Gravity field mapping using laser-coupled quantum accelerometers in space

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Abstract
The emergence of quantum technologies, including cold atom-based accelerometers, offers an opportunity to improve the performances of space geodesy missions. In this context, CNES initiated an assessment study called GRICE (GRadiométrie à Interféromètres quantiques Corrélés pour l’Espace) in order to evaluate the contribution of cold atom technologies to space geodesy and to the end users of geodetic data. In this paper, we present mission scenario for gravity field mapping based on a long baseline gradiometer. The mission is based on a constellation of two satellites, flying at an altitude of 373 km, each equipped with a cold atom accelerometer with a sensitivity of $6 \times 10^{-10} \text{ms}^{-2} \tau^{-1/2}$. A laser link measures the distance between the two satellites and couples these two instruments in order to produce a correlated differential acceleration measurement. The main parameters, determining the performances of the payload, have been investigated. We carried out a general study of satellite architecture and simulations of the mission performances in terms of restitution of the gravity field. The simulations show that this concept would give its best performance in terms of monthly gravity fields recovery under 1000 km resolution. In the resolution band between 1000 and 222 km, the improvement of the GRICE gradient approach over the traditional range-rate approach is globally in the order of 10 to 25%.

Keywords Geodesy · Cold atoms · Atom interferometry · Gravity

1 Introduction
The knowledge of the gravity field, which reflects the mass distribution of the Earth, enables to investigate the internal structure of our planet and the dynamics of its external fluid layers: atmosphere, oceans, polar caps, hydrosphere. Observations on the gravity field thus contribute to the understanding of the geological and climatic evolutions of our planet (Tapley et al. 2019). Meanwhile, the emergence of quantum technologies, including cold atom-based accelerometers, offers an opportunity to improve the performances of space geodesy missions (e.g., GOCE, GRACE, GRACE-FO) (Douch et al. 2018). In this context, we initiated an assessment study (Lévèque 2019) called GRICE (GRadiométrie à Interféromètres quantiques Corrélés pour l’Espace) in order to evaluate the potential contribution of cold atom technologies to space geodesy and to the end users of the geodetic data.

The Earth is a complex system composed of an internal structure (core, mantle, crust) and its external fluid layers (atmosphere, oceans, polar caps, hydrosphere). This system is submitted to permanent interactions between its components over different scales of time and space. These mass transfers result in temporal variations of the gravity field, followed since 2002 with a global coverage by the GRACE mission (Tapley et al. 2004). On the other hand, the undu-
lutions of the so-called static field, mapped by the GOCE mission (Drinkwater et al. 2007), up to about 85 km of resolution (Bruinsma et al. 2014) provide information on the internal distribution of the masses of the planet and its evolution at the geological scales. It also provides the equipotential reference surface, called geoid, to which the altitudes and the dynamic topography of the oceans relate. The use of the space gravimetry data is a real challenge because of the wealth of its signal. Indeed, its exploitation requires to identify the contributions of superimposed sources in the measurement of the field. The GOCE mission has shown the relevance of the gravity gradient measurement to identify the different sources based on their geometric characteristics. More generally, the method used for the separation of superimposed contributions to the gravity field is based on the identification of specific spatial and temporal patterns. These are combined to observations and models on the individual components of the system (satellite data of soil moisture, altimetry, soil deformations, etc.). It has been shown that these analyses benefit from an increase in the spatial resolution, particularly for discriminating the deformations of the solid Earth from that of the water cycle in the fluid envelopes. It can also contribute to distinguish the signal of the postglacial rebound from that of the ice melt in the polar areas. An increase in spatial resolution also enables to reduce the “spatial leakage” at the boundaries between areas of different variability, and therefore, to better estimate their mass balances. Thus, the scientific needs for the future space missions, in terms of measurement of the Earth’s gravity field, are first, the continuation of long-term observations, which are crucial for the study of the climate change. Second, it is necessary to increase the spatial resolution of this kind of mission in order to close mass balances at all scales of time and space (Pail 2015). Finally, the scientific community would benefit from a mission determining the temporal variations of the gravity field with increased performances compared to GRACE.

Here, we present a space mission concept for mapping the Earth’s gravity field based on a long baseline gravity gradiometer. The mission uses a constellation of two satellites each equipped with a cold atom accelerometer. A laser link enables to measure the distance between the two satellites and then couples these two instruments in order to produce a differential acceleration measurement. This mission, composed of one pair of satellites, aims to improve the space resolution of Earth’s gravity field. A complete analysis of this concept was carried out in order to determine the performances of the instrument and the main mission parameters. A first satellite layout definition has been realized, and the performances of the mission in terms of restitution of the Earth’s gravity field have been evaluated. A 5-year mission lifetime is envisioned.

The article is structured as follows. In the first part, we present the instrument concept for a future gravity mission which enables to realize a long baseline gravity gradient measurement on a twin satellite constellation. The main performances of the payload are presented. The second part is dedicated to the definition of the mission parameters. We then determine the orbit of the satellites in order to maximize the sensitivity to the gravity field and to obtain an optimal coverage of the Earth. The needs in terms of onboard energy (solar panel, battery) and propulsion are deducted from the orbit. This analysis enables to end up with a preliminary design of the satellite platform. The last section is dedicated to a numerical simulation in order to retrieve the mission performances that could be achieved in terms of restitution of the Earth’s gravity field.

2 Satellite payload

One of the main scientific needs, in terms of gravity field mapping from space, is related to an increase in the spatial resolution of the measurements. Moreover, stability and accuracy would ease long time data treatments and combinations with other data types. Finally, the gravity gradient is the most relevant observable to increase the spatial resolution of the mission as it will be discussed in Sect. 4.8. In this context, the development of cold atom-based accelerometers offers an opportunity to improve the performances of space geodesy missions. Indeed, these instruments provide stable and accurate measurements and are particularly adapted to differential mode operations needed for the gravity gradient measurement (Snadden et al. 1998). In this part, we describe the use of quantum technologies in a specific configuration that would enable an increase in the performances of space geodesy missions.

2.1 General description

Two main space mission concepts have been investigated for mapping the Earth’s gravity field using cold atom instruments (Carraz et al. 2014; Trimeche et al. 2019; Chiow et al. 2015; Hogan and Kasevich 2016; Abrykosov 2019; Migliaccio et al. 2019). The first one, which has been proposed in Carraz et al. (2014) and studied in Trimeche et al. (2019), is based on a 3-axis atomic gradiometer of 0.5-meter baseline on board a single satellite. It consists in a GOCE-like mission at a low altitude of 239 km. This instrument would enable to achieve a sensitivity of 5 mE Hz$^{-1/2}$ and then a twofold improvement on the gravity field recovery for degrees above 50 (Trimeche et al. 2019). The second concept, which has been proposed in Chiow et al. (2015), involves a long baseline gradiometric measurement between two satellites (Keller and Sharifi 2005). A similar twin-satellite concept has also been proposed for gravitational wave detection from space in Hogan and Kasevich (2016).

In this paper, we investigate a twin-satellite mission concept inspired by Chiow et al. (2015). The instrument, depicted
Gravity field mapping using laser-coupled quantum accelerometers in space

Fig. 1 Mission scenario involving twin satellites each equipped with a cold atom accelerometer. A laser telemeter is used to monitor the distance between the two satellites and couples the measurements of the two atom interferometers. In Fig. 1, is based on a composite gradiometric measurement. The mission uses a constellation of two satellites each equipped with a cold atom accelerometer located at its center of mass. A laser link monitors the relative displacement between the two satellites in order to perform a differential measurement between the two distant accelerometers and reject the common-mode noises. The absence of drifts on the atomic measurements allows for the accurate determination of the differential inertial acceleration between the two satellites out of the combination of these accelerometric data. The mission payload is then functionally close to the GRACE and GRACE follow-on missions (Kornfeld et al. 2019). This measurement concept offers the prospect of improved mission performances as it will be shown in Sect. 4.

In this instrument concept, each atom accelerometer measures the acceleration $a_{M1}$ and $a_{M2}$, between an inertial frame, defined by the free falling atomic cloud, and the frame of its satellite, materialized by the retro-reflection mirror of the Raman beam, which is rigidly attached to the satellite. We define the signals delivered by the atom accelerometers $S_1$ and $S_2$ as:

$$S_1 = a_{M1}; \quad S_2 = a_{M2}. \quad (1)$$

The laser link provides a simultaneous telemetry measurement between the two frames of the satellites, i.e., between the two mirrors of the atom accelerometers:

$$L = d_{M1/M2}. \quad (2)$$

Thus, the combination of the two acceleration measurements with the second derivative of the telemetry signal enables to determine the gradient of acceleration between the two distant atomic clouds along the track of the satellites ($a_x$):

$$\frac{\partial a_x}{\partial x} = \frac{1}{L} \left( S_1 - S_2 + \frac{d^2 L}{dt^2} \right). \quad (3)$$

In order to provide the best performances for the complete instrument, the derivative treatment of the telemetry signal has to be included in the data processing as it is crucial to guarantee the coupling between the three instruments. The gravity gradient measurement expressed in Eq. (3) becomes more sensitive when the baseline of the instrument (i.e., the distance between the two accelerometers) increases. This payload configuration, which distributes the atom accelerometers on two satellites, allows a significant increase in the baseline compared to an instrument that would have to stand on a unique platform. In this constellation, each satellite is implemented at low orbit in a nadir pointing mode and then submitted to an orbital rotation $\Omega_{orb}$ of $1.16 \times 10^{-3}$ rad s$^{-1}$ at an altitude of 373 km.

2.2 Atom accelerometers

For this mission, we consider atom accelerometers based on sources of laser-cooled rubidium atoms manipulated by Raman transitions (Fig. 2). Rubidium atoms are first prepared from a vapor to form an ultra-cold sample, using atom chip techniques (Schuldt 2015; Becker et al. 2018; Müntinga et al. 2013). The atomic cloud is then manipulated, so as to move it away from the surface of the chip (Amri et al. 2018) and collimate it via magnetic lensing (Kovachy et al. 2015). Then, the free falling rubidium cloud interacts successively three times with a unique pair of retro-reflected
Raman beams, which acts on matter waves as beam splitters or mirrors. This creates an interferometer of 2T total interaction time in a so-called double-diffraction configuration (Lévèque et al. 2009). This interferometer scheme is particularly adapted to space instruments. Indeed, for inertial sensors in zero-gravity environments, the Doppler effect cannot be used to select one or the other effective Raman transitions in the current retro-reflected configuration needed for accuracy. The atomic phase shift is then obtained from the population in each output port of the interferometer, which is measured by a fluorescence technique. The atomic phase shift $\Delta \Phi$ is given by:

$$\Delta \Phi = \Phi_1 - 2\Phi_2 + \Phi_3,$$

where $\Phi_{1,2,3}$ is the local Raman laser phase seen by the atom along their classical path. This phase shift is then related to the acceleration between the atomic cloud and the equiphases of the Raman laser, along its propagation direction, by:

$$\Delta \Phi = 2k_{\text{eff}}aT^2,$$

where $k_{\text{eff}}$ corresponds to the effective wave vector of the Raman beam. The sensitivity of the acceleration measurement is related to the interaction time ($T$), the fringe contrast ($C$), the duration of the measurement cycle ($t_c$) and the detection noise. In our study, we consider that the determination of the output phase shift of the interferometer is limited by the quantum projection noise (Gauguet et al. 2009). Indeed, other noise contributions can be reduced down to the quantum projection noise limit when using atom interferometers in different modalities (Sorrentino et al. 2014). Moreover, the double-diffraction configuration further suppresses the influence of many systematics and noise sources with respect to single-k diffraction interferometers. The quantum projection noise is linked to the atom number ($N_{\text{det}}$) at the output of the interferometer, so that the acceleration sensitivity ($\sigma_a$) is given by:

$$\sigma_a = \frac{1}{k_{\text{eff}}T^2} \frac{1}{C\sqrt{N_{\text{det}}}} \frac{1}{\sqrt{t_c}}.$$  

The fringe contrast depends on the atomic cloud temperature and decreases with the rotation of the satellite and the interaction time as described in Barrett et al. (2016). Indeed, in the scenario that we consider, the sensitivity axis of the instruments is orientated along the line of sight of the laser ranging instrument, while the spacecraft operates in a nadir pointing mode. The rotation of the instrument during the orbit causes a loss of contrast due to the separation of the wavepacket trajectories and the resulting imperfect overlap during the final light pulse. The contrast is then reduced due to the rotation of the spacecraft in the orbital plane. In the case of an atom interferometer in double-diffraction configuration, this loss of contrast is given by:

$$C \propto \exp \left[ - (k_{\text{eff}} \sigma_r (2T))^2 (\Omega_x^2 + \Omega_y^2) \right]$$

where $\sigma_r (t)$ is the size of the atomic wavepacket given by:

$$\sigma_r (t) = \sqrt{\left( \frac{h}{\sigma_p} \right)^2 + \left( \frac{\sigma_p t}{M} \right)^2} \quad \text{and} \quad \sigma_p = \sqrt{2k_B M \theta}.$$  

In this expression, $k_B$ corresponds to the Boltzmann constant, $M$ to the mass of the $^87$Rb atoms and $\theta$ to the temperature of the atomic cloud. In order to evaluate the atom accelerometer performances needed for this mission, we list the key parameters of the instrument. In Table 1, we report a set of parameters, achievable by existing technologies, which enables to reach a sensitivity on the acceleration measurement of $6 \times 10^{-10} \text{m s}^{-2} \text{rad}^{-1/2}$, where $\tau$ is the averaging time. This sensitivity corresponds to a noise of 2.3 mrad per shot on the phase measurement of the atom interferometer. The interaction time $T$ of 0.5 s is chosen so as to ensure a high sensitivity to accelerations while limiting the loss of contrast induced by nadir rotation. This set of parameters guarantees the existence of a signal at the output of the atom interferometer independently of any rotation compensation system.

### 2.3 Laser ranging instrument

The laser ranging instrument is used to determine the relative displacement of the two atom accelerometers in order to retrieve the gravity gradient measurement along the track of the satellites. As the measurement is realized between the two retro-reflection Raman mirrors of the accelerometers, the gravity gradient measurement extracted from the combination of atomic accelerometer and LRI data is, in principle, insensitive to non-gravitational forces acting on the satellites.
The laser ranging instrument considered in our study is based on a heterodyne optical interferometer working at a wavelength $\lambda$ of 1.5 $\mu$m. The principle of this instrument (Fig. 1) is similar to the one used in the GRACE FO mission (Sheard et al. 2012; Sanjuan et al. 2015). A first laser L1, on board the satellite A, is frequency locked on a spectroscopic reference. This laser is sent to the satellite B after a retro-reflection on the local Raman mirror. On board the satellite B, the laser L1 is received and superimposed to a local laser L2 in order to form a beat note on a fast photodiode at a few tenth of MHz. This signal is used to lock the phase of the laser L2 onto the laser L1. The laser L2 is then sent back to the satellite A where its phase is measured with respect to the local arm of the laser L1 through a beat note technique. The inter-satellite displacement $\delta L$ is derived from the phase shift $\delta \Phi$ of the beat note:

$$\delta L = \frac{\lambda}{2\pi} \delta \Phi$$ (9)

For generating the composite gradient of acceleration, the signals coming from the laser ranging and from the accelerometers have to be recorded simultaneously. The ranging signal is then differentiated twice and filtered taking into account the sensitivity function of the atom accelerometers (Cheinet et al. 2008; Lautier et al. 2014). This treatment is an important aspect which has to be included in the data processing in order to guarantee an optimal coupling between the instruments. This laser ranging system also integrates a steering mirror in order to perform the fine pointing of the beam on the distant satellite (Koch et al. 2018).

In the frame of the proposed mission, the laser ranging instrument has to be able to operate following the relative motion of the satellites along track which corresponds to the ranges reported in Table 2. The performance required for the laser ranging measurement is of 40 nm Hz$^{-1/2}$ in order not to limit the performance of the atom accelerometers as it is shown in Sect. 2.4. This performance is close to the orbit performance of the GRACE-FO mission laser ranging instrument (Abich 2019). Nevertheless, low-frequency increase in the laser ranging instrument noise below 40 mHz is not considered here as the overall noise of the gravity gradient measurement will be dominated by the atom accelerometers contribution in this frequency domain.

In order to guarantee this level of performances, the relative frequency stability of the reference laser has to be better than $10^{-13}$ Hz$^{-1/2}$. This level of performances is currently achievable at $\lambda = 1.5 \mu$m by frequency locking a laser either on a high-finesse Fabry–Perot cavity (Argence et al. 2012) or on saturated absorption spectroscopy in iodine (Philippe et al. 2017).

### 2.4 Instrument performances

The gravity gradient measurement is retrieved from the combination of the acceleration data given by the atomic instruments and of the laser ranging instrument. The laser ranging instrument, which provides a distance measurement, is derived twice in order to determine the relative acceleration between the two satellites. This derivation results in an increase in the measurement noise with the frequency. Finally, the sensitivity of the combined instrument is limited by the atom accelerometers from $10^{-5}$ to $10^{-2}$ Hz and by the laser ranging instrument noise from $10^{-2}$ to 1 Hz. The sensitivity of the instrument to the gravity field gradient is reported in Fig. 3.
Fig. 3 Amplitude spectral density of noise of the complete instrument (red) expressed in sensitivity to the gravity gradient along the track. Individual noise contributions of atom accelerometers (black) and laser ranging (blue) to the complete instrument noise are also represented.

2.5 Impact of the satellite motion on the measurement

In this part, we determine how to guarantee the performance of the constellation when the satellites are in orbit. To do so, we quantify the impact of the satellite pointing errors on the instrument measurement. We also quantify the impact of the nadir rotation on the phase shift of the atom interferometer.

2.5.1 Non-gravitational accelerations

Both satellites are submitted to non-gravitational forces affecting the three directions. These forces are mainly due to drag effect of the residual atmosphere in low orbit. In this part, we evaluate the impact of non-gravitational accelerations on the gravity gradient measurement.

In order to determine the order of magnitude of the non-gravitational accelerations affecting the satellites, we consider the acceleration levels recorded by the GOCE mission. These levels are not directly comparable to our mission scenario as the altitude of GOCE is lower (260 km). Nevertheless, they can be considered as a worse case for our study as GOCE was submitted to a stronger interaction with the residual atmosphere. The typical levels used for our calculations are reported in Table 3.

In the following analysis, we consider that the satellite attitude is controlled within ±4 mrad, while the absolute attitude is measured with a sensitivity of ±0.1 mrad. The performances considered here are those of the GRACE mission (Herman et al. 2004) and are therefore technically very conservative.

When the two satellites are perfectly aligned as shown in Fig. 4, the measurement axis of the laser link is coincident with the atom accelerometers axis. In this case, the non-gravitational accelerations affecting the sensitivity axis are, in principle, perfectly cancelled due to common-mode rejection. However, when one of the satellites is affected by a pointing error, the measuring axis of the atom accelerometer is affected by a projection of transverse non-gravitational forces that are not measured by the laser link and thus induce noise on the acceleration gradient measurement. In Table 4, we report the mean values of the parasitic accelerations induced on the atom accelerometer measurement considering satellite pointing errors of 4 mrad along the three axis of rotation.

In all cases, the parasitic accelerations induced by the non-gravitational accelerations on the atomic interferometer are lower than the sensitivity level of the instrument (6 × 10⁻¹⁰ m s⁻²). These therefore have no impact on the measurements performed by the constellation. The conserva-

| Component | Acceleration mean value [m s⁻²] | Standard deviation [m s⁻²] |
|-----------|---------------------------------|---------------------------|
| ax (along track) | -4.45 × 10⁻⁶ | 1.4 × 10⁻⁶ |
| ay (cross track) | 1.16 × 10⁻⁷ | 1.5 × 10⁻⁷ |
| az (Nadir) | -2.14 × 10⁻⁸ | 1.7 × 10⁻⁸ |

Table 3 Mean values and standard deviation of the non-gravitational accelerations affecting the GOCE satellite along each direction, for the month of April 2012.
The main contribution of the nadir rotation on the atom interferometer phase shift comes from the Sagnac effect induced by the velocity in the $z$ direction. As the sensitivity of the atom accelerometer is at the level of 2.3 mrad per shot, a technical solution for limiting the impact of the rotations on the measurement would be necessary. Several solutions can be envisaged. First, an active control of the center of mass of the satellite would be performed in order to cancel the effect of long-term variations of the relative centering between the center of mass of the spacecraft and the atomic cloud. Second, a high-performance fiber gyroscope would be implemented on board in order to measure the rotation and cancel its contribution on the output phase shift, assuming a simultaneous measurement of the residual atomic cloud velocity. Finally, the Sagnac phase and centrifugal acceleration could be limited by an active rotation of the reference mirror in order to compensate for orbital rotation (Lan et al. 2012). This last solution would in addition restore the full contrast of the interferometer and improve its sensitivity.

### 2.6 Instrument architecture

The atom accelerometer is divided into several subsystems that can be analyzed separately to establish a payload layout.
plan. The function of the different subsystems depicted in Fig. 5 is described below:

The Physics Package (PP) consists of a titanium vacuum chamber in which the rubidium atoms are manipulated. It also includes the pumping system to keep the chamber under ultra-high vacuum, the rubidium dispenser and the coils which enable to generate the magnetic fields during the instrument operation. The physics package is surrounded by two layers of magnetic shields.

The Laser Source (LS) has three main functions during the accelerometer operation: the cold atom preparation, the Raman pulses generation and the atomic state detection. It is made of a main laser subsystem for generating 2D MOT, 3D MOT, selection, detection and Raman beams. It also contains a more powerful auxiliary laser system dedicated to the realization of an optical dipole trap which would be necessary for manipulating and preparing the cold atom sample before the interferometer sequence.

The Microwave Source (MS) generates the 6,8 GHz reference signal necessary for the phase/frequency locking of the lasers. This subsystem also enables to generate the electromagnetic signals for driving microwave transitions during the operation of the instrument. It integrates an ultra-stable oscillator (USO) and a microwave synthesis system.

The Electronics Unit (EU) contains all the electronic subsystems to ensure the operation of the PP. In particular, it includes the ion pump power supply, the magnetic field coil control system, the photodiode acquisition electronics and an accelerometer for hybrid measurements.

The On-board Computer (OC) allows to control the instrument by driving the measurement sequences (atomic cooling, Raman pulses and output phase shift measurement). It also enables to perform the on-board data acquisition and treatment.

In Table 6, we give an estimation of the power, mass and volume budget of the payload on board each satellite.

| Mass (kg) | Power (W) | Volume (l) |
|----------|-----------|------------|
| Physics package | 45 | 8 | 82 |
| Electronics | 5 | 57 | 5 |
| Laser source | 25 | 96 | 33 |
| Microwave source | 7 | 26 | 9 |
| Onboard computer | 6 | 10 | 7 |
| Laser link | 20 | 20 | 8 |
| Total | 108 | 217 | 144 |

3 Space mission concept

The orbit of satellites is determined in order to maximize the sensitivity to the gravity field, via the altitude, while performing an optimal coverage of the Earth. From the orbit, we derive the needs in terms of onboard energy (solar panel, battery) and propulsion for keeping the orbit during the mission. This analysis enables to end up with a preliminary design of the satellite platform.

3.1 Mission analysis

3.1.1 Choosing the orbit

In a low–low satellite-to-satellite tracking mission dedicated to gravity recovery, the baseline of the instrument (i.e., the separation between the satellites) contributes to determine the shortest measurable wavelength of the gravity field. Thus, in our study, an inter-satellite separation of 100 km has been chosen in order to increase the sensitivity to the short-wavelength part. Moreover, so as to optimize the gravity field restitution, the orbit of the two satellites has to fulfill the following requirements:

*Inclination* a full coverage of the ice polar caps is necessary as their gravity observations are of major scientific interest.

*Altitude* the altitude should be as low as possible to maximize the sensitivity of the constellation to the high spatial frequencies of the gravity field while limiting the effect of the residual atmosphere. The altitude must be chosen in the range 320–420 km. The altitude must also be constant. An altitude of 373 km was chosen, based on initial analytical simulations of performance for the recovery of the gravity field.

*Ground track* the orbit is optimized to have an inter-track lower than 10 km and a regular ground track on a weekly and monthly basis. This requires a very long phasing cycle of about a year, with two subcycles: one of about 30 days and the other of about a week. This specification enables to recover a homogeneous ground track densification along the mission lifetime. This homogeneous distribution of the ground tracks allows an optimal gravity map restitution on several timescales.

These constraints set several orbital parameters: an inclination of 90° since the orbit is polar and a frozen eccentricity since the orbit shall be quasi-circular (constant altitude).
Hence, the argument of the perigee is set to 90° and the eccentricity value will be directly determined by the specific altitude chosen. The altitude and the right ascension of the ascending node (RAAN) are determined taking into account the constraints of the mission. The final orbit parameters are indicated in Table 7.

### Table 7 Orbit parameters of the mission

| Parameter       | Value     | Unit |
|-----------------|-----------|------|
| Semimajor axis  | 6751 440.4| m    |
| Eccentricity    | 0.0011    | –    |
| Inclination     | 90        | deg. |
| RAAN            | 0         | deg. |
| Argument of perigee | 90    | deg. |

During the mission, the two satellites will be controlled in altitude and in phasing. The orbital plane will not be controlled as the drifts will remain small. The main change in velocity ($\Delta V$) cost then comes from the effects of atmospheric drag. In this mission, the worst case of area to mass ratio for the satellites is estimated at $2.9 \times 10^{-2}$ m$^2$ kg$^{-1}$ according to the GRACE satellite parameters (Herman et al. 2004). From this parameter, we compute rates of semimajor axis decay between 50 m/day and 300 m/day depending on solar activity. The worst case scenario yields a required $\Delta V$ budget of up to 60 m s$^{-1}$ per year. Over a 5-year mission, this translates into a total budget of 300 m s$^{-1}$. In order to limit the dephasing with respect to the ground track to less than 1 km, these maneuvers must occur, at most in the worst case scenario, every 48 h. This very high frequency of maneuvers leads to envision an autonomous, onboard system. The number of other satellites flying at such a low altitude is limited; therefore, such a system would be satisfying even with debris catalog updated sparsely through mission-driven contacts with the ground.

### 3.2 Spacecraft

In this part, we present an analysis of the spacecraft architecture, including all the subsystems, which was carried out according to the mission scenario.

#### 3.2.1 Impact of mission parameters on satellite architecture

The described space geodesy mission induces significant constraints on satellite architecture. First, the measurements of gravity field require a low-altitude polar orbit which implies a high atmospheric density and then a strong drag that have to be minimized. Second, the cold atom accelerometer must be placed at the center of mass of the satellite which must be kept at the same position throughout the life of the satellite. Finally, all dynamic disturbances must be minimized due to the high sensitivity of the cold atom accelerometer. These constraints have a strong impact on the spacecraft layout and then induce an interpenetration of the payload and the platform that have to be designed together. *Satellite shape* The mission constraints preclude having deployable solar panels, which would inevitably create vibrations on the instrument, so they must be on the spacecraft body. The mission also requires the satellite to be in constant geocentric pointing. As the orbital plane is inertial, the Sun’s angle, with respect to this orbit, rotates of 360° per year. The size of the frontal area of the spacecraft is a direct driver of atmospheric drag. Thus, this surface needs to be minimized and the spacecraft cannot be arbitrarily high or wide. We looked for the best spacecraft shape to accommodate as much power as possible, limiting our search of spacecraft shape to right trapezoidal prism (as the GRACE spacecraft is), with solar panels set on the top face and the two side faces. For a constant frontal area, our numerical simulations have shown that the best shape to collect sunlight is actually to forego the top face entirely and have a right triangular prism. However, such a shape is not optimal for the equipment accommodation. Moreover, this optimal shape is a shallow maximum: still for constant frontal area, flattening the top of the triangle only affect the overall budget by a few percent. The shape of GRACE is actually a very good, albeit not perfect, optimum for maximizing the total amount of energy obtain from Sun.

Over the course of the year, this amount of energy is approximately constant. The solar array efficiency averaged over an orbit is displayed in Fig. 6. The solar array efficiency is the ratio between the installed power and the average power obtained over one orbit. For instance, an efficiency of 25% means that 4 kW of installed power would only generate 1 kW on average over the orbit. Although the RAAN plays a role in such a calculation, because we are at a near optimum configuration, its impact is not significant enough to alter the power budget calculation if changed. We see that, in our situation, the minimum efficiency, i.e., design point, of the solar array is of 23.2% and is obtained neither at the equinoxes or solstices but in between. In other terms, for every 1 W of power required over the orbit, we should install 4.3 W of solar cells.

*Propulsion and attitude control subsystem:* The last two constraints listed above have also an impact on the propulsion and attitude control subsystems. The center of mass position would be modified if using liquid propulsion. The liquid sloshing would also affect accelerometric measurements. So, for propulsion, only two possibilities remain: cold gas and electric propulsion. The trade-off between these two options is realized for the attitude control and for the orbit control...
mode against the $\Delta V$ to maintain the orbit. For the attitude control subsystem, the choice of the actuators is driven by the third constraint. As micro-vibration analysis was not carried out at this stage of the study, two options have been investigated, based on reaction wheels or propulsion. The attitude control subsystem and associated modes are described in “Appendix.”

### 3.2.2 Spacecraft layout

The envisioned platform (Fig. 7) provides all the necessary housekeeping functions to perform the mission: payload support, electrical power, thermal control, command, data handling and storage, attitude and orbit control. The platform structure has a trapezoidal shape. Its size is determined by the amount of energy necessary for the mission ($\approx 500$ W in mission mode, $\approx 800$ W in orbit control mode). Three faces support the solar cells coupled with a battery in order to provide the energy to the platform. A radiator is accommodated on the lower face for thermal control. The different parts of the payload are accommodated on the lower panel near the middle of the platform. The sensor part of the payload is placed at the satellite center of mass and supported by a transversal bench. The instrument, supported by a transversal bench, is installed on board the satellite so as to minimize off-centering between the position of the atomic cloud at the middle light pulse of the interferometer and the center of mass of the satellite. Six center-of-mass trim assemblies (CMT, two per axis) are used to adjust center of mass as needed during the flight. The platform is based on 3-axis stabilization with nadir pointing. The attitude measurement is done by stellar sensor and gyroscopes. For the control, two options are identified: reaction wheels or propulsion (cold gaz) with magnetotorquers. At this stage of the study, the cold gaz is the reference solution with eight thrusters placed on the velocity and anti-velocity faces of the platform associated with three magnetotorquers, one on each axe. For orbit control, two plasmic thrusters are accommodated on the anti-velocity face in order to cover the total $\Delta V$ budget of $300$ m s$^{-1}$. The data handling architecture is based on a central computer (OBC). The housekeeping architecture is based on a central computer (OBC). The housekeeping TM/TC (telemetry/telecommand) and the payload telemetry are realized in S band. For the precise orbit determination, a bi-frequency GNSS receiver and laser reflector are also integrated on the platform. The mass of the satellite is around 900 kg.

### 4 Gravity field restitution

#### 4.1 Simulation procedure

Analytical simulations with dedicated methods can roughly express the expected contribution of a new mission design, but they are often optimistic. In the case of GRICE, many peculiarities intervene in the recovery of the gravity field such as the uncertainties in the ancillary computation models (tides in particular) or the rapid gravity changes in the Earth’s surface layers (atmosphere–ocean interaction) that introduce aliasing phenomena.

This is why we conduct a numerical simulation to better examine the benefit of the GRICE concept in a supposedly realistic environment projected in 10 to 15 years time. In the context of this future mission, the noises impacting the different models are then supposed to be a fraction of the present-day model differences. Moreover, the specified instrumental noises are introduced in the numerical simulation.

The goal is to evaluate the gravity gradient approach which involves two kinds of data: absolute acceleration measurements performed by the cold atoms accelerometers coupled with inter-satellite acceleration measurement obtained from the double differentiation of the inter-satellite laser link range measurement.
The numerical simulation relies on two sets of data: first, a set of “true” data including gravitational accelerations computed from un-perturbed gravity models, non-gravitational accelerations computed from drag, solar and terrestrial radiations in the visible spectrum as well as Earth’s infrared radiations and inter-satellite distance measurements computed from orbit propagated with the set of “true” models; second, a set of “perturbed” data including noisy a priori gravity models, noisy accelerometer and inter-satellites measurements.

4.2 Orbit scenario

The mission orbit scenario has been presented before. It comes under several geometrical and dynamical considerations such as a global Earth’s coverage, a monthly orbit repeatability and an optimal satellite inter-range for gravity field sensitivity. In the frame of the numerical simulation, the inclination is chosen to be 89° in order to keep a small crossing angle of the ground tracks, without losing any signal at the poles. The choice of flying altitude was not challenged during the numerical simulations that were carried out afterward to assess more precisely the performances of GRICE with regards to the gravity field.

4.3 Algorithms

In the case of the GRACE mission, gravity field spherical harmonic coefficients (Bettadpur 2018; Tapley et al. 2004; Flechtner 2010; Dahle et al. 2014) are usually adjusted numerically from the range measurement between satellite B and satellite A provided by the K-band ranging (KBR) instrument. The non-gravitational accelerations measured by the accelerometers onboard each satellite are added to the one of the gravitational models for performing a double integration (using the Cowell method for instance) which provides orbit positions of both satellites. The projected difference on the line of sight (L) can then be compared with the KBR observations taking into account all geometric or physical corrections:

\[
L = ||r_B - r_A||. \tag{10}
\]

with \(r_{A,B}\) being the geocentric distance of the satellites A and B. But the most general approach is to use the time derivative \((dL/dt)\), called K-band range rate (KBRR), in order to reduce observation biases and to increase the gravitational orbit perturbation sensitivity:

\[
\frac{dL}{dt} = \frac{(v_B - v_A) \cdot (r_B - r_A)}{L}. \tag{11}
\]

with \(v_{A,B}\) being the satellite velocities. As one would be able to derive twice the inter-satellite distance with enough precision, it is possible to work at the acceleration level considering bias-free accelerometers to form directly the gravity gradient observable. The range acceleration is then:

\[
\frac{d^2 L}{dt^2} = \frac{(a_B - a_A) \cdot (r_B - r_A) + ||v_B - v_A||^2 - \frac{dL^2}{dt^2}}{L}, \tag{12}
\]

The projection of gravity acceleration differences on the line of sight is equivalent to the gradient of the gravitational potential at the middle of the inter-satellite distance. If one isolates the modelled part of gravity \((a^{GH})\) from the un-modelled part (hydrology, \(a^{GM}\)), it yields:

\[
\frac{(a_B - a_A)^{GH} \cdot (r_B - r_A)}{L} = \left(\frac{d^2 L}{dt^2}\right)^2 + \frac{dL^2}{dt^2} - ||v_B - v_A||^2 \frac{L}{L} - \frac{(a_B - a_A)^{GM} \cdot (r_B - r_A)}{L} \tag{14}
\]

where the different terms represent the second derivative of the inter-satellite distance measurement, the velocity cor-
Table 8 Models used in data simulation (A) and for the gravity field adjustment (B)

| Static gravity field | "True" models used to generate the "true" orbit (A) | Alternative models (B) | A priori models used in the adjustment procedure |
|----------------------|---------------------------------------------------|------------------------|-------------------------------------------------|
| Ocean tides          | FES2014 (20 waves)                                | ITSG-Grace2014         | A+20% (B–A) for degrees 91 to 120 (omission error) |
| Atmospheric pressure | ECMWF/6h                                           | GOT4.8 (20 waves)      | A+20% (B–A)                                      |
| Ocean response       | T-UGOm/6h                                         | MERRA2/6h              | A+20% (B–A)                                      |
| Hydrology            | GLDAS/6h signal completed by GRACE data at high latitudes | OMCT/6h               | A+20% (B–A)                                      |

The gradient approach is promising if the numerical derivation of the inter-satellite distance can be maintained at a low level of noise. In this respect, the noise of GRACE KBR measurements appears to be too high, which explains that the classical KBRR approach is generally used (Goswami et al. 2018). This is no more the case for laser-range interferometry (LRI) which is planned to be used for GRICE, with a noise supposed to be up to 100 times lower. Hence, the GRICE simulations are performed with the gradient approach over 2 years of data (2006–2007 environmental conditions). Monthly gravity field models are computed and compared to a priori hydrology models as defined in the following. However, a comparison with the range-rate approach is worked out as well in order to assess the benefits of the gradient approach.

4.4 Model noise

The simulation of model noise is built from the model differences for ocean tides, atmosphere pressure, ocean response to surface pressure and wind stress, hydrology (Table 8). The full difference of nowadays models (Mayer-Gürr et al. 2018) appears as too large for a mission to be launched in a decade or more, so we consider only 20% of the model differences. Moreover, we amplify by a factor 2 the hydrology signal to be detected because the current hydrological models (GLDAS, WGHM, etc.) seem to be too weak in amplitude compared to GRACE’s solutions (Landerer and Swenson 2012). An example of comparison (in terms of spherical harmonic expansion expressed in equivalent water height), at one particular date, between the hydrology signal and the model errors is shown in Fig. 9.

4.5 Instrumental noise

Instrumental noise is directly applied on the “true” measurement files. As presented in the part 2, we consider that the two types of instruments involved in the measurement (i.e., atom accelerometers and LRI) are white noise limited. The gravity recovery algorithm used for these numerical simulations needs input data with a sampling time of 5 s. As the cycling time considered for our instruments is about 2 s, we consider the following noise levels:

- $2 \times 10^{-8}$ m at 1 $\sigma$ for inter-satellites range measurements. Range-rate and accelerations measurements are derived using a polynomial approach over 100 s in sliding window.

- $3 \times 10^{-10}$ m s$^{-2}$ at 1 $\sigma$ for accelerometer measurements. No calibration parameters are introduced considering the absolute nature of atomic accelerometers. Indeed, atom accelerometers are known for having an excellent accuracy (i.e., stable and well-characterized bias) as shown in Karcher et al. (2018), Louchet-Chauvet et al. (2011). Therefore, neither systematics nor drifts are introduced in the simulation.

Although the gravity field recovery is done with only the knowledge of accelerometer and inter-satellites range data,
it is useful to constraint the absolute 3D-position of GRICE orbit at some centimeter level (i.e., equivalent to GNSS positioning). In fact, we introduce a priori ephemeris data with a random noise of 1 cm (at 1 σ) per coordinate. The double derivation of the range introduces of course some additional noise which is taken into account in the simulation. This noise is compared to the accelerometer noise in Fig. 10. Systematic noises are then automatically introduced through the imperfections of the variable gravity models.

4.6 Methodology

In this work, 730 one-day arcs are computed with the nominal models to generate the “true” measurement files. Once the measurements have been noised and derived, the same 1-day arcs are computed using the alternative models with the following partial derivatives:

- Initial arc parameters;
- Empirical once-per-revolution periodic coefficients in each direction (radial, tangential and normal) per day;
- Gravity field spherical harmonic coefficients from degree 2 to 90.

Normal equations are formed per day and stacked per month. The 24 monthly normal equations are then solved using two different inversion methods. The first one is the Cholesky decomposition without any constraint. The second one is the singular value decomposition method (SVD) eliminating the smallest non-significant eigenvalues in order to stabilize the solutions.

4.7 Results

The inherent problem of the spherical harmonic representation in orbital adjustment is the loss of sensitivity at the higher degrees. A global inversion through the Cholesky decomposition, for instance, is not stable enough and requires some stabilization process afterward. We prefer to use the SVD method which produces cleaner solutions provided that we truncate the number of eigenvalues to the most significant ones. The level of truncation is empirically fixed according to noise and track artifact apparition. The number of linear combinations of Stokes coefficients that are solved is much higher for the acceleration approach, for which 75% of eigenvalues are retained, than for the range-rate approach where, on the opposite, 75% of eigenvalues are eliminated. Figure 11 shows the spectra of the 24 monthly gravity field solutions (in equivalent water height) for both cases, Cholesky and SVD, obtained from acceleration data (in red) and LRI range-rate data (in blue). It appears that the SVD method reduces the noise up to one order of magnitude from degrees 50 to 90 and that the acceleration method gives systematically less noisy results.
However, some artifacts remain in the acceleration solutions as shown in Fig. 12 (top) (modeled hydrology signal for March 2006) and Fig. 12 (middle) (recovered signal after SVD inversion). In fact, it appears that acceleration measurements are less sensitive to very low spherical harmonic degrees. This fact tends to attenuate the long wavelength signal in the solution and generates some artifacts mainly at the poles. If one assumes that the degree 2 to 20 are known by any other type of data (like ranging data), the acceleration method allows to recover almost perfectly the hydrology signal originally introduced (Fig. 12 (bottom)).

Another way to assess the gravity field solutions is to look at the agreement in terms of spherical harmonic coefficients. The bloc diagram of coefficient amplitudes given in terms of

Fig. 13 Hydrological signal to be recovered (a) with respect to spectral degrees between 0 and 90 for March 2006, differences with the adjusted coefficients (b) and with the solution without any noise (c)
equivalent water height compared to the hydrology signal to be recovered is a good visual indicator of the quality of the adjustments. Thus, Fig. 13a shows coefficients amplitudes of the modeled hydrology signal for March 2006 corresponding to Fig. 12 (top). Figure 13b presents the difference obtained after adjustment. It is clear that the low sectorial and near-sectorial coefficients (under degree 20) are not well adjusted and that their errors propagate around order 9. In fact, this is due to the technique itself as it proved by a zero test (Fig. 13c) that consists of adjusting the hydrology signal without any simulated noise either from models or from the instruments.

Compared to current GRACE/GRACE-FO solutions, the combination of a low flying altitude for GRICE and of the gravity gradient approach allows a reduction of amplitude of the degree-error curves in the higher degrees and hence a better spatial resolution. This was verified by comparing the red curve of Fig. 11 (top), with the formal error curves of the ITSG-Grace2018 solution from the Institut fur Geodesie (IfG) of Graz Technical University (TUGRAZ), for the years 2006–2007. We selected this solution because the computation of the formal error for these models is done with extreme care and represents a realistic estimate of the true error of the GRACE solutions (Mayer-Gürr et al. 2018). While the error curves of ITSG-Grace2018 align almost on a straight line (in a linear-log plot) between 1 mm EWH at degree 3 and 30–40 cm EWH at degree 90, with an intermediate point at 2 cm EWH at degree 45, it can be seen in Fig. 11 that the error curves of GRICE display a lower slope after degree 45, ending up at 7–8 cm EWH at degree 90. As a result, GRICE allows a better determination, by a factor ~5, of the shorter wavelengths of the gravity field than GRACE/GRACE-FO.

### 4.8 Comparison between acceleration and range-rate methods

All computations presented here have been done without empirical periodic parameter adjustment. In that case, the acceleration method applied from degree 2 to 90 performs much better than the range-rate method as described in Table 9 (up to almost 5 times). However, introducing once-per-revolution parameters per day in the adjustment reduces the difference to 8% between both approaches. The results are approaching the level when no instrumental or model errors are introduced.

Nevertheless, the improvement is not global. As depicted before, the acceleration method tones down the low degrees compared to the range-rate method usually applied to the GRACE KBR data, that is, indicated in Table 10 when splitting the harmonic decomposition in 3 brackets of low degrees (2 to 20), medium degrees (21 to 60) and higher degrees (61 to 90).

### 4.9 Contribution of atom accelerometers to the mission performances

As presented in Sect. 2, the use of cold atom accelerometers is required for retrieving a long baseline gradiometric measurement. Indeed, two main issues can arise when coupling the instruments to retrieve the gravity gradient measurement. The first one is the imperfect rejection of common-mode noises, and the second is the knowledge of the bias affecting each accelerometer and their drift over time. In this work, we highlight two main aspects of this mission concept that ease the coupling between these instruments. First, the use of common mirrors between the laser link and the atom accelerometers ensures a high rejection of common-mode noises. Second, atom accelerometers are known to have an excellent accuracy (i.e., stable and well-characterized bias). This characteristic eases the combination of the two measurements. Therefore, the gradient of acceleration will not be affected by any significant bias, specially over long timescales. Thus, the performances of the atom accelerometers, which provide a low-bias and low-drift measurement, are the keystone of this gradient approach.

In order to point out the interest of atom accelerometers in the specific case of this mission scenario, we run a simulation to determine the performances which would be achieved by a drifting accelerometer in the same mission scenario. In particular, we focus on the impact of long-term stability, in the 10−6 Hz to 10−5 Hz frequency band, on the gravity restitution. Indeed, accelerometers used in space missions generally exhibit an increase of the noise at low frequency, induced by thermal effects on the system (Bergé et al. 2013), that could be avoided using atom accelerometers (Lautier et al. 2014). The methodology used for this simulation consists in adding drifts in the instrumental noise considered at the input of the numerical calculation. This approach is a simple method for modeling the behavior of these instruments in the spectral band of interest. Therefore, this simulation enables to retrieve the gravity field in the same conditions for both a drifting accelerometer and quantum technologies.

In order to simulate the performances of a drifting accelerometer, a drift and a periodic signal have been introduced on the accelerometric measurements (different on each satellite). When considered on a few days timescale, a typical drift of 10−9 m s−2 per day has been added. We also introduce a periodic signal at 10,800 s with an amplitude of about 3 × 10−10 m s−2, corresponding to a typical perturbation related to the orbital period.

All computations were carried out without any empirical periodic parameter adjustment in the gravity field restitution. This approach enables to compare the technologies with the same level of signal post-processing. Indeed, the adjustment of empirical parameters, which generally limit the effect of
Table 9  Impact of adding once-per-revolution acceleration parameters in radial/tangential/normal directions on range-rate and acceleration residuals worldwide, expressed in cm of EWH for spherical harmonics (SH) 2 to 90

|                                | Global RMS error (cm of EWH) | Range rate (SH 2–90) | Acceleration (SH 2–90) |
|--------------------------------|------------------------------|----------------------|------------------------|
| Without any additional parameters | 21.26                        | 4.35                 |
| With once-per-rev. parameters   | 3.98                         | 3.63                 |
| Without any instrumental or model noises | 3.40                        | 3.12                 |

Table 10  RMS errors by brackets of spherical harmonic expansion over oceans, continents and global for both adjustment types in terms of range-rate and acceleration measurements

| SH degrees | Oceans (cm) | Continents (cm) | Global (cm) |
|------------|-------------|-----------------|-------------|
|            | Range rate  | Acceleration    | Range rate  | Acceleration |
| 2–20       | 1.36        | 1.34            | 2.50        | 2.90        |
| 21–60      | 1.79        | 1.22            | 2.71        | 2.16        |
| 61–90      | 2.12        | 2.16            | 3.61        | 3.02        |
| Global: 2–90 | 3.00       | 2.79            | 5.26        | 4.84        |

instrumental drifts, is also likely to affect the gravitational signal.

Figure 14 exhibits the error spectra of the gravity field solution in equivalent water height for both quantum and drifting accelerometers. We also report the gravity maps related to each case. It appears that the quantum accelerometers enable to reduce the noise up to a factor of 5 from degree 50 with respect to the drifting accelerometers. The global RMS error drop from 23.64 cm of EWH for the drifting technology to 6.72 cm of EWH for the cold atomic sensor, improving the quality of the gravity map restitution.

Further simulation also shows that the gap between these two results is reduced by adjusting empirical parameters in the gravity field restitution. However, this study highlights that the long-term stability of accelerometers is an important parameter for space geodesy missions. Moreover, atom accelerometers need no calibration of the scale factor which is perfectly known and stable over time. Thus, these characteristics would enable to get rid of empirical parameters during data processing and thus potentially improve the quality of the gravity signal restitution.

5 Conclusion

We carried out an assessment study of a space mission for gravity field mapping involving innovative atomic technologies. The related mission scenario implies a constellation of two satellites each equipped with a cold atom accelerometer placed at their center of mass. A laser link measures the distance between the two satellites and couples these two instruments in order to produce a correlated differential measurement. This instrument enables to retrieve a long baseline gradiometric measurement along the track of the two satellites which is likely to increase the resolution of the high harmonics of the Earth’s gravity field. The main parameters of the instruments have been estimated in order to determine the final performances of the payload. To ensure an optimal temporal resolution, the mission’s orbit has been optimized to provide a consistent coverage over different timescales. Finally, a preliminary design of the platform has been realized. This design integrates all the subsystems which enable the satellite to operate at low orbit during 5 years and demonstrates the technical feasibility of this concept.

Further work has been performed in order to simulate the gravity field recovery. For this, a specific data processing method has been developed taking advantage of the correlation between the two atomic accelerometers by a gradient approach. The simulations have shown that, with the given level of instrument specifications and with the chosen model type errors, this method would be able to give its best performance in terms of monthly gravity fields recovery under 1000 km resolution. In the resolution band between 1000 and 222 km, the improvement of the GRICE gradient approach over the traditional range-rate approach is globally in the order of 10 to 25%. An alternative method would be necessary for the lowest degrees either in iteration or in combination with a KBRR-type determination, for an optimal recovery of the time variable gravity signal. The use of GNSS-type inter-satellite range data for determining the low degree coefficients might be the best complementary approach.

This study highlights the potential of atom accelerometers in space for Earth’s science applications. In particular, it shows that long-term stability and accuracy specific to these instruments could improve the determination of the Earth’s gravity field. Such improvement of the gravity map resolution would pave the way for new fields of applications in hydrology, glaciology, oceanography and internal geophysics. Indeed, it would enable a better monitoring of mass...
displacements in both internal structure and external fluid layers and then contribute to a better understanding of the Earth’s system and, in particular, its global climate change.

To a greater extent, the mastery of atom accelerometers in space will become a determining scientific and technical stakes in the next decades. In this context, other work will be conducted at CNES in order to study a pathfinder mission for this type of instrument. This new proposition of mission, called CARIOQA (Cold Atom Rubidium Interferometer in Orbit for Quantum Accelerometry) will demonstrate the utilization of a quantum inertial sensor in space paving the way for future space applications in Earth’s science and fundamental physics.

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Author contributions TL and CF conducted the study and wrote the paper; MM defined the general scientific scope of the study; FPDS, PB and BB determined the performances of the atom accelerometers; ST performed the optimization of orbital parameters; SD and AP designed the satellite platform; and RB, JML and SB conducted the simulations of Earth’s gravity field recovery.

Data Availability Statement All data sets are available upon reasonable request from the authors.

Appendix: Attitude control subsystem

The mission definition and constraints have a strong impact on the attitude control system (ACS) architecture. The nadir pointing shall be maintained within 4 mrad and 50 μrad/s. This nonetheless ensures a correct alignment of the payloads of the two spacecraft with respect to one another, but it also minimizes the impact of air drag (and thus increases the mission lifetime). ACS shall also provide the estimated attitude to the instrument with a knowledge error lower than 0.1 mrad. Such requirements are not particularly stringent, but due to the low altitude of the orbit and the elongated shape of the spacecraft, the impact of orbit-related perturbations has to be carefully considered in the ACS design. Indeed, with such a long shape, the air drag and gravity gradient torques are strongly dependent of the pointing error. The spacecraft layout also plays an important part in the air drag torque: Any offset between the center of mass and the aerodynamic center increases the aerodynamic torque linearly. Moreover, in addition to the attitude control during the measurements sessions, ACS shall ensure a correct pointing during the orbit correction maneuvers and in case of failure. To reply to all these considerations, ACS is defined to provide three-axis stabilized Earth-pointing attitude control during all mission phases. Thus, ACS architecture is composed of a set of equipment and onboard software used within the control closed loop. Sensors are used to measure the attitude and actuators to correct it. Estimation and attitude control algorithms are implemented onboard to compute the estimated attitude and the control law. Three ACS modes are defined depending on the phase of the mission described below:

Mission mode This is the mode during which the mission is carried out. In this mode, the control loop uses measurements from a 3-head Star-Tracker (STR) and a 3-axis optical gyroscope to estimate the attitude. As the orbital plane is inertial, a 3-head STR is needed: The satellite-Sun angle varies along the mission (which induces successive dazzling of each panel of the satellite), and the heads are thus set on the lateral and upper panels to ensure at least one available STR. Both measurements are hybridized in a gyro-stellar filter to compute the attitude.

Concerning the actuators, in order to minimize the perturbations toward the payload, two options are considered to realize the commanded torques: a set of 4 reaction wheels in pyramidal configuration (to ensure the torque capacity and a redundancy in case of failure) or a cold gas propulsion subsystem (CGPS). The CGPS configuration is composed of
4 × 2 thrusters on the front and back sides of the satellite. Magnetotorquer bars (MTBs) are also used in both options: either to unload the wheels or to help the CGPS to compensate for the Earth’s magnetic torque.

Safehold mode This mode is enabled just after separation to stabilize the satellite (inertial acquisition) and in case of failure detection or attitude loss. The satellite remains in geocentric pointing to minimize air drag perturbations, but with relaxed pointing and stability performances. The attitude estimation uses a set of Sun sensors and magnetometers. Six Coarse Earth Sun Sensors (one head on each of the six sides of the satellite) are used to give omnidirectional and coarse attitude estimation. In addition, 2 fine Sun sensors located on the lateral sides are used to provide a precise attitude once the satellite is stabilized. Two magnetometers are used to estimate the angular rate. The 3-axis stabilization is performed by MTBs.

Orbit control mode During the orbit control maneuvers, the thrust direction has to be maintained along track to counter air drag effect. Consequently, the satellite has to remain in nadir pointing. The orbit control propulsion system is not suited to realize attitude control by off-modulation. ACS uses the same sensors and actuators as the mission mode (star tracker and gyroscope as sensors, and reaction wheels or CGPS as actuators), while the thrust is performed by electric propulsion. During this mode, no scientific measurements will be realized.

In this early phase of the project, particular attention is paid to the dimensioning of the actuators (MTBs, reaction wheels, CGPS). Indeed, the satellite layout is strongly dependent of the size and mass of the actuators. The dimensioning process consists in checking that the capacities of the chosen actuators are compliant with the main perturbations encountered for each mode.

In Safehold mode, MTBs have to deal with potential huge pointing errors and thus high aerodynamic and gravity gradient torques. Nevertheless, in Mission and Orbit Control modes, the main constraint is the unload duration of the reaction wheels that has to be lower than a quarter orbital period.

For the reaction wheels and the CGPS, the dimensioning constraint comes from the orbit control maneuvers. Indeed, the thrust torque is directly linked to the offset between the center of mass and the aerodynamic center of the platform: With a 300 m/s ΔV budget over 5 years shared with maneuvers every 48 h, each 2h-spread tangential thrust generates a constant 2.5 mN m torque in the case of 50 mm offset (in pitch/yaw). Moreover, concerning the reaction wheels option, an unloading by MTBs during the maneuver is necessary to be compatible with the wheels’ angular momentum. Concerning the CGPS option, it is also important for the satellite layout to estimate the gas consumption during both modes: With the considered hypotheses (5-year mission, 50 mm offset, 55 s Isp, 1000 kg satellite mass), the foreseen gas consumption is 35 kg in Mission mode and 15 kg in Orbit Control mode. This consumption is a major factor in the trade-off between the two options.

The trade-off between reaction wheels and cold gas propulsion system is not completely done at this stage of the project. Both solutions are compatible with the mission needs. Reaction wheels are often used as main actuator in ACS, mainly because they do not need any propellant. They also provide the satellite attitude control system with a high torque capacity. However, they induce micro-vibrations which may be incompatible with atomic accelerometer measurements (the effect of micro-vibrations on the instrument has not been characterized yet). On the other hand, CGPS certainly induces a gas consumption which has to be taken into account in the satellite layout, but the feedback of high precision missions such as microscope (Delavault et al. 2019) or GAIA showed a very satisfactory level of performance well suited to GRICE needs. At this stage of the study, the CGPS is then the reference solution for the satellite layout.

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Gravity field mapping using laser-coupled quantum accelerometers in space

15

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