SIMULATION BASED CONCEPT FOR LOW COST ATTITUDE OPERATIONS OF EQUATOR-S

Uwe Feucht, Ralf Faller, Wolfgang Grimm
DLR Oberpfaffenhofen
German Space Operations Center
82234 Wessling, Germany
E-mail: uwe.feucht@dlr.de

Abstract

This Paper describes the attitude operations concept for the scientific satellite Equator-S. The original strategy requires attitude determination and commanding capability once per orbit to operate the onboard algorithm.

Due to operational problems and in order to reduce ground station time and operator workload a different approach was implemented by using dedicated flight dynamics simulation and attitude determination algorithms.

An object oriented approach for the flight dynamics simulation had been developed for mission analysis and preparation. The capabilities of this system turned out to be the key for low cost attitude operations. Its performance had been validated to be precise enough to predict the attitude dynamics for 4 to 6 days.

In this paper the results of this more offline-oriented low cost operations strategy are presented and it is shown how precise the attitude control requirements could be met. It is demonstrated how operational resources can be saved by efforts in mission preparations and sophisticated flight dynamics systems even and especially for small satellite missions.

***Key words**:* Attitude Dynamics, Flight Dynamics Simulation, Object Oriented Modeling

Introduction

Equator-S is a scientific satellite for the investigation of the magnetosphere of the earth under the responsibility of the Max-Planck-Institute fuer Extraterrestrische Physik (MPE) in Garching, Germany. Mission Operations are performed by DLR's German Space Operations Center. The spacecraft has been launched on Dec. 4, 1997 into a final highly eccentric 500 x 65000 km orbit where it was spin-stabilised with magnetic torquers to control both spin rate and spin axis direction. The torquing maneuvers took place within 2.5 earth radii altitude around perigee.

After separation from the Ariane-4 launcher and a kick motor firing to reach the final orbit the main attitude maneuvers were the changes of the spin axis attitude from nearly parallel to the ecliptic up to 90°. There is a constraint for the sun-aspect-angle (SAA) of the solar panels mounted on the surface of the cylindrical spacecraft. Therefore it is urgent to control not only the final spin axis orientation but also the trajectory from the initial to the final attitude.

For this process the often used Shigehara algorithm had been implemented which uses the actual attitude, a selectable target attitude, orbit information and a model of the earth's magnetic field. By reducing the error in the angular momentum using an asymptotic stability criteria a switching function for the magnetic coils for periods around perigee is derived[1].

Due to problems with the onboard main processors the routine operational phase of the spacecraft ended on May 1st 1998 and could not be resumed so far.

All magnetic torquing is only possible within a range of 2.5 earth radii around perigee. Following the sun-aspect-angle requirements the satellite's spin axis must be erected to stay perpendicular to the orbital plane (which is in fact almost parallel to the equatorial plane). Only this attitude guarantees sufficient power supply by the solar cells independently from the season. This major attitude maneuver of the mission is performed with the aid of a ring coil aligned parallel to the spin axis. See Fig.1 for a geometry overview.

Initially the spin axis is closely parallel to the ecliptic plane with the SAA between the spin axis and the sun direction of approx. 74° (optimum would be 75° with a tolerance band of +/- 25°).

Due to the motion of the sun w.r.t. the inertially fixed orientation of the spinning spacecraft an increment of 1° per day at an ecliptical latitude of the spin axis of 0° must be compensated.

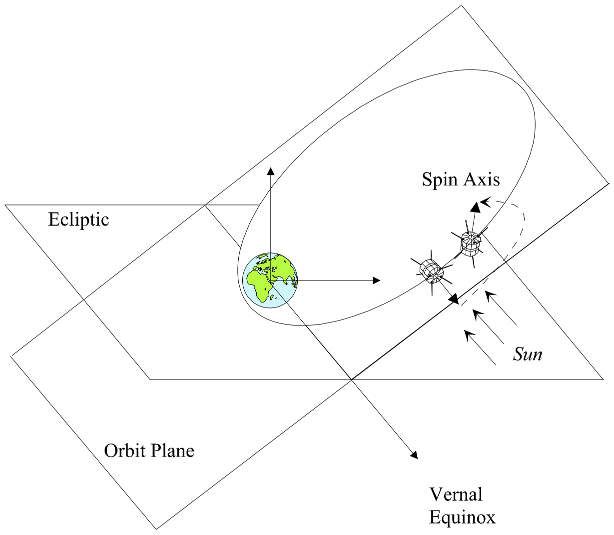

Figure 1: Spin axis erection geometry

But the main target of the erection process is an ecliptical latitude of 90° which makes the SAA independent from the motion of the sun.

A generic algorithm for the interaction of magnetic torquers with the earth magnetic field to control the attitude of a spinning satellite was developed by M. Shigehara[2].

To control the spin axis direction and rate the error vector **E** between the desired spin axis direction $\mathbf{k_f}$ and the actual orientation (expressed by the angular momentum **H**)

$$\mathbf{E} = \mathbf{k_f} - \frac{\mathbf{H}}{H} \qquad \textbf{(1)}$$

should be reduced to zero.

Differentiating (1) twice with respect to time and using asymptotic stability as a sufficient condition yields after some transformations the general control criteria

$$U\mathbf{E}\,(\mathbf{k_B} \times \mathbf{B}) \geq 0 \qquad \textbf{(2)}$$

with U as the coil polarity and $\mathbf{k_B}$ as the spin axis vector in body-coordinates.

The control of the spin axis direction is then realised by the following switching function for the torquers:

$$s = (\mathbf{I}_S \omega \mathbf{k} - \mathbf{H})(\mathbf{m} \times \mathbf{B}) \qquad \textbf{(3)}$$

with $\mathbf{I_s}$ as the tensor of principle moment of inertia around the spin axis, ω as the spin rate **m** the effective magnetic moment perpendicular to the spin axis and **B** as the vector of the earth magnetic field.

I.e. this control law which follows closely Shigehara requires the actual and the desired attitude of the spin axis, orbit information and a model of the earth magnetic field to derive a switching function for the magnetic coils for periods around perigee.

The key for making effective use of this strategy is the selection of adequate target orientations for each step of the torquing process (i.e. each perigee crossing).

The main opposite constraints are a small erection time, the effective and power-saving use of the magnetic torquers, and finally the sun aspect angle between the spacecraft spin axis and the axis satellite - sun.

Simulation System

Under these premises the key element for designing the attitude ground system baselines is a simulation and optimisation tool[3].

In contrary to previous missions where Fortran- or C-code was directly generated for modeling parts of the satellite's hardware and software the Equator-S mission is using a new object oriented modeling approach.

In order to use the knowledge and expertise in modeling and simulation acquired inside DLR but outside GSOC the choice was to use the Dymola[6] environment for the modeling and simulation tasks of Equator-S.

Satellites should be described in a notation close to physics. The most natural way of modeling interacting physical systems is to describe them as individual physical objects connected according to their physical energy flow interaction. This has the advantage over modeling physical systems via signal flows or input-output block diagrams as conveniently used for controller modeling (e.g. within Simulink). I.e. that mathematical equations of the overall system have not to be ordered beforehand.

The step from the simulation to an operational software could be realized by a feature of Dymola which allows to generate standard code extracts (e.g. Fortran,

C) from the simulation program. Even the complete simulation code can be extracted.

Under aspects of operational reliability and testability only the differential equations were extracted in Fortran-77 code and used for the operational software in a corresponding environment. Thus already approved modules (e.g. numeric integrator, geomagnetic field and orbit state vector predictors) could be used in order to minimize the effort in testing and maintenance.
The final design was flexible to adaptations without the need of a complete module and integration retest.

The main operational output were the switching commands for the magnetic torquer (on/off/polarity) and the onboard antennas in a time-tag format.

The commands were stored in an ASCII file which could be directly used by the command system.

Attitude Determination

The attitude solution file was generated for the EQUATOR-S science teams, as input for the spin axis erection torquing strategy, and for spacecraft spin axis surveillance purposes. It comprises time information and the spin axis direction parameterized in right ascension and declination, related to the Earth Mean Equator and Equinox of epoch J2000.0 (EME2000) coordinate system.

Due to spacecraft mass memory capacity constraints only 20 min of attitude determination relevant sensor telemetry data were available per orbit. Therefore attitude determination was performed in two 10 min time intervals, before torquing begin (40 to 30 min before perigee) and after torquing end (30 to 40 min after perigee); the respective spacecraft distances were 2.1 to 2.4 Earth radii.

The sun sensor outputs yield the sun aspect angle, i.e. the angle between spin axis and sun vector. The magnetometer measures the magnetic field vector outside the spacecraft, the x-component of the measured magnetic field unit vector yields the angle between spin axis and Earth magnetic field vector. The Earth magnetic field model used is the IGRF 1996[4].

Considering the limited attitude sensor information (only two measurements to determine two attitude angles) and the inaccuracy of the Earth magnetic field model in higher altitudes, a simple deterministic single axis attitude determination algorithm was sufficient. Here the algorithm according to Grubin[5] was used. The intersection of two conical surfaces yields two attitude solution vectors, and the real attitude is chosen either from an a priori estimate or from a check on the time evolution of the solutions.

The accuracies of the SAA and the magnetometer measurements were about ± 0.1 deg resp. ± 100 nT, which translated into an attitude accuracy in RA/DEC of about ± 0.6 deg.

Attitude Operations

Originally the attitude maneuvers were planned to be realised in a 3-step procedure:

- *Attitude Determination:*

  Telemetry data dumps were performed around the apogee, and the attitude determination yielded the spacecraft attitude after the last torquing maneuver which nominally stays inertially fixed up to the next perigee.

- *Simulation:*

  With a planned spin axis orientation as the other input the application of the Shigehara algorithm yields the switching times of the magnetic coils[1].

- *Commanding:*

  Before the spacecraft enters a distance of 2.5 earth radii (where the torquing maneuvers start to become efficient) these coil switching times are sent to the spacecraft as time-tagged telecommands.

This procedure was required to be used once for each orbit.

The crucial parameters for this algorithm are the input values of the actual and the planned spin axis orientation. The desired attitude is the result of a planning process[1] whereas the real attitude is depending from actual telemetry measurements.

The only direct attitude parameter appearing in the telemetry is the SAA measured by the sun sensor. It increases by 1°/day due to the relative motion of the sun.

Realtime telemetry cannot be received precisely at the times of perigee crossing due to the characteristics of the highly eccentric orbit.

The torquing software on ground requires the final values of the past torquing maneuver as a main input. This input can be delivered by online telemetry during phases where the coils are already switched off but the magnetic field is still strong enough to provide sufficiently precise magnetometer measurements. This leads to an earlier than planned switching off of the coils. The loss in torquing efficiency turned out to be neglectable.

The other possibility to get this data would be the storage of the magnetometer measurements by the onboard mass memory. The following dump must then occur early enough to run the ground torquing software before the next perigee requires new torquing commands.

These ground-generated time-tagged telecommands control the switching times of the onboard coils during the next perigee crossing.

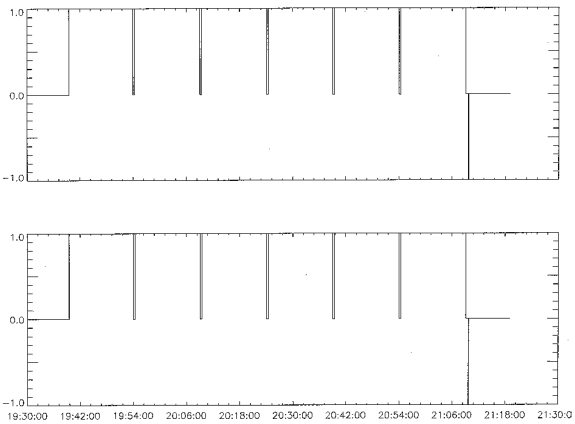

Figure 2: Coil polarities during an early torquing

At the beginning of the torquing process with an ecliptical latitude of the spin axis close to 0° there is no change in the coil polarity necessary. The current is simply switched on and off at +/- 2.1 earth radii, except for some short periods. These non-torquing times around perigee were regarded necessary to enable precise scientific magnetometer measurements. Later during the mission it turned out that this approach did not work satisfactory.

Figure 2 shows the status of the 2 coils, the short time with negative polarity after the end of the maneuver is required for demagnetisation. Otherwise the some remanent magnetism of the coils would disturb the scientific measurement.

Though for the first 30° of spin axis erection relative to the ecliptic plane the ground algorithm only has to determine the sign of the coil current and the begin and end times of each torquing action.

The rarely available telemetry from the sun sensor is shown in figure 3.

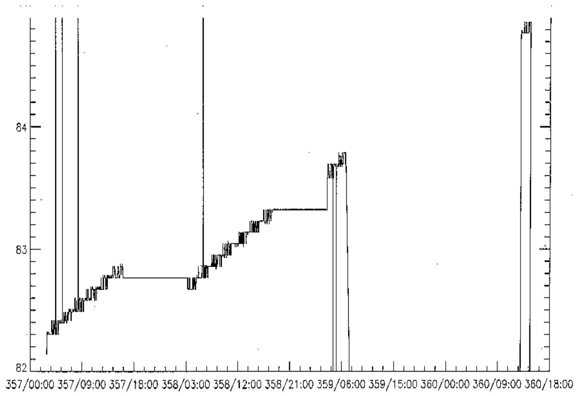

Figure 3: SAA for perigees #34 to #37

Two longer and two shorter periods of sun-aspect-angle measurements can be identified. The linear change in the SAA is due to the relative motion of the sun around the spacecraft system. In between there are the non-measurement periods around the 4 regarded perigees including the torquing activities. In the center of the graph (perigee #35) the efficiency of the torquing maneuver (resulting in a reduction of the SAA) is almost zero, in fact during this perigee crossing the coils were switched on only for a short period.

The operational problems during the first weeks made it impossible to have access to the latest sun sensor and magnetometer measurements, i.e. for the preparation of the next torquing commands the actual orientation of the spacecraft was not known from telemetry for most of the perigee crossings.

As described above this was not important for the first weeks of the erection process because once switched on there was no change in the coil polarity sign required.

The determination of the sign at that time was only roughly dependant from the actual attitude and the on/off times for the torquing maneuver could be derived from orbit calculations.

This easy situation turned out to become more challenging as the spin axis was moved out of the ecliptical plane higher than 30°.

Now it became crucial for the efficiency of the torquing maneuvers to work with a varying coil polarity during each perigee crossing.

Remaining operational problems, an increasing demand for mass memory resources by the scientific instruments and further approaches to save ground station time made it impossible to follow the originally planned strategy of using actual attitude determination data for the generation of torquing commands.

It became a stringent requirement to predict the torquing activities for usually 4 days.

I.e. the latest attitude input data for the ground torquing algorithm could be as old as 96h or more than 4 orbits. Since the Shigehara based algorithm requires the actual spin axis orientation a precise flight dynamics simulation facility became an urgent part of the ground operation system to meet the challenges of this changed operational environment.

Due to these different operational and onboard problems it turned out that actual telemetry was not constantly available for the required times.

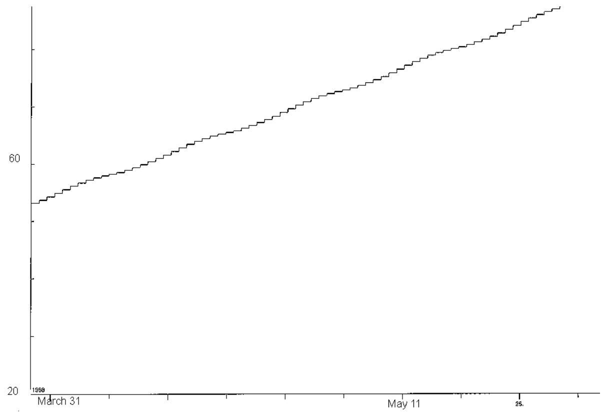

Figure 4: Ecliptical latitude

Same difficulties lead to a lack of commanding capabilities with the result that also not before each perigee crossing the coil switching commands could be sent properly.

Thus the (sometimes) contradictory requirements of both erecting the spin axis and keeping the SAA within its limits could only be followed by closely observing the simulated satellite dynamics.

The higher the ecliptical latitude was the more precise the simulation had to meet the real spacecraft situation to generate the correct commands. Figures 4 and 5 show the long-term simulation for the erection from 53° close to the final 90° within 2 months.

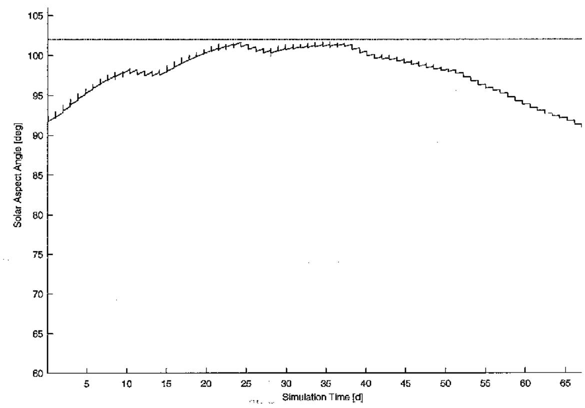

Figure 5: SAA

Figure 4 gives the ecliptical latitude of the spin axis for the simulated last 2 months of the erection process. Each step represents a perigee crossing.

Figure 5 shows the development of the SAA for the same period. The upper line is the limit of 102° which would not be reached in this scenario.

The attitude trajectory can be visualised in equatorial inertial coordinates in figure 6. It shows the typical behaviour of curving up to the ecliptical north pole.

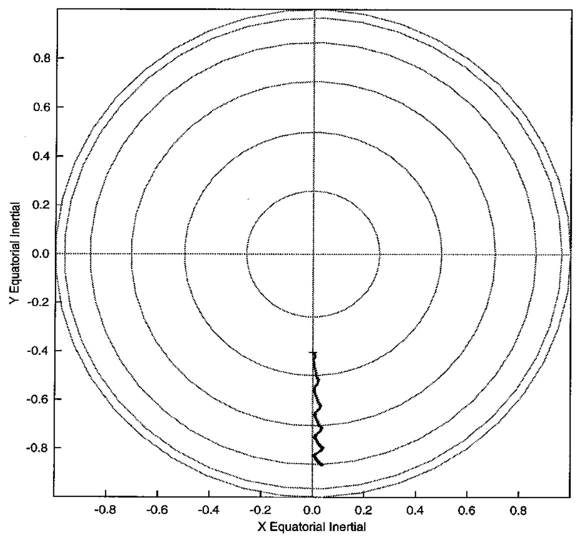

Figure 6: Attitude trajectory

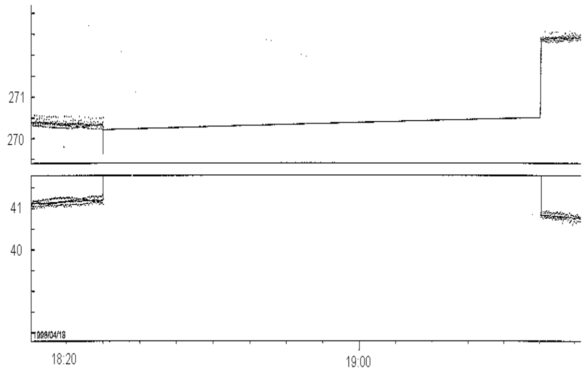

Figure 7: Right ascension and declination of the spin axis in the equatorial inertial system

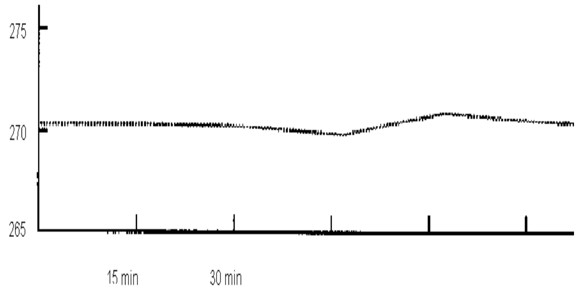

Figure 8: Simulated right ascension

To compare the simulated satellite dynamics with the behaviour of the real satellite a typical perigee crossing of April 18, 1998 (perigee #158) is taken as an example.

Figure 7 shows the result of the attitude determination system before and after the coil activation. The periods in between without usable data result from mass memory constraints.

The equivalent change in the simulated right ascension can be seen in figure 8.

Figure 9: SAA from telemetry

A comparison between reality and simulation for the SAA show figure 9 and 10 for the same torquing maneuver.

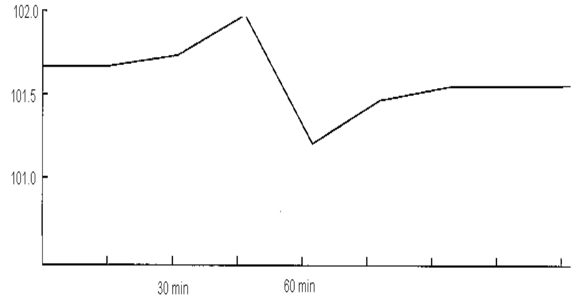

Figure 10: Simulated SAA

Also for the SAA there are no measurements available for about 1 hour around perigee.

Another failure of the second and last onboard main processor put a (hopefully preliminary) end to the Equator-S mission during the night from April 30 to 1st of May 1998 [7].

This event triggered the (up to now) last analysis of the satellite's attitude dynamics. Finally the development of the SAA had to be investigated under the constraint that no more torquing maneuvers were possible in the following weeks.

Figure 11 gives the development of the SAA starting 1st of May for the next 3 months:

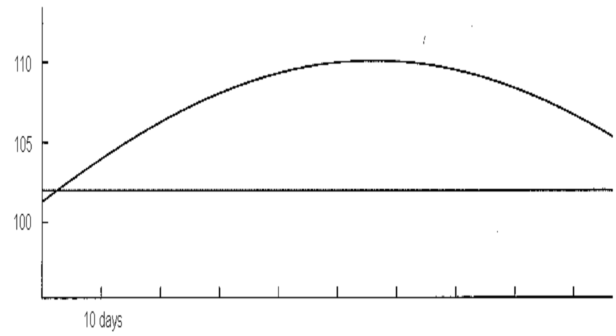

Figure 11: SAA development after 1st of May

At that time the spin axis had already reached an ecliptical latitude of 70°. Thus the effect of the seasonal variation in the sun incidence angle on the spacecraft was only a minor one.

The maximum of 110° SAA represents no significant danger to the spacecraft or the experiments, this value is reached only for a short period, after June 22 the SAA is decreasing again. The required limit for the SAA of 102° is violated for more than 3 months, nevertheless the required low energy level onboard due to no experimental nor torquing activities would make a survival of the spacecraft possible.

Analysis, Conclusion

An analysis of the simulation accuracy yields the following results:

- Simulations of up to 1 week simulated time showed no significant orbit deviations. I.e. the influence of the orbit error on both the coil switching times and the derivation of the magnetic field vector is neglectable and far within the measurement accuracy e.g. of the magnetometer.

- The attitude dynamics, represented by its state variables and derivated parameters like the SAA showed a maximum error of approx. 1.5°. This is slightly higher than the measurement accuracy of 1° and is reached at times of coil activity. The reasons are deviations in the modeling of the magnetic fields (both from the satellite and from the earth) counteracting with the coils.

Summary: The torquing performance would not have been better if the originally planned strategy of calculating the coil switching times once per orbit before each perigee would have been possible:

- The performance of the attitude dynamics simulation made it possible to follow the planned erection process despite different onboard and ground problems.

- With the original strategy the mission would have come into serious trouble already in January 1998 when the SAA came close to its upper limit for the first time.

- During the routine phase of the mission the performance of the simulation environment made it possible to resign from most of the scheduled ground station contacts and thus save remarkable operation costs.

Even for so called small satellites a sophisticated flight dynamics simulation system is worth the money. It can save mission costs and probably the mission itself.

Today object oriented simulation approaches make such developments possible. They can be easily reused for other missions.

References

[1] Feucht U., Enderle W., Moormann D., Equator-S Attitude Dynamics Simulation, *12th International Symposium on Space Flight Dynamics,* Darmstadt, 1997

[2] Shigehara M., Geomagnetic Attitude Control of an Axissymetric Spinning Satellite, *J. Spacecraft*, Vol.9, 1972

[3] Enderle W., Feucht U., Moormann D., Attitude Control Simulation for Equator-S using Object Oriented Modeling, *3rd International Conference on Spacecraft Guidance Navigation and Control Systems*, Noordwijk, 1996

[4] C.E.Barton, Revision of International Geomagnetic Reference Field Released; *EOS Trans. 77 # 16*, 1996

[5] C.Grubin, Simple Algorithm for Intersecting two Conical Surfaces, *J. Spacecraft, Vol. 14,* 1977

[6] Elmqvist H., Object Oriented Modeling and Automatic Formula Manipulation in Dymola, *Scandinavian Simulation Society SIMS'93,* Kongsberg, 1993

[7] Melzer F., Hoefner H., Equator-S Failure Report, *Max-Planck-Institut für extraterrestrische Physik*, Garching, 1998