System Test and Flight Results from Infrared Astronomical Satellite External Thermal Subsystem

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The Infrared Astronomical Satellite (IRAS) received its system level thermal testing in a solar simulator in September and October 1982 and was launched in January 1983. In order to reduce costs and meet a tight launch schedule, a system thermal test approach was employed which combined the goals of thermal design development tests, thermal balance tests, and thermal vacuum qualification tests. This approach relied heavily on the use of a computer-controlled automatic data handling and heater control system which reduced reliance on human monitoring of test data in comparison to past projects. The validity of this approach has been borne out by flight results. During the whole mission most temperatures were within the desired range and close to predictions. Included under the external thermal subsystem are room temperature electronics and radiatively cooled surfaces with temperatures below 100 and 200K.

Introduction

THE Infrared Astronomical Satellite (IRAS) is a cooperative project involving three countries as well as several private companies and NASA organizations. The scientific objectives and project organization were described by Clegg[1] and Van Holtz,[2] respectively. The telescope and spacecraft were described by Irace and Rosing[3] and Pouw.[4] Several other papers[5-9] describing different aspects of the project and its mission are contained in a special issue of the *Journal of the British Interplanetary Society*.

The satellite is shown in Fig. 1. The telescope consists of 1) a superfluid helium Dewar, to cool the optics and detectors, whose outer shell must be cooled in orbit to below 200K; 2) a set of boxes containing signal processing and other electronics which must be kept near room temperature; and 3) a sun shade which protects the telescope aperture from direct sun and Earth radiation. The sun shade interior is a gold-coated surface with a design temperature below 100K in orbit.

The orbit is approximately 900 km, near polar, twilight, sun synchronous. During the course of the mission, the orbit beta angle (smallest angle between the orbit plane and the sun line) may cover the range from minimum of 58 deg to a maximum of 90 deg.

The telescope may be pointed in any direction within a space bounded by both of the following constraints: 1) the angle between the telescope boresight and the zenith must be less than 28.8 deg, and 2) the angle between the sun line and the telescope boresight must be between 60 and 120 deg.

The thermal design of the telescope exterior and sun shade were described by Anderson and Hilton[10] and Anderson,[11] respectively. Thermal design and modeling of the cryogen system were described by Hopkins and Brooks,[12] flight results by Urbach and Mason.[13]

Briefly, the telescope exterior shell is cooled by means of a white-painted radiator which is oriented to face away from the sun within the constraints imposed by the orbital parameters. Earth heating is minimized by means of an Earth shade below the radiator. Thermal isolation from the room temperature spacecraft is provided by multilayer thermal blankets and low-conductance titanium support trusses.

Room temperature electronics boxes are mounted separately around the telescope on low-conductance fiberglass support trusses. Radiative isolation from the telescope is provided by multilayer insulation blankets. The electronics boxes are passively controlled, either by radiators sized to the power requirement of the individual box or by being thermally connected with copper straps to a box with a radiator. Other than these radiators, the boxes are all covered by multilayer insulation blankets. Each of the radiators is covered by a combination of either high-emittance teflon tape or white paint and low-emittance aluminized tape. The use of these combinations of surfaces allows adjustment of the effective emittance of the radiator surface either up or down by changing the ratio of the high- and low-emittance surfaces.

Most of the electronics boxes are mounted in close proximity to each other on the $+Z$ side of the telescope behind the solar panel as shown in Fig. 1. One box, the Dutch Additional Experiment (DAX) electronics, is mounted by itself on the $-Z$ side.

The telescope supports a low-conductance fiberglass mounting structure for the sun shade. This has three radiator stages surrounding the telescope aperture, the coldest of which is designed to operate below 100K. The cold stage consists of a radiator and gold-covered cone section covering the interior of the sun shade, which is visible to the telescope aperture.

During all ground operations, including the thermal balance test, the telescope aperture was closed by a liquid helium-filled cover. Attached to this was a thermal blanket covering the two coldest radiator stages. During the first few days in flight, the aperture cover, filled with supercritical helium, protected the cold telescope interior from contamination due to out-gassing from the satellite as well as providing a cold calibration target for the detectors.

The challenge in satellite thermal design stems from the fact that the thermal regimes described above must be effectively isolated from each other for the telescope system to perform correctly, that is, for the sunshade to provide a low thermal input to the telescope, for the helium boiloff rate to be low enough to meet the desired lifetime, and for the signal processing, power, and switching electronics to function correctly.

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The thermal balance test was intended mainly to demonstrate the thermal compatibility of the spacecraft and telescope. This was also the only time before flight to test the performance of the cryogen system, that is, the helium boiloff rate under flight-like conditions. Thus the cryogen system was required to have as long an undisturbed period in the chamber as possible so as to allow the telescope exterior to cool down and the helium flow rate to approach flight-like levels. This meant that if at all possible, no breaks should be made in the test to fix or adjust the thermal hardware. Finally, a phase of high-temperature functional testing was required to verify the behavior of boxes which had experienced minor modifications after being individually qualification tested.

The result of these requirements was a test which combined the objectives of different types of tests which in the past had been done separately, namely thermal design development tests, thermal balance tests, and system thermal vacuum qualification tests (see Ref. 14 for a full explanation of these terms). Briefly the purposes of these tests are the following:

1) Thermal design development test to determine the characteristics of parts of a thermal design which are difficult to determine by analysis.

2) Thermal balance test to verify the ability of the thermal subsystem as a whole to maintain all hardware elements within established limits and to verify the analytical model.

3) System thermal vacuum qualification test to insure the ability of flight hardware to function correctly over a temperature range greater than that expected in flight.

Thermal Balance Test Design

The satellite thermal balance test plan consisted of four test phases. The first was a warm test with the telescope normal to the solar beam and a high level of solar intensity (1420 W/m^2). The second was a cool test with the telescope tilted 30 deg with respect to the solar beam and a low level of solar intensity (1290 W/m^2). The third phase called for half the electronics to be turned off, as would occur in a potential failure mode, with the sun at low intensity, the telescope tilted 30 deg, and a simulated solar eclipse. The final phase called for all the electronics boxes to be heated to +40°C for a period of 12 h of functional testing.

The purpose of the first two tests was to evaluate the response of the thermal subsystem to conditions similar to the extremes expected in orbit. The purpose of the third test was to evaluate the thermal effect of the simulated failure of part of the electronics on the remaining operational electronics, in the worst case including a simulated 15-minute solar eclipse. The high-temperature phase was intended to demonstrate correct operation of all external electronics at a temperature higher than that expected in orbit. Although each separate box was predicted to have a different maximum temperature in orbit, the single maximum test temperature of +40°C was chosen, first, to simplify the test procedure, second, because it was acknowledged that there was considerable uncertainty in the predictions, and third, because nearly all the telescope electronics were qualified to the same temperature range.

A thermal model simulation of the entire test was run in order to provide a standard for comparison of test temperatures and to assist in planning the length of each test phase. Except for the solar eclipse test, it was intended that near steady-state temperatures should be achieved in each phase. Orbital operations are such that changes in temperature occur slowly, mostly over periods of half a day or more, or in response to seasonal variations in the orientation of the sun to the orbit plane.

The test hardware consisted of a support structure which held the satellite in a horizontal position relative to the vertical solar beam and provided the tilt of 30 deg; cryogen lines and electrical cables; thermal blankets covering all of the aforementioned hardware; test thermocouples located on all thermally separate pieces of hardware; and film heaters also located on each thermally separate piece of hardware.

The test hardware setup was designed to provide a flight-like simulation, except for Earth heating, where the heat flows at all external hardware interfaces would be controlled to zero by combinations of guard heaters, radiators, and thermal blankets. The purpose of the purely thermal test hardware, heaters and thermocouples, was to monitor all important temperatures and insure the safety of flight hardware with respect to maximum and minimum limits.

The heaters were automatically controlled by the data handling system computer so that the reading from each differential thermocouple would be zero. The radiator insured that without heating the item in question would cool down below the telescope structure so as to provide a positive control margin. An absolute thermocouple was attached so that the test team could monitor the function of the guard heater and insure that the absolute temperature of the support or cable was near that of the satellite structure to which it was attached.

A heater and thermocouple were attached to each electronics box. These were programmed for automatic operation at a temperature 5°C above the lower qualification temperature.

Heaters were also attached around the exterior shell of the cryogen Dewar and to each of the stages of the sun shade so that they could be brought up to room temperature quickly when the test was completed. This protected the surfaces from contamination as well as providing a means to end the test quickly.

No simulated Earth heating was applied to the telescope exterior because of the associated expense and complication. The thermal model was used to make the transition from test temperatures to flight temperatures. The effect of Earth heating was expected to be small, since the geometry of the design was intended to minimize this contribution.

As part of contingency planning it was decided that in the case where the electronics boxes were too cold the heaters would be manipulated to characterize box thermal behavior empirically and help to determine the required change in thermal design. The thermal design of the electronics boxes, as mentioned, depends on conductive and radiative isolation

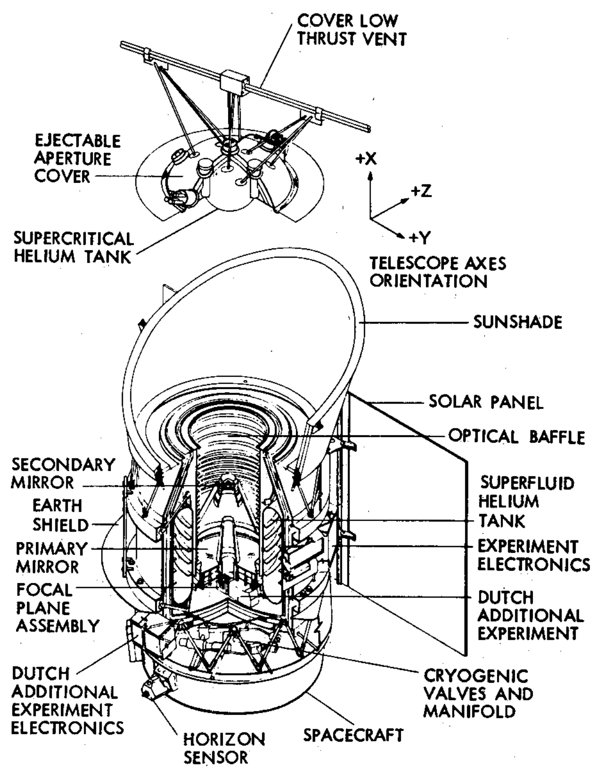

Fig. 1 Satellite configuration.

Table 1 Predicted equilibrium temperature vs actual temperature in first thermal test phase

	Predicted	Actual
External electronics boxes, °C	−8 + 32	−15- + 3
Cryogen Dewar exterior shell, K	173	184
Sun shade cold stage, K	143	153

Table 2 Predicted vs actual temperature during first five months in orbit

	Predicted	Actual
External electronics boxes, °C	−13- + 24	−9- + 21
Cryogen Dewar exterior shell, K	183-202	195-198
Sun shade cold stage, K	81-96	96-102

from the cryogen Dewar and radiators facing away from the telescope. By changing the ratio of high-emittance tape or paint to low-emittance tape, a certain amount of temperature adjustment is available irrespective of the cause of the incorrect temperature.

In summary, the test was designed to meet the following objectives simultaneously: a) characterization of the thermal design as built, b) minimization of the danger to one-of-a-kind flight hardware by means of automatically controlled heaters, c) minimization of the possibility of an interruption once the test was begun, d) empirical determination of required thermal design fixes so as to minimize the need for a retest, e) providing of efficient methods of making the transition between test phases and warming up to room temperature once the test was complete, and f) demonstration of the correct operation of the external electronics at a temperature above the maximum expected in flight.

The approach to meeting these objectives relied on the following major elements: a) prioritizing test objectives such that the most important is carried out first; later and less important objectives might be compromised in order to achieve the most important objectives; b) providing a flight-like simulation such that not too much reliance is placed on a still-unproven thermal model; c) using a single maximum electronics soak temperature; d) developing contingency plans designed to minimize the possibility of interrupting the test, once begun; e) automatic control of heaters to prevent electronics from exceeding their lower qualification limits; and f) automatic logging, plotting, and display of all temperatures and heater power by the data handling system.

The guiding philosophy behind the facility setup called for minimizing the use of test engineers to perform data monitoring while maximizing the freedom of the engineers to use a limited test period in the way that yielded the most information.

Test Results

The results of the first test phase with high solar intensity and the satellite normal to the solar beam are summarized in Table 1. The consistently low values of the electronics and the high value of the main shell temperature indicated that there was more heat flow between these two than had been expected from the thermal modeling.

The sun shade temperature was considered reasonably close to the prediction, since modeling of this item in the test configuration with the two coldest radiator stages covered by a multi-layer blanket was difficult. The sunshade was tested alone previously as described by Anderson.[11]

At this point it was decided not to proceed with the test as planned but to exercise the contingency plans to characterize the thermal design empirically. Over the remainder of the test period, four separate equilibrium data points were achieved at two temperatures in each of the two satellite orientations. The equilibrium points were achieved by using the automatically controlled heaters in the following way: all boxes with radiators were driven to a set temperature while those without radiators were first driven to a predicted temperature and then allowed to float with no external heat. The heater power for each equilibrium point was logged automatically. This information was to be the basis for correcting first the thermal model and then the design.

The remaining test phases were carried out as planned except for minor modifications to accommodate the failure of one of two redundant microprocessors in the telescope electronics. Recovery from this problem was possible because the external heaters allowed the essential thermal information to be obtained in a different way from that planned.

Test Data Analysis

The thermal design problem revealed by the test results was that heat paths from the room temperature electronics to the cold telescope were much larger than expected. The philosophy followed in correcting the thermal design was to make the quickest change to the electronics boxes that would have a high probability of success, thereby minimizing the need for any further thermal testing. The test data were used to determine the heat radiated from the electronics box radiators and then to reduce the overall emittance of the radiators by increasing the portion of low-emittance aluminized tape.

On the basis of the results of the first test phase, a simplified thermal analysis was begun which considered each electronics box separately. This was intended to verify quickly that a fix was possible by changing the radiator patterns, and to produce an estimate of the magnitude of the change.

The complete satellite thermal model had been produced by combining that for the telescope with that for the spacecraft. This model has 186 nodes and 1840 thermal connectors. Because of its size and complication, a simplified analysis was expected to produce quicker answers, provide more physical insight, and provide a check on the computer model.

On the basis of the results of the full test, 33 additional thermal connectors were added to the model, primarily among the electronics boxes and between the boxes and the cryogen Dewar exterior shell. The addition of these connectors provided better agreement with the test data. The model predictions were further improved by using a model correction routine developed by L.C. Wen[15] which automatically varied a selected group of connectors in order to produce a solution which closely matched the test data at the steady-state data points. The complete thermal model was then used with orbital heat inputs computed with the LOHARP[16] program to determine the effect on flight temperatures of the design changes which had been derived from the simplified analysis.

The resulting reductions in radiator emittances ranged between 7 and 61%. The box which had been the farthest from predictions, the DAX electronics, had a radiative fin removed, gaps in thermal blankets covered, and also had an additional heater capable of being commanded from the ground installed. This box contains a thermostatically controlled heater; plans were made to allow the set point to be adjustable by ground command.

Flight Results

A comparison between predicted and actual temperatures for the the first five months in orbit is given in Table 2. The external electronics boxes have matched the predictions very closely, with the only difference being that the predicted extremes for worst-case environments have not occurred. The exterior shell of the cryogen Dewar has been from 4 to 9K above the predicted value, with the difference being near 4 to 5K in the period between three and five months after launch.

The sun shade has consistently been 12 to 13 K above the predicted value. Both the cryogen Dewar exterior shell and the sun shade temperature variations have been less than predicted.

The reason for all temperature extremes being less than predicted is that the predictions are generated by placing the satellite in a fixed orientation throughout an orbit in order to generate the orbital heat inputs for the thermal model. During orbital operation, the telescope is not fixed but is moved through a number of orientations during any given orbit. Thus, the extremes in orbital heat loads tend to be smoothed out, with the same effect on temperatures.

The electronics box which had the greatest variation from the prediction was the DAX electronics, just as in the thermal balance test. This box was kept close to the desired value only by turning on the extra heater which had been added after the test.

The greatest difference between predicted and actual temperature has been on the sun shade. This was tested separately as noted previously.[11] The results of this test were somewhat ambiguous due to the failure of some of the test hardware, namely detachment of high-emittance tape and film heaters from the surfaces to which they had been attached. The general conclusion from these test results was that the ability of the sun shade to meet the design limit of 70 to 100 K in orbit had been verified. This, in fact, has been very nearly achieved.

Conclusions

The successful flight results shown by IRAS have demonstrated the validity of several aspects of the program of designing, building, and testing the satellite. The satellite thermal test showed that it was possible to combine partially the functions of a thermal development, a thermal balance, and a thermal vacuum test, while still maintaining control of schedule and budget. This was made possible by the use of automated data handling and heater control systems, but depended most basically on the commitment to use the test period in a dynamic fashion to gain knowledge of the design, rather than as a pass-fail type of operation.

The test was completed on schedule, even though there were serious discrepancies from predictions, with the result that the thermal subsystem was characterized well enough to make significant design changes with high confidence.

The design has shown itself in orbit to be a basically stable and robust one which has, in most cases, exceeded expectations. All system level functions of the thermal subsystem have been achieved successfully, even where there were deviations from predictions.

The sun shade flight results have shown that it is possible to use a relatively uncomplicated design for a large radiative cooler to achieve temperatures near 100 K in low-Earth orbit. Even though the predicted temperatures were missed by some 12 K, the temperatures achieved must be considered a significant step, both in terms of cooler size and difficulty of orbital environment.

Acknowledgment

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