Life with a weak Heart

Prolonging the Grace Mission despite degraded Batteries

J. Herman[1]
DLR, 82234 Wessling, Germany

A. Davis[2]
Center for Space Research (CSR), University of Texas at Austin, USA

K. B. Chin[3]
Jet Propulsion Laboratory (JPL), California Institute of Technology, USA

M. Kinzler[4]
EADS, 88039 Friedrichshafen, Germany

and

S. Scholz[5] and M. Steinhoff[6]
DLR, 82234 Wessling, Germany

The two Grace satellites were successfully launched on March 17, 2002 by a Russian Rockot launcher. GRACE not only was the first dual-satellite mission operated by GSOC, but it also was the first formation-flying occurring at an altitude below 500 km. The mission is extremely successful from a scientific point of view and the originally envisaged mission duration of 5 years has more than doubled by now. A follow-on mission is planned by the same partners for 2016 and JPL projects a new generation in the twenties, so there is a strong incentive to prolong GRACE and try to bridge the gap.

Infirmity comes with age and several components have deteriorated or are defunct. Nevertheless, the scientific goals can still be obtained to nigh on 100%. The major challenge for operations is posed by the degradation of the NiH_2 batteries. These are comprised of 20 cells packaged in the common pressure vessel (CPV) configuration. However, two cells have shorted out on Grace 1 and one on Grace 2. The available capacity of the operational cells is also severely degraded. The current operational capacity of the batteries is limited to ≤ 3 Ah as compared to the original nameplate capacity of 16 Ah.

This paper describes the special operations needed to prolong the mission despite the considerable power constraints. The first Section gives a general overview with emphasis on

[1]AOCS team lead, German Space Operations Centre, Oberpfaffenhofen, 82234 Wessling, Germany/ Jaap.Herman@dlr.de
[2]Grace systems engineer/ adavis@csr.utexas.edu
[3]Power subsystem and battery engineer/ keith.b.chin@jpl.nasa.gov
[4]PTS engineer, Astrium-EADS, 88039 Friedrichshafen, Germany/ Markus.Kinzler@astrium.eads.net
[5]PTS team lead, German Space Operations Centre, Oberpfaffenhofen, 82234 Wessling, Germany/ Sina.Scholz@dlr.de
[6]Grace flight operations manager, German Space Operations Centre, Oberpfaffenhofen, 82234 Wessling, Germany/ Michael.Steinhoff@dlr.de

the components relevant to this paper. The battery, its current state and the mission specific circumstances which require special handling are described in Section 2. The several threats to and failure mechanisms of NiH_2 batteries are also presented here. The third Section then contains a detailed description of all measures taken to pamper the batteries. This includes heater and parameter settings, special on-board macros, orbit-to-orbit charge regulation, but also physical actions such as turning the satellites away from the Sun to force battery discharging and subsequent charging. The fourth Section, finally, presents conclusions, recommendations and an estimation of how long the Grace mission can be prolonged.

I. Introduction

The GRACE mission - "**G**ravity **R**ecovery **A**nd **C**limate **E**xperiment" - is a scientific co-operation between the USA and Germany. The two identical satellites were designed and built by Astrium in Germany. All operations are carried out at the German Space Operations Centre (DLR-GSOC), whereas the scientists are from the University of Austin in Texas and the Geoforschungszentrum Potsdam, Germany. The on-board instrument processing unit (IPU) is under the responsibility of JPL (Jet Propulsion Laboratory, USA).
The main scientific goal of the mission is to collect data for creating both static and time-varying Earth gravitational field models of unprecedented accuracy. This is done by measuring relative variations in satellite separation down to 1 μm/sec, using a microwave link between the two spacecraft that are flying on a polar orbit at an altitude of currently ~450 km and that are kept at a distance of 220±50 km. Study of time dependencies, yielding e.g. the long-term development of polar or glacier ice masses, or of the water masses in the Amazonas basin, gained in importance over the years and is by now the strongest incentive to prolong the mission as long as possible.
A schematic view of Grace is shown in Fig. 1. Of specific interest in this paper is the location of the solar panels. One smaller panel with an effective area 1.6 m^2 is on top of the satellite and the two larger panels, which have an effective area of 2.5 m^2 each, are on the 50° slanted sides.

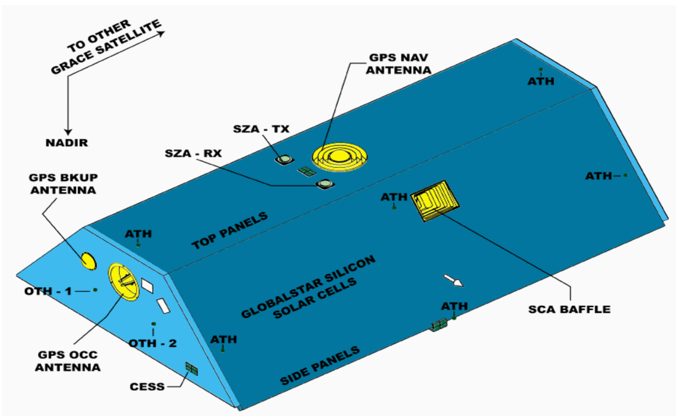


Figure 1. **Schematic of one of the Grace satellites**. *Solar panels are located on both sides and on the top. The micro-wave assembly providing the link to the other satellite is at the front and not visible in this picture. The 18 kg NiH_2 battery is on the bottom plate near the centre of the satellite.*

The mission now continues in its 11^{th} year and still resources are sufficient for at least 3 – 4 years more. Orbital height and front-end oxidation of the "follower" both benefit from the relatively weak solar maximum – without orbit raise or satellite swap life predictions still reach into 2016. More than 1/3 of the cold gas fuel is still available (see Herman *et al.*, 2004 and also Herman and Steinhoff; these proceedings).

Table 1 gives an overview of the several components that started to show signs of weakening with age. A few failed altogether (or are suspected to have failed, which has not yet been confirmed), but for each of these there still is a fully operational back-up in place. A major OBDH software upgrade was implemented in 2006, primarily in order to

be prepared for a further degradation of the CESS (which luckily did not happen). The battery design allowed for 22000 charge/discharge cycles of nominal depth (Mank et al., 2001), by now far more than 40000 cycles have been made. The considerable degradation of the available capacity furthermore implies that the *relative* DoD (depth of discharge) is now much higher than at the beginning, a fact that the battery appears to cope with surprisingly well.
Eq. 1, 2 and 3 show that nominally the battery yields 20x1.4 = 28 V with variations due to temperature and location in the orbit of the order of ±4 V. The loss of one cell decreases the average voltage by ~1.4 V.

GRACE 1	Since	Redundancy / Fully operational	**GRACE 2**	Since	Redundancy / fully operational
Failed components			**Failed components**		
IMU	Launch	No / No			
USO main	17.03.02	Yes / Yes			
IPU redundant (?)	10.03.12	Yes / Yes	IPU redundant (TBC)	04.05.07	Yes / Yes
ICU main	22.05.02	Yes / Yes	ICU main (TBC)	12.09.06	Yes / Yes
3 CESS thermistors	2004 until	Yes / Yes	2 CESS thermistors	2004 until	Yes / Yes
(2 black –z; 1 silver +x)	2005		(1 black –z; 1 black –x)	2005	
2 battery cells	>22.08.10	Yes / No	1 battery cell	>13.08.07	Yes / No
(fade over several weeks)	>16.04.11		(fade over several weeks)		
			Transmitter main		Yes / Yes
Degraded components			**Degraded components**		
Battery	gradual	≤3 Ah (~15%)	Battery	gradual	≤3 Ah (~17%)
Star cameras	gradual	Enhanced noise	Star cameras	gradual	Enhanced noise
		(axis dependent)			(axis dependent)

Table 1. **Overview of components which do not meet the original specifications anymore**. *The major restriction is the natural degradation of the batteries, which requires special operations in the presence of eclipses. Otherwise full operations are still guaranteed. In some cases a switch to the back-up component was made after experiencing problems with the main, but the latter might still be functional. This, however, will not be tested as long as the back-up is working satisfactory (denoted by TBC in the table).*

II. Battery

The Grace polar orbit has a precession of about 1.1 °/day, or a 322 day period in the inertial frame. This means that roughly every five months the direction to the Sun is in the orbital plane (denoted as beta prime angle β' = 0°) and the satellites experience eclipses of up to 37 minutes at an orbital period of 94. The Sun illuminates primarily the top panel for not more than half an orbit (the rest it shines on bottom, front and back; the elevation runs from -20° to 90° and back to -20° again). In between │β'│ becomes >70°, which means the Sun is visible to the satellites all the time and primarily illuminates one of the side panels (see Fig. 2). Both geometries bring their own peculiarities for handling the batteries.

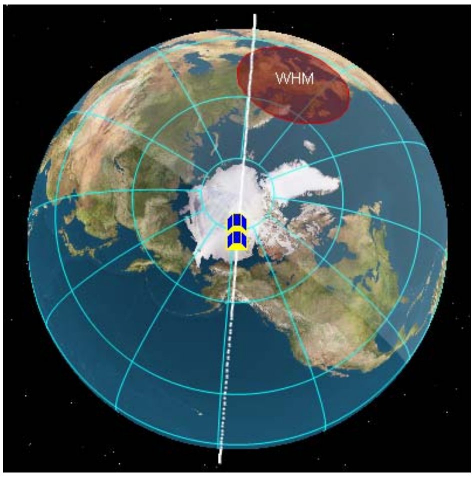


Figure 2. Grace polar orbit. *β is the angle between orbital plane and terminator. β' is the complement of β, i.e. β' = β ± 90° (Figure: Heavens-Above).*

Full-sun orbit
The solar panels provide more than enough energy for the spacecraft to operate. The batteries are fully loaded after the eclipse season ended and then receive only a small trickle charge rate in the order of 2% of the name plate capacity. There is no discharge whatsoever, which leads to a threatening situation for the weak cells present. There is no possibility on the Grace satellites to control the charge rate in this situation and so to compensate for the dynamics of degradation processes. Leakage will then result in an irreversible cell collapse due to electrolyte bridging if no counter measures are taken.

Eclipses
Power during eclipses is delivered by the batteries. The remaining capacity is used to the utmost, which is ~15%, or 2.6 Ah. Full re-charging must be done in the half orbit with sunlight with the additional restriction that the end-of-charge level (EoC) can only be set in discrete steps of ~0.3 Volt. The condition of the batteries puts stringent limits on the temperature, the amount of overcharge and the DoD, each shortly discussed below. The basic chemical reactions for a NiH_2 battery are as follows:

anode $\quad 1/2\, H_2 + OH^- \leftrightarrow H_2O + e^- \quad +0.83V$ **(1)**

cathode $\quad H_2O + e^- + NiOOH \leftrightarrow Ni(OH)_2 + OH^- \quad +0.52V$ **(2)**

net reaction $\quad 1/2\, H_2 + NiOOH \leftrightarrow Ni(OH)_2 \quad +1.35V$ **(3)**

where the arrows pointing towards the right show the processes when the battery is discharging, towards the left when it is charging. Around 20°C the efficiency is ~90%, but at higher temperatures a larger fraction of the current at the electrodes is converted into O_2. The oxygen recombines with H_2 into water and heat, thus leading to a thermal run-away (see Fig. 8 for an example) . The decrease in efficiency is a direct function of the production of oxygen. Temperatures have to be kept low therefore, ideally <10°C. However, the longer eclipses become, the higher the DoD and the higher the EoC level required for a sufficient re-charge. In practice this means that somewhat higher temperatures have to be accepted (≤15°C; see Fig. 7), but even then not enough charge can be obtained anymore for a period of about four weeks around the longest eclipses.

For a long time the only failure mechanism for NiH_2 batteries was thought to be loss of a cell due to a short circuit. Two factors were known to accelerate this, high operating temperatures and/or excessive overcharge (which also increases the temperature). Zimmerman[7] identified in 2009 a second failure mechanism for NiH_2 batteries in the CPV configuration (i.e. two cells share a common pressure vessel), namely electrolyte migration from one cell to the other. Historical life test data was used to determine the degree to which electrolyte migration gets worse by having inadequate overcharge. The failure mode has been reported in a number of missions[8].

Consequently there are two threats to the battery's health, one aggravated by too much and one by too little overcharge. This suggests that there should be an optimum level of over-charge, were the risk of failure is balanced between the two mechanisms. Zimmerman (2010) analysed the battery data for the entire mission of both Grace satellites and determined the point at which the risk for a cell short becomes higher than the risk of electrolyte bridging. The failure risk trade-off point is a function of the battery discharge load and orbit average temperature. The interesting part of this study for Grace battery management is above discharge loads of 0.5 Ah and is plotted in Fig. 3 separately for each satellite.

Because average battery temperature increases with the amount of overcharge, the operational objective is to keep the amount of overcharge between 60 and 80% of the point where the risk of shorting a cell is aggravated by too much overcharge. The empirical target for the overcharge of the Grace 1 battery is slightly higher than on the Grace 2 battery. The 60 to 80% operating rule has been applied for more than two β' cycles of 322 days each by now.

[7] A. Zimmerman; JPL internal investigation (2009)
[8] WMAP has a CPV failure due to electrolyte bridging, CloudSat a cell failure due to electrolyte dry-out attributable to excessive overcharge and DAWN is having CPV problems due to undercharge while still in cruise stage (see NESC, 2010 and references cited therein)

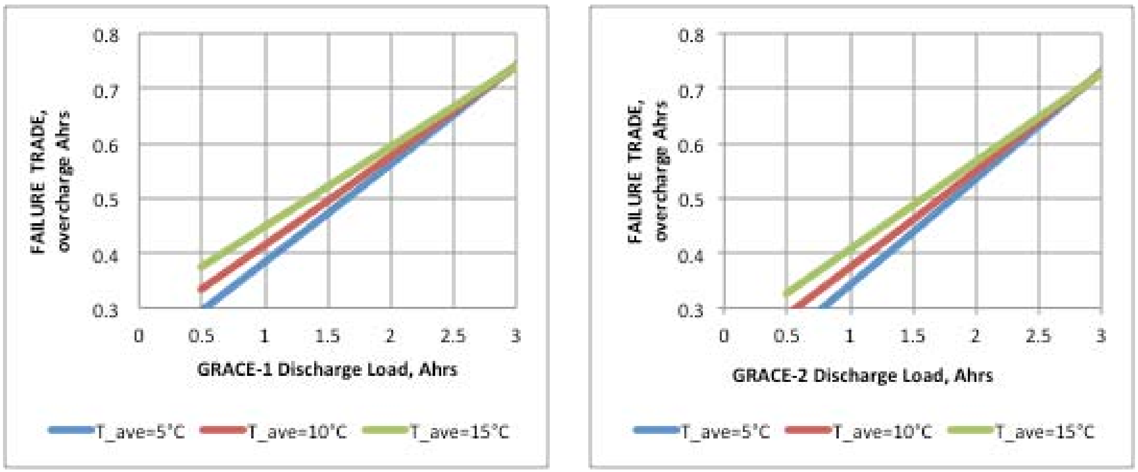


Figure 3. **Failure trade-off**. *The graphs show the point at which the risk of cell failure due to a short is larger than due to electrolyte migration. It defines the maximum amount of overcharge allowed as a function of battery discharge load and temperature (Grace 1 left, Grace 2 right).*

III. Operational solutions

All mitigation measures and solutions have to comply with the following operational constraints:

- discrete EoC steps (0, …, 15 are possible inputs), which determine the end-of-charge voltage depending upon temperature and battery charge current:

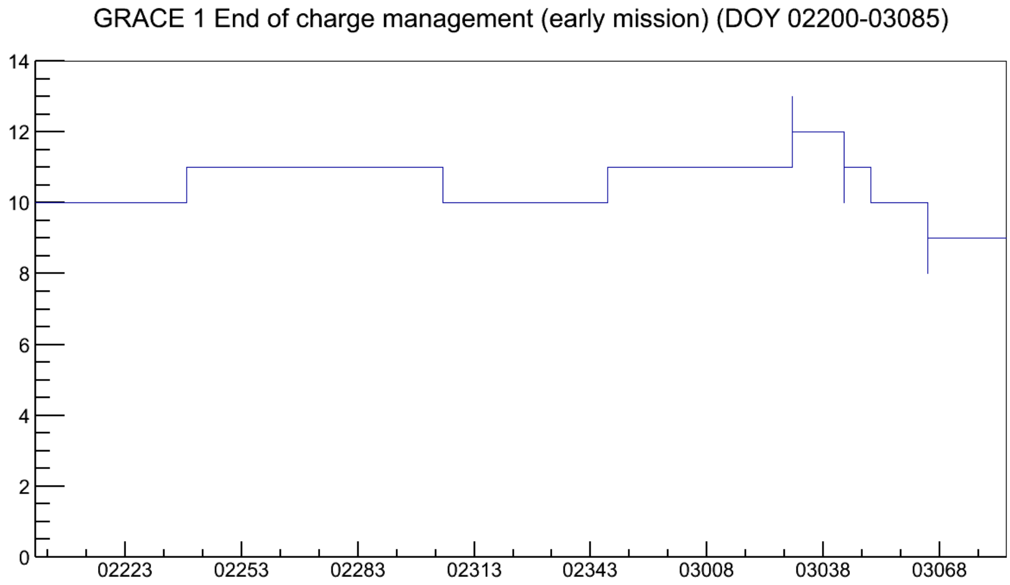
  $$V_{EoC} = 27.05 + 0.3*EoC + 0.2*(I-I_o) - 0.09*(T-T_o), \quad \textbf{(4)}$$

  where all coefficients are hard-wired and only EoC can be set (see Fig. 4 for an example)
- trickle charge, which operates once the specified V_{EoC} is reached, is fixed at 0.28A (see Fig. 6; the rate of 0.28A corresponds to ~2% of the nameplate capacity)
- high rate charge current can not be controlled; 18 shunts regulate the solar panel currents autonomously
- telemetry is limited to overall properties such as battery voltage or pressure, but offers additionally
  - the ½ battery voltage; monitoring 10 coherent cells simultaneously and building the ratio with the full voltage of all 20 cells: a failed cell will show up as a change of ±0.025 with respect to 0.5 (e.g. 9/19 = 0.474, or 10/19 = 0.526), whereby the sign indicates on what side the problem is located
  - a separate measure of charge- and discharge- current
  - the battery pressure as measured on two CPVs only
  - the temperatures measured on the base-plate
- data recorded on-board have to be dumped, transferred, processed and interpreted before improved commands for the battery settings can be sent to the satellite. The latency amounts to roughly 12 hours

Eq. 4 shows that in the original design the voltage at end-of-charge (V_{EoC}) would automatically be adjusted according to battery charge current (I) and temperature (T), i.e. adaptive to orbital phase. This was indeed maintained at the beginning of the mission with few manual changes of the EoC level (see Fig. 4, upper panel).

Current handling is completely different as is illustrated in the lower panel of Fig. 4. The V_{EoC} is controlled to values some 2 – 3 V lower than previously, on the one hand dictated by the loss of two cells on Grace 1 and on the other by the more stringent temperature requirements. The latter condition also implies a larger variation in EoC level between full-sun- and eclipse- phases.

The general strategy must of course be to minimise the amount of energy drawn from the battery by managing the depth of discharge per cycle. The load is reduced to a level that still guarantees full scientific return. The IMU on Grace 2 is permanently switched off unless the satellite goes to AOCS safe mode. Hardly any heaters are active anymore apart from the ones for the instrument. The satellites use the heat dissipated from the powered components and active heating only occurs if temperatures drop below certain minimum values. The transmitter is switched on for station contacts only with a margin which is just compatible with the accuracy of the predictions.



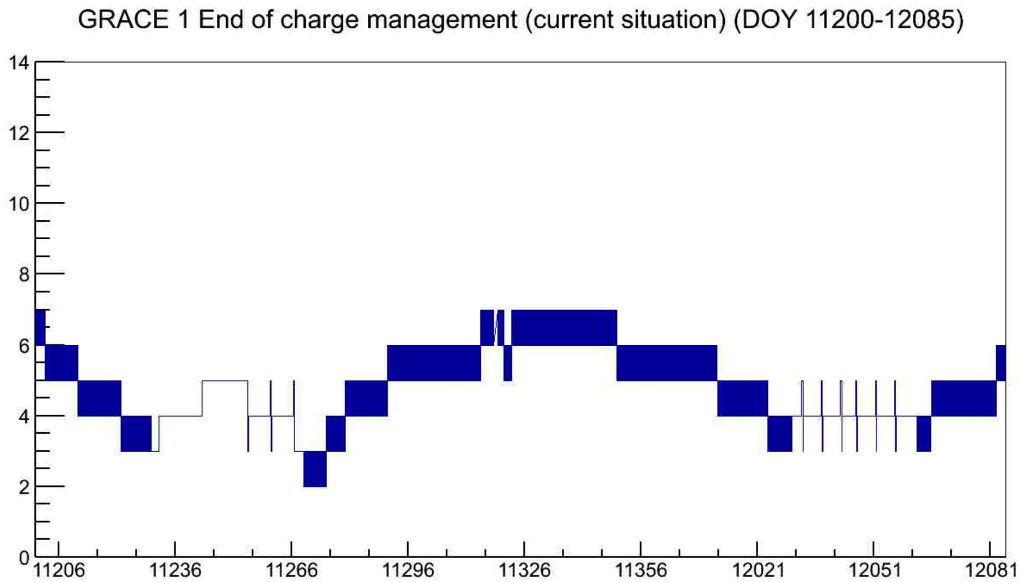


Figure 4. **Management of battery end-of-charge level in two instances on Grace 1**. *The upper panel shows a 250 day period at the beginning of the mission, the lower panel the current situation over an identical interval. Changes in EoC level at the beginning of the mission were few and far between. The mean EoC level of 10 was occasionnaly adjusted by plus or minus one step. The current situation is completely different. The average EoC level is appreciably lower, because Grace 1 lost two battery cells. The rapid EoC modulation with changes in each orbit are apparent by the seemingly filled areas (see text for full details). The seasonal variation stems from the 322 day orbit precession. The EoC settings first become lower when the satellites approach full-sun orbit with* $|\beta'| > 70°$ *and then modulation is stopped. The changes for the artificially induced eclipses, obtained by slewing the satellites in yaw, are clearly seen. In the last full-sun orbit six such yaw manoeuvres were made (see text for further details).*

The two phases, the one with eclipses and the other with full-sun (see Section 2.), each require different measures in addition to the ones listed above to keep the batteries alive.

Battery management in full-sun orbit

Fig. 5. illustrates the risk for a weak battery cell when it is not used in full-sun orbit. A second cell on Grace 1 started to fade on DOY 078 (19 March 2011) as diagnosed by the ½ battery voltage dropping below the 0.475 level (-0.025 corresponds to the first cell failed; the direction shows that the second weak cell occurs on the same side of the battery). The plot also shows the effect of the mitigation measures. The solar panels were turned away from the Sun by making -90° yaw-slews. This results in an artificial eclipse with the Sun illuminating the rear side of the satellite (near β' = ±90° there is also no Sun on the top panel). During the slew, "eclipse" and slew back the battery is thus forced to discharge and recharge with a high current rate to facilitate rapid cell recovery.

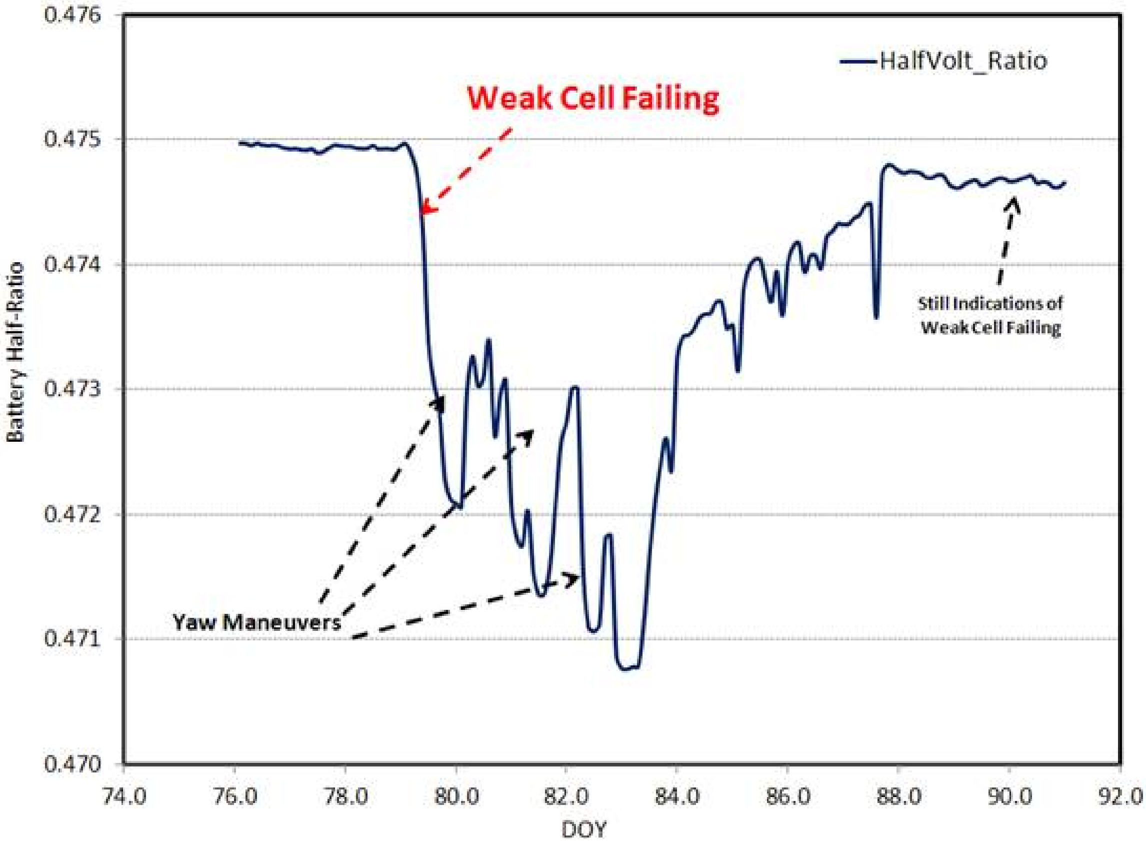


Figure 5. **Plot showing the battery half-voltage over a full-sun period in 2011**. *A second cell of the Grace 1 battery started to fade due to electrolyte bridging because there was no charge/discharge cycle. Artificial eclipses were introduced by making several 90° yaw slews which turn the side panel away from the Sun. The beneficial effect of this strategy is apparent although the cell did not fully recover (one completely failed cell corresponds to a change in the half-voltage of 0.025). Despite this successful recovery the cell was nevertheless lost a couple of weeks later.*

A detailed example of such an artificially introduced discharge/charge cycle on the battery is shown in Fig. 6. The green line is the yaw offset with respect to the flight direction with an extreme value of -1.5708 (-90°). Once the satellite starts turning away from the Sun, the current from the solar arrays (light blue) decreases until it reaches 0 A near 90° offset. At the same time the load on the battery rises (red curve) until the complete load is provided by the battery alone. The satellite is moved back to its nominal orientation after a short stay at -90° offset. The solar array current starts rising and the battery discharge current decreases accordingly. Once the SA provides enough power to serve the entire satellite, battery discharge becomes 0 and the remaining current from the solar arrays is used to re-charge the battery (dark blue). The battery charge process is terminated by a pre-set EoC level and the charge current returns to the fixed 0.28A trickle charge. Shunts regulate the SA current to the level needed for the load only.

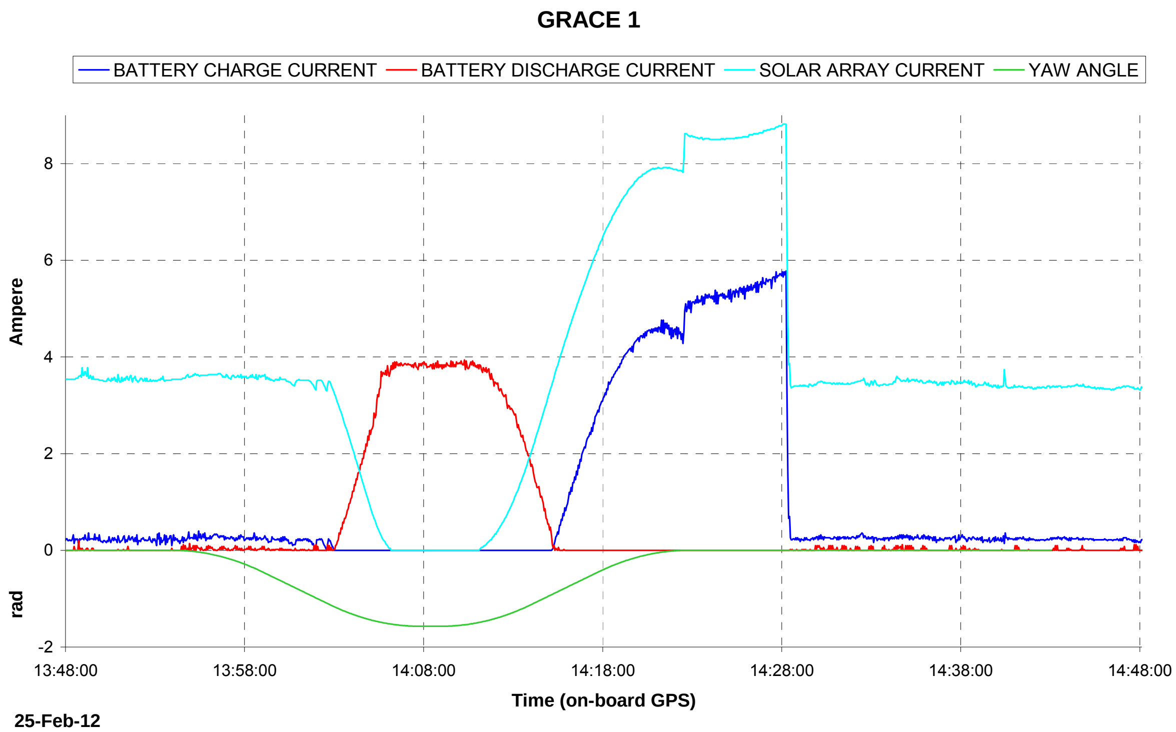


Figure 6. The effect of a -90° yaw turn (bottom) in full-sun orbit is shown. *Battery charge- and discharge-currents are plotted together with the input from the solar array. The solar array current is provided solely by the +y-side panel during the slew. The small jump at the end of the slew is possibly caused by additional current from the top panel, or by thermal expansion during the warm-up phase in sunlight.*

Optimum settings were obtained by the development of increasingly more complicated offline tools (A. Davis; 2012) and by in-orbit trial and error. The first manoeuvre was made 10.4.2007, when only one turn on one satellite – Grace 1 – was made in the middle of the full-sun period. By now, both satellites are turned away from the Sun every 4th or 5th day when in full-sun orbit (~6 such manoeuvres in total; see for example Fig. 4). A total discharge of 0.5 – 0.7 Ah was found to be optimum for battery management. The length of the discharge period and the specific moment during the slew back that re-charging starts, were optimised over the last few years.

Battery management in the presence of eclipses

Instances where station contacts come in quick succession or even overlap are reduced in order to minimise the on-time of the transmitter. Especially so-called “triplets” (these occur if a Grace satellite passes and uses Weilheim, NeuStrelitz and NyAlesund in one orbit) near the end of the eclipse are avoided.

The general measures to save energy soon proved to be insufficient to keep the state of the battery within the desired regime. The major challenge is to get high enough voltage at end-of-charge (V_{EoC}), yet to keep the temperature below 15°C (or even better <10°C). The control mechanism provided by Eq. 4. turns out to be too coarse; either the V_{EoC} is high enough but at the cost of too high temperatures, or the temperature is fine but the charge level too low.

The first try to achieve a finer granulation was to use EoC modulation – alteration between two levels, e.g. one orbit on 7 and two orbits on 6 to get approximately an effective EoC level of 6.33. This strategy was first applied end of

2010 and continued until 1 March 2011. The beneficial effect on the maximum temperature is apparent from Fig. 7; whereas before up to 24°C were reached near β'=0° now temperatures could be kept below ~15°C. The disadvantage of this method is the relatively long time-base. There might be two full orbits with too little and then one orbit with too much overcharge.

The next and currently used strategy is therefore to devise a method which optimises temperature and battery charge and overcharge each orbit. To this purpose the desired (over-)charge is determined from theoretical curves combined with extrapolated values of recent data (normally the last 24 hours available are taken).

$$\text{Desired} = (-0.003451 * \langle T \rangle + 0.212215) * (\text{Load} - 2.89) + 0.71 \quad [\text{Ah}] \tag{5}$$

The average temperature <T> and Load are the values measured on-board over the last day. Note that the linear fit is not valid for all parameter regimes, but applies only for the specific conditions on the Grace satellites..

This then yields a desired V_{EoC}, which is not necessarily corresponding to one of the discrete EoC levels. As an example consider the situation where one wants to reach a V_{EoC} of 29 V. This would according to Eq. 4 require an EoC of 6.5, assuming nominal temperature and charge current. The system only allows 6 (⇔ 28.85V) or 7 (⇔ 29.15V) as input for the EoC level. EoC 7 is commanded at the end of the eclipse and the battery starts loading and will reach successively increasing V_{EoC} values. Sometime, maybe 25 minutes after the end of the eclipse, the level corresponding to “EoC” = 6.5 is reached. Right at this moment a time-tagged command switches the charge regulator to an EoC one or two levels lower. The charge process will stop immediately and the desired decimal end-of-charge level is thus obtained exactly.
If by chance the desired level is at or near an integral EoC step, then the battery is allowed to continue charging until this level is actually reached. A safety switch back to a lower EoC level is set shortly after the predicted moment in this case to prevent a thermal run-away if the desired level for some reason could not be reached (might for example occur if one cell suddenly starts showing signs of weakness; see Fig. 8 for an example).

Several practical measures accompany this strategy. First of all, the EoC levels are set by commanding single bits (0000 through 1111) to either the prime- or back-up- charge regulator. This presented no problem as long as few changes per year were made (see Fig. 4), but 32 switches or 128 commands per day at a limited number of station contacts mean a considerable extra stress and also operational risk, because missing one command could result in a completely erroneous EoC level. On-board macros were designed therefore that set the correct EoC level with one single ground command. They are in use since June 2011. The macros have the additional benefit that if a ground command to the spacecraft fails, the setting is never off by more than one EoC step.

Another macro that was designed monitors the battery temperature. In case a pre-defined value is exceeded (at the moment of writing set at 17°C, but operational settings vary from ~10° to 20°C depending on orbital phase) a switch to the redundant charge regulator will be made, which is always kept at a lower EoC level than the main. This guarantees that the charge process will immediately be terminated if the temperature gets too high, in which case it will slowly return to normal (see Fig. 8).

The standard safety measures against a low voltage drop from the beginning of the mission also had to be revised. Obviously the limit for DNEL (disconnect non-essential loads) had to be lowered due to the failed cells and is currently at 21 V for Grace 1 and at 22 V for Grace 2. DNEL, which internally still has four subsequent levels of severity, is the last resort in case of low battery voltage and therefore another security is in place some 1.5 V higher. However, DSHL (disconnect standard heater lines) is no longer effective, because in essence all heaters are already turned off (see above). An intermediate step was therefore introduced which at a level 1.2 V above DNEL switches off two of the instruments together drawing about 0.8 – 1.0 A. The distance to DNEL is chosen such that normally this switch-off will allow the battery to recover in time.
At still higher level (e.g. 26.5 and 28.2 V for Grace 1) two further macros were built that toggle between two different heater tables. A thermal survival table and hence less power will be used when the voltage drops below the lower limit and a switch to the nominal settings with higher temperature set-points occurs as soon as the higher limit is reached again. The low heater set-points that are currently used make this ineffective at the moment and so these macros are disabled for the time being.

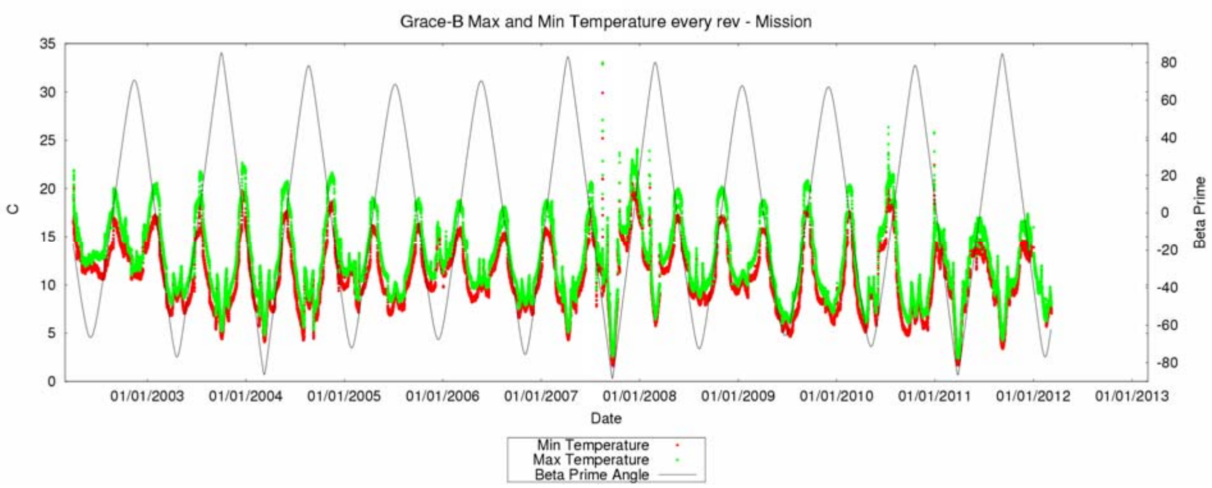


Figure 7. **Grace 2 battery temperatures over the entire mission**. *Temperatures in excess of 20° occurred regularly until stringent battery management was begun in 2010. The ~5° decrease is apparent from the plot. Also shown is the β' angle (axis on the right hand side). Full-sun orbit occurs for |β'|>70°. The longest eclipses, the highest load on the battery, largest DoD, highest EoC and highest temperatures occur near β' = 0°.*

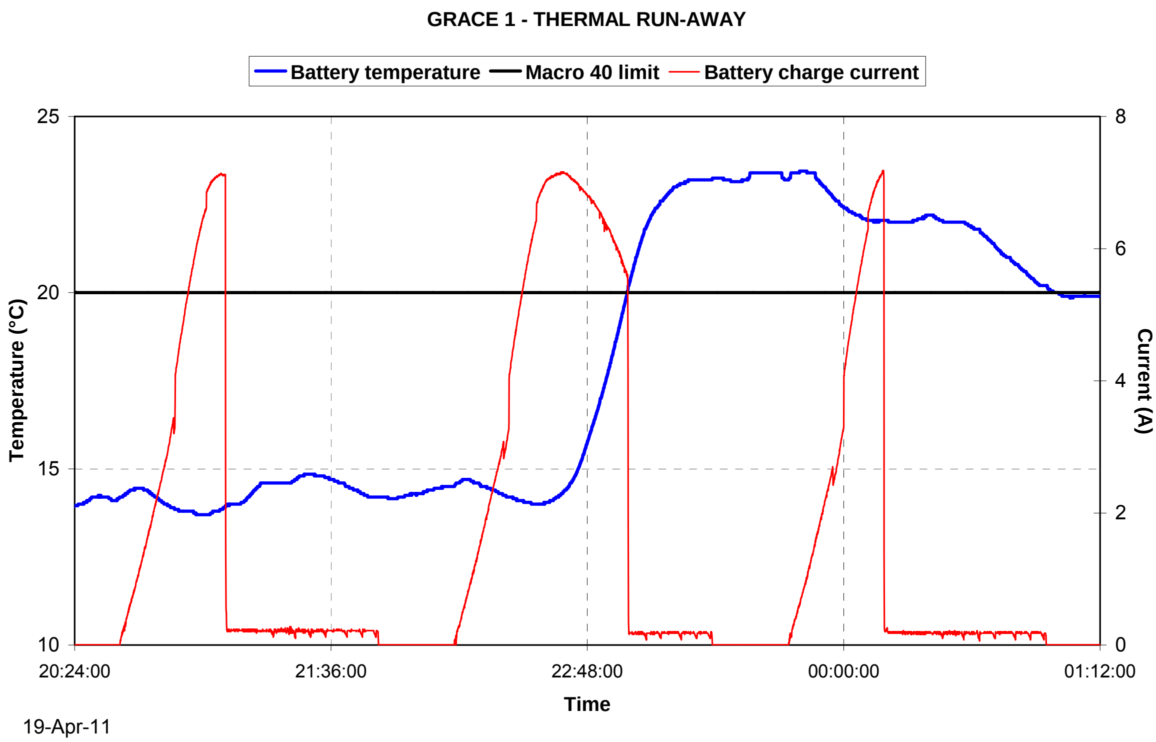


Figure 8. **On-board safety mechanism against thermal run-away**. *In the second orbit the pre-set V_{EoC} was apparently not reached and the battery continued to charge (red line). The temperature (blue line) was seen to rise sharply. At 20°C the on-board safety triggered a switch to the redundant charge regulator set at a lower level. Charge current immediately dropped to trickle and somewhat later to 0 A in eclipse. The temperature increase was stopped and then a slow recovery started which took several orbits.*

This strategy proved quite successful in keeping the battery alive and thus continuing the science on Grace as much and as long as possible. Only around β' = 0°, the phase with the longest eclipses and the worst geometry for providing solar array energy, additional measures are still necessary which unfortunately interrupt science. Two weeks before reaching β' = 0° the instrument control unit (ICU, using ~0.35 A) is switched off. In the future it might also become necessary to temporarily switch off the microwave assembly. The MWA draws ~0.55 A and is essential for the highest quality science data. Degraded scientific results are still possible as long as the MWA is operational. The interruption has been minimised by the continuous improvements described above. The first interruption in 2010 still lasted for 41 days, but around the last β' = 0° crossing Grace 1 instrument was off for 25 and Grace 2 for 19 days only. The real interruption is somewhat longer though, because it takes several days before the SuperStar accelerometers reach their stable operational temperature again.

Some ideas are around to improve on this even further. However, most are quite expensive in terms of operational effort and fuel expenditure. It would for example be possible to slew the satellites 90° in yaw at the end of each eclipse. This turns the more effective side panel towards the Sun and also catches sunlight earlier than the top panel does thanks to the 50° slant. This would allow us to keep the instrument completely powered, but a permanent yaw-bias interrupts science just the same! Two slews per orbit would allow science for about 30% of the time, but would deplete the fuel budget in two to three β' = 0° crossings. Discussions on possible solutions are still ongoing.

IV. Conclusions, outlook and recommendations

The current situation allows the mission to continue with little restrictions for at least a couple of years. One more cell on Grace 2 collapsed briefly near the end of 2011 but could be fully recovered. A failure of a second cell on Grace 2 would give the same situation as it is on Grace 1 now. Such occurrence would essentially not have much impact on the mission. Failure of a third cell on either satellite would bring about severe restrictions and limit science operations to periods with full-sun orbit. Fortunately, the careful daily optimisation of the battery promises to keep the remaining cells healthy to the end of the mission albeit with a severely degraded capacity of course. The natural degradation of the capacity could be slowed down by optimisation of the battery handling. Keeping the operational temperatures low, minimized electrolyte bridging and the rate of self discharge. Control of the amount of overcharge clearly stabilized the degradation rate associated with electrolyte bridging and other undesirable side reactions. No appreciable decrease has been observed on either satellite for the last couple of years.

The most important parameter for managing battery charge control was found to be the difference between the charge- and discharge- capacity [Ah] in each orbit. Operational constraints on the down-load of the data, the processing, transfer and interpretation lead to a reaction time of ~12 hours. It is strongly suggested that in future missions the charge/discharge difference is computed on-board and included in the telemetry. Direct telemetry for the accumulated heater on-times and e.g. their daily averages would also be advantageous.

Acknowledgements

The authors would like to thank S. Löw for his careful reading of the manuscript and for his helpful comments. The company Heavens-Above GmbH provided Figure 2 and Boris Grishechkin provided a version of the plot program ROOT for Grace.

References

Chin, K., Bugga, R., Davis, A., Gaston, R.: "*The Efficacy of Battery Management Strategies to Extend Life on the Grace Project"*, NASA Battery workshop; 2011
Davis, A.: *Grace Battery Analysis Tool* (project internal); 2012
Dunlop, J.D., Rao, G.M., Yi, T.Y.; "*NASA Handbook for Nickel-Hydrogen Batteries*", NASA-RP-1314; 1993
Herman, J., Presti, D., Codazzi, A., Belle, C.: "*Attitude control for Grace. The first low-flying satellite formation*", 18th International Symposium on Space Flight Dynamics, ESA SP-548, Munich, Germany, 2004
Herman, J., Steinhoff, M.: "Balancing, Turning, Saving; Special AOCS Operations to extend the GRACE Mission", These Proceedings, 2012
Mank, H., Fehrenbach, M., Petruschke, U.; 2001, "*GRACE Design & Interface DescriptionSatellite Power & Avionics*", GR-DSS-DID-002, Astrium, 2001
NESC (NASA Engineering and Safety Centre); "*Nickel-Hydrogen (NiH2) Common Pressure Vessel (CPV) Cell Capacity Loss and Voltage Collapse*", Technical Bulletin No. 11-01, 2010
Zimmerman, A.: "*Report on the investigation of the Grace mission battery data*", (NASA unpublished); 2010

Heavens-Above: website http://www.heavens-above.com/ and telemetry display program *satmon@home*