A non-drag-free gravitational wave mission architecture

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Abstract

The architecture for the LISA gravitational wave mission is based on three spacecraft separated by 5 million km in heliocentric orbits chosen such that the spacecraft form a triangle inclined to the ecliptic by 60°. The spacecraft use laser interferometry to measure changes of distance between spacecraft due to gravitational waves and use drag-free systems to reduce spacecraft acceleration noise due to non-gravitational effects. The use of three spacecraft is required to reduce laser frequency noise to an acceptable level. Here we present an alternative concept with three spacecraft separated by 260 million km with orbits in the ecliptic plane and no drag-free system. With suitable calibration this architecture may be able to recover a significant fraction of the LISA science goals without the need to develop and test drag-free technology.

1. Introduction

Gravitational waves are predicted by Einstein's theory of general relativity. Gravitational waves have not yet been directly detected though with recent advances in ground based-detectors such detection may occur in the coming decade. The LISA mission was first proposed in 1993 with the goal of detecting gravitational waves from known sources and a wide variety of expected sources in a frequency band complementary to ground-based detectors.

Gravitational waves causes a change in distance (or light-travel time) between two objects. The displacement is proportional to both the amplitude of the gravitational wave and the average separation between the objects. The accuracy of measurement of light-time between two objects is limited by the stability of the clock (time standard) used. Because gravitational wave signal amplitudes are very small, even the best clocks are not stable enough to detect gravitational waves from expected sources. Instead the difference in distances from one object to each of two other objects can be measured, with the clock noise largely canceling in the difference measurement.

The LISA architecture [Danzmann et al. 1997] uses three spacecraft in heliocentric orbits forming an equilateral triangle with lengths of 5 million km inclined 60° to the ecliptic, rotating about the center once per year. This formation is achieved by careful selection of the eccentricity and inclination relative to the ecliptic for each spacecraft.

The acceleration noise goal for LISA is 3×10^{-15} m/s^2/√Hz, which is far lower than the 1×10^{-10} m/s^2/√Hz achieved for the accelerometers for the GRACE mission [Tapley et al. 2004], and the 1×10^{-11} m/s^2/√Hz differential acceleration noise achieved on the GOCE mission [Albertella et al. 2002]. This goal for acceleration noise is far enough beyond what has currently been demonstrated that several committees reviewing the LISA mission have recommended a precursor mission to demonstrate the capability before proceeding with the LISA mission. This has led to the ESA LISA Pathfinder mission currently under development [McNamara et al. 2008].

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The drag-free technology requires development of both a gravitational reference sensor, which provides a test mass shielded from external effects and a means of measuring separation of the test mass from the spacecraft, and micro-propulsion thrusters capable of controlling the spacecraft position with noise power spectral density of ~10 nm/√Hz. Both of these technologies have had development challenges, and place significant design constraints on the rest of the spacecraft.

We suggest here an alternate architecture with larger separation between spacecraft and no drag-free system employed. A natural choice for a larger spacecraft separation is made by forming an equilateral triangle about the Sun in the plane of the ecliptic with the spacecraft one astronomical unit from the Sun. This geometry has been suggested before in concepts including a drag-free system and targeting a lower signal frequency than for LISA [Cornish et al. 2002; Husa et al. 2000]. Without a drag-free system this geometry appears suitable for many of the LISA science goals with a simpler space system.

Assuming an acceleration noise 50 times higher than for LISA, and arm length 50 times longer with the same laser power and telescope diameters, the non-drag-free architecture with the same laser and optics as for LISA results in a sensitivity about a factor of 4 higher (worse) than for LISA at signal frequencies below 2 mHz and with further reduced sensitivity at higher frequencies, as shown in Figure 1. Also, confining the spacecraft to the plane of the ecliptic provides less ability to determine the direction to sources than the LISA geometry provides [Moore and Hellings, 2002]. This reduction of sensitivity is the cost of simplification resulting from dropping the drag-free system.

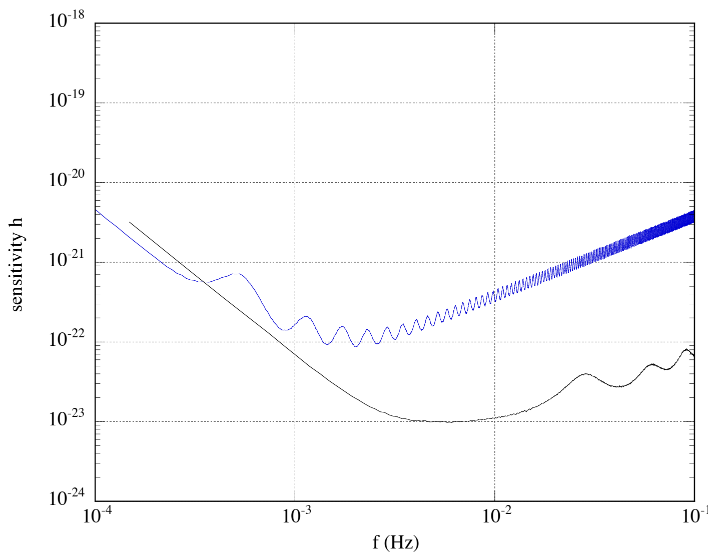

Figure 1 – Gravitational wave sensitivity for one year integration, averaged over polarization and direction, and signal-to-noise ratio of five. Lower (black) curve is for nominal LISA parameters (from Danzmann et al. 1997). Upper (blue) curve is for the non-drag-free case.

The sensitivity indicated for the non-drag-free configuration is sufficient to observe coalescence of massive black holes with masses greater than 10^6 $M_\odot$ with a signal-to-noise ratio comparable with that of LISA, since such coalescences occur at low enough frequency for LISA sensitivity to be limited by galactic binary confusion noise [Vecchio, 1997; Bender and Hils 1997]. Coalescences of massive black holes with lower masses will be observed with high SNR but reduced with respect to LISA by the reduced sensitivity. The sensitivity shown in Figure 1 is sufficient to detect many binary sources within our galaxy. Capture of stellar-mass compact objects by massive black holes would probably not be observable with this sensitivity. We note that a sensitivity very close to the LISA sensitivity could be achieved by increasing the effective laser power above the nominal level assumed for LISA, but such an improvement would also have a corresponding improvement for LISA so is not considered further here.

The gravitational-wave sensitivity shown in Figure 1 is based on an assumption of acceleration noise of 1.5×10^{-13} m/s^2/√Hz without a drag free system. This is three orders of magnitude lower than expected for the GRAIL mission to the Moon [Zuber et al. 2008], which uses two spacecraft with a microwave ranging system to measure the lunar gravity field. The similar system at Earth for the GRACE mission uses an accelerometer to measure atmospheric drag fluctuations with a noise of 1×10^{-10} m/s^2/√Hz. For GRAIL the leading acceleration noise is due to variations in solar luminosity. Extensive studies for GRAIL have been done to provide evidence that thermally-induced variations and other forces are not larger than the solar variations. As shown below it appears possible to calibrate the solar luminosity variations and other forces well enough to reach the sensitivity level shown in Figure 1. Several tests of other noise sources would have to be done to provide confidence that they would be small enough to reach the target level.

The laser interferometry for the non-drag-free architecture is simpler in most ways than for LISA. With a 50 times longer arm length, the displacement noise requirement is 50 times larger than for the LISA mission. Thus dimensional stability and thermal stability requirements on the spacecraft are significantly reduced. However the received light power is 2500 times lower than for LISA, so there are increased requirements on photo-diode preamplifiers and signal tracking. Preliminary orbit estimates show Doppler shifts due to arm-length variation and change in pointing angles are comparable with those for LISA.

Below we describe some assumptions on the mission deployment, propulsion system, and acceleration noise sources, and indicate areas that would need further study.

2. Mission Description

A candidate spacecraft orbit configuration is indicated in Figure 2. One spacecraft leads the Earth by 45°, the second leads the Earth by 165°, and the third completes the equilateral triangle. While a trajectory design has not been done, the three spacecraft might be launched on a single rocket to a very high Earth-centered trajectory as with the GRAIL mission. Spacecraft 2 would fly by the Moon is such a way as to get a gravity assist to speed ahead of the Earth. Spacecraft 3 would fly by the moon two weeks later to get a gravity assist to slow it relative to the Earth. Each spacecraft would need propulsion to complete its trajectory and change its velocity to co-orbit the Sun with the same angular velocity.

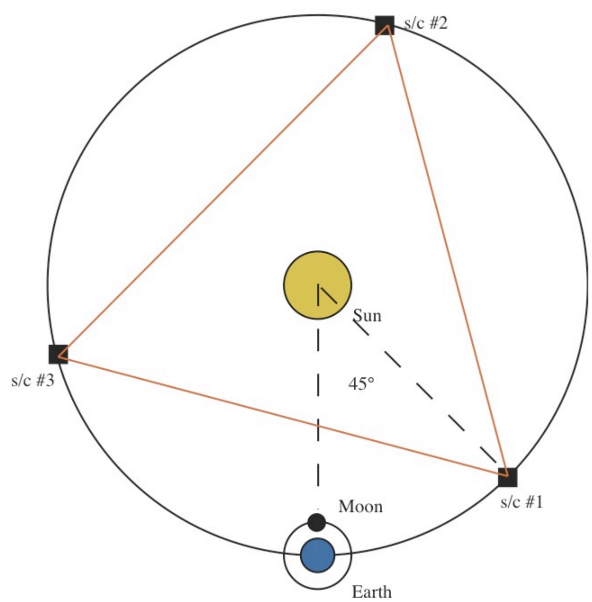

Figure 2: Spacecraft orbit configuration

Once in the correct orbit, no further deterministic maneuvers would be needed. While any propulsion system could be used, there are several reasons to use a electric propulsion system with a gas propellant such as Xe as used in several planetary missions. Since acceleration must be minimized in the final orbits, the propulsion system must be completely turned off. This can be done by venting the propellant so that, once completely vented, no further outgassing takes place. Venting of a chemical propulsion system might have some propellant freeze when vented with the frozen propellant providing a source of a long term evaporation higher than the allowed acceleration noise goal. An electric propulsion system also has much lower mass than a chemical propulsion system, which is an advantage for launch.

Attitude control would be done using reaction wheels. Because the spacecraft will maintain a fixed attitude with respect to the Sun during operations, it will be possible to design the solar panels to minimize torque on the spacecraft. The reaction wheels will still saturate occasionally, perhaps once per month requiring continued use of small thrusters.

A preliminary orbit design has been performed to estimate the rates of changes of the lengths of the arms and the subtended angles due to the influence of the Earth and other planets. The rates of change of up to about 4 m/s are about a factor of two less than the rates of change of the LISA arm lengths. Because of the lower rates of change of arm lengths and the longer arm lengths, the angles between any two arms vary much less than the ~±0.5° variation of the LISA triangle.

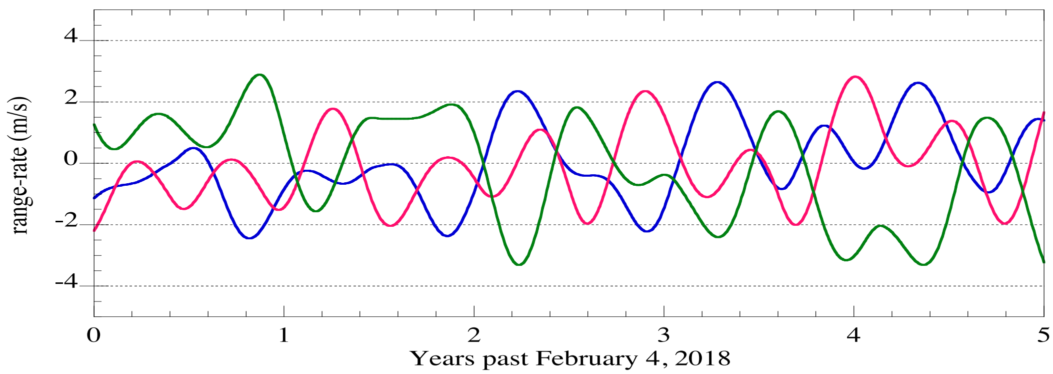

Figure 3: Rate-of-change of arm lengths

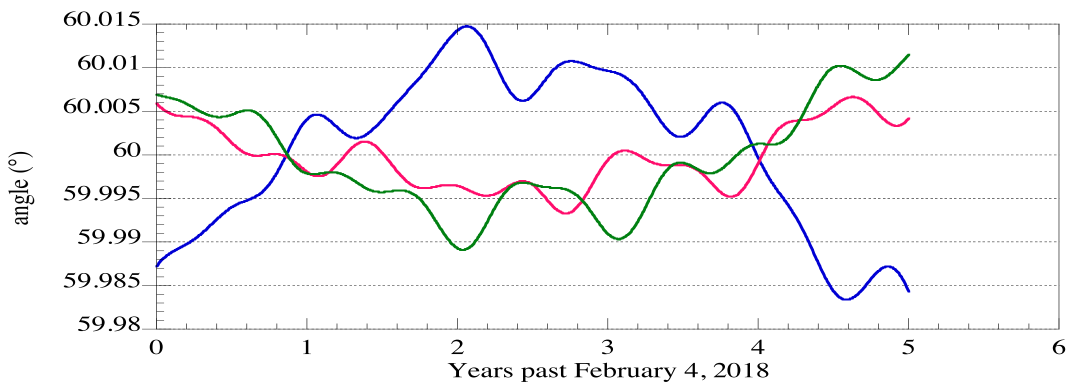

Figure 4: Angle between arms

3. Acceleration Noise

The largest force acting on the spacecraft after propulsion is turned off is due to absorbed and reflected sunlight. A constant force can be removed from the observed data but variations in force will show up as noise in the gravitational-wave measurement. Figure 5 shows the estimated power spectral density of the leading acceleration noise sources. Each is discussed in the following sections.

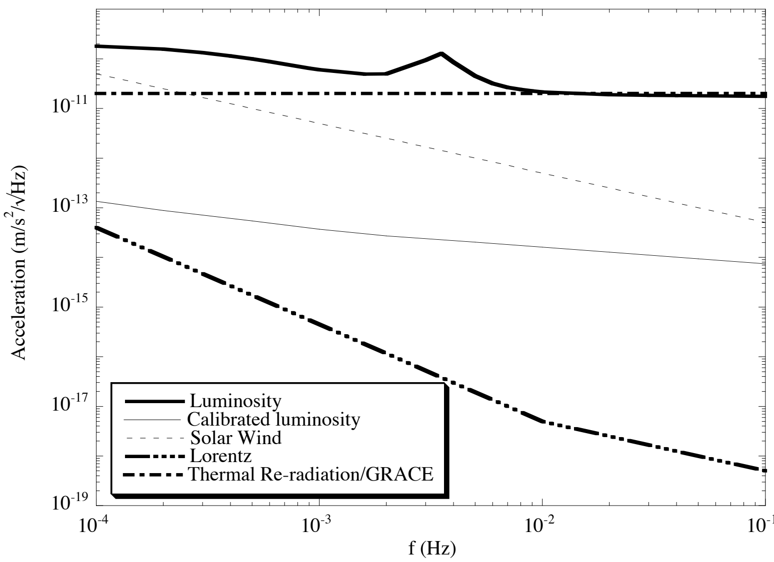

Figure 5: Major acceleration noise sources before and after calibration

3.1 Solar Luminosity

The largest acceleration noise shown in Figure 5 is from variation in solar luminosity. The estimated power spectral density is based on a time series of luminosity measurements [Pap et al. 1999] and the assumption of a spacecraft area of 3 m^2 with half of incident sunlight absorbed and half reflected, with a spacecraft mass of 300 kg. If the solar luminosity, with a nominal value of 1350 W/m^2 was measured with a fractional stability achieved by a thermally stabilized voltage reference [Fleddermann et al. 2009], the acceleration noise due to luminosity variations would be reduced to below 10^{-13} m/s^2 as indicated in Figure 5. Variations in response from a suitable photodiode has not been specifically measured and so could cause an increase in noise. The observed noise in the Virgo radiometer that flew on the SOHO mission [Appourchaux et al,. 1997] had a power spectral noise approximately a factor of ten larger than shown.

3.2 Charged particles

Charged particles in the solar wind cause a total force on the spacecraft much smaller than the solar luminosity but with larger fractional variations. Figure 5 shows an estimated power spectral density of acceleration noise from charged particles measured on the ACE spacecraft [McComas et al. 1998]. The acceleration noise estimated needed to be reduced by a factor of one hundred through calibration by measurement on the spacecraft. The SWEPM instrument used on ACE has suitable measurement accuracy and is fairly small (<3 kg). However the charged particle force varies in direction. The SWEPAM instrument relies on rotation of the ACE spacecraft to provide variable sensing direction of the charged particle flux. So suitable calibration of solar wind will require a somewhat more complex instrument to determine charge particle directions on an inertially pointed spacecraft needed for gravitational-wave detection.

3.3 Thermal Radiation

Since the orientation of the spacecraft relative to the Sun will be constant, temperature variations will be dominated by fluctuations in the solar luminosity. This will cause the temperature of the solar panels to fluctuate. The effects could be calibrated using a precise thermometer, or reduced by actively controlling the temperature of the solar panels. The temperature of other panels will remain stable if the solar panel temperature is controlled and the electrical power dissipation of the spacecraft is sufficiently regulated. Figure 5 shows a power spectral density of acceleration based on measured temperature variations from one point on the GRACE spacecraft that uses active thermal control on various elements of the spacecraft, and assuming a 300 kg spacecraft with area 3 m^2. This needs to be reduced by a factor of one hundred for gravitational-wave detection. GRACE thermal control is based on a temperature resolution of only 0.1 K while many sensors exist with 1 mK resolution.

3.4 Lorentz acceleration

A typical spacecraft in orbit about the Sun at distance 1 AU will have charge build up to the point where it has a potential of about 30 V, where emission of electrons released by sunlight balances inflow from the solar plasma [e.g. Torkar et al. 2001]. The charged spacecraft moving through the variable solar magnetic field will experience a variable Lorentz acceleration. Taking the spacecraft capacitance to be that of a sphere 3 m in diameter at a potential of 30 V and magnetic field variations from the ACE magnetic field instrument [Smith et al. 1998] the estimated acceleration noise is shown in Figure 5, which is low enough that no calibration is needed.

3.5 Other noise sources

Other sources of acceleration noise are expected to be small enough to not require calibration, but specific measurements are needed to confirm acceleration noise levels. The acceleration noise for the warm gas thrusters used on the GRAIL mission are expected to be below the acceleration assumed for this mission concept based on measured leak rates [N. Doha, private communication]. Those thrusters might be suitable, or pulsed plasma thrusters such as used on the Earth Observing 1 mission [Zakrzwski et al. 2002] might be preferred as not using any fluids. The materials used for spacecraft construction can have significant outgassing, but should reach an acceptable level after several months in space, once the desired orbits are reached. The interaction of the fluctuating charge of the spacecraft times the average magnetic field should be less than the Lorentz effect, as should be the interaction of magnetic materials on the spacecraft with the variable solar magnetic field, but these need to be confirmed for the materials chosen.

4.0 Interferometry

We assume an interferometer system similar to that from the LISA Pre-Phase A report [Bender et al. 1995] in order to point out differences solely due to the arm length. Thus we assume 1 W laser power, 30 cm diameter transmit/receive telescopes, and an optical efficiency of 0.3.

4.1 Shot noise

For LISA the assumed laser power, telescope diameter, and optical efficiency give a received light power of 200 pW and a distance measurement limited by laser shot noise of 11 pm/√Hz.

For the architecture discussed here, the longer arm lengths result in a received power of 80 fW and a distance measurement limited by shot noise of 550 pm/√Hz, since the received power scales as the inverse square of the arm length and the shot noise scales as the square root of the received power. The received power is much lower than typically used for closed-loop operation. However the signal-to-noise ratio is still quite high compared with spacecraft radio tracking. The ratio of signal power to shot-noise is 50 dB Hz, which is comparable to the GRACE mission microwave ranging signal-to-noise ratio, and much higher than the 15 dB Hz level used reliably for planetary spacecraft tracking by NASA's Deep Space Network. Closed-loop tracking of coherent lasers for optical communication purposes at levels as low as 40 fW has been demonstrated in laboratory tests [Dick et al. 2008].

The low received power will place more constraints on other sources of measurement noise such as dark current and amplifier noise than in the case for LISA, and possibly more constraints on filter design.

4.2 Down-conversion noise

With rate of change of arm lengths of up to 4 m/s, assuming an ultra stable oscillator with Allan deviation of 1×10^{-13} at time scales of 1000 s, the noise introduced by down conversion of the Doppler shifted laser signals is about 1700 pm/√Hz at a signal frequency of 1 mHz. This is three times the shot noise level so would require a clock calibration scheme using side tones on the laser beams as for LISA [Hellings et al. 1996] to be smaller than the shot noise.

It might be possible to find alternate orbits, or might be acceptable to perform maneuvers once or twice per year to significantly reduce the Doppler shifts and possibly eliminate the complexity needed for the clock calibration.

4.3 Optical path length

Since the distance measurement goal is 50 times less sensitive that for LISA, comparably larger thermal distortions can be allowed. This might allow for use of materials with higher thermal expansion coefficients and/or reduction in thermal control requirements to reduce effects due to optical path length changes within the spacecraft.

5. Technology development

The mission concept described is based on existing capabilities but extrapolating performance to a lower acceleration noise and lower light levels than previously required.

The major technical risk is judged to be calibration of the solar wind magnitude and direction, which may require significant modification from existing instruments. Calibration of solar luminosity variations with ten times greater accuracy than currently obtained in space will require laboratory verification of the stability of photodiode response. Measurements of thruster and spacecraft material leak and outgassing rates to the required levels are needed but based on measurements for GRAIL are expected to require no new developments.

The received light level expected here is much lower than for LISA, and while some laboratory tests have shown basic feasibility, much more work will be needed to demonstrate performance in a relevant environment. Any laser interferometer mission requires a moderate power laser. While 1 W lasers for space are commercially available, the supply is limited and new technologies for alternate lasers are attractive. Development of a laser with >2 W power at a wavelength shorter than the 1064 nm currently available would be advantageous, since the received light power is proportional to the wavelength to the 5/2 power. A shorter wavelength laser than 1064 nm may also be suitable for locking to an atomic line which would significantly simplify initialization of the measurement process.

6. Cost estimate

The cost estimate presented in this document does not constitute an implementation-cost commitment on the part of JPL or Caltech. The estimate was prepared without consideration of potential industry participation and derived using a combination of parametric estimates and analogies to comparable historical mission actual costs. The accuracy of the cost estimate is commensurate with the level of understanding of a Pre-Phase A mission concept and requires more engineering work to be done on the payload, trajectory design, and propulsion subsystem.

A constellation of four single-string spacecraft, as shown in Figure 6, is a feasible alternative to the three dual-string spacecraft shown in Figure 1. While it is not clear which would cost less, the constellation of four single-string spacecraft is used to develop the cost estimate based on a parametric mass-based cost model calibrated to the GRAIL mass and mission cost of $497M at launch, inflated to FY2012 dollars. The GRAIL mission has two spacecraft in orbit about the Moon with a precision inter-satellite microwave ranging instrument. Each satellite is single-string, which is judged acceptable risk since the mission duration is less than one year.

With four spacecraft the main gravitational-wave science goals would be met even with the failure of one spacecraft, so that sufficient mission reliability could be achieved with four single-string spacecraft (where the spacecraft support equipment such as the main computer, attitude control system, etc. would not have any redundancy). With three spacecraft each of which must work to achieve most science goals, normal practice would be to use redundant spacecraft equipment (space computer, spare attitude control subsystem, etc.). With four single-string spacecraft the cost of the spacecraft support equipment will be lower than for three redundant spacecraft (e.g. four computers versus six) but require more payload equipment (e.g. eight telescopes instead of six).

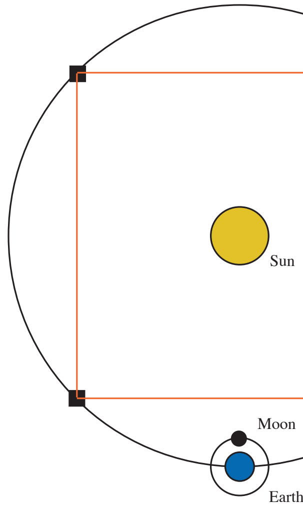

Figure 6: Alternate spacecraft orbit configuration

Each of the four spacecraft has been assumed to have two identical assemblies, each assembly consisting of a laser, optical bench, telescope, measurement electronics, etc. This payload assembly is similar to the laser ranging instrument under development for the GRACE Follow-On mission [Sheard et al. 2011]. The GRACE-FO mission is currently under development as a partnership between the United States and Germany. The laser ranging instrument has a measurement requirements of 50 nm/√Hz. An earlier technology development demonstrated 1 nm/√Hz measurement noise, only twice as high as assumed for this mission concept using a prototype optical bench and a breadboard version of the GRACE-FO electronics [Stephens et al. 2006].

Table 1 list the estimated mass and power of a single payload assembly. The parameters for the optical bench, steering mirror electronics, modulator, electronics, ultra-stable oscillator, and cables are taken from the GRACE-Follow-On mission. The telescope and laser are taken from an earlier LISA study [Danzmann et al. 2000]. A proton energy counter is based on the SWEPAM instrument for the ACE mission [McComas et al. 1998], and the thermal control sized based on GRACE but sized for a constant solar illumination angle.

Table 1: Mass and power estimated of single payload assembly

Unit	Mass[kg] CBE	P[W] CBE	Comment
Telescope	6.6	-	LISA cornerstone study, page 7-27
Optical Bench	3.0	-	GRACE-FO, incl. Photodiode and Steering Mirror
Steering Mirror Elec.	1.0	5	GRACE-FO
Laser	1.5	20	LISA cornerstone study, page 4-33
Modulator	0.5	-	GRACE-FO
Atomic cell	1.0	5	
Electronics	2.2	12	GRACE-FO
USO	1.5	2	GRACE-FO
Cables & fibers	1.5	-	GRACE-FO
Proton counter	3.7	4	ACE/SWEPAM
Thermal Controller	2	10	
Total	17.3	58	

A payload cost estimate for eight assemblies as listed in Table 1 was made using the paramteric mass-based model with an assumed 50% inheritance from GRACE Follow-On and a higher level of complexity than the GRAIL microwave instrument. The cost of four spacecraft was scaled from the cost of the two GRAIL spacecraft. The GRAIL spacecraft provides adequate power for the new payload, since GRAIL must accommodate operations in the Moons shadow while the gravitational-wave mission is in constant sunlight. The GRAIL chemical propulsion system probably is smaller than needed for the gravitational-wave mission, but more work is needed to assess the mass impact, especially if electric propulsion is considered.

The total cost estimate is given in Table 2. The project management, system engineering, and safety and mission assurance cost are based on historical ratios to payload and spacecraft cost. The mission operations cost is larger than the GRAIL mission operations cost since there are twice as many spacecraft and the mission duration is taken to be two years longer than the one-year GRAIL mission duration. The launch cost is taken from a recent LISA study, which is based on a Delta-IV medium rocket that can launch more than 4000 kg into an Earth-escape trajectory. By comparison the mass of the two GRAIL spacecraft plus launch adaptors was <800 kg.

Table 2: Cost estimate based on GRAIL with increased payload mass and complexity, to more spacecraft, and increased mission duration

WBS Element	Dev Cost (FY12$M)	Ops Cost (FY12$M)	Cost (FY12$M)
PM/SE/SMA	53.95	3.3	57.2
Science	25.93	24.3	50.3
Payload	132.8		132.8
Spacecraft plus ATLO	266.8		266.8
MOS/GDS	28.29	22.3	50.5
Launch Vehicle	194.6		194.6
Education & Public Outreach	5.1	0.5	5.6
Reserves	153.9	12.58	166.4
Total	**861.3**	**62.9**	**924.2**

7. Conclusions

A gravitational-wave detection mission without a drag-free system with sensitivity to gravitational waves comparable with that of the LISA mission concept appears feasible. There are many factors that need further investigation, such as other sources of acceleration noise and issues with low received light levels.

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