4. LAUNCH AND REVISED MISSION DEFINITION

Following a perfectly nominal launch of the Hipparcos satellite, by Ariane 4 flight V33, from Kourou on 8 August 1989, the satellite did not reach its intended geostationary orbit due to the failure of the apogee boost motor. The scientific mission was destined to be conducted from the highly elliptical non-nominal geostationary transfer orbit, with only small orbital changes possible (making use of the remaining hydrazine fuel intended for station keeping). The activities necessary to recover an operational mission, and to carry out the first phases of the planned mission operations—sun acquisition, real-time attitude determination, scanning law acquisition, and the start of the payload calibrations—are described. At this stage of the mission, the expected lifetime was highly uncertain, in particular due to the difficulty of predicting the long-term solar array performances in view of the hostile radiation environment in the geostationary transfer orbit.

4.1. Introduction

Hipparcos was launched as the lower passenger by Ariane 4 flight V33 at 23:25:53 UTC on 8 August 1989 from the launch site at Kourou, French Guyana. Hipparcos and the second passenger satellite (TV-SAT2) were injected by the Ariane launch vehicle without problems. The transfer orbit into which the satellite was launched had a period of about 10 hours reaching an apogee of 36 000 km (120 km lower than nominal) and a perigee of 200 km. The nominal injection attitude after launch was $\alpha = 200\overset{\circ}{.}816$, $\delta = -8\overset{\circ}{.}882$.

At first acquisition of signal at the Malindi ground station a spin rate of 6.92 rpm was calculated. The solar aspect angle was $70\overset{\circ}{.}5$, approximately 5° lower than expected. The estimated nutation at injection was about $1\overset{\circ}{.}0$, a value which reduced to $0\overset{\circ}{.}5$ through the first spin-up of the satellite.

On 10 August 1989 at 12:14:00 UTC, the telecommands to initiate the firing of the apogee boost motor were executed from the operations control centre at ESOC. However, no subsequent change in Doppler rate was observed at the supporting ground stations, and it was evident that the apogee boost motor firing had not taken place. The uplinked commands were immediately verified and confirmed to be correct, so the failure was almost certainly on-board the satellite. Further unsuccessful firing attempts were conducted in subsequent orbits.

During the following weeks detailed investigations were performed in order to understand what had happened on-board. A reformatting of the telemetry allowed the sampling of the currents in the pyrotechnic relay unit at intervals of 5 ms. From the data downlinked during subsequent apogee boost motor firing attempts, it was possible to reconstruct the behaviour of the firing relays, and the fault was located to be somewhere in the pyrotechnics chain. A total of five firing attempts (each consisting of multiple firings) were made until 25 August 1989 without any success.

The most optimistic outlook for the mission at the time was that, if a 'hibernation mode' for the first long eclipse season could be devised (involving a reduction in spacecraft heating and deactivation of the payload to save power), there could be a chance of satellite survival until the next eclipses starting in mid-1990. By that time, it was expected that the on-board power generation would be so degraded that meaningful science data could no longer be collected.

Under the latter scenario, the outcome of the mission would have been reduced to something between 5 per cent and 10 per cent of the original mission objectives, but this would have still represented an achievement which would be hard to accomplish with ground-based astronomical observations. As a minimum, it was hoped that these measurements would still be useful in order to validate the Hipparcos mission concept. In reality, the mission lasted much longer.

4.2. Operations Until Revised Mission Implementation

This section summarises the main events which happened, day by day, through the attempts to fire the apogee boost motor, leading up to the decision to terminate further firing attempts, and concentrate on the implementation of a 'revised', or 'recovery', mission.

89-08-08: Launch of Ariane flight V33 at 23:25:53 UTC.

89-08-08: At 23:49:36 UTC, the spacecraft was successfully ejected from the launcher.

89-08-09: The first sun sensor data (at 00:21:00 UTC) registered a spin rate of 6.92 rpm and a solar aspect angle of $70\overset{\circ}{.}5$ (compared with the expected value of $75\overset{\circ}{.}15$). A series of spin-up manoeuvres were carried out to increase the spacecraft spin rate in a number of steps: firstly to 18.57 rpm, then to 62.33 rpm, followed by a correction manoeuvre to achieve the target spin rate of 60 rpm. At 02:52:00 UTC, the ground system derived an attitude (in right ascension and declination) of $\alpha = 204\overset{\circ}{.}9$, $\delta = -10\overset{\circ}{.}1$. The spacecraft was commanded to slew (in two stages) to the required attitude for the apogee boost motor firing attempt at apogee number 4 namely $\alpha = 213\overset{\circ}{.}91$, $\delta = -8\overset{\circ}{.}87$. Between the two slews, a spin trim manoeuvre to 60.13 rpm was performed. These manoeuvres were completed by 01:50:00 UTC on 89-08-10.

89-08-10: The first apogee boost motor firing attempt was performed at 12:14:00 UTC without success. At 21:45:00 UTC, the spacecraft was slewed to the required attitude for a second apogee boost motor firing attempt at apogee number 6, namely $\alpha = 212\overset{\circ}{.}29$, $\delta = -8\overset{\circ}{.}36$.

89-08-11: Three further apogee boost motor firing attempts were performed between 09:10:00 and 09:25:00 UTC, without success. At 10:15:00 UTC, the spacecraft was slewed to the required attitude for a further apogee boost motor firing attempt at apogee number 8, namely $\alpha = 211\overset{\circ}{.}49$, $\delta = -8\overset{\circ}{.}32$. In the event, the decision was later taken not to make a further attempt at apogee 8.

89-08-13: A further apogee boost motor firing attempt was performed at apogee number 11 (13:51:00 UTC), without success. During this attempt, a newly designed high-rate telemetry format (Format 4) was selected to allow oversampling of the apogee boost motor firing ignition current in order to investigate the failure. After the failed attempt, a slew manoeuvre to an 'average' attitude of $\alpha = 214\overset{\circ}{.}40, \delta = -9\overset{\circ}{.}60$ was deemed to be sufficient for subsequent firing attempts.

89-08-14: To reduce the strain on various components caused by the prolonged exposure to high spin rates, a spin-down of the spacecraft from 60 to 30 rpm was performed in two steps. The final attitude was $\alpha = 213\overset{\circ}{.}63$, $\delta = -9\overset{\circ}{.}50$.

89-08-17: Further apogee boost motor firing attempts were performed at apogee number 20 (between 11:55:00 and 12:05:00 UTC), without success. The Format 4 telemetry was used during the attempts to gather extra information.

89-08-18: It was decided that the next firing attempt should be made after heating the apogee boost motor. This was achieved by slewing the spacecraft such that the motor was oriented more toward the Sun (the solar aspect angle was reduced from 115 to 95°). This attitude ($\alpha = 277\overset{\circ}{.}8$, $\delta = -13\overset{\circ}{.}7$) was maintained for 6 days prior to the next apogee boost motor firing attempt.

89-08-19: The fill-in antenna was tested at 11:46 UTC.

89-08-24: The spacecraft was slewed to the apogee boost motor firing attitude ($\alpha = 217\overset{\circ}{.}69$, $\delta = -8\overset{\circ}{.}11$) changing the solar aspect angle from 103° to 113°.

89-08-25: The spin rate of the spacecraft was increased once more from 29.9 to 37.0 rpm at 08:20:00 UTC, prior to the fifth apogee boost motor firing attempt at apogee number 38 at 08:35:00 UTC. The Format 4 telemetry was used during this final attempt in order to gather extra information. One hour after apogee 38, a slew manoeuvre was performed to an attitude of $\alpha = 235\overset{\circ}{.}88$, $\delta = -14\overset{\circ}{.}73$. The attitude achieved meant that the daily drift of the solar aspect angle could be left uncompensated for several days thereafter.

89-08-26: The decision was taken to abandon any hope of successful boost motor firing, and all effort was dedicated to designing and optimising a 'recovery' or 'revised' mission operations scenario.

4.3. Apogee Boost Motor Failure Investigations

An enquiry board was set up in September 1989 to examine all aspects of the apogee boost motor anomaly, establish the most probable cause or causes, and to make appropriate recommendations to prevent re-occurrences.

The enquiry board identified and studied the most likely causes of the failure in the apogee boost motor pyrotechnic subsystem, which were: (a) mechanical failure of the flexible explosive transfer assembly under vibration load; and (b) failure of the 'through-bulkhead initiators' (the pyrotechnic initiators). A characterisation programme to determine the detonator electrical resistance after firing was performed to establish whether the failure was of electrical or pyrotechnical nature. The characterisation programme consisted of manufacturing, testing and firing 40 detonators. Flexible explosive transfer assemblies were vibrated up to the qualification level of 20*g* at resonance, X-rayed and inspected. They did not show any deterioration. Then they were successfully fired. Study of the electrical resistance after the firing was in accordance with the telemetry values proving that the detonators had indeed been fired.

Through-bulkhead initiator tests were thorough and very complex. Drop tests were performed which proved that the through-bulkhead initiators were not damaged by such shocks. After the two through-bulkhead initiator firings performed in 1989, three more from the Hipparcos lot were fired and one of them failed. In total six through-bulkhead initiators from the Hipparcos lot were fired and two failed. However due to the limited number available it was not possible to conclude the direct cause of these failures.

The enquiry board, recognising that both redundant through-bulkhead initiators must have failed, concluded that the boost motor failure was related to the non-functioning of the through-bulkhead initiators.

4.4. Revised Mission Definition

Immediately following the first firing attempts, investigations were initiated to check the possibility of fulfilling at least part of the mission objectives within the given constraints. The prospects were not very promising, since the available hydrazine fuel (intended for station acquisition) would allow the orbit to be raised by not more than a few hundred kilometres: therefore, the satellite would be exposed to the effects of the van Allen radiation belts during each orbit. It was expected that the radiation would degrade the solar arrays to such an extent that the on-board power would not be sufficient for the satellite to survive the long eclipses (of over 100 minutes duration) starting in March 1990 (Figure 4.1).

A concentrated effort of ESOC, ESTEC, and industry experts was devoted to all relevant system, spacecraft, payload and associated operational aspects of a revised mission. At the same time, the Hipparcos Science Team performed investigations into the science implications of the revised mission. Their simulations indicated clearly that the ground station coverage should be increased to the maximum extent: only with close to full time station coverage, would it be possible to establish a full sky network of star measurements within a limited mission lifetime using the nominal attitude scanning law and star observation strategy which were implemented in the on-board software.

The objective of the mission redesign efforts was to recover as much as possible of the original Hipparcos mission objectives. The operational problems to be addressed were essentially those of a new mission with a few significant differences: (i) there was little flexibility left on the satellite system design: only limited on-board software modifications were possible within the practical constraints available; (ii) only narrow margins were available on the orbit selection because of fuel constraints; and finally, (iii) any day spent on the definition of the mission represented a full day of lost mission time.

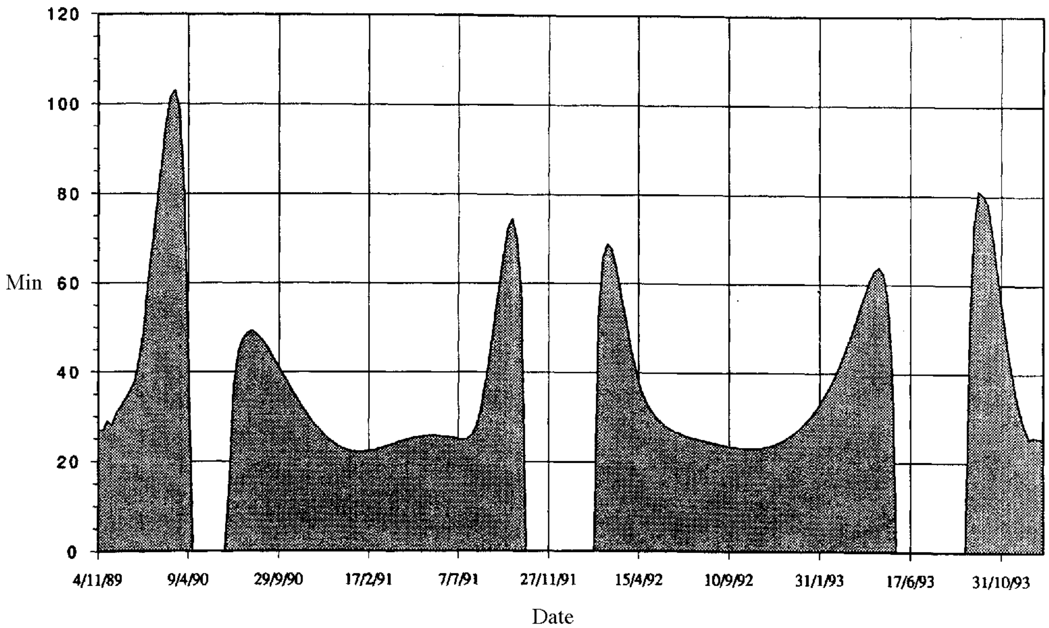

Figure 4.1. *Eclipse duration due to the Earth as a function of time throughout the operational period. The extended eclipse periods in early 1990, late 1991, early 1992, and early 1993 imposed constraints on the available satellite power, which had to be handled by shutting down specific on-board subsystems during some of these intervals.*

A number of problems and uncertainties contributed to generally pessimistic mission prospects at the time:

(i) the limited ground contact from the Hipparcos-dedicated antenna at Odenwald (about 32 per cent of the time) would not be sufficient for significant science data collection;

(ii) the limited station coverage from Odenwald would severely complicate the execution of the operations and endanger the satellite's safety outside the coverage interval, so a number of precautions would have to be implemented before each loss of contact (the satellite design was based on essentially full ground station coverage with occasional interruptions of not more than 0.5 hour);

(iii) the long periods without ground station contact would necessitate frequent initialisations of the autonomous attitude determination process from ground, which would further reduce the time for science data collection and would require a larger operations team;

(iv) the high radiation in the van Allen belts (which were crossed about four times per day) would increase the detectors background noise levels, complicating the autonomous star detection, and inducing a darkening of the payload optics which would degrade the photometric results;

(v) long eclipse periods would occur in March 1990. The associated power problems were expected to require a special and still to be designed 'hibernation mode';

(vi) the occultations caused by the Earth being in the field of view would be significantly longer and more frequent than in the nominal geostationary orbit. This would make the maintenance of the on-board attitude knowledge more difficult;

(vii) the Earth albedo effect would provide a thermal input to the payload in the perigee region. This could lead, for example, to significant distortions in the fine modulating grid of the payload;

(viii) the high perturbing torques in the perigee region would introduce difficulties for the attitude control. In any case, it would require the inhibition of the satellite emergency mode triggering on the basis of too high thruster actuation demands.

The following list summarises a few of the most significant mission design aspects studied at ESOC during the first few weeks after the failure:

(a) evolution of the geostationary transfer orbit and the satellite attitude orientation for subsequent apogee boost motor firing attempts and solar aspect angle violations;

(b) redesign of the attitude acquisition sequence (directly from the attitude required for apogee boost motor firing to the sun-pointing attitude);

(c) station coverage predictions of the transfer orbit and potential recovery orbits;

(d) eclipse and occultation predictions of the potential recovery orbits;

(e) potential implications of modifying the nominal scanning law and the star observation strategy;

(f) fuel budget requirements and predictions under the revised orbit and attitude acquisition strategies;

(g) strategies for covering the long periods without ground contact, and the implementation of a sparse programme star file as a means of survival of the on-board attitude knowledge;

(h) new attitude control parameters designed to cope with the high disturbance torques in the perigee region.

At the same time, ESTEC and contractor experts were evaluating the radiation effects of the van Allen belts on solar array and payload degradation. Thermal and power aspects of the new attitude acquisition strategy, and associated risks, were also being assessed and evaluated.

Immediately following the first apogee boost motor firing failure on 10 August 1989, a team of mission analysis and flight dynamics experts started investigations concerning potential recovery orbits. After it was found that 12-hour or 8-hour orbital periods were not practically feasible, the orbit selection concentrated on achieving a perigee altitude above 400 km taking account of fuel constraints and to reduce the influences of air drag and atomic oxygen with, at the same time, a synchronisation between the scanning law period (2 hrs 8 mins) and the orbital period: this was considered to be beneficial as it would limit the effects of albedo heating on the payload and minimise the duration of payload occultations in the perigee region by a proper tuning of the initial scan angle.

Fortunately, a 10 hour 39 min orbital period fulfilled both criteria, and was within reach of the available fuel resources. The necessary orbit adjustments were performed after implementation, validation and simulation of the required new software and control procedures, in a series of five manoeuvres during the period 7–18 September 1989 (see Figure 4.2).

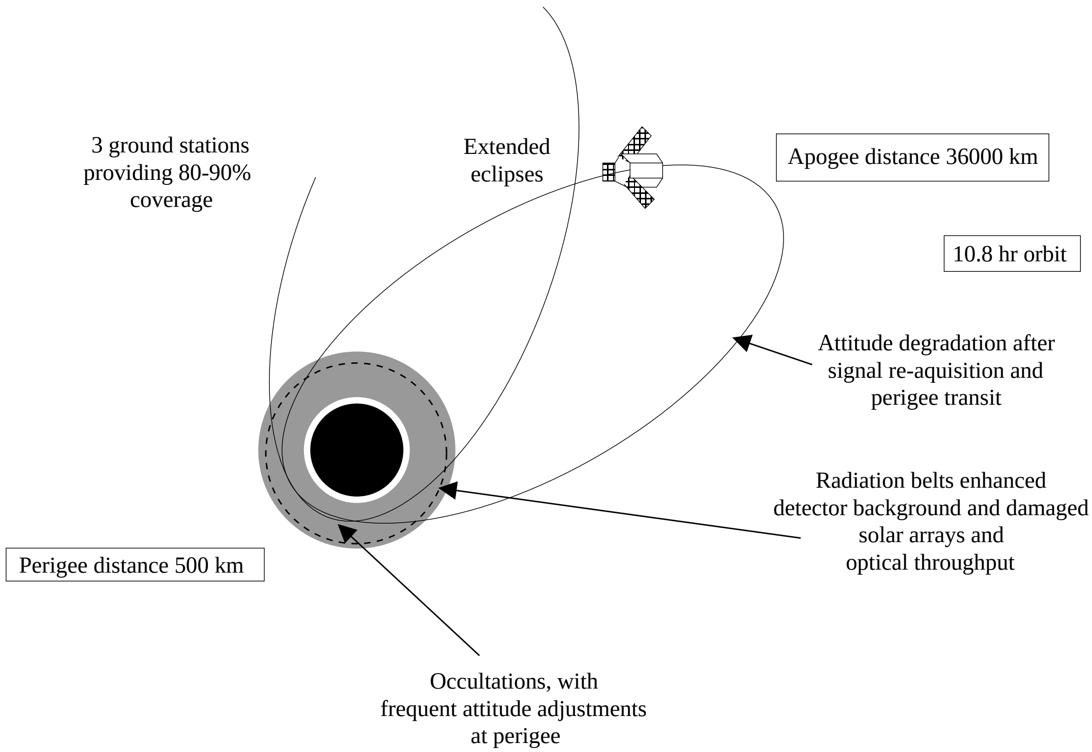

Figure 4.2. *The 'revised' or 'recovery' orbit from which the Hipparcos observations were conducted. The features of the orbit which provided considerable operational complications are indicated. The satellite could be observed for about 80–90 per cent of the time with three ground stations, compared with 100 per cent of the time in the intended geostationary orbit.*

4.5. Ground Station Utilisation

During the launch and early orbit phase, the ESTRACK launch and early orbit phase S-band network was used for telemetry acquisition, telecommanding and tone-ranging. The launch and early orbit phase ground stations were operated under control from the operations control centre. The launch and early orbit phase network consisted of the following ground stations: Kourou-100 (French Guyana), Malindi (Kenya), and Perth (Australia). Back-up support was given by the Odenwald ground station and the Villafranca transportable station when the satellite was visible.

For routine operations, the Hipparcos ground station network eventually consisted of five S-band stations: three stations belonging to the ESA ground station network (ESTRACK) and located in Germany (Odenwald), Australia (Perth) and French Guyana (Kourou). A fourth ground station, located in the U.S.A. (Goldstone) and belonging to the NASA deep space network, was added in the first half of 1990. The Kourou station, whose coverage was largely superseded by the Goldstone station, was subsequently taken out of the network in mid-July 1990. However, during the last months of the satellite's lifetime, Kourou was re-activated. All ground stations provided full tracking measurement, satellite data (telemetry) acquisition operations, and commanding for

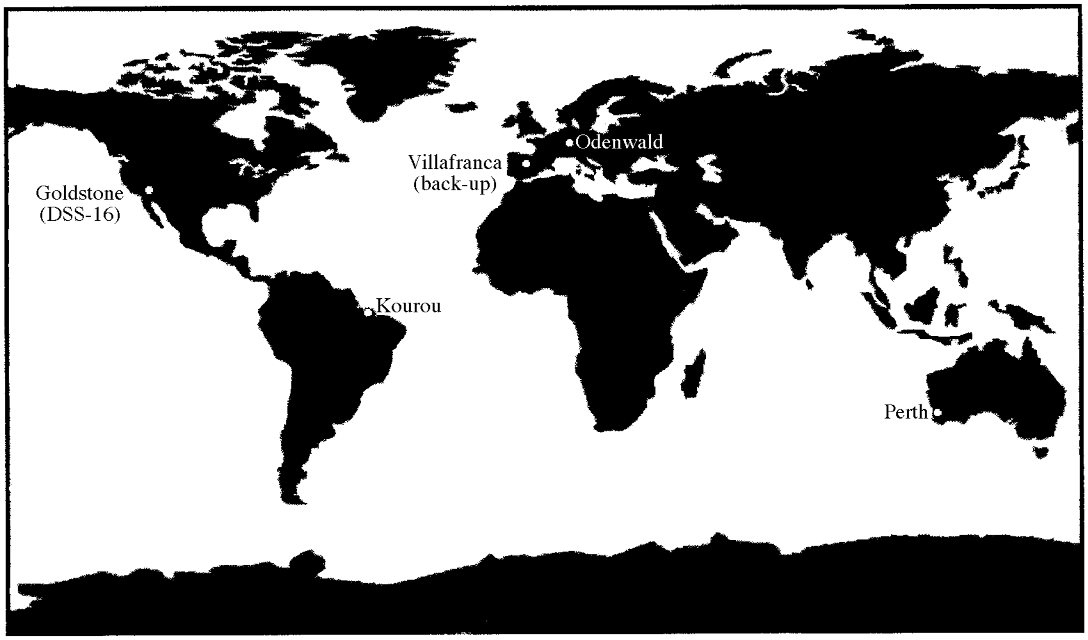

Figure 4.3. *Hipparcos ground stations. Only the Odenwald station had been originally foreseen for the geostationary orbit operations: the other ground stations were brought into the operational network only after a decision to proceed with the revised mission had been taken.*

the Hipparcos mission. A fifth ground station, located in Spain (Villafranca), provided a back-up to the Odenwald station. The ground station locations are illustrated in Figure 4.3.

The ground stations were operated in a transparent mode, i.e. the transactions between the operations control centre and the satellite were supported automatically with no local operator intervention at the station unless, in case of control centre or communication link failure, limited local commanding and telemetry readout functions were necessary to safeguard the satellite.

The ground communications facilities with the Kourou and Perth ground stations consisted of telecommand link for command transmission at 2 kbps using X25 level 3 protocol; low-rate telemetry link for telemetry transmission at 1.44 kbps using X25 level 3 protocol; high rate telemetry link for transmission at 23.04 kbps using X25 level 3 protocol (Perth and Kourou only); and voice loop, telex and telephone communications facilities.

The ground communication facilities with the Odenwald ground station consisted of telecommand link for command transmission at 2 kbps using X25 level 3 protocol; low- and high-rate telemetry link for telemetry transmission at 1.44 and 23.04 kbps using a digital PTT line with High Level Data Link Control (HDCL) protocol (integrated with the METEOSAT data traffic); ranging link allowing ranging request and measurement transmission using X25 level 3 protocol; link for antenna pointing transmission using X25 level 3 protocol; monitor and control link between the station computer at ESOC and the advanced monitor and control modules at the ground station, multiplexed with the high-rate telemetry.

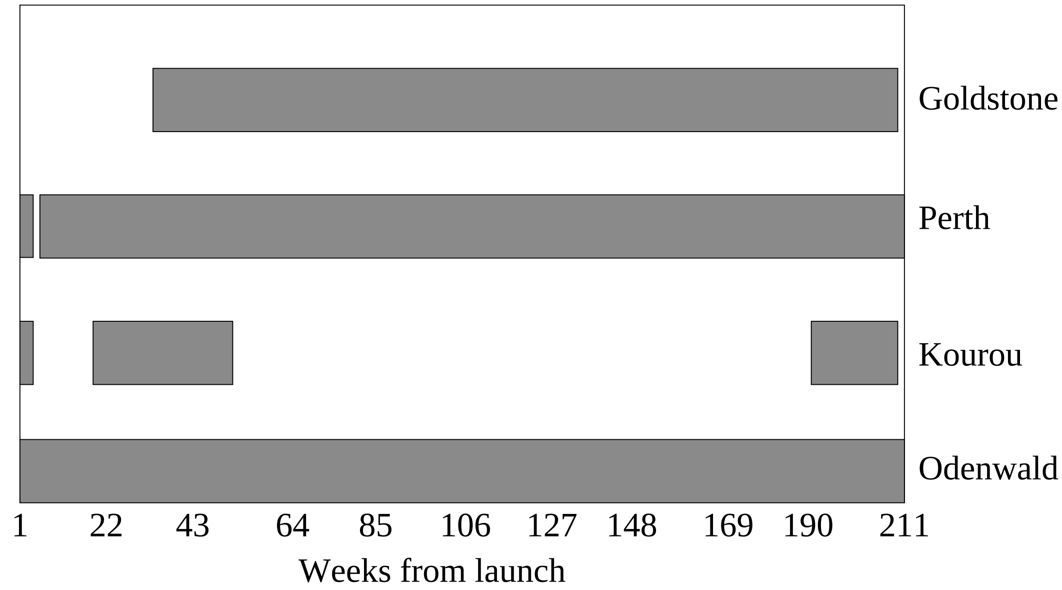

Figure 4.4. *Ground station support throughout the operational phase. The (baselined) Odenwald station was used throughout the mission. The Perth and Kourou stations were part of the ESA network. The Goldstone station was provided by NASA.*

The ground communication facilities with the Villafranca ground station, used as a back-up to Odenwald ground station, consisted of telecommand link for command transmission at 2 kbps using X25 level 3 protocol; low-rate telemetry link at 1.44 kbps using X25 level 3 protocol; communication of antenna pointing data; voice loop and telephone communications facilities in connection with back-up; commanding and telemetry quick-look functions.

The ground communication facilities with the Goldstone ground station consisted of telecommand link for command transmission at 2 kbps using X25 level 3 protocol; high-rate telemetry link for transmission at 23.04 kbps using X25 level 3 protocol; voice loop, telex and telephone communications facilities.

During the routine phase the ground stations at Odenwald, Perth, Kourou and Goldstone provided support. Routine phase back-up support was given by the Villafranca transportable station. The overall ground station support throughout the mission is illustrated in Figure 4.4.

The results for station coverage predictions did not vary significantly between the various possible orbits, and indicated that Odenwald would have contact with the satellite for not more than about 32 per cent of the time. Adding Perth and Kourou would extend the coverage to about 81 per cent (but even with those extra stations, long gaps of the order of 9 to 11 hours would have to be tolerated once every four days).

Preparations were immediately started to bring the Perth ground station on-line for telecommanding, high-rate telemetry reception and tracking purposes. This involved the installation at Perth of equipment which was used in pre-launch telemetry and telecommand interface tests between the satellite and the control centre at ESOC. In addition, dedicated communications lines were ordered from the PTT. Testing of the real-time interfaces with the control centre was performed during the first weeks of

September 1989. By mid-September, Perth was declared to be fully operational, and contact with the satellite was then available for about 62 per cent of the time.

Subsequently, similar activities were initiated to bring the Kourou Galliot station on-line for high-rate telemetry and telecommanding functions, the necessary equipment being procured or retrieved from other stations. By early November 1989, this station was also able to support all required real-time interactions between the satellite and ESOC. The support from Kourou was essential for obtaining a better understanding of the attitude control performance in the perigee region, as it was the only feasible station to provide occasional coverage in this region, being located near the equator.

Coverage was subsequently further improved by the assistance of NASA, through their offer of the Goldstone Deep Space Station (DSS-16) station in the Californian Mojave desert for the Hipparcos revised mission. In order to ensure that the data interfaces with ESOC were the same as for the other stations used in the Hipparcos revised mission, it was necessary to install ESA telemetry, telecommand, and data communications equipment on-site at Goldstone. The equipment was procured and integrated at ESOC before it was transported to Goldstone at the beginning of 1990. The real-time communications interface with ESOC was quite complex, and it took considerable effort on both ESA and NASA sides to get this station in a satisfactory operational condition (in the period from mid-March to the beginning of May 1990).

Since the Kourou station coverage intervals were almost completely contained within those of Goldstone, the Kourou support was not considered to be cost-effective within the existing network of stations. Kourou support to the Hipparcos revised mission was therefore terminated by mid-1990. During the last few months of the satellite's lifetime, the station was re-implemented into the ground stations network because of reduced Goldstone support due to competing demands.

Contact with the satellite became possible for about 90 per cent of the time and the duration of loss-of-contact periods was limited to not more than 1.5 hours: this eased the operations somewhat since the on-board attitude knowledge was less likely to be lost during the non-coverage periods. However, the complexity of the revised mission operations remained significantly above what would have been considered acceptable during the design phase. The coverage percentages are shown in Figure 4.5.

4.6. Mission Planning

The Hipparcos Mission Plan was regularly generated for a seven-day period starting always five days in advance, using a specifically designed software tool on the off-line computer system at ESOC. It included all orbit and attitude related events as well as ground-triggered ones (e.g. ground station non-availability).

The generation of a mission plan was based on both manual inputs e.g. payload calibrations, ground station availability times, and automatically computed events. Validity checks were performed according to pre-defined rules and any inconsistency was brought to the attention of the operator who could then modify the inputs. The output of the mission plan tool was a time-ordered sequence of events (ground station visibility, payload calibrations, antenna switching, occultations, eclipses and apogee/perigee control parameters changing) which could trigger possible satellite or ground segment

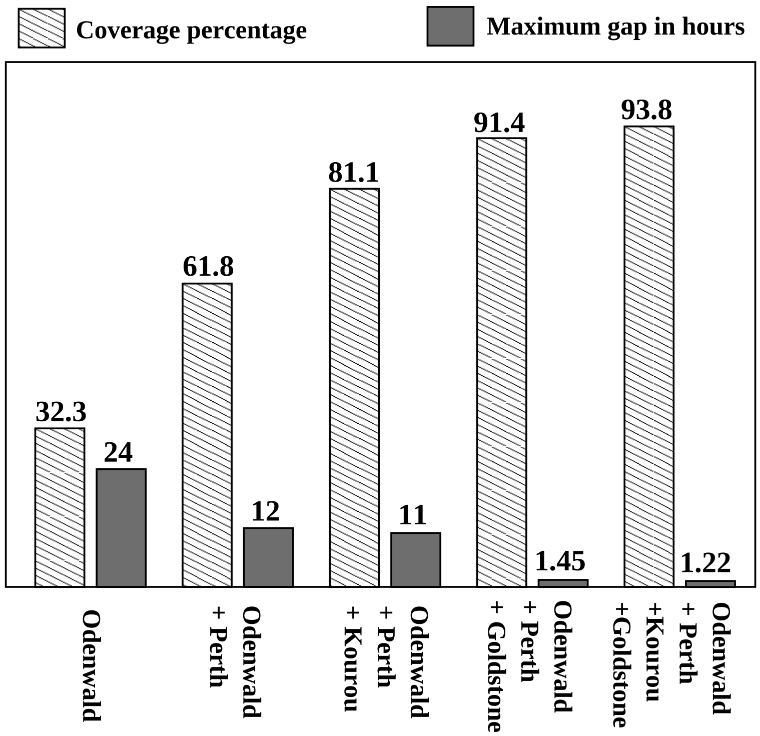

Figure 4.5. *Coverage percentage and maximum data gaps for the various ground station combinations. The Odenwald/Perth/Goldstone combination was used for most of the mission (the Kourou station adding little to this combination) with a resulting coverage of more than 90 per cent, and maximum data gaps around perigee of approximately 1.5 hours.*

related operations. From this mission plan, a chronological telecommand plan was produced automatically using pre-defined command sequences. Conflicts between the command sequences related to different events were resolved automatically according to pre-defined rules. The commands established in this manner were then transferred from the off-line computer to the on-line machine via a Hyperbus link, and were uplinked to the spacecraft at the specified times.

The mission plan had to be greatly enhanced to support routine operations for the non-nominal orbit with its period of 10 hours 39 minutes. In particular the list of computer generated events was augmented with the following items: (a) station coverage from Odenwald, Villafranca, Perth, Goldstone and Kourou; (b) spacecraft antenna to use in conjunction with the above stations; (c) periods of telemetry loss; (d) periods of commanding loss; (e) periods of perigee passage (spacecraft altitude below 6000 km); (f) periods of uninterrupted observation time; and (g) eclipse start/exit times.

The use of time-tagged commands had also been extended in order to perform spacecraft operations outside ground contact such as closing and opening of shutters for occultations, switching the perigee controller, updating the orbital oscillators, handling eclipses, etc.

A significant portion of the flight operations procedures had to be rewritten to take account of the new mission sequences adopted in the acquisition phase: the perigee raise and orbit adjustment manoeuvres, attitude acquisition sequence up to sun-pointing, and the sequence of events after solar array deployment. A considerable system analysis effort involving individuals from ESOC, ESTEC and external experts was needed to ensure that the operations were compatible with the satellite design characteristics. All operations had to be arranged into sequences which would fit within the interval of ground station coverage (about 8 – 10 hours for Odenwald). Furthermore, it was essential that, before contact was lost, the satellite was brought into a safe configuration which would allow it to survive the long periods without ground contact.

The operations were by necessity extremely risky since the time pressure did not allow full assessment of all possible eventualities, nor to prepare for potential contingencies. Validation (as far as feasible) of the new procedures was carried out using the ESOC software simulator, often not more than one day prior to their execution in actual operations.

The failure or back-up modes for the real-time attitude control had to be revised as the nominal criterion (using the interval between thruster pulses) was no longer valid because of the high torques in the perigee region. The redesign of the operations for the revised mission had to take account of the following new constraints: (i) station contact was frequently interrupted: this resulted in the need for procedures to ensure the spacecraft and payload safety during periods without ground contact; (ii) eclipse periods were of a much longer duration (maximum of 104 minutes) and the more frequent occurrence demanded a careful monitoring of the on-board power generation and associated battery management functions; (iii) payload occultation periods occurred with higher frequency (up to 28 per day) and longer duration: this led to a higher risk in losing the on-board attitude knowledge and required more frequent support from ground. Also, more telecommand uplinks had to be prepared (to open and close the payload shutters); and (iv) crossing of the van Allen radiation belts often led to difficulties in attitude knowledge (re)-acquisitions and the data collected over these periods were not of the required quality to be forwarded to the scientific data analysis teams.

The operational support of the on-board attitude determination process was one of the most demanding tasks. Because of the long ground station loss-of-contact periods and due to the noise in the star mapper signals induced by the radiation belts, frequent loss of attitude knowledge had to be anticipated: in that case, ground control had to re-establish the on-board attitude knowledge using downlinked star mapper signals and a star pattern recognition technique. A new team for supporting these unforeseen activities around the clock was recruited and trained within a very short time. In the early days of the revised mission, this specific support was required after almost every perigee passage. New on-ground support software was implemented to prepare the manoeuvres required for achieving the recovery orbit, i.e. the perigee raise and the period adjustment manoeuvres, and the new attitude acquisition sequence.

The mission planning software had to be redesigned and extended to include many more unforeseen events related to the specific orbit characteristics and van Allen belts passage. These included (a) ground station coverage start and end times, (b) switching of the on-board torque model in the perigee region, (c) implementation of a special attitude control strategy in perigee region, (d) switching of shutters and antennae outside ground station contact periods, (e) calculation of revised eclipse durations, and (f) prediction of payload occultation intervals.

Updates to the on-board software were necessary in order to extend both the time-tagged telecommand and the programme star file buffers (see Section 8.4), which had to be sized to accommodate the many commands needed during the long periods without ground contact. More sophisticated ground software was installed to allow real-time monitoring of the on-board attitude convergence using payload measurements.

The data interface definition between ESOC and the scientific data reduction consortia was also significantly affected by the revised mission: the necessary modifications to the interface were implemented and validated (using actual telemetry tapes) in the course of 1990.

4.7. Orbit Manoeuvres and Commissioning Activities

The initial payload switch-on and initialisation of the payload operations were conducted in the last week of September and the beginning of October 1989. The initialisation was severely hampered by extremely high solar activity with an exceptional solar flare occurring just at the time when the first star mapper samples were received: it was enough to ensure that no star transits could be observed.

After scanning law acquisition and successful calibration of the payload star mapper and main (image dissector tube) detectors, the actual scientific data collection started on 26 November 1989. The initial data recovery rate was about 50 per cent: this meant that 'good' data which could be used for the scientific data reduction were collected over about 48 hours during every four-day period, after which the ground station coverage pattern was repeated.

The computer system installed for the ground segment support was severely overloaded because of the many additional software tasks required, and the machine was replaced by a more powerful VAX computer in the beginning of 1990. This exchange resulted in extremely stable computer support conditions thereafter.

By summer 1990, the science data recovery rate had increased to about 65 per cent with the addition of the Goldstone station and because of further improvements and tuning of the operations support, mainly in the area of initialisation of on-board attitude determination and payload control.

Perigee Raising Manoeuvres

This section summarises the main events which happened, day by day, during the recovery mission implementation.

89-09-06: At 12:00:00 UTC, the spacecraft was de-spun from 37.03 to 30.05 rpm (12:00:00 UTC). The solar aspect angle at that time was $72\overset{\circ}{.}46$, with nutation less than $0\overset{\circ}{.}05$.

89-09-07: The decision having been taken to use almost all of the remaining hydrazine fuel to raise the perigee height, the first manoeuvre took place at 10:30:00 UTC and raised the perigee by 98 km.

89-09-08: A second perigee raising manoeuvre at 07:25:00 UTC raised the perigee by 102 km.

89-09-11: A third perigee raising manoeuvre at 09:14:00 UTC raised the perigee to the final target of 532 km.

Hydrazine Sun Acquisition

89-09-11: On the same day as the final perigee raising manoeuvre, the spacecraft was de-spun in two further manoeuvres. The first spin-down manoeuvre at 10:45:00 UTC reduced the spin rate from 29.49 to 19.90 rpm. The second spin-down manoeuvre at 11:45:00 UTC reduced the spin rate from 19.90 to 10 rpm.

89-09-12: Between 04:30:00 and 04:50:00 UTC, the two-stage slew to reorient the spin-axis to point at the Sun was performed. The first stage 'spin axis orientation' was performed using the on-board Earth/Sun sensor. The second stage 'hydrazine sun acquisition' used the sun acquisition sensor output. The solar panels were deployed at 05:02:00 UTC resulting in a decrease in the spin rate from 10.0 to 8.9 rpm. The fill-in antenna was then deployed.

89-09-13: The Perth ground station became operational for high-rate data telemetry.

Orbit Period Adjustment

89-09-14: The orbit was adjusted using the remaining hydrazine fuel to give a period which was commensurate with station coverage. In order to keep the spin rate within operational limits during the orbit period adjustment manoeuvres, the spacecraft was spun-up to 11.6 rpm. The first orbit adjustment manoeuvre started at 14:22:30.

89-09-15: The Perth ground station became operational for commanding.

89-09-16: The central on-board software was loaded and activated.

89-09-18: The central on-board software and payload were activated. The second orbit adjustment manoeuvre started at 13:52:00.

Hydrazine Fuel Dump

To prevent fuel sloshing during payload operations, the remaining hydrazine fuel supply was emptied out of the tanks while the spacecraft retained some degree of spin stabilisation. Following this, the helium pressurant was also drained.

89-09-19: Between 04:50:00 and 10:20:00 UTC, the first part of the remaining hydrazine was drained through cycles of firings of the hot gas thrusters. Nutation damping was performed using the cold gas thrusters.

89-09-20: Between 03:50:00 and 07:05:00 UTC, the second part of the remaining hydrazine was drained. Between 07:10:00 and 08:10:00 UTC, the helium was dumped. During the course of the fuel dumping exercise, the spin rate increased from 11.0 to 12.0 rpm.

Cold Gas Despin and Sun Acquisition

89-09-21: Between 09:30:00 and 14:14:00 UTC, the spacecraft was de-spun from 11.9 to 5.6 rpm using the cold gas thrusters.

89-09-22: Between 06:02:00 and 12:55:00 UTC, the spacecraft was de-spun in five manoeuvres from 5.6 to 0.2 rpm. After the third manouevre at 0.73 rpm, nutation damping was performed. At 13:03:00 UTC, the on-board sun acquisition software was triggered causing a final de-spin to the nominal spin rate of 168 arcsec s^{-1}. At 13:25:00 UTC, the on-board sun acquisition software was triggered orienting the spin axis of the spacecraft towards the direction of the Sun.

Real-Time Attitude Determination Initialisation

Real-time attitude determination activities proceeded as follows:

89-09-23: By collecting data at the nominal spin rate and at zero spin rate, a calibration of the biases of the sun acquisition sensor, and of the drifts on gyros 1 and 2, was performed. The sun acquisition sensor biases were uplinked to the spacecraft to optimise the sun-pointing accuracy, and the gyro drift estimates were used as first estimates for the ground real-time attitude determination software. Payload baffles were ejected on 26 September 1989, and star mapper transits were immediately detected.

89-09-27: The first attempt was made to run the ground real-time attitude determination software. Star pattern matching was achieved after tuning of the ground software. The basic angle separating the two fields of view was also calibrated.

89-09-30: The second attempt to run ground real-time attitude determination was aborted due to a solar flare giving extremely high background counts in the star mapper detectors. This phenomenon lasted several days.

89-10-04: The third attempt to run ground real-time attitude determination resulted in convergence of the ground software and an accurate attitude determination. Several attempts were made to command real-time attitude determination initialisation on-board. Convergence of on-board real-time attitude determination was not however assured, lasting for less than a minute after each attempt. The problem was subsequently traced to a problem in the ground software which was using the incorrect coordinate system for the on-board gyro drift representation.

89-10-08: The fourth attempt to run ground real-time attitude determination resulted in convergence of the ground software leading to the first successful initialisation of on-board real-time attitude determination at 10:30 UTC.

Between 9 and 31 October 1989, further successful attempts at real-time attitude determination initialisations were achieved, allowing payload calibration activities to proceed. In particular, a newly calibrated star mapper single-slit response was used to improve the performance of both real-time attitude determination and ground real-time attitude determination software.

During the same period, experiments were performed by varying the density of the programme star file. Real-time attitude determination performance was found to be satisfactory even for losses of signal lasting longer than one complete orbit. Real-time attitude determination performance was shown to deteriorate through the van Allen belts. The need for real-time attitude determination initialisation activities after every perigee passage was demonstrated.

Scanning Law Acquisition

To bring the spacecraft from sun-pointing mode to the nominal scanning law in which the solar aspect angle was maintained close to 43°, it was necessary to perform a 'scanning law acquisition manoeuvre'. This manoeuvre was first performed on 1 November 1989, as part of the planned commissioning activities and subsequently repeated at various times in the mission when spacecraft contingencies required a return to sun-pointing mode.

The main sequence of events for the first scanning law acquisition manoeuvre were as follows:

08:15:00 The spacecraft was de-spun from 168.75 to 0.0 arcsec s^{-1}. This lasted two minutes, but there then followed a wait period of 25 minutes to account for damping of any overshoot and to prepare the next set of commands.

08:42:00 The spacecraft slewed about the −Y axis from a solar aspect angle of 0° one of 43°. A further wait period lasting 24 minutes was required to damp any overshoot.

09:22:00 The spacecraft was spun-up from 0.0 to 168.75 arcsec s^{-1}.

09:23:00 The spacecraft normal mode controller was selected to prevent any drift away from the nominal scanning law (see Chapter 8).

Before the manoeuvre start, preparation of nominal scanning law-based mission planning and programme star file was performed.

Real-Time Attitude Determination Initialisation at 43 Degrees

On 1 November 1989, having completed the scanning law acquisition manoeuvre, operations were immediately started to initialise real-time attitude determination at a solar aspect angle of 43° (i.e. according to the nominal scanning law).

Between 1 and 26 November 1989, further real-time attitude determination initialisation were achieved, allowing payload calibration activities to proceed. The commissioning phase was formally terminated on 26 November 1989, and routine operations started from that time.

Payload Calibration

The initial payload calibration activities were carried out between 4–26 November 1989, as described in detail in Chapter 5. Following this phase, routine science data collection was performed.

The following additional events took place in this period:

89-11-14: The parameters affecting the performance of the thrusters within the normal mode control software were altered for perigee passage. The transition between normal and perigee controllers was set at 18 000 km altitude.

89-11-15: Gyro 3 was activated for the first time, without problem.

89-11-20: The real-time attitude determination and central on-board software processing was enabled during the entire orbit for the first time.

89-11-23: A revised input star catalogue for the recovery mission was received.

89-11-24: The innovation thresholds within which the real-time attitude determination software would accept star mapper transit information were increased for both narrow and extended window modes.

89-11-25: Gyro 5 was activated for the first time, without problem.