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Design of the GG satellite

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Abstract

The design of the satellite for the GG experiment was addressed in phase-A level studies in 1997–2000, based on an equatorial orbit, and more recently re-addressed for sun-synchronous orbit (SSO). The mission consists of an experiment running uninterrupted with few operational modes, small telemetry rates, easily controlled by one ground station. The satellite is small, low-weight, with low power demand. The configuration, resembling a spinning top, is very compact and stiff. The main requirements are for thermal stability, drag-free control and spin rate control. The reconfiguration to SSO makes the mission suitable for a low-cost launch, and improves the thermal performance.

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1. Introduction

GG is a small scientific satellite mission with the objective of testing the equivalence principle (EP) to 1 part in 10^{17}, four orders of magnitude better than the best experiments to date. The experiment [1,2] consists in testing the universality of free fall by means of two concentric hollow cylinders of different composition (the test masses) in orbit in the gravitational field of the Earth. Two additional cylinders, also concentric, made from the same materials, provide the zero-check. The cylinders spin around their common axis, perpendicular to the orbit plane, at a rate (2 Hz) much larger than the orbit rate. Weak suspensions constrain the motion to the plane perpendicular to the spin axis. Suitable sensors (capacitance plates) located between the test masses detect the relative displacements. A violation of EP would manifest itself as a displacement of the order of 1 pm, constant in the orbit frame and directed toward the center of the Earth.

The test bodies, their mechanical coupling and the capacitance read-out are the core of the mission. Once the experimental design has been conceived, the required features of the spacecraft and its attitude and orbit follow. The spacecraft must provide a suitable accommodation to the experiment, provide specific and stable mass properties, shield it from thermal and dynamic perturbations within specified limits. The spin axis must be nearly perpendicular to the orbit plane, gyroscopic stiffness providing stability against external attitude disturbances. The orbit must be near circular, the altitude and inclination being selected as the best compromise between experiment needs and technical and cost considerations.

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The GG mission in equatorial orbit was studied at phase A level [1] as a candidate for the Small Mission Program of the Italian Space Agency (ASI). More recently, a supplementary study addressed the implementation of the mission in sun-synchronous orbit. A prototype of the GG payload was built to demonstrate the validity of the physical principles on which the proposed space experiment relies. The ground experiment is operational in the laboratories of LABEN in Florence. A second-generation, more sensitive prototype is under testing [2].

2. Orbit design

Since the intensity of the putative signal depends on the inverse-square of the distance from the center of the Earth, the orbit altitude should be as low as possible, and the orbit should be near-circular. A lower limit to the altitude is set by the need to limit the disturbing effects from the residual air density. The criterion adopted is that of an altitude at which the air drag is of comparable magnitude to the solar radiation pressure (that does not depend on height). The mean altitude adopted is 520 km and the period is 95 min. At 520 km, the magnitude of the perturbing forces (with the given spacecraft cross section) is less than 50 μN. This value also sets the requirements of the actuators dedicated to balancing the perturbations (drag-free control micro-thrusters). A residual orbit eccentricity will be induced by both natural perturbations due to the non-spherical potential of the Earth and launcher injection errors. The expected upper limit of the eccentricity, of some parts in 10^3, is tolerable by the experiment. As is well known, the flattening of the Earth J_2 induces a secular precession of the line of nodes of an orbit of radius r and inclination i at the rate

$$\dot{\Omega} \approx -\frac{3}{2}\left(\frac{R}{r}\right)^2 J_2 n \cos(i),$$

where R is the radius of the Earth and n the satellite mean motion. Hence for any inclination different from 0 or $\pi/2$, the spin axis of the satellite, initially parallel to the normal to the orbit plane, will move away from it up to a maximum separation $2i$ reached after a time $\pi/\dot{\Omega}$. For this reason, GG was originally conceived for an equatorial orbit [1]. In this orbit, the attitude of the spin axis perpendicular to the orbit plane would be naturally maintained. However, low-cost launch opportunities for equatorial orbits are not available. Sun-synchronous orbit (SSO) is far more frequently visited and currently the cheapest option. Hence the GG mission was redesigned for SSO. In SSO, the inclination and radius are chosen such that $\dot{\Omega}$ equals the apparent rotation rate of the sun in an inertial frame centered in the Earth. Thus the orbit plane maintains its relative position with respect to the sun. A special case is the so-called dawn-dusk orbit, where the initial conditions are set such that the projection of the sun vector on the equator makes an angle of $\pi/2$ to the line of nodes, i.e., the orbit rides the terminator. The sun-synchronous inclination at $r = R + 520$ km is 97.5°.

The effect of the inertially fixed attitude of the spin axis is that the 'node' of the spin axis (intersection of the plane perpendicular to the spin axis with the equatorial plane) lags behind the orbit node. Thus the spin axis will move away from its nominal position, aligned with the orbit normal, at a rate of about $1°$/day. This misalignment leads to a growing component of the EP signal out of the equatorial plane of the satellite where the sensors are located. Hence it must be corrected. The misalignment will be corrected every 20 days (1 measurement interval) by a dedicated spin axis turn maneuver executed by the on-board thrusters. The average amplitude of the re-orientation maneuvers is 18.5° and they are executed at intervals selected in such a way that the maximum misalignment with respect to the ideal orientation does not exceed 10° (Fig. 1). The nominal launch time for dawn-dusk SSO is 6 a.m. or 6 p.m., local solar time (LST) at the launch site. The typical daily

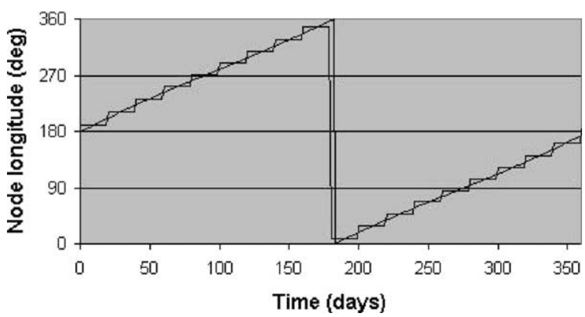

Fig. 1. Evolution of longitude of ascending node (continuous line) and spin axis node (step function). Dawn-dusk orbit, initial epoch winter solstice.

launch window for such orbits is $\pm 1/2$ hour. The launch can occur at any epoch in the year. The orbit has one eclipse season per year, occurring (a) around the winter solstice if $\Omega = \alpha_{\text{sun}} - 1/2\pi$ (ascending node at dawn), or (b) around the summer solstice if $\Omega = \alpha_{\text{sun}} + 1/2\pi$ (ascending node at dusk). In both cases, the peak duration of eclipse is about 23 min. In principle, condition (a) can be realized by launching due North at 6 a.m. or launching due South at 6 p.m., and condition (b) the other way round. Because of the geographical constraints of the launch site, only the South-bound launch is allowed. Therefore there is one launch opportunity per day, at 6 p.m. LST for the orbit of type (a), and at 6 a.m. LST for the orbit of type (b). Depending on the launch epoch, the type of orbit can be selected so as to place the eclipse season as judged more convenient for the early orbit operations.

An important advantage of SSO is a much more stable thermal environment. In the equatorial orbit, the satellite enters the Earth's shadow once per orbit, all year round. This is a source of severe thermal variations at orbit frequency, which affect the experiment and must be damped by design. The SSO instead is eclipse free for about 265 days in a year, and the solar array can be used to protect the spacecraft body from direct illumination by the sun, easing the task of the thermal control. At the selected altitude, the eclipse in SSO (23 min) is significantly shorter than in equatorial orbit (36 min). Thus, a smaller battery can be employed. The size of the solar array used as sun shade is determined by the solar aspect angle (SAA) of the spin axis. The maximum SAA occurs around a solstice, winter solstice in an orbit of type (a) or summer solstice in type (b). As a consequence of the non-ideal orientation, this maximum increases slightly, from 31° to 32.5°. The maximum SAA is coincident with the eclipse season. Therefore this is the worst case for thermal variations (temperature excursion due to shadow + maximum incidence of the sun). Outside of the eclipse season, the maximum SAA never exceeds 23°.

Atmospheric drag will cause a secular decay of the semi-major axis of the GG orbit, eventually leading to re-entry. The semi-major axis drop will have an effect on the eclipse pattern and the ground contact times. However, for the 2-year nominal life of the mission these effects are negligible. The next minimum of the solar cycle occurs in mid-2007 and through the period June 2005–June 2008 the solar activity index $F_{10.7}$ remains below 100. This contributes to making the orbit decay particularly small. With an area-to-mass ratio of 0.005 m^2/kg re-entry occurs 7.6 years after launch. The mean altitude drop is 8 km after 1 year, 18 km after 2 years and 29 km after 3 years. After 3 years, the eclipse season is 3 days longer and the maximum duration of eclipse per orbit is half a minute longer.

The mission design assumes a single ground station. The data flow with ground was examined with reference to the near-equatorial ground station of San Marco (Malindi, Kenya). Contacts occur in a regular pattern, on two successive orbits, after which a gap of 6 orbits (10 hours) follows, and then the pattern repeats. Occasionally, one of the two successive contacts is missing. The average duration of a contact is 9.4 minutes, and the sum of two successive contact times is nearly constant, around 19 minutes. The daily average is 35 minutes.

During the normal science mode, the data to be transmitted to ground are accumulated at a constant rate of about 8 kbit/s. This rate is dominated by the housekeeping data (including the record of thruster and damper commands). The accumulated data in 1 day are about 700 Mbit, and, with 35 min/day available for the downlink, the minimum required telemetry data rate is less than 400 kbit/s, easily within the capability of a standard S-band station. The mass memory is sized by the maximum duration of a contact gap, about 12 hours, which leads to about 350 Mbit. Hence the required mass memory is about 1.5 times the size of that designed for the equatorial orbit.

The reference launch vehicle is Dnepr. Dnepr can inject directly the satellite into the required circular, 520-km altitude, sun-synchronous orbit. The payload capability is close to 1500 kg. Given the low mass and small size of GG, a double or even triple launch can be envisaged. The typical injection errors have no impact on the GG mission. The flight reliability, based on 159 launches, is 97%.

3. Spacecraft configuration and thermal design

The cylindrical symmetry of test masses and their enclosure (Pico-Gravity Box, PGB), and the spin re-

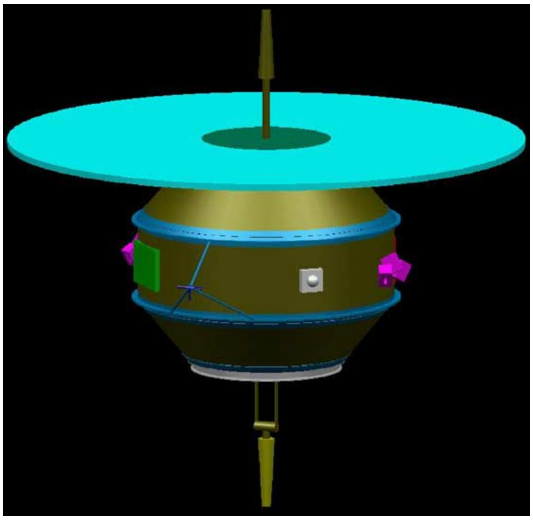

Fig. 2. 3D view of the GG satellite (thermal covers not shown). The configuration is symmetric about the longitudinal axis (Z-axis, $+Z$ direction in the hemisphere containing the sun). In orbit, the Z-axis is always maintained within 10° of the normal to the orbit plane. The 2-m diameter solar panel provides electrical power to the satellite and shields it from direct illumination from the sun at all permitted solar aspect angles. Two deployable S-band antennas on the $\pm Z$ sides provide for telecommunications. Two clusters of FEEP thrusters are used for in-plane and out-of-plane drag-free control. Two more clusters of cold-gas thrusters, mounted on dedicated triangular arm structures, provide for initial spin up and periodic reorientation of the spin axis. The cold gas is drawn from two toroidal tanks equally distanced from the center of mass. The satellite equipment boxes are mounted to the central cylinder and to the upper and lower conical caps.

quired to provide high frequency signal modulation lead to a spacecraft of cylindrical symmetry too, stabilized by one-axis rotation about the symmetry axis. Fig. 2 shows a 3D view of the satellite. The spacecraft body is 1 m wide and 0.8 m high. The experimental apparatus is accommodated in a nested arrangement inside the body, as shown in outline in the cross section of Fig. 3. The structure is made up of a central cylinder and an upper and lower truncated cone. The upper cone is removable to allow the integration of the PGB with its suspension springs; the lower cone supports the launcher interface ring. The equipment is mounted to the central cylinder and to the upper and lower cones. The solar cells are mounted to a 2 m-diameter circular platform on top of the satellite, which also performs the function of a sunshade.

This spacecraft structure, similar to a spinning top, is exceptionally compact and stiff. Carbon fiber is used to minimize thermal distortions. Total structural mass is 57 kg including 20% margin. The calculated first axial mode with the structure constrained for launch is 49.7 Hz (against > 20 Hz required by launcher) and the first lateral mode is 60 Hz (> 10 Hz required). The first unconstrained normal mode is an axial mode of the solar array at 51.9 Hz, thus totally decoupling the structure from the bandwidth of the controls. The design maximizes the moment of inertia J_Z with respect to the symmetry axis, thereby providing passive spin stabilization around it, and meets the requirement of a ratio $(J_Z - J_X)/J_X > 0.3$.

The thermal control is passive, using multi-layer insulation blankets and heaters. The satellite body never sees the sun, and the environmental conditions are exceptionally stable throughout the year (design "cold cases" and "hot cases" differ by very little). The largest thermal load is from incident sunlight and the Earth IR flux. Three sets of multi-layer insulation, on the exterior and interior of the spacecraft shell and the exterior of the PGB, protect the test masses from the external fluxes. The conductive link from the payload compartment to the main structure is via tiny springs. A very small amount of power is dissipated within the PGB, the bulk of the electrical power dissipation occurring in the spacecraft compartments. Thus the test mass temperatures are practically unaffected by variations of the external and internal environment. A thermal mathematical model of the entire spacecraft was formed and used to assess the thermal control performance. In the worst-case environmental conditions (external fluxes and internal dissipations), the following results were found. The test mass axial temperature gradient is always $< 10^{-3}$ K/m (against a requirement of < 4 K/m). The test mass azimuthal temperature difference is always $< 10^{-6}$ K, limited by model resolution (requirement: $< 1.6 \times 10^{-6}$ K). Any test mass mean temperature variation is $< 5 \times 10^{-3}$ K/day (requirement: < 0.1 K/day). The thermal time constant of the payload enclosure is so large that the satellite must be launched at close to orbital temperature (about 5 °C at the test masses), otherwise the drift to thermal equilibrium would dominate the test mass temperatures. With this provision, the experiment runs at near-constant temperature all through its orbital life.

Gallium arsenide (GaAs) solar cells with photovoltaic conversion efficiency of 19% provide primary power. After accounting for radiation, pointing and

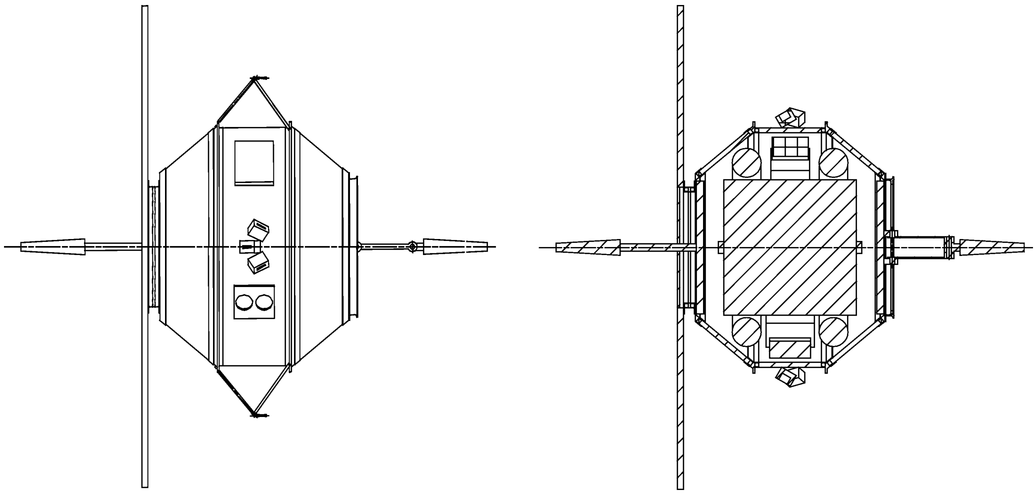

Fig. 3. Layout of the reorientation thrusters and cross section showing the toroidal tanks for symmetric mass depletion.

packing factor losses, the specific power is on the order of 150 W/m^2. A preliminary assessment of the power demand of the spacecraft gives about 200 W, including battery charging and margins. Therefore, the minimum required array area is about 1.4 m^2. More than 3 m^2 are available for cells in the configuration of Fig. 2, where the disk-shaped array is sized for its use as sun shield.

The spacecraft data management unit handles the ground link (telemetry and command) and executes the software performing the platform attitude control, data handling and fault detection and recovery tasks via a 1553 data bus. Data are stored in a modular solid-state mass memory unit. A dedicated computer handles the payload control and processing tasks. The total mass amounts to 276 kg (316 kg with system margins).

4. Attitude and drag-free control

The launcher releases the satellite with a small spin rate and the spin axis roughly oriented in the required direction. An autonomous system provides for initial correction of angular momentum depointing and damping of residual angular rates and for spin-up to the nominal spin rate of 120 rpm. Two (+2 redundant) circumferential 200 mN nitrogen thrusters are provided for this purpose. The maneuver lasts 36 minutes and uses 1.3 kg propellant. Another pair of 200 mN nitrogen thrusters, at right angles to the former set, is used for the periodic maneuvers to realign the spin axis to the orbit normal. For this function the thrusters, located at 0.7 m from the spin axis, are operated in pulsed mode for 3.4 hours. Before each reorientation maneuver, the test masses are locked by means of inch worms (use of the launch-lock mechanism is not necessary). After the maneuver, the inch worms are released in a controlled way, using the pressure sensors associated to them for measuring the displacement from the symmetry axis until it is small enough for the capacitive sensors/actuators to take over whirl control. Finally, the along-track drag signal is used for proof mass balancing and capacitive plate centering, thus restoring the nominal common-mode rejection ratio. In the mission operations plan, one day is assigned to each reorientation maneuver (including both thruster actuation and lock-unlock-rebalancing procedure). The propellant mass required is 0.38 kg per maneuver, which translates into 6.8 kg/year. The total propellant budget for 2-year operation is therefore 15 kg. The nitrogen propellant is stored at 550 bar, giving a required volume of 40 liters. Two toroidal tanks will be employed, placed as shown in Fig. 3, symmetric with respect the spacecraft center of mass. The propellant for the periodic maneuvers is drawn from both tanks so as to produce negligible displacements of the center of mass.

Because of the position of the solar array, the thrusters for the re-orientation maneuvers cannot be

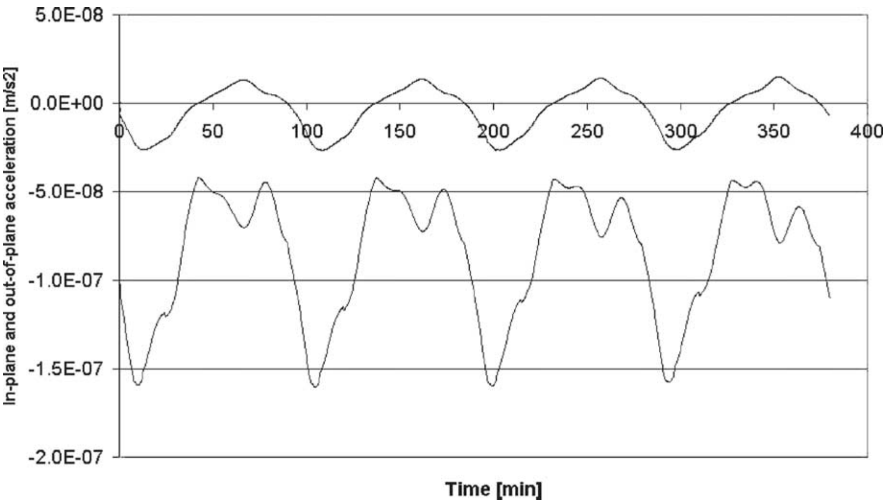

Fig. 4. Perturbing accelerations in the measurement plane and perpendicular to it, when the spin axis is 10° away from the orbit normal (summer solstice 2005). The magnitude of the out-of-plane acceleration (top) is less than 1/6 of the in-plane component (bottom).

balanced (they produce torques as well as forces). Therefore they will produce an effect on the orbit. The main component of the thrust is along the orbit normal. At first order, thrust in the direction of the orbit normal affects the inclination and longitude of ascending node. The magnitude of the perturbation is about 0.12 milliradians per maneuver, completely negligible. Moreover, the orbital positions at which the maneuvers are executed can be selected in an appropriate way so as to make the net effect zero over a number of maneuvers.

The kinetic spin energy at the nominal rate of 2 Hz is so high compared to all natural torques that they would need a very long time to even slightly displace the spin axis. Therefore, active control of the direction of the spin axis in space is not needed during measurement periods.

The main perturbing accelerations are from solar radiation pressure and atmospheric drag. Fig. 4 shows the pattern of the accelerations in worst-case conditions (epoch 2005, spin axis at maximum angle to orbit normal). The effect of a 10° offset of the spin axis with respect to the orbit normal is an increase of the magnitude of the in-plane forces by a factor of 2, and an increase of the magnitude of the out-of-plane forces by a factor of 5. The out-of-plane force remains less than $1/6$ of the in-plane component.

Drag-free control must reduce the common-mode drag force acting on the satellite in a narrow bandwidth (notch filter) centered at the orbital frequency (1/5700 s), by a factor $1/10\,000$ in the XY plane and by a factor $1/400$ on the Z axis. A supplementary requirement is to reduce by a factor $1/100$ the external disturbance acting in the XY plane at the test-mass differential frequency ($1/540$ s). By reducing the acceleration, the amplitude of oscillation of spacecraft-PGB is reduced by the same amount, at the same frequency. Phase differences due to a differential rotation rate between the spacecraft outer shell and the PGB are mostly compensated passively. Residuals are detected by displacement sensors and corrected by the drag-free control thrusters.

The actuator system for drag-free control must produce finely tuned (in magnitude and frequency) forces, using tiny amounts of propellant in order to minimize perturbations on the test bodies from nearby moving masses. The solution adopted is to use Field Emission Electric Propulsion (FEEP) mini-thrusters with high specific impulse (gas velocity at nozzle exit > 6 km/s) [3]. A few grams of cesium suffice for the entire 2-year duration of the mission. The drag-free control thrusters are accommodated in two clusters of 3 each, with orientations providing sufficient torque and force authority on all axes. Fig. 5 shows the performance required of the notch filter implementing the drag-free control. Detailed time simulations have shown the adequacy of the adopted control laws and the selected actuators [1].

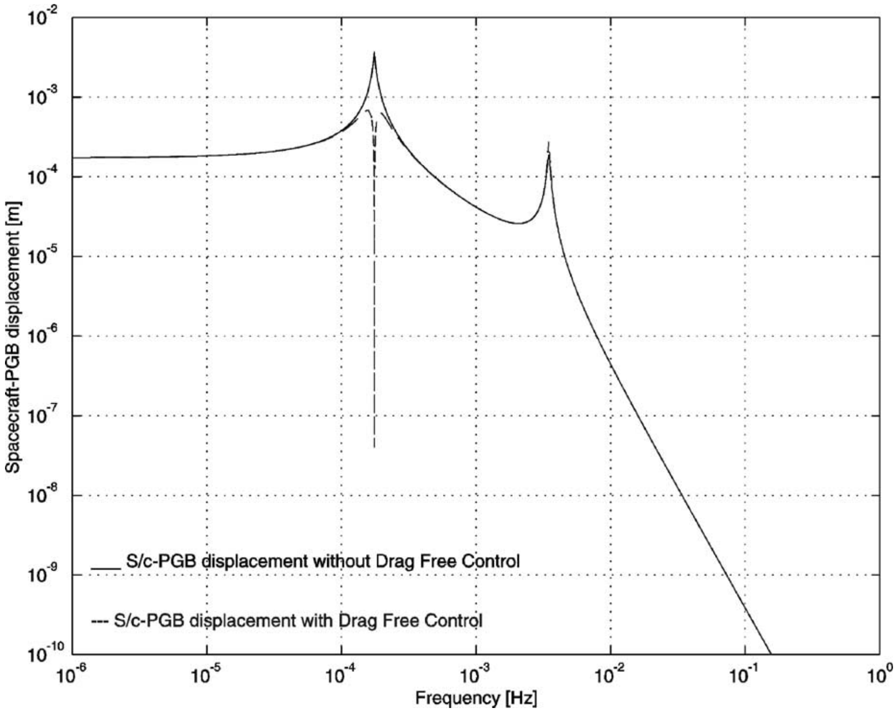

Fig. 5. Relative displacement between PGB and spacecraft without (continuous line) and with (dashed line) drag-free control implemented by notch filter at orbital frequency.

The drag-free and attitude control system is subjected to demanding requirements on the measurement and control of the spin rate ω_S and the orientation of the spin axis. The standard deviation of the spin angular frequency determines the residual drag at the orbital frequency (placement of the notch filter). This translates into a control requirement of $\Delta\omega/\omega_S \leqslant 10^{-4}$, equivalent to detecting a phase shift of $\Delta\varphi = 6 \times 10^{-4}$ rad (0.035°) in one spin period. On the other hand, a constant error in the determination of the spin angular frequency is easily removed from the controller input. The requirement for measuring the separation between the spin rate and the fundamental frequency of the test mass differential mode is one order of magnitude less demanding, $\Delta\omega/\omega_S \cong 10^{-3}$.

The satellite is made up of six co-aligned, co-rotating cylinders, coupled by springs. The spin frequency is much higher than the natural frequency due to the mechanical coupling of the springs, in order to produce supercritical rotation, which is the closest possible condition to that of free rotating cylinders. External torques applied to the outermost cylinder (the 'spacecraft') will tend to change its orientation relative to the cylinder surrounding the test masses (the PGB). Differential spin rates between the spacecraft and the PGB must not be allowed to produce an angular displacement in the plane perpendicular to the spin axis $> 10^{-2}$ rad, and a tilt of the spin axis of the spacecraft with respect to that of the PGB $< 10^{-2}$ rad. The tilt of the spin axis during measurement periods is negligible, because of the high ratio of spin kinetic energy to magnitude of the perturbing torques. Hence, spin axis control is not needed during measurement periods. The largest effect potentially producing a change in the spin rate of the spacecraft is due to thermal expansion and contraction of the satellite. The eclipse-sun and sun-eclipse transitions induce such thermal expansions and contractions, which change the spacecraft inertia and hence, to conserve angular momentum, the spin rate. With the given SSO orbit and thermal design, this effect is minimized and easily managed by a dedicated spin controller (proportional-derivative controller plus low pass filter). The system for detecting the relative phase between spacecraft and PGB consists of two symmetrically placed pairs of light-emitting diodes. The resolution is better than 10 μm at a distance of 30 cm (43 arcseconds). The phase coming from the average of the two sensor pairs is used as input for the spin controller.

Both the orientation of the spin axis and the direction of the Earth with respect to an inertial frame must be measured with an accuracy of 6×10^{-3} rad (0.35°). An Earth elevation sensor is envisaged for measuring the inertial direction of the Earth's center. For a satellite with momentum bias perpendicular to the orbit plane, there is no indetermination in yaw angle measurement with just one sensor. The satellite orientation is inertially fixed because of the gyroscopic stiffness, and therefore an error in the yaw angle would be detected, after a quarter turn, as an error in roll. The sensor is fixed to the satellite with two optical axes perpendicular to the spin direction, and the satellite spin provides the required scanning motion. The required Earth sensor must provide a systematic error $< 0.05°$ and a random error $< 0.01°$ (3σ).

5. Conclusions

The design of the satellite for the GG experiment was addressed in phase-A level studies in 1997–2000 (based on an equatorial orbit) and more recently (2001) re-addressed for sun-synchronous orbit. The satellite is small, low-weight, with low power demand. The service functions can be provided by standard, low-cost equipment. The mission consists of an experiment running uninterrupted with few operational modes, small telemetry rates, easily controlled by one ground station. The experiment requirements lead to an ad hoc configuration, resembling a spinning top, very compact and stiff. The requirements for drag-free control, spin rate control, thermal stability are demanding but manageable, as shown by extended analysis and simulations. The reconfiguration to SSO makes the mission suitable for a low-cost launch, and improves the thermal performance. Enabling technology items include the drag-free control thrusters and attitude sensors, for both of which state-of-the-art candidates exist.

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