AIAA 80-1500R

OTS: Two Years of Thermal Control Experience in Orbit

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Since 1978 a thermal in-orbit test has been performed every three months with the geostationary, three-axis stabilized communication spacecraft OTS-2. By direct comparison and through computer-aided analysis the initial performance of the thermal subsystem and its variation during operation has been observed. From the measured temperatures and operational data, the degradation of thermal control surfaces, in particular that of the second-surface-mirror radiators, has been determined.

Nomenclature

t, t_0 = time
α = absorptivity
ΔT = temperature deviation
σ = standard deviation

Introduction

THE Orbital Test Satellite OTS-2 was conceived to validate, in a three-year orbital test program, the telecommunication technologies and spacecraft hardware intended for use in the European Space Agency's communication satellite program. The OTS spacecraft is three-axis stabilized and was positioned in May 1978 in a geostationary orbit at 10°E longitude.

Although most of the tests performed to date within the Orbital Test Program have been dedicated to the telecommunication payload, test time has been allocated every three months at each solstice and equinox to observe the thermal subsystem performance in a unique series of tests under stable and well-known operational conditions.

After a short description of the spacecraft and its thermal control subsystem, this paper presents the thermal test results and their interpretation.

System and Thermal Concept Description

With the future development of European Communication Satellite (ECS) and other space communication systems in mind, the OTS spacecraft was conceived as a "bus" (Fig. 1) which houses all service subsystems and carries a communication module dedicated to the particular mission. The total power dissipation in the spacecraft is typically 580 W in solstice, dropping to 90 W during eclipses in equinox periods. The temperatures to be maintained by passive thermal control are given in Fig. 2. The following system characteristics are important with regard to a thermal evaluation.

In the payload a maximum of six traveling wave tubes (TWT) can be operated during sun illumination and two channels during eclipse. As traffic demand varies, the power output of each TWT is variable, as is its internal power dissipation. This variation is a few watts per tube and, at most, around 1 W for its associated electronic power conditioner (EPC). Simulation heaters automatically replace the power dissipation of the TWT and EPC when they are switched off. In the service module noticeable operational variations occur in the power conditioning unit (PCU), the dissipation of which depends on the payload consumption and the switchover from one fixed momentum wheel (FMW) to the redundant one, which introduces a dissymmetry in the local distribution but not in the total level of dissipation.

The equipment with the most important power dissipation excursion is the shunt which controls the solar array output. Its dissipation varies between 40 and 145 W. Although the shunt electronics are mounted in a dedicated radiator area on the north face of the spacecraft, it is by far the largest contributor to changes in the spacecraft's temperature level. It renders direct orbit test comparison difficult for tests which have identical operational status but different shunt power dissipation resulting from the slow degradation of the solar array output.

The thermal control concept selected under these constraints is summarized in Fig. 3. Two platforms supported by a central tube carry the electronic components. The north/south walls are used as the primary means to radiate the dissipated power into space. Dedicated radiators enable

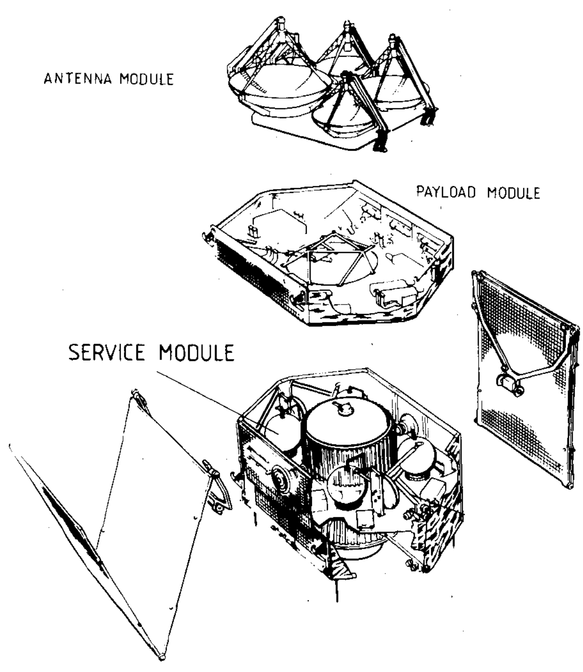

Fig. 1 OTS configuration.

Presented as Paper 80-1500 at the AIAA 15th Thermophysics Conference, Snowmass, Colo., July 14-16, 1980; submitted Jan. 14, 1981, revision received July 20, 1981.

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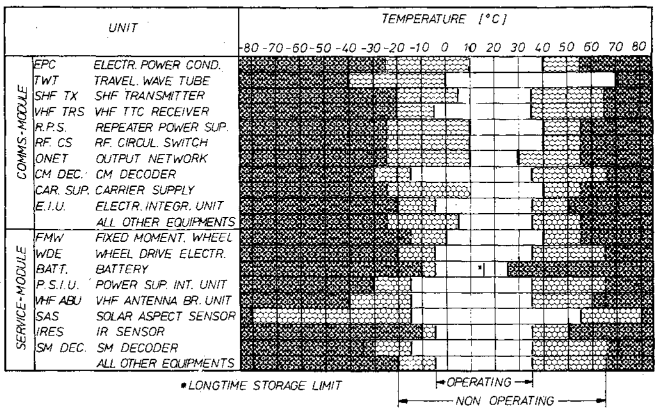


Fig. 2 OTS temperature limits.

the TWTs to make efficient use of their elevated operating temperature and reduce their influence on the spacecraft. The shunt electronics also has its own radiator because of its highly concentrated and widely varying power dissipation. All radiator surfaces are covered with a rigid quartz optical solar reflector [second surface mirrors (SSM)] to limit the solar input. The east/west faces are insulated with multilayer insulation (MLI). On the Earth-viewing panel, a farm of white (S13G-LO) painted antennas is mounted on a rigid antenna platform via low-conductance standoffs. On the surface facing away from Earth, the launcher adapter, the apogee kick motor (AKM), the vhf antennas, and 12 hydrazine thrusters break the MLI blanket continuity.

Tests Performed

During the course of the OTS program, thermal tests were performed on the ground, simulating space temperatures and solar irradiation at extreme equinox and solstice conditions under selected fixed solar input angles (steady-state tests). The purpose of these tests was to verify in a simulated space environment that the spacecraft performance was within acceptable limits and that the thermal mathematical model used for the extreme hot and cold design cases was capable of reasonably predicting the test results, thus giving confidence in the orbital predictions.

OTS being a test satellite, it had been decided early in the program to include the thermal subsystem in the in-orbit performance tests. The objectives of such an endeavor were somewhat different from the objective of the tests performed on the ground. They can be summarized as follows:

1) To assess the adequacy of the thermal control subsystem in providing the proper environment for the different equipment on the spacecraft.

2) To compare flight temperature with analytical predictions for the same electrical configuration, and to ascertain the degree of accuracy of such orbital predictions.

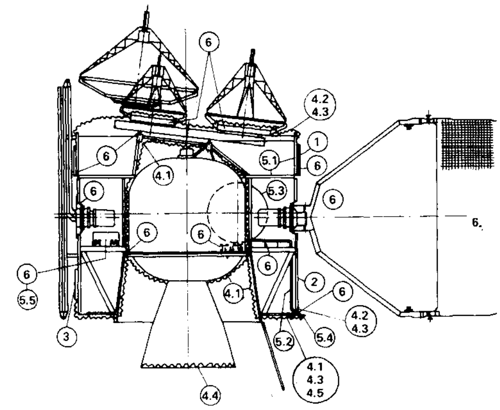
3) To provide the possibility of assessing the thermal distortion of a large antenna dish.

4) To detect and quantify at specified intervals any degradation of the thermal coatings under the influence of the space environment.

Since launch in May 1978, dedicated thermal tests have been performed (Table 1) every equinox and solstice and at other ad hoc periods. To establish thermally representative conditions, the dedicated tests were carried out for approximately two days with a well-defined and constant power dissipation and repeater status.

Available Test Data

To achieve the combined objectives stated in the preceding paragraph, the spacecraft was instrumented with 152 thermistors, thermocouples, and platinum resistance sensors to map the temperatures. The summary of sensor apportionment



1. TWT RADIATORS
   - PROFILED HONEYCOMB SANDWICH
2. CONTROL RADIATORS
   - SSMs
   - FOIL FOR TRIMMING
3. SHUNT RADIATOR
   CONDUCTIVELY DECOUPLED FROM STRUCTURE
4. MULTILAYER INSULATION
   4.1 KAPTON CONCEPT HIGH TEMPERATURE
   4.2 MYLAR CONCEPT
   4.3 VHF SHIELD
   4.4 BLOW-OFF INSULATION FOR ABM NOZZLE
5. HEATERS
   5.1 PAYLOAD SIMULATION
   5.2 HYDRAZINE LINE HEATERS
   5.3 TANK HEATERS
   5.4 HYDRAZINE VALVE HEATERS
   5.5 BATTERY HEATERS
6. LOW CONDUCTANCE FLANGES & BRACKETS
   - ANTENNA DISHES
   - RADIATORS
   - BAPTA
   - BATTERIES
   - TANKS
   - LINES
   - FCVs
   - ABM
   - ANTENNA PLATFORM

Fig. 3 OTS thermal control concept.

Table 1 Orbital tests performed

Test date	Remarks
May 25-26, 1978	First orbital assessments, payload operated normally
Summer solstice 1978	Dedicated thermal test
Autumn equinox 1978	Dedicated thermal test
Winter solstice 1978	Dedicated thermal test
Vernal equinox 1979	Dedicated thermal test
May 25-26, 1979	Repeat of previous year (cyclic conditions not fully reached)
Summer solstice 1979	Dedicated thermal test
Autumn equinox 1979	Dedicated thermal test
Winter solstice 1979	Dedicated thermal test
Vernal equinox 1980	Dedicated thermal test
May 25-26, 1980	Repeat of previous year (fairly stable conditions)
Summer solstice 1980	Dedicated thermal test

is given in Table 2 for the different subsystems, including 36 sensors which are internal to the equipment and required by the manufacturers. They do, however, provide the thermal subsystem with some additional temperature information. Of these 152 sensors, 60 are mounted on components outside the main body of the spacecraft, such as the antennas and solar array.

All sensors have been calibrated but, due to the data formating for telemetry, an uncertainty of between 0.3 and 0.8 °C, depending on the type of sensor, remains. The larger values are essentially attributed to the platinum sensors mounted on equipment with large temperature swings, such as the antenna, solar array, and shunt radiator.

During the two days of each thermal test the reading of all sensors are telemetered in 25.6 s intervals. The cyclic reproduction of the temperature timeline is verified by comparing the temperatures at discrete times of the two consecutive days. Table 3 shows that reasonably stable conditions have been achieved in all tests, although distinct differences are noticeable. Only the four hydrazine tanks, which have a high thermal capacity and are well insulated, did not systematically reach a thermal equilibrium during the 48 h of test.

One major problem confronting the thermal analyst when making temperature predictions is the accuracy of the power dissipations introduced in the mathematical model. Individual power dissipations are usually known from previous testing at the component or system level but are very rarely determined for the exact flight configuration to be studied and analyzed. In the OTS the dissipation of the TWTs, EPCs, PCU, batteries, and shunt and, to a lesser extent, of filters and operating networks is effectively a function of the rf power output which is not as accurately known as the dissipation of a heater.

From the power subsystem telemetry of OTS, it is possible to estimate the power capability of the solar array and also to calculate the global values of the spacecraft consumption. The currents leaving the active and inactive sections of the solar arrays are measured and from these data the power generated can be determined. On the spacecraft side, the power consumption is calculated at that point on the electrical bus where current and voltage are measured. The energy used before the current-measuring point has to be evaluated separately for the power dissipations of the shunt, battery, and PCU. A comparison can be performed between the capability of the arrays and the spacecraft consumption. A typical example is given for summer solstice of 1978 in Table 4, which illustrates that a few watts difference (uncertainty) exists between the dissipation determined from orbit measurements (columns 1 and 2) and between these measurements and the dissipation expected from the ground tests (column 3).

e ve largest individual uncertainty in power determination exists on the rf output. Therefore, in the thermal tests the channel dissipation has been replaced to the maximum practicable extent by simulation heaters or those three channels have been used which permit a measurement of the rf output.

Table 2 Sensor distribution

Subsystem	Sensors
Reaction control system	38
Power	13
AKM	2
AOCS (attitude and orbit control system)	14
TTC (telemetry and telecommand system)	9
Antenna	24
Repeater	40
Structure	12
Total	152

An analysis of the statistical variations in the various data used for power calculation has shown that the uncertainty is approximately 8 W, which is equivalent to almost 1°C temperature level of uncertainty in the spacecraft.

Operating Conditions of the Spacecraft

For all thermal tests the equipment operation has been very well defined before hand and it has been tried particularly to keep the repeater activity at a minimum. Nevertheless, due to operational constraints, the aim to have identical operating conditions in all comparable orbit tests has not been fully achieved. Table 5 indicates the main differences which cause significant variations in dissipation. Of course, any switching during the test time is closely monitored and the switching times are kept identical for each day. In this respect the monitoring of the shunt radiator is also important because while the switching of a small consumer may have a relatively minor influence on the shunt dissipation if it is operating in the middle of its range, near the upper or lower limit of the shunt capability it may require one solar array section more or less, with a flip of the shunt operating point to the other side of its range (a variation of more than 100 W).

Orbit Test Evaluation

A spacecraft equipped with so many sensors and subjected to such a comprehensive series of tests provides answers to detailed questions which are of great interest to the thermal designer.

As the performance of a thermal control subsystem cannot be fully verified on the ground, the first point of interest is verification that the temperatures of the electronic and other components lie within the tolerances computed, using worst case assumptions for the full mission of the spacecraft. The next important question is in defining the precision of the thermal mathematical model, which is formulated analytically and improved by ground tests, for a given operational configuration of the spacecraft.

In view of the extended lifetime of a communication spacecraft, the most important questions regarding thermal control concern the long-term stability of the employed thermal technologies in the geostationary orbit environment. It is known from ground tests performed during the last two decades [e.g., those of the Department d'Etude et de Recherche en Technologie Spatiale (DERTS[1])] and also from orbit observation[2] that the thermo-optical properties of materials degrade under the electromagnetic and particle environment in space. The extent of this degradation depends on the selected material, the technology of application, the cleanliness of the surface, and disturbances coming from the spacecraft (contamination either directly or by reflection at the appendages). It is further suspected and the possibility has been demonstrated in ground test[3] that the differential electrostatic charging and subsequent arc discharges which may occur on spacecraft surfaces in geostationary orbits may damage such thermal control materials as MLI and SSM.

Table 3 Quality of cyclic temperature reproduction over 24 h periods

Date	Amount of sensors	Identical	Reading differences +1 BIT	−1 BIT	+ >1 BIT	− >1 BIT
May 1978	152	75	19	27	21	10
Summer solstice 1978	152	130	7	14	–	1
Autumn equinox 1978	140	99	21	16	2	2
Winter solstice 1978	151	92	55	–	2	2
Vernal equinox 1979	140	87	24	26	1	2
May 1979	138	16	26	6	68	22
Summer solstice 1979	150	118	29	1	2	–
Autumn equinox 1979	150	91	44	10	5	–
Winter solstice 1979	138	93	14	10	20	1
Vernal equinox 1980	138	92	24	10	2	10
May 1980	138	60	63	4	7	4
Summer solstice 1980	144	117	24	1	1	1

The data available from the OTS thermal tests performed in orbit provide some answers to these questions that are valid not only for this particular spacecraft but also with certain reservations for other three-axis stabilized spacecraft in geostationary orbits having thermal control materials and technologies similar to those of OTS.

Method of Evaluation

The initial thermal subsystem performance or, more precisely, the accuracy of the mathematical model, can be verified by a straightforward comparison of the actual orbit temperatures with those predicted by the mathematical model when exercised under identical operating and environment conditions.

Degradation mainly affects the reflectivity of surfaces in the solar spectrum and generally leads to a reduction in that reflectivity. Changes in the infrared spectral region are, with few exceptions, of comparatively minor importance. Thus degradation results in additional absorbed heat in the spacecraft and consequently in a temperature rise during the life of a spacecraft. To assess the effect of degradation, a threefold use can be made of the OTS orbit thermal test data by:

1) Direct comparison of temperatures under identical external conditions. This presumes that the operational configuration and in particular the power dissipation of the spacecraft is identical in the cases compared. If this is not the case, knowledge of the sensitivity of the spacecraft temperatures to power variations is a prerequisite for this approach.

2) Duplication of the orbit measurements by a mathematical model which takes into account the actual operational configuration of each test and indicates the effect of degradation by increasing differences between measurement and analytical results. An alternative to this would be to maintain the initial correlation of measurement and analysis constant by allocating suitable degradation rates to the various surfaces.

3) Direct comparison of individual sensor readings from areas that are mainly influenced by external heat input and for which the possible internal coupling to the spacecraft is either negligible or quantifiable.

The first two approaches have an integrating character in the sense that the increase of temperature results from the degradation of every surface of the spacecraft. The last method allows precise determination of local degradation and is often used for dedicated experiments with calorimeters.

Table 4 Power distribution during summer solstice 1978, W

1 Array power – shunt – rf output	2 Spacecraft load – rf output	3 Sum of unit dissipation from ground test
427.8	422.4	411.3

Initial Thermal Performance

The mathematical models needed to design and verify the OTS thermal control system were used to predict both steady-state thermal conditions and the transient temperature response to the daily cycle of the spacecraft relative to the sun and to variations in power dissipation. Reports[4,5] have been written on the quality of these models and therefore only a summary of this information is presented here.

After launch from the Eastern Test Range on May 11, 1978, the first in-orbit thermal interpretation was conducted on May 25-26 after the spacecraft had reached its final position and was set into its normal operational mode. Histograms of the temperature deviation between the orbit data and the analysis were generated which show that the mean diurnal temperature was approximately 1.5 °C above predictions with a standard deviation of the order of 5 °C. The diurnal temperature variation was approximately 3.0 °C larger than predicted.

Whereas the average values and standard deviation are similar to other spacecraft data[6] and were considered satisfactory, the larger diurnal variation was a surprise that could be traced back by detailed investigation[7,8] to an underestimated solar input into the spacecraft adapter and the cavity of the AKM. This area of the spacecraft had never been exposed to the sun during ground tests because the adapter was used to mount the spacecraft on the test facility. Corrections in the solar input and in the area of the AKM brought the following improvement:

1) The average diurnal mean temperature difference between the orbital experience and the prediction was reduced from 1.5 to 0.7 °C.

2) The deviation of the average diurnal variation was improved from 3.0 to 1.8 °C.

3) The standard deviation was also improved by approximately 0.5 °C.

All subsequent orbit test evaluations have been performed with this corrected model.

Thermal Degradation

Direct Comparison

The simplest and most unambiguous method of determining any effects of thermal degradation on the spacecraft is a comparison of temperatures on exactly the same day in consecutive years under exactly the same operating conditions. Unfortunately, even OTS with its dedicated thermal test program did not exactly achieve such conditions during all tests.

In every test the shunt electronics, which dissipate the excess power from the solar array, have different power

Table 5 Operation variations

	SS 78	SS 79	SS 80	WS 78	WS 79	AE 78	VE 79	AE 79	VE 80
Payload									
TWT A2 A	On	H	H	H		H	H	H	H
TWT A2 B					On				
TWT A2 bar	H*					H	H	H	H
TWT A4	No drive	H	H	H	On	H	H	On	H
TWT A4 bar A	On	On	On	On	On	On	On	On	On
TWT A4 bar B									
TWT A BCRL (Beacon right-left)	H	H	No drive	H	No drive	H	H	H	On
TWT A BCLR (Beacon left-right)	No drive	No drive	H	No drive	H	No drive	No drive	No drive	H
Power subsystem									
Power conditioning unit, W	18.2	18.2	20.1	18.2	18.2	20.3	25.8	26.2	26.7
Shunt dissipation,[a] W	107.1	111.8	130.1	97.8	90.1	79.1	72.3	121.1	115.3

H—simulation heater; SS—summer solstice; WS—winter solstice; AE—autumnal equinox; VE—vernal equinox.

[a] 24 h average value.

Table 6 Degradation effect derived by direct comparison

	Average temperature difference, ΔT, °C	σ, °C	Spacecraft dissipation difference, W	Shunt dissipation difference, W	Apparent degradation, °C
May 1979-80	3.48	2.08	−11.4	−30.8	5.78
Summer solstice 1979-80	2.63	1.5	−0.7	21.5	1.99
Autumnal equinox 1979-80	4.20	2.03	−1.6	42.6	2.96
Vernal equinox 1979-80	3.29	2.05	−2.7	37.8	2.33
Autumal equinox 1979-vernal equinox 1980	4.83	2.2	0.4	35.8	3.60

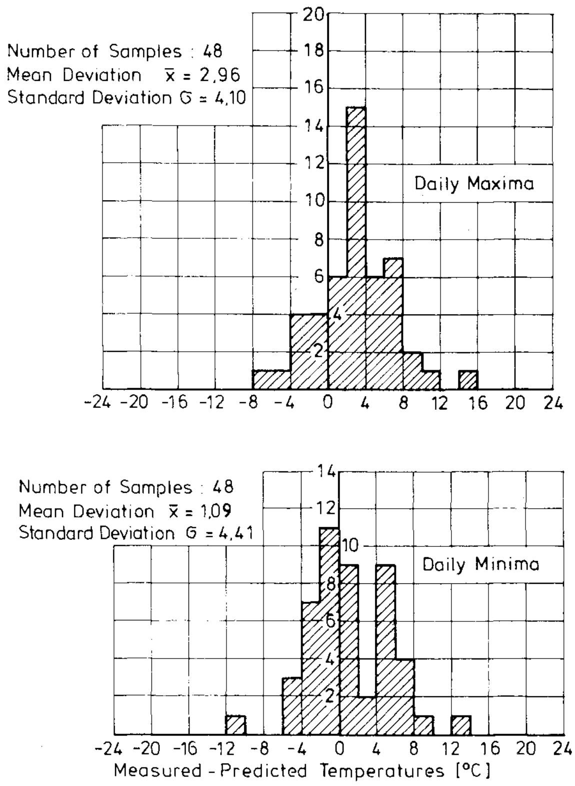


Fig. 4 Histograms for summer solstice 1978.

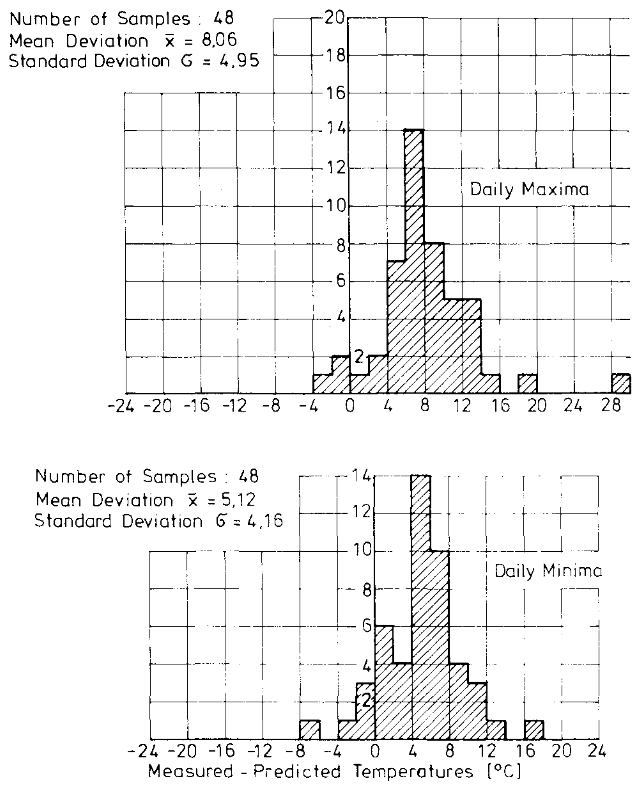


Fig. 5 Histograms for summer solstice 1980.

dissipations (somewhere between 40 and 145 W) because of the aging of the solar array, even if the overall spacecraft consumption is constant. As the shunt radiator is not completely decoupled from the spacecraft, its power variation influences the temperature level. Also the dissipation in the operating TWT is not always identical but depends on the traffic demand. Nevertheless, some tests can be compared directly and an average temperature difference of the spacecraft due to degradation can be derived.

The temperature sensors taken for this global assessment are carefully selected so that units which suffer from strong arbitrary variations in dissipation or which do not reach a thermal equilibrium during the 48 h of test are eliminated. Measurements obtained from outside the spacecraft (such as from sensors on the arrays, antenna, thrusters, and AKM) are also excluded. Thus 48 of the sensors are useful for this purpose. For these, the average deviation between comparable tests is determined together with a standard deviation which is a measure of the accuracy of the result. This average value is corrected for the known influence of dissipation difference via the spacecraft sensitivity of 1°C/9 W of internal dissipation and 1°C/15 W of shunt dissipation. Results of comparable seasons are tabulated in Table 6, indicating a net rise in both equinoxes and solstices. A discussion of these effects will follow in the next section after presentation of the results of the computer-aided comparison.

Computer-Aided Comparison

By performing thermal analysis for the exact operational condition of an in-orbit test, one can define a difference between the test and the "beginning of life" analysis results. Interpretation of this difference as a function of time in orbit can then lead to the determination of the temperature effect of the degradation of the thermal control materials. The results of the computer-aided comparison are interpreted statistically using histograms constructed by plotting the temperature difference between the observed flight temperatures and the predicted temperatures of the same 48 sensors used in the previous section.

The operating conditions implemented in the analysis are determined from orbit measurements as far as available. For each piece of equipment, the power dissipation known from ground testing is entered into the mathematical model. (See table 4, column 3.)

Typical histograms are reproduced in Figs. 4 and 5 for the daily maximum and minimum of the first dedicated test of the 1978 summer solstice and the most recent one of 1980. Some comments should be made on these histograms. The spread of each histogram quantified by the standard deviation is reasonable for a three axis stabilized satellite as mentioned earlier. One has to remember that what is represented by the histograms is a difference with respect to a beginning-of-life (BOL) prediction and does not indicate that equipment 1 or 2

σ away from the mean temperature is outside the acceptable operating limits.

The histograms are a good tool with which to obtain the variation of the mean temperature as a function of time spent in orbit. Interpretation at solstice is performed with diurnal temperature maxima and minima independent of the time at which they occur. It had been found that comparison at a specific orbital time is in general better than comparison using the maxima and minima. At equinox, the interpretation is performed with temperature maxima occurring during the sun-illuminated portion of the orbit and at the specific orbital times of the beginning and end of an eclipse.

The mean deviation between test and analysis is displayed over the time in orbit in Fig. 6. It shows the temperature increase after two years in orbit already observed by direct comparison. The temperature increase during equinox with four points of data seems to be of the order of 3°C for the first 18 months in orbit. In both solstices, summer and winter, as well as in May the temperature increase is significantly larger than in equinoxes.

As the degradation must reach a finite value after prolonged exposure, it has been attempted to fit an exponential decay function of the type

$$\Delta T = A\,(1 - e^{-B(t-t_0)})$$

by least square fit to the available data. Unfortunately the number of data points on the time scale is still too low to achieve mathematically precise results and indeed the functions obtained are physically unreasonable in that they give a

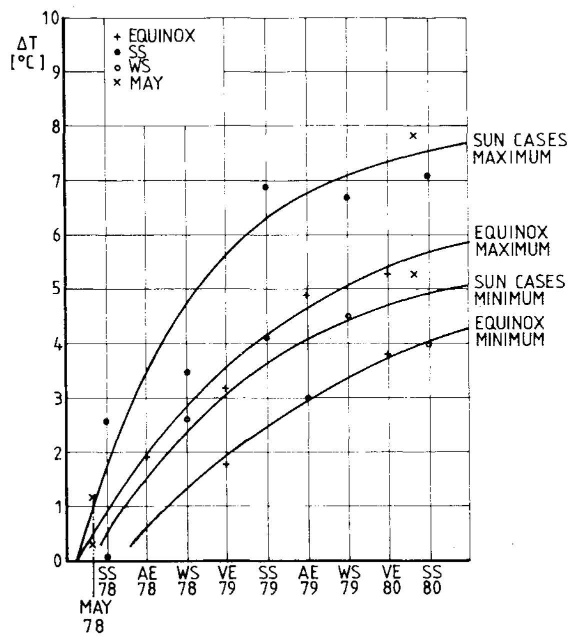


Fig. 6 Orbital mean temperature increase.

crossover of the degradation in equinox and solstice. Further test points have to be awaited before a degradation prediction of this type may be successful.

Interpretation of Results

When interpreting the results one has to bear in mind that over and above the usual uncertainties in the mathematical model an uncertainty equivalent to $\pm 1°C$ is introduced by the inaccuracy of the power dissipation as outlined above and an additional uncertainty of approximately 0.4°C from temperature recording. Nevertheless, the following interpretation uses the results "as is."

The May 1978 comparison of orbit and model data demonstrate that the present model is very well set to calculate orbit temperatures and should be a good tool to determine temperature changes with mission time. From a comparison of identical seasons, the following observations can be made.

The best statistic is available on the equinox where four comparable tests have been performed. They indicate a continuous temperature rise with only very slowly decreasing slope. Presuming that degradation manifests itself in an increase in absorbed solar energy, then the surfaces which participate in degradation during equinox periods are multilayer insulation (MLI), painted antennas, the spacecraft adapter, and some small Earth- and sun-sensor apertures. In solstices the same areas are illuminated plus the north and south surfaces, respectively, which are covered by rigid SSM and MLI. Interpretation of the observed spacecraft degradation by reference to the degradation of individual surfaces as known from ground tests has been attempted. Table 7 lists the degradation determined for OTS in ground tests for a three-year geostationary mission. By implementation of these degraded surface characteristics in the analytical model, the sensitivity of the OTS temperature level has been determined.[9]

Even allowing for some marginal additional degradation due to the small sensor apertures, the temperature increase in equinox is larger than can be explained by the ground test results. No degradation mechanism other than the absorptivity increase can be deduced from the orbit data, because such effects as loss of crinkling or evaporation of the MLI aluminum layer due to electrostatic discharges through the blankets would generate not only an increase in absorbed heat but also a larger heat leak at the cold side which would significantly widen the diurnal variation and ultimately decrease the temperature level rather than raise it. The small increase in the diurnal temperature variation in Fig. 6, which

Table 7 Design baseline degradation values for 3 yr mission, $\Delta\alpha$

Antenna paint	0.27
Aluminized Teflon	0.12
Aluminized Kapton	0.10
Quartz SSM	0.03

Table 8 Diurnal temperature variation of some components

	Equinox			
Unit name	Autumnal 1978	Vernal 1979	Autumnal 1979	Vernal 1980
uhf TTC Rx (uhf telecommand receiver)	4.3	4.5	4.7	4.9
EPC-BCRL (electronic power conditioner for Deacon right-left) $+z$ floor	6.6	6.8	7.3	—[a]
	9.3	9.6	10.0	11.1
WDE (wheel drive electronic)	12.3	12.5	14.0	14.2
BMCU (battery monitoring and control unit)	13.7	14.0	14.1	13.0

[a] Changed operation mode.

is more pronounced for individual units located close to insulated walls (Table 8), also supports this hypothesis.

The two May tests show that there is additional degradation caused by the SSM and MLI surfaces which are now illuminated. From the three summer solstice tests it is evident that the first year accounts for much more of this degradation than the second one. The two winter solstice cases confirm this by their ΔT which lies in between that of the first and second year between summer solstices.

The sensitivity analysis[9] showed that the MLI degradation on the north/south side would account for approximately 0.3°C average temperature level increase after two years in orbit. The remaining difference points to a possible degradation of the SSMs through first year until the 1979 summer solstice of about $\Delta\alpha = 0.051$ and through two years until that of 1980 of about $\Delta\alpha = 0.063$, which is significantly above ground test results.[1] It must be borne in mind that the ground tests have been performed in a clean facility free of any contamination sources other than the SSM and their adhesive. In this context it is also interesting to note that between the first thermal test on May 25-26, 1978, and the first solstice less than one month later there appears to be a significant increase in the deviation. This might be taken as an indication of a fast degradation of the spacecraft surfaces at the beginning of the mission. Unfortunately, none of the temperature sensors is sufficiently decoupled from the spacecraft to permit the observation of SSM degradation from launch through the first weeks of the mission.

The exact cause of the higher than tested SSM degradation is not known but it could be:

1) Pure degradation of the SSM. The space environment could generate higher degradation than ground tests.

2) Contamination of the SSM with subsequent degradation of the deposit. This contamination could come from the outgassing of spacecraft materials or from contaminants collected prior to launch.

3) Contamination as above but stemming from expendables, i.e., the AKM and/or attitude control thrusters.

4) Damage caused by electrostatic discharges. In ground tests destruction of both the quartz surface and the silver layer of the mirrors has been produced under electron charging and arc discharge.[3]

Some arguments can be presented with these potential causes, but a conclusion is not possible on the basis of the available data. Pure degradation and contamination from thermal control materials such as paints and adhesives are considered unlikely because the thermal design authority for OTS has been used on other spacecraft (e.g., HELIOS[10]) the same low-outgassing materials in an environment which is—at least for solar ultraviolet and high-energy particles—more severe than the geostationary orbit. Degradation on those spacecraft which had no AKM and used cold-gas attitude control thrusters was close to the ground test results. The ground contamination hazard has been controlled by manufacturing the thermal subsystem and integrating the spacecraft under clean conditions (class 100000, Federal Standard 209B) and by isopropyl-alcohol cleaning of the radiator surfaces just prior to closing the launcher fairing. The possibility of contamination by expendables, in particular from the AKM, is supported by the steep rise in degradation observed in the first short period of the mission. This is also assumed to be the cause for the degradation of many American spacecraft, both geostationary and Earth orbiting.[2,11] Although the mirrors are of the nonconductive quartz type, severe electrostatic discharges are unlikely because no effect in the spacecraft which could be traced back to electrical interference of an arc discharge has been observed in the first two years.

Individual Assessment of Degradation Effects

Given the uncertainty of the cause as well as of the amount of SSM degradation and the relatively high importance of this material to the temperature increase of the spacecraft, the investigations on individual components have been thus far largely concentrated on further evidence of SSM degradation.

There was no degradation experiment flown on OTS nor are sensors on the spacecraft surface available which are sufficiently decoupled from the interior to permit a straightforward allocation of temperature timelines to surface degradation. Four sensors are located on the SSM radiators in areas in which the major thermal influence is solar heat input where local spacecraft dissipation is well defined. Using a thermal analytical model of the area local to flight thermistors, the following method has been applied[12] to determine degradation:

1) Derivation of mean internal spacecraft temperature local to the sensor location from flight data for the seasons to be compared.

2) By variation of the mean internal temperature in the analytical model, obtain identical (test/analysis) radiator temperatures for the first tests in 1978.

3) Applying the annual temperature differential obtained in step 1, calculate the mean environment to be used in the mathematical model for the identical 1979 and 1980 seasons. (It is thought that due to the few temperature sensors in the environment of the SSM radiator, the absolute internal spacecraft temperature recorded may not be meaningful, whereas the difference between successive years is reliable.)

4) Adjust SSM absorptance in the mathematical model so that with the environments determined in step 3 identical (test/analysis) radiator temperatures are obtained.

The required adjustment of absorptance then indicates the SSM degradation. Results obtained are shown in Table 9. The method (steps 1-3) has been checked by application in subsequent years to the nonilluminated radiator and a very good predictability of the radiator temperature by means of its environment has been found (±0.3°C). Nevertheless, this possible error and the inaccuracy of the digital temperature transmission account for an uncertainty of at least ±0.013 in the absorptance changes given in Table 9.

These results for three seasons, where the May and summer solstice used two sensors on the north radiator and the winter solstice used two other sensors on the south radiator, agree to each other very well. They confirm the total degradation determined from the global spacecraft analyses presented here for two years in orbit. They also indicate a clear reduction of degradation in the second year, which supports the results of Fig. 6 and Table 6.

Further clarification of the contamination influence was expected from the TWT radiators which always run about

Table 9 Degradation of control radiator SSM

Control radiator quadrant	Season	Degradation, $\Delta\alpha$ 1978-79	1979-80	1978-80
$-y, -x$	May 25/26	0.047	0.024	0.073
$-y, +x$	May 25/26	0.044	0.028	0.074
$-y, -x$	Summer solstice	0.041	0.024	0.066
$-y, +x$	Summer solstice	0.036	0.024	0.061
$+y, +x$	Winter solstice	0.044		
$+y, +x$	Winter solstice	0.042		

35 °C hotter than the spacecraft radiator and therefore should have a different degradation. This is because volatile products will condense more on cold surfaces.

The method used to determine the effects of SSM degradation of TWT radiator SSMs was similar in concept to that used in control radiator analysis. However, as only one thermistor is mounted on that radiator and the radiator has a complicated structure profile, a quite detailed model had to be constructed in order to obtain meaningful results. A summary of the results is annual SSM degradation, 1979-1980 (summer solstice): ~ 0.018; biennial SSM degradation, 1978-1980 (May) ~ 0.051. From these results it is possible to conclude that annual degradation is less in the TWT radiators compared to the control radiators, although it must be noted that the difference is within the uncertainty of the results.

Conclusion

After two years in orbit, valuable information has been collected in 10 thermal tests with respect to the thermal performance of OTS which can be summarized as follows:

1) The initial performance of the thermal subsystem was as close to predictions as can be expected with a passive thermal subsystem.

2) As observed in the case of many other geosynchronous spacecraft, OTS experienced a higher than predicted temperature rise shortly after launch. This higher temperature level may be explained by a sudden increase in absorptance due to early outgassing, AKM efflux, or a similar cause, but no definite conclusion is permitted from the data base available.

3) The equinox data show a significant increase in temperature over a period of 18 months, a rise that exhibits a slow tendency to taper off. This behavior is related to a degradation of the coatings on the surfaces rotating with respect to the sun vector (white paint on the antennas, outer Kapton layer of the MLI, spacecraft adapter, etc.).

4) Between successive solstice seasons, the increase in temperature has been larger than expected but it tapers off in the second year; this seems to be the result of increase in absorptance of the SSM, which is higher than obtained in ground tests and from flight experience but similar in magnitude to values reported by other sources. The exact reason for this degradation is not yet understood, but it is likely that it is caused by some contamination generated at the beginning of the mission.

The increase in temperature experienced during the first two years in orbit of the OTS spacecraft is expected to continue, but with a decreasing slope. The operation of OTS, which has been very successful, is now proposed to be continued beyond its original goal of three years in orbit, which will give us more insight and information about longer term degradation effects on the surface materials and the technology employed. OTS has already shown, and will continue to prove, that European technology is ready to support the future telecommunication program.

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