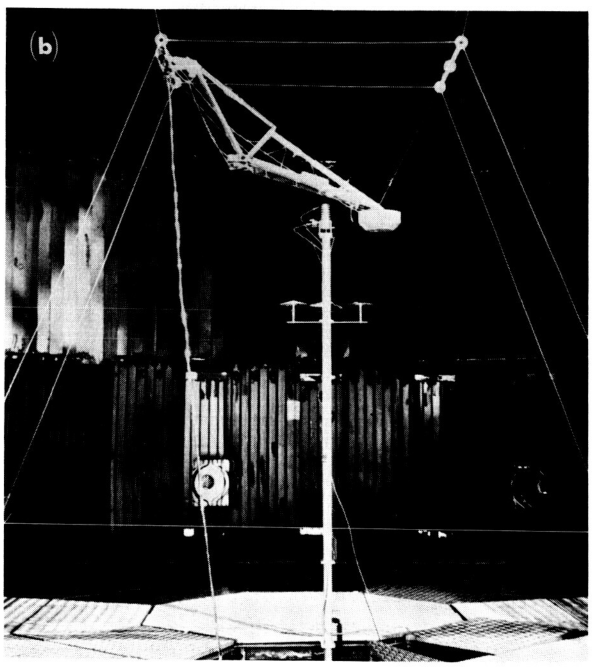


Fig. 8. *Surveyor* parts in the 25-ft space simulator: (a) Sector I and (b) landing leg

The space simulation phase of the ground test program is typically broken into two parts: tests to verify the adequacy of the temperature control system and mission tests to verify that the spacecraft electronic system will operate satisfactorily during a simulated mission.

The initial temperature control system design, which may be passive and/or active, is built into the TCM, which is a full-scale, thermal equivalent of the flight spacecraft. Heat loads, which are ordinarily generated by the spacecraft electronic components, etc., are simulated by electric heaters in the TCM. Extensive tests are then performed on the TCM in the space simulator. These tests may include, but are not limited to:

(1) Tests at expected solar intensity.

(2) Tests at higher than expected solar intensity.

(3) Tests at lower than expected solar intensity.

(4) Tests simulating a temporary Earth shadow condition.

(5) Tests simulating midcourse and/or terminal maneuvers.

(6) Tests with and without dummy solar panels to determine the infrared (IR) input to the bus from this source.

(7) Tests with internal heater settings set at higher and then at lower than expected energy levels.

Hopefully, the TCM testing is scheduled early enough in the spacecraft program so that resultant design changes can be conveniently incorporated into the PTM and flight spacecraft.

The PTM spacecraft is used for flight spacecraft design verification. This prototype of the final flight spacecraft is subjected to a variety of tests, many at more severe environmental conditions than would normally be expected in flight, in order to verify spacecraft design and performance. The primary objective of PTM testing is to determine the degree to which the assembled spacecraft meets the design criteria and restraints in both ambient and simulated flight environments. Secondary objectives include demonstrations of subsystem performance and compatibility and failure mode operation, determination of design safety margins, development and verification of flight procedures, and training of spacecraft operations personnel. In general, when PTM testing reveals the need for a design change, the change will be incorporated and tested on the PTM before incorporation into the flight spacecraft.

Finally, each flight spacecraft is tested to verify that its equipment operates normally relative to the design specifications. The testing is primarily concerned with qualification and uses the PTM test data as a standard against which to check flight spacecraft performance.

The particular objectives of the space simulator tests are to (1) establish the functional integrity of the spacecraft while operating in a simulated space environment; and (2) perform a final verification of the ability of the temperature control subsystem to maintain spacecraft temperatures within design limits and to verify that spacecraft performance in a simulated space environment is not affected by test cables.

The length of tests in the space simulator is a function of a spacecraft's mission and is limited by practical considerations. In the case of the *Ranger's* 66-hr flight to the Moon, it is practical to perform simulated mission tests in real time. On the other hand, it is presently impractical because of the long flight times to simulate entire planetary missions in real time. Consequently, only parts of a planetary mission such as the 8-mo *Mariner* flight to Mars are simulated. These simulated parts include launch, midcourse and terminal maneuvers, and cruise conditions at several solar intensities. Life tests of various components such as batteries are performed at the subsystem level for planetary missions.

IV. SIMULATOR OPERATING PROBLEMS

The operating problems, which have occurred with the JPL 25-ft space simulator, have generally resulted from an understandable lack of experience among the operating personnel and from the fact that space simulator testing required new techniques and combinations of equipment. In order to accomplish the testing required by the *Ranger*, *Mariner*, and *Surveyor* Projects, it has been necessary to staff the 25-ft simulator 24 hr/day and 7 days/wk. The minimum operating crew for each of three shifts consists of one operations engineer, one crew chief, one vacuum technician, one cryogenic technician, one solar technician, one scribe and relief man, two compressor plant technicians, and two electricians. Additional personnel are, of course, required to cover absences due to sickness and to limit each man's workweek to about 45 hr.

The preceding comments about working hours and personnel requirements have been made primarily as an introduction to the problems of personnel safety and training. Personnel safety and training cannot be overstressed in any discussion of space simulator operation, and the following brief list of hazards which come with most space simulators will sober the most casual facility operator.

Cryogenic

(1) Frostbite.
(2) Explosion.
(3) Anoxia.

Electrical

(1) Shock.
(2) Fire.

Solar

(1) Sunburn (eyes and skin).
(2) Ozone.
(3) Mercury vapor from compact arc lamps.
(4) Compact arc lamp explosion.

Vacuum

(1) Implosion.
(2) Low-pressure piping.
(3) Anoxia.

High-pressure nitrogen

(1) Explosion.
(2) Anoxia.

The safety problem is aggravated, and the training routine is complicated by the number and especially the variety of hazards and material involved in space simulator operation. For example, one must explain electrical hazards to solar system technicians, cryogenic hazards to electricians, and the ultraviolet or sunburn problem to everyone.

All JPL simulator operating personnel are now given regular physical checkups in order to detect the onset of mercury poisoning or other physical problems which would be aggravated by or impair a man's performance in his work around the simulator.

The test facility operator is usually required to start testing work immediately after completion of the facility construction contract. The time which was originally allotted to a thorough shakedown and calibration is usually shortened by construction delays, flight program changes, etc., to a barely tolerable minimum. In many cases the initial tests are far from ideal, but certainly better than no tests at all. The abbreviated shakedown and calibration period results in abbreviated operating experience and procedure shakedown and development prior to testing expensive and irreplaceable spacecraft. An unfortunate example of spacecraft damage occurred during a test in the 25-ft space simulator. The spacecraft was operating under simulated flight conditions when a mechanical vacuum pump stopped; this, by itself, was not serious, since there was some redundancy in the pumping system. However, while making an effort to return the pump to operation, some air at atmospheric pressure was inadvertently let into the vacuum system. This air caused the diffusion pumps to backstream briefly, and also temporarily raised the pressure in the simulator. The diffusion pump oil was removed from the spacecraft during a 5-day refurbishing period; however, the loss in test time resulted in only minimum tests being run on the electronic systems which replaced those damaged by arcing during the simulator pressure rise. As a consequence of this experience, equipment has been installed which will automatically turn off the spacecraft high voltage whenever the simulator pressure rises above a pre-set level. The troubleshooting procedure for the vacuum system has also been overhauled. Both of these operational improvements could very well have been generated in a longer shakedown and training period than actually occurred in the case of the 25-ft space simulator. In review, it is clear that, at the outset of operations, one should spend considerable time (e.g., 3 mo) training one's staff and developing operational procedures. This is being done in the case of the new JPL 10-ft space simulator which is now under construction.

Another problem which has been treated at JPL is that of contaminating the test item with dirt during a test in the space simulator. The "dirt" consists primarily of components outgassed from test cable insulation and test item potting compounds under simulated space conditions. This material will generally collect on the cold simulator wall, but may later be deposited on the test item if one is careless about his warm-up procedure. This sort of test-item contamination can be minimized by keeping the test item warmer than the simulator walls during warm-up. The warm-up process can then be "cleanly" accelerated by raising the simulator pressure slightly with dry nitrogen to a level where the mean free path of the molecules in the simulator is much less than its internal diameter. The nitrogen serves to protect the test item while the contaminants leave the wall and are pumped out of the simulator. In the 25-ft space simulator, pumping is continued at a simulator pressure of 10^{-3} torr until the wall or shroud temperature reaches 50°F. Then the pressure is raised to 400 torr with dry nitrogen, which tends to keep the remaining "wall-mounted" contaminants near the wall while the wall temperature is raised above the dew point temperature of the dry air used to eventually raise the pressure to one atmosphere. Four other steps which are taken to reduce contamination are:

(1) Pre-condition cables in a small washable vacuum tank.

(2) Suspend a liquid nitrogen-cooled plate between the simulator wall and test item optical systems, e.g., the *Ranger* TV system as in Fig. 4.

(3) Vacuum clean and bake out the simulator prior to a test.

(4) Eliminate or minimize the use of materials which outgas under test conditions.

The function of the vacuum conditioning is to eliminate some outgassing material, while the liquid nitrogen-cooled plate intercepts and traps contaminants in the area of the test item that it is to protect. Many of the contaminants which remain on the simulator walls after a test are removed just prior to the next test by evacuating the simulator to 10^{-6} torr for 24 hr, while the shroud is heated to 125°F. The 25-ft simulator is limited to this low bakeout temperature because the collimating mirror elements are epoxy replications which outgas at temperatures slightly above 125°F. The bakeout would obviously be more effective if it could be performed at a higher temperature.

The vacuum system, which can approximate a launch pressure profile and provides an ultimate vacuum level in the 10^{-6}- to 10^{-7}-torr range, has had no significant problems. It is worth noting that there has been no trouble with the large (15 ft $\times$ 25 ft) loading door in the simulator (Fig. 3). This door is sealed by a one-piece molded seal and, when coupled with an overhead crane, makes spacecraft test installations relatively simple and convenient relative to top or bottom-loading schemes which are initially less expensive.

Difficulties with the cryogenic system have been confined primarily to occasional leaks in the stainless steel to aluminum joints which are a part of the present system. The piping is stainless steel and the shroud panels are aluminum. Welded repairs to the aluminum panels are made difficult by scale inside the cooling tubes. Cryogenic leaks inside the simulator are found by evacuating this part of the cryogenic system and then noting where helium is sucked into the system. The helium leak detector is placed well downstream of the suspected leak.

The solar simulation system has been the largest source of trouble. The difficulties can be divided into two groups:

(1) Necessity to improve the "as-built" performance.

(2) Care and feeding of the present system.

The solar intensity level and uniformity of the 1962 as-built system were intolerable. The average intensity was about 60 w/ft^2, and there was a monstrous dip in intensity below the virtual source. The present off-axis arrangement shown in Fig. 2 was salvaged from this system by concentrating all of the light in a smaller beam and by eliminating the necessity to go through and around the virtual source. This rather economical modification and its performance are described in Ref. 2. Since the performance of the modified system was then generally acceptable by 1962 standards, JPL was able to concentrate on the other solar simulation problems.

The major complication in caring for a solar simulation system is that the components are somewhat unique and generally in limited supply with attendant long delivery times and often seemingly high cost. Most of the present components are not production line items.

The useful life of the 2.5-kw mercury-Xenon compact arc lamps has been extended from 200 hr to approximately 800 hr in the 25-ft simulator by attending to the following details:

(1) Elimination of reflections through the lamp envelope.

(2) Careful lamp cooling by modest forced convection.

(3) Regular inspection of lamps to find flaws in the envelope, etc.

(4) Care in handling, e.g., no fingerprints on the lamps.

Another technique for reducing lamp consumption was developed and used satisfactorily in mission tests of *Ranger* spacecraft. The spacecraft temperature distribution, which had been determined with solar simulation and the TCM and PTM, was duplicated on the flight spacecraft with electrical heaters. For certain tests such as a simulated mission test, the method of duplicating the previously determined temperature distribution is of no consequence to the spacecraft electronic system.

Lamps are arbitrarily removed from service after 800 hr because their original output has decreased by 30%, which results in insufficient reserve power for normal test purposes. Elimination of installed lamps which have developed flaws is particularly important because a lamp explosion usually damages the headlamp reflector; the falling debris then damages one or more of the 33-in. dia. turning mirrors (Fig. 2).

Because of the long time required to get replacement lamps and mirrors, a complete set of spares is stocked for most components in the solar simulation system. Regular maintenance and mirror refinishing permits operation with less than 100% spares for some components including the lamp power supplies and starters.

Solar instrumentation such as radiometers for measuring solar intensity and for use in collimation and spectrum measuring instruments has been another problem area. The most urgent problem here is the necessity to buy and/or develop a reliable calibration standard for in-house use. It is often impractical to return the instrument to its manufacturer for a check calibration. JPL is preparing to use a pyrheliometer as a secondary standard for calibration purposes in sunlight.

V. LACK OF IDEAL SPACE SIMULATION

The major shortcoming of present space simulation tests is the lack of ideal solar simulation. The ill effects of a large space simulator's cold black wall at $-320°F$ and vacuum level in the 10^{-6}-torr range have been negligible for all practical purposes in tests of JPL spacecraft which operate at room temperature level. It should be noted that, as a spacecraft component's operating temperature approaches the wall temperature, wall effects must be considered. The solar qualities which are of primary concern are intensity, spectrum, and collimation. Fortunately, some practical techniques have been developed which result in satisfactory test results in spite of nonideal solar simulation.

For example, the amount of tolerable nonuniformity in solar intensity is a function of the test item configuration. The variation of intensity incident on an aluminum box is effectively averaged by conduction through the aluminum. In a more complicated test item, this averaging may not occur and the experimenter may correct an artificial temperature problem which resulted from say a local cold or hot spot in the simulated solar flux. Such a correction to an artificial cold spot, for example, will then result in a hot spot during the actual space flight mission. This problem can be reduced by comparing the amplitude and frequency of the solar flux nonuniformities with the physical characteristics of the test item. Samples of the intensity distribution in the present JPL solar beam are shown in Fig. 9. In addition, the solar intensity is actually measured or mapped with a relatively small solar cell at a number of locations on the test item, as shown in Fig. 5. These measurements are required in the analysis of test item temperatures in which effects due to nonuniformities, reflections, and beam divergence must be considered.

The simplest way to eliminate ill effects because of a nonideal spectrum is to make all the sunlit surfaces black or nearly independent of wavelength. This was not possible in the thermal control system of the recent JPL spacecraft where such an approach would have overheated the spacecraft. Therefore, some sunlit area was made of polished metal or painted white in order to reflect unwanted solar energy. The amount of energy absorbed by the various spacecraft surface finishes is determined experimentally in the simulated sunlight of the space simulator under vacuum and cold-wall conditions. Surface samples are allowed to reach their equilibrium temperatures; then the solar simulator is turned off and internal heaters duplicate the equilibrium temperatures. The energy absorbed in the simulated sunlight can then be calculated and used in comparing the test item's performance in the space simulator with its performance in real flight.

In order to determine the thermal and electrical performance of a solar panel at an Earth intensity level in the 25-ft space simulator, it has been necessary to test at two intensity levels. This has been necessary because the spectrum of our present solar simulator is deficient (relative to actual sunlight) in energy at the wavelength which causes the solar cells to generate electricity. This approach has not been entirely satisfactory because it is necessary to almost overheat the solar panel during the electrical performance phase of the test and because the real interaction of the thermal performance on the electrical performance must be inferred.

Differences in collimation between real and simulated sunlight appear as differences in irradiation on shaded spacecraft surfaces during actual and simulated space flight. The maximum beam divergence of the present solar beam in the 25-ft space simulator is $\pm 5.3°$. Consequently, it has occasionally been necessary to alter Sun shades and other shadow producing parts of the spacecraft for tests in the space simulator.

It is worth noting that is has been possible to perform satisfactory temperature control and mission tests on some spacecraft whose overall projected area was larger than the cross-section of the simulated solar beam. In the case of *Ranger* and *Mariner*, the solar panels protruded beyond the edge of the solar beam, but their effect on the spacecraft bus was accurately simulated by stubby, electrically heated dummy panels.

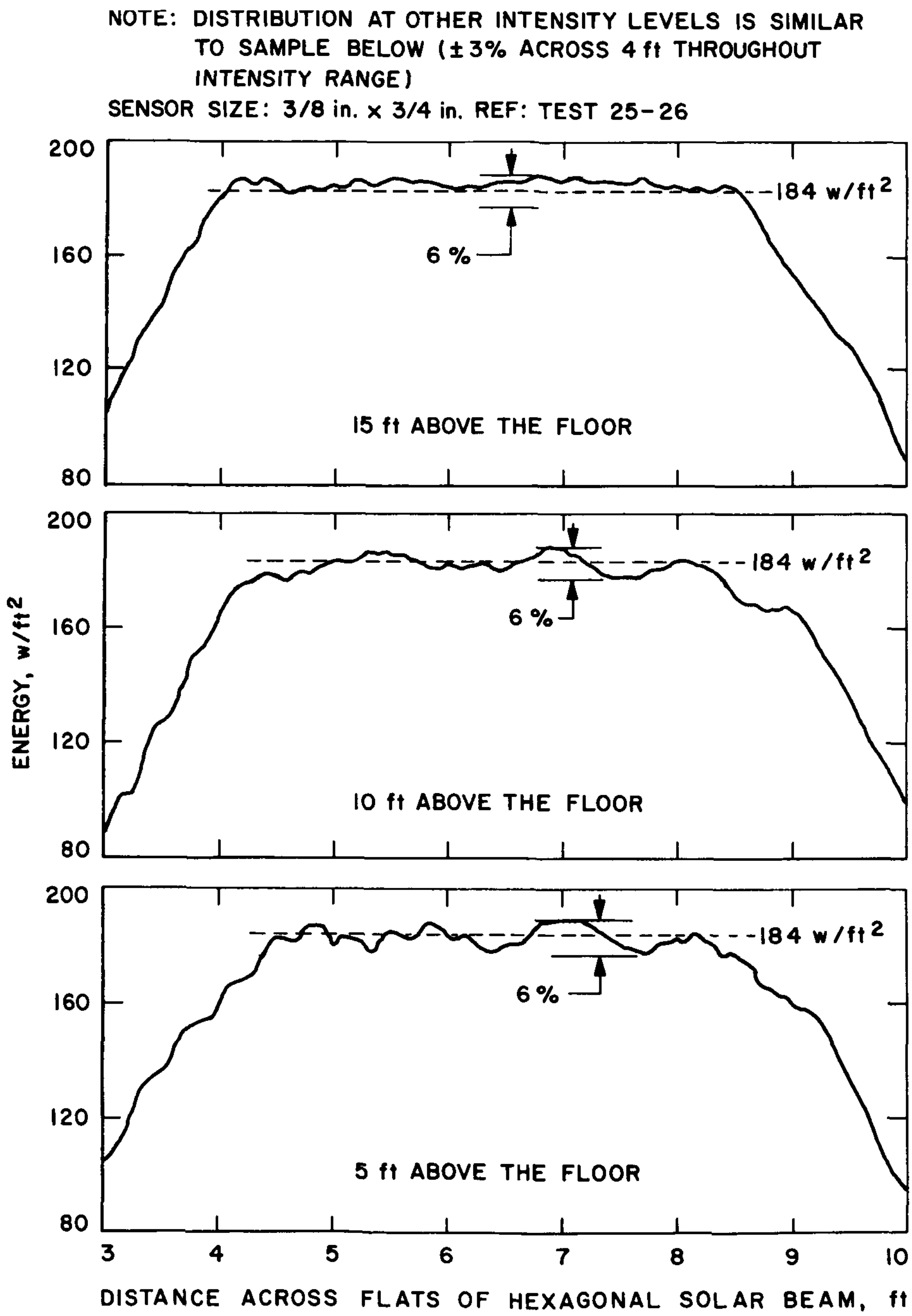


Fig. 9. Solar intensity distribution in the 25-ft space simulator

VI. CONCLUDING REMARKS

The significant value of testing spacecraft in simulated space environments has become obvious in the last few years. Since it is presently impractical to accurately calculate the temperature distribution for some of today's complicated spacecraft, ground testing in a simulated space environment which includes artificial sunlight is a necessity.

Properly interpreted spacecraft temperature data from space simulation tests compare satisfactorily with flight data as shown in Ref. 3. As time goes on and spacecraft become more complicated, it will become increasingly important to reduce the differences between ground tests and actual flight. The most significant differences are presently in the solar environment; but, fortunately, considerable progress is being made in this area as exemplified by the new JPL solar simulator, Type A (Ref. 4). This system is presently being installed in the new JPL 10-ft space simulator in which the following solar performance is expected:

Intensity: Venus level (270 w/ft^2).

Uniformity: $\pm 5\%$ throughout the test volume.

Beam divergence: $2°$ or less.

Spectrum: Xe and Hg–Xe compact arc lamps.

The problem of properly operating space simulation facilities can be greatly simplified by adequate attention to operating procedures, personnel training, and experimental techniques. These several items must not only be "talked about" but must be thoroughly practiced prior to spacecraft testing in the facilities. Then as test work continues, one must continually review and improve his procedures and training programs and develop new techniques in the light of his experience.

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