Test/Flight Thermal Data Comparisons for Viking and SCATHA

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Thermal flight data from two Viking Mars Landers and from the P78-2 (SCATHA) spacecraft tested at Martin Marietta Space Simulation Laboratory are compared with solar thermal vacuum test data. Various orbit/mission conditions are evaluated by the data comparisons and reported. Vehicle performance as well as payload/mission performance as predicted by solar thermal vacuum test are discussed. Fidelity of space simulation capabilities are reported as a conclusion of the data comparisons. Incorporation of space vehicle engineering and testing into overall vehicle mission for future programs is mentioned.

Introduction

THREE space vehicles, two of which were identical and totally different from the third, were built, tested, and subsequently successfully launched. The two identical vehicles were the Mars Landers of the Viking program and the third was the Space Charging At High Altitudes (SCATHA) Earth-orbit satellite. The mission as well as the tests for the two programs were entirely different and constitute a wide scope in test philosophy and flight configurations. For this reason, they are chosen for comparison of flight and test thermal data to be reported in this paper.

Viking Program

The primary mission of the Viking program was to place two identical Viking spacecraft, each consisting of an Orbiter and a sterilized Lander capsule into the Martian environment (see, for example, Ref. 1). Launch was in mid-1975 with Mars orbit insertion and landing in mid-1976. The major mission events for the program are shown in Fig. 1.

Due to the number of different configurations required by the mission, a completely comprehensive test program would be too costly, too long, and a simulation of the Martian environment especially in the landed configuration would not be guaranteed. The following discussion highlights the general thermal test sequence arrived at, with a brief summary of the philosophy used in its derivation. This development is discussed in detail in Ref. 2.

Viking Thermal Test Philosophy

Due to the above-mentioned (and ever-present) constraints, a test philosophy, which allowed simultaneous testing of a model with fabrication of a qualification unit, was necessary. This required primarily the development of a thermal model, which could be verified and adjusted by the testing of a simulator and whose parameters allowed for a broad range of values and initial conditions. This model would then be used to predict vehicle thermal behavior as the initial conditions became available during the mission.

The simulator which was the object of the development test, would verify the Lander thermal control system and verify the applicability of the proposed techniques of Martian environment simulation to the thermal vacuum chamber.

The qualification unit, which included primary flight-type hardware, was then extensively tested. As per normal test procedure, extensive testing of the actual flight articles was not necessary.

The science team who would operate the payloads of the Lander by ground command received extensive training to thoroughly acquaint them with Lander operations.

Test Constraints

Even in simulation of actual mission environments, there was a good deal of uncertainty. The environment on the surface of Mars was simulated but since the flight articles themselves were not tested in the landed configuration, no comparisons of landed flight data are presented in this paper.

Figure 2 shows an overview of the Viking Lander Capsule (VLC) test program adopted. Only where the symbol SIM appears were mission conditions actually simulated.

Mathematical thermal modeling was verified and developed mostly by testing the simulator and only near the end of the program by actual flight articles. It was also impossible to completely construct the various configurations for entry into the Martian environment for the flight articles. The complexity of necessary mission configurations can be appreciated by looking at Fig. 2.

During the 10-11 month cruise phase of the mission, the VLC would be attached to the Orbiter and experience very little activity other than some routine checkout. After Martian orbit was achieved, the landing capsule would be separated, at which time its thermal environment would change drastically.

The long thermal time constant required by the cruise phase would prevent precise testing of the thermal response after entry since the whole entry-to-touchdown time would be only about 10 min. This fact required an accurate yet "open-ended" thermal math model to predict the thermal behavior or at least bound it acceptably once actual conditions became available on mission. As this model was developed and tested by the simulator and by the qualification unit in the solar thermal vacuum chamber (STVC), there was confidence in its scope and fidelity to predict VLC thermal behavior for almost any given set of conditions by the time the testing of the flight articles was at hand.

For futher discussion of Viking thermal vacuum testing and the associated facilities, see Refs. 3-12.

Factors Affecting Validity of Comparison of STVC and Flight Data

The only configuration in which the flight articles were tested was the cruise mode. Strictly speaking, this would be the only phase in which valid flight and test data comparisons could be made. This would include the mission phases of interplanetary cruise, Martian orbit, and all preseparation sequences. Since, however, the test was a mathematical model

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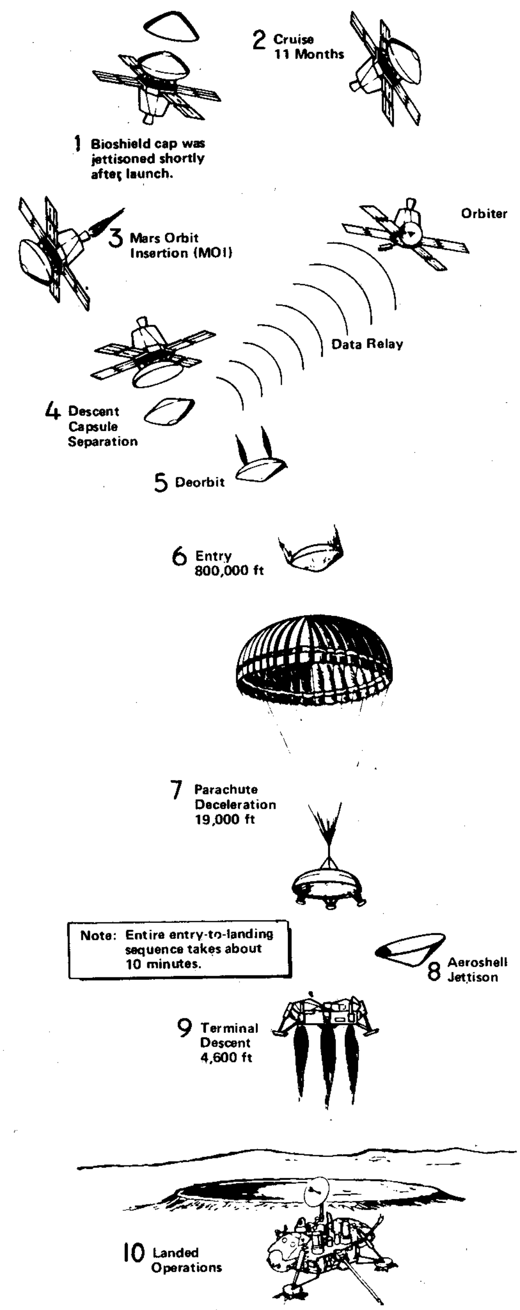

Fig. 1 VLC mission sequence.

checkpoint as well as an acceptance test, valid comparisons may also be made in the separation through touchdown phases if the applicable discrepancies between the STVC and mission environments are properly accounted for.

These discrepancies are basically fourfold:

1) The STVC test was conducted with the VLC still attached to the Orbiter thermal effects simulator (OTES), while the VLC was put through its separation through landing sequence with respect to internal power consumption. Therefore, the bioshield base would act as additional heat shielding and cause the VLC temperature to run slightly higher in the STVC.

2) The STVC test included no Martian atmosphere which would have a net effect of heating the aeroshell upon entry, and convective cooling to the Lander after the aeroshell was jettisoned. This effect would be small on mission, however, since entry to touchdown is only about 10 min and most of the atmospheric impact is experienced by the aeroshell, which shields the Lander, itself having a large thermal response time.

3) The internal power consumption could not be totally predicted after separation. Since the uhf radio tended to run hot, power was reduced on mission compared to STVC. Also the test required the PCDA (power conditioning and distribution assembly) to be on during the entire STVC, whereas on mission it was on only about one fourth of the time. This would tend to make the test temperatures run slightly hotter than on mission.

Finally, the precise initial conditions going into separation could not be known until mission. These were fed into the math model on mission and all systems were "separation go" for both VLCs during the mission. The smaller power consumption on mission than in test was a major difference in the initial thermal conditions for separation.

Data Comparisons

The only portion of the thermal vacuum test that could actually be called a thermal balance test for the flight articles was the cruise test. This test was run for 3.5 days for VLC-1 and 4.5 days for VLC-2. Figure 3 shows the cruise test equipment plate cooldown profile vs time compared with the flight telemetry data and the model (LTEMP) predictions for VLC-1 adjusted upwards by 2°F (1.1°C). The locations of the sensors are shown in Fig. 4. The profile shown in Fig. 3 is the average temperature of compartments 1, 2, 4, 5, and 6. Since all through the interplanetary cruise VLC-2 was 2°F (1.1°C) hotter than VLC-1, the model was only applied to VLC-1 in the data, and the result simply increased by 2°F (1.1°C) for VLC-2. The agreement between flight and test data is very good. Both Landers had highly repeatable internal equipment plate temperature for the same power configurations even after more than 9 months.

Figures 5-8 are plots of the uhf radio interface and battery interface temperatures for the preseparation through touchdown phases of the mission for each of the Landers. These are compartments 1 and 2, respectively, in Fig. 4. The thermat vacuum test data in each of the plots are from test thermocouples rather than flight telemetry instrumentation. In each of the plots the test data and post test analysis are represented by a solid line, flight data by dots, and LTEMP on mission predictions by the dashed line. In the case of VLC-2, the on-mission LTEMP predictions are again simply the VLC-1 values increased by 2°F (1.1°C). Each of the figures has ten time markers labeled A-J, respectively. These denote the following portions of the mission:

A-B Preseparation checkout
B-C Battery charge
D-E Sep – 9½ h update
F-G Sep – 3½ h update
G Separation (*NOTE:* Flight data to the left of this point are strictly comparable with test data.)
G-H Deorbit burn
H-I Coast
I-J Entry into Martian atmosphere
J Touchdown

The flight data are in very good agreement with the thermal vacuum test data. Since as little power as possible was used in this phase of the mission due to concern about the uhf radio overheating, flight data are slightly lower than test temperatures. Also, for VLC-1 the test started at 2°F (1.1°C) cooler (ambient in lab was cooler that day) than for VLC-2, and this is reflected in the data. Strictly speaking, test and flight data are comparable only up to separation, or point G. After that the discrepancies mentioned earlier are in force, each of which tends to make test data hotter than flight data.

Upon entry to Martian atmosphere (800,000 ft), some heating was experienced by the aeroshell, and when the aeroshell was jettisoned at about 4800 ft the Landers experienced a small amount of convective cooling.

The test data last slightly longer because originally it was thought to have about a 4 h elapsed time from separation to touchdown, but concern about overheating the uhf radio

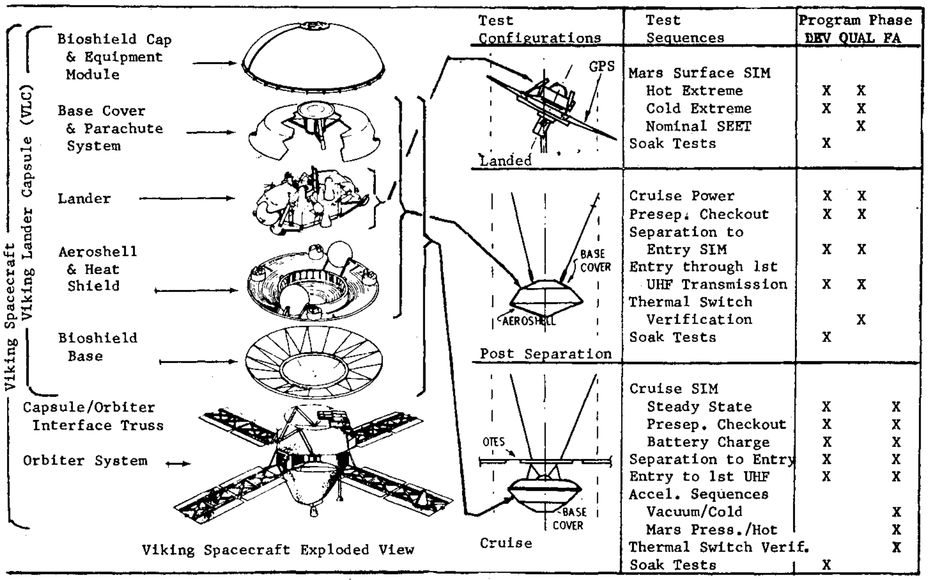

Fig. 2 VLC thermal vacuum test program overview.

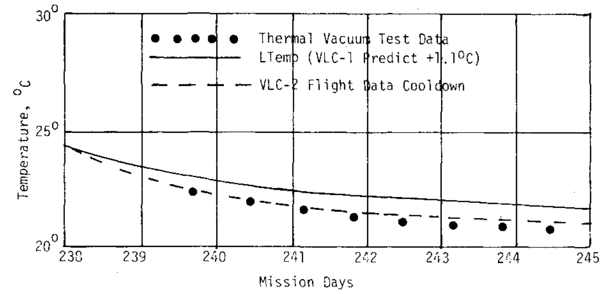

Fig. 3 VLC-2 average equipment plate cooldown profile.

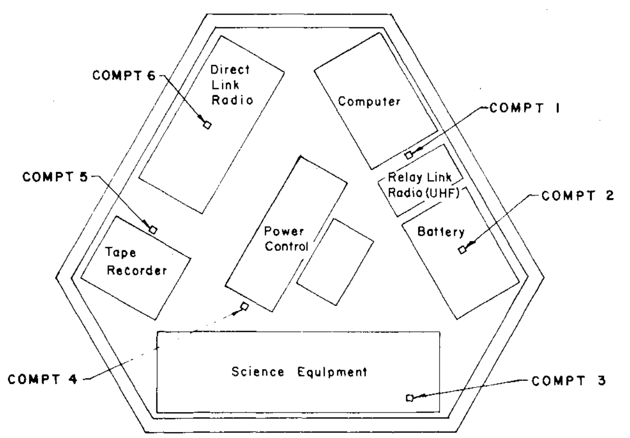

Fig. 4 Lander interior equipment plate temperature sensor locations.

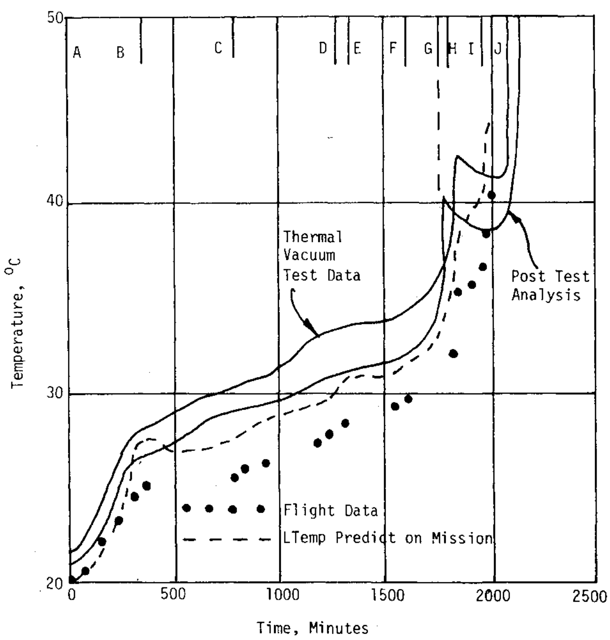

Fig. 5 Temperature profile from preseparation checkout to touchdown for uhf radio interface on VLC-1.

caused this time to be reduced by almost an hour.

As would be expected, there is less of a pronounced difference between the preseparation checkout and battery charge phase as sensed by the battery interface than as sensed by the radio interface. Also, the uhf radio, which is used during the terminal descent and upon landing, gets hotter than the batteries at this latter portion of the descent mission.

SCATHA Program

The P78-2 (SCATHA) program was conceived to investigate certain space charging and discharging phenomena observed in the early 1970s by numerous satellites in near-synchronous Earth orbit. The mission of the spacecraft is fully described in Ref. 13. The program was unique in that the vehicle was basically a one-of-a-kind "build it and fly it" item. The vehicle was successfully launched in January 1979 for a planned mission of one year. It still continues to operate nominally in the middle of this, its third year in orbit.

The vehicle is spin stabilized and placed in a near equatorial, near synchronous (period 23.5 h) orbit. Its orientation is such that its spin axis is parallel to its velocity vector, and thus its orientation with respect to the sun changes by $360/365 = 0.986$ deg per day. The vehicle supports 12 scientific payloads (experiments) which measure, monitor, and calibrate the various parameters associated with the charging events under investigation. These experiments were numbered SC-1-SC-11 and ML12. The experiments as well as a quality of data and analysis are found in Refs. 14-22.

The body of the spacecraft was cylindrical with the addition of five booms, two long rectractable tape antennas, and a few smaller instrument and hardware protrusions. Figure 9 shows the space vehicle.

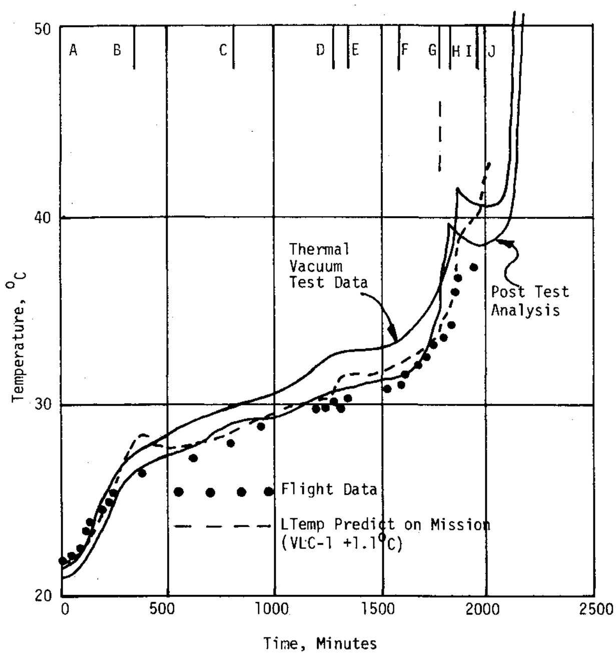

Fig. 6 Temperature profile from preseparation checkout to touchdown for uhf radio interface on VLC-2.

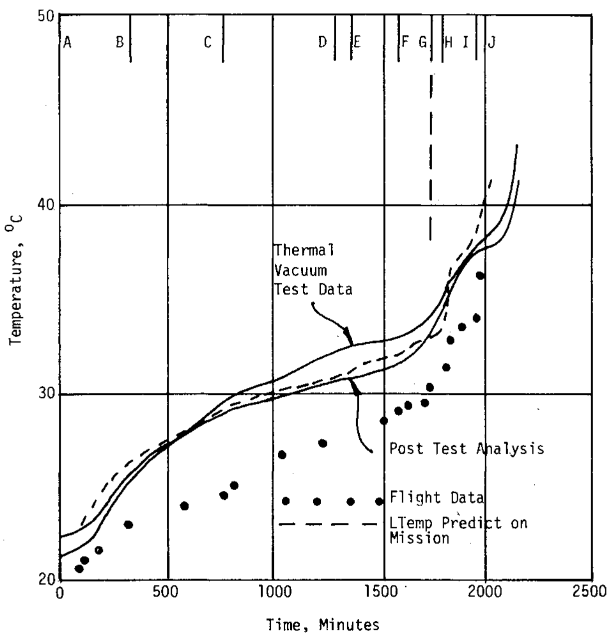

Fig. 7 Temperature profile from preseparation checkout to touchdown for battery C/D interface on VLC-1.

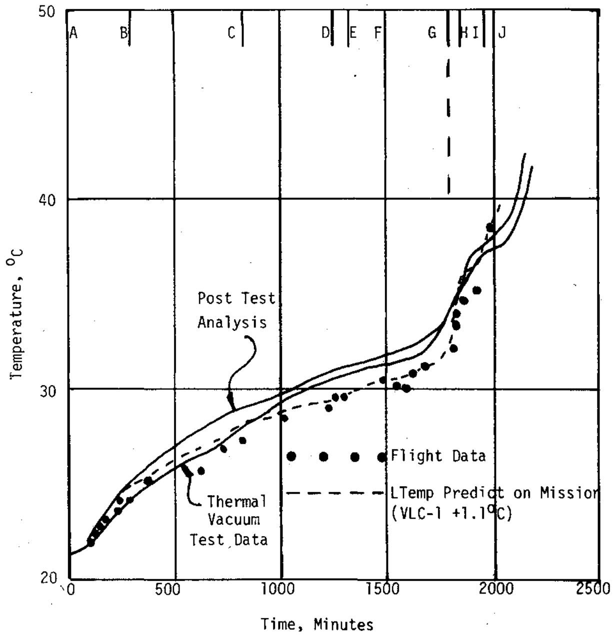

Fig. 8 Temperature profile from preseparation checkout to touchdown for battery C/D interface on VLC-2.

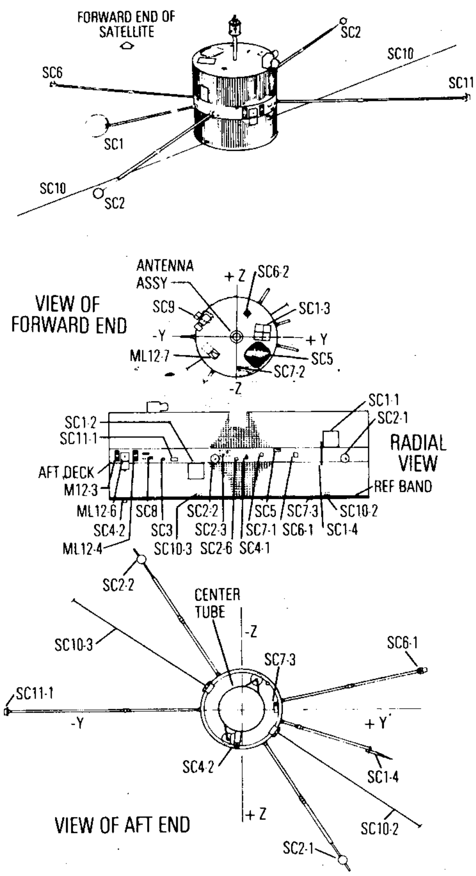

Fig. 9 SCATHA space vehicle.

SCATHA Test Program

The test program for SCATHA was unique for the same reason as mentioned previously; namely, that no qualification unit or test models were built. The thermal vacuum test was both the qualification and the flight acceptance test. This fact required extra caution because the testing always involved flight hardware. At the same time, since the program was much smaller than the Viking program, and since only flight hardware was monitored, it is much easier to compare the data from test with flight data.

Much of the thermal vacuum test program for SCATHA involved thermal balance testing. This subjected the vehicle to actual on-orbit environments for sufficient time for all portions of the vehicle to approach thermal equilibrium. Such a test validates modeling and demonstrates the performance of the onboard thermal control system. Two special cases were also run in the thermal vacuum test; the transfer orbit simulation which was a thermal balance test, and eclipse events where the "sun" was removed for longer than the longest on-orbit eclipse duration.

Test Requirements

The test profile for the program was determined by several factors. First, to draw out marginal equipment design, certain

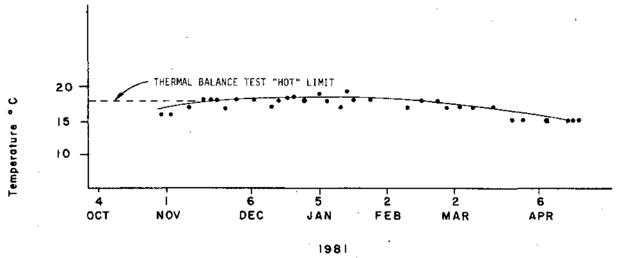

Fig. 10 Seasonal trend for SC2-3BT.

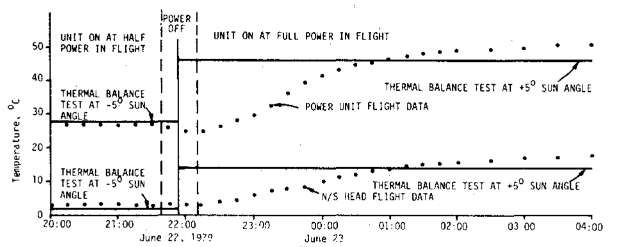

Fig. 11 Temperature profile for power unit and N/S head of SC9 going through precession maneuver.

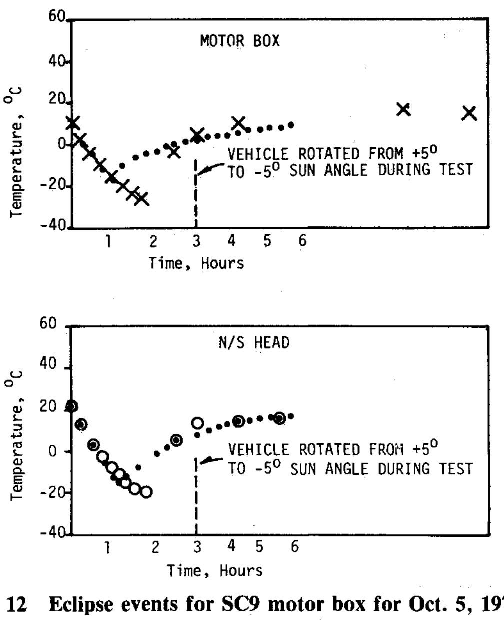

Fig. 12 Eclipse events for SC9 motor box for Oct. 5, 1979 (above) and for N/S head for March 28, 1979 (below). The motor box eclipse duration was 1:07:13 on flight and the N/S head eclipse was 1:00:18 on flight. The eclipse in test was 1:42:00. Flight data are dots, and test data are X's and circles.

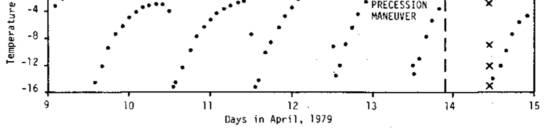

Fig. 13 Thermal profile of consecutive eclipse events for SC11-1 in April of 1979, going through a precession maneuver.

worst cases were forced. These included primarily an eclipse condition followed by a 1.25 solar constant forced hot cycle. The variation in solar intensity due to the eccentricity of the Earth's orbit was simulated by 0.96 solar constants for the "cold" case and 1.04 solar constants for the "hot" case. The mission plan was to keep the vehicle within ±5 deg of normal sun angle by periodic precession maneuvers. Therefore, the vehicle was tested at solar aspect angles of ±5 deg to simulate these worst cases. Also, to insure that no components would go out of thermal specifications during the transfer orbit, a nominal +45 deg solar aspect angle was chosen to simulate this phase of the mission. The sun angle was varied by rotating the tilt axis of a two-axis gimbal, while the azimuthal axis of the gimbal was rotated at approximately 1 rpm to provide axial spin for the vehicle. The solar aspect angle or "sun angle" is the angle between the normal to the vehicle axis (the spin axis) and the sun. This angle is defined to be positive if the sun is forward of the normal, and negative if the sun is aft of the normal.

Validity of Comparison of Test/Flight Data

Since so much of the test program was designed to simulate actual on-orbit environment, the test data may be compared to the flight data with very little discrepancy. Most discrepancy arises from multiple power dissipation levels of the various payloads, some of which may not have been chosen in test, and in one case, a boom was shortened in order to fit the vehicle into the test volume with proper clearance for reorientation during test.

Data Comparisons

Many space vehicle missions do not include the downlinking of extensive engineering data. Some only downlink such data if there are scientific data to be transmitted. Of the flight data available, comparison with test values for thermal balance test is not best displayed graphically, but rather stated. Comparisons of flight and test data showing seasonal trend, solar aspect angle dependence, eclipse events, or other events of interest are presented graphically. In many cases, flight data were only available at particular points rather than with sufficient frequency in time to demonstrate trends. In these cases the data comparisons are simply stated and discussed, if such discussion is warranted.

Seasonal Hot and Cold Limits

Data from various portions of the SC2 payload demonstrate hot and cold limit thermal responses for the vehicle. This payload measured spacecraft electrical sheath fields and had an energetic proton detector as well. Sensors were located at various positions on the spacecraft as shown in Fig. 9. SC2-6 flight data range from −7°C for the cold case to 0°C for the hot case. Corresponding test values are −6°C and 0°C, respectively. SC2-4 flight data indicate +37-+39°C while balance test values range from 34-39°C, respectively. SC2-3BT thermal response is discussed in the next section. The agreement for thermal balance hot and cold limits between flight and test is very good.

Seasonal Trend

Three payloads provided data at sufficient frequency to establish a seasonal temperature trend. The first, SC2-3BT, which is the bracket temperature for experiment number 3 of payload 2, is plotted in Fig. 10. Data here represent the vehicle leaving the "cold" and going through the "hot" portion of the Earth's orbit around the sun. The hot case thermal balance test temperature of +18°C is also plotted. The agreement is very good. This test portion was run with a simulated solar intensity of 1.04 solar constants.

The second payload, SC11-1, was a magnetic sensor located on a boom 4 m from the vehicle, which makes it particularly sensitive to small solar intensity changes. The flight values varied from −2-+4°C going from cold to hot seasons. The test values were somewhat higher, +5 and +12°C, respectively, due to the fact that in test, only a 4-ft "stub" boom was used in order to allow the vehicle to be rotated and tilted in the chamber, and this closer distance increased the radiative thermal coupling between the payload and the vehicle. Also, the solar aspect angles varied during the test, which would tend to bring the 12°C value down to +8°C to be comparable.

A third payload, SC-3, a high energy particle spectrometer, also logged a trend from 13-18°C. The test values were 23°C and 18°C, respectively. It is not known why the discrepancy is of such a magnitude, although the difference of 5°C is the same for both.

Solar Aspect Variation Thermal Response Limits

As mentioned earlier, the vehicle naturally "degraded" slightly less than 1 deg per day in solar aspect angle. The orbit was such that the degradation would move the aft end of the vehicle into the sun and the forward end out of the sun. Several payloads whose thermal response favored one end or the other provided thermal data near the times of precession maneuvers. These maneuvers oriented the vehicle from sun nominally 5 deg aft to the sun nominally 5 deg forward and the payload would thermally respond accordingly. Precession maneuvers were not always 10-deg maneuvers, but were nominally so.

Payload SC7, which is a light ion mass spectrometer, had one sensor on the forward end of the vehicle and one on the aft end. When the aft end of the vehicle was in the sun that sensor operated at +10°C while the forward head operated at 8°C. The test data for these were 14°C and 8°C, respectively. When the forward end was in the sun after a precession maneuver, the forward sensor operated at +14°C and the aft head at +7°C. Test values were +14°C and 8°C, respectively. Not only is the agreement very good, but the test predicted the greater difference for the forward sun case as well.

For the case of the long (100M) "extrudable" electric field detectors in payload SC10, they varied from +2-+3°C on flight and from 0-3°C in test. The cold case in test was not as warm as in flight due to the fact that they could not be deployed appreciably, and therefore did not have the radiative coupling to the sun due to available surface area as they did in flight.

Solar Aspect Effect on Sensitive Payloads

Two payloads provided sufficient data to demonstrate the thermal dependence on solar aspec angle. The first, SC9, which measures energy and angular distribution of electrons and protons in the spacecraft environment, mounted on the forward end of the vehicle, had a drastic dependence on solar angle which favored the sun in a forward position. A plot of temperature for one of the heads and the power unit going through a precession firing on orbit for June 22, 1979 is shown in Fig. 11. In this plot the vehicle is precessed from −5 deg solar angle to +5 deg. The unit is turned off through the actual maneuver as indicated and then turned on again to a power consumption higher than test, which accounts for the final values being higher than the test values as indicated. The drastic dependence of the temperature on solar aspect angle is vivid in this particular plot.

Payload SC11-1 provided a very *sensitive* thermal response to solar angle as well as seasonal trend since it was located so far (4 m) from the vehicle. This location caused values to be lower in flight than in test as has already been discussed. The temperature was found to be least in flight at 0 deg aspect angle. No test was run at this configuration, but rather at +5 and −5 deg. As the test predicted, this payload ran hottest with the sun aft. This particular temperature response to sun angle was repeatable even during eclipse season and will be discussed later.

Eclipse Thermal Response

One of the special events simulated in the thermal vacuum test was the eclipse. This was produced by closing the "douser" of the solar simulator, and later opening it. Due to schedule constraint, sun angles were changed when the douser was reopened, making only the "cooldown" portion of the eclipse test data strictly comparable with flight data. Also, to provide sufficient margins, test eclipses were typically run for 1¾-2 h, and on orbit the longest eclipse was a little over 72 min. If the test eclipse data are superimposed over flight so that the eclipse entry time on flight corresponds to the douser closing time, the data are then strictly comparable up to the point in time when the vehicle exited the Earth's penumbra.

An eclipse plot for one of the detector heads and the associated drive motor box for SC9 is shown in Fig. 12. The agreement is excellent. Different days were chosen so that flight eclipses *started* at the same temperature values as in test.

SC11-1 is again a good and sensitive indicator for eclipse. A plot of five consecutive on-orbit eclipse events is shown in Fig. 13. The events also include a precession firing which reoriented the vehicle to +5 deg sun angle where the test data were taken. The corresponding test data are superimposed for April 14. It is interesting to see the repeatable dependence on solar aspect angle again as discussed previously. The minimum temperature occurs at 0 deg sun angle, and the difference in temperature from negative to positive sun angle is slightly over 2°C. The difference between 0 deg sun angle and sun aft is slightly over 4°C. If one takes into consideration that the test will be somewhat hotter due to the shortened stub test boom, the test eclipse data are in good agreement with flight data.

Transfer Orbit Thermal Response

Some data were available in the transfer orbit portion of the mission. These were from SC9 and were generally in good agreement with the simulated transfer orbit balance test run at a nominal +45 deg sun angle. One head and the motor box ran at about 36°C and 34°C, exactly the same as in test. The other head ran at 26°C, while in test it ran at 34°C. The power unit whose temperature is drastically dependent upon the operating mode varied from 33-54°C while the test value was 52°C.

Thermally Controlled Components

The propulsion system had several components which were kept within thermal specifications by heaters. In the case of the catalyst bed heaters, which prepared the rocket engine modules (REMs) for firing, the heaters were ground commanded, and in the other cases they were cycled automatically. In all cases the cycled temperatures were essentially in exact agreement with those from the test. Examples are as follows with test values in parentheses.

For the valve modules 18°C and 24°C (19°C and 24°C), for REM No. 2 19°C and 27°C (19°C and 27°C), and for the hydrazine thermostat blocks 21°C and 29°C (21°C and 29°C).

Conclusions

Flight data and test data were generally in very good agreement for both the Viking and SCATHA programs, where comparisons are valid. For a program as complex as Viking, the mission has confirmed both the philosophy of the test program adopted and the applicability of the space simulation chamber for such a diverse test profile. For a program as unique as SCATHA, the fidelity of the space simulation capabilities of the STVC have been demonstrated to be excellent in both simulating environment, and verifying vehicle "health" systems such as thermal control. Both the Viking Lander 1 and SCATHA vehicles remain healthy and operational today, way beyond their planned mission durations. This fact is, in part, due to a good test program and facility.

In order to be able to proceed to a modular space structure vehicle fabrication approach, which will be required as space systems and structures increase in size, future test programs should be geared to "off the shelf" items whose thermal response when integrated into a larger system is known or can be predicted from proper modeling approaches. This will be greatly accelerated if proper engineering data are carefully integrated into the mission of presently planned programs as part of normal downlink. Such a plan will provide the necessary data base to model the thermal response of systems too large to be tested on Earth.

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