Bus Design of the Microsatellite BIRD for Infrared Earth Observation

C. Stelter, B. Biering, B. Heym
DLR, Institute of Space Sensor Technology
Rudower Chaussee 5, D-12684 Berlin, Germany
Phone: +49-30-67055601
e-mail: Christian.Stelter@DLR.de

Abstract. The DLR plans to launch the microsatellite BIRD in 1999 as part of a Earth remote sensing mission. This project represents the begin of a line of small satellite missions with ambitious scientific and technological objectives by application of new technology and respecting the limitations of microsatellites.

The spacecraft bus design is based on the proposed orbit and the payload requirements. The scientific payload is a novel multi-spectral sensor system, consisting of two cooled infrared sensor arrays and the Wide Angle Optoelectronic Stereo Scanner (WAOSS). A serious constrain of the satellite design is the required compatibility to a piggyback launch. The concept of the satellite bus fits to the requirements with the satellite dimensions of about 550x610x620 mm^3 and a total mass of approx. 85kg.

The presentation describes the approach for the system design of the satellite bus with focus on the mission profile and the requirements of the payload.

A special attention is paid to the bus structure, the attitude control and the thermal subsystem with its components and sensors under consideration as a low-cost mission.

1 Introduction

With stressed national funding and the need to apply new and advanced - but not yet space proofed - technology in the area of space based remote sensor systems the BIRD satellite demonstrates a new form of approach to this problem by establishing a small satellite program within the DLR community. The BIRD mission shall demonstrate the scientific and technological value and the technical and programmatic feasibility of a small satellite mission conception under low budget constrains. This mission follow strictly a design-to-cost philosophy where iterative loops are carried out between total mission costs, technical plan, scientific objectives and operational aspects under the constrain of a microsatellite mission.

1.1 Mission Objectives

A short summary of the primary mission objectives are given below:[4]

- test of a new generation of infrared array sensors adapted to Earth remote sensing objectives by means of small satellites
- detection and scientific investigation of hot spots (forest fires, volcanic activities, burning oil wells or coal seams)
- thematic on-board data processing, test of a neural network classificator in orbit[3]

The unique combination of a stereo optical and two infrared cameras gives the opportunity to aquire

- more precise information about leaf area mass and photosynthesis for early diagnosis of vegetation condition and changes
- real-time discrimination between smoke and water clouds, cloud analysis

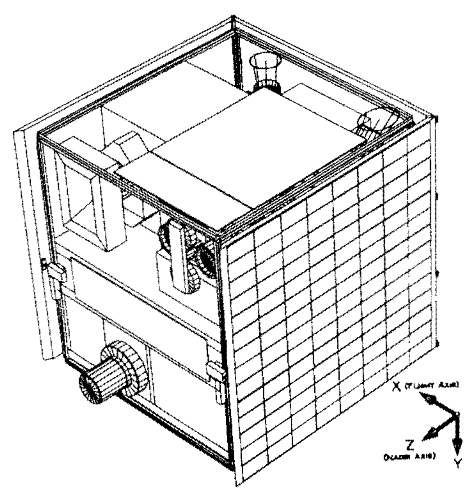


Fig. 1-1 View of BIRD with folded Solar Arrays (Launch Configuration) and w/o MLI

1.2 The BIRD Mission

The microsatellite BIRD will be launched as a piggyback into a low earth orbit. The payload will consist of a VIS/NIR Wide Angle Optoelectronic Stereo Scanner (WAOSS), which is a modified spare part of the MARS 96 mission, and two new developed, cooled infrared sensor arrays at 8.5µm and 3.7µm Wavelength. The sensor data of the observed areas are stored onboard together with the attitude and position data until downlink to a groundstation located in Germany.

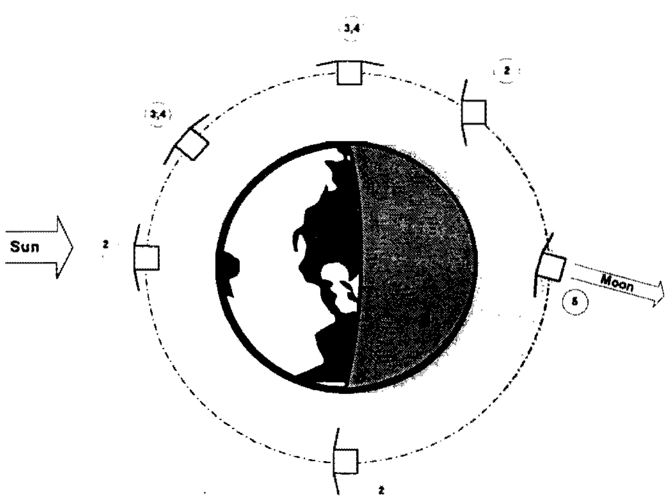
1.3 Operation Modes

The BIRD mission has to consider a lot of constrains, either funding or payload related.
Therefore it was necessary to design a Attitude Control System (ACS) which has to switch easily between several operational modes.
The satellite has to be oriented in a pushbroom mode during Earth Observation. Because of the high power surge during this time (table 1-1), the satellite should be capable to be inertial oriented to the sun during the non-observation period for efficient recharge of the drained batteries. This will keep the costs for expensive hardware like solar cells or batteries and secondary costs like increased launch mass low.

The drawback will be a reduced observation time of 10..15min for every observation period, with 5 periods a day.
All implemented operational modes are listed in table 1-2.

Tab. 1-1 Mission Summary

Function	Technological Satellite ; IR & VIS/NIR Earth Observation
development time	< 3 years
lifetime	> 1 year in orbit
launch constrains	piggyback launch into LEO; price range: approx. 6000US$/kg
Design Orbit	470km / 83° inclination (CHAMP-Mission)
Power Consumption	40W nom.; 205W peak (Observation Mode)
Solar Array Power	120W max.
Satellite Mass	85 kg
Satellite Dimension	folded: 550x610x620 mm^3
Communication	S-Band 19.2 kbps Uplink; 2.5 Mbps Downlink
Funding	DLR



Tab. 1-2 Operation Modes of BIRD

1. Initialisation Mode	The satellite will be inertial stabilised and its solar array oriented to the sun
2. Charging Mode	Orientation of the solar arrays to the sun for efficient charge of the batteries
3. Earth-pointing Mode G	Coarse adjustment along nadir of the payload as preparation of the earth observation, or for high speed downlink
4. Earth-pointing Mode F	Fine adjustment of the payload along nadir for earth observation in pushbroom mode
5. Calibration Mode	Calibration of the payload by using the moon and dark space
6. Emergency Mode	Reset of the SBC initiating Initialization Mode

1.4 Design Philosophy

The design philosophy is based on a design-to-cost-approach with avoiding of qualified and therefore expensive systems. Commercially available parts and components are applied as much as possible.
The used commercial components are sample tested for their radiation and vacuum-tolerance and - if necessary - protective measurements up to a replacement with a functional equivalent type are arranged.
A redundancy concept is only applied where critical key sections are involved and the additionally required resources in volume, mass and energy are acceptable. If possible, a functional redundancy with fault tolerance of the whole system is preferred.

2 Structure

2.1 General Objectives

The primary objective for the development of the satellite structure are small dimensions, as this is essential for an economical piggyback launch.
Secondarily, the development costs for the satellite have to be kept as low as possible. Therefore, readily available component parts are used. Own development of components will be done if the appropriate know-how is available.
A further possibility to reduce costs is to apply reasonably priced components and qualify them for space use.
The cost-benefit-ratio has to be taken into account when designing the load bearing parts. Expensive materials and costly manufacturing processes are only applicable if the structural performance will improved.
The small satellite size requires a special design for single parts integrating many functions to reduce weight, raise stiffness and load strength and eliminate fastener.

The satellite structure will be qualified performing dynamic load tests and thermal tests in vacuum. The concept demands not only tests of the complete satellite but also the qualification of parts and components before assembly.

The structure of BIRD can serve as a basis for further microsatellite missions.

2.2 Components of BIRD

The following components have to be integrated into the satellite.

Components of the payload:

- WAOSS - camera
- MWIR, LWIR - cameras
- stirling cooler for MWIR/LWIR
- radiator (heat dissipation)
- electronics for MWIR/LWIR (data acquisition)
- payload computer (data processing)

Components of the service system:

- solar panels (Power supply)
- batteries (power accumulation)
- laser gyro (attitude measurement)
- GPS-parts (position measurement)
- star sensors (attitude measurement)
- reaction wheels (attitude control)
- magnetic coils (attitude control)
- board computer (attitude control)
- S - band (telemetry)
- heatpipes (heat transfer)
- radiator (heat control)
- multiple layer insulation (heat control)

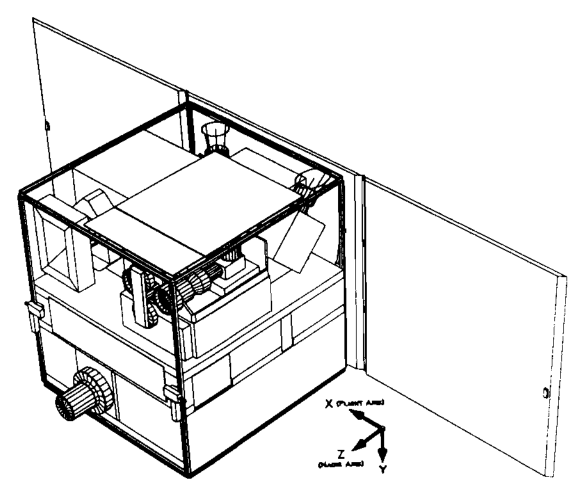


Fig. 2-1 Flight Configuration of BIRD w/o MLI

2.3 Requirements

With respect to the aims of the mission, there are many requirements on the structure; a selection follows:

Dynamic requirements:

- high resonance frequencies > 100 Hz
- mass maximum of 90 kg
- bearing the launch acceleration forces to the separation system
- linear acceleration in launch direction: 10 g (10 min)
- linear acceleration perpendicular to the launch direction: 1g (10 min)
- test accelerations:
  sinusoidal 5 - 100 Hz, up to 2 g
  random vibration 10 - 2000 Hz, up to 0,2 g^2 / Hz
  shock response 100 - 3000 Hz, 10 g up to 1000 g
- acoustic test 30 - 8000 Hz, 120-140 dB

Kinematic requirements:

- two foldable solar panels (90° each, launch config.: 620×550×620 mm;
  flight config.: 620×1600×620 mm
- separation system (spring acceleration to the speed of 1 m/s)

Geometric requirements:

- adjustment of the three optics (to reach complete pixel overlapping)
- holding the adjusted positions during assembly, launch and operation
- exact positioning of the reaction wheels, the inertial measurement unit and the star sensors in reference to the optics

Electrical requirements:

- insulating batteries
- grounding of the structure

Environmental requirements:

- thermal protection (Controlling of the satellite temperature, see TCS)
- heat radiating surfaces
- vacuum qualified materials
- temperature range qualified materials
- radiation shielding
- micro meteorite shielding

2.4 Design Concept and Realization

The core of the satellite bus can be devided into three sections:

- the payload - platform as a base for experiments
- the satellite service system with mostly ACS-components and batteries
- a segment with all necessary onboard electronics for payload and satellite

The electronics compartment and the service system are building the core of the spacecraft bus.
The service system is positioned directly on the separation system to get short guide lines for mass forces.
The lower mass of the payload will be coupled by a defined interface on the service system.
The star sensors are mounted on the payload platform for optical reasons.
The payload area contains a volume of 450×450×250 mm (less of star sensor volumes). It carries most of the payload components, as they are built as stand-alone units and no additional housing is required.

The infrared objectives are assembled in one shared housing to match extremely accurate (see geometric requirements), while the detectors are cooled down to 80 - 90 K.

The payload platform will be developed as a sandwich structure; it combines high stiffness, good strength properties and good thermal conductivity to two side heat-pipe flanges. As a result of this, fixed positions of components are guaranteed after the calibration process.

The interface consists in supports with an evenness tolerance of ± 20 µm.
The thermal loss of shape is calculated by FEM - calculations and holds the tolerances as well.

The segments of the service system are shown in figure 2-2; the service segment as the lower part with concentration of mechanical parts and the electronic segment as the upper part.

They are will be build of two housing halves milled out of aluminum and joined together with approx. 80 Titanium screws. The position of flanges and fins observes the requirement to lock payload bolts from the bottom.
Therefore, the stiffness is very high against bending and torsion in all three axes.

Mounting parts are used for the assembly of components, fulfilling special requirements like tetrahedral angles etc.
In the service segment, the mounting parts shall guide forces directly to the bottom of the body. This leads to a reduction of the remaining structural loads.

The completed housing is the base for structure parts of the electronic segment, built by separated parts of aluminum. The subdivisions are determined by the European standard layout for electronic boards of 160 x 100 mm. There are two sections, each containing twelve boards and one motherboard. The electronic boards have a fixed thermal interface to the structure.

Due to the high thermal conductivity, the system brings heat to the surface quite effectively. The bottom of the service segment will be equipped with radiator surfaces. Sideways, the bottom carries heat-pipe flanges for the heat connection of the payload.

The satellite structure will be completed by solar panels with hinges and release mechanisms, magnetic torque coils, antennas and a separation system.

During the payload integration, cables, two heat pipes and the multiple layer insulation will be assembled.

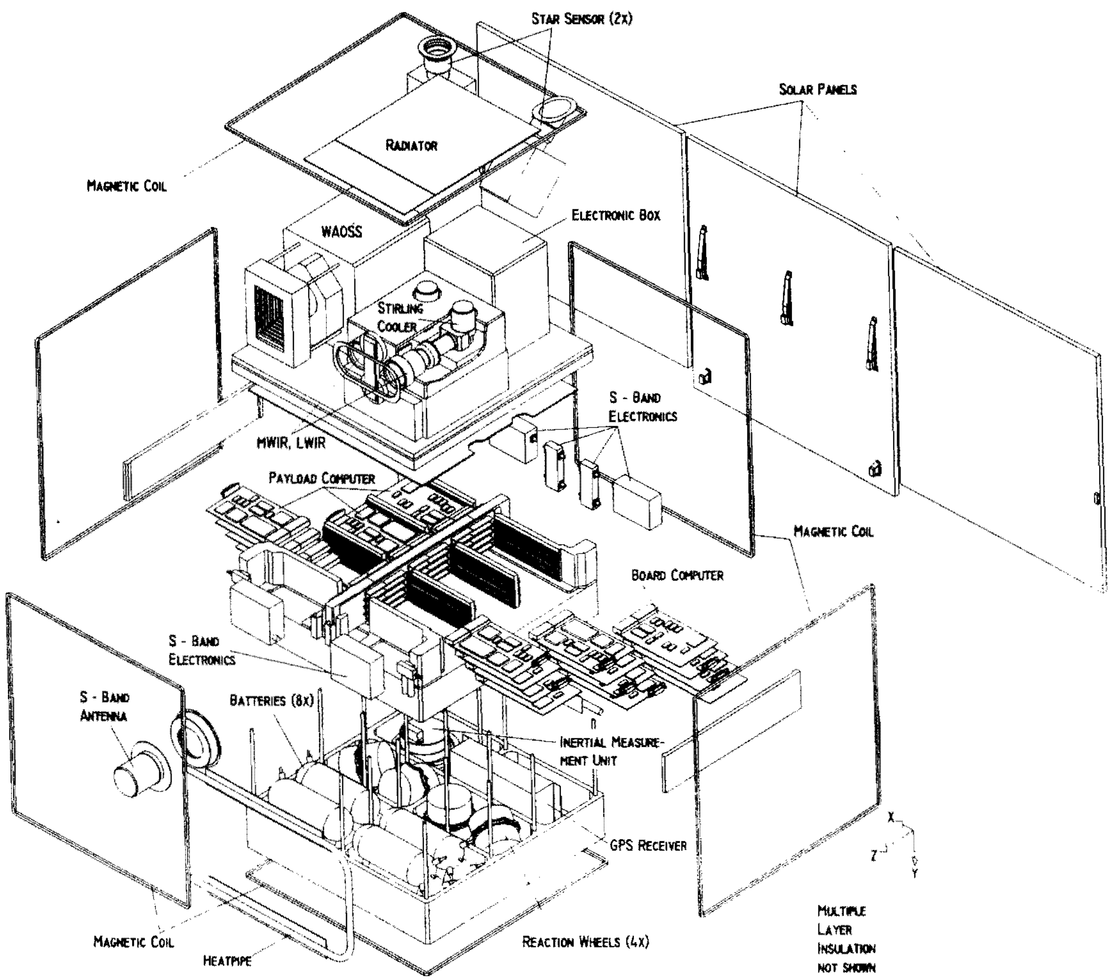


Fig. 2-2 Exploded View of BIRD

3 Attitude Control System Design

3.1 Attitude Requirements

The Attitude Control System (ACS) should be capable to function as a flexible system with different orientation modes and their transitions. The requirements are summarized in table 3-1.

Tab. 3-1 ACS Attitude Requirements

Pointing Directions	nadir, moon, deep space, sun
Absolute Pointing Error	7.0 arcmin (Earth Observation)
Pointing Jitters	1 arcmin/sec
Measurement Accuracy	0.2 arcmin
Slew Rate	1°/sec

3.2 ACS Equipment

Table 3-2 provides an overview of the used components with their typical performance.

Four reaction wheels in a pyramidal configuration are used for BIRD, giving the system the possibility to compensate the loss of one unit. For desaturation a system consisting of 3 pairs of magnetic coils are used.
As attitude sensors a commercially available 3-axis Laser Gyro and 2 star sensors are applied. In conjunction with GPS position information, the nadir direction could be calculated.

All sensors and actuators will be connected with serial interfaces to the ACS and Space Bus Controller (SBC) processor systems. The identical build SBC and the ACS processor system are based on the same electronic-board, developed originally for the control of the Sounding Rockets by the DLR.

The payload will have its own computer system, which is connected over serial interfaces with ACS and SBC. [1]

Tab. 3-2 Overview of the ACS Components

Component	Units	Performance Parameters
Inertial Measurement Unit (3-axis Laser Gyro)	1	• bias stability: 1°/h (1σ) • update rate: 50Hz • Noise: 0.125°/√hr max • Resolution: 16 arcsec
Star Sensor	2	• accuracy: 2-5 arcsec • magnitude: 0-7 • FOV: 15° • range: 1°/sec • integr. time: 150ms
Reaction Wheels	4	• momentum: > 0.2 Nms • torque: 20mNm • spin rate: 2..5000 rpm • resolution < 1 rpm
Magnetic Coils	6	• linear dipole: 1.5 Am^2 • torque: $1.2*10^{-6}$ Nm
GPS	1	• position: 100m • velocity: 2 m/sec
ACS/SBC-Processor System	2	• SAB80C167 Embedded Controler • RAM: 256 Kbytes • ROM: 512 Kbytes • A/D Converter • Modem

The BIRD mission and future missions, with increasing attitude requirements, require access to inexpensive, but highly accurate sensors and actuators. With the heritage in design and development of instruments at the Institute of Space Sensor Technology it was decided to develop a low-cost star sensor in cooperation with the industry.

3.3 Attitude Concept

Because of the flexibility in the orientation, a zero-momentum stabilization will be applied. All four wheels work around a nominal spin-rate of about 1/3 of their maximum, summarized to a zero momentum vector. Distortions are distributed between all four wheels until desaturation. This will ensure that no wheel will spin down to 0rpm during the observation period.

For better correlation between sensor and actuator data, the whole ACS activity is highly synchronized with a 500ms clock cycle.

3.4 Control Algorithm

The gyro unit will be used as a dynamic model of the satellite, while the star sensors correct the gyro dependent drift.
The attitude estimation will be done in several steps by using a Kalman Filter algorithm:
The two star sensors provide their attitude information with a delay of 500ms, (1 clock cycle) because of their dwell and processing time.
The information is correlated with the 1 clock cycle delayed gyro data to estimate the attitude parameters (λ, ω) at this time.
The present attitude will be extrapolated with the actual gyro data. This is the predicted attitude based on its history and will be parameter for the control activity of the ACS by determining an error vector (λ_e, ω_e).
The control activity will result from the Euler-Equation for ω and the differential equation for the Euler-Parameters λ.
To solve this equations a linearisation is done, by extracting their undisturbed components. The remaining, disturbed part of the equation will be used for determination of the necessary control activity to compensate the occurring errors λ_e, ω_e in a two step process.

3.5 Redundancy Concept

To save resources in mass, volume and energy a joint concept of the ADC and SBC will be applied, giving both systems a functional redundancy.
Nevertheless in operational mode both functions are separated and performed individually. If one of the systems get lost, the remaining computer system will perform both tasks in a time-sharing mode with reduced requirements.

The redundancy of the ACS components are on a minimal level and are designed as hot, in-system redundancies. The whole system is capable to compensate the failure of one component with limited restrictions, depending of the type of fault.

Due to the short development time, the autonomous failure processing on board will be limited to basic operations, which ensure a survival of the satellite. Further steps of handling the individual failure would be taken after close examination of the system

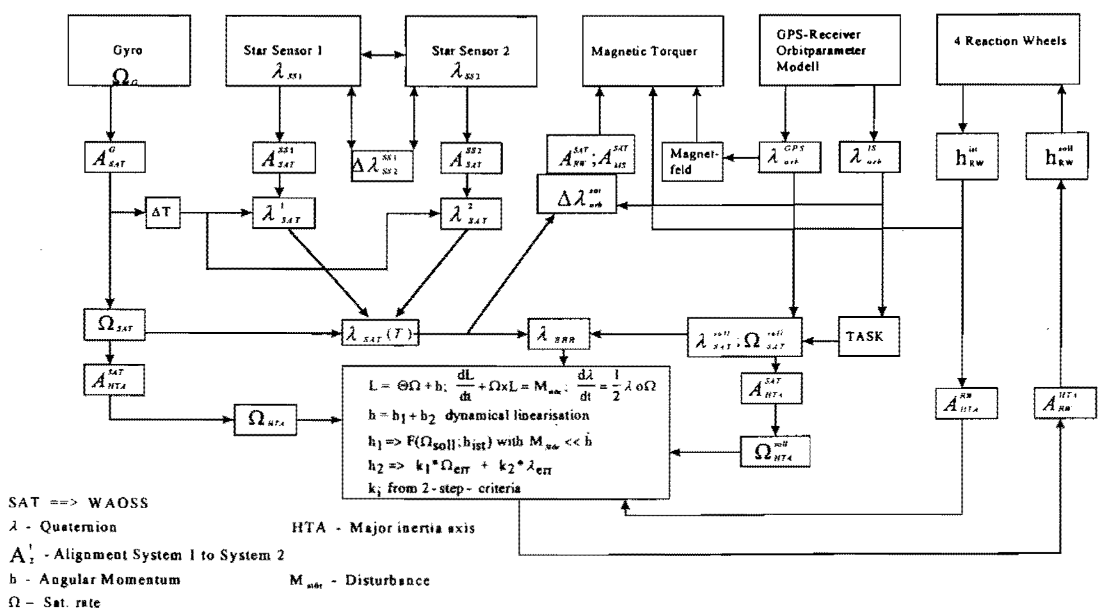


Fig. 3-1 Control Algorithm Diagram

4 Thermal Control System (TCS)

4.1 General Objectives

The TCS has to ensure the required working temperature ranges for all instruments, components and subsystems of the satellite in all Mission-Phases. BIRD-TCS should be passive and uses merely the dissipated heat of all components of the S/C system and Payloads. The fundamental structural subdivision of the inner satellite structure on the three segment planes „Service-Segment“, „Electronics-Segment“ and „Payload-Segment“ shall be kept up for the TCS in principal also. The TCS for the IR Payload Complex will be designed as an autonomous one. A Stirling Cooler is used for the cooling of the IR-Sensor, its operating heat loss is radiated to free space via a separate complex of heat storage-heat transfer system and a radiator.

The following table gives typical temperature requirements for some (regular) bus subsystems.

Tab. 4-1 typical temperature requirements for some bus subsystems

Subsystem	Temperature Range (°C)		
	pre launch/ launch	in orbit	
		operation	non operation
ACS	as op.	-15 to+55	as op.
Power	-15 to +35	-10 to +20	-15 to +35
Structure	-50 to +100	-50 to +100	-50 to +100
SBC	-20 to +80	-10 to +20	-20 to +80
Eject Mech.	tbd	tbd	tbd

4.2 Thermal Conditions in Consequence of the BIRD-Orbit-Phases

The probable but not finally ensured orbit conditions are :

Inclination: $i = 83^0$

Right ascension of the ascending node Ω ca.16^0

Eccentricity: $e = 0$

Orbit Altitude: $H = 470$ km

Orbit Cycle Time: $t = 94$ min.

The change of β (the minimum angle between the orbit plane and the solar vector) is in the range 86^0 and -76^0. For these orbit parameters the thermal conditions are strongly variable but also the measurement opportunities are more restricted compared with a sun synchronous orbit.

The satellite has to be orientated „sun pointing“ outside the shadow phases and the measuring phases, so that the sun vector is perpendicular to the solar panels. The duration of sun and shadow cycles very varies in one year.

The shadow cycles for a 94 min. orbit increase during the first 45 days after launching from about 21 minutes up to about 35 minutes and than become shorter and is omitted between the 85th and 100th day.

The shadow times vary between 0 and 35 minutes in 90 day cycle.

But there are shadow free cycles only then when a measurement is nearly impossible caused by incidence angle of the sun.

A specified alignment of the satellite to the solar vector is essential for the stabilisation of the thermal conditions but also for adequate power generation by the solar panels. This is a considerable requirement for the satellite attitude control.

For the concept of TCS is also to take into account the varying disturbances on the radiators by albedo and earth radiation

4.3 Conditions during the Operational Phases

The dissipated heat for all operational phases for a 24h period is presented in figure 4-1:

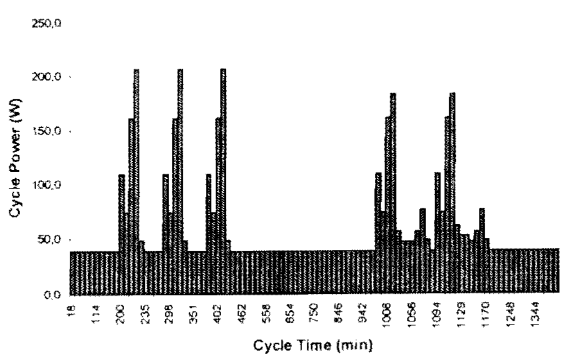


Fig. 4-1 Power cycle diagram for 24h (about15 orbits)

For the corresponding modes see table 1-1

- **Charging Mode (Mode 2)**

The orientation of the satellite is sun pointing, so that a sun incidence angle of 90° is realised. A constant load of 39W is taken into account for the phase of battery charging.

- **Preparation of Measurement (Mode 3)**

The satellite changes its orientation to the direction earth pointing . The power dissipation increases by 121W. The changing of orientation has to be performed at the end of this phase (result: nadir orientation).

- **Earth Observation (Mode 4)**

The satellite is orientated to nadir regarding to its instrumentation. The solar panels „see" the sun (only) with angels $0° \leq \beta < 90°$ for ten minutes. The power dissipation increases in such phase by about 206 W / 183 W

- **Data Transmission (Mode 3)**

The satellite orientation is the same as in the measurement phase. But the power dissipation is decreased to 76W.

4.4 TCS Design

The TCS consists of two parts. It will be realised by two separate TC sub Systems because the thermal requirements for the IR payload (including stirling coolers) are very different from the other payload requirements and the service system. So, it has to realise a „cold" regime for the whole IR complex and a „warm" regime for all other systems. The IR - TC sub System will be integrated in the thermal analysis as an autonomous one with defined thermal interfaces.

Both systems should work without use of additional heaters. Therefore a careful insulation of the whole system against free space is necessary. There are only heat fluxes permissible over special heat transfer components and radiators, which have to be optimised in each case for the „cold" and the „warm" regime. Heat sources are the power dissipations of the several components of the satellite only, where the basic power will be about 39W.

The balanced thermal budget is achieved by using a (main) radiator. It has to be designed so that it will be possible to radiate enough power against the environment in each mission phase to guarantee the required operational temperature ranges for all systems. In this relation the thermal optical coating of the radiator is very important. It seems to be clear that conventional thermal coatings do not satisfy the requirement that the radiator should be not larger than one plane of satellite cube. Two heat pipes are used for the heat transfer between the (two edges of) payload segment and the main radiator.

For the conditioning of the service- and electronic segment the conductivity of the AL structure is used for the heat transfer to the main radiator.

The interaction between satellite structure and solar panels will be minimised by using both thermal isolators for their mounting to limit heat conduction and covering the satellite body by MLI to limit the heat exchange by radiation.

The radiator coating, the special coating of the back side of the solar panels and the detailed design of the heat pipes are subjects of further investigations and thermal analysis.

4.5 Thermal Analysis

The thermal modeling is based on the lumped parameter method in the frame of the software package ESATAN (ESA PSS-03-105).

For the simulation of the several thermal influences of the orbit is used the ESATAN extension ESARAD.

When the design of structure is created more detailed it is foreseen to use the FEM package MSC NASTRAN for solving of special problems (e.g. for the estimation of the thermal expansion of the payload segment in dependence of current mission phases).

4.6 Thermal Conditions and their Modeling

The structure of the satellite is built like a tower of service-, electronic- and payload-segment. It is foreseen a main support plate for each segment which is braced by perpendicular to the plate mounted transverse structures. Both the service segment and the electronic segment will be realized by milling from a solid Al block.

The outer side of the base plate of the service segment is used as the main radiator of the satellite.

The support plate of the payload-segment will be a complex carbon fiber composites (CFC) sandwich. It will be consisted of two

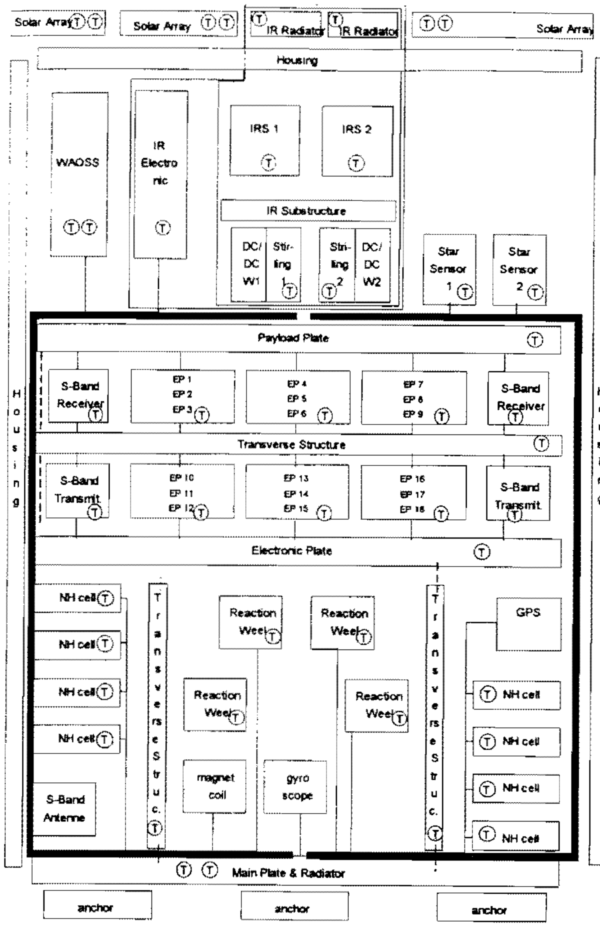


Fig. 4-3 Nodal model

CFC/NOMEX sandwiches and a core of good heat conductivity.

For the thermal modeling (lumped parameter method) of the satellite it was simplified that the thermal mass of whole satellite structure is concentrated proportional in the 3 support plates. Each support plate is defined as thermal node. The transverse structures are modeled as thermal conductors between the nodes.

There are 2 heat pipes for the good thermal connections of the payload segment with the main radiator.

The satellite body is cube-shaped and is covered with MLI completely excluding the gaps for apertures and radiators.

The solar panels are mounted on the satellite structure with usage of thermal insulators.

The effect of MLI is simulated in the nodal model by using of (fictitious) AL-sheets. These sheets are assigned the thermal-optical properties of MLI (very low effective emittance). The sheets are defined as thermal nodes with moderate heat-conducting ability connected with the nodes of the satellite structure.

The parameters of the orbit (470km/83°) cause complex irradiation conditions of the satellite surface. It were calculated the values of incident heat flux density from solar irradiation, albedo and earth radiation for each outside plane of the cube-shaped satellite. These irradiation influences were calculated for the two important orientations of the satellite (solar panels perpendicular to the sun vector ; nadir regarding to the instrumentation).

A current 54-node ESATAN model combines the switch on / switch off cycles of the several units and their power dissipation and their internal thermal connections with the orbit caused irradiation loads of the surfaces of satellite.

So the thermal state of the satellite can be simulated for any time.

Figure 4-2 shows a scheme of current nodal model.

4.7 Results of the Thermal Calculations

The figure 4-3 shows the temperatures for several selected units and payloads respectively for a period of 60 orbits (about 5600 min). It is to seen that from the arbitrary chosen start point the temperature advances an average (quasi steady state) unit temperature which is caused by the mode of satellite and the thermal environment of the unit.

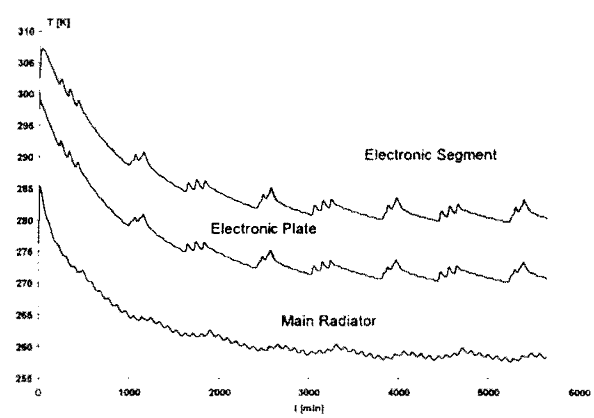


Fig. 4-2 Temperatures for service plate (main radiator), electronic plate, and a typical electronic plate of the electronic segment

4.8 Thermal Borderline Cases

The discussion of the following borderline cases is necessary since caused by the chosen orbit it is impossible to avoid the irradiation of the main radiator by partial considerable irradiation densities of earth radiation. The properties of usually used radiator coatings are adjusted to the frequencies of the solar spectrum, it is to be feared that the long wave earth radiation will cause considerable disturbances of the thermal regime.

To estimate these disturbances the thermal regime of the satellite was calculated for a minimum absorptance ($\alpha = 0.2$) and a maximum absorptance ($\alpha = 0.8$) of the main radiator, α is here the absorptance for earth radiation.

The resulting differences in the temperature of the main radiator for 60 orbits are illustrated in figure 4-4.

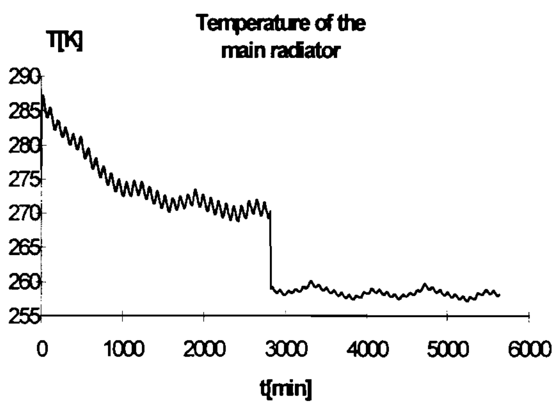


Fig. 4-4 Difference of temperature of the main radiator by changing its absorptance for earth radiation.

The first part of the curve is calculated for an $\alpha = 0.8$, the second part for an $\alpha = 0.2$.

It is visible that the increasing of the absorptance of the main radiator (only for the long wave earth radiation) causes a increasing of temperature level of the main radiator of about 15°. That means an increasing of the temperature level of all connected nodes.

4.9 Tests of TCS

The model- and test philosophy is the basis for all tests and their documentation. The specification of some tests will be caused by advanced results of thermal analysis. There are planed 4 groups of tests:

- Technological tests for the investigation of thermal optical properties of coatings and
- conductivity tests of thermal interfaces.
- Qualification tests. The TCS components will be qualified by using **STEM** (Structure Thermal Engineering Model) of the satellite.
- Acceptance tests will be made with the **PFM** (Proto Flight Model).

5 Project Status

The project finished its Phase B study March '97.

The kickoff of Phase C/D will be in September of the same year.

References

1. C.Stelter, F.Rollenhagen and K.Schmelovsky: „Concept of an Attitude Control System for the BIRD Mission"; IAA Conference on Small Satellites for Earth Observation, Nov. 1996

2. C. Stelter and I.Walter: „Concept of a Microsatellite Bus for Cooled Infrared Sensors"; IAA Conference on Small Satellites for Earth Observation, Nov. 1996

3. W.Halle: „Neuronal Network Application on Board the Micro Satellite BIRD"; IAA Conference on Small Satellites for Earth Observation, Nov. 1996

4. K.Brieß, H.Jahn, H.P.Röser: „A DLR Small Satellite Mission for Investigation of Hot Spots, Vegetation and Clouds"; IAA Conference on Small Satellites for Earth Observation, Nov. 1996