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End–of–Life Power Management on the Grace Satellites with several failed Battery Cells

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The two Grace satellites were launched on March 17, 2002 by a Russian Rockot. Both were operated for more than 15 years and delivered science data until the middle of 2017. The mission was successful from a scientific point of view and the originally envisaged mission duration of five years finally more than tripled. The data quality at the end was still at least an order of magnitude better than from any other source. A follow-on mission has been launched by the same partners in 2018 and NASA and DLR project a new generation of satellites for gravity field measurements in the next decade. Continuity in these measurements is highly desirable, especially for the time dependencies in the hydrological cycle, and thus there was a strong incentive to prolong the Grace mission as long as possible. This paper describes the power related activities to achieve this goal. A summary of the operational approaches that helped to ensure safe operations is provided, as are some conclusions and recommendations for the power management on the Grace Follow-On mission. A complementary paper that covers attitude operations is also presented at this conference.

Section 1 provides the background to the Grace mission. In Section 2 the battery, its failure modes and operational history are presented.

The degradation of the NiH_2 battery was one of the two main factors (the other was the amount of fuel left and its enhanced expenditure towards the end of the mission) that set the operational constraints. Standard charge and discharge regulation had not been possible since 2011. A description is given of the several alternatives that were tried. Battery handling required unconventional charge termination, special on/off cycles of the instrument, re-work of the on-board FDIR and daily adjustment of their settings. Several areas other than charge regulation itself were also affected by the low-power situation. The voltage in eclipses dropped so low that the transmitter was no longer working and the instrument computer shut down (with direct consequences for attitude control, thus aggravating the power situation). It even led to a power–down of the On-Board Computer in some instances. Here, only very special measures that comprised a wide palette touching computer, power/thermal, attitude and payload settings could mitigate the problem and re-start basic operations. These measures will be discussed in detail in Section 3.

Acronyms/Abbreviations

Battery Charge Control (BCC)
Battery Charge Regulator (BCR, main or redundant)
Common Pressure Vessel (CPV)
Deutsches Zentrum für Luft-und Raumfahrt (DLR)
Electrically Erasable Programmable Read-Only Memory (EEPROM)
Electromotive Force (EMF)
End-of-Charge (EoC)
Geoforschungszentrum Potsdam (GFZ)
German Space Operations Center (GSOC)
GRACE (Gravity Recovery and Climate Experiment)
Instrument Control Unit (ICU)
Instrument Processing Unit (IPU)
Jet Propulsion Laboratory (JPL)
Microwave Assembly (MWA)
On-Board Data Handling (OBDH)
Safe Guard Memory (SGM)
Ultra Stable Oscillator (USO)

1. Introduction

The GRACE mission—“Gravity Recovery And Climate Experiment”— was a scientific co-operation between the USA and Germany. The two identical satellites were designed and built by Airbus Defence and Space (formerly Astrium) in Germany. All operations were carried out at the German Space Operations Center (DLR-GSOC), whereas the scientists were from the University of Austin in Texas and the Geoforschungszentrum Potsdam (GFZ), Germany. The instrument processing unit (IPU) was under the responsibility of JPL (Jet Propulsion Laboratory, USA).

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The Grace satellites had a polar orbit with an inclination of 89° and an altitude of originally 480 km. This inclination caused the orbital plane to rotate in inertial space once every 322 days and consequently the eclipse duration changed throughout each cycle. Full sun orbits without eclipses occurred when $|\beta'| \geq 70°$ which was the case in $\sim$ 2⁄3 of the cycles. The orientation of the orbital plane with respect to the Earth-Sun line is denoted as the β' angle. At $\beta' = 0°$ the eclipses are longest, i.e. then the most energy from the battery was needed. Such $\beta' = 0°$ crossings occurred every 161 days.

Contacts for downlink were frequent: since 2006 up to 16 passes per day (distributed over both satellites) were provided by GFZ over the Ny-Ålesund antennae on Spitsbergen. The Weilheim and Neustrelitz ground stations in Germany were normally used for uplink in up to four passes per spacecraft per day. Passes over stations of NASA's Near Earth Network, or over other DLR stations could be ordered if additional contacts were needed (on short notice if necessary).

The main scientific goal of the mission was to collect data to create both static and time-dependent models of the gravity field of the Earth with unprecedented accuracy. This was done by measuring variations in the separation of the satellites down to $1\,\mu\mathrm{m/s}$ using a microwave link. The distance of the satellites was kept within (220 ± 50) km.

The study of time dependencies gained in importance over the years and became the strongest incentive to prolong the mission as long as possible. It yields e.g. the long-term development of polar or glacier ice masses or of the water masses in the Amazon basin, but also indirect results such as the prediction of droughts and floods, forecasts of seasonal stream-flow and the analysis of the over-exploitation of ground water resources. This justified a follow-on mission ("Grace Follow-On") which was launched on May 22nd 2018. A schematic view of a Grace satellite is shown in Figure 1.

Several units had to be active to generate scientific output:

- The Ultra-stable Oscillator (USO). It provides the time signal necessary for science operations.
- The Instrument Processing Unit (IPU) which handles the data of the Microwave Instrument, the GPS receivers and the Star Camera. It is therefore essential for science but also for orbit and attitude determination.
- The Microwave Assembly (MWA) which is the microwave transmitter and receiver located at the front-side of each satellite.
- The Instrument Control Unit (ICU). This unit processes the data from the accelerometer (ACC) which are used to correct for non-gravitational forces such as air drag or thruster torques.

Also needed were:

- The frontends of both satellites had to point at each other within roughly 5 mrad to maintain the microwave link.
- Use of the transmitter had to be possible to download the collected science (and housekeeping) data.

Each of these units was a candidate for being switched off temporarily (or even permanently) when power became scarce. Therefore they played a role in the power related operations described in this paper.

The Grace mission ended in October of 2017 after more than 15 1⁄2 years. Within the last year of operations the life-limiting resource was not only the battery but also the remaining fuel. Eventually, the battery of Grace 2 deteriorated beyond being operationally useful. The fuel was depleted shortly afterward. Therefore, it was decided to end the mission in order to achieve a controlled decommissioning. Grace 2 re-entered the atmosphere on December 24th, 2017. Grace 1 followed on March 10th, 2018.

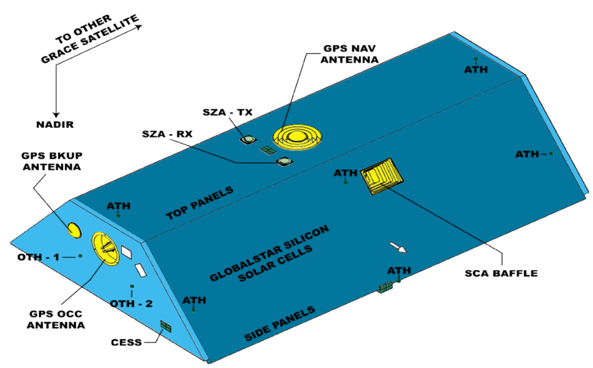

Figure 1: Schematic view of one of the Grace satellites. Solar panels were located on both sides and on the top. The 18 Ah NiH_2 battery was on the bottom plate near the center of the satellite. Note that the placement and orientation of the solar panels were such that during a full sun phase only one of the side panels was continuously illuminated and at $\beta' = 0°$ all panels were illuminated but at varying angles.

2. The Battery and its operational history

2.1. *The battery*

The battery comprised 20 nickel–hydrogen cells arranged in ten common pressure vessels (CPVs) with two cells each. A cell releases gaseous hydrogen when it is charged and therefore must be contained in a sealed vessel.

The charge was delivered by three body-mounted solar panels, whereby one, two or all three were illuminated by the Sun depending on the β' angle (and the satellite's attitude; see Figure 1). The charge process was terminated

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automatically on board at a commandable end-of-charge level. This level was one of 16 integer values, each one representing a voltage U_{eoc} between 27.04 V and 31.54 V in increments of 0.3 V. The actually achieved end-of-charge voltage also depended upon the battery temperature and charge current (Eq. 1). High temperatures decrease but high charge currents increase the effective termination voltage.

$$U_{\text{termination}} = U_{\text{eoc}} - 0.09\,\frac{\text{V}}{^{\circ}\text{C}}T + 0.2\,\frac{\text{V}}{\text{A}}I_{\text{charge}} \quad (1)$$

The exothermic reaction of a discharging Nickel–Hydrogen battery increases the temperature and thereby the pressure. As a precaution the termination voltage was reduced if the temperature rose due to external influences.

During the charge process the voltage seen at the battery terminal is higher than the actually available voltage or electromotive force (EMF). The difference is the product of charge current and internal resistance of the battery as seen in equation (2). The lower the charge current the closer the voltage will be to the EMF.

$$U = \text{EMF} + I_{\text{charge}}R_{\text{int}} \quad (2)$$

Once the termination voltage $U_{\text{termination}}$ was reached the charge regulator would go into a trickle charge mode by closing shunts. These were opened again when entering eclipse, detected by a solar array current close to zero.

The battery charge control (BCC) and one of two charge regulators (BCR) had to be on for this process. Each BCR could be set to a different termination level and was linked to a separate voltage measurement. Termination would not occur if BCC was off and charging would resume if it was switched off after the end-of-charge level was reached, i.e. the full charge mode was active whenever BCC was off.

2.2. *Failure modes*

One failure mode for Nickel-Hydrogen batteries was known already: the collapse of a cell due to a short circuit. This is generally caused by high temperatures or excessive overcharging (which in turn also increases the temperature). Zimmerman[3] identified an additional failure mode that he named “electrolyte bridging” in 2010; the migration of electrolyte between cells in the same CPV. This failure mode is driven by insufficient charging.

Precise control of the amount of overcharge was required to avoid both failure modes and became a core concept for battery operations. It is defined as the sum of the ampere hours (Ah) put into the battery minus the sum of Ah taken out of it in an orbit.

The charge termination method with its granulation of 0.3 V had to be refined. The target value was given by Zimmerman[4] as the “failure trade” overcharge between the two failure modes in relation to the desired discharge amount. This is shown in Fig. 2 for three different temperatures. The required settings to achieve the target were provided by JPL on a daily basis from September 2010 on.

Additional care had to be taken of the temperature because the battery is affected by it. Higher temperatures increase undesired side reactions and it was desirable to achieve a better control that limited the maximum temperature to about 15 °C.

2.3. *Initial overcharge control*

Early in the Grace mission these failure modes were not taken into account. The battery temperature reached values above 20 °C during the $\beta' = 0^{\circ}$ crossings.[1]

Indicators for the weakening of a cell were primarily the battery temperature and the “half voltage” defined as the ratio half voltage/full voltage. The half voltage indicated not only a change in behavior but also in which half of the battery stack a weak cell was located by either increasing > 0.5 (cell problem in other half) or decreasing < 0.5 (cell in measured half). The relative

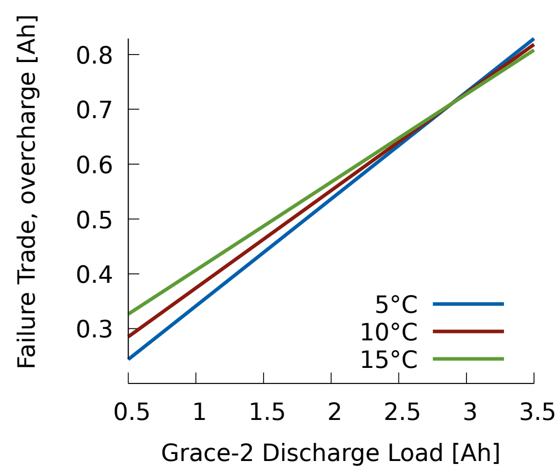

Figure 2: Grace 2 overcharge tradeoff curves by Zimmerman. The appropriate target overcharge can be determined depending upon the used discharge load per orbit and the average battery temperature. The curves are similar for both satellites but not exactly the same due to differences in the battery properties.

amount of overcharge increases if a cell weakens or collapses completely. This also causes the temperature to increase which is detrimental to the battery and should be avoided. An autonomous on-board reaction had to be implemented because a cell collapse could happen at any time out of the visibility of a ground station. The Grace satellites allowed for the creation of command sequences

[1] Implies longer eclipse periods, more battery discharge and higher temperatures due to the exothermic reaction.

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that could be uploaded to the spacecraft and started either by telecommand or automatically based on certain conditions. Such a macro was built with the goal to lower the end-of-charge setting of the regulator when the battery temperature exceeded a pre-defined limit. It would in such a case enable the charge control, disable the main charge regulator and switch to the redundant one, which was set to a lower EoC value. The temperature limit of this macro was adjusted frequently in accordance with the varying battery temperature.

The load on the battery was continuously changing twice in a 322 day cycle: from zero in full sun orbits to a maximum close to $\beta' = 0°$. The value of the maximum depends upon the active units and other settings on board. It was no longer sufficient to merely set a fixed end-of-charge voltage. The 0.3 V step size of the charge regulator resulted in too much difference between target and achieved result. Therefore, an approach was chosen that covered two scenarios. Each orbit the end-of charge limit was set twice with time tagged commands. During eclipses it was set to a level n and at a later time in the sun phase this was reduced to $n-1$. If the target overcharge required a termination close to an existing EOC level then the automatic voltage controlled termination happened there. The time tagged reduction would then serve as a backup. If the target overcharge required a termination in between two existing levels, the time tagged reduction was placed such that the voltage was between the levels of n and $n-1$. Charge termination occurred immediately after the level had been reduced. This is labeled method 2 in Table 1.

The battery was not needed to provide power in full sun phases when one solar panel was illuminated all the time. The BCR was in trickle charge with a small charge current remaining. The battery would not be discharged for several weeks which was deemed detrimental to its health. Artificial eclipses were introduced to draw a discharge current from the battery[1]. This was done by rotating the satellite such that none of the solar panels was illuminated for some time. (method 6 in Table 1:).

3. The handling of a weakening battery

3.1. *Refined charge termination methods*

A method was described in section 2.3 that allowed for a finer granulation of the charge level than the original 0.3 V. This provided good control over the amount of overcharge but not over the profile of the charge current. A new scheme was therefore developed which was first used in October 2013. The main incentive was to avoid high charge currents when the battery was nearly full because that reduces the efficiency (a higher percentage of the current will be transformed into heat).

This is labeled method 3 in Table 1:

- The EoC level was set to zero in eclipse, i.e. to the lowest level possible
- Charging of the battery would start shortly after the beginning of the Sun phase. The charge current would increase rapidly as the solar panels became ever more illuminated by the Sun (see Figure 1 for the geometry).
- Charging stopped when EoC level 0 was reached.
- Some time later the EoC level was set to the actually desired value.
- A change of EoC level *after* charging has finished has no effect and thus charging was forced to resume by a switch to the redundant and back to the main BCR.
- The second charge termination would then occur either when the EoC level was reached or when a time-tagged command reducing the EoC level came earlier.

Effectively, this scheme created an interruption of the charge process where the charge current would otherwise have been relatively high (see Figure 5a). The complete filling of the battery occured closer to the entry of the next eclipse where the charge currents became smaller again.

The prerequisites were a change of the EoC level and a switch of the charge regulator twice per orbit. This implied between 75 and 80 time-tagged commands per day[2], significantly more than for the previous scheme. The placing of the commands as well as the desired EoC levels had to be computed and adjusted on a daily basis.

In order to simplify operations a replacement method was developed that had the same effect but reduced the number of commands to be sent each day by a factor of two. This is labeled method 4 in Table 1:

- The EoC level was set to its desired value.
- The increase of the charge current with higher Sun angles reduces the termination voltage (see Eq. 1).
- The battery would therefore not be really full at the moment of charge termination.
- A second phase of charging was then triggered by a time-tagged disable of the BCR in the second half of the Sun phase (i.e. when the current decreases again.)
- Re-enabling the BCR after a few minutes would then terminate charging at a higher voltage than in the first phase.

[2]The important switch back to the main charge regulator was sent twice for safety reasons.

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Table 1: Summary of methods for improved battery handling. The phases in which the methods were employed can also be seen plotted as color coded boxes in Figure 3.

	Method	Time Span	Description	Update Frequency
1	Coarse EoC control	2002-May 2011	constant EoC	Adjustment if needed, e.g. for full Sun phases.
2	Fine EoC control	May 2011- Oct. 2013	Switching of EoC level for charge termination. Termination between levels possible	Daily upload of 2 commands per orbit.
3	Fine EoC and current control	Oct. 2013- Apr. 2014	additional current regulation	Daily upload of 5 commands per orbit.
4	Simplified fine EoC and current control	Apr. 2014-Jun. 2016 (Grace 1 until end of mission)	additional current regulation	Daily upload of 2 commands per orbit.
4 / 5	Simplified fine EoC and current control & current diversion by heater switching for safety	Dec. 2015-Jun. 2016 (Grace 2 only)	additional current regulation and backup termination	Daily upload of 2 commands per orbit. Adjustment of heater diversion if needed.
5	Current diversion by heater switching	Jun. 2016 - Aug. 2017 (Grace 2 only)	Charge control no longer works otherwise	Daily adjustment if needed.
6	Forced discharge	2012-2015	Forced discharge during full sun phases by yawing the satellite 90°	Significant portion of one orbit for selected orbits during the full sun period.

Again the place of the two commands in each orbit had to be adjusted at least once per day. A comparison of these schemes can be seen in Figures 5a and 5b. For Grace 1 this method was used until the end of the mission.

The result of these schemes was an overall improved control of the overcharge as can be seen in Figure 3. It shows when the several methods of battery handling were applied, and further the amount of overcharge and the overcharge ratio, i.e. charge/discharge per orbit over the mission as well as the corresponding minimum battery voltage and the maximum battery temperature per orbit. This figure also shows the history of the battery voltage. There is a trend visible from 2013 onwards indicative of a degradation besides the failure of individual cells. Nonetheless, the control of the temperature could be improved due to the methods employed (see Table 1).

3.2. *Switching off instruments*

The overall goal was to keep the temperature below 15 °C (see Section 2.2). This was at odds with operating the instruments during the long eclipses around $\beta' = 0°$ because the deep discharges inevitably led to an increase of the battery temperature.

It became therefore necessary to reduce the amount of discharge during the longest eclipses. The first step was to switch the MWA off in eclipse and restart it in the sun phase. This was not possible for the ICU because its stabilization would take several days after power up. A macro was developed to handle the power cycles of the MWA depending upon available voltage and current.

Table 2 provides an overview of the power consumption of each unit and an approximate corresponding gain in the voltage when it was switched off.

Table 2: Power usage of the instruments and the approximate increase in the minimum voltage that could be obtained by switching off these units.

Instrument	Power	≈ Increase of V_{min}
ICU	7.3 W	0.5 V
MWA	17.5 W	1.0 V
IPU	14.6 W	0.8 V

Around $\beta' = 0°$ the MWA and ICU were switched off entirely to maintain a proper temperature range. This period was about four weeks in duration when it was first begun in 2012 (see Figure 4). The switch-off period increased with further battery degradation as did the period of partial operation. In 2016 it was no longer practicable to have the MWA active during an entire orbit even at favorable β' angles. It had to be switched on and off either via time tagged telecommand or on-board macros. The ICU had to be switched off entirely. For science, the accelerometer data were transplanted from Grace 1 to Grace 2 on the ground, i.e. the measurements from Grace 1 were shifted in time such that they fit the separation of the Grace 2. Naturally, this reduced the quality of the data.

At all transitions between operating periods, changes to the battery charging scheme becamse necessary. Also, the setpoints for low-voltage protection had to be adjusted.

The instrument status during the last months of the mission can be seen in Figure 6.

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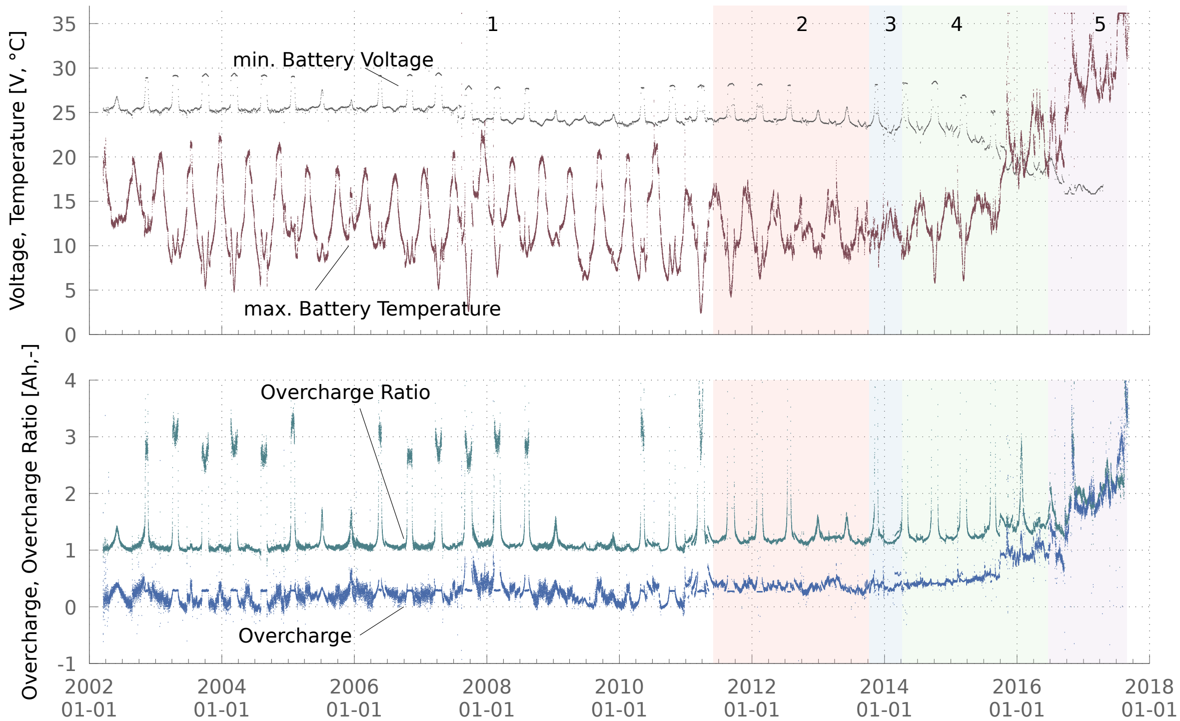

Figure 3: The minimum battery voltage, maximum battery temperature, overcharge ratio (Ah in/Ah out) and overcharge (Ah in - Ah out) per orbit for Grace 2 over the entire mission. The phases in which the various battery handling methods were employed are numbered and color coded in accordance to Table 1.The spikes in the Overcharge Ratio correspond to short eclipses changing into full sun phases where the battery was not or only slightly discharged. Starting in mid-2011, the overcharge was controlled more accurately which reduced its variation and the maximum battery temperature. As more battery cells began to fail in the second half of 2015 the overcharge amount was deliberately increased to counteract further cell failures (at the risk of short circuits). Towards the end of the mission the heaters were used to perform battery charge regulation, further increasing the temperature.

3.3. *Charge control by diverting the current from the battery (Method 5)*

By November 2015 Grace 2 had lost four battery cells reducing the minimum voltage to 18.5 V. The eclipse duration was increasing and the MWA and ICU had been switched off to save power. The EoC level was set to 0 on both charge regulators. The heaters were using the survival settings that used hardly any power at all.

On November 11th, the voltage controlled charge termination on Battery Charge Regulator (BCR) Main started to become unreliable due to another weakening cell. The charge termination could not be reached in an orbit (see Figure 7). The resultant temperature rise triggered the safety macro described in Section 2.3 which eventually caused a charge termination by the redundant BCR. This was peculiar because the termination condition—battery voltage above the termination voltage [see equation (1)]—was not met on either BCR.

It was assumed that the *combination* of BCC OFF and BCC ON commands caused the charge termination, irrespective of whether or not the condition was met. This could be confirmed in a test and it appeared there was the possibility to have full time tagged control over the battery charge process. This approach was then implemented in the same way as method 4, described in section 3.1 as they are identical.

Unfortunately, this method was not as reliable as had been hoped. An analysis showed that the termination depended on the difference between battery– and termination–voltage. When it dropped below about -0.7 V termination would not work anymore. This could be verified in a few instances in the telemetry. Charge termination was governed entirely by hardware and it was speculated that the observed behavior and the telemetry values might not be entirely consistent in the narrow range around the termination point.

Orbits where this type of commanded charge termination did not work at first only occurred close to $\beta' = 0°$ crossings but it was clear that an alternative method was needed.

GRACE used two tables for the control of 64 heaters, a nominal and a survival heater table. The set-points of each individual heater were configurable and it was possible to switch between the two tables.

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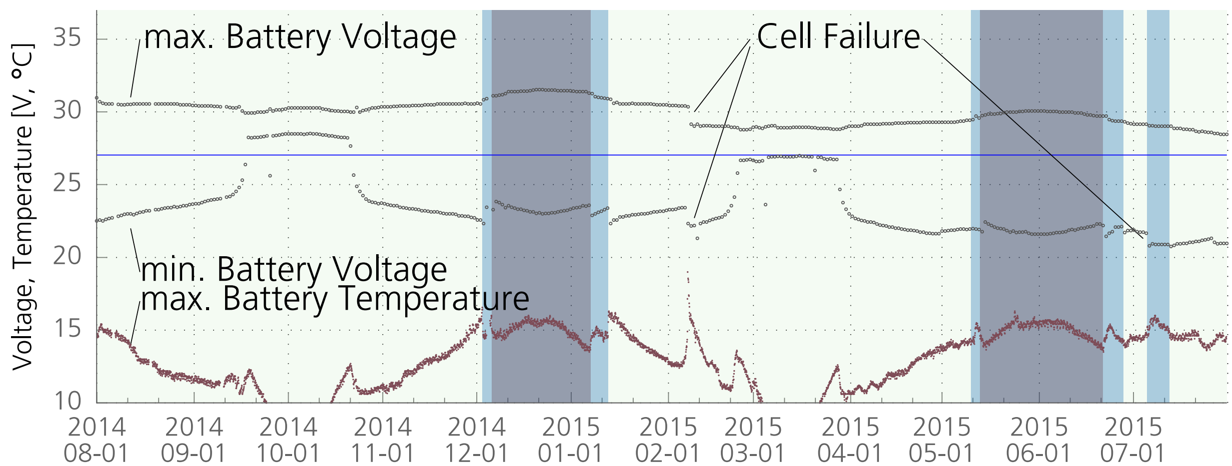

Figure 4: Zoomed in view of Figure 3 showing roughly one β'-cycle. The maximum voltage in each orbit is shown additionally as is the lowest EoC level denoted by the blue horizontal line (as a proxy for charge termination). Method 4 was used during the time. A goal of charge control was to prevent the temperature from rising much above 15 °C. Units were switched off in order to reduce the load on the battery despite the fact that charge termination was still roughly possible indicated by the maximum voltage being above the blue horinzontal line. The blue bar denotes where the MWA was switched off, the purple bar within the blue one were the ICU was switched off. Two cell failures (of the second and third cell to fail) can also be seen, one in February, the other in July 2015. During the February failure the cell is gone entirely. During the failure in July it still contributed to the maximum (charge) but not the minumum voltage (discharge). An on-board macro switched off the MWA as a response. Due to the cell failures and overall degradation the maximum voltage is reduced and gets closer to the minimum EoC level.

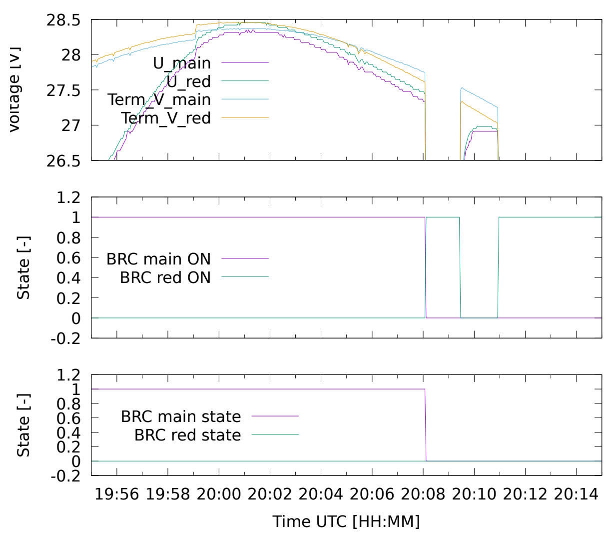

Figure 7: The top plot shows the main and redundant voltage measurements and their corresponding termination voltage according to Equation (1). Termination was expected to occur on BCR around 20:01 but this did not happen. An increase in temperature (not shown) triggered a safety macro at 20:08 which switched from BCR main to BCR redundant and triggered a power cycle on the BCC. This caused a termination of the charge process. On-board commands switched the BCC off and on again at 20:09:30 and 20:11:00, respectively. The latter command again caused a termination. This was unexpected because in both instances the termination condition was not met.

A temporary switch could thus be used to divert current from the battery to the heaters and regulate the charge current. The timing of the switches and the setpoints of the heaters had to be adjusted in a way to prevent a collapse of the battery due to under-voltage and a thermal run-away due to over-current on the other.

This method raised the battery temperature because of the needed number of activated heaters, including the battery heater itself. This was undesirable but remained the better option compared to a thermal run-away due to excessive overcharging. Naturally, the heaters should not be enabled in eclipse when the battery was discharged and some caution needed to be used to not end up in such a situation. The option to switch tables was realized by a chain of three on-board macros which were triggered when an associated monitoring detected certain conditions for a set persistence (see below).

Charge termination via the BCC OFF and ON commands (method 4) still worked when this new option was developed. It was first used as a back-up in case charge termination failed. The new method should therefore not interfere with the old one. The persistence of the macros was therefore set to values that exceeded the expected time of charge termination.

The sequence of method 5 was as follows (the values used here are in line with what can be seen in Figure 5c):

- The battery charge current is monitored. If it is larger than 3 A for more than 1200 seconds, the first

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macro is triggered.

- This resets its own timer and switches to the other heater table. Selected heaters are now active and current will be diverted from the battery.
- A monitoring function surveilles the time of the current diversion. When it exceeds 820 seconds, the second macro will be started.
- This macro resets the second timer and switches back to the original heater table. The current diversion stops.
- If the current of the solar array is lower than 0.5 A for more than 30 seconds (due to eclipse entry), the third macro is triggered.
- This also resets its timer and re-enables the first monitoring function. The sequence can start over in the next orbit.

The timing of the BCC ON/OFF commands and the persistencies of the monitoring functions had to be adapted on a daily basis. The set-points in the heater table had to be updated occasionally as well to adjust the amount of diverted current. Sometimes it was even necessary to change the settings twice in a day.

The usage of scheme 4 and 5 in parallel was done until the end of June 2016 when the 5th cell weakened on Grace 2. Thereafter, the commanded charge termination no longer worked and the only option to limit the overcharge was the method to divert the current. This scheme was employed until the end of the mission on Grace 2 (see Table 1).

3.4. *Transmitter switch on, data gaps, hibernation and OBDH reboots*

Seven cells had failed on Grace 2 by August 2017 which reduced the minimum voltage around that $\beta' = 0°$ crossing to <16 V. The 13 remaining cells would barely suffice to support science operations once the illumination conditions would improve again in October. Also, ≈300 g of fuel would be needed for the attitude control in science mode. The ICU and MWA had been switched off. Even the IPU—which was needed to keep fuel expenditure low[3]—had to be switched off for a period of up to 15 minutes each orbit via configurable on-board macros. The load was reduced to an absolute minimum and only the usage of the heaters each orbit to draw current away from the battery could be continued.

[3] Grace 1 accelerometer data were used on Grace 2 after its ICU was powered off. See the corresponding paper[2] in these proceedings for this and other mitigation measures for AOCS.

3.4.1. *Transmitter switch on and data gaps*

Standard procedure was to switch on the transmitter before a contact with time-tagged commands (usually preloaded for a complete week). Ground stations for Grace 2 were already selected based on their location with respect to the eclipses. The timing of switch-on was shifted to higher voltages and a little was gained by a re-ordering of the switch-on sequence. Now, however, the voltage regularly dropped below 18.2 V, the limit for switch on.

The only work-around was to send so-called “high priority” commands that are addressed to the transmitter directly at the start of a contact and to keep repeating this “blind acquisition” until the transmitter was powered on.

When the voltage dropped below 15.8 V the telemetry stream on board was interrupted because the I/O board stopped working until the voltage increased to 16.9 V. Such data gaps were not recoverable and had serious consequences on satellite control (see [2]).

3.4.2. *Hibernation and Recovery*

An eighth battery cell was presumably lost on Grace 2 in early September 2017. No telemetry was received for about four days. It was assumed that the lack of telemetry was due to data gaps, complete power down of the OBDH, a transmitter problem, or a combination of these.

The fuel reserves were close to being depleted and the latch valves to the tanks had therefore been closed just prior to this event. The satellite was using the magnetic torquers for attitude control. The valves would be open again if a reboot from EEPROM occurred (possible if the on-board computer several times).

Commands to switch on the transmitter, to close the latch valves or to stabilize the satellite in case it was spinning were sent in the blind. Only once, a signal was reported but still no telemetry was received. This was similar to an anomaly that had been observed earlier in the mission. In that case the processor module had switched over autonomously to the redundant side and produced only snippets of telemetry. Recovery was achieved in that instance by a switch back to the main processor.

Therefore, a switch to processor module B and back to A after one minute was commanded in the blind. This succeeded in re-establishing communication with Grace 2. It was discovered that no fuel had been used in the intervening four days and that no boot of the OBDH had occurred. Apparently, it had remained active, albeit in an undefined state which was then dubbed “hibernation”.

The lost cell appeared to have recovered and 13 cells were again contributing to the voltage.

At that time of unexpected recovery it was still a month to go until a short enough eclipse duration to permit switching on the units for science operations. Unfortunately, the “hibernation mode” could not be operationally

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triggered. It had to come about by itself.

3.4.3. *Frequent Reboots of the OBDH*

The on-board computer started to reboot frequently in the days following the recovery from "hibernation". In a scenario were several boots occur in sequence, there would at first be a warm reset of the active processor module, then a cold boot of the active processor module and then a switch to the redundant side of the OBDH. As long as a counter was > 0 the content of the non-volatile Safe Guard Memory (SGM) was used. Otherwise the contents of the EEPROM would be loaded. The counter was reduced by one at every reboot. It was undesirable to use the configuration from EEPROM because it differed in a number of settings, most notably those of the heater tables.

At each pass the commands to enable the transmitter and to reset the boot counter were sent, into the blind if no telemetry was received. Housekeeping data could be dumped only rarely. Still, reboots from EEPROM could not be avoided. As long as the hibernation state did not occur, these reboots had to be handled.

A sequence to recover the satellite to a nominal state after an OBDH reboot was available already during routine operations. The activities to recover could be executed within three to four passes. This was compressed to a single pass, addressing only the most urgent settings.

The used recovery sequence was as follows:

- set the safe guard memory counter to its maximum value (10) to avoid boots from EEPROM which would revert the heater settings to their defaults.
- switch off the Inertial Measurement Unit (IMU, which was on by default) to save power.
- re-upload all needed macros (user-defined macros are lost after a reboot) and adjust the their settings.
- reset the on-board computer watchdog counter.
- synchronize with GPS time
- adapt the persistence for a mode drop to the desired value to avoid unnecessary mode drops.

This sequence provided a very basic configuration.

3.5. *Final Attempt to recover Grace 2*

Grace 2 did finally fall back into hibernation in late September. This was indicated by a detectable signal but lack of telemetry in several contacts. It was decided to leave it in this mode until October 4th, 2017 when the duration of the eclipses would be well below twenty minutes and the time spent in the sun was expected to provide sufficient margin for a recovery.

The attempted recovery consisted of two parts. Part one was to bring the satellite back into a state in which it could be commanded and telemetry reception was possible. Part two was the configuration of the satellite to use the thrusters as little as possible (see [2]). The first part was successfully performed on October 4th but just when the activities of the second part had begun the eighth battery cell was lost again. One more attempt was made on October 9th but—presumably because even more battery cells stopped functioning—this was not successful. Fuel had been depleted entirely after the second attempt.

Decommissioning was done on Grace 2 immediately after the last attempt and it re-entered the atmosphere on December 24th, 2017 west of Alaska. Several tests were performed on Grace 1 until fuel was almost completely exhausted. Decommissioning was started in December and reentry occurred on March 10, 2018 south of Africa.

4. Conclusions, Outlook and Recommendations

The final years of Grace demonstrated that permanent analysis, planning and performing of operational activities can prolong a mission far beyond the original design. Science data could be collected until six months before re-entry which occurred at a moment where fuel and battery were both completely exhausted.

Grace Follow-On was launched May 22nd 2018 meaning the gap to its predecessor was less than a year. Its design is the same as for Grace (with modifications on bus and software). It uses a 78 Ah Li-Ion battery instead of the 18 Ah NiH_2 design of Grace. Further improvements comprise the addition of a third star camera and a second IMU. A laser ranging instrument serves as a technology demonstrator for future gravity missions.

A few items have been identified that have helped Grace operations and have either been applied to Grace Follow-On or are relevant for future missions.:

- Contrary to conventional practice, the GRACE Project did not hesitate to cycle power on operating flight units. There were no failures induced by power cycling.
- The macro capability of Astrium's flexbus flight computer was critical to extending the life of the mission. It had the ability to create commands based on telemetry values, and create action sequences with relative time offsets (persistences). Macros were used extensively to managing the load on the battery and to optimize battery charging. It is a "must have" capability. Basic math functions were not available, and were not needed.
- Commands to recover from reboots of the on-board computer should be available. Depending on the situation, this should also include commands to restore a particular configuration.

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- The availability of a simulator over the entire mission is important. The Grace simulator became unusable in 2012, mainly due to obsolete software and unavailable hardware. All tests and experiments performed afterwards had to be made directly in orbit. This made their preparation and execution far more complex and risky. Grace Follow-On uses a pure software simulator that is easier to maintain. In addition, there is a hardware realtime testbed maintained by Airbus.
- The BCC was hard wired. A more flexible software controlled system would be desirable in the future.

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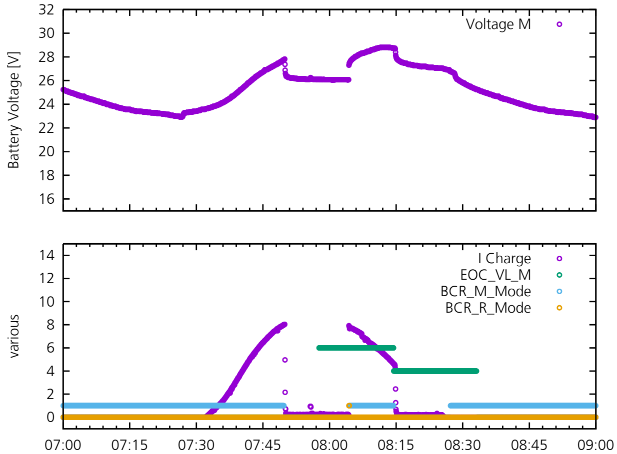
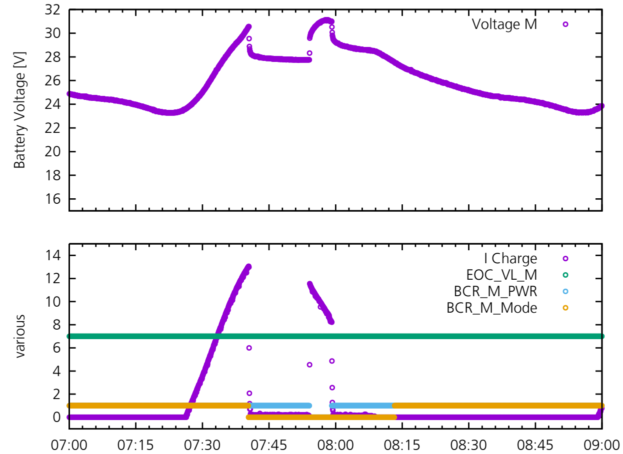
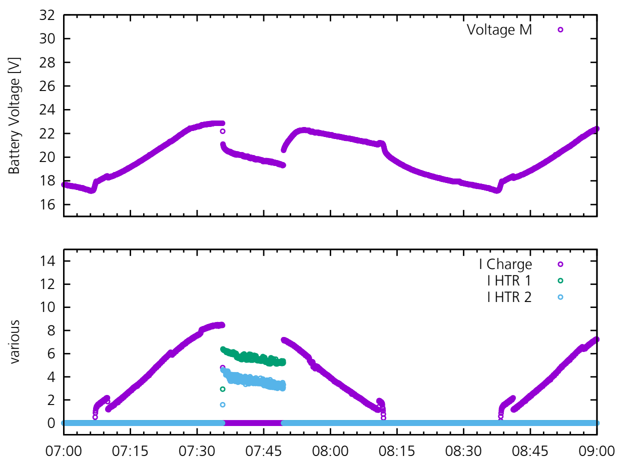


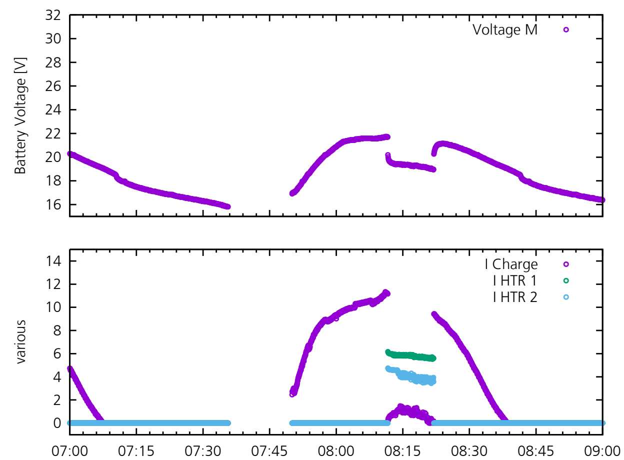
(a) Initial charging scheme to control the charge current, here for Grace 1. EoC to minimum, Charge termination at EoC, EoC to desired value, re-activation of charging by toggling BCR, charging stops again either at EoC or by backup time-tag.



(b) Simplified charging scheme to control the charge current, here for Grace 2. EoC to desired value, charge termination at EoC, time-tagged disabling of BCC causes charging to continue at the desired charge current, time-tagged enabling of BCC terminates charging.



(c) Charging scheme utilizing the heaters, here for Grace 2. The maximum battery voltage is no longer sufficient to trigger charge termination. Heaters are activated via timed on-board macros to re-route the current that would otherwise be available for charging the battery. Switching on the heaters also heats the battery and increases its temperature significantly.



(d) Same charging scheme as in Figure 5c but with data gaps due to low voltage.

Figure 5: Different battery charging schemes. In Figure 5a the more complicated approach is shown, in 5b the simpler one. In both cases the result is the same: two charge phases interrupted for a period when the charge current is highest Figure 5c shows the charging scheme using battery heaters which was employed once the regular charge termination no longer functioned properly.

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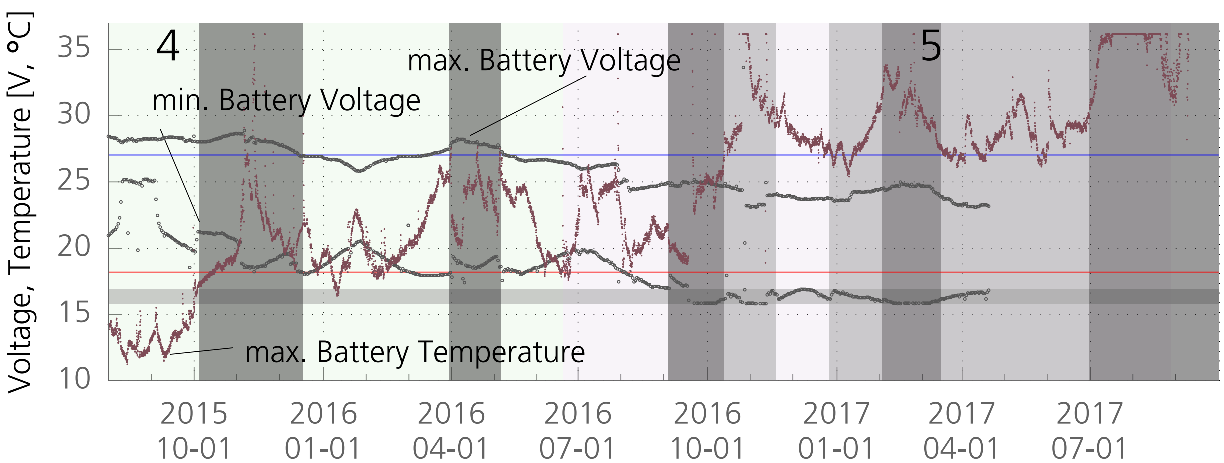

Figure 6: The last two years of Grace 2. Shown is a zoomed in view of Figure 3 from September 2015 to September 2017. Again, the minimum battery voltage and maximum temperature are plotted. Note that these are the minimum and maximum values for each day. The blue horizontal line denotes the lowest EOC level at 27.04 V. Below the red horizontal line (18.2 V) the transmitter could not be switched on. The horizontal grey bar denotes the area below which the OBDH could no longer process data and gaps occurred. The dark vertical bars show where the MWA was switched off. The vertical light grey bars show were it was partially active.