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Long term behaviour of the solar panels of INTEGRAL

Bachelor Thesis

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1 Introduction

The first artificial satellite was launched by the Soviet Union on 4 October 1957.[1] In this context of human spaceflight, artificial was introduced to distinguish objects which have been placed into the Earth orbit from natural satellites like the Moon. According to David Rising until November 2013 about 6,600 satellites have been launched.[2] [3] There exist different groups of satellites defined by August Grant and Jennifer Meadows. For example military and non-military satellites. The non-military satellites can be separated in three basic categories: fixed satellite services, mobile satellite systems and scientific research satellites (2004, Communication Technology Update). Some of them use nuclear power systems but the most common way of power supply is the usage of solar panels as the handling of nuclear material is very dangerous. Furthermore most of the satellites perform a save re-entry after their mission is over to stop the growing amount of space debris. This wouldn't be possible if there is nuclear material on board. Consequently the usage of solar cells established oneself. Solar energy is converted in electrical power and stored in batteries, to be released when the satellite is in the Earth's shadow or is not pointed to the sun because of astronomical observation. A successful scientific mission requires a well working satellite and therefore the power supply system is very important to achieve this. Consequently the solar panels have to be designed very carefully, which means they have to work long enough to achieve the missions timetable and they have to withstand the harming conditions in space. Modern solar panels of satellites are protected against such influences as good as possible, but nevertheless get damaged over the mission time. Hence it is important to investigate what harms the solar panels and how strong is the damaging influence during a whole mission. Therefore this work deals with the long term behaviour of the solar panels of the INTEGRAL satellite.

1 http://iml.jou.ufl.edu/projects/fall99/coffey/history.htm
2 http://www.esa.int/Our_Activities/Operations/Space_Debris/About_space_debris
3 https://web.archive.org/web/20131112013308/http://seattletimes.com/html/nationworld/2022236028-_apxfallingsatellite.html

2 Solar panels in space

Solar panels in space have to deal with other conditions than solar panel on Earth. The conditions in space implicate both advantages and disadvantages. There is no atmosphere which acts like an filter between the solar panel an the solar radiation. Also there is no restriction due to day and night cycle. Consequently the solar energy reaching the panels in space is higher than on the Earth's surface. However there are some disadvantages. It's very expensive to bring objects into space, which leads to a maximum reduction in size and weight of the satellite. As the panels are attached to the satellite there is a limit to its size. Furthermore it's not cost-efficient to repair the panels in space or to exchange them. The panels have to deal with harsh conditions such as the variation in temperature. Also they provide power for extremely expensive equipment. All these aspects lead to one point. The solar panels have to be made up of the best materials available. From the mission point of view this is very good, but it also will be much more expensive. Another aspect is the influence of space debris and micro meteorites which damage the solar panels by impact driving. The number of space debris decreases with increasing height. Most parts are at 600 km to 1500 km height. However there is also a serious amount of space debris up to 36000 km height.[1] Consequently the solar panels of INTEGRAL are in danger of being damaged by space trash and micro meteorites. The most important aspect is the influence of radiation. On the one hand the charged particles out of the solar flux harm the solar panels. On the other hand the satellite has to pass the Van Allen radiation belts including its high energetic particles. In detail it consists of two permanent belts as seen in Fig 2.1 . The inner belt extends from 1000 km to 6000 km above the Earth and is made up mainly of high energetic protons. The range of the outer belt is from 13000 km to 60000 km above the Earth's surface and consists of high energy electrons with energies from 0.1 to 10 MeV. Both belts are created because of the Earth's magnetic field when it traps charged particles from solar wind and cosmic rays. Afterwards these particles build two belts with space in between. Their intensity increases or decreases in response to the incoming energy

1 http://www.space.com/16518-space-junk.html

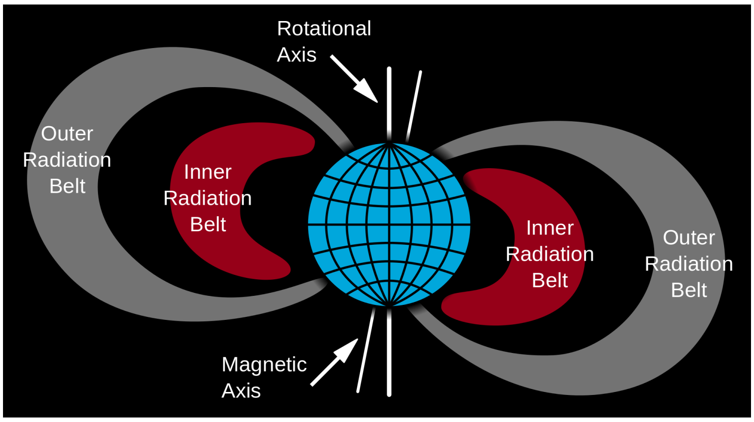

Figure 2.1: Van Allen belts shown in transversal section (from `https://upload.wikimedia.org/wikipedia/commons/thumb/0/02/Van_Allen_radiation_belt.svg/500px-Van_Allen_radiation_belt.svg.png`)

from the sun. In extreme cases the sun has such an influence that a temporary third belt is created or destroyed. A coronal mass ejection of the sun splits the second belt into two clear belts separated with space in between. This third belt lasts for about a month till it is finally destroyed by an interplanetary shock wave from the Sun. In case of INTEGRAL all three belts are interesting as the satellite moves between 3000 km to 153000 km above the Earth. The solar panels are damaged by high energy protons and electrons when INTEGRAL passes trough these belts. [2] The solar cells used on INTEGRAL's solar arrays are silicon solar cells. The functional principle of a silicon solar cell is shown in Fig 2.2. The semi conductor material has to be doped which means to carefully integrate chemical elements. As a result you create either a positive charge carrier excess or a negative charge carrier excess in the semi conductor material. The former case is the p-conducting semiconductor layer with holes the latter is the n-conducting semiconductor layer with free electrons. Fig 2.2 names the n-semiconductor layer "emitter", the p-n intersection "junction" and the p-semiconductor layer "base". If you put the p- and n-semiconductor layer together some electrons from the n-layer wander to the holes in the p-layer and create imperfections in the n-layer. Some of these electrons recombine with holes near the boundary creating a negative charge carrier excess in the p-layer. Now electrons are missing in the n-layer creating a positive charge carrier excess. This is only in an small area near the boundary as the increasing negative charge carrier in the p-layer hinder further electrons to come over. Consequently an electric field is created at the intersection. The n-layer is pointed to the sun and very thin as the photons have to reach the p-n intersection. The incoming photons

2 https://www.nasa.gov/content/goddard/van-allen-probes-spot-impenetrable-barrier-in-space

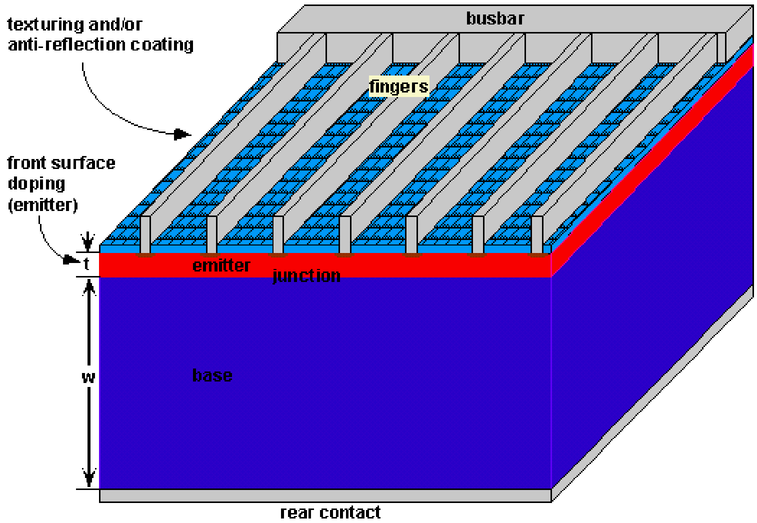

Figure 2.2: Basic schematic of a silicon solar cell (`http://www.pveducation.org/sites/default/files/PVCDROM/Design/Images/CELLSCH.GIF`)

split up the covalent bonding between silicon atoms creating further free electrons and holes. Due to the electric field the electrons are transported to the n-layer and the holes are forced to the p-layer. This leads to a positive charge of the p-layer and a negative charge of the n-layer. Closing the electric circuit leads to an electron flow trough the conductor and therefore to an charge equalisation. On the upper side the busbar with the contact fingers is the minus pole. On the bottom side the continuous metal layer is the positive pole. A transparent anti-reflection coating helps to protect the cell and reduces the reflection loss. 3 4

3 http://hyperphysics.phy-astr.gsu.edu/hbase/solids/pnjun.html
4 http://www.sfv.de/lokal/mails/phj/solarzel.htm

3 Detailed information about INTEGRAL

INTEGRAL is an abbreviation for INTErnational Gamma-Ray Astrophysics Laboratory and was launched on 17 October 2002 by a Russian Proton rocket into an unusually high Earth orbit. Only because of this high eccentric 72-hours orbit it is possible that INTEGRAL spends most of its time above 40000 km and consequently reducing the radiation background effects. This is necessary as its instruments work with the coded mask method and any radiation influence will downgrade the results. [1] Tab 3.1 gives a short overview about the satellite and its orbit.

Table 3.1: Short overview about INTEGRAL and its Orbit (from `http://sci.esa.int/integral/47360-fact-sheet/`)

Satellite	Launch mass	4 tonnes
	Height	5 metres
	Diameter	3.7 metres
	Solar panels	16 metres across
Orbit	Perigee	9000 km
	Apogee	153000 km
	Inclination	51.6 degrees
	Orbit type	highly eccentric 72-hour orbit around Earth

The orbit parameters which are displayed in Tab 3.1 are from the start of the mission.[2] During its time in space perigee, apogee and inclination are often changed to achieve different mission aims. The eccentricity is a dimensionless parameter between 0 and 1. It expresses the deviation from an ideal circle. If the eccentricity is 0 the orbit is a perfect circle whereas an eccentricity of 1 is an exact parabolic orbit. Fig 3.1 proofs that the orbit of

1 http://sci.esa.int/integral/47360-fact-sheet/
2 http://sci.esa.int/integral/31289-orbit-navigation/

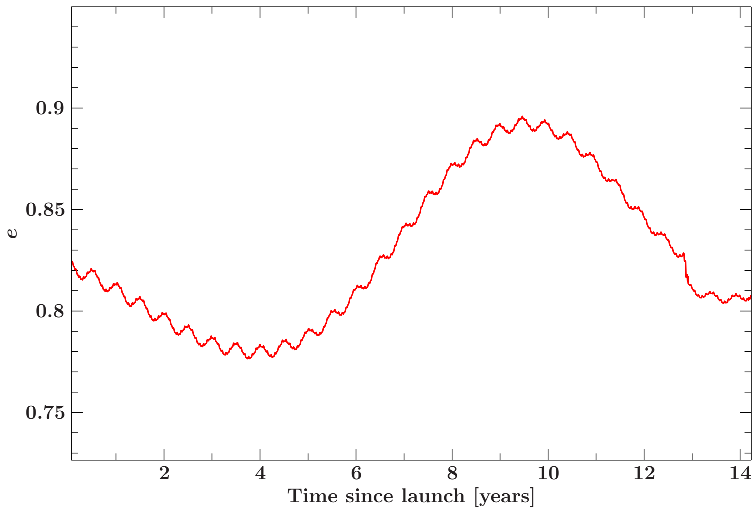

Figure 3.1: Eccentricity variation since launch

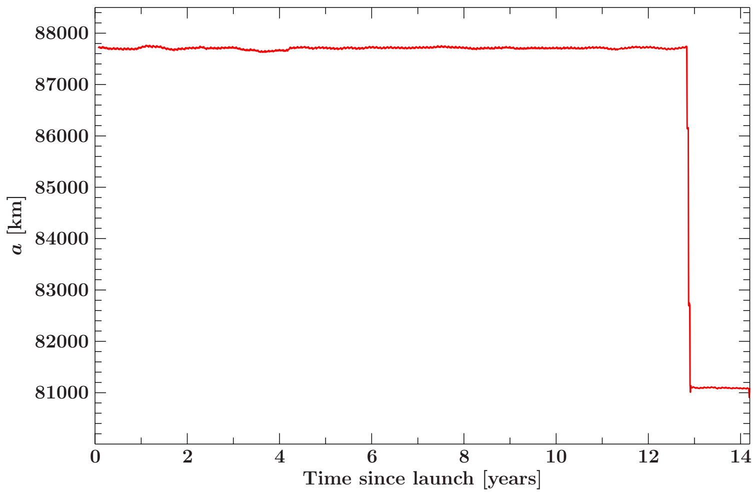

Figure 3.2: Semi-major axis change in the year 2015

INTEGRAL is highly eccentric as the eccentricity e varies between 0.78 and 0.9. The current state orbit type is the result of an shift in 2015, as the 72-hours orbit was changed to an 64-hour orbit. This was necessary to guarantee a save re-entry in 2029 and to exploit the fuel budget. Fig 3.2 shows this orbit shift by displaying the semi-major axis change in the year 2015. Originally the duration of the mission was planned to be two years. Until then the whole project was a great success and the components are construed to last much longer. Consequently the duration was extend several times. Meanwhile the mission is prolonged till 2018, but plans using INTEGRAL even exceed this date.[3]

3 http://sci.esa.int/integral/47360-fact-sheet/

INTEGRAL is an astrophysics laboratory for observation in the gamma ray section and was launched into space because the gamma rays cannot penetrate the Earth atmosphere. This high-energy electromagnetic radiation occurs in many different objects because of various physical processes. Thus INTEGRAL's observations include a wide variety of objects and research fields:[4]

- compact objects such as white dwarfs, neutron stars, black holes and gamma-ray bursts
- the galactic centre
- particle process and acceleration (beams and jets)
- stellar and explosive nukleosysnthesis such as supernovae and novae
- structures in the milky way galaxy

To achieve its objectives INTEGRAL is equiped with different instruments. Out of the four main instruments, IBIS (Imager on-Board Integral Satellite) and SPI (Spectrometer on INTEGRAL) build the major payload. Together with the third instrument JEM-X (Joint European X-Ray Monitor) all have in common that they work with the coded mask method, which is shown in Fig 3.3. This technique relies on the shadow model and is necessary as there is no possibility to take an optical image in the gamma range with the help of mirrors or lenses. JEM-X supports the two main instruments providing complementary observations in the X-ray range. The fourth main instrument is the OMC (Optical Monitoring Camera) which takes images from the observed object in the visible spectral range. All four instruments simultaneously monitor the same field of vision. At last there is the particle radiation environment monitor (IREM) which measures charged particles fluxes from the environment around the satellite. This helps to assess the background and to optimize the sensitivity and performance of the instruments. Furthermore it is necessary to protect the device if the radiation exceeds a critical value by shutting down the instruments.[5] [6] [7]

The solar arrays of INTEGRAL convert the incoming solar radiation into electrical power and supply the spacecraft during sunlight periods. In combination with a battery the

4 http://sci.esa.int/integral/31169-objectives/
5 http://www.dlr.de/rd/desktopdefault.aspx/tabid-2448/3635_read-5473/
6 http://www.cosmos.esa.int/web/integral/instruments
7 http://sci.esa.int/integral/47360-fact-sheet/

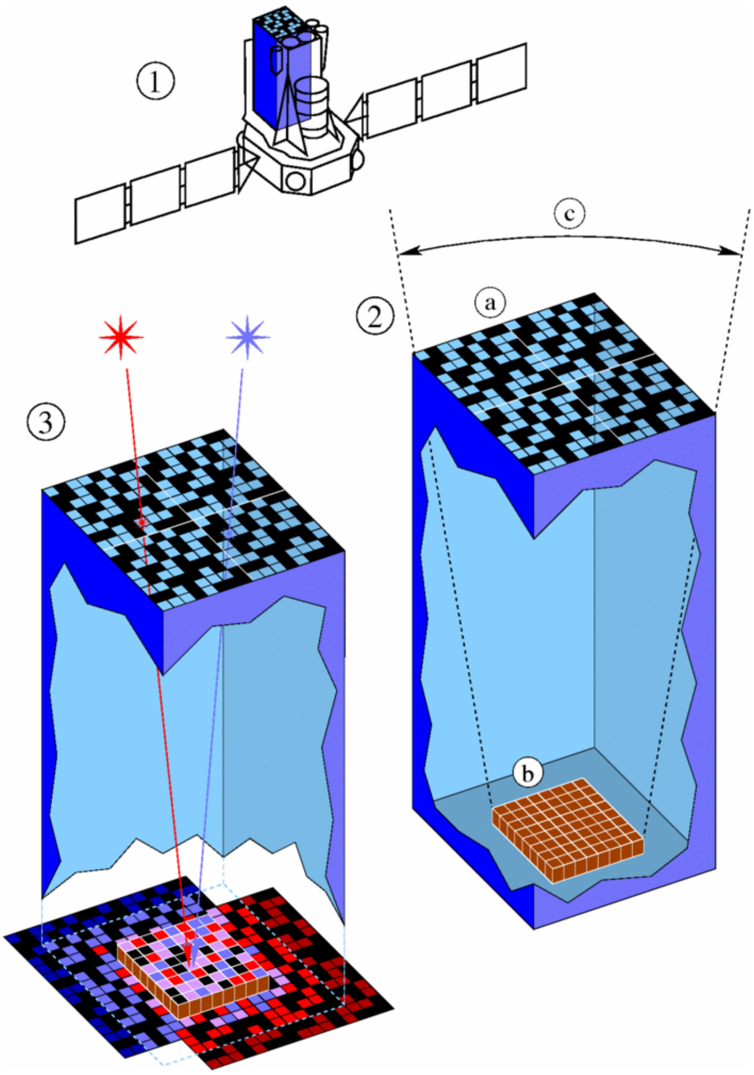

Figure 3.3: The coded mask method (from `http://www.isdc.unige.ch/integral/images/medium/codedMask.gif`)

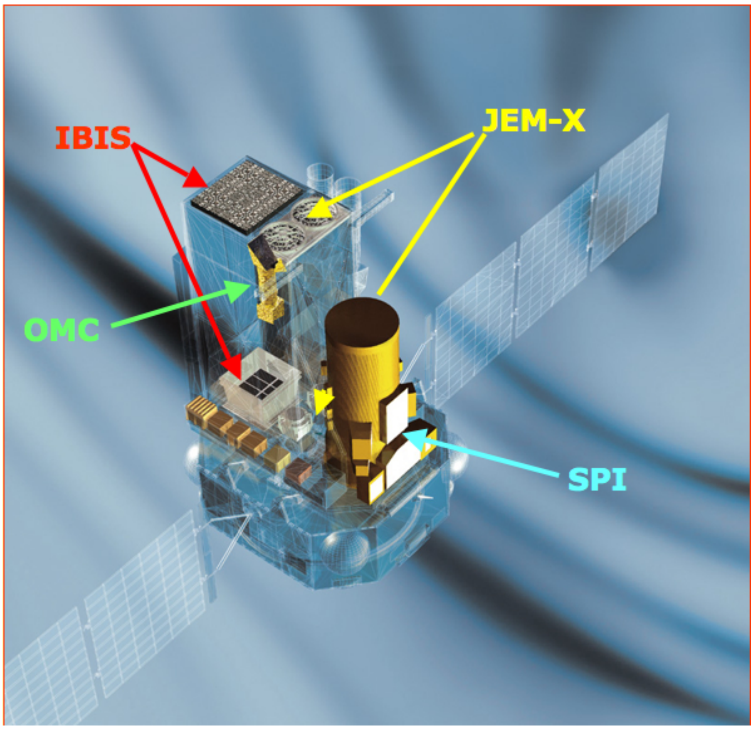

Figure 3.4: The four main instruments of INTEGRAL (http://www.cosmos.esa.int/documents/332075/979553/IntegralTransparentLabeled.jpg)

electrical power can be stored when the wings point to the sun and is released when the satellite is in the Earth's shadow or the wings can't be adjusted to the sun. According to the INTEGRAL user manual issue 5 the solar arrays consist of two wings which are covered with the silicon solar cells. Both wings are kept against the spacecraft with 4 kevlar cables in stowed configuration. After INTEGRAL separated from the launcher they are cut by thermal knives and the wings deploy. Each wing contains 3 rigid panels which are covered with solar cells. Each Panel is divided into 4 sections which are further subdivided into four solar cell strings. Fig 3.5 shows the schematic structure of the solar array (2002, INTEGRAL user manual issue 5).

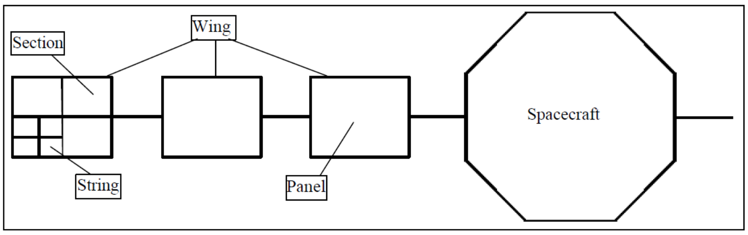

Figure 3.5: Basic schematic of a silicon solar cell (2002, INTEGRAL user manual issue 5)

4 Data Analysis

The following charts, tables and evaluations are made on the basis of the collected data of INTEGRAL from the launch of the satellite until now. These data include:

- the on-board time
- the distance from the satellite to the surface of the Earth
- the power of the different sections of the wings
- the temperature of the different sections of the wings
- the revolution number
- the x,y,z-position of the satellite to the surface of the Earth
- the x,y,z-velocity
- different particle counter of IREM

4.1 Orbit evaluation and radiation overlay

INTEGRAL stops collecting data when it comes close to the Earth. While the on-board time is recorded continuously, the position data tracking ends at some point and starts again when leaving the region near the Earth. The radiation monitor data are also not continuous, but the data gap is smaller than in the case of the position. Furthermore we have to know the position of INTEGRAL at any time to estimate whether the satellite has to deal with the radiation influence of the solar flux or of the Van Allen belts. Consequently an evaluation of the complete orbit with the help of the given data is necessary. As discussed in "Space Mission Analysis And Design" by James R.Wertz and Wiley J. Larson the motion of celestial bodies has challenged observers all over the world since many centuries. They especially dealt with the explanation of the planets motion. Since the early Greeks many other famous astronomer tried to solve this problem with their thesis. With the help of the research of Nicolaus Copernicus and especially Tycho Brahe's observational data, Johannes Kepler was able to describe elliptical planetary orbits about the sun. He published his three laws of planetary motion which also apply to satellites orbiting the Earth. With the information of chapter 6.1 "Introduction to Astrodynamics" by Daryl G. Boden from the United States Naval Academy it is possible, using the satellite equations of motion, to calculate the full orbit out of the following data:

- the incomplete x,y,z-position
- the incomplete x,y,z-velocity
- the complete on board time

As the satellite stops recording data when it comes close to the Earth, the values at the apogee of one revolution are used to calculate the missing parameter. The equation for the standard gravitational parameter is

$$\mu = G \cdot M \tag{4.1}$$

where G is the universal constant of gravitation and M is the mass of the Earth. Using the two-body equation of motion it is possible to assess several constants of motion of a satellite orbit. Amongst others the total specific energy is

$$\epsilon = \frac{v^2}{2} - \frac{\mu}{r} \tag{4.2}$$

ϵ is also called the mechanical energy per unit mass for the system and is the sum of the kinetic energy per unit mass and potential energy per unit mass. Eq. 2.2 is called the energy equation and can be transformed into

$$V = 2 \cdot \sqrt{\epsilon + \frac{\mu}{r}} \tag{4.3}$$

Consequently the energy equation reveals that a satellite moves fastest at perigee of the orbit and slowest at apogee. The next constant of motion which is linked to a satellite orbit is the specific angular momentum. In detail it's the satellite's angular momentum per

Table 4.1: Explanation of the classical orbital elements from(`https://marine.rutgers.edu/cool/education/class/paul/orbits.html#3`) and (2010, Space Mission Analysis And Design, Chapter 6.1)

a	semi-major axis: describes the size of the orbit.
e	eccentricity: defines the shape of the orbit. For e = 0 the orbit is an perfect cycle
i	inclination: the angle between a satellite's orbital plane and the equator of the Earth. This defines the orientation of the orbit related to the Earth's equator.
Ω	Right Ascension of the Ascending Node: the ascending node is the point where the satellite crosses the Earth's equator while going from the Southern Hemisphere to the Northern Hemisphere. Because of the Earth's rotation a fixed object in space is needed. This can be the Aries constellation or the vernal equinox as it is the same location. The angle between the vernal equinox and the ascending node is called the right ascension of the ascending node.
ω	argument of perigee: the angle formed between the ascending node and the perigee direction respectively the eccentricity vector.
θ	true anomaly: the angle from the perigee direction and respectively the eccentricity vector to the satellite position vector. It is also possible to calculate the true anomaly with the help of the time since perigee passage T.

mass. The cross product of the position and velocity vectors

$$\mathbf{h} = \mathbf{r} \times \mathbf{v} \tag{4.4}$$

provides the specific angular momentum.

The next step is the evaluation of the classical orbital elements.

With the help of Fig 4.1 and the definitions in Tab 4.1 it is possible to solve for the orbital elements if the satellite position and the velocity vectors are given. The energy and the angular momentum vector are already determined with help of the equations 2.2 and 2.4. However two parameter are still needed. The nodal vector in the direction of the ascending node is

$$\mathbf{n} = \mathbf{Z} \times \mathbf{h} \tag{4.5}$$

with $\mathbf{Z} = \begin{pmatrix} 0 \\ 0 \\ 1 \end{pmatrix}$

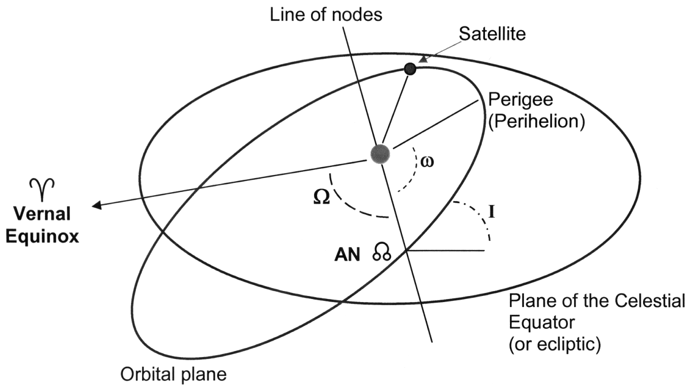

Figure 4.1: The orbital elements
(`http://spaceflight1.nasa.gov/realdata/elements/orbital_elements.gif`)

The eccentricity vector can be calculated with

$$\mathbf{e} = \frac{1}{\mu} \cdot \left\{ \left(V^2 - \frac{\mu}{r} \right) \mathbf{r} - (\mathbf{r} \cdot \mathbf{v})\, \mathbf{v} \right\} \tag{4.6}$$

Now all parameter are known to solve the equations for the classical orbital elements. The semi-major axis is

$$a = -\frac{\mu}{2\epsilon} \tag{4.7}$$

with the standard gravitational parameter μ and the total specific energy ϵ . Followed by the eccentricity

$$e =\mid \mathbf{e} \mid \tag{4.8}$$

The third classic orbital element is the inclination

$$i = \cos^{-1}\left(\frac{h_z}{h}\right) \tag{4.9}$$

The right ascension of the ascending node is

$$\Omega = \cos^{-1}\left(\frac{n_x}{n}\right) \tag{4.10}$$

Now the argument of perigee can be calculated with

$$\omega = \cos^{-1}\left[\frac{(\mathbf{n} \cdot \mathbf{e})}{(n \cdot e)}\right] \tag{4.11}$$

The last of the six elements is the true anomaly

$$\theta = \cos^{-1}\left[\frac{(\mathbf{e}\cdot\mathbf{r})}{(e\cdot r)}\right] \tag{4.12}$$

Now all the classic orbital elements are known (2010, Space Mission Analysis And Design, Chapter 6.1). An additional Parameter, the orbital period can be calculated with

$$T = 2\cdot\pi\sqrt{\frac{a^3}{\mu}} \tag{4.13}$$

However there is one further parameter needed. Since INTEGRAL stops the position tracking when it comes close to the Earth, the time of perigee τ is not known. As there is no direct formula, this parameter has to be calculated with the help of the true anomaly θ . Therefore we define a τ_{test} which is, at first, equal to the on board time. A step size $\Delta\tau$ as well as a θ_{test} is defined. At last we need a defined accuracy. Now the mean anomaly can be calculated with

$$M = \frac{2\cdot\pi\,(t-\tau)}{T} \tag{4.14}$$

where t is the on board time, tau is τ_{test} and T is the orbital period. The eccentric anomaly E can be calculated out of the Kepler equation

$$M = E - e\cdot\sin(E) \tag{4.15}$$

The true anomaly is

$$\theta = 2\cdot\tan^{-1}\left(\sqrt{\frac{1+e}{1-e}}\cdot\tan\left(\frac{E}{2}\right)\right) \tag{4.16}$$

and is from now on the new θ_{test}. Afterwards a comparison is made. If the difference from θ_{test} and θ we get out of Eq. 2.12 is bigger than the accuracy, τ_{test} is reduced by $\Delta\tau$. Now the equations 2.14, 2.15 and 2.16 have to be calculated again. This will go on till θ_{test} is bigger than the θ we calculated out of Eq. 2.12. Then the last step size has do be added to τ_{test} and afterwards the step size is divided by two. Then the whole process continues with the half step size. It finishes if θ_{test} is near the θ which is derived in equation 2.12 with respect to the chosen accuracy. The discovered τ_{test} is the wanted time of perigee τ . With the time of perigee and equation 2.14 a new mean anomaly can be calculated. After that we get a new eccentric anomaly out of the Kepler equation. The semi-minor axis results

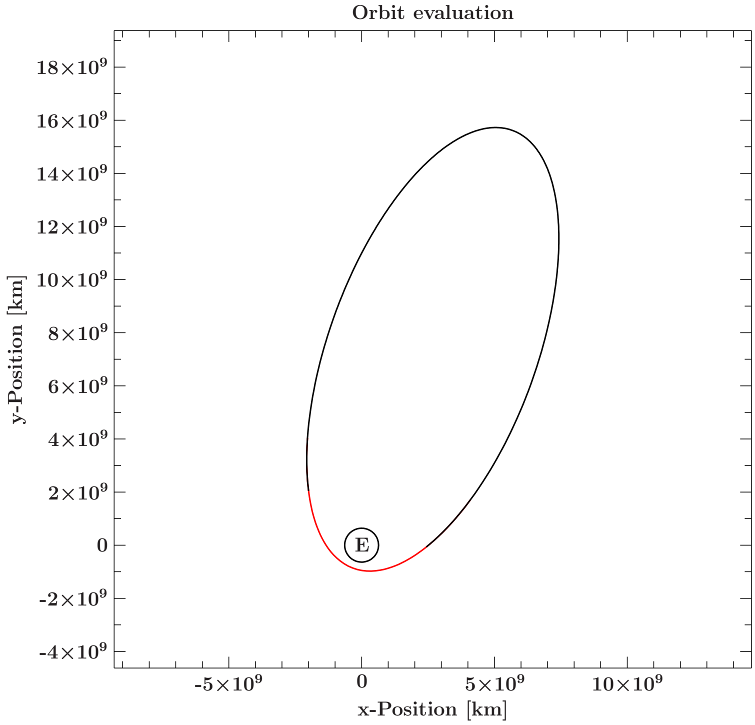

Figure 4.2: Full orbit in red and incomplete orbit in black for revolution 1100

from

$$b = \sqrt{a^2 \cdot (1 - e^2)} \tag{4.17}$$

All evaluations and the generation of the ellipse are made with the help of isis. Therefore isis creates an ellipsis out of the calculated data by using the semi-major axis a, the semi-minor axis b, the argument of perigee ω and the eccentric anomaly E. Fig 4.2 shows the orbit evaluation for the revolution number 1100 which is about 9 years after launch. The black incomplete ellipsis is made with the help of INTEGRAL data, whereas the red curve is the calculated orbit. Having the full orbit it is possible to overlay the radiation over the orbit. Fig 4.3 shows the complete orbit in red and the radiation in blue. Consequently you can observe where INTEGRAL has to deal with the highest rate of radiation. Therefore the blue curve is fitted to the orbit data and two scale curves are inserted. The outer scale is the maximum value which the radiation monitor detected during this revolution whereas the inner scale curve is the mean value. This value represents a count rate which means that this is a number of particles detected during this revolution. Whether it is a proton, an electron or an ion and in which energy range it is, depends on the chosen monitor. In Fig 4.3 the TC1 monitor of IREM is used. The shown radiation curve consists of protons in the energy range above 27 MeV. As expected the most radiation is detected in the region

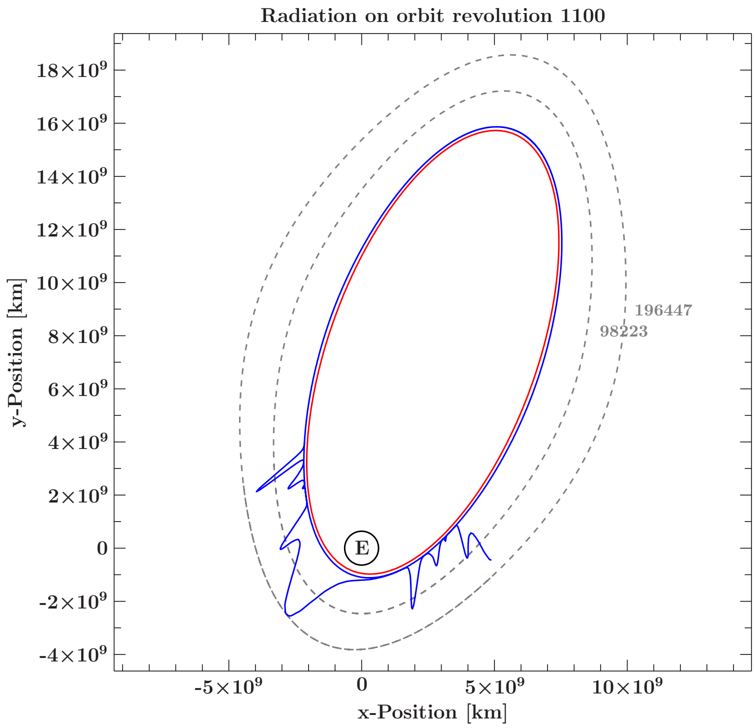

Figure 4.3: Radiation over orbit revolution 1100

near the Earth whereas the particle counter doesn't spot much radiation during the rest of the orbit.

4.2 Degradation explanation

The following data and plots are the unfiltered raw data. Fig 4.4 shows the current output degradation since launch exemplarily for one wing and one section. As expected the solar array current output constantly decreases since the satellite's launch. This degradation can be seen for both wings and all sections. Tab 4.2 shows the maximum current output within a year since launch. These values cannot be used to give information about the detailed energy household or to anticipate the future energy household as therefore the average values have to be taken. Furthermore you have to consider the orientation of the satellite, the direction of the solar panels and the ageing of units if you draw a comparison. However it is definitely possible to show the rate of degradation, as the maximum current output of the prior year is never achieved again. In general you see with Fig 4.4 that even the average output from the prior year is never achieved in the follow year again.

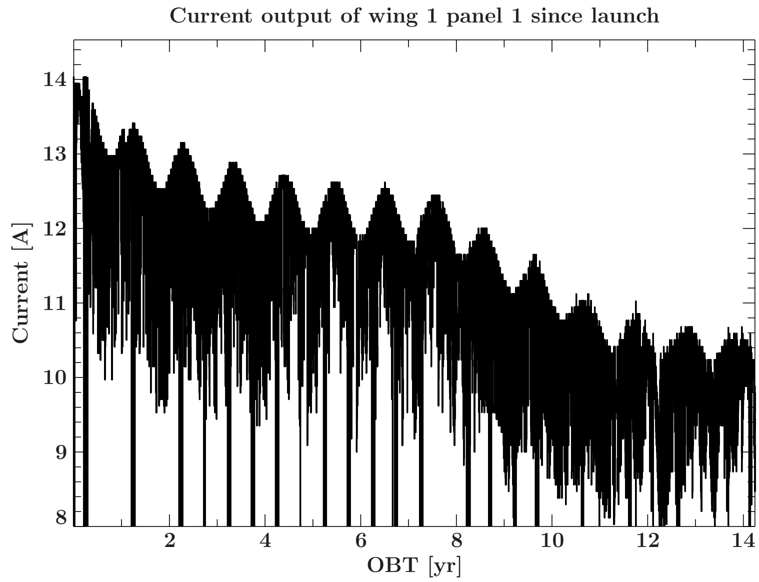

Figure 4.4: Current output wing one section one during since satellite launch

Table 4.2: Current output degradation with the help of maximum current output within a year [A]

		Maximum current within a year													
		1	2	3	4	5	6	7	8	9	10	11	12	13	14
Wing 1	P1	14.2	13.4	13.2	12.9	12.7	12.6	12.5	12.4	12	11.65	11	10.8	10.7	10.6
	P2	14.2	13.4	13.2	12.9	12.8	12.7	12.6	12.5	12.19	11.8	11.2	11	10.8	10.7
	P3	14.2	13.6	13.5	13.4	13.35	13.2	13.1	13	12.5	12.2	11.5	11.3	11.2	11.1
Wing 2	P1	14.2	13.7	13.6	13.4	13.3	13.25	13.2	13.2	12.9	12.75	12.4	12.25	12.2	12.1
	P2	14.2	14	13.8	13.6	13.55	13.5	13.4	13.4	13.1	12.8	12.55	12.3	12.2	12.1
	P3	14.2	14	13.75	13.5	13.4	13.3	13.2	13.1	12.9	12.7	12.4	12.25	12.15	12.05

Consequently we can conclude that it is not the case that the satellites energy management simply didn't want to achieve the output of the prior year but it was not possible to reach it again. As the Main Power Bus voltage is always 28 V it is possible to determine the maximum array output power as shown in Tab 4.3.

As expected the solar array output current has degraded since launch and this is influenced by several factors. The first aspect is the influence of the outer Van Allen belt. With the help of the literature values out of Tab 3.1 for the apogee and perigee it can be seen that the satellite has to pass the outer Van Allen belt within every orbit revolution. With this radiation, mainly high energetic electrons an some protons, the satellite has to deal its whole mission time since launch as well as in the future. This factor is responsible for

Table 4.3: Current output degradation with the help of maximum current output within a year [A]

Years since satellite launch	1	2	3	4	5	6	7
Maximum current of both wings within a year [A]	84.8	82.1	81.05	79.7	79.05	78.55	77.9
Maximum array output power within a year [W]	2374.4	2298.8	2269.4	2231.6	2213.4	2199.4	2181.2
Years since satellite launch	8	9	10	11	12	13	14
Maximum current of both wings within a year [A]	77.6	75.59	73.9	71.05	69.9	69.25	68.65
Maximum array output power within a year [W]	2172.8	2116.52	2069.2	1989.4	1957.2	1939	1922.2

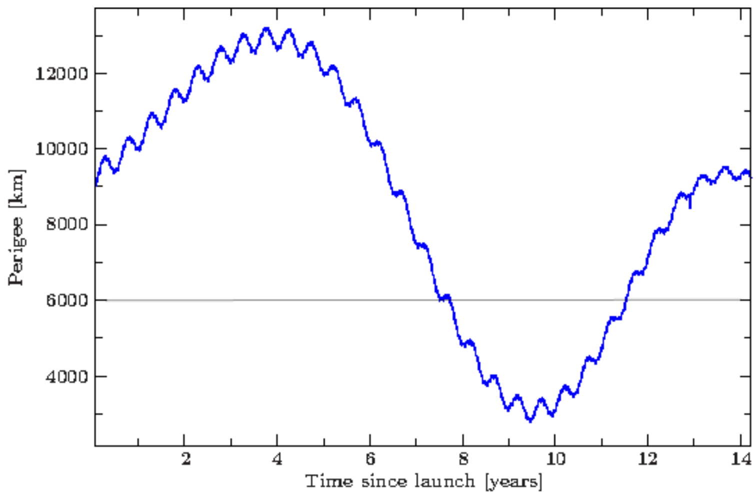

Figure 4.5: Development of INTEGRAL's perigee since launch

constant slow degradation during the whole time INTEGRAL is in space. Furthermore there are some aspects which only lead to a degradation in an special time frame. For example there exists a primary very rapid phase of degradation in the time shortly after the satellite's launch. For this behaviour i have no explanation. Then the above mentioned slow and steady degradation starts. The next exceptional higher rate of degradation occurs from early 2010 to October 2013. This clearly can be seen in Tab 4.2 examining the total output current within the years 8 to 12 since mission start. This degradation can partially be explained with Fig 4.5, which shows the perigee of INTEGRAL during the whole mission time. Till the period of higher degradation the perigee was above 6000 km. By reducing the perigee to 6000 km and below, the degradation rises up as INTEGRAL now enters the inner

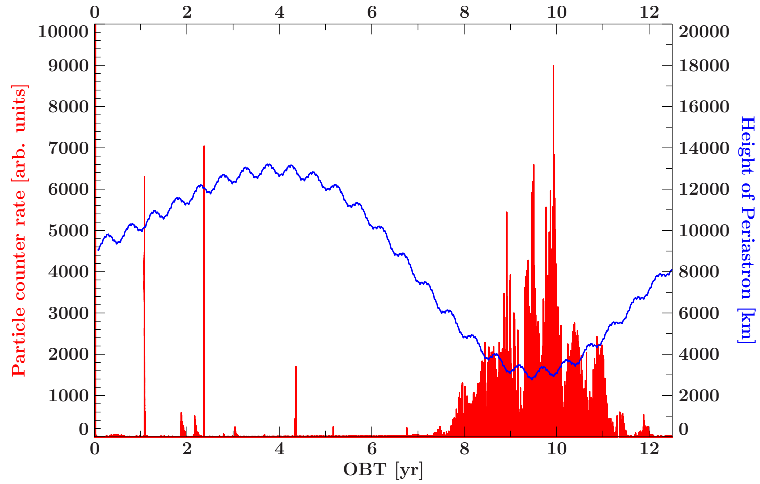

Figure 4.6: INTEGRAL's perigee and high energy proton particle counter C3

Van Allen belt. Now high energy protons harm the solar panels additionally. Fig 4.6 shows the IREM particle counter C3 which detects protons in the energy range between 450 MeV and 76 MeV together with the satellites perigee. This is the most sensitive proton detector as all other counters measure the count rate of lower energy ranges. Consequently these are high energy protons which lead to major damage at the solar panels. As expected the counter remains silent most of the time, changing its behaviour when INTEGRAL comes closer to the 6000 km perigee in early 2010. There it detects many high energy protons in the environment around INTEGRAL till the satellites apogee is above 6000 km again. After that the Fig 4.4 shows that the rate of degradation decreases to the normal slow and steady degradation as the perigee is changed to 6000 km and above. Now the output current data are normalized to 1 AU and the results are displayed in Fig 4.7. Additional to the above mentioned degradation sections a very rapid output current loss 1 year after launch can now be seen. This was the time between October and November 7 2003 called the "Halloween Storms of 2003". During this time some of the most powerful solar flares ever recorded erupted on the sun. Such flares cause the sun's magnetic field lines to expand and suddenly flicks beyond its limit. As a result coronal mass ejections were released. These are enormous explosion on the sun's surface which float huge amounts of electrified gas and subatomic particles with high speed into space. This space weather caused re-routing of aircraft, radio and communication problems plus power outages.[1] At least one of these solar flares caused damage at INTEGRAL's solar panels resulting in an rapid output current drop. However the degradation is not only caused because of the

1 http://www.nasa.gov/topics/solarsystem/features/halloween_storms.html

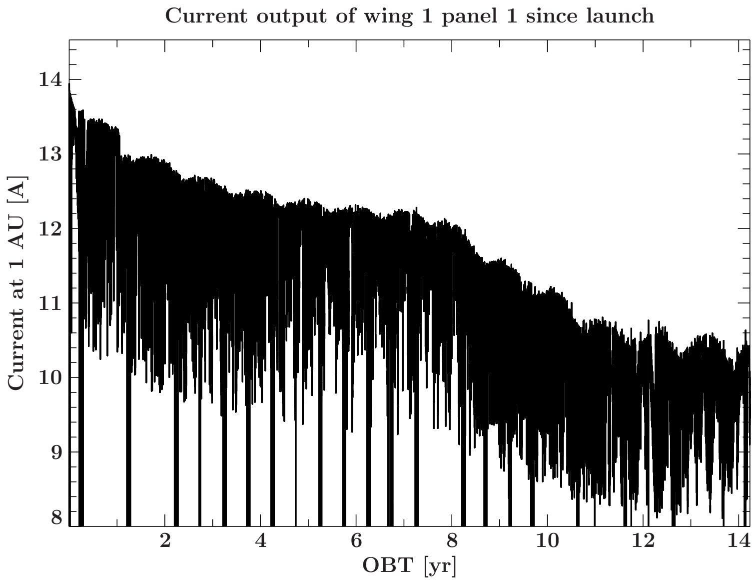

Figure 4.7: Output current of wing 1 panel 1 normalized to 1 AU

the Van Allen belts and the perigee variation. In fact the solar activity is directly linked to the degradation rate. An comparison of the 10.7 cm solar flux and the output current normalized to 1 AU reveals this connection. Therefore the Sun's 10.7 cm solar flux, also called radio flux, is shown in Fig 4.8 together with the output current of wing 1 panel 1. The 10.7 cm solar flux is proportional to the solar activity. In Fig 4.8 a clearly general trend can be seen. The slow and steady rate of degradation till early 2010 is directly linked to the decreasing solar activity. As the exceptional higher rate of degradation occurs after about 8 years since launch, the solar activity also increases. An eye-catching event is the already mentioned influence because of the solar flares in October 2003. Fig 4.8 shows that one year after launch the output current suddenly decreases while the solar flux and therefore the solar activity has an exceptional high peak. To sum it up the degradation is caused by two main reasons. On the one hand there is the influence of the Earth's radiation belts and on the other hand it is directly linked to the solar activity. Furthermore INTEGRAL is in danger of being damaged by space debris or micro meteorites which hit the panel, but an actual damage cannot be proven.

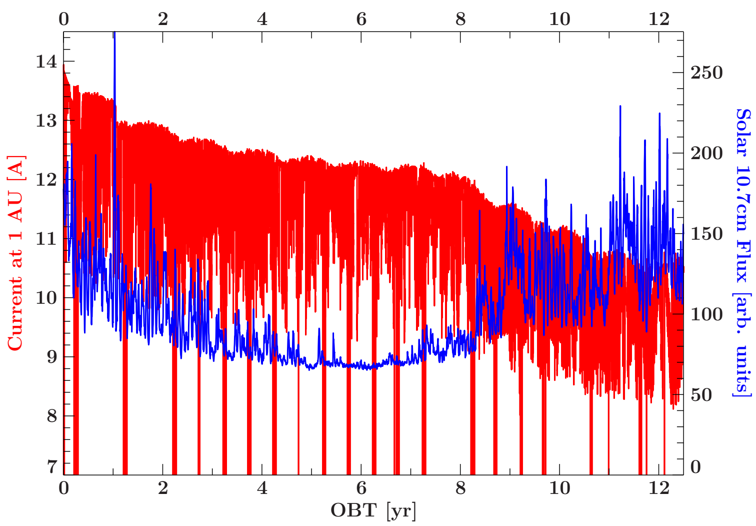

Figure 4.8: 10.7 cm solar flux compared with the output current of wing 1 panel 1

5 Summary

From early 2014 till now INTEGRAL has remained above an altitude of 6000 km and consequently was harmed only by the normal slow and steady degradation of the outer Van Allen belt. This will approximately continue till the middle of the year 2018. Then the perigee will again drop among 6000 km. Consequently the estimated degradation because of the lower perigee will increase like in the years from early 2010 to 2013. To predict the degradation caused by the solar flux is however more complex. According to Richard Southworth the sum of all arrays currents normalised to 40 degrees pitch angle will drop below the maximum observed main bus current. Providing that all other components will work properly, the satellite should be able to continue its work without any restrictions till 2020. After 2020 however the energy household has to be planned very carefully. This means that the mission still can go on but not all instruments can be used simultaneously. The inevitable end of INTEGRAL will be in 2029 when it performs a save re-entry in the Earth's atmosphere.

A Appendix

A.1 Orbit evaluation and radiation on orbit overlay

```
% function to calculate the true anomaly at a given time,
% t, depending on the time of perigee passage, tau,
% the eccentricity, ecc, and orbital period, porb.
% call: trueanomaly(t, tau, ecc, porb);
define trueanomaly(t, tau, ecc, porb) {
  % mean anomaly
  variable M = 2*PI*((t - tau) mod porb / porb);
  while (M < 0) { M += 2*PI; }
  % eccentric anomaly
  variable eccanom = KeplerEquation(M, ecc);
  % true anomaly
  variable theta = 2*atan(sqrt((1.+ecc)/(1.-ecc)) * tan(.5*eccanom[0]));
  if (theta < 0) { theta += (2*PI); }
  return theta;
}

% determine the orbital parameters from the position and
% velocity vector of the satellite at a the given time,
% t, in the gravitational field of a central mass, M.
% call: orbit_from_r_v(r, v, M, t);
%   or  orbit_from_r_v(rx, ry, rz, vx, vy, vz, M, t);
define orbit_from_r_v() {
  variable r,v,M,t = NULL;
  variable x,y,z,vx,vy,vz;
  switch(_NARGS)
  { case 3: (r,v,M) = (); }
  { case 4: (r,v,M,t) = (); }
  { case 7: (x,y,z,vx,vy,vz,M) = ();
          r = vector(x,y,z); v = vector(vx,vy,vz); }
  { case 8: (x,y,z,vx,vy,vz,M,t) = ();
          r = vector(x,y,z); v = vector(vx,vy,vz); }

  % potential energy (eq. 6-1)
  variable mu = Const_G*M;
  % total energy (eq. 6-4)
  variable e = vector_sqr(v)/2 - mu / vector_norm(r);
  % angular momentum (eq. 6-7)
  variable h = crossprod(r, v);
  % nodal vector (eq. 6-9)
  variable n = crossprod(vector(0,0,1), h);
  % eccentricity vector
  variable ev = 1./mu * ((vector_sqr(v) - mu / vector_norm(r))*r - dotprod(r,v)*v);
  % orbital elements (table 6.2)
  variable orb = struct {
    a     = -.5*mu / e,
    ecc   = vector_norm(ev),
    i     = acos(h.z / vector_norm(h)),
    Omega = abs((n.y < 0 ? 2*PI : 0) - acos(n.x / vector_norm(n))),
    omega = abs((ev.z < 0 ? 2*PI : 0) - acos(dotprod(n,ev) / (vector_norm(n) * vector_norm(ev)))),
    porb,
    theta = abs((dotprod(r,v) < 0 ? 2*PI : 0) - acos(dotprod(ev,r) / (vector_norm(ev) * vector_norm(r))))
  };
  orb.porb = 2*PI * sqrt((orb.a)^3 / mu);

  % time of perigee pessage
  if (t != NULL) {
    variable tau_test = t, delta_tau = orb.porb/10.;
    variable theta_test = 10;
    variable eps = 1e-9 ;
    while (abs(theta_test - orb.theta) > eps) {
       tau_test = tau_test-delta_tau;
       theta_test = trueanomaly(t, tau_test, orb.ecc, orb.porb);
       %vmessage("theta=%.2f theta_test=%.2f tau_test=%.0f", orb.theta, theta_test, tau_test);
       if (theta_test > orb.theta) {
         tau_test = tau_test+delta_tau;
         delta_tau=delta_tau/2.;
       }
    }
  }
  %   return tau_test
  orb = struct_combine(orb, struct { tau = tau_test });
  return orb;
}

define get_r_v(t, orb) {

  variable M = 2*PI*((t - orb.tau) mod orb.porb / orb.porb);
  variable eccanom = KeplerEquation(M, orb.ecc);
```

```
  variable b = sqrt(orb.a^2 * (1-orb.ecc^2));
  variable r = vector(0,0,0);
  (r.x, r.y) = ellipse(orb.a, b, orb.omega, eccanom);
  r.x -= (orb.a*orb.ecc*cos(orb.omega));
  r.y -= (orb.a*orb.ecc*sin(orb.omega));
  r = vector_rotate(r, vector(1,0,0), orb.i);
  r = vector_rotate(r, vector(0,0,1), orb.Omega);
  return r;
}


% Laden gebraeuchlicher Skripte
require("isisscripts");
% load our new functions
()=evalfile("orbitparameter.sl");
% load data
variable data = fits_read_table("/userdata/data/kreykenbohm/integral/panels/panel_database_reduced2.fits");
variable data2 = fits_read_table("/userdata/data/kreykenbohm/integral/panels/panel_database_coarse.fits");
variable W=14; % Breite des Plot-Fensters


% mass of the Earth (in g)
variable m = 5.972e27;

% array element for testing
variable revolnum = 1100;
variable i = where(abs(data2.revol - revolnum) < 1);
% calculate the orbital parameters with the new function
variable k = i[where_max(data2.distance[i])];
variable orb = orbit_from_r_v( % all positions and velocities from km -> cm
  data2.xpos[k]*1e5, data2.ypos[k]*1e5, data2.zpos[k]*1e5,
  data2.xvel[k]*1e5, data2.yvel[k]*1e5, data2.zvel[k]*1e5,
  m, data2.obt[k]/1e6 % time from micro s -> s
);

%print(orb);

% kuenstliches drehen des orbits in die xy-ebene
variable orb_original = COPY(orb);
orb.i = 0; orb.Omega = 0;

% vollstaendigen orbit plotten
variable t = data2.obt[i[0]]/1e6 + [0:orb.porb:#1000];
variable r_test = get_r_v(t, orb);
% zentrum der ellipse/orbits
variable x0 = .5*sum([min_max(r_test.x)]), y0 = .5*sum([min_max(r_test.y)]);%, z0 = mean(r_test.z);
% galbe breite des plots
variable wrld = 12e9;
% plot initialisieren
variable pl=xfig_plot_new(W,W);
pl.world(x0-wrld, x0+wrld, y0-wrld, y0+wrld);
variable wrld = abs(diff(pl.get_world()[[1,0]])[0]);

pl.plot(r_test.x, r_test.y; color = "red");

% zentrum der ellipse plotten
% pl.plot(x0, y0; sym = "x");

% erde plotten
variable Rerde = 6370.*1000*100;
pl.plot(ellipse(Rerde, Rerde, 0, [0:2*PI:#100]); fillcolor = "steelblue", fill);

% perigaeum
% variable per = get_r_v(orb.tau, orb);
% pl.plot(per.x, per.y; sym = "x");

% orbit nochmal fuer jeden strahlungspunkt berechnen
i = where(abs(data.revol - revolnum) < 1);
t = data.obt[i]/1e6;
r_test = get_r_v(t, orb);

% Einfuegen der Strahlungsdaten
variable radiation = data.tc1_sum[i];

% Plotten der Strahlungsdaten
% variable k;
variable minz = -10000, maxz = 200000.;
variable scal = 3e9;

% Vektor vom Zentrum der Ellipse zu allen Position d. Satelliten
variable v = vector(r_test.x-x0, r_test.y-y0, 0);
variable len = vector_norm(v);
% die laengen
% nun zu jedem Vektor die Strahlung drauf addieren (in Richtung des Normalvektors)
variable drauf = (radiation - minz) / (maxz - minz) * scal;
variable vS = v + vector(v.x/len*drauf, v.y/len*drauf, 0);
pl.plot(vS.x+x0, vS.y+y0; color = "blue");
variable anzahl = 2;
variable rMax = max(radiation);
_for i (1, anzahl, 1) {
  drauf = (rMax*1.*i/anzahl - minz) / (maxz - minz) * scal;
  variable vG = v + vector(v.x/len*drauf, v.y/len*drauf, 0);
  pl.plot(vG.x+x0, vG.y+y0; color = "gray", line = 1);
  pl.xylabel(max(vG.x) + x0, y0 + i*8e8, sprintf("%.0f", rMax*1.*i/anzahl),
    -.6, 0; color = "gray", size = "footnotesize"
  );
}

pl.title("Radiation on orbit revolution 1100");
pl.xlabel("x-Position [km]");
pl.ylabel("y-Position [km]");
```

```
pl.xylabel(0,0,"E");

% Dateiausgabe
pl.render("radiationonorbitfullfile.pdf");
```

A.2 Output current normalised to 1 AU and 10.7 cm solar flux implementation

```
#!/usr/bin/env isis

require("isisscripts");
variable a=fits_read_table("/userdata/data/kreykenbohm/integral/panels/panel_database_coarse_new.fits[HKData]");

variable y=a.power_w1c1*0.08825; % conversion to Amperes
variable mjdlaunch=MJDofDate(2002,10,17);
variable mjd=mjdlaunch+a.obt/1e6/86400.;

%variable ndx=where(y>7); % eliminate missing data
%y=y[ndx];
%mjd=mjd[ndx];

% earth-sun distance in AU per
% http://physics.stackexchange.com/questions/177949/earth-sun-distance-on-a-given-day-of-the-year
% note: this is a hack, for production we should do this calculation
% exactly, but this is ok given that the submission of the BSc
% is only a week away...
variable jdyear=MJDofDate(2002,1,1); %date of periastron
variable dd=a.mjd-jdyear;

% distance earth-sun
variable dist=1.-0.01672*cos(2*PI*(dd-4.)/365.25);


% read 10.7cm flux of the Sun from
% http://lasp.colorado.edu/lisird/tss/noaa_radio_flux.dat?&time%3E=2002-10-01&time%3C2016-02-01

variable yy;
variable mm;
variable dd;
variable radio;

()=readascii("noaa_radio_flux.dat",&yy,&mm,&dd,&radio);
variable ndx=where(radio>-99998.);
yy=yy[ndx];
mm=mm[ndx];
dd=dd[ndx];
radio=radio[ndx];

variable mjdradio=array_map(Double_Type,&MJDofDate,int(yy),int(mm),int(dd));

variable pl=xfig_plot_new(14,10);
pl.world(0.,12.5,7.,14.5);
pl.world2(0.,12.5,0.,max(radio));

pl.xlabel("OBT␣[yr]");
pl.ylabel("Current␣at␣1␣AU␣[A]";color="red");
pl.y2label("Solar␣10.7cm␣Flux␣[arb.␣units]";color="blue");

pl.plot((mjdradio-mjdlaunch)/365.25,radio;depth=50,world2,color="blue");

pl.plot((mjd-mjdlaunch)/365.25,y*dist^2.;depth=100,color="red");

pl.render("radioflux.pdf");
```

Erklärung

Hiermit versichere ich, dass ich die vorliegende Arbeit selbstständig verfasst und keine anderen als die angegebenen Quellen und Hilfsmittel benutzt habe, dass alle Stellen der Arbeit, die wörtlich oder sinngemäß aus anderen Quellen übernommen wurden, als solche kenntlich gemacht und dass die Arbeit in gleicher oder ähnlicher Form noch keiner Prüfungsbehörde vorgelegt wurde.

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I declare that I have authored this thesis independently, that I have not used other than the declared sources / resources, and that I have explicitly marked all material which has been quoted either literally or by content from the used sources.

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