A simplified gravitational reference sensor for satellite geodesy

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Abstract
A simplified gravitational reference sensor (S-GRS) is an ultra-precise inertial sensor for future Earth geodesy missions. These sensors measure or compensate for all non-gravitational accelerations of the host spacecraft to remove them in the data analysis and recover spacecraft motion due to Earth’s gravity field. Low–low satellite-to-satellite tracking missions like GRACE-FO that use laser ranging interferometers (LRI) are limited by the acceleration noise performance of their electrostatic accelerometers and temporal aliasing associated with Earth’s gravity field. The current accelerometers, used in the GRACE missions, have a limited sensitivity of \(10^{-10} \text{ m/s}^2/\text{Hz}^{1/2}\) around 1 mHz. The S-GRS is estimated to be at least 40 times more sensitive than the GRACE accelerometers and over 500 times more sensitive if operated on a drag-compensated platform. This improvement is enabled by increasing the mass of the sensor’s test mass, increasing the gap between the test mass and its electrode housing, removing the grounding wire used in GRACE, and replacing it with a UV LED-based charge management system. This allows future missions to take advantage of the sensitivity of the GRACE-FO LRI in the gravity recovery analysis. The S-GRS concept is a simplified version of the flight-proven LISA Pathfinder (LPF) GRS. Performance estimates are based on models vetted during the LPF flight and the expected spacecraft environment based on GRACE-FO data. The relatively low volume \(\sim 10^4 \text{ cm}^3\), mass \(\sim 13 \text{ kg}\), and power \(\sim 20 \text{ W}\) enable the use of S-GRS on microsatellites, reducing launch costs and allowing more satellite pairs to improve the temporal resolution of gravity field maps. The S-GRS design and analysis, as well as its gravity recovery performance in two candidate mission architectures, are discussed in this article.

Keywords Satellite geodesy · Gravity measurements · Accelerometers · Drag compensation

1 Introduction

Precision inertial sensing is important for many space science missions, including fundamental physics experi-

ments (Everitt et al. 2011; Touboul et al. 2001) and gravitational wave observation (Danzmann and Rüdiger 2003). The S-GRS presented here specifically targets future satellite geodesy missions, following the successful CHAMP (Reigber et al. 2002), GRACE (Tapley et al. 2004), GOCE (Drinkwater et al. 2003), and GRACE-FO (Sheard et al. 2012) missions.

The GRACE (2002–2017) and GRACE-FO (2018–present) missions in particular have provided a nearly 20-year climate data record of Earth system mass-changes. The utility of the data is incredibly diverse for serving both scientific and societal applications. Contributions include quantifying the rate of ice sheet and glacier accumulation and ablation globally, identifying and quantifying areas of unsustainable groundwater withdrawal due primarily to heavy irrigation, quantifying global and regional sea level changes and ocean
heat content (when combined with satellite altimetry), monitoring drought severity through operational assimilation in the U.S. Drought Monitor, and assessment of geohazards, including mapping mass redistributions due to large earthquakes (Tapley et al. 2019). The importance of mass change observations was recognized in the 2018 Decadal Survey for Earth Science and Applications from Space, where Mass Change was listed as a Designated Observable (National Academies of Sciences Engineering and Medicine 2018), and is now a core component of NASA’s Earth System Observatory to be launched in the next decade.

The quality of mass change data products derived from a gravity mission depends on the orbital characteristics of the mission (altitude, inclination, number of satellite pairs, etc.), as well as the precision of the on-board measurement system (inter-satellite ranging system, accelerometer, attitude and orbit determination). For GRACE-FO, it is well understood that the limiting source of error is due to temporal aliasing, i.e., undersampling of high-frequency mass variations in the Earth system such as ocean tides and weather systems (Fig. 1, top, solid green curve includes total temporal aliasing), and improving the precision of the measurement system does not improve the quality of the gravity fields. There are two possible ways to reduce the impact of temporal aliasing error on derived gravity fields: 1) sample more frequently via more satellite pairs (Daras and Pail 2017; Elsaka et al. 2013; Wiese and Nerem 2011; Wiese et al. 2011a, b) as shown in Fig. 1 (bottom), and 2) process the data in an “along-the-orbital track” approach rather than accumulating observations over a finite number of days, which is traditionally done. Solution 1) is well understood, having been extensively studied in the literature, while solution 2) is less well understood, but recent analysis suggests it is a promising approach. For example, first “along-the-orbital track” analysis of the GRACE-FO LRI data revealed differences between geopotential models and characteristics of the error signals in them (Ghobadi-Far et al. 2020). This method can also be applied to increase the sampling of short-period mass change processes that would otherwise produce biased estimates with a standard method (Han et al. 2021a, b).

Future mission implementations should seek to minimize the impact of temporal aliasing error via one of the above two solutions. The precision of the accelerometer becomes important in such a design, as those in GRACE-FO limit both approaches. The GRACE-FO electrostatic accelerometer is the largest source of error contained within the measurement system (Fig. 1, top, blue dashed curve), and has been shown in previous studies (Purkhauser et al. 2020) to be insufficient for multi-pair implementations at lower altitudes (~350km), limiting the accuracy of the recovered gravity fields. We refer the reader to the results of the NASA Mass Change Designated Observable Study (Wiese et al. 2021) for a complete list of accelerometer technologies that are suitable for multi-pair low-altitude implementation. The S-GRS offers a technological pathway to improve the measurement of non-gravitational accelerations (Fig. 1, bottom, dark blue curve) to an acceptable level for such constellation-type implementation. Furthermore, and perhaps even more importantly, the S-GRS offers a technological solution to measuring non-gravitational forces at a level that contributes equally to the gravity recovery error when compared to the LRI that was successfully demonstrated on GRACE-FO (Fig. 1 shows comparable magnitude for the solid-blue and red-dashed curves) (Abich et al. 2019). Such a match in performance between the accelerometer and the inter-satellite ranging instrument has potential to allow for full exploitation of the data in future “along-the-orbital track” data analysis, as discussed in Spero (2021). The current mismatch in performance between the LRI and the accelerometer on GRACE-FO limits interpretation of “along-the-orbital track” signals to smaller spatial scales only; such analysis is currently not valid for the longest wavelengths in the gravity field, where the signal amplitudes are the largest and the
accelerometer error is present along with the dominant residual centrifugal acceleration (Ghobadi-Far et al. 2018, 2022).

Figure 2 uses methods described in Hauk and Wiese (2020) to show the maximum potential improvement the S-GRS offers in the architectures listed in Fig. 1. Maximum scientific potential can be assessed by looking at the measurement system performance (neglecting temporal aliasing error and orbit dynamics error); this assumes temporal aliasing error is mitigated via “along-the-orbital track” data analysis or future improvements in models of high-frequency mass variations. The potential for improvement offered by the S-GRS in the dual-pair architecture, in particular, could advance the measurement parameters beyond those targeted in the Decadal Survey and identified in the Baseline Measurement Parameters in the Science and Applications Traceability Matrix identified by the Mass Change (MC) Study Team for the next observing system implementation (NASA 2020a, b).

1.1 Simplified gravitational reference sensor

The S-GRS design follows that of the flight-proven LPF gravitational reference sensor (GRS) that represents the state of the art in precision inertial sensors (Dolesi et al. 2003). The LPF GRS used a 2 kg, Au/Pt test mass (TM) inside a molybdenum electrode housing (EH). The housing held 12 gold-coated electrodes to differentially sense the position and orientation of the cube via capacitive sensing and to control it using electrostatic actuation. Six “injection electrodes” were driven with a 100 kHz AC bias voltage to frequency shift the capacitive measurement to high frequency. The 4 mm gap (in LPF) between the TM and housing was defined by a trade-off; it was large enough to reduce noise sources, such as uncontrolled potentials on the electrodes, and small enough to measure TM displacement at the nanometer-level over the measurement bandwidth. A caging and venting mechanism used a series of mechanical fingers to secure the TM during launch and release it in orbit (Bortoluzzi et al. 2013). During science operations, the TM charge was controlled by a charge management system (CMS) based on UV photoemission using Hg vapor lamps as the UV source (Armano et al. 2018b). The CMS eliminates the need for the small grounding wire used in the ONERA accelerometers that both limits their performance and causes challenges during integration and testing (Touboul et al. 1999).

The primary performance metric for these instruments is the acceleration noise from the residual spurious forces acting on the TM, often defined by an amplitude spectral density (ASD). LPF, launched in 2015, exceeded requirements by more than a factor of ten with a measured acceleration noise below $3 \times 10^{-15} \text{m/s}^2/\text{Hz}^{1/2}$ between 0.4 mHz and 20 mHz (Armano et al. 2018a), the same frequency band that is important for Earth geodesy. This represents a factor of $3 \times 10^4$ improvement over the performance of GRACE-FO and GRACE-FO, and $10^3$ over that of the GOCE instrument (Christophe et al. 2010).

The S-GRS is a scaled-down version of the LPF GRS, with reduced mass and complexity, and it is optimized with respect to performance for future geodesy missions (see Fig. 1). Key features of the design and its implementation that improve sensitivity with minimal size, mass, and power are:

- The TM grounding wire used in previous accelerometers is removed, TM charge is instead controlled via non-contact UV photoemission as in LPF. The grounding wire is one primary source of acceleration noise in current electrostatic accelerometers.
- Removing this grounding wire allows an increase in the TM-to-EH gap from $\sim 100 \mu \text{m}$ to $\sim 1 \text{mm}$. The TM mass can also be increased from $\sim 100 \text{g}$ to $> 500 \text{g}$ by changing the dimensions (30 mm cube) and using a high-density material like the Au–Pt alloy. Acceleration noise (for surface forces) scales inversely with TM size and density, and to first order, with the gap size between the TM and EH.
- Careful control of the environment surrounding the TM, including venting to space to reduce residual pressure, and shielding against Earth’s magnetic field and thermal fluctuations. This will be done through a vacuum enclosure and magnetic shield that are part of the S-GRS.
- Tightly integrating the S-GRS and LRI, eliminating the microwave ranging instrument used on GRACE-FO to minimize size, mass, and power.
- Design for the possibility of operating the S-GRS on a drag-compensated platform to further reduce acceleration noise, while simultaneously maintaining orbit altitude.
The instrument consists of an S-GRS Head and an Electronics Unit. The detailed CAD design of the S-GRS Head is shown in Fig. 3. It is the mechanical unit containing the TM and EH, the caging mechanism, and the vacuum enclosure with a compact version of the triple mirror assembly used on GRACE-FO. The Electronics Unit performs all control and readout functionalities and is separated from the S-GRS head to minimize electromagnetic and thermal disturbances to the TM. In the remaining sections of the paper, each sub-element of the S-GRS will be described, beginning with the S-GRS head in Sect. 2 and the S-GRS electronics unit in Sect. 3. In Sect. 4 we present a breakdown of the S-GRS acceleration noise performance followed by simulations of the test mass position control algorithms in Sect. 5 and conclusions in Sect. 6.

The S-GRS is proposed to be the inertial sensor of future inter-satellite ranging missions like GRACE-FO. It may be used within a pair of low orbiting satellites using drag compensation or even multi-pair missions. Performance estimates of the instrument are based on an Earth orbiting environment, but it may be used for geodesy in other planets (Genova 2020).

## 2 S-GRS head

### 2.1 Test mass and electrode housing

Detailed performance analyses determined that a 30 mm cubic Au–Pt TM with a mass of 540 g results in the desired acceleration noise performance described in Sect. 4, while keeping mass and volume of the sensor to a minimum. The TM is a gold-coated cube with two indented features on opposite sides, one conical and the other pyramidal, used to hold the test mass on ground and during launch.

The electrode housing design is a scaled-down version of the LISA-like EH prototype previously fabricated by the University of Florida (UF) team and shown in Fig. 4. The inner dimension of the cubic EH is 32 mm creating a 1 mm gap between the TM and EH, and its wall thickness is approximately 10 mm. The technology readiness level (TRL) 5 EH will be fabricated from gold-coated molybdenum with sapphire electrode spacers (see Appendix B for information on the TRL system). Molybdenum is chosen for its low magnetic properties, machinability, and high thermal conductivity, which reduces thermal gradients across the sensor. Sapphire is chosen because its coefficient of thermal expansion is nearly the same as that of molybdenum.

The electrode geometry, shown in Fig. 5, differs from that of LPF. The S-GRS design has only one large electrode to sense the test mass motion in the sensitive direction. This larger electrode area maximizes sensitivity in this direction and minimizes cross coupling with other degrees-of-freedom that occurs when more electrodes are present. Around the holes used for the CM fingers, both faces normal to the Y-axis have 2 injection signal electrodes and 2 sensing electrodes that can measure TM motion along the Y-axis and rotation along the Z-axis. Faces normal to the Z-axis have 1 injection electrode and 4 sensing electrodes to detect TM motion along the Z-axis and rotation along the X and Y axes.

### 2.2 Triple mirror assembly

The Triple Mirror Assembly (TMA), shown in Fig. 6, is an optics mounting structure located on the +X face of the vac-
uum enclosure (see Sect. 2.3). It acts as a retroreflector at the ends of the “racetrack” configuration for the LRI that measures the distance between the pair of satellites (Dahl et al. 2017). The input beam is reflected in three orthogonal planes, producing an output beam at an offset distance in an antiparallel orientation. The TMA is designed so that the virtual vertex at which the three planes intersect is coincident with the center of mass of the spacecraft’s inertial reference body, the TM, when it is located in the center of the EH. This way, the LRI’s distance measurement is done between the reference masses of each satellite.

The concept for the TMA is based on an equivalent component of the GRACE-FO mission (Abich et al. 2015; Kornfeld et al. 2019). The corner-cube principle is similarly implemented in both designs, with the purpose of returning the LRI beam, regardless of pointing variations. The proposed TMA is significantly smaller than the one used in GRACE-FO. This is because in GRACE-FO, the laser interferometer had to be built around a Ka-band microwave link which was used as the primary interferometer.

The TMA is made of titanium alloy, with three sapphire mirrors supported by titanium alloy retainers and PEEK spacers. It has a yaw angle of 45° from the chamber wall for a symmetric aperture of the mirrors in the input and output planes. It also has a pitch angle of 20° that reduces the volume of the body that supports the mirrors. There is a mean distance of 129 mm between the input and output beams.

An advantage of the symmetric geometry of the TMA and its large contact area with the vacuum chamber wall of the same material (see Sect. 2.3), is the athermal behavior on the vertex position with respect to the TM center. Any thermal deformation of the chamber would similarly affect the TMA planes, reducing any changes in the vertex position. The goal is to maintain the TMA vertex within 50 μm of the TM center and a co-alignment error of less than 50 μrad.

2.3 Vacuum enclosure

The S-GRS electrode housing and TM are enclosed within a vacuum enclosure, shown in Fig. 3 to limit the effects of Brownian noise and improve the sensing system’s noise environment (McNamara et al. 2013) with a goal pressure of 10⁻⁵ Pa. While on ground, the enclosure can be pumped out for testing purposes. On orbit, residual gas in the chamber will be vented to space through a duct to separate the system from the outgassing of the rest of the vehicle. The chamber is also the main supporting structure of the S-GRS components; the TM is located at its center, the two sections of the caging mechanism are positioned on opposite sides of the EH, and the TMA is mounted on the external +X face of the enclosure (see Sect. 2.2).

The vacuum enclosure is designed to support the TMA on its exterior and contains the TM, EH, caging mechanism, magnetic shield, and their supports. The EH is fastened and aligned through its frame to stiffener-like surfaces within the chamber that are placed away from the TM center. These surfaces are also used to support the standoffs and fasteners for the magnetic shield that surrounds the EH. Twenty coaxial SMA feedthroughs that connect to the EH electrodes are distributed around the top and bottom sections of the enclosure, except on the TMA face. Two D-Sub 25 pin feedthroughs that connect to the caging mechanism actuators are placed on the −X face. The UV injector is attached to a ConFlat-type (CF) flange located at the central section of the enclosure between the +Z and −X faces at a 45 deg angle along the +Y axis. The enclosure vent duct is attached to a flange on the +Z face.

The chamber is based on the LPF design, but with a smaller size and rectangular shape. The 5 mm thick walls were selected to keep the maximum deformation from pressure and thermal loads below 10 μm. With the CF-type square
ends, the chamber is 235 mm long, with an internal square section of 78 mm at the center.

The UV injector feedthrough and flange are based on commercially available CF options, similar to the one used in the LISA Inertial Sensor UV Kit (ISUK) (Armano et al. 2018b). The 65 mm injector is supported at the flange and extends through a hole on the EH without making contact with it. Random vibration simulation results show that the diameter and length of the injector are such that no additional support is required to avoid contact with the EH during launch. The injector is located at the center of the EH along the Y-axis and oriented 45 deg from the TM +Z face so that the UV light is reflected between the TM and the +Z injection electrode of the EH.

In addition to the protective walls of the vacuum enclosure, a magnetic shield is required to attenuate the magnetic field at the TM (see Fig. 3). The cubic shield fully envelops the sensor with the exception of gaps for the support stiffeners and holes for the caging mechanism grabbing fingers (GF), coaxial cables, UV injector, and shield fasteners. The shield consists of two pieces in a clamshell configuration that fit tightly within the chamber wall, with a total mass of ~207 g that accounts for 2.7% of the S-GRS head total mass. Acceleration noise modeling described in Sect. 4 shows that attenuation of the Earth’s magnetic field by at least a factor of ten is needed to meet performance goals.

### 2.4 Caging mechanism

The caging mechanism must secure the TM against launch loads and gently release it once in orbit. The technical challenge is that upon release, the TM must have a low enough velocity relative to the spacecraft so that the S-GRS control system can electrostatically capture it. To achieve such a low relative velocity, the design must minimize both adhesion between the GF and TM gold surfaces, and any asymmetry in the mechanical preloads just prior to release (Bortoluzzi et al. 2013). The S-GRS caging mechanism reduces the three-stage LISA system to a simpler two-stage system. The first stage uses standard aerospace launch lock actuators to handle launch loads, while the second stage utilizes a commercial vacuum-compatible piezo-walker motor. The first stage applies 200 N to a pair of opposing titanium alloy, hard gold-plated fingers that engage the TM at the conical and pyramidal indents, as seen in Fig. 7.

After the one-time first stage launch lock is released, the second stage retracts the gold alloy fingers until the TM is held with a < 1 N preload by spring-loaded release tips (RT), as seen in Fig. 8. The spring stiffness is designed to ensure that the adhesion between the gold-coated TM and the gold alloy fingers is broken. The final RTs are fabricated from a hard, non-stick silicon nitride material to avoid adhesion with the TM. Damping is added to the system so that the TM motion is minimized upon final retraction and release. To minimize potential electrostatic interactions between the silicon nitride RTs and the TM, the caging mechanism retracts the tips ~10 mm inside the EH.

The simplification to a two-stage system is possible because of the lighter 0.54 kg mass of the S-GRS TM compared to the 1.96 kg LPF TM. Static and random vibration finite element analysis simulations validate the use of only two fingers to support the TM during launch. The static simulations apply a 15 g acceleration to the system in three orthogonal directions. The random vibration simulations use the 14.1 g\text{rms} ASD from the NASA GEVS document (NASA 2021a). The result proves the feasibility of the two-finger approach with a high margin of safety. The 200 N holding
force safely satisfies the low TM displacement requirements, especially for the critical case of random vibration along the $Y$-axis (along the length of the fingers).

In addition to its caging capabilities, the mechanism must also release the TM at a low residual velocity. A maximum allowable residual velocity is obtained from a 2 DOF dynamics simulation that starts immediately after the TM is released and stops when it has been captured by the electrostatic actuation system or collides with the EH. The simulation iterates through a range of initial TM linear and angular velocities, actuation voltages, and directions of motion. The objective is to find the highest initial velocity values that can still be captured by the actuator system. In the model, the TM is treated as a square of four rigidly grouped particles that represent its four corners in the plane of motion. As the square moves because of its initial velocity, the actuator system applies a force that attempts to cancel the TM motion. The electrostatic force is a nonlinear function, dependent on the voltage and distance between the TM and actuator electrodes, and the electrode area (Ciani 2008). Since the S-GRS has different electrode configurations on each face, the electrode areas and locations vary for each plane of motion. The greater the electrode area and the smaller the gap to the TM, the higher the electrostatic force. Assuming a 10 V actuation voltage, the residual linear velocity must be $<4.5 \, \mu\text{m/s}$ and the angular residual velocity $<300 \, \mu\text{rad/s}$.

With a known residual velocity requirement, a second dynamics model is created to simulate the actual retraction and release process. The 1 DOF model is based on that used for the LPF Grabbing Positioning and Release Mechanism (GPRM) (Bortoluzzi et al. 2013) but with the Ball Aerospace spring-tip concept. The system uses the retraction speed of both GFs as an input and simulates the RT and TM dynamics. Assuming all conditions are symmetric between both sections of the caging mechanism, the result is an ideal release where the TM remains centered in the EH. It is only when asymmetries are included in the system, that the TM drifts to one direction. The worst-case scenario for this asymmetry involves having the maximum adhesion force profile on one GF and no adhesion with a retraction delay on the other. The retraction asymmetry causes a kick from the delayed GF, which acts like a compressed spring (Bortoluzzi et al. 2016). The one-sided adhesion force keeps the TM attached to only one GF, pulling the TM as it is retracted. This requires assuming that the gold adhesion force profiles described in the GPRM model apply between the GF and TM. Other than the mechanical spring and damping forces between the GFs and their respective RTs, all bodies in contact are treated as compressed springs to simulate contact forces with the penalty method (Fourment et al. 1999). An additional electrostatic spring-like force acts upon the TM and is a function of the squared sensing injection bias (Ciani 2008). The difference in shape between the PGF and CGF is not included in the model, since their interaction with the TM is treated as a point contact; the labels GF1 and GF2 are used instead, which retractor to the $+Y$ and $-Y$ directions, respectively.

The model’s output contains the location, velocity, and forces of each component along the $Y$-axis (direction of release). These can be used to identify the necessary spring and damping coefficients, retraction velocity, and understand the performance of the current design. The spring-tip concept worst-case simulation uses the 200 N preload, a 50 $\mu$s delay in the retraction of one GF (as used in the GPRM model), and the strongest adhesion force profile on the other GF. For the resulting time series in Fig. 9, a damping ratio of 3, a spring constant of 131 N/m, an initial actuator speed of 50 $\mu$m/s, and an injection signal bias of 4 V is used. In the figure, a time series describes the displacement, speed, and the GF to TM contact forces. Initially, the GFs are preloaded against the TM until retraction starts. Because of the adhesion force, the TM sticks to GF2 as it is retracted (seen in point 1 of the top plot). In point 2, RT1 is pulled back by GF1 as it is constrained by its geometry inside the GF. This decreases RT1’s preload to the TM, allowing RT2 to push the TM and break the adhesion force. With no contact between the GFs and TM, the retraction speed is decreased at point 3 to extend the time in which the RTs dissipate the TM kinetic energy. As seen in the bottom plot, the TM speed amplitude decays while GF retraction continues. At point 4, both RTs are constrained and pulled by their respective GFs until the TM is fully released in point 5.

The adhesion force profile and retraction delay time would be random and uncontrollable inputs in the actual release process, so a Monte Carlo simulation is used to determine the probability of safe release. The adhesion force maxima range from 0.1 to 0.15 N. The elongation or separation between the bonded surfaces in which the maximum force occurs varies from 0.4 to 0.8 $\mu$m. The retraction delay time ranges from 0 to 100 $\mu$s. Using a damping ratio of 3 and assuming all three random inputs are uniformly distributed, results in a mean TM residual velocity of 1.11 $\mu$m/s with a standard deviation of 0.67 $\mu$m/s for 100 points. A histogram of the results can be seen in Fig. 10, where the TM is safely released for all points with a minimum safety factor of 1.7 in the right-most bar. A multiple DOF model is being developed to estimate the TM angular velocity distribution.

### 2.5 Charge management system

The University of Florida team has been developing the LISA CMS for over four years (Kenyon et al. 2021). The CMS utilizes newer UV LEDs that both resolve technical issues related to the Hg-lamp system used by LPF, and reduce the size, weight, and power of this subsystem. Compared to the Hg lamps used in LPF, UV LEDs are smaller, lighter, consume less power, and have a higher dynamic range, with
Fig. 9 Displacement (top) and speed (bottom) time series of the spring-tip concept release simulation results. At point 1, preload from the GFs drops to 0 N as they retract. The RTs follow the TM as the adhesion force pulls it to GF2. RT1 reaches the length it can extend from the GF1 housing at point 2. As each RT reaches its extension limit, the TM is pulled back and forth between them. At point 3, the GF retraction speed input is reduced to extend the time RTs have to dissipate the TM motion. At point 4, both RTs reach their extension limit and move with their respective GF. Point 5 is the moment of TM release; the residual speed can be seen in the bottom plot. Legend description: RT1 = top release tip, RT2 = bottom release tip, GF1 = top grabbing finger, GF2 = bottom grabbing finger, TM = test mass

Fig. 10 Histogram of TM residual speed from a 100 point Monte Carlo analysis

at least an order of magnitude improvement in each performance area (Olatunde et al. 2015). Recent work on the lifetime testing of UV LEDs has demonstrated that it is a viable option for multi-year missions (Hollington et al. 2017). The S-GRS will use the LISA CMS with an advanced pulsed, continuous charge control scheme with a low power (nW) UV light, pulsed in a low duty cycle (ratio between the time a signal is active and its period) that is synchronized to the 100 kHz injection electric field (see Sect. 3) (Inchauspé et al. 2020). Photoelectron flow between the TM and EH will anti-align with the injected field, causing the phase of the UV light pulses with respect to the field to determine the net flow of photoelectrons and the resulting equilibrium charge of the TM. The UV pulse phase is then selected to produce a passively stable TM equilibrium charge of 0 Coulombs.

The CMS unit, shown in Fig. 4, consists of an FPGA-based controller, a power system, the UV LEDs and their current sources, a mechanical enclosure, and a fiber optic harness that delivers the UV light to the S-GRS head. The LISA CMS will reach TRL 6 by the end of 2023. In terms of S-GRS development, the CMS only needs to be tested in integration with the TM and EH because of the difference in geometry to the LISA GRS. In addition to being smaller in size, the S-GRS will only have a single UV light feedthrough (instead of three for LISA) that will be oriented to maximize discharge robustness against variations of the gold surfaces of the TM and EH.

3 Electronics unit

The S-GRS electronics, which comprises both analog and digital elements, will perform the following functions:

- Generate (digitally) the TM injection bias to frequency shift the capacitive readout to the injection frequency (100 kHz)
- Perform the six degree-of-freedom position readout of the TM by differencing opposing pairs of sensing/actuation electrodes via analog electronics, and demodulating the differenced signals at the injection frequency via digital electronics
- Apply actuation voltages via a digital-to-analog converter to the sensing and actuation electrodes at audio frequencies to control the position and orientation of the TM
- Drive the current source for the UV LEDs to discharge the TM
- Drive the caging and positioning mechanism
The S-GRS can be operated in two primary modes: drag compensation and accelerometer mode. In accelerometer mode, all control algorithms are embedded within the S-GRS so that it may be treated as a black box that provides spacecraft acceleration estimates. The TM position and orientation are estimated, and a control algorithm commands the actuation system to keep the TM centered in its housing. The command forces applied to the TM and the residual spacecraft-to-TM motion are used to estimate the non-gravitational spacecraft acceleration. In this mode, no spacecraft propulsion is required.

Drag compensation is a spacecraft-level control loop that commands thrusters using S-GRS data to keep the spacecraft centered on the TM without applying actuation forces to the TM. In this mode, external forces on the spacecraft are directly canceled by a propulsion system, minimizing the applied forces and therefore the force noise on the TM. In addition to improved acceleration noise performance, drag compensation also simplifies the data analysis, since there is no non-gravitational spacecraft acceleration that must be accounted for. For an infinite-gain closed-loop system, there is no non-gravitational spacecraft acceleration in this mode, external forces on the spacecraft centered on the TM without applying actuation forces.

3.1 Capacitive sensing electronics

The TM position in the EH is measured by the capacitive sensing electronics. The basic principle consists of measuring the differential changes between the capacitances on opposite sides of the TM. Similarly, the rotation of the TM can be measured by combining the information of electrode pairs.

One of the limiting noise sources is the intrinsic noise in the capacitive sensing electronics. The main goal is to reach noise levels close to those of the GRACE-FO LRI system, i.e., around 1 nm/Hz$^{1/2}$ in the frequency range of approximately 0.35 mHz to 50 mHz. The proposed capacitive sensor electronics system is shown in Fig. 11. It is based on an AC differential charge amplifier (Lotter et al. 1999), unlike the differential transformer designs used in other accelerometers (Josselin et al. 1999; Speake and Andrews 1997; Weber et al. 2002).

The TM sensing bias is applied on opposite sides of the TM at $f_0 = 100$ kHz using the injection electrodes, which induce an oscillating voltage in the TM, $V_{TM}$, of about one-tenth of the voltage applied to the electrodes (Mance 2012). The two charge amplifiers at opposite faces of the TM sense the capacitances between the TM and the sensing electrodes. The outputs for each arm are (at 100kHz) $V_i = \frac{C_i}{C_f} V_{TM}$ ($i = 1, 2$), where $C_f$ is the feedback capacitor in the charge amplifier that defines the gain and using a parallel plate approximation, $C_i = \varepsilon_0 \frac{A}{d_x}$ where $A$, $d$ and $x$ are the area of the sensing electrode, the gap between the TM and the electrode, and TM motion, respectively. The $\pm$ symbol indicates that the change in the capacitance is antisymmetric: TM motion in one direction causes an increase in $C_1$ and a decrease in $C_2$ and vice versa. The small changes in the TM position are encoded in the differential signal, which is obtained using an instrumentation amplifier (IA). The output of the IA is band-pass filtered with a center frequency of 100kHz and sampled with an analog-to-digital converter (ADC). The amplitude and phase of the 100kHz signal are obtained after I/Q demodulation. The amplitude is proportional to the TM motion,

$$\Delta V \simeq 2\frac{GV_{TM}}{d} \frac{C_0}{C_f} \Delta x$$

where $G = 40$ includes the gain of the IA and of the band-pass filter at 100kHz. $C_0$ is the capacitance between the TM and the electrode for $x = 0$.

The most critical parts of the electronics are the charge amplifiers. For the values given in Fig. 11, the noise levels at 100kHz referred to the input of the IA are 51 nV/Hz$^{1/2}$ and are dominated by the charge amplifier stages (IA, BPF, and ADC noise levels are negligible). The equivalent position sensing noise is readily calculated by considering the position sensitivity

$$\frac{d \Delta V}{G} \simeq 2 \frac{C_0 V_{TM}}{d} \frac{C_f}{C_f}$$

$$\Delta V \simeq 2\frac{GV_{TM}}{d} \frac{C_0}{C_f} \Delta x$$

Figure 11 Capacitive sensing and actuation electronics for one degree-of-freedom. Two charge amplifiers (with $C_f = 10$ pF and $R_f = 1$ MΩ) and an IA are used to measure the differential change in the capacitances $C_1$ and $C_2$ due to TM motion along $x$. A multi-feedback band-pass filter after the IA serves as an anti-aliasing filter and adds extra gain to minimize ADC quantization noise. The components of the design are chosen such that they have space qualified equivalent versions. Actuation signals are applied at $\sim 100$Hz through a DAC and a band-pass filter. $R_d = 1$ MΩ and $C_d = 1$ nF decouple the sensing and actuating signals.
For $C_0 = 3.9 \text{pF}$ (capacitance in the most sensitive axis, $x$ direction), $d = 1 \text{mm}$, $V_{\text{TM}} = 0.5 \text{V}$ and $C_1 = 10 \text{pF}$, it is 390 V/m, which yields a noise equivalent position of $0.13 \text{nm}/\text{Hz}^{1/2}$, i.e., well below the $1 \text{nm}/\text{Hz}^{1/2}$ goal. A similar analysis for the angular readout system results in an angular sensitivity of $\sim 50 \text{nrad}/\text{Hz}^{1/2}$.

Another source of error is due to common-mode rejection (CMR) limitations and, more importantly, their fluctuations at the mHz frequency due to temperature variations. In this case, CMR fluctuations are dominated by the gain mismatch between the two charge amplifiers. The induced error in the TM position is $\delta x = d \alpha C_T \delta T$ where $\alpha C_T$ is the temperature coefficient of $C_T$ and $\delta T$ is the temperature fluctuation. This implies that the temperature coefficient has to be at the ppm/K level for $1 \text{K}/\text{Hz}^{1/2}$ temperature stability.

The experimental results for a single degree-of-freedom electronics prototype are shown in Fig. 12. The upper plot shows the ASD for a 6-hour measurement. The conversion to position noise has been done assuming $V_{\text{TM}} = 0.5 \text{V}$. The noise levels for $f > 20 \text{mHz}$ are $0.5 \text{nm}/\text{Hz}^{1/2}$, i.e., a factor $~4$ higher than the theoretical value, yet below $1 \text{nm}/\text{Hz}^{1/2}$. For $f < 20 \text{mHz}$, the noise levels increase due to temperature fluctuations and exhibit a plateau at $3 \text{nm}/\text{Hz}^{1/2}$ between $0.1 \text{mHz}$ and $3 \text{mHz}$. The plot on the bottom shows the equivalent position changes when injecting temperature changes. The temperature coefficient is $3.8 \text{nm}/\text{K}$ ($3.8 \text{ppm}/\text{K}$ for $d = 1 \text{mm}$). The temperature sensitivity of the charge amplifiers can be further reduced by selecting matching capacitors $C_T$.

### 3.2 Electrostatic actuation

Electrostatic actuation is needed to keep the TM centered in the EH. The performance of such actuation is especially important in non-drag-compensated mode, since the applied electrostatic force is the signal used to derive the non-inertial forces acting on the spacecraft. The electrostatic forces are applied to the sensing electrodes at lower frequencies (50-300 Hz) to avoid interfering with the 100 kHz sensing signal. The circuit for the actuating signals is rather simple—see Fig. 11: a digital-to-analog converter (DAC) with a high-stability voltage reference and an active band-pass filter with gain. However, the requirements are stringent. The force on the TM for a single electrode is $F \simeq \frac{1}{2} C_0 V_{\text{act}}^2$ where $V_{\text{act}}$ is the actuation signal at $\sim 100 \text{Hz}$. In acceleration mode, the electrostatic force on the TM must be equal to the drag on the spacecraft to avoid a collision between them. At an altitude of 500 km, the acceleration due to drag on a GRACE-like spacecraft is about $0.3 \mu \text{m}/\text{s}^2$ (Mehta et al. 2017), which requires an applied voltage of $V_{\text{act}} \simeq 12 \text{V}$ with relative stability requirements at the mHz band and at $2 f_{\text{act}}$ below $10^{-6} \text{Hz}^{1/2}$. At $f_{\text{act}}$, the requirement is about $20 \mu \text{V}/\text{Hz}^{1/2}$. The stability at $f_{\text{act}}$ and $2 f_{\text{act}}$ is not a problem for the actuator circuit. However, the requirement in the mHz band is challenging and, at a minimum, depends on the quality of the voltage reference of the DAC. Ultra-stable voltage references are available at the $10^{-6} \text{Hz}^{1/2}$ level (Halloin et al. 2013). The requirement at lower altitudes, e.g., 350 km is more demanding (Montenbruck and Gill 2000), which means $10^{-12} \text{m/s}^2/\text{Hz}^{1/2}$ noise levels cannot be reached unless a drag-free scheme is considered.

### 4 Acceleration noise performance

A detailed acceleration noise model for the S-GRS has been developed based on two operational scenarios: non-drag-compensated at an orbit altitude of 500 km (similar to GRACE and GRACE-FO), and drag-compensated at an altitude of 350 km. The former and latter cases represent the worst and best realistic operational scenarios in terms of gravity recovery sensitivity. The model accounts for roughly 30 different noise sources that are relevant at acceleration noise levels below $1 \text{pm/s}^2/\text{Hz}^{1/2}$ (Gerardi et al. 2014). This model is based on both torsion pendulum experiments, flight experiments performed on the LPF mission (Armano et al. 2018a) and measurements of the GRACE environment in orbit. The UF torsion pendulum, equipped with a LISA-like GRS, shown in Fig. 4, has a demonstrated force noise performance below $1 \text{pN/Hz}^{1/2}$ at around $1 \text{mHz}$ acting on a lightweight test mass in one degree-of-freedom (Ciani et al. 2017).

Using the developed acceleration noise models and a spacecraft environment equivalent to that of GRACE-FO, Fig. 13a shows the acceleration noise performance of the simplified GRS if operated on a drag-free platform at 350 km. Figure 13b shows the equivalent performance in a GRACE-
Fig. 13 Acceleration noise estimate for the simplified GRS, operated on a drag-compensated platform at 350 km altitude (top) and operated on a GRACE-like spacecraft at 500 km (bottom), based on the measured GRACE-FO flight acceleration environment. The bottom plot shows the GRACE-FO accelerometer noise in a light red curve for comparison. See Eqs. 3 and 4 for details.

Like mission at 500 km with no drag compensation. The GRACE-FO data is from a period of minimum solar activity and therefore represents the most favorable drag environment for acceleration noise performance. The performance of the S-GRS is expected to be roughly 40 times better than the GRACE accelerometers in the same mission configuration. If the sensor were operated on a drag-compensated platform, its performance would improve by an additional factor of 8 to about $6 \times 10^{-13} \text{m/s}^2/\text{Hz}^{1/2}$ around 1 mHz.

The ASD of the acceleration noise, $S_u^{1/2}$, in the drag-free environment can be described by the function of frequency in Eq. 3. For the non-drag-compensated 500 km environment, the equivalent function describing the acceleration noise performance is seen in Eq. 4.

$$S_u^{1/2} = 5 \times 10^{-13} \text{m/s}^2/\text{Hz}^{1/2} \sqrt{1 + \left(\frac{1 \text{ Hz}}{f}\right)^{2/3}}$$

$$S_u^{1/2} = 4 \times 10^{-13} \text{m/s}^2/\text{Hz}^{1/2} \sqrt{1 + \frac{700 \mu\text{Hz}}{f} + \left(\frac{300 \mu\text{Hz}}{f}\right)^2}.$$  

These two ASDs are shown as black bold curves in Fig. 13 and indicate the acceleration noise budget for the S-GRS in the two orbital environments. These curves have been used to evaluate the gravity recovery capability of the S-GRS in Sect. 4 over the frequency band of 0.1 mHz up to 0.1 Hz, the bandwidth of interest for these measurements.

The individual terms for the S-GRS noise model are grouped into six categories: stiffness, magnetic, thermal, Brownian, electrostatic, and actuation (Gerardi et al. 2014). Stiffness describes the coupling between the relative motion of spacecraft and TM and force gradients to produce an acceleration. Magnetic forces arise through the interaction of the bulk material of the TM with the magnetic field at the TM, comprising contributions from the Earth field and spacecraft components. Fluctuating temperatures and thermal gradients in the S-GRS create TM acceleration noise, through residual gas, radiation pressure, and outgassing. Brownian noise is caused by random impacts of the residual gas molecules inside the vacuum chamber on the TM. Electrostatic forces produced by the variation of surface potentials and charge build-up on and around the TM create unwanted accelerations. Finally, noisy forces needed to control the TM position contribute to the total acceleration noise, either directly in the sensitive axis or by cross-talk from orthogonal degrees-of-freedom. A calculation of each of these noise sources is included in Fig. 13 as colored traces, and the best estimate of the total acceleration noise is shown by the light purple curve (top plot). Actuation noise is not included in Fig. 13a because test mass actuation along the sensitive axis of the instrument is not required in a drag-free configuration. In Fig. 13b, the combined thermal, magnetic, Brownian, and electrical effects are identical to the drag-free case and are shown as the light blue curve. The actuation noise and stiffness contributions in this mode are significantly higher because of the need to control the test mass position electrostatically, and are therefore plotted explicitly. Also shown in Fig. 13a,b are the displacement measurement noise curves for the LRI and S-GRS capacitive readout, twice differentiated to convert them to acceleration noise. Note that since the LRI and S-GRS readout noise spectra are quite similar, neither one dominates the overall gravity recovery sensitivity.

4.1 Detailed noise simulations

In the following sections, we summarize the calculations of the major noise sources contributing to the acceleration noise budget of the S-GRS.

Thermal effects Thermal acceleration noise effects in the S-GRS are driven by the fluctuation in temperature gradients.
across the TM. There are three physical sources of thermal gradient-induced noise: the radiometer effect, in which the residual gas in the gaps between the TM and EH has different pressure on opposite sides of the TM. Radiation pressure, caused by the difference in energy of photons emitted by the hot and cold sides of the TM and EH, and the difference in temperature-dependent outgassing rates between hot and cold surfaces. Radiation pressure is well described by analytical and geometrical expressions adapted from those derived for LISA and LPF (Carbone et al. 2007). Both radiometer and outgassing effects depend on the geometry-dependent gas conductances of the EH. Because of this, the effects are simulated with the numerical analysis software MolFlow (Kersevan and Pons 2009). Results from these simulations are used to obtain temperature coupling coefficients for pressure related effects. Using a simplified geometry of the TM and inner EH, the force per unit temperature gradient is determined by applying temperatures to the EH faces and calculating the pressure difference on the TM faces. Assuming water is the dominant outgassing species, an operational temperature of 293 K, and a residual gas pressure of $10^{-5}$ Pa, temperature gradient coupling parameters can be calculated for all three effects. To calculate the expected TM acceleration, noise assumptions must be made about the stability of the temperature gradient across the EH. Data from the GRACE-FO mission, shows that the temperature gradient stability at the accelerometer instrument is better than the GRACE-FO mission, shows that the temperature gradient is around $10^{-5}$ Pa and a residual gas pressure of $10^{-5}$ Pa, temperature gradient coupling parameters can be calculated for all three effects. To calculate the expected TM acceleration, noise assumptions must be made about the stability of the temperature gradient across the EH. Data from the GRACE-FO mission, shows that the temperature gradient stability at the accelerometer instrument is better than 5 mK/Hz$^{1/2}$ at 1 mHz. The S-GRS design will be designed to meet this thermal stability requirement. Assuming a performance at this level gives an upper limit of 0.05 pm/s$^2$/Hz$^{1/2}$ for thermal effects at 1 mHz. Based on GRACE-FO data, a frequency dependence of $1/f^{1/2}$ is assumed for the stability of the temperature gradient.

Gravitational effects The mass distribution on-board the spacecraft will produce a residual gravitational force and force gradient at the TM. The resulting acceleration of the TM is likely to be small compared to the drag on the spacecraft, so that in non-drag-free mode the acceleration is a small fraction of the actuation force. In a drag-compensated mode of operation, however, a continuous acceleration of the TM along the line of sight between spacecraft may eventually affect the inter-spacecraft laser metrology and require corrective maneuvers. This constraint will enforce some level of gravitational balancing on the spacecraft system. The LISA Pathfinder mission demonstrated the ability to compensate TM accelerations at the sub-pm/s$^2$ level (Armano et al. 2016) but S-GRS performance likely does not require such fidelity.

In order to estimate the residual gravitational force on the TM, a mapping of components from a geometrical model of the GRACE-FO spacecraft was analyzed to estimate the volume and center of mass of each component, the S-GRS CAD design was also broken down into 127 mass elements.

The gravitational force on the TM due to each component and S-GRS mass element was calculated, resulting in a net residual dc acceleration on the order of $30 \text{nm/s}^2$. This level of acceleration would result in an accumulated length change in the inter-spacecraft distance of around 10 km in 10 days, but could be mitigated by better gravitational balancing, station-keeping, or compensation by electrostatic actuation of the TM. The gravitational force also produces a force gradient at the TM of 13.5 nN/m in the initial estimate. This is an important contribution to the total stiffness of the TM. In-band fluctuations of the gravitational force due to thermal distortion of the spacecraft were calculated and found to be negligible. This preliminary calculation provides only an order of magnitude estimation of the gravitational effects, given that a flight platform for the S-GRS has not been defined.

Magnetic effects The magnetic force $F_{B,x}$ on the TM along the $x$-axis can be written as

\[ F_{B,x} = \left( \frac{\chi_{TM} L_{TM}^3}{\mu_0} B + M \right) \cdot \frac{\partial B}{\partial x}, \]  

where $M$ is the remnant magnetization of the TM, $\chi_{TM}$ is the magnetic susceptibility, $L_{TM}$ is the length of the TM side, and $B$ is the magnetic field at the TM. The remnant magnetization $M$ of the gold platinum TM is expected to be around $10^{-6}$ A m$^2$, while the susceptibility $\chi$ is around $10^{-5}$. The magnetic field at the TM originates from components on-board the spacecraft, and from Earth. The field gradient is dominated by the on-board component. The time-varying field at the spacecraft due to its motion through the Earth field, coupled with the static local field gradient, gives rise to a magnetic force signal at harmonics of the orbital frequency. Stochastic noise in $F_{B,x}$ arises from the coupling between the noise in the field gradient due to spacecraft equipment and the Earth field.

In order to estimate the magnitude of the magnetic effects, a magnetic map of the spacecraft has been created, similar to the gravitational analysis described above. Each spacecraft component is represented as a magnetic dipole at a position determined from a spacecraft geometrical model. The magnitude of the dipole is assigned to each element of the spacecraft equipment based on published measurements made on similar components of the LPF mission (Martin 2015). A simulation was developed, assigning each dipole a uniformly distributed random direction and calculating the field and gradient at the TM. 10,000 field realizations were calculated, which resulted in a local field magnitude of $\pm 0.5 \mu T$ and a gradient of $\pm 2.7 \mu T/m (1 - \sigma)$. This is likely to be an optimistic estimate based on the simplicity of the geometry used. Given that the LISA Pathfinder gradient requirement was $5 \mu T/m$ with stringent magnetic cleanliness.
requirements, we assume a worst-case gradient of 500 \text{\mu T/m} in our acceleration noise calculations. The Earth field is measured on-board by GRACE-FO. The along-track component of the field has a peak-to-peak amplitude of around 55\text{\mu T}. Combining these assumptions in Eq. (5) results in the dashed red curve in Fig. 13a which dominates the acceleration noise budget. A magnetic shield is designed as part of the S-GRS head to reduce the effects of these noise sources.

Initial estimates using a realistic geometry simulated in the COMSOL simulation toolkit (version 5.6 of a finite element analysis and multiphysics simulation software) indicate the magnetic shield will reduce the field and gradient at the TM by up to a factor of 100 leading to an acceleration noise contribution represented by the red curve in Fig. 13a, suppressed by a factor of $10^4$ compared to the unshielded case. Incorporating a magnetic shield relaxes significantly the magnetic cleanliness requirements compared to LISA Pathfinder, making it easier to accommodate the S-GRS as a secondary payload or on a spacecraft requiring an electrical thruster for drag compensation. Additional details on the magnetic shield can be found at the end of Sect. 2.3.

**Other acceleration noise sources** Electrostatic interactions between the charged TM and electric fields in the EH result in forces on the TM. Both the stray electric field and the charge build-up on the TM are noisy, resulting in an acceleration noise. The effect has been well studied in the context of the LISA mission (Sumner et al. 2020). Applying geometric scaling factors to those model predictions to account for the reduction in size of the S-GRS TM and gaps, and assuming a well-performing CMS, we expect a noise of around $20\text{fm/s}^2/\text{Hz}^{1/2}$. This level allows continuous TM charge control using photoemission from UV light. In this case, the acceleration noise is dominated by the discharging current shot noise, with a $1/f$ shape and a magnitude that is expected to be similar to that expected for LISA. Assuming a worst case of uncompensated stray electric fields in the S-GRS leads to the dark green curve of Fig. 13a.

Electrostatic actuation forces applied to control the TM will introduce force noise through the imperfect stability of the voltages applied to actuation electrodes in a non-drag-compensated mode of operation. The expected relative voltage stability for actuation voltages is around $5 \times 10^{-6}/\text{Hz}^{1/2}$ resulting in a force noise with a spectral density at a level of around $10^{-5}$ times the applied actuation force. The magnitude of this effect is calculated from the control and actuation simulations described in Sect. 5 below. In a non-drag-compensated operational mode, this is a limiting contributor to the S-GRS acceleration noise performance, shown by the dark red line in Fig. 13b.

As well as force noise, the electrostatic force needed to control the test mass position creates a significant force gradient upon the TM. From the electrostatic control simulations, the commanded voltages on the electrodes produce a force gradient of $3 \times 10^{-4}\text{N/m}$, while for a drag-compensated mode, the effect is reduced to around $10^{-6}\text{N/m}$, contributed by the gravitational forces and the electrostatic sensing bias of the TM. Coupling these gradient values with the relative motion of the TM and S/C shown in Fig. 15 produces the light green traces of Fig. 13a, b.

It is noted that in missions like GRACE, an additional noise source is found in the form of thermally induced “twangs” caused by the foil in the bottom of the satellite. These could mostly be observed in the radial component of the accelerometer data (Petersen 2014). The presented noise simulations are based on a GRACE-FO environment, where changes are made in the design to avoid these features (Landser et al. 2020). Because of this, the twangs are not included in the current simulations. However, other transient disturbances may be included in further simulation work.

### 4.2 Mission architecture considerations

Like previous Earth geodesy missions (and space gravitational wave interferometers), the proposed concept that includes the S-GRS, LRI, GPS receiver, and spacecraft forms a single “instrument.” The performance of this instrument is directly tied to its attitude and possible drag-compensation control performance; as well as the thermal, electromagnetic, and gravitational environment provided for the S-GRS and LRI. Therefore, to maximize scientific performance, both the instrument and spacecraft platform must be considered. A spacecraft environment equivalent to that of GRACE-FO is assumed for the performance model described in the previous section. The S-GRS provides significant improvement when placed in the GRACE environment. The mitigation steps taken in the S-GRS design include a small mu-metal magnetic shield surrounding the EH and the vacuum enclosure (see Fig. 3). Including these measures, the key environmental parameters needed to achieve the stated performance are listed in Table 1.

The performance of the instrument in terms of gravity recovery accuracy is also closely tied to orbit selection and the number of orbits (satellite pairs). The number of pairs of satellites that is financially viable depends on the launch mass of each spacecraft and therefore the size, weight, and power of the S-GRS and LRI. Finally, mission lifetime and orbit altitude also depend on whether a drag-compensated platform is chosen.

The estimated mass of the S-GRS Head and electronics unit is $\sim 8\text{ kg}$ and $\sim 5\text{ kg}$, respectively. The power consumption of the S-GRS electronics unit is $\sim 20\text{ W}$. The total mass of the GRACE-FO LRI units per spacecraft is $25\text{ kg}$, and the nominal power consumption is $35\text{ W}$. The sum of these results is a mass of $\sim 38\text{ kg}$ and power consumption of $\sim 55\text{ W}$. These values already put the mass and power budget of the S-GRS and LRI within levels typically accommodated on ESPA-
class microsatellites (Evolved Expendable Launch Vehicle Secondary Payload Adapter (Haskett et al. 1999)), whose size and mass are 1/3 of the GRACE-FO spacecraft. However, it is noted that implementing the sensor within this architecture would still be limited by other system requirements (NASA 2021b).

5 S-GRS and drag-compensation control

In both drag-compensated and non-drag-compensated modes of operation, the TM position must be controlled, either by actuation of the spacecraft with micro-Newton thrusters or by electrostatic actuation of the TM. The residual jitter of the TM position relative to the spacecraft drives the stiffness contribution to the acceleration noise budget and, in the non-drag-compensated case, the noisy TM actuation force is an important noise contribution. We have developed a control system for both modes of operation, and here we show the performance in the x (SC-SC) direction.

The performance is evaluated through a numerical simulation that uses realistic sensing noise, actuation noise, and spacecraft environment. The control of the spacecraft is simulated in both the non-drag-compensated and drag-compensated case. In both cases, a state space representation of a simple equation of motion for the linear satellite dynamics is used.

\[ \ddot{x} = a_{SC} + a_{TM} + \frac{K}{m_{TM}} x \]  

(6)

where \(a_{SC}\) is the environmental disturbance on the spacecraft, \(a_{TM}\) is the direct TM force noise, \(K\) is the stiffness, \(m_{TM}\) is the mass of the TM and \(x\) is the relative position of the TM with respect to the S/C. The position sensor noise and applied actuation forces are accounted for when applying this equation in the simulation.

The controls simulation is done using MATLAB (version R2019b) and Simulink to simulate around one day of data. For the non-drag-compensated case at 500 km altitude, the spacecraft acceleration noise is taken from previously recorded GRACE-FO Level-1B accelerometer data along the x direction, which measures the relative acceleration between the test mass and spacecraft, and we assume this measurement is dominated by the spacecraft acceleration signal. At 350 km we assume the along-track disturbance is 10 times greater. The position sensor noise in both cases is modeled as a Gaussian white noise with a level of 1 nm/Hz\(^{1/2}\) while the direct forces on the test mass, excluding actuation, are approximated by a white noise of 0.2 pm/s\(^2\)/Hz\(^{1/2}\). The latter value is the upper limit for all forces on the TM at 0.1 mHz. A spring constant of 1 \(\mu\)N/m, is used for the drag-compensated case in which electrostatic stiffness from TM actuation forces will be small, in a non-drag-compensated case where large voltages are applied to sensor electrodes, a value of 310 \(\mu\)N/m is used. This figure was calculated from the commanded voltages output from the simulation described below, iterating from an initial value of 100 \(\mu\)N/m for the stiffness.

The feedback control block diagram is shown in Fig. 14 indicating the modeled disturbances, sensing and actuation noises. In order to reduce stiffness contributions to the acceleration noise budget, the aim of the controller in both control schemes is to reduce the in-band motion of the test mass to below 50 nm/Hz\(^{1/2}\) for drag-compensated and below 2 \(\mu\)m/Hz\(^{1/2}\) for non-drag-compensated. In both cases, an H-infinity optimization method is employed to determine the control effort needed to stabilize the motion. This method allows for the disturbances and noise to be accounted for in the control algorithm by using a Multi-Input Multi-Output system and indicating which error outputs are to be kept small. Using the function \texttt{hinfssyn} in MATLAB, we minimized the position readout and velocity jitter.

The thruster model for the drag-compensated simulation is based on cold-gas thrusters used in the Gaia, Microscope, and LISA Pathfinder missions (Armano et al. 2019). Our model is capable of simulating 6-DOF S/C control, although here we present only the results for the critical x direction. Similar to the GRACE thruster system, four thrusters actuate each degree-of-freedom, \((x, y, z)\), linear thrust is achieved by actuating thrusters on the same face of the spacecraft, torques are obtained commanding thrust to opposite pairs. The thrusters are positioned so that translational thrust produces no torque and torque applied to the spacecraft produces no net force. Once the command to each individual thruster has been calculated, the constraints of a cold-gas thruster model are applied. A resolution of 0.1 \(\mu\)N and range of 2 to 1000 \(\mu\)N is used, with a time response of 250 ms to 63% of the commanded thrust level. A thrust noise of 0.17 \(\mu\)N/Hz\(^{1/2}\) is also added in this step. The maximum thrust used here is extended beyond the capability of the system included on
LISA Pathfinder. It remains to be investigated whether there are technical barriers to achieving this thrust level, however, a number of other micro-thruster technologies are under development which may address the required thrust range and resolution.

For the non-drag-compensated case, the S-GRS electrodes are utilized as actuators on the test mass. Commanded forces and torques are converted to electrode voltages using a simple electrostatic model. Assuming a maximum applied voltage of 15 V with 16 bits of resolution, the maximum force that can be applied to the TM in our model is about 0.4 µN with resolution of 0.1 fN. The ASD of the voltage noise is calculated as 5 ppm of the rms commanded voltage on each electrode. This noise is added to the commanded force before calculating the resulting force on the test mass.

The results of the x-axis position simulation for drag-compensated and non-drag-compensated cases are shown in Fig. 15. The drag-compensated simulation results show that the larger input noise from the 350 km orbit is greater than the sensor noise limit. The control loop gain is insufficient to reduce this, but the jitter requirement of 50 nm/Hz$^{1/2}$ is met. The non-drag-compensated simulation shows that the in-loop position measurement is at a level below the measurement noise, successfully suppressing stiffness contributions to the acceleration noise budget. The controller performs optimally so that the sensor noise level is the limiting factor and there is no wasted control effort. They both make a negligible contribution to the acceleration noise budget.

By further analyzing the non-drag-compensated case, we can identify the sources limiting the sensitivity of the S-GRS from the simulated data. In this mode of operation, the spacecraft acceleration would be determined from the S-GRS measurements and the applied control forces. Figure 16 shows the ASD of various sources of apparent TM acceleration in the simulation. The black curve shows the input S/C acceleration to the simulation. After subtracting the commanded control forces (orange curve), the residual acceleration is given by the dark red curve. This represents the performance limit of the S-GRS at $3 \times 10^{-12}$ m/s$^2$/Hz$^{1/2}$ and is limited by the actuation noise (light blue curve) and position sensor noise (dark blue curve). Also shown are the sub-dominant contributions of stiffness (yellow), direct TM forces (green), and in-loop TM motion estimate (purple).

### 6 Conclusions and outlook

We have developed a novel simplified gravitational reference sensor optimized for future geodesy missions. This sensor is a scaled-down version of the flight LPF GRS with reduced complexity. As discussed in Sect. 4, the performance is roughly 40 times better than that of the accelerometers flown on GRACE and GRACE-FO when operated on a non-drag-compensated spacecraft in a 500 km altitude Earth...
Table 2  Force models used in numerical simulations

| Measurement                        | Truth model          | Nominal model          |
|-----------------------------------|----------------------|------------------------|
| Static gravity field              | gif48                | gif48 (Ries et al., 2011) |
| Ocean tides                       | GOT4.8 (Ray, 1999)   | FES2004 (Lyard et al., 2006) |
| Nontidal atmosphere and ocean (AOD) | AOD RL05 (Flechtner et al., 2014) | AOerr + DEAL (Dobslaw et al., 2016) |
| Hydrology + ice                   | ESA earth system model (Dobslaw et al., 2015) | N/A |

Table 3  TRL of S-GRS Components

| Component        | Current TRL | TRL goal      |
|------------------|-------------|---------------|
| S-GRS head       | 3           | 5–6 (mid-2020s) |
| Caging mechanism | 3           | 5–6 (mid-2020s) |
| CMS              | 5           | 6 (2023)      |

orbit. When operated on a drag-compensated spacecraft, the performance improvement is a factor of 500. This is achieved by eliminating the small grounding wire used in prior accelerometers and instead using a UV photoemission-based TM charge control scheme. In addition to eliminating the noise associated with the grounding wire, this also allows for a larger TM and a larger TM-to-electrode housing gap.

We described the mechanical and electrical design of the S-GRS and provided details of our performance assessment, which is based on models validated by LPF and torsion pendulum experiments, and used flight environment data from GRACE-FO. Gravity recovery simulations reveal that the S-GRS noise and the GRACE-FO LRI noise contribute equally to the gravity recovery sensitivity of a mission that utilizes both. Therefore, neither the S-GRS nor the LRI would limit measurement sensitivity.

The next steps in the S-GRS development consist in manufacturing a prototype-like testbed of the S-GRS head. The EH and other components will be placed on a hexapod while the TM will be fixed to a static frame. This will allow us to test the capacitive sensing within the limits of a 1 g environment. The torsion pendulum facility at UF allows for a test of the S-GRS actuation electronics with scaled-up versions of the S-GRS EH and TM. The sensitivity of the pendulum is sufficient to demonstrate the required performance level in one degree-of-freedom. Other testbeds are already in use to characterize the components used in the caging mechanism assembly. Ball Aerospace will continue the development of the caging mechanism design and contribute in the structural and thermal testing of the S-GRS head.

A team led by the University of Florida, in collaboration with Caltech/JPL and Ball Aerospace, is advancing the technology readiness level of the S-GRS. Currently considered TRL 3-5, depending on the component, we plan to reach TRL 5-6 in three years and flight readiness by the mid-to-late 2020s.

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Data Availability  The datasets generated and analyzed during the current study are available from the corresponding author on reasonable request.

Appendix A: Models for performance estimates

High-fidelity numerical simulations relying on the same software suite used to process GRACE and GRACE-FO data at JPL were run to quantify performance in Figs. 1 and 2. The simulations consist of a truth run to create a set of synthetic satellite observations based on a realistic flight environment, followed by a nominal run where the truth measurements are perturbed in some way. The error in the recovered gravity field due to this perturbation is then quantified via a large linear least squares estimation process, as is commonly used in the GRACE and GRACE-FO data processing. Force models used in the truth and nominal runs are given in Table 2 when temporal aliasing error is included in the simulation. When only measurement system error is considered, the nominal models in Table 2 are set equivalent to the truth models, so there is no perturbation among the model. The simulation timeframe is January 1–29, 2006. The gravity estimation process is a 2-step process, where in the first step, a set of “local” parameters are estimated using the tracking data to converge to the best fitting orbit. These parameters consist of daily position and velocity of each spacecraft, daily accelerome-
ter scale factors and biases, and range-rate biases, drifts, and one cycle per revolution each orbital revolution. In the second step of the gravity estimation process, these same parameters are again adjusted along with a 29-day mean gravity field expressed to spherical harmonic degree and order 180. A diagonal weighting matrix is used for the tracking observations (LRI range-rate and kinematic orbits); and relative weights between the two data types are estimated in an optimal fashion as described in Section 3.4 of Watkins et al. (2015).

Measurement system errors ingested into the simulation process rely largely on heritage information from GRACE-FO. Orbit error is introduced by adding white noise with a 1 cm standard deviation in all 3-axes with a 5-min sampling time. Attitude Error is derived using the difference of two competing data products used to define the GRACE-FO attitude. For pitch and yaw, the difference between an attitude solution that combines star tracker and IMU data (as in the GRACE-FO v04 SCA1B data product) with star tracker and laser steering mirror data (from the LRI) is used to define the error (Goswami et al. 2021). Since the laser steering mirror is insensitive to roll variations, roll error is defined as the difference between an attitude solution that uses only star tracker data with one that uses both star tracker and IMU data. LRI error is derived from GRACE-FO flight data (Abich et al. 2019), where laser frequency noise dominates high frequencies and tilt-to-length coupling error dominates lower frequencies (Wegener et al. 2020). The LRI error curves with “improved alignment” assume the tilt-to-length coupling error is driven to zero via improved alignments relative to the center of mass of the spacecraft. The GRACE-FO accelerometer error is taken as the best estimate of performance prior to launch of GRACE-FO and is approximately a factor 3 better than the requirement discussed in Kornfeld et al. (2019). The S-GRS error is described in detail in Sect. 4. Satellites have a separation distance of 300 km.

Many previous studies have compared single and dual-pair architectures for recovering time variations in the Earth’s gravity field. While differences in simulation setups across studies make it difficult to compare results at a granular level, results can be compared on a macro level. We note that the simulation results we show (relative improvement of dual-pair over single pair, and relative contribution of different measurement system components to the overall error budget) agree with previously published studies at the macro scale (Wiese et al. 2011b; Purkhauser et al. 2020; Hauk and Wiese 2020).

**Appendix B: Technology readiness level**

Projects are classified depending on the level of maturity of their technology through a system called the Technology

| Table 4 | Table of abbreviations |
|---------|------------------------|
| **Abbreviation** | **Meaning** |
| AC | Alternating current |
| ADC | Analog-to-digital converter |
| ASD | Amplitude spectral density |
| BPF | Band-pass filter |
| CAD | Computer aided design |
| CF | ConFlat (type of flange) |
| CGF | Conical grabbing finger |
| CHAMP | Challenging minisatellite payload |
| CM | Caging mechanism |
| CMS | Charge management system |
| CMR | Common mode rejection |
| DAC | Digital-to-analog converter |
| DOF | Degrees-of-freedom |
| EELV | Evolved expendable launch vehicle |
| EH | Electrode housing |
| ESPA | EELV secondary payload adapter |
| EWH | Equivalent water height |
| FPGA | Field-programmable gate array |
| GEVS | General environmental verification standard |
| GF | Grabbing finger |
| GOCE | Gravity field and steady-state ocean circulation explorer |
| GPS | Global positioning system |
| GPRM | Grabbing, position, and release mechanism |
| GRACE | Gravity recovery and climate experiment |
| GRACE-FO | GRACE Follow On |
| GRS | Gravitational reference sensor |
| IA | Instrumentation amplifier |
| IMU | Inertial measurement unit |
| ISUK | Inertial sensor UV kit |
| I/Q | In-phase and quadrature |
| LISA | Laser interferometer space antenna |
| LPF | LISA Pathfinder |
| LRI | Laser ranging interferometer |
| MC | Mass change |
| PEEK | Polyether Ether ketone |
| PGF | Pyramidal grabbing finger |
| RT | Release tip |
| SC or S/C | Spacecraft |
| S-GRS | Simplified GRS |
| SMA | Sub-miniature version A |
| TM | Test mass |
| TMA | Triple mirror assembly |
| TRL | Technology readiness level |
| TVAC | Thermal vacuum chamber |
| UF | University of Florida |
| UV LED | Ultraviolet light-emitting diode |
Readiness Level (TRL) (NASA 2010). The system uses a numeric scale that starts at 1 for “reporting and observation of basic principles” and ends at 9 for flight-proven technology. For a technology to be included into a new mission, it must progress through the TRL scale until it is flight ready (TRL 8). Table 3 shows the current and goal TRL values for the S-GRS head, the caging mechanism, and CMS.

Appendix C: Abbreviations

See Table 4.

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