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Attitude and Orbit Control of the Grace Satellites at extremely low power

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Abstract

The two GRACE satellites were launched on March 17, 2002 by a Russian Rockot. GRACE not only was the first dual-satellite mission operated by German Space Operations Center (GSOC), but it also was the first formation flying at an altitude below 500 km. The mission was successful from a scientific point of view and the originally envisaged mission duration of five years finally became almost sixteen. A follow-on mission with the same partners started 2018 and JPL projects a new generation of satellites for gravity measurements in the twenties, so there was a strong incentive to prolong the GRACE mission as long as possible in order to minimize the gap.

Infirmity comes with age and several components deteriorated towards the end or even were defunct. The scientific goals were still obtained to a very high level until June 2017, four months before decommissioning.

The major challenge for operations was posed by the degradation of the NiH_2 batteries. These are comprised of twenty cells packaged in the common pressure vessel (CPV) configuration. Three cells had shorted out on Grace 1 and seven on Grace 2 by the middle of 2017. The charge capacity of the operational cells was severely degraded. The longevity and the occurrence of several anomalies due to power problems depleted fuel ever more rapidly.
Keywords: low-Earth orbit, mission extension, satellite operations, low-power, fuel expenditure

Acronyms/Abbreviations

Accelerometer (ACC)
Attitude Hold Mode (AHM)
Attitude thruster (ATH)
Center of Mass (CoM)
Coarse Earth and Sun Sensor (CESS)
CESS and Magnetometer Coarse Pointing Mode (CMCPM)
Coarse Pointing Mode (CPM)
Depth of Discharge (DoD)
End of Charge (EoC)
Failure Detection, Isolation, and Recovery (FDIR)
German Space Operations Center (GSOC)
GRACE Follow-On (GFO)
Half Power Bandwidth (HPBW)
High Priority Command (HPC)
Electrically Erasable Programmable Read-Only Memory (EEPROM)
Inertial Measurement Unit (IMU)
Instrument Control Unit (ICU)
Instrument Processing Unit (IPU)
Jet Propulsion Laboratory (JPL)
K-Band Ranging (KBR)
Magnetorquer (MTQ)
Mission Elapsed Time (MET)
Microwave Assembly (MWA)
On-board Data Handler (OBDH)
On-board Orbit Propagator (OOP)
Orbit thruster (OTH)
Proportional Derivative (PD)
Rate Damping (RD)
Science Mode (SM)
Star Camera (SCA)
Star Tracker (STR)
Star Tracker Head (STRS)
Two-Line element (TLE)
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 Defense and Space (former Astrium) in Germany. All operations were carried out at DLR-GSOC, whereas the scientists were from the University of Austin in Texas and the Research Centre for Geosciences Potsdam, Germany. The IPU was under the responsibility of JPL.
The main scientific goal of the mission was to collect data for creating both static and time-varying models of the gravity field of the Earth with unprecedented accuracy. This was done by measuring variations in the separation between the two satellites down to 1 μm/sec using a microwave link. Both flew on a polar orbit at an altitude of originally 480 km and were kept at a distance of 220±50 km. Study of time

dependencies, yielding e.g. the long-term development of polar or glacier ice mass, or of the water content in the Amazonas basin, gained in importance over the years and became the strongest incentive to prolong the mission as long as possible and to start a follow-on mission as soon as possible.

A schematic view of one of the satellites is shown in Fig. 1. Of specific interest in this paper is the location of the SCAs and of the thrusters for attitude and orbit control (ATH and OTH). Each satellite had two star cameras, one on the +y and one on the –y side. Both orbit control thrusters were located at the rear (-x) side. Attitude control was performed by a combination of twelve ATHs (two branches with six thrusters each) and three MTQs, one for each axis.

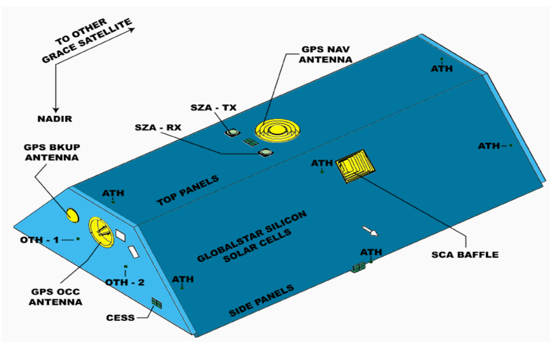


Fig. 1. Schematic of one of the Grace satellites. Solar panels were located on both sides and on the top. The micro-wave assembly providing the link to the other satellite was at the front and is not visible in this picture. The 18 Ah NiH_2 battery was mounted on the bottom plate near the center of the satellite. The two star cameras are located at the +y and –y side (only the baffle at the +y side is visible).The location of both orbit control thrusters is indicated on the rear panel and six (out of twelve) attitude control thrusters are shown as well.

The mission ended by the end of 2017 after more than 15½ years. Not only the battery, but also the rapidly depleting fuel led to the decision to end the mission after the full-Sun orbit in October 2017, in order to achieve a controlled decommissioning of both satellites.

Table 1 gives an overview of the several components that started to show signs of weakening with age. A few failed altogether (or were suspected to have failed), but for each of these there was a fully operational back-up in place until the very end. A major OBDH software upgrade was implemented in 2006, primarily 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, by the end of the mission far more than 70000 cycles had been made. The natural degradation of the capacity furthermore implied that the *relative* DoD was much higher than at the beginning, something the battery appeared to cope with surprisingly well. During the last years of operations, especially GR2 experienced several OBDH reboots and data outages as a consequence of low battery voltage. For more information on the battery design and performance, see [8].

Table 1: Overview of components that did not meet the original specifications in the latter years of the mission (status as of June 2017). The major impact on the mission stemmed from the natural degradation of the batteries coupled with the loss of several cells, which required special operations in the presence of eclipses. The name-plate capacity was 18 Ah. Scientific operations were still possible with few restrictions until shortly before decommissioning.

GRACE 1	Date	Redundancy / fully operational
Failed components		
IMU	Launch	
USO main	17.03.02	Yes / Yes
IPU redundant	10.03.12	Yes / Yes
ICU main	22.05.02	Yes / Yes
3 CESS thermistors (2 black –z; 1 silver +x)	2004 until 2005	Yes / Yes
3 battery cells	26.06.17	Yes / No
Transmitter main	27.06.17	Yes / Yes
Degraded components		
Battery	gradual	≤3 Ah (~15%)
Star cameras	gradual	Enhanced noise
		(axis dependent)
GRACE 2		
Failed components		
IPU redundant	04.05.07	Yes / Yes
ICU main	12.09.06	Yes / Yes
2 CESS thermistors (1 black –z; 1 black –x)	2004 until 2005	Yes / Yes
7 battery cells	16.08.17	Yes / No
Transmitter main		Yes / Yes
Degraded components		
Battery	gradual	≤3 Ah (~17%)
Star cameras	gradual	Enhanced noise
		(axis dependent)

2. Special AOCS operations at low-power

The emphasis during the first years of AOCS operations was on having a good balance between thrusters and magnetorquers (see [5]) and minimizing their disturbances on the accelerometer. It was discovered shortly after launch that the MTQs caused disturbances that were too large for optimal science operations.

The influence of the MTQs was minimised by a) tuning of the control parameters, b) a S/W upload in 2006 that led to improved magnetometer measurements and hence less spurious MTQ currents, and c) by shifting the control balance towards the thrusters.

Point c) was good for science (it is possible to subtract the disturbance of each single thruster pulse from the accelerometer measurements) but lead to an increase of fuel expenditure.

Both satellites carried ~33.2 kg of cold gaseous N_2, an allowance of 6 kg/yr for a five year mission. Fig. 2 shows that the actual expenditure was much lower thereby making a much longer mission feasible. Efforts to minimize fuel expenditure continued throughout the mission though.

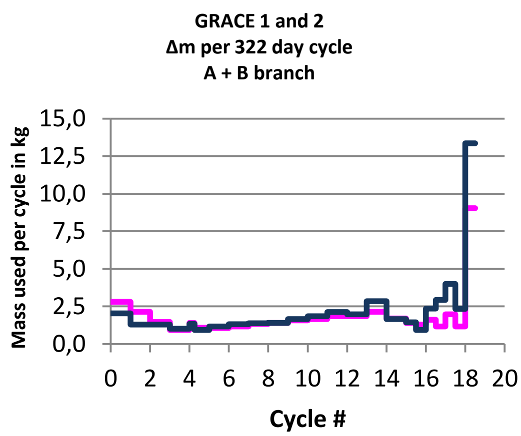


Fig. 2: Shown is the fuel consumption over the complete mission lifetime in terms of cycles. Each cycle is ~322 days. During the first couple of years the optimization of fuel consumption can clearly be seen (cycles 0 – 4). After that a quiescent phase with slowly deteriorating star cameras took place (cycles 4 – 14); Cycles 14 – 16 show the improvement due to parameter tuning and start of swaps (see 3.2); Deterioration of GR2 (dark blue) due to low-power can clearly be seen in cycles 16-18, at the very end there was a high expenditure on both satellites.

There is an additional dependency of expenditure on the star camera that is used for control within a cycle (see 3.2). There certainly also was an influence of the solar cycle. GRACE was launched in 2002 around solar maximum and still lived to see the low, but extended solar maximum 2013-2015. The orbital height of 480 km in cycle 0 declined to 300 km at decommissioning

Star Camera and GPS data were handled and processed by the IPU. The part needed by the AOCS was forwarded to the OBDH. Right after launch the IPU was doing well, but already during the commissioning phase the scientists requested the introduction of 1 Hz operations on both star cameras in order to get data to facilitate more accurate attitude determination. The default setting in the IPU was 1 Hz on the prime and 0.1 Hz on the backup camera. Dual 1 Hz operations led to overstrain, which then additionally led to more frequent IPU resets. Also the GPS data showed numerous invalid points which did not have a negative impact on the fuel consumption but regularly increase the time for re-acquisition after an IPU reset. At the beginning of the mission the IPU experienced frequent resets (once or twice per day). Roughly 10% of the resets resulted in a hang-up of the IPU caused by SCA and GPS re-acquisition problems. Additional FDIRs were introduced later in the mission in order to mitigate the problem of hang-ups. In 2006 the IPU software was updated, solving the problems with the Dual 1 Hz operations and invalid GPS data. The interval between resets became then so long, that after roughly 900 hours an IPU reset had to be commanded to keep clock biases within 500 ms (a requirement for science)

In the beginning of the mission the SCA experienced a lot of invalid points (up to 2%). As this often led to intervals without control, this behavior cost fuel. The amount of errors and invalid points was reduced by tuning parameters within the IPU (the trade-off was to accept somewhat higher noise in the SCA data). The star camera sometimes also experienced a so-called “hole-in-the-sky”. A hole-in-the-sky occurred when not enough stars were visible for the attitude to be calculated correctly. This happens in regions near the galactic poles. Additional head switches were introduced to cope with that. Over the whole mission life time always new ideas were brought up by the operations team to save fuel, reduce the noise in the data or to allow the satellites to acquire data at low power. An extrapolation of the fuel expenditure during the first years of mission showed an expected life time of more than 15 years.

The situation started to change after about ten years, whereas no extraordinary AOCS measures were needed during the first part of the mission (keep both satellites pointing at each other in science mode). Both GRACE 1 and 2 started to lose battery capacity (see [4]). During full-Sun orbits the battery was not used at all and it was in the state of trickle charge. During these phases some charge was still lost and the danger of a cell collapse or short circuit

increased, because it was not possible to recharge the battery then (see [4]).
It was decided to turn the satellite by 90° around the z-axis to cope with this situation and keep it at this offset for a few seconds to minutes (equivalent of a short eclipse) a few times during the full-Sun period. Thus the battery experienced a real discharge and charge cycle. Flying with such an attitude for a longer time required a change of the AOCS concept. The head which was turned away from the Sun was used during nominal operations. It had to be analyzed which direction would be the desired one before turning the spacecraft by 90° in order to avoid Sun blinding of the used star tracker. Additionally, as this attitude was not foreseen in the beginning, the FDIR concept had to be adapted to that situation. This concept was used from 2010 – 2014 until the battery started to degrade further. The concept of yaw turns could no longer be used for battery management after additional cells faded out. New macros (autonomously executed on-board procedures) were introduced, uploaded, and activated which allowed to start and end charging without the use of EoC levels (for detailed explanation see [4]).

The reduced workload on AOCS side only lasted roughly one year until the battery on GR2 started to degrade further and a fifth cell was lost in July 2016. It was discovered that below a battery voltage of 18.2V the interface between the transmitter and the OBDH was no longer working, with the loss of the telemetry link. Those downlink passes were selected when the satellite was in sunlight whereby the switch-on of the transmitter was delayed until 18.2V was reached. The impact on the collection and forwarding of science data could be minimized.
A 6th cell was lost on GR2 only two months later. The sudden decrease in power led to a shut-down of the OBDH and an autonomous restart on the redundant side once enough voltage was available in sunlight. However, only single TM frames were seen and no stable TM/TC link could be established. The satellite appeared not to be tumbling, though, so the root cause was an unknown failure on the redundant OBDH. It was decided to perform a switch back to the nominal OBDH. This was successful and allowed nominal commanding again. The satellite was brought back into a configuration that would support science operations. Data gaps and IPU resets occurred in almost every orbit with only 14 of 20 cells left. The interface between the IPU and the OBDH was seen to stop below 15.8V implying that no attitude control could be performed during such gaps. The IPU performed an autonomous power cycle once the voltage rose above 16.5V again and started to deliver data after ~3 – 5 min. The time below 15.8V was of the order of 3-4 min. during eclipses <10 min, implying a total outage for attitude control of up to 9 min. The instruments could still be switched on in the sun-phase and fuel expenditure was not yet excessive.
The time <15.8V increased to 20 min. for the longest eclipses of 30 min. Several tests were executed to see if an automatic IPU power cycled after every outage could be prevented. None was successful and therefore the FDIR settings were changed (e.g., for the star trackers) in order to prevent drops to coarse pointing mode (where fuel consumption was ~100 times higher).
A small amount of fuel was wasted each time the IPU restarted. The AOCS software stored the last valid quaternion from the star camera. It derived an attitude rate signal by differencing the current quaternion with the previous value. The error in this attitude rate signal is a function of the duration of the data gap and decreases rapidly if regular updates become available again. The impact of this anomalous, but transient, attitude-rate signal on fuel consumption became a concern. An additional macro chain was designed to mitigate this. It was triggered by the IPU power cycle and inhibited attitude control torques to be applied for the first ten seconds after the data flow resumed.
GR2 lost another cell (#7) in October 2016. The OBDH shut down several times and experienced numerous reboots. At this point it was very unlikely to recover the satellite to a state which would allow science operations to resume and first plans to decommission GR2 were made. Two weeks later the IPU on GR2 showed an outage of more than 11 minutes which led to roll deviations of 180° (up-side-down) and therefore to several orbits without on-board computer. The OBDH once more restarted itself and miraculously suddenly 15 cells were found to provide voltage during charging and 14 during discharge. The reason for this is still not understood but it allowed the operations team to continue acquiring science data. This fortuitous situation continued over the complete β' cycle from November 2016 until July 2017. Then, the approach of the next phase with eclipses forced the switch off of the ICU and MWA, this time final.

3. Minimizing fuel expenditure

Prerequisite for a mission extension far beyond the originally planned five years was a careful management of available resources and the balancing of the several effects that limit the lifetime. Despite the battery as an obvious factor also the fuel usage quite soon after launch was identified as a possible life limiting resource. Some effects could not be controlled of course, but a short discussion of those that could regarding fuel consumption will be given here (a more detailed explanation can be found in [5]). The influence of the battery and defunct components is presented in [4].

3.1. Parameter tuning

The two satellites were in SM for ≥93% of the mission. All efforts to reach the best possible performance by parameter tuning concentrated on this mode. Here only one idea is presented as an example (see [5] for an overview of other implementations such as the balancing of thruster and magnetic torques).

Narrower dead bands imply smaller attitude deviations but also lead to more frequent thruster actuations and therefore to higher fuel consumption. It was possible to transfer thruster activity from the yaw- to the roll-channels by changing their relative dead bands. This was advantageous because control in yaw was more expensive than in roll[1]. Several tests were performed over the years and allowed to find the optimum ratio between roll and yaw dead bands.

The gain on fuel consumption was in the order of 0.1 g/d.

3.2. Exchange of relative satellite position (swaps)

The precession of the orbital plane had an inertial period of ~322 days (β´ cycle) implying that for ~161 days the Sun is on one side and for ~161 days on the other side of the satellite. β´ denotes the angle between the orbital plane and the direction to the Sun. Thus, half of the time the Sun was on the side of SCA#1 and half of the time of SCA#2. The exclusion zone for intrusions increased from ±40° to more than ±50° during the mission, meaning that ¼ of the orbit the camera on the sun-side was blinded. The prime head was switched at every β'=0° crossing and used as secondary only to avoid Moon intrusions well outside the exclusion zone for the Sun. The leading satellite always had a yaw bias of 180° in SM in order to maintain the link between the MWA of both satellites which were mounted on the +x side. As a consequence, when the leading spacecraft used SCA#1 as prime, the follower had SCA#2 prime, and the other way around.

The performance of the star cameras degraded slowly (see Fig.3 and 4). SCA#1 on GR2 started to deteriorate in the 8th year of the mission. The resulting difference in daily fuel expenditure wasn't worrisome at first but reached a factor of two after two to three more years. That meant that the extra fuel usage with SCA#1 on control reached ~1 kg per cycle.

An exchange of the relative positions at β´=0° crossings would allow GR2 to always use the better camera. A swap of leader and follower also costed fuel, of course, because one of the satellites had to be maneuvered over a distance of ~440 km. Collection of science data is impossible for inter-satellite distances ≤100 km (see 3.2.1).The exact cost depended upon initial and final conditions, upon the time slot allocated (the faster the more expensive) and upon the tangential distance required for a safe fly-by. The satellites were put in AHM once below 100 km. The goal was to perform swaps within 4-6 weeks in order to lose no more than one month of data. A swap required five 180° yaw turns if the orbit maneuvers were all made on one satellite. The other satellite then had to be turned only once.

The total cost of such a swap operation turned out to be in the order of 150 – 200 grams, a considerable saving when compared to the use of the poorer camera.

This would be true only if the other camera on GR2 (SCA#1) was not showing the same deterioration, which luckily turned out not to be the case (see Fig.4). Both cameras on GR1 showed almost identical behavior with a very slight degradation on SCA#2 towards the end. This meant that the swaps kept the better cameras prime on both.

Between 2013 and 2017 eight swaps were performed in order to keep the fuel consumption below 10 g/d in SM.

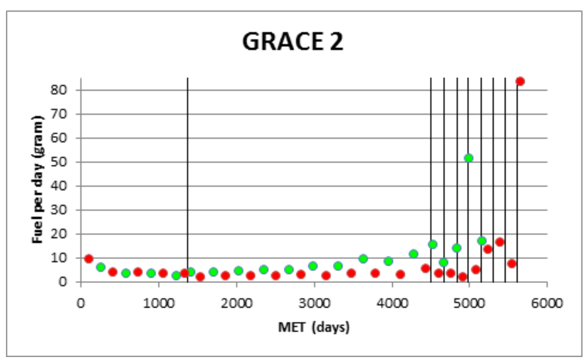


Fig. 3. *Fuel expenditure on GR2 as a function of mission elapsed time (MET). The prime star camera changes every 161 days due to the precession of the orbital plane (green dots: SCA#1 prime; red dots: SCA#2 prime). The deterioration of SCA#1 starts around MET 3000 (the 8th year of the mission). Satellite swaps (indicated by the vertical black lines) were initiated once the difference reached a factor of*

[1] The coupling between the axes is due to the fact that the magnetic field lines are not parallel to the flight direction

two. The first swap in 2006 was per design to equalize the wearing of the front-end. Towards the end eight swaps were made to keep the better camera on control.

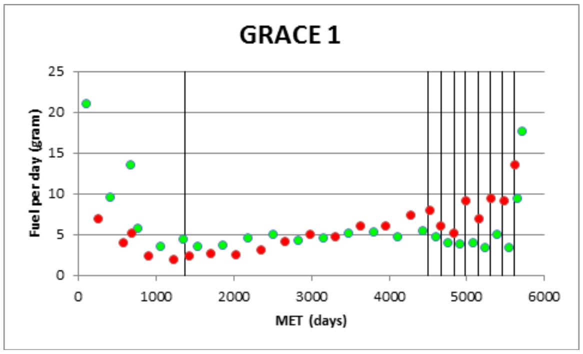


Fig. 4. *Fuel expenditure on GR1 as a function of MET. Again the green points indicate SCA#1 on control, the red dots SCA#2 and the vertical lines the satellite swaps. The same scale as in Fig. 3 is used for comparison.*

3.2.1. Instruments (ICU, MWA) on full-time

Pointing in SM is based upon information from a SGP4 propagator fed daily with the two-line elements (TLE) of both satellites. The resulting pitch bias with respect to the flight direction is ~-1° for the nominal inter-satellite distance of 220 km. Whereas the pitch bias decreases for smaller inter-satellite distances the yaw bias increases. This sets a practical limit of ~100 km below which SM is not operable.

SM was maintained as long as possible during the first swaps but AHM, aligning the satellites with the orbit, was used for inter-satellite distances <100 km. Fuel consumption in AHM was ~25% higher than in SM because it used a PD controller, a simple propagator and parameters had not been optimizes as in SM. Therefore, so-called modified TLEs were introduced from the 3rd swap onwards (MET 4843); leader and follower both used a virtual companion with the same eccentricity and inclination at a moderate inter-satellite distance (100-150km). By using such modified TLEs the biases were independent of the real inter-satellite distance and the satellites could stay in SM throughout. An additional benefit was that two of five yaw turns could be saved by continuing to use the virtual companion until the drift stop maneuvers.

Science data could be collected as long as both satellites had the correct orientation and the real companion was within the HPBW of the MWA.

The only disadvantage of this concept was that the poorer star camera had to be used on the satellite making the maneuvers for one to two weeks after β'=0° crossing. All maneuvers were made on GR2 for the first 14 years of the mission because its tanks contained 2kg more fuel. From 2016 onwards, however, orbit maneuvers were switched to GR1 which had two comparable star cameras and a better battery.

3.2.2. *Reducing the pitch bias during swaps*

Each cycle the virtual companion was put at a smaller distance in order to minimize the pitch bias. The modified TLEs had to be uploaded two or three times per week, due to the relative drift. The virtual distance was not allowed to become smaller than 50 km because a failed upload might then lead to a 180° yaw slew!

3.2.3. Instruments (partially) off

The eclipses were longest around β′=0° crossings. The low power situation on both satellites meant that it was not possible to keep instruments on during this time. A typical scenario would be the following: at first the MWA would be operated in Sunlight only and autonomously switched off in eclipses. Specific macros were written to switch it off when the battery voltage went below a certain limit and to switch it on when the voltage increased again after eclipse exit. The second stage would be the switch off of the ACC. A similar turn-on in sunlight was not feasible because the ACC needs several orbits to stabilize. The third and final step would be the complete power down of the MWA. Note that the three phases did not necessarily coincide on both satellites. The better conditions on GR1 allowed for a shorter interruption. The interval with minimal scientific return was used for the swap maneuvers.

3.3. Introducing a pitch bias in SM

The period without the ACC became increasingly longer on GR2 with every further failed cell. Gravity models could still be produced, however, as long as the ACC on GR1 remained on by transplanting the measured non-gravitational disturbances to GR2.

This was difficult due to the different yaw orientation of leader and follower, and due to the ~-1° pitch bias because the front-ends were pointing towards each other. A positive pitch offset was commanded in SM to align both satellites with the orbit, thus facilitating use of GR1 data on GR2.

A positive side-effect was the slightly lower fuel consumption because of the smaller drag. The microwave link still worked perfectly well with the introduced offset of ~1° (see 3.2.1).

3.4. Maneuvers

The exchange of leader and follower was always done around β'=0°, i.e. in the period with the longest eclipses when the collection of science data was either reduced or stopped altogether. The available timeslot for the swap was typically four to five weeks thus dictating a drift rate of ~450 / 30 ≈ 15 km/day. The drift start and stop maneuvers were normally split into two parts to control the eccentricity (important to guarantee a safe fly-by).

A particularity of the standard formation was the orientation of the satellites with the front-ends facing each other. The two orbit thrusters were located at the back (see Fig. 1). Thus 180° yaw turns were required for the drift start *and* for the drift stop maneuvers, independently whether they were made on leader or follower (assuming standard orientation at the moment of maneuvering).
All maneuvers were made on one satellite only, choosing the one with most remaining fuel. This was GR2 until 2015, but from 2016 onwards all maneuvers were made on GR1. The OCTs hadn't been used on GR1 since LEOP but were found to work perfectly well after fourteen years of inactivity.

The drift stop maneuvers were planned to minimize fuel expenditure. They were made at the lower end of the range – prescribed formation was 220 ± 50 km – in such a way that the natural variation in inter-satellite distance would yield a maximum after about two months and then return to ~170 km after a further 65 days. The next swap would thus benefit from favorable initial conditions. Savings were of the order of 25-30%, or ~50 grams.

Fuel was also saved by reducing the number of 180° yaw turns from five to three. This was achieved by using modified TLEs (see 3.2.1) which allowed continuous use of science mode but with the "wrong" orientation after fly-by (wrong for the standard formation but correct for the drift stop maneuvers).
Start and end times of the yaw slews were planned in the vicinity of the equator, thus maximizing the influence of the torquers during the acceleration and deceleration phases. The cost of each slew was thus decreased from ~15 to ~10 grams.

3.5. Data gaps

After the loss of the 6th cell on GR2 in September 2016 voltages dropped so low during eclipse that some components stopped working altogether (e.g., the transmitter: see [8] for more details). Two effects marred AOCS in particular when the voltage went below 15.8V. The delivery by the I/O board of SCA and GPS data from IPU to OBDH was interrupted for voltages <15.8V and the OBDH was autonomously switched off below 13.5V. The latter could, of course, only be deduced because no telemetry would be visible anymore.

3.5.1. *First mitigation measures*

The most obvious action that could be taken was to minimize the duration of the individual gaps and of the interval around β'=0° where they occur (see also [8]). The AOCS stayed passive when no data were delivered. Although the disturbance torques in eclipse are relatively small, still a substantial deviation occured in pitch. It was in the order of 0.5 * 360° * (gap duration)/P (about half of the difference between inertial- and nadir- pointing). A 10 min. outage thus led to a pitch offset of some 20°, 100 times larger than the dead band in SM.
The basic safe mode (coarse pointing mode, CPM) also worked during data gaps because the required CESS and magnetometer data were not delivered via the I/O board. However, CPM cost 100-200 times more fuel than SM. The persistency for a transition to CPM in case of deviations in SM was therefore originally set to 10 – 11 minutes already. Now it was even set higher, to ~1/4 orbit. Attitude limits and persistencies were all configurable and the FDIRs were adapted accordingly. The cost to recover from, e.g., a 90° offset in SM was lower than for a drop to CPM and subsequent recovery.

3.5.2. Commanded IPU off times

Resumption of the data flow over the I/O board required an IPU reset which was triggered autonomously on-board but added another 2 – 5 min. to the gap.
IPU resets could even take longer than this, mostly due to the re-acquisition of GPS satellites when the constellation happened to be unfavorable. An internal watchdog would trigger another reset after 15 min. The probability of an IPU hang-up increased with the length of the preceding data gap (the saved constellation becoming obsolete).

A commanded switch off of the IPU towards the end of an eclipse had the advantage that it saved power and restarted in a controlled manner. Several hundred power cycles were commanded on GR2 starting in November 2015.

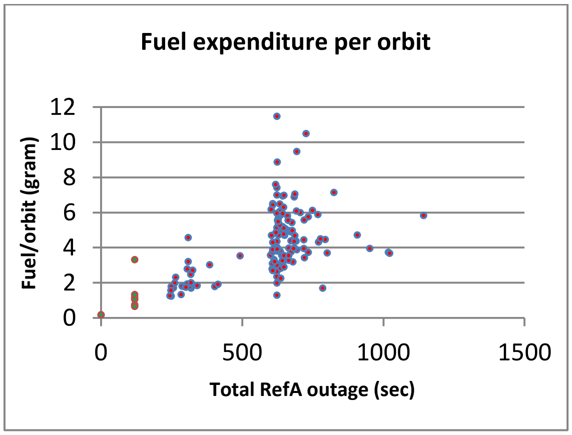


Fig. 5. Fuel expenditure per orbit as a function of the total time without AOCS control, i.e. the commanded outage plus recovery. The length of the commanded outages was 120, 180, 480 and 720 sec. Recovery after a power cycle took 2 – 5 min., slightly longer than after a soft reset.

But also this method had some disadvantages. The IPU off command had to be executed just before the critical voltage was reached and the power on had to be at a moment that high enough voltage was guaranteed again. Also, the recovery after an IPU power cycle took longer than after a soft reset. For outages with a total duration >600 sec the fuel consumption increased to ~100 grams/day (see Fig. 5). The length of eclipses increased slowly towards β′=0° and thus the off time had to be adapted regularly. Also, two additional commands per orbit had to be sent.

This approach had to be discontinued in November 2016 after 1½ cycles. The necessary off times of the IPU became longer than ten minutes due to the degrading battery and several times the recovery took longer than average (due to poor GPS constellation). This led to attitude offsets of ≥45° being reached regularly, which delayed the charging in Sunlight and hence reduced the voltage at next eclipse entry: a vicious circle. The ultimate consequence was that in one orbit the battery was not charged at all during the Sun phase. The battery voltage went below 13.5 V and the OBDH was powered off

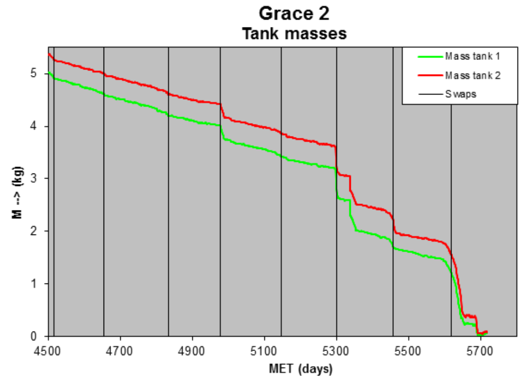


Fig. 6. Tank masses on Grace 2 as a function of MET. The events around November 2016 (MET 5300) when commanded IPU outages were used are clearly visible. The improvement at the next β'=0° crossing (MET 5450) with accepted data gaps accompanied by on-board mitigation measures is also apparent.

3.5.3. Further mitigation measures by on-board macros

Star camera data were the first to be delivered after an IPU reset. GPS data typically returned five to ten seconds later. Analysis revealed that attitude control resumed with the first sample of star camera data. This led to enhanced activity because obsolete GPS information was used.

A three-step macro chain was built that inhibited thruster activity for ten seconds after the return of star camera data. The length of the inhibition could be set. Fuel consumption decreased considerably to ~35 grams/day for outages of ~ 10 min, a gain of at least a factor of two.

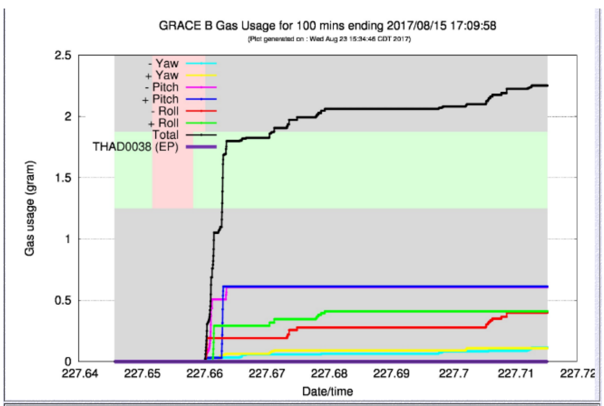
4. Desperate measures, hibernation and chase maneuvers

On August 3rd (MET 5618) 2017 β'=0° was crossed and thereafter eclipses became shorter again. The strategy described in 3.5.3 went well and the situation was still looking promising. The team started to prepare for the next phase with science operations, which could possibly start in September 2017, when another cell was lost on August 17th (see Table 1) This time it turned out to be a permanent failure.
By the middle of August 2017 GR2 had roughly 2.6 kg of fuel left and the daily consumption was 30-35 grams due to the IPU data gaps (see Fig.7 top panel). That was still enough to reach the next full Sun period and to collect 3 – 4 months of science data. In an extremely optimistic scenario it might even have

been possible to survive another β'=0° crossing and to continue in 2018. The final loss of the 7th cell changed this completely.

The satellite could this time be recovered relatively quickly after the OBDH restart, but unfortunately the data gaps became ~20 minutes with only 13 out of 20 cells providing voltage and eclipses still >30 minutes. The attitude deviations became > 35° for >200 seconds and mode drops to CPM occurred almost every orbit. Fuel consumption increased to about 100 grams/day (see Fig.7 lower panel). At this point GR2 had thus roughly 2kg of fuel left. The fuel was more or less equally distributed between tanks (1.00+/.0.07kg per branch). Unfortunately it was not clear how much of the remaining fuel could be used, as an unknown quantity (estimations were about 0.5kg) was not usable (ullage). Also low pressure decreased significantly making attitude control less effective. With all these information the guess was there was fuel left for 20 days and the full Sun phase was still more than 5 weeks away.

It was still possible, though complicated and time consuming, to operate the satellite with only 13 cells left but there would not be any fuel left when the collection of science data could be resumed. Immediate action without lengthy analysis was required with fuel depletion at 5% per day.



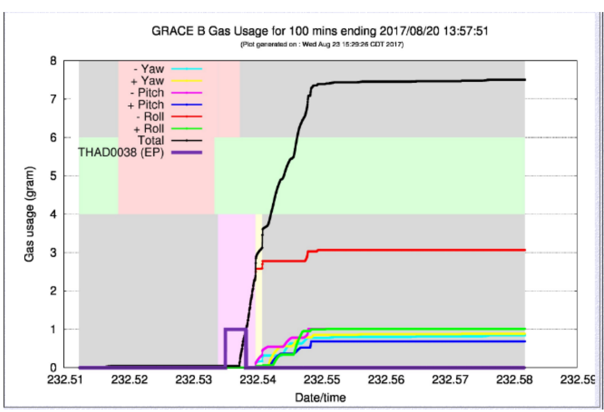


Fig. 7: The two plots above show the difference in fuel consumption on Aug 15th (one day before losing the 7th cell) and Aug. 20th over an interval of 100 minutes. The light red area in the upper left visualizes the length of the IPU data gap. In the upper panel this was ~9.5 min. and in the lower panel ~20 min. , Attitude control resumed as soon as the IPU had a valid GPS solution and thruster actuations increase (black line). Before the loss of the 7th cell the data gaps were short enough to stay in AHM. The extra gas consumption was about two grams per orbit, i.e. ~30g/d. The gaps became more than twice as long after the loss of the 7th cell and a drop to CPM would occur (pink bar on the bottom). Recovery cost ~ 8 grams, four times more than before.

So called "desperate measures" were started on GR2 within a few days in the hope to save enough fuel to reach the full-Sun phase in Oct/Nov 2017.

4.1. *Change of FDIR settings*

The first step to cope with the loss of the 7th cell was the adaption of those monitoring functions pertaining to the AOCS and IPU. By increasing the maximum time allowed for SCA outages and for an invalid reference attitude the number of drops to CPM could be reduced. However, too large persistencies led to such large attitude deviations that the fuel consumption to recover to nominal mode was unacceptably high.

4.2. *Introduction of an attitude bias to gain power*

The Sun is in the x-z-plane of the satellites near β′=0° (see Fig. 1). It illuminates the top panel full, but the side-panels under a 40° angle only. The effective area is roughly 50% when compared with the situation where the Sun is perpendicular to one of the side-panels. It takes ~5 min. after eclipse exit before the Sun illuminates the panels and a similar interval is lost before the entry of the next eclipse.

4.2.1. *Yaw bias*

A possible solution would be to introduce a 90° yaw bias. This would shorten the intervals without Sun on the panels and increase the effective area, in total a gain of a factor of two which would shorten the off-times on the IPU by a factor of two also. The tools to implement the above had all been developed, but several serious drawbacks prevented their practical use.

A 90° offset would have increased the area-to-drag ratio by a factor of two which at the altitude of 320 km at this time would imply a significantly higher decay rate. To keep GR1 in the vicinity would have been possible only if it was also set to yaw = 90°. The prime star camera would have to be switched exactly in the middle of the Sun phase. Some 60 additional commands would have to be sent and timing would definitively have been an issue. Also, the poorer camera would have to be used for ~1/4 orbit each time. In the end it was decided against this option.

4.2.2. Roll bias

A roll bias also increases the effective area, because the side-panel is turned towards the Sun (see Fig. 1). It also allows the MWA link to be maintained. Charging would still be delayed by ~5 min after eclipse exit, however. Several settings were tried in orbit, ranging from 5° to 30°. The smaller biases did not produce a significant gain in power and for some reason (not fully understood) attitude control became instable at an offset ≥25°.

4.2.3. Pitch bias

A pitch bias of 30° achieves the same integrated effect as a 90° yaw bias (see Fig. 1). Charging starts almost immediately after eclipse exit, but stops earlier. The latter is no problem because at the end of the Sun phase the battery is full. The area-to-drag ratio increases, but not so much as for a 90° offset.
Tests were made with biases ranging from 10° to 25°. The effect on the battery was as expected (see [8]) but fuel expenditure increased to 45 g/d for the largest offset.
This method could, therefore, not be used on GR2 due to its shortage of fuel, but it was applied on GR1 (see Table 2).

Table 2: Pitch bias as implemented on GR 1 in order to to gain power but also to increase the area-to-drag ratio deliberately to decrease the difference in decay rate between the two satellites

Pitch bias	2017 (DoY)	Fuel (g/d)
-10°	261 - 263	6.0
-20°	263 - 264	10.1
-25°	264 - 269	36.6
-27°	269 (2 orbit test)	45.7
-25°	269 - 285	32.8

The pitch bias on GR1 was not only used to get better charging of the battery but also to assimilate its drag to that on GR2 which was in hibernation mode during this time (see Sect. 4.5).

4.3. Shifting the Center of Mass

The data gaps on GR2 remained at the 30 min. level despite all measures tried. A final idea was to minimize the disturbance torques in pitch in order to minimize the thruster activity after gaps. The connecting valve between the two tanks (which are located on the x-axis; see Fig. 1) was opened but, although telemetry had indicated a mass difference between them, this did not equalize. The two trim masses on the x-axis were moved as far as possible to the front of the satellite, but no beneficial effect was seen in the fuel expenditure.

4.4. Rate damping without thrusters

Another attempt to save some fuel was to command the S/C to perform RD in CPM with magnetorquers only. While usually in CPM mostly the thrusters are used for RD and Earth orientation the aim of this command was to reduce the thruster activity, especially in the time period after the mode drop. As mentioned earlier, CPM on GRACE was very expensive in terms of fuel expenditure. As the default CMCPM only worked properly in orbit phases with Sun illumination, a so called emergency RD mode was implemented in order to keep at least the rates low during such periods. Emergency RD was never used before during the mission, but it turned out to work surprisingly well. Few days of data collection and fine-tuning showed first results and fuel consumption decreased by about 30% (~60g/day). This concept was successfully followed for about one week until 2nd September, 2017. By then the IPU outages due to power problems became short enough again to stay in SM without having to go to CPM.

4.5. *Hibernation*

The amount of remaining fuel continued to decrease rapidly on GR2 after the loss of the 7th battery cell despite all mitigation measures as described in the previous sections. The full-Sun phase would only be reached in the beginning of October 2017 (DoY 286), still more than a month away.
Another OBDH reboot took place early September but this time the satellite was not seen for several days. The TM/TC link could only be re-established after three days by switching to the redundant processor module and back. It was discovered that during these three days no fuel had been used at all. The satellite had survived in an unknown mode that no one can describe because we had no telemetry. This mode was called "hibernation" by the project

later. Therefore, a plan was devised to send the satellite deliberately into hibernation again until the eclipses would become short enough to allow science operations.

- Both high pressure latch valves were closed thus preventing any fuel use until re-activation.
- The FDIRs were re-configured to prevent an autonomous re-enabling of the thruster or opening of the valves..
- The IPU was switched off
- The satellite put into AOCS safe mode.
- The FDIRs were re-configured to prevent an autonomous re-enabling of the thruster or opening of the valves.

Attitude control was still active but with the MTQ only. The GR2 satellite tumbled but its rates remained at ≤2 mrad/s. The remaining fuel was judged to be sufficient to stabilize the satellite after re-activation and to operate it throughout the ensuing full-Sun phase.

The uncontrolled attitude enhanced the area-to-drag ratio which led to an increase in the decay rate from ~100 m/d to >200 m/d (see Fig. 8).

4.6. Chase maneuvers on GR1

GR2 was the leading satellite after the last swap. The inter-satellite distance started to increase rapidly after hibernation was initiated.

Efforts were made to assimilate the decay rates of the two satellites by introducing a pitch offset on GR1 (see Table 2). This offset was beneficial for the battery on GR1 but the orbit decay remained smaller than on GR2 and it was certainly not possible to decrease the inter-satellite distance with this method. It had grown to >1000 km in less than a week precluding the resumption of science which required 220 ± 50 km..

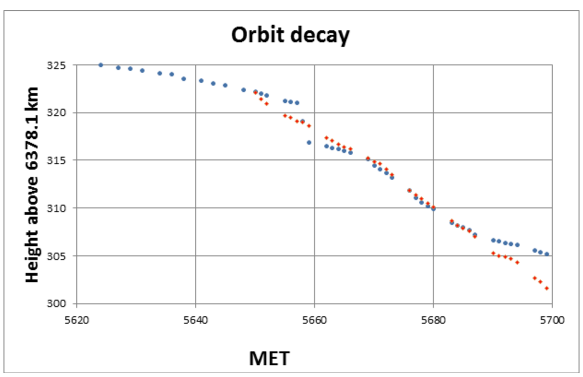


Figure 8. Decay rate as a function of MET during the last weeks of the mission. It was the same for both satellites until MET 5650 when GR2 was sent in hibernation mode (from this moment GR2 is shown by the red points). The lower decay on GR1 (blue points) had to be increased by implementing a pitch bias of up to 25° and by making numerous "chase" maneuvers (large step-down). Finally, at MET 5685, GR2 had to be abandoned at which moment GR1 returned to its nominal attitude and decay rate.

Therefore, a total of thirtyseven "chase" maneuvers were made in addition to the commanded pitch bias from 13 September (MET5659) until 6 October (MET5682 – see Fig. 8). The total cost amounted to 2.2 kg and reduced the amount of remaining fuel on GR1 to ~600 grams, the same level as on GR2. The standard inter-satellite distance was finally re-established on 6 October.

The efficiency of the maneuvers started to decrease from ~ 100% at the beginning to 97% by the middle of September and to 90% on October 1st. The efficiency of the last two maneuvers dropped to 69% and 62%. The decrease in mass-flow-rate was caused by the decrease in low-pressure which in turn was caused by tank pressures below the operational limit for the pressure regulators.

It was obvious that, if the decay rate on GR2 was not stopped at this point, a further catch-up with maneuvers on GR1 would be impossible.

4.7. Re-activation of Grace 2

It was decided to start the re-activation of GR2 on 04.10.17. The situation at that moment was

- Remaining fuel 500g on GR2 and 600g on GR1 (deemed to be enough for two months of data collection in SM)
- Inter-satellite distance 410 km (decreasing by ~30km/day)
- Duration of eclipses <10 min.

A detailed command sequence had been prepared to re-open the valves and to re-activate the thrusters. The IPU had to be switched on to allow a quick transition into AHM. The IMU, which was powered on autonomously after each cold OBDH restart, had to be switched off as soon as possible.

The battery situation was good at the moment the recovery was started, but it deteriorated instantaneously at the moment that the IPU was switched on. Each time a further OBDH boot was triggered, worsening fuel and power situation.

The recovery was interrupted and a (final) attempt was made on October 9. Now the situation was

- Remaining fuel on GR2 300g
- Inter-satellite distance 207km
- Duration of eclipses <2 minutes

This last attempt was not successful and the remaining fuel on GR2 completely depleted. Decommissioning started immediately by executing an already prepared decommissioning sequence of events.

The remaining 600 g of fuel on GR1 were used to prepare for the follow-on mission. A series of calibration maneuvers were made to characterize the accelerometer at larger values than had been possible during the mission. These were performed in the four weeks before the decommissioning of GR1 was started on December 1 2017.

5. Grace Follow-on: improvements and performance

The two satellites of the Grace Follow-on mission were successfully launched on 22 May 2018 in a polar orbit at an altitude of 491 km. The new mission continues the measurements of the gravity field by GRACE. The design benefits from the experience gained from their predecessors. In particular, the number and quality of the sensors has been improved. The use of an IMU with high performance enhances the accuracy of rate measurements with four independent axes, whereas on GRACE only one IMU channel was available. The four inertial sensors are arranged in a tetrahedral configuration offering one-failure tolerance. Three sensors are used during nominal operations, the fourth only after a transition into safe mode.
GRACE had two star cameras of which only one was actively used for AOCS.
GFO has three star cameras with a separation between their bore sights of 80.4° to 100°. They are used in hot redundancy. This configuration optimizes the coverage and increases the accuracy in all three axes. The new generation of cameras can handle partial intrusions and the on-board software autonomously delivers fused data from one, two or three cameras depending upon validity.
Fused data from all three cameras were delivered for more than 60% of the time in the first year of the mission, whereas 0.2% of the time attitude data were derived from a single camera only (the other two being blinded by the Sun and the Moon simultaneously; see [Table 3]).
The individual cameras deliver valid data for 83 - 94% of the time (see Table xx), whereas on GRACE the camera on the sun-side was blinded for ¼ orbit (the exclusion angle was ±40° but increased to ±50° towards the end of the mission).
The three cameras are operated by one of two electronic units and the data are handled by the OBC directly. This implies that the attitude determination on GFO is independent of the IPU in contrast to the situation on GRACE.

Table 3: percentage of quaternion samples derived from the fusion of the data of 3, 2 or 1 star tracker head (on top) and percentage of valid samples delivered by each star tracker head (on the bottom) with respect to the unit on-time, after one year in orbit.

	% of on-time	
	GFO1	GFO2
Fusion type:		
3 star cameras	69.60	65.85
2 star cameras	30.22	32.39
1 star camera	0.18	0.18
Validity:		
STR 1	93.33	91.79
STR 2	86.45	84.19
STR 3	89.62	86.51

GPS data are still provided by the IPU and used to initialize the on-board orbit propagator at each cycle. The first FDIR reaction in case of GPS outages is triggered after 24h (configurable) only, again limiting the dependency of attitude control on the IPU.
The attitude is maintained primarily by a set of cold redundant MTQs. These are aligned with the axes of the satellite and located at maximum distance from the magnetometers in order to minimize disturbances. The MTQs are supplemented by a cold gas propulsion system with a set of twelve 10 mN ACTs, separated into two branches but operated simultaneously to ensure an even depletion of the tanks. The design is inherited from GRACE with state of the art enhancements and small improvements like a two-stage pressure regulator that assures a constant feed pressure for the thrusters over the whole range of tank pressures.

The fuel consumption in all attitude modes is considerably smaller on GFO than it was on GRACE. In AOCS safe mode the satellite is oriented with the z-axis directed towards the Earth and a yaw angle of 0° or 180°, depending on which value is closer to the current attitude. The more powerful battery on GFO (78Ah name-plate capacity on GFO compared to 18Ah on GRACE) tolerates temporary yaw deviations of up to 90° in safe mode. No "yaw-steering" concept, forcing one of the side panels towards the Sun, is applied on GFO leading to a gain of a factor of 20 - 100 in fuel consumption as compared to GRACE. Also, the nominal modes perform better on GFO than on GRACE with an expenditure <1 g/d in fine pointing mode (cf. 3 – 5

g/d in the equivalent SM on GRACE) and ~0.5 g/d in attitude hold mode (cf. 10 g/d in AHM on GRACE).
The use of TLEs to point towards the partner satellite is on GFO not restricted to the fine pointing (science) mode. Pointing to the other satellite depends upon the frame of reference used. This can be set by command in any mode (apart from safe mode), thus allowing for extra flexibility when performing special tests.
A very useful feature on GRACE was its ability to perform an auto-recovery after a drop to AOCS safe mode. This minimized the interruption of payload operations and fuel expenditure because the satellite returned autonomously to science mode within 10 – 30 min. in most cases. This possibility is currently not used on GFO. However, the fuel consumption is small in AOCS safe mode so the only consequence is an interruption of the acquisition of data for science.

6. Conclusions

The GRACE mission was continued more than ten years longer than originally planned. Many ideas and creative solutions were needed to make the fuel last for almost 16 years. The fact that ~7% (3.5% per year) of all fuel was used during the last two years shows how successful this was.
The GFO mission uses the experience gained on Grace:

1. The integration of the Star Camera software in the IPU software led to the situation that with every IPU problem there also occurred an SCA and therefore an attitude control problem. In order to improve the situation for GFO the Star Cameras software are decoupled from the IPU software. Hence AOCS performance is not anymore influenced by any IPU issue (except for GPS data which is supported by an OOP).
2. One of the two star camera heads on Grace was blinded by the Sun up to 25 minutes per orbit depending on the Sun angle. A switch of the prime camera e.g., in case of Moon intrusions or $\beta'=0°$ crossing, had to be commanded. GFO has three cameras of which at least one is valid for 99.94% of the time. The AOCS safe mode on GRACE used up to 1kg of fuel per day which was 100-200 times more than the nominal modes. This was due to the concept of “yaw-steering”. The more powerful battery, the more relaxed control and a more intensive use of the magnetic torque rods reduce expenditure in safe mode on GFO to ~8g/d.
3. Weight thruster / torquers was shifted to the latter (more power, less fuel usage). Improvement was possible due to reduced interaction of MTQ with magnetometers
4. IMU: determination of rates is more stable (see 5)

Acknowledgements

GRACE represents an outstanding example of international cooperation with a common goal in mind. The authors would therefore like to thank the every GRACE team member for their contributions over the many years of the mission.

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