Determination of the accelerometer metrological characteristics on board the METRIC mission

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***Abstract*—METRIC is a proposed mission that aims to improve the knowledge of atmospheric density, general relativity and geodesy. The mission foresees a small spherical satellite placed in a polar eccentric orbit, with the apogee at 1200 km of altitude and perigee at 400-450 km. The spacecraft will be tracked from ground and space and have an on-board 3-axis accelerometer. The accelerometer will measure non-gravitational accelerations (due mainly to neutral drag and solar radiation pressure). A dynamical simulator has been implemented with known models (e.g. higher-harmonics gravity field, upper atmosphere density, solar pressure) in order to estimate the non-gravitational accelerations acting on the satellite. These are then analyzed in order to extract their main characteristics: intensity and frequency spectrum. Consequently, an envelope for the accelerometer features — dynamic range, required accuracy and frequency bandwidth — can be set. The data collected by the accelerometer will be used, in combination with tracking data from GNSS and SLR, to improve the atmospheric density knowledge over a wide range of altitudes, to perform an accurate measurement of the orbital precessions predicted by general relativity and implement a space-based tie of geodetic reference frames.**

***Index Terms*—accelerometer, metrological characteristics, atmospheric density, general relativity, navigation, geodesy**

I. Introduction

The near-Earth environment has been an important field of study for many years and nowadays there are several mathematical models that describe it, as for instance atmospheric density, general relativity effects connected to the Earth's gravity field. However, very few missions have measured the non-gravitational accelerations in-situ with high accuracy (e.g., CHAMP, GRACE [1]) but have done so in circular low Earth orbits (LEO). In-situ measurements at altitudes that span a wide region of LEO are missing and are necessary for improving the models. The use of circular orbits has also limited the impact of missions to measure the effects of general relativity because the position of the pericenter and consequently the precession of the apsidal line are not well defined/measurable in those orbits.

The proposed mission named METRIC [2] (Measurement of EnvironmenTal and Relativistic In-orbit preCessions) aims to contribute to these scientific goals with a particular architecture that mixes different technologies from previous missions. For instance, similarly to LAGEOS [3] and LARES [4] missions (launched for the first time respectively in 1976 and 2013), METRIC is thought to be a well-trackable satellite with retroreflectors that allow the tracking of its position with laser techniques from ground (Satellite Laser Ranging – SLR). Generally, the evaluation of the precise gravitational effects on a test mass is based on a platform that is not disturbed by non-gravitational forces. This issue can be dealt with a drag-free or drag-compensated satellite (e.g., in GOCE the drag-compensation system [5] requires fine actuators controlled through the accelerations measured by an on-board accelerometer) in order to cancel or mostly compensate the effects of non-gravitational forces.

However, a virtually drag-free satellite represents a simpler solution to be realized. CHAMP and GRACE already used this solution to evaluate the variations of the gravity field in both space and time. This technique does not require a propulsion system, because the satellite is left orbiting around the primary, affected by the non-gravitational forces. An accurate accelerometer is needed to measure the disturbing accelerations, in order to remove their effects post-flight. METRIC aims to implement a virtually drag-free satellite in order to reach its scientific goals as detailed later on.

In this paper, we present a numerical analysis that allows to determine the metrological characteristics of the accelerometer on board the proposed mission in terms of dynamic range, required accuracy and frequency bandwidth. The accelerometer measurements are intended to be combined with Precise Orbit Determination (POD) techniques to improve the accuracy of the models of atmospheric density and test general relativity. After a more specific description of the METRIC objectives and its onboard instrumentation (section II), the methodology of analysis is introduced in section III. Section IV shows the simulation results and the accelerometer metrological characteristics proposed for METRIC. Conclusions are outlined in section V.

II. The METRIC Mission

A. METRIC main objectives

METRIC [2] is a proposed scientific mission that focuses its goals on the study of near-Earth environment's physics and characteristics. Specifically, METRIC's main objectives can be summarized as follows: 1) map the atmospheric density in an orbital region of great interest to satellite lifetimes and debris mitigation; 2) conduct fundamental physics tests verifying the validity of the general relativistic equation of motion; and 3) establish a space-based tie among different space geodesy techniques.

The first goal requires measurements by a high-accuracy accelerometer of the non-gravitational accelerations so that the contribution due to drag can be extracted from them. The choice of an appropriately-chosen elliptical orbit allows the separation of the drag contribution as explained later on. The second objective is the reason for which the satellite has to become a virtually drag-free test mass. In this case, differently from the previous goal, data obtained by the accelerometer are disturbances for the parameters of interest and their effects on the satellite motion will be removed from tracking data with the goal of estimating the effects of general relativity. The last objective is related to spacecraft tracking and concurrent measurements. The main concept is to realize the space counterpart of what is known as a local tie in geodesy by creating a well-known metrological platform that may host various space geodesy techniques. The architecture and the orbit of METRIC are chosen in order to reach the goals of the mission.

B. METRIC orbit and on-board instrumentation

The METRIC satellite should fly on a polar eccentric orbit with the apogee at 1200 km of altitude and the perigee at 400-450 km (Fig. 1). This orbit has been selected for specific reasons. First, the inclination of $90°$ allows canceling the even terms of the Earth's gravity series expansion (i.e., C_{20}, C_{40}, etc) [6], which mostly contribute to orbit's perturbation, leading to a precession of the nodal line that becomes to be dominated by non-gravitational forces and not by Newtonian gravity effects, thus making possible a better analysis of the relativistic effects. Second, the eccentricity of the orbit simplifies the estimation of the acceleration due to atmospheric drag that can be separated from that due to solar radiation pressure. Indeed, near the perigee the acceleration due to drag is much higher than the one due to solar radiation pressure (SRP) [6]; on the contrary, near apogee the effects of solar pressure are much more important than those of atmospheric drag. The acceleration due to SRP acting on a spherical satellite is also nearly constant (except when the satellite goes into eclipse), while the one related to drag changes periodically over the orbit. The selected orbit (i.e., perigee and apogee

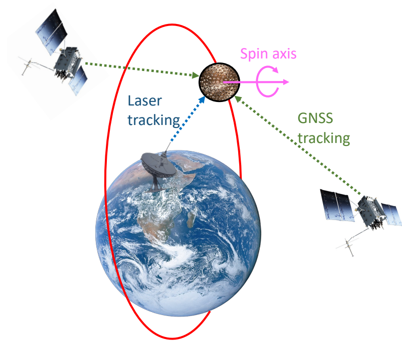

Fig. 1. The orbit of the METRIC mission with the spacecraft tracked from both SLR network and GNSS constellations.

altitudes) spans a region of great interest for satellite orbital lifetimes and the mitigation of orbital debris. Satellites in the mid of this region may or may not violate the 25-year rule for satellite deorbiting. Moreover, conjunction analyses of satellite impacts will benefit greatly from a better knowledge of the atmospheric density. There is a need for more accurate drag models, obtained also through accurate in-situ measurements of atmospheric drag deceleration.

The satellite design foresees a small sphere with diameter in the range $50–60\,\mathrm{cm}$ and corner cube retroreflectors on the surface to allow accurate laser tracking from Earth. The expected mass is estimated to be less than $100\,\mathrm{kg}$. Due to its spherical shape, the spacecraft cross section is constant for both SRP and drag and this simplifies the post-processing analysis since the non-gravitational accelerations become decoupled from the spacecraft attitude.

The satellite hosts a 3-axes accelerometer that measures the non-gravitational accelerations acting on the spacecraft and is fundamental for the achievement of the mission goals that its metrological characteristics are properly selected.

Another key instrument is a GNSS (Global Navigation Satellite System) receiver, that helps the tracking of the spacecraft from space. By combining GNSS and SLR measurements, the position of the satellite can be estimated continuously leading to a precise orbit determination. Orbital parameters, especially the rotation of both the nodal line and the perigee, can be precisely estimated in order to evaluate the effects of general relativity post-flight. The Attitude Determination and Control System also includes magnetorquers to impose a spin to the satellite. Thanks to the spin, the accelerations measured by the accelerometer are modulated, so that also constant (in magnitude) contributions appear as oscillating signals, improving the frequency separation of the measured acceleration contributions and the accurate estimate of the acceleration phase. The spacecraft should also carry a pressure gauge on board to help estimate the not perfectly known drag coefficient of the satellite through the combination of in-situ acceleration and pressure measurements.

III. Methodology of Analysis

A software simulator of orbital dynamics has been implemented in Matlab© in order to compute the expected accelerations acting on the spacecraft, to determine the metrological characteristics of the accelerometer vis a vis the requirements stemming from the mission objectives. The equation of motion for the satellite is:

$$\ddot{\mathbf{r}} = -\frac{\mu}{r^3}\mathbf{r} + \mathbf{p} \tag{1}$$

where $\mathbf{r}$ is the satellite geocentric position vector and the perturbing acceleration $\mathbf{p}$ includes: 1) the effects due to the non-homogeneity of the Earth's gravity field $\mathbf{a}_{\mathrm{GF}}$; 2) the non-gravitational accelerations due to drag $\mathbf{a}_{\mathrm{drag}}$; 3) the non-gravitational accelerations due to solar radiation pressure $\mathbf{a}_{\mathrm{SRP}}$; and 4) the relativistic effects $\mathbf{a}_{\mathrm{rel}}$.

Specifically, $\mathbf{a}_{\mathrm{GF}}$ includes the effects related to the two zonal coefficients C_{20} and C_{30} that are the most significant ones and cause a rotation of the nodal line and the periapsis; these contributions are computed with well-known formulas, as reported, for example, in [6]. The acceleration $\mathbf{a}_{\mathrm{drag}}$ is important near the perigee (here taken at 450 km of altitude) and is computed as $\mathbf{a}_{\mathrm{drag}} = \frac{1}{2}\rho v \frac{A}{m} C_D \mathbf{v}_{\mathrm{rel}}$, where ρ is the atmosphere density, $\mathbf{v}_{\mathrm{rel}}$ is the vector of the spacecraft velocity relative to the atmosphere (v is its norm), A is the spacecraft cross-section, which is constant because of the satellite geometry, m is the satellite mass and C_{D} is the drag coefficient, assumed constant and equal to the typical mean value of 2.2 at these altitudes. The atmospheric density ρ depends on both the spacecraft position and the epoch because of the solar activity, and it can be computed from known empirical models; a good approximation is provided by the NRLMSISE-00 model implemented in Matlab© in the function *atmosnrlmsise00* [7].

In addition, the acceleration due to the SRP is computed as $\mathbf{a}_{\mathrm{SRP}} = \nu C_{\mathrm{R}} \frac{A}{m} P_0 \frac{\mathbf{r}_\odot}{r_\odot}$, where ν is a coefficient related to eclipses, C_{R} is the reflectivity of the spacecraft, P_0 is the solar pressure per unit area at a distance of $1\,\mathrm{au}$ ($4.56 \times 10^{-6}\,\mathrm{N\,m^{-2}}$), $\mathbf{r}_\odot$ is the position vector of the satellite with respect to Sun ($r_\odot$ is its norm). The ν coefficient is obtained with a function that evaluates if the Sun is visible by the spacecraft or shadowed by Earth or Moon through geometrical considerations. The value of ν is 1 when the Sun is visible and so the solar radiation pressure acts on the satellite, while $\nu = 0$ when the satellite is in shadow. The cross-section area is constant due to the spacecraft geometry.

Following the IERS (International Earth Rotation and Reference Systems Service) Convention (2010) [8] the relativistic acceleration that must be added to correct the dynamical model can be divided into three contributions: 1) the gravitoelectric contribution, also called Schwarzschild term, is the dominant contribution between the relativistic effects and influences the rotation of the perigee; 2) the gravitomagnetic contribution, also called Lense-Thirring term, is one or two orders of magnitude smaller than the Schwarzschild contribution and acts on the precession of nodal line; and 3) the geodetic contribution, also known as De Sitter term, is the less intense effect in the case of the METRIC orbit. The formulation of the relativistic accelerations can be written as:

$$\begin{aligned}\mathbf{a}_{\mathrm{rel}} = \frac{\mu_\oplus}{c^2 r^3}\left\{\left[2(\beta+\gamma)\frac{\mu_\oplus}{r} - \gamma\dot{\mathbf{r}}\cdot\dot{\mathbf{r}}\right]\mathbf{r} + 2(1+\gamma)(\mathbf{r}\cdot\dot{\mathbf{r}})\dot{\mathbf{r}}\right\} + \\ (1+\gamma)\frac{\mu_\oplus}{c^2 r^3}\left[\frac{3}{r^2}(\mathbf{r}\times\dot{\mathbf{r}})(\mathbf{r}\cdot\mathbf{J}_\oplus) + (\dot{\mathbf{r}}\times\mathbf{J}_\oplus)\right] + \\ \left\{(1+2\gamma)\left[\dot{\mathbf{R}}\times\left(\frac{\mu_\odot\mathbf{R}}{c^2R^3}\right)\right]\times\dot{\mathbf{r}}\right\}\end{aligned} \tag{2}$$

in which $\mu_\oplus$ and $\mu_\odot$ are the gravitational coefficients of Earth and Sun respectively, c is the speed of light, β and γ are parameterized post-Newtonian (PPN) parameters, $J_\oplus$ is the Earth's angular momentum per unit mass and $\mathbf{R}$ is the position vector of the Earth with respect to the Sun (R is its absolute value). β and γ are set equal to 1 with good approximation (error in the order of 10^{-5}, as obtained through different missions and experiments about general relativity). The Earth's angular momentum per unit mass is a vector defined as $J_\oplus = [0\ 0\ \frac{2}{5}R_\oplus^2\omega_{\mathrm{e}}]^T$, assuming, only in this case, that the Earth is perfectly spherical, with ω_{e} the rotational velocity of the Earth and $R_\oplus$ the Earth mean equatorial radius. All three terms are quite small (orders of magnitude between 10^{-12} and $10^{-9}\,\mathrm{m\,s^{-2}}$) leading to small accelerations; it is well known that the rotations of nodal and apsidal lines due to relativity are in the order of arcsec year^{-1}.

Since (1) is a second-order differential equation, a numerical integrator is needed to compute both the state vector (i.e., all spatial components of position and velocity) and the perturbing accelerations step by step. In this way, it is possible to link non-gravitational and relativistic accelerations to the satellite position and velocity.

The orbit of the spacecraft should be practically determined with different POD techniques. The typical measurement precision is nowadays in the order of cm or less. In our simulation procedure, the orbit is converted from Cartesian coordinates to Keplerian orbital parameters, hence evaluating the Argument Of Perigee (AOP) and the Right Ascension of Ascending Node (RAAN) and their drifts over time. Removing

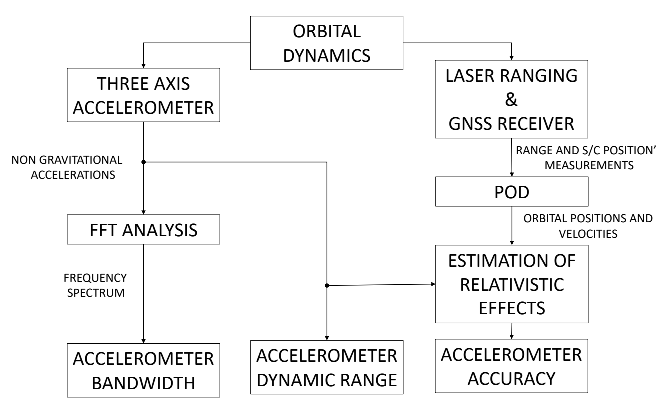

Fig. 2. Block diagram showing the computations done to determine the accelerometer metrological characteristics.

from these values the relevant contributions obtained from the measurements provided by the accelerometer, the effects of relativity can be estimated (Fig. 2).

Drifts of AOP and RAAN are simulated considering the simplified case in which the satellite is affected only by relativistic effects and non-gravitational accelerations below specific thresholds, that represent different accuracies of the accelerometer. Assuming the simulation with only the spherical term of gravity and the general relativity as the reference case (i.e., assuming an ideal accelerometer), residuals on the drifts can be determined based on the effect of the accelerometer actual accuracy/resolution values that impact the accuracy of the estimation of the relativistic effects.

Finally, the acceleration components measured by the accelerometer are processed by a Fast Fourier Transform in order to obtain their frequency spectrum and determine the required bandwidth of the accelerometer.

IV. Simulations and Results

The effects of the accelerations due to SRP and atmospheric drag change by varying both the orientation of the solar rays with respect to the orbital plane and the epoch due to the variability of the solar activity. For these reasons, the orbital dynamics of the spacecraft is simulated over one year, with a focus on different orientations of the orbit both at equinoxes and solstices. Moreover, the orientation of the orbital plane is assumed once parallel and once perpendicular to the direction of the solar rays, in order to evaluate the range of possible accelerations due to non-gravitational forces. The attitude of the spacecraft is not considered in this computations because it does not change the magnitude of the acceleration: it only adds a spin-frequency modulation to the environmental acceleration, that does not influence for this particular analysis the accelerometer characteristics. In a follow-on analysis the spin frequency will have to be carefully selected and related to the centering requirements of the instrument inside the spacecraft.

The following results refer to an orbital plane parallel to the solar rays and across the spring equinox. Specifically, Figures 3, 4 and 5 show the components of the non-gravitational (drag and SRP) and the relativistic accelerations

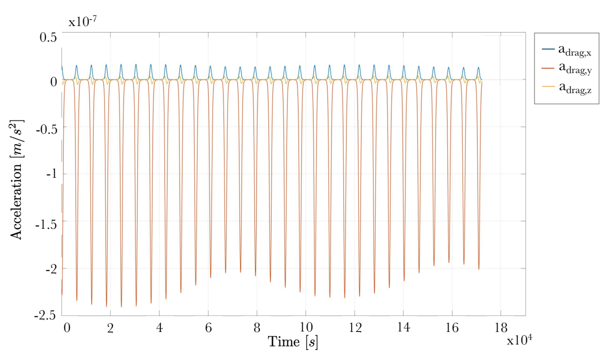

Fig. 3. Components of the acceleration due to atmospheric drag for 28 orbits across the spring equinox.

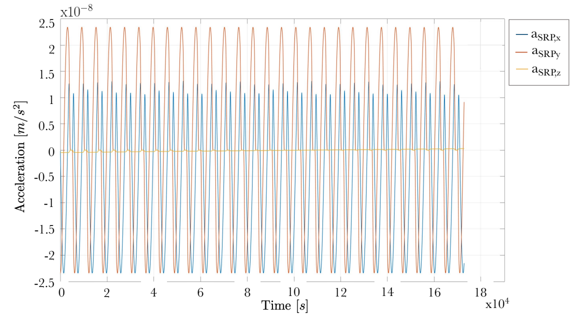

Fig. 4. Components of the acceleration due to solar radiation pressure (SRP) for 28 orbits across the spring equinox.

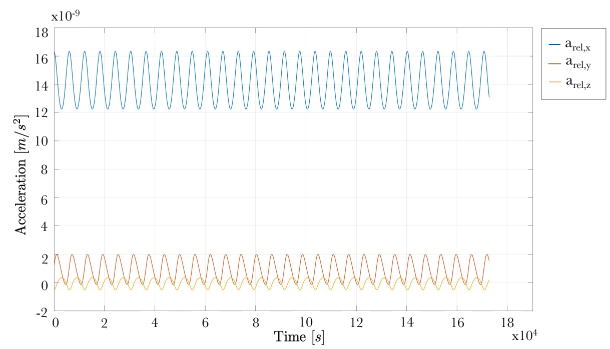

Fig. 5. Components of the acceleration due to general relativity for 28 orbits across the spring equinox.

as a function of time near the spring equinox. The components of the accelerations are expressed in the Local-Vertical Local-Horizontal (LVLH) frame of reference attached to the satellite, with the x-axis that lies on the local vertical direction and points towards the zenith, the z-axis normal to the orbital plane and parallel to the angular momentum of the orbit, and the y-axis completing the triad (it is on the local horizon towards the flight direction). The non-gravitational accelerations measured by the accelerometer are utilized to extract, at different levels of accuracy, the effects of general relativity on the satellite motion (that is one of the goals of the mission). Figure 6 shows a comparison between the magnitude of the non-gravitational and the relativistic accelerations over 3 orbital periods.

The first important thing to underline is the order of magnitude of each contribution: $10^{-7}\,\mathrm{m\,s^{-2}}$ for drag, $10^{-8}\,\mathrm{m\,s^{-2}}$ for SRP and 10^{-9}–$10^{-8}\,\mathrm{m\,s^{-2}}$ for relativistic effects. In addition, in the previous figures, we can observe a periodicity of the acceleration components related to the orbital period of METRIC (about 1 hour and 40 minutes). Specifically, the drag-related acceleration is the highest; all of its components present peaks when the satellite is near the perigee and flat parts very close to zero when the satellite is near the apogee. The most important component is the y component, which is almost along the velocity, and reaches maximum values of $(2 \times 10^{-7}$–$2.5 \times 10^{-7})\,\mathrm{m\,s^{-2}}$ in magnitude. On the contrary, the acceleration due to SRP is dominant at the

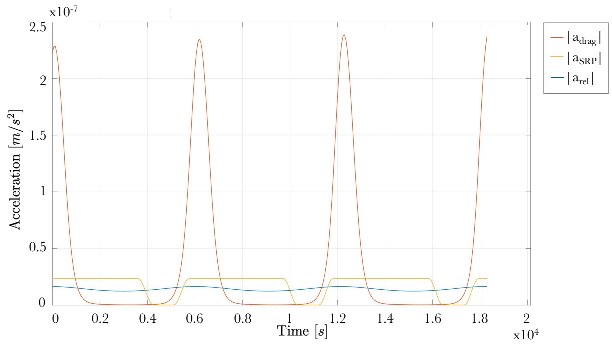

Fig. 6. Comparison between the acceleration magnitude due to atmospheric drag (red line), solar radiation pressure (yellow line) and general relativity (blue line).

apogee and presents x and y components oscillating, while the z component is almost zero in this case. This is because the LVLH frame is a rotating frame where x and y axes exchange between each other with orbital periodicity and the z-axis remains almost parallel to itself. In addition, the z component is null because in this configuration the z-axis is always perpendicular to the direction of the solar rays.

Regarding the relativistic acceleration (Fig. 5) each component in the LVLH frame is oscillating with a precise periodicity. The most important effect is along the zenith because the higher term in (2) is the Schwarzschild contribution (in particular the first part of that term) that is aligned with the position vector **r** (parallel to the x-axis of the LVLH frame). Indeed, its contribution is at least one order of magnitude larger than the other two components. The acceleration due to general relativity is the smallest effect among all the perturbation terms present in the equation of motion.

Accelerations experienced by the satellite related to other epochs and orbital geometric configurations have been evaluated and show a trend similar to the one already discussed. The most significant differences are the intensity of drag contribution, associated with a different solar activity, that changes significantly the atmospheric density [6], and the SRP contribution, which can present a constant value along one axis, as Figure 7 shows, depending on the orientation of the orbital plane with respect to the solar rays.

Figure 8 shows the Power Spectral Density (PSD) of the accelerations due only to drag and SRP that are used to evaluate the bandwidth of the onboard accelerometer. The acceleration measured by the accelerometer includes several terms with increasing frequency and decreasing amplitude; the most important of them is in correspondence of the peak at about 1.6×10^{-4} Hz (marked with a vertical dashed line) and is at the orbital frequency. As just mentioned, some acceleration components can be constant, especially those due to the SRP, and consequently they could not be measured accurately by a "stationary" accelerometer. This issue is solved by spinning the spacecraft as reported in Section I.

The accuracy of the accelerometer affects the accuracy in estimating the drift of the RAAN $\dot{\Omega}$ and AOP $\dot{\omega}$ (Fig. 9). An accurate estimation of those drifts is fundamental for enabling a good estimation of the relativistic effects. With an accuracy of the accelerometer of $10^{-10}\,\mathrm{m\,s^{-2}}$, the relative errors on $\dot{\Omega}$ and $\dot{\omega}$ are 2% and 4.5% respectively.

The metrological characteristics required for the 3-axis accelerometer for METRIC are: dynamic range $\pm10^{-6}\,\mathrm{m\,s^{-2}}$, bandwidth 10^{-4}–10^{-1} Hz and accuracy $10^{-10}\,\mathrm{m\,s^{-2}}$. Table I provides a comparison between the accelerometer proposed for METRIC and the accelerometers used in other past or current missions in terms of dynamic range, bandwidth and accuracy. The 3-axis accelerometer of METRIC will be derived from the ISA accelerometer [9, 10] developed for the BepiColombo mission to Mercury.

TABLE I
METROLOGICAL CHARACTERISTICS OF METRIC ACCELEROMETER COMPARED TO THE ONES USED IN PAST OR CURRENT MISSIONS. * FOR TWO AXIS, † FOR THE LESS SENSITIVE AXIS, § INSTRUMENT LABORATORY PERFORMANCE.

Mission	Dynamic range $[\mathrm{m\,s^{-2}}]$	Bandwidth [Hz]	Accuracy $[\mathrm{m\,s^{-2}}]$	Orbit
CHAMP	$\pm10^{-4}$	10^{-4} to 10^{-1}	$3\cdot10^{-9}$ * $3\cdot10^{-8}$ †	445 km circular
GRACE	/	$2\cdot10^{-4}$ to 10^{-1}	$3\cdot10^{-10}$ * $3\cdot10^{-9}$ †	495 km circular
GOCE	$\pm3\cdot10^{-6}$	$5\cdot10^{-3}$ to 10^{-1}	10^{-12}	255 km circluar
Bepi Colombo	$\pm3\cdot10^{-6}$	$3\cdot10^{-5}$ to 10^{-1}	10^{-9} §	interplanetary
METRIC	$\pm10^{-6}$	10^{-4} to 10^{-1}	10^{-10}	450x1200 km elliptical

V. CONCLUSIONS

The focus of this paper is on the 3-axis accelerometer, which is a crucial scientific payload for the proposed METRIC

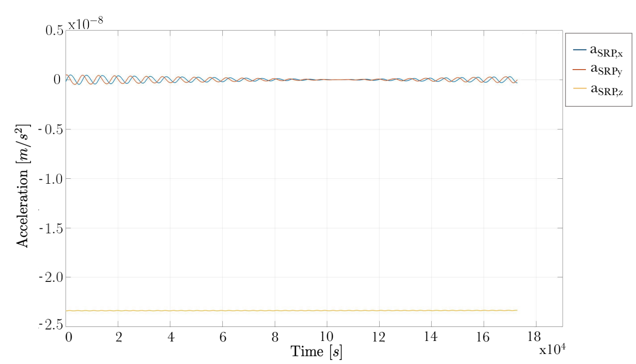

Fig. 7. Components of the acceleration due to solar radiation pressure (SRP) for 28 orbits across the spring equinox with orbital plane perpendicular to solar rays direction.

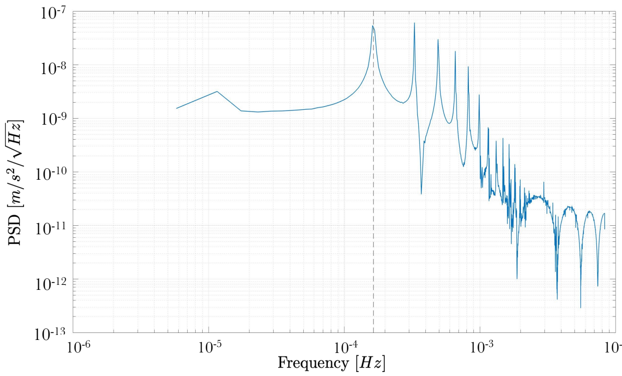

Fig. 8. PSD of the acceleration measured by the onboard accelerometer and due to drag and SRP.

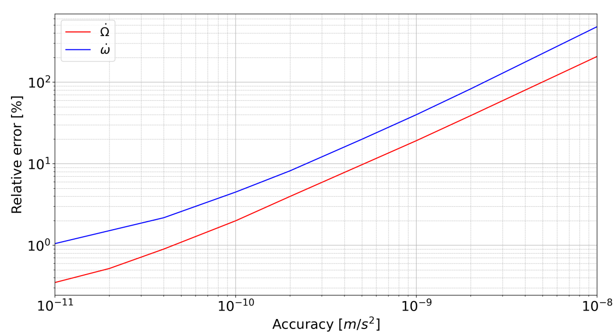

Fig. 9. Relative errors on $\dot{\Omega}$ (red solid line) and $\dot{\omega}$ (blue solid line) as a function of the accuracy of the accelerometer.

mission. Derived from the ISA accelerometer developed for the BepiColombo mission to Mercury, the 3-axis accelerometer package is asked to deliver an outstanding performance to accurately measure the non-gravitational accelerations acting on the spacecraft in the orbital altitude range of 400–1200 km. The accelerometer measurements will be combined with GNSS and SLR data to enhance our understanding of Earth's atmospheric density, geodesy, and general relativity.

The study utilized Matlab® to conduct accurate simulations that calculated non-gravitational accelerations experienced in different orbital plane orientations relative to the solar rays. The simulations accounted for various acceleration contributions, such as non-uniformity in Earth's gravity field, atmospheric drag, solar radiation pressure and the relativistic effects. The satellite state vector has been obtained from the orbital propagator and used to determine two key quantities: the non-gravitational accelerations measured by the accelerometer and the drifts over time of the AOP and RAAN caused by the relativistic effects. The 3-axis accelerometer metrological characteristics have been identified and specifically: dynamic range of $\pm 10^{-6}\,\mathrm{m\,s^{-2}}$, a bandwidth of 10^{-4}–10^{-1} Hz and an accuracy of $10^{-10}\,\mathrm{m\,s^{-2}}$.

The METRIC accelerometer demands an accuracy improvement of the BepiColombo ISA accelerometer by one order of magnitude (considering its instrumental accuracy). By achieving this enhanced accuracy, the accelerometer will allow estimating the drift of both RAAN and AOP with remarkable precision, i.e., relative errors of 2% and 4.5%, respectively.

References

[1] Jose van den IJssel and Pieter Visser. "Performance of GPS-based accelerometry: CHAMP and GRACE". In: *Advances in Space Research* 39.10 (Jan. 2007), pp. 1597–1603. DOI: 10.1016/j.asr.2006.12.027.

[2] R. Peron and E. C. Lorenzini. "METRIC: A Dedicated Earth-Orbiting Spacecraft for Investigating Gravitational Physics and the Space Environment". In: *Aerospace UK* 4.3 (July 2017). DOI: 10.3390/aerospace4030038.

[3] S. C. Cohen and D. E. Smith. "LAGEOS Scientific results: Introduction". In: *Journal of Geophysical Research* 90.B11 (Sept. 1985), pp. 9217–9220. DOI: 10.1029/JB090iB11p09217.

[4] Antonio Paolozzi and Ignazio Ciufolini. "LARES successfully launched in orbit: Satellite and mission description". In: *Acta Astronautica* 91 (Oct. 2013), pp. 313–321. DOI: 10.1016/j.actaastro.2013.05.011. arXiv: 1305.6823 [astro-ph.IM].

[5] ESA. *GOCE*. URL: https://www.eoportal.org/satellite-missions/goce#eop-quick-facts-section. (last view: 10.10.2022).

[6] David A. Vallado. *Fundamentals of Astrodynamics and Application*. Space Technology Library. Microcosm Press, 2013. ISBN: 9781881883180.

[7] Picone J. M. et al. "NRLMSISE-00 empirical model of the atmosphere: Statistical comparisons and scientific issues". In: *Journal of Geophysical Research: Space Physics* 107 (2002). ISSN: 2156-2202. DOI: 10.1029/2002JA009430.

[8] Gérard Petit and Brian Luzum. "IERS Conventions (2010)". In: *IERS Technical Note* 36 (Jan. 2010), p. 1.

[9] Francesco Santoli et al. "ISA, a High Sensitivity Accelerometer in the Interplanetary Space". In: *Space Science Reviews* 216.8, 145 (Dec. 2020), p. 145. DOI: 10.1007/s11214-020-00768-6.

[10] V. Iafolla et al. "Italian Spring Accelerometer (ISA): A fundamental support to BepiColombo Radio Science Experiments". In: *Planetary and Space Science* 58.1-2 (Jan. 2010), pp. 300–308. DOI: 10.1016/j.pss.2009.04.005.