17. END OF MISSION ACTIVITIES AND CONCLUSIONS

Satellite operations were continued under increasingly difficult technical circumstances, in the face of solar array and power degradation, gyroscope and thermal control failures, all attributable to the harsh radiation environment in which the Hipparcos satellite was destined to carry out its measurements. Nevertheless, operations were maintained for a total duration of four years, longer than the nominal 2.5-year operational period intended before launch, and long enough for all of the scientific mission goals to be significantly surpassed. Operations were terminated on 15 August 1993, after repeated attempts to communicate with the satellite on-board computer were finally abandoned. The end of the satellite operations was accompanied by a series of diagnostic tests undertaken to establish the final performances of several of the spacecraft and payload subsystems. An assessment of the overall satellite reliability, the data archiving policy, and some considerations of the mission in retrospect are presented.

17.1. End-of-Life History

On 24 June 1993, the central on-board software halted. Subsequent attempts to reload prime and redundant computers failed. Without the central on-board software, no scientific data could be obtained. Investigations indicated that the fault lay with a failed opto-coupler in the on-board computer. These components are radiation sensitive and may have been the cause of the earlier demise of on-board computer 2. It was decided to cool the spacecraft to improve the performance of the opto-couplers.

On 8 July 1993, on-board computer 1 accepted a run command and restarted. Over the next few days, the entire central on-board software was reloaded, although the command failure rate remained high. Once the central on-board software was reloaded and started, investigation began into reducing the high command rejection rate. The problem was traced to the checksum packet at the end of each command being corrupted. It was then possible to load some parameters by slightly modifying the values, thereby changing the checksum. The payload was re-heated to allow ground real-time attitude determination to be performed and to attempt real-time attitude determination initialisation. This was partially successful with real-time attitude determination remaining converged for one hour in one period. However, commanding was still failing frequently, making initialisation and maintenance of real-time attitude determination convergence impossible for most of the time.

It was realised that a central on-board software patch would be required to provide robust enough commanding for a reasonable science return. Before this possibility had been properly assessed, however, a further deterioration in the on-board computer became apparent, when the last checksum packets were rejected without exception, making any further commanding of that type impossible. A swap of central data units was performed, without improvement. This required that on-board computer 1 would have to be restarted. This was not possible.

No solution was found to work around the failure of the two on-board computers, and the decision was taken by the ESA Project Manager and the ESOC Spacecraft Operations Manager to terminate operations. Before shut down, certain end-of-life tests were carried out, and subsequently the spacecraft was spun up to 10 rpm—there being insufficient fuel to de-orbit the satellite or to change the perigee height significantly. At mission termination all end-of-life tests had been completed and the payload was turned off. The attitude and control system remained on, and the power and data handling sub-systems were nominal. No fuel remained in one tank, while the other had 20 bars (0.75 kg of cold gas) remaining.

Operations of the Hipparcos satellite ceased on 15 August 1993 at 16:00 UTC, when the transmitter was switched off for the last time.

17.2. End-of-Life Tests

In the last two weeks of operations, the satellite was subject to a series of engineering tests. These tests were used to assess the conditions of the various subsystems after the four year period in the revised transfer orbit with its regular passages through the van Allen belts. The tests involved many on-board re-configurations in order to verify units which had never been used before, or units that had already failed in the past and had been substituted by the back-up ones. All test were carried out via mode 3 memory load commands since packet commands were not usable due to the on-board computer failure.

The following sections describe the tests that were carried out, ordered by subsystem. Unless specifically noted otherwise, the tests carried out were terminated after the stated number of cycles (rather than failing at that point).

Payload

Image Dissector Tube 1: the status of image dissector tube 1 was last verified on 29 July 1993 using the internal star pattern assembly. Image dissector tube 1 was still working, but both the electronics and the high voltage power supply voltage readouts were not correct, probably due to a faulty input multiplexer on the payload remote terminal unit.

Image Dissector Tube 1 Chromaticity Filter: the image dissector tube 1 chromaticity filter underwent a destructive test starting on 12 August 1993. It was commanded to the calibration position and then to the wide band again (using a purpose built command sequence) for 288 cycles without any problem. The sequence contained the memory load, reset, and execute commands spaced to allow the verification of each, and the verification of the filter position via the telemetry.

Image Dissector Tube 1 Shutter: the image dissector tube 1 shutter underwent a destructive test starting on 10 August 1993. It was commanded open and then closed again (using a purpose built command sequence) for 244 cycles without any problem. The sequence contained the memory load, the reset, and the execute commands spaced to allow the verification of each, and the verification of the status of the shutter via the telemetry.

Image Dissector Tube 2 Shutter: the image dissector tube 2 shutter underwent a destructive test starting on 4 August 1993. The shutter was commanded open and then closed again (using a purpose-built command sequence) for 60 cycles before the telemetry readout stopped updating. Then, the internal star pattern assembly light sources were turned on to verify the position of the shutter. The sequence contained the memory load, the reset, and the execute commands spaced to allow the verification of each, and the verification of the status of the shutter via the telemetry.

Internal Star Pattern Assembly: the internal star pattern assembly lights underwent a destructive test starting on 13 August 1993. The switching mirror was moved toward image dissector tube 1 thus allowing image dissector tube 2 to detect the light; then the internal star pattern assembly lights were switched on and off again (using a purpose built command sequence) for 720 cycles without any problem. The commands were verified via the light's associated telemetry parameter and via the image dissector tube raw output.

Electro-Magnetic Torquer: the status of the mechanism drive electronics 2 electromagnetic torquer was last verified on 13 August 1993. The test showed that, apart from the star mapper 2 shutter which went to an intermediate position, none of the mechanisms actually worked.

Mechanism Drive Electronics: the status of the mechanism drive electronics 1 and 2 were last verified on 13 August 1993. The test confirmed that the units were still in working condition, although the secondary voltage of mechanism drive electronics 1 had dropped to 1.44 V instead of the 1.88 V registered at the time of its first anomaly.

Refocusing Mechanism: the functioning of the refocusing mechanism was last tested on 11 August 1993. The results of this test confirmed the correct functioning of the refocusing mechanism, but they showed a discrepancy in the maximum and minimum limits reachable by the grid as the calibration curve indicated a minimum of 20 per cent and a maximum of 80 per cent of the allowed range which were not achieved.

Thermal Control Electronics: the status of the thermal control electronics 1 was last verified on 6 August 1993. The results of this test were in line with the previous tests conducted on thermal control electronics 1, and they confirmed that the unit was not working. The status of the thermal control electronics 2 was last verified on 29 July 1993 for each of the 21 actively controlled areas. Of the 24 controlled areas (21 with connected heaters and 3 spare with no heaters) only 4 still functioned.

Star Mapper 2 Shutter: the star mapper 2 shutter underwent a destructive test starting on 14 August 1993. The shutter was commanded open and then closed again (using a purpose built command sequence) for 160 cycles without any problem. The sequence contained the memory load, the reset, and the execute commands spaced to allow the verification of each, and the verification of the status of the shutter via the telemetry.

Radio Frequency

Transmitter 2: the status of transmitter 2 was last verified on 14 August 1993. The results of this test indicated that transmitter 2 was working without any problem, but it also showed a lower on-board automatic gain control reading for receiver 2. This situation did not appear to hinder the commanding function of the satellite, and it had already been observed in the past during ground station checks.

Solar Arrays

Solar Panel Degradation Test: the degradation level of the solar arrays was last verified on 10 August 1993 using the open circuit test. The results of this test indicated that there was no major change in the performance of the solar arrays since the previous tests had been conducted.

Solar Panel Shadowing Test: the solar panel shadowing test was carried out on 12 August 1993, using the modified sun acquisition manoeuvre phase 4 controller. Using the modified sun acquisition manoeuvre phase 4 control loop, the spacecraft was first moved to 55° where its status was checked out; using the same method, the satellite was then moved to 65° where the charge current started to drop, but no discharge was observed. The satellite was finally moved to about 75° where the charge current dropped completely, and the batteries started to discharge. Once the desired solar aspect angle had been reached, the attitude control software was configured to standby mode 2, and the spacecraft was spun up to about 500 arcsec s^{-1}. Shortly afterwards, sun acquisition sensor 3 entered its shadow area, and the spacecraft started to drift while it was trying to re-acquire the Sun; from the output of both sun acquisition sensor 1 and sun acquisition sensor 2 it was seen that the satellite completed a revolution around the x axis. As soon as sun acquisition sensor 3 re-acquired the Sun, an emergency sun reacquisition was initiated from ground, and the satellite was brought back to the Sun. At the end of the test, no noticeable difference was seen in the behaviour of the solar arrays.

Attitude and Orbit Control System

Accelerometer Package and Earth/Sun Sensor: the Earth/Sun sensor 2 and the accelerometer package B, were last tested during the spin-up manoeuvre carried out between 13 and 14 August 1993. The results of this test showed that the units were still working, although it was not possible to fully test the Earth/Sun sensor due to the satellite's sun-pointing position and spin rate.

Control Processing Electronic Cross Strap Configuration: the two control law electronics systems were last tested in a cross-strapped configuration on 12 August 1993. The results of this test verified the feasibility of this unorthodox configuration.

Gyro Oversampling: the gyros outputs were last tested, using the 10 Hz oversampling system (format 3), on 4 and 5 August 1993; the data samples related to the tests were forwarded to ESTEC, Matra Marconi Space and the 'Laboratoire de Recherches Balistiques et Aérodynamiques' for further analysis.

Gyro Heaters: the status of the gyro heaters was verified in two parts on 27 July and 5 August 1993, due to the gyro mechanism electronics temperature problem. The results of this test confirmed the correct functioning of the heaters of gyros 1, 2, 4 and 5 and the failure in the heater of gyro 3.

Latching Valve: the status of the latching valve A1 was last verified on 29 July. The results of this test confirmed the correct functioning of the latching valve.

Spin-up Manoeuvre: the spacecraft was spun-up to its final rotation speed of 10 rpm in several steps starting on 13 August 1993, via direct thruster firing commands. As a first step, the satellite was moved 8° away from the sun-pointing position using the sun acquisition manoeuvre phase 4 controller and direct thruster firing commands on the $-x$ axis. Once the rates had abated to an acceptable level, the first $+z$ direct thruster firing command was sent bringing the spin rate to 0.25 rpm; this first manoeuvre also introduced a nutation of about 9° half-cone which had to be reduced before progressing any further.

After the nutation had been successfully reduced to less than 1° half-cone, a second step was started bringing the spin rate to 1 rpm. A third step brought the satellite to 3.02 rpm, and was immediately followed by another step which should have brought the satellite to a spin rate of about 5 rpm. During this period, in order to prepare for a different test, a series of attitude and orbit control system re-configurations were carried out; unfortunately, one of them caused the manoeuvre to abort prematurely (the attitude and orbit control system telemetry format was changed from high rate to low rate). Once the re-configuration was completed the spin-up was resumed with a fifth step which brought the rotation rate from 3.75 to 5.1 rpm.

Two more steps were carried out, bringing the spin rate to 9.07 rpm. On 14 August, a spin-down was performed to introduce nutation around the z-axis in order to verify the accelerometer package; during this manoeuvre, tank B was emptied, and had to be swapped for tank A. The de-spin was manually stopped at about 8.52 rpm as there was little or no effect on the nutation amplitude. The last step of the spin-up manoeuvre, was then carried out bringing the spin rate to its final value of 10.04 rpm and leaving about 21 bars in tank A.

Data Handling Subsystem

Central Decoding Unit: the status of central decoding unit 2 was last verified on 23 and 24 July 1993. The result of the test indicated that there was a problem with the commanding chain of central decoding unit 2 (mode 3 commands), thus making it completely unusable.

Memory Bank Module: the status of memory bank module 3 was last verified on 23 and 24 July 1993. The results of these tests indicated that memory bank module 3 was working correctly.

Spacecraft Remote Terminal Unit 2: the status of spacecraft remote terminal unit 2 was last verified on 14 August 1993. The results of this test indicated that spacecraft remote terminal unit 2 worked without any problem.

17.3. Satellite Reliability Assessment

A summary study of the reliability of the Hipparcos satellite during its three years of operation in geostationary transfer orbit compared to the expected reliability after three years in the nominal geostationary orbit was undertaken by Matra Marconi Space. The following hypotheses were assumed for the study: (a) the computed anomalies referred to satellite anomalies only, with ground-segment anomalies not taken into account, and (b) the satellite was considered to be one instrument, rather than an association of different subsystems each with their own reliability.

Figure 17.1 represents the evolution of the reliability of the satellite versus time in geostationary orbit (nominal mission), and the geostationary transfer orbit (revised mission). Figure 17.2 represents the evolution of the failure rate versus time in geostationary transfer orbit (revised mission). It is apparent that the reliability decreased quickly and remained very low after three years in geostationary transfer orbit (between 0 and 0.2) compared to the expected reliability (around 0.6) for the nominal mission in geostationary orbit. The failure rate continuously increased.

17.4. Data Archiving Policy

In accordance with the general data archiving policy at ESOC, the raw satellite data will be archived for a period of 10 years. It is, however, considered most unlikely that access to the raw satellite data at ESOC would again be needed. The data processing tasks undertaken by the data reduction teams were so lengthy, and so complex, that a repeat of the Hipparcos data analysis starting from the raw data will almost certainly never be embarked upon. Intermediate data archives generated by the data reduction teams would be the natural starting point for further analysis, should this ever be considered appropriate.

17.5. Miscellaneous Considerations

Design and Development of the Satellite

Concerning the relationship between the instrument design, manufacture and operation on the one hand, and the data analysis effort on the other, several features of the Hipparcos mission implementation contributed to the mission's success:

(i) acceptance of the mission within the ESA scientific programme was only undertaken after a very thorough scientific and technical evaluation of the end-to-end system (up to the final product) had been made;

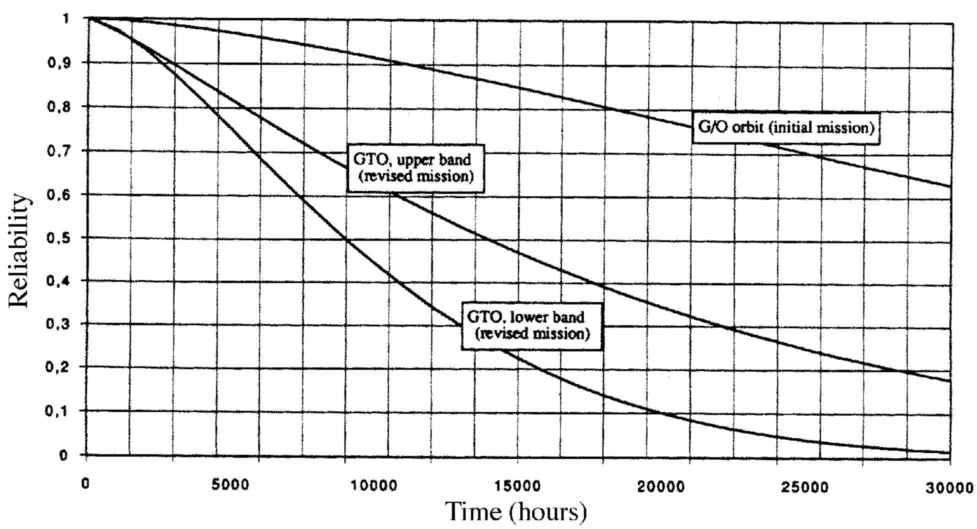

Figure 17.1. *Evolution of the nominal reliability of the satellite as anticipated in geostationary orbit (top curve), and as inferred in the geostationary transfer orbit.*

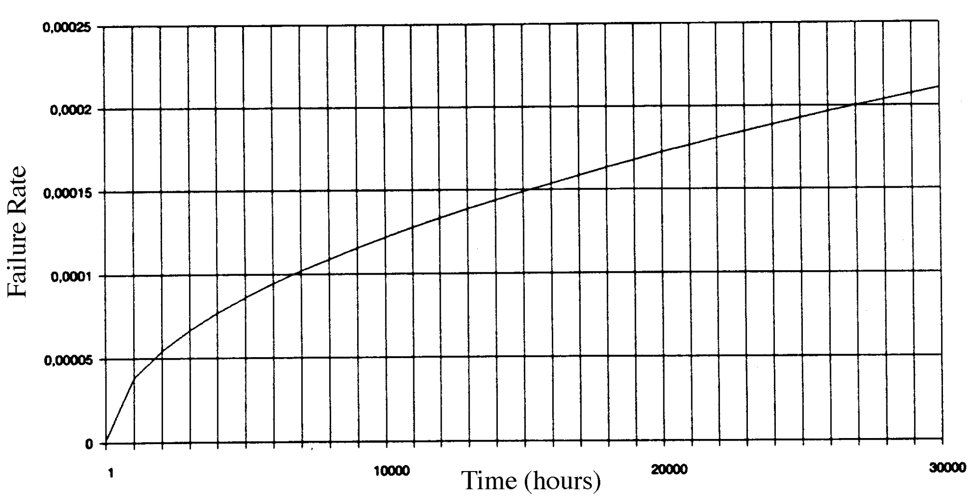

Figure 17.2. *Failure rate, expressed as the probability of a satellite failure occurring per hour, versus time.*

(ii) sole responsibility for the total success of the spacecraft and payload was entrusted to a single prime contractor, who placed great emphasis on the overall (system) error budgets;

(iii) the observing programme definition and data analysis preparations took place in parallel with the satellite design and manufacture;

(iv) for about a year during the early satellite design phase, the industrial prime contractor employed the services of a Science Advisory Group, comprising some of the same scientists that advised ESA through their participation in the Hipparcos Science Team;

(v) a single scientific advisory group with responsibility for all scientific aspects of the mission (the Hipparcos Science Team), and an ESA project scientist with overall responsibility for all scientific aspects of the project from the start of the mission to the final catalogue publication, were important elements in the proper functioning of this very complex mission.

Some of these aspects are not 'standard' ESA policy. Thus while the Hipparcos spacecraft and payload were constructed and calibrated by a single prime contractor, ESA is more typically requested to provide experimental opportunities on specified platforms, with the scientific space research laboratories building the scientific instruments.

It may have appeared superficially questionable for both customer and manufacturer to be advised by the same individuals. That this structure worked may be attributed to the fact that the scientists and engineers concerned, whether employed by the prime contractor, as part of ESA's Project Team, or through scientific institutes as members of the Hipparcos Science Team, were capable of pursuing the difficult challenges of the mission in a collegial atmosphere oriented towards success. This atmosphere tended to dominate the very few incidents where commercial or other aspects surfaced.

A constructive rivalry was present throughout the mission between the scientific consortia, and it engendered a positive collaborative spirit that has been with the Hipparcos mission since its inception. The mission has also maintained a persistent level of independent parallel dedicated effort, conducted in a manner that may well have differed from that encountered in missions where the Agency leaves the scientific instruments to the respective institutes and the platform to the contractor, finding itself in the middle with the task of integration.

Satellite Margins

With the final results in the hostile geostationary transfer orbit being basically superior to those targeted at the time of the project's acceptance by ESA, it is natural to reflect on what the final astrometric results would have been had the correct orbit been achieved. At the same time the question of whether the mission was 'over specified', with corresponding cost implications, should be posed. Although it seems evident in retrospect to conclude that some specifications could have been relaxed while still meeting the target mission goals in the nominal orbit, the full answer to these questions is not so trivial. The satellite would undoubtedly have survived for significantly longer, providing high quality data for perhaps 1.5 – 2 times as long as it did. However that fact was not fully apparent during the satellite design phase: the consumption rate of the cold-gas in particular was the most obvious life limiting factor, budgeted by the prime contractor and installed with a corresponding safety margin. Performing the analysis of this budget to the level required to make a significantly more precise cold-gas budget would very soon surpass in cost the savings that might be made on cold-gas mass. In particular, improvements beyond the already very sophisticated attitude control strategy were not foreseen, and would have required an in-depth study of ideas like gyroless operations, before creative hopes could have been turned into reality.

There was actually one significant concern that was eliminated by the failure of the apogee motor: the contribution to the pre-launch error budget arising from a potentially significant jitter source through the interior 'bouncing' of the remaining propellant flakes after burnout. Design efforts had been aimed at making the power spectrum of the jitter as weak as possible at the higher frequencies, and the apogee boost motor residuals were the most significant remaining jitter concern. For the same reason the hot-gas (hydrazine) system for orbit manoeuvres was replaced by the cold gas for the operational phase of the mission. The cold-gas thrusters were specifically designed for the low thrusts required, again basically to minimise their impulsive character. Even now it is not clear how much the apogee motor residuals would have affected the measurements. Undoubtedly, the absence of the boost motor mass, and the presence of the residuals, would also have affected the cold-gas consumption. The benefits of a smooth attitude motion, free from such residuals, had been anticipated, but these could never be guaranteed.

In most other respects any 'over specification' can only be identified in retrospect. The photon noise budget, for example, took into account the radiation darkening of the optical elements, allowing for a certain number of solar events during the period of solar maximum which could never be properly predicted before launch. Indeed, for most identified error sources, a small margin was normally demanded, and the cumulative effects proved significant, outweighing the negative effects of a reduction of observing time due to the lack of full orbital coverage.

The mission operational life had been specified as 2.5 years, and the survival for almost 4 years in this harsh environment is certainly more than might have been hoped. Yet a specification of mission survivability for at least 3 years is not the same as specifying its demise after 3.1 years, as evident from experience with other missions such as IUE. It was also clear from the beginning that a longer mission duration would lead to improvements in the proper motions according to time to the power of 1.5, rather than with time to the power of 0.5 as for the parallaxes and positions. These facts may well have provided at least the scientific advisors with an incentive to optimise the margins affecting the mission duration, but only to convincing levels for the prime contractor and the ESA Project Team.

The fact that the mission, even in its adverse orbit, fully surpassed its original goals is basically due to its impressive survival of the harsh conditions, with an unprecedented creativity of the operations team in overcoming increasingly failing hardware. Thus in the end the mission provided as much data as foreseen, but stretched over a somewhat longer time.

Orbit and Perturbing Torques

The successful operation of the Hipparcos satellite in its geostationary transfer orbit provided an unprecedented experiment providing insight into operational possibilities for scientific missions in this harsh environment. It gave information about the radiation conditions at solar maximum in considerable detail, and also gave details of the decay of semiconductor-based systems with inadequate shielding. The radiation-induced deterioration of the optics, including induced stresses deforming the highly accurate mirrors, was modelled in detail.

The carefully designed thermal control subsystem performed well within the revised mission, avoiding significant deterioration of the science data. Also the detailed description of the disturbing torques, such as due to the magnetic field and gravity gradient torques, will facilitate the future design of attitude control systems for very demanding missions in such orbits. In order to avoid the atmospheric drag, which is very difficult to model, the minimum perigee height should be around 550 km.

Many other aspects of the satellite functioned flawlessly, including the image dissector tube and photomultiplier (star mapper) detectors, and their associated electronics.

The mass implications for going to geostationary orbit instead of operating in the transfer orbit are very significant, amounting to almost 50 per cent. It may be that the dedicated design of a mission and its associated hardware might be optimised for such an orbit, with resulting benefits even when taking into consideration the significant additional operational constraints. Mission planning with more forward uplink, supporting significant periods of loss of direct satellite contact, could minimise operational implications in a manner which was impossible for Hipparcos. Improved uplink and downlink storage capabilities on board, and increased uplink and downlink budgets during periods of visibility, may evolve in the future to make that less of a drawback.

A future astrometric mission aiming at improved accuracies of tens of microarcsec will face as large an advance in terms of precision as has been achieved with Hipparcos. It is not at all evident that such goals could be achieved with 'only' a Hipparcos-type understanding of the hardware and mission analysis. The demands of such a mission would evidently not tolerate a geostationary transfer orbit operation, and a geostationary or L2 orbit would be vastly preferable, given the environmental stability. A solid boost motor would not be employed, although this choice for Hipparcos was determined by its record for dependability as well as cost. In retrospect it might well have been a better choice to have included a larger supply of hydrazine; even the rather low thrust of the hydrazine thrusters was good enough for the perigee raise manoeuvre in only a few orbits. The high thrust of the original solid fuel boost motor also resulted in certain design constraints: acceleration forces of more than 6*g*, and the potential jitter source of the motor residuals referred to above. A lower thrust propellant would demand a longer period required to reach the final orbit, but already with the solar generators deployed.

Another issue that will merit a more detailed analysis for a future high-accuracy astrometric mission is operation with the gyroscopes turned off. A rotating spacecraft which 'observes' its own attitude with high accuracy is potentially a better gyro than almost any 'regular' high accuracy gyroscopes system. The performance would mainly depend on the maximum unmodelled angular accelerations, i.e. on external disturbing torques and moments of inertia. These contributions can be kept small enough when the satellite is kept well away from the Earth surface, but become too complicated to handle in a geostationary transfer orbit.

While clearly required during the early orbit and station acquisition phases, gyroscopes present two drawbacks thereafter: residual torques (depending on the configuration of the active gyros), consuming cold gas or other attitude actuations, and moving parts, at undesirable angular frequencies which may generate jitter in the measurements. For a future mission targeting two orders of magnitude higher precision such concerns will need to be carefully addressed.

Understanding the Hipparcos Payload

A driving concern during the design and construction of the satellite and payload was always that of some unforeseen and 'indescribable' effect appearing in the satellite or in its data stream. In engineering terms, the mission's success centred around the concept of a measurement apparatus, mechanically defining a measurement reference (albeit with an optical readout, to about 1 nm in a device of 1.5 metre in size). This concern resulted in the satellite and payload specifications being formulated in very fine detail (reflected in the 'Invitation to Tender' prepared by ESA for industry in 1981). While all significant effects were indeed duly accounted for, at least three smaller unforeseen effects became apparent in the data. Although these were so well 'documented' in the measurements themselves, and facilities existed in either the on-board or on-ground data analysis, or on-board hardware, to correct for them, their unforeseen appearance should be noted:

(i) differential grid rotation: this had not been anticipated, although the software for both data reduction consortia were very easily able to accommodate the unforeseen effect of different rotation angles of the main grid about the optical axis for the two different fields of view;

(ii) (differential) flexure of mirrors due to radiation exposure. The effect is described in Chapter 10. In addition to a common linear component, an additional differential effect was caused by non-uniform shielding of the back of the folding flat mirror, and caused both a differential evolution of the focus for the two fields and an evolving discrepant phase difference between the first and the second harmonic of the modulation. Such an effect qualified as one of the effects for which it was mandatory to downlink the raw photon counts, rather than just compressed modulation parameters;

(iii) along-scan attitude jitter due to mechanical relaxation in the solar arrays due to thermal effects after emergence from eclipses (see Chapter 11). In practice, this affected an almost insignificant fraction of the overall data set, which was easily recognised as corrupted, and duly discarded.

With the emphasis of the satellite design being focussed on the accurate control of random and systematic errors, it is noted that two other means of suppressing potential systematic errors were discussed in advance of launch, but never implemented:

(i) the sense of the satellite rotation was not strictly part of the instrumental design, and thus reversing the spin could have helped determine and suppress errors affected by the spin sense, or by the sequence of observations resulting from a particular spin sense. This was considered (but never placed as a requirement) during the mission design and planning, but was abandoned around 1986 for reasons of induced additional complexity in the on-board and on-ground software. In practice, no effect that could only be traced by spin reversal had been identified, and even in retrospect, no parameter affecting the final results has been identified that could have been (better) determined by means of spin reversal;

(ii) another feature that might have been exploited more rigorously was the *independent* use of the first and the second harmonics of the modulation by the main grid at the level of the great circle-reductions and the sphere solution. These measurements almost correspond to independent instruments, the discussion of which might well have contributed to the detailed understanding of the payload. That little, if any, astrometric weight was lost using the adopted data reduction procedures is, however, evident.

On-Ground Versus In-Orbit Calibration

A sizeable ground-based calibration programme was carried out for Hipparcos, in common with all space programmes, and these revealed certain payload problems, most notably a problem of vignetting in one of the fields of view due to misalignment. Thermal cycling was also demonstrated to leave hysteresis in the glue originally selected for the mounting pads, a problem avoided by an alternative choice of glue, and resulting in the spare flat folding mirror becoming the flight model. Although effective in putting a well adjusted and properly verified payload together, the on-ground calibration thereafter strictly provided only starting values for the parameters to be evaluated in flight. Without exception the in-orbit calibration proved more extensive than what had been achieved on ground: in particular improved photometric and spectral responses were derived in orbit. These in-orbit improvements were not surprising, and even modifications to certain of the calibration parameters had been foreseen—thus it was considered unlikely that the payload would retain its calibrated parameters through the heavy load phase of launch and early orbit manoeuvre, and moisture outgassing was expected to lead to an evolution of the best focus position after launch.

The only on-ground calibration that was not fully re-performed in flight, although it was subject to considerable verification using the in-flight data, was the medium-scale grid calibration. The verifications carried out confirmed the details of the on-ground calibration, and proved the grid to be of very high quality. Any remaining grid-induced errors were not capable of even marginally acting as an additional 'noise source' in the data.

On-Board Data Compression

At the time of the mission design the downlink budget, amongst other things, was driven by a requirement to avoid significant data processing on board: this was considered to be too high a risk for such an advance in measurement precision in an area that had been hampered historically by significant systematic errors. Any data reduction on board (beyond the actual coding of the photon counts as used for Hipparcos) would implicitly have imposed some model into the data. 'Undoing' such a model on-ground would have been impossible, and this risk far outweighed other considerations such as the consequent demands on the downlink budget, both in terms of photon counts (the dominant contribution to the telemetry), as well as in many instrumental parameters, such as the on-board thermistors.

Subsequent experience of the Hipparcos data analysis has confirmed that the end result of the complex data reductions has indeed led to an understanding and modelling of the missions measurements, the payload mechanics, the optics, and the detection systems, and that these may be considered to have been essentially 'perfectly' understood, resulting in photon noise largely dominating the final error source. That experience was guided throughout the scientific data reductions by the detailed comparisons between the two reduction consortia. It seems evident that any errors that would have remained in any (encoded) data analysis algorithms would have seriously weakened this conclusion. While the data could have been downlinked after compression of the photon counts according to Equation 1.1, such a conclusion might not have been reached without the 'luxury' of the raw data, and indeed without the two reduction consortia with basically independent approaches to the data analysis.

17.6. Overall Success of the Hipparcos Mission

Targeted for an operational lifetime of two and a half years, more than three years of high-quality star measurements were eventually accumulated. The Hipparcos mission finally superseded all of its original scientific goals, resulting in milliarcsec astrometric measurements of positions, distances and space motions for more than 100 000 stars, and an additional catalogue of high-precision astrometric and photometric data for more than one million stars.