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Laser metrology concept consolidation for NGGM

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ABSTRACT

Measurement of the static and temporal variation of Earth's gravity field yields important information on water storage, seasonal and sub-seasonal water cycles, their impact on water levels and delivers key data to Earth's climate models. The satellite mission GOCE (ESA) and GRACE (US-GER) resulted in in a significant improvement on our understanding of the system Earth. On GRACE and GRACE Follow-On two satellites are following each other on the same orbit with approx. 200 km distance to each other. A microwave inter-satellite ranging system measures the variation of the inter-satellite distance from which the gravity field is derived. In addition, on GRACE Follow-On, which has been launched May 22nd 2018, a laser interferometer is added as an experiment to demonstrate the capability of this system to improve the ranging accuracy by at least one order of magnitude. To significantly improve the gravity field measurement accuracy, ESA is investigating the concept of a 'Next generation gravity mission' (NGGM), consisting of two pairs of satellites and a laser interferometer as the sole inter-satellite ranging system. Based on the heritage of the development of the laser ranging interferometer for GRACE Follow-On and the former and ongoing studies for NGGM, several concepts for the laser metrology instrument (LMI) for NGGM, namely the on- and off-axis variants of the transponder and the retroreflector concept have been investigated in detail with respect to their application for an inter-satellite distance of approx. 100 km. This paper presents the results of the detailed tradeoff between different concepts, including laser link acquisition, ranging noise contributors, instrument performance analyses, technology readiness levels of the individual instrument units and an instrument reliability assessment.

Keywords: NGGM, GOCE, GRACE, Gravity Measurement, Laser Ranging Interferometer, Laser Metrology Instrument

1. INTRODUCTION

The applied principle for measuring the distance variation Δd of two satellites, flying approximately 100 km apart from each other at the same orbit with nanometer resolution, is heterodyne laser interferometry, similar to the one foreseen for LISA. Laser beams are exchanged between the satellites and small distance variations are measured. The distance variation is caused by residual air drag and differences in the local gravity forces. The variation the optical path travelled by the laser beams can be detected as a variation of the phase angle of the interferometer beat signals. The achievable measurement resolution is a small fraction of the laser wavelength. To derive the gravity component of the distance variation, the air drag component is measured with an accelerometer at the center of mass of each satellite and subtracted from the interferometer ranging signal.

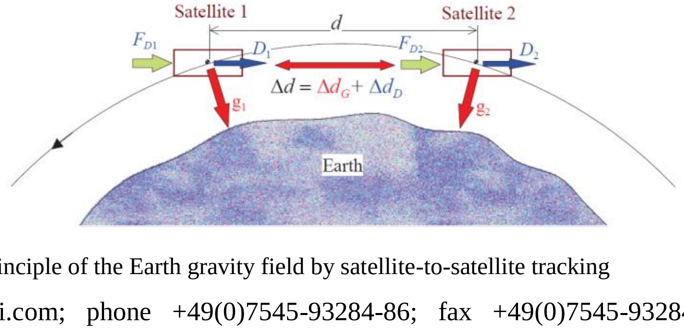

Figure 1. Measurement principle of the Earth gravity field by satellite-to-satellite tracking

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Two basic schemes of laser interferometers have been envisaged for application to NGGM:

- Optical transponder scheme, very similar to the scheme applied on GRACE-FO [1][2][3][4] in off-axis configuration, where the laser beam transmitted by S1 is "regenerated" on S2 by a second laser source locked in phase to the incoming beam and retransmitted towards S1.
- Retroreflector scheme consisting of a heterodyne laser interferometer where the laser transmitted by Satellite 1 (S1) is passively back-reflected by Satellite 2 (S2), as investigated in [5][6] for an on-axis configuration.

Figure 2 shows the basic principle of the two schemes under discussion

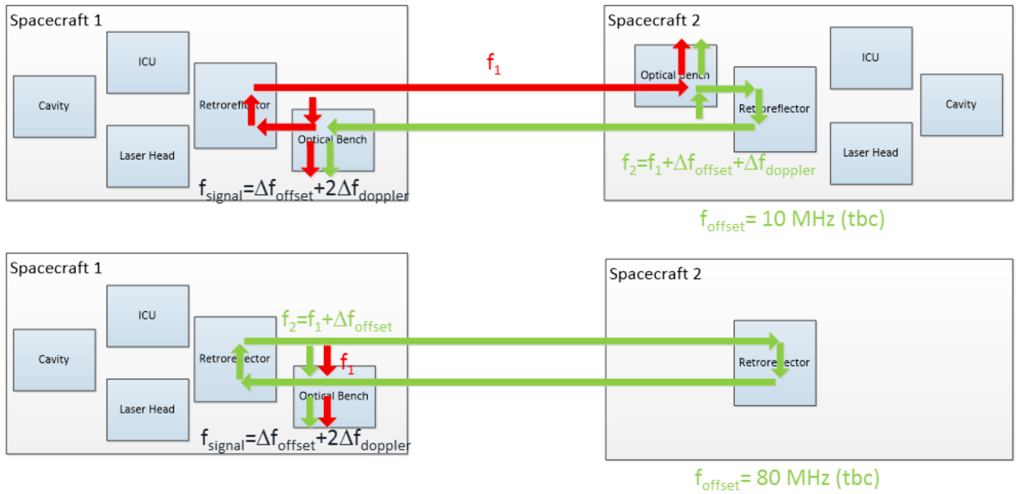

Figure 2: basic transponder (top) and retroreflector (bottom) interferometer schemes

Both concepts can be implemented with the in on- or off-axis configuration concerning the separation of the outgoing (S1→S2) and the incoming (S1←S2) laser beams. In the On-axis configuration the beam separation is achieved via different polarisation status of the outgoing/incoming laser beams, while in the off-axis (also called racetrack) configuration the beam separation is achieved via different optical path travelled by the outgoing/incoming laser beams.

For both schemes, the off-axis variants have been identified as superior due to less criticality with respect to polarisation, straylight, complexity of setup/number of optical elements in the beam path and complexity of operation. Therefore the on-axis options are no longer followed.

Below the, the consolidated design, lower level requirements, the reliability assessment, laser link acquisition, and expected performance are given in detail.

2. RANGING NOISE REQUIREMENTS

The requirement for the measurement noise of the distance variation between the COMs of the two satellites has been specified as fraction of the baseline (= the distance between the two spacecraft). Transferring this fraction of the baseline into LMI ranging performance, it is important to understand that the LMI measurement includes twice this distance (=full roundtrip). We define the single link as the full roundtrip divided by two.

A threshold and a goal level have been established for the distance variation requirement:

- Threshold: relative error power spectral density of 2E-13 1/√Hz at all frequencies > 10 mHz. Below 10 mHz, the relative error is allowed to grow one order of magnitude per decade, e.g. 2E-12 1/√Hz at 1 mHz. For the given baseline length of 100 km between the satellites this corresponds to an inter-satellite ranging noise (single link) of (taking over the Nomenclature of HSL)

$$\tilde{x}_{thr} = 20\frac{nm}{\sqrt{Hz}}\ x\ \sqrt{1+\left(\frac{10mHz}{f}\right)^2} \quad for\ 1x10^{-4}Hz < f < 1\,Hz$$

- Goal: relative error power spectral density < 1E-13 1/√Hz RMS at all frequencies > 10 mHz. Below 10 mHz, the relative error is allowed to grow one order of magnitude per decade, e.g. < 1E-12 1/√Hz RMS at 1 mHz. For the given baseline length of 100 km between the satellites this corresponds to an inter-satellite ranging noise (single link) of

$$\tilde{x}_{goal} = 10\frac{nm}{\sqrt{Hz}}\ x\ \sqrt{1+\left(\frac{10mHz}{f}\right)^2} \quad for\ 1x10^{-4}Hz < f < 1\,Hz$$

We define the noise shape function NSF(f) as $NSF(f) = \sqrt{1+\left(\frac{10mHz}{f}\right)^2}$ and the frequency range between 0.1 mHz and 1 Hz as science measurement band.

3. INSTRUMENT SETUP

The two LMI schemes consist of the same basic units (see Figure 4 and Table 1), with individual variations depending on the scheme. The basic instrument configuration consists of a single frequency laser (the laser head – LH), that is stabilized to a highly stable reference cavity (CAV) to some Hz/√Hz. This stabilised laser signal is launched onto an optical bench assembly (OBA) and split up into a local oscillator and a transmit beam (depending in the scheme one of the beams is frequency shifted). The transmit beam is sent out the second spacecraft, via a retroreflector (RR) with vertex at the center of mass of the spacecraft. At the second spacecraft the beam is retroreflected back to the first one. Either directly by use of a passive retroreflector, again with the vertex in the center of mass, or by a setup identical to the setup to the first spacecraft, with a laser that is frequency offset locked to the incoming beam and sends back this offset locked beam to the first spacecraft. On the first spacecraft, the received beam of the second spacecraft is superimposed with the local oscillator beam onto quadrant photodiodes.

The main signal of the instrument is the resulting difference frequency of the two beams, the heterodyne signal. It contains the fixed frequency offset plus twice the Doppler shift caused by the relative motion of the two spacecraft. From this signal, the accelerometer signals and potentially further calibration signals and models the gravity field is ultimately derived.
In addition, the phase difference between the individual quadrants of the photoreceivers is evaluated and used to point the spacecraft with microrad accuracy towards each other.
This approach enables an interferometric measurement from CoM of one spacecraft to the CoM of the other spacecraft with “nanometre per root Hertz” accuracy.

Initial Acquisition
As the heterodyne signal amplitude is strongly depending on the angle between the two laser beams (local oscillator and received beam) and drops below detection level approximately around 150 µrad, a dedicated link acquisition system is used to reduce the initial satellite pointing accuracy of 3 mrad to less than 100 µrad. This system consists of an acquisition light source (a diode laser with 3 mrad beam divergence and about 1 W of output power) and an acquisition light detector (a position sensing detector or a pixel array detector) with at least 3 mrad field of view. The laser source acts as an ‘artificial’ guide star to which the respective other spacecraft aligns itself to.

Instrument Scheme differences
The two schemes differ in individual unit details, mainly in the placement of the acquisition light detector and light source, the Baffle configuration, the detailed setup of the optics of the optical bench assembly, the Power laser head with or without optical amplifier and Frequency shifter, and the electrical power demand. In addition the redundancy

approach for the two schemes is different, with the transponder scheme foreseeing partial redundancy of critical components and the retroreflector scheme with full instrument redundancy.

3.1 Transponder configuration

In the transponder LMI, the laser head on S1 is looked to the cavity ("Master Laser"), providing 25 mW of single mode single frequency signal to the optical bench. The Laser head is partially (in the case of the TESAT NPRO) or fully redundant (in the case of the ECDL MO). On the second spacecraft, the laser frequency is offset-looked to the first laser by some MHz ("Slave Laser")

Figure 3 shows an overview of the two satellite configuration of the LMI in transponder scheme.

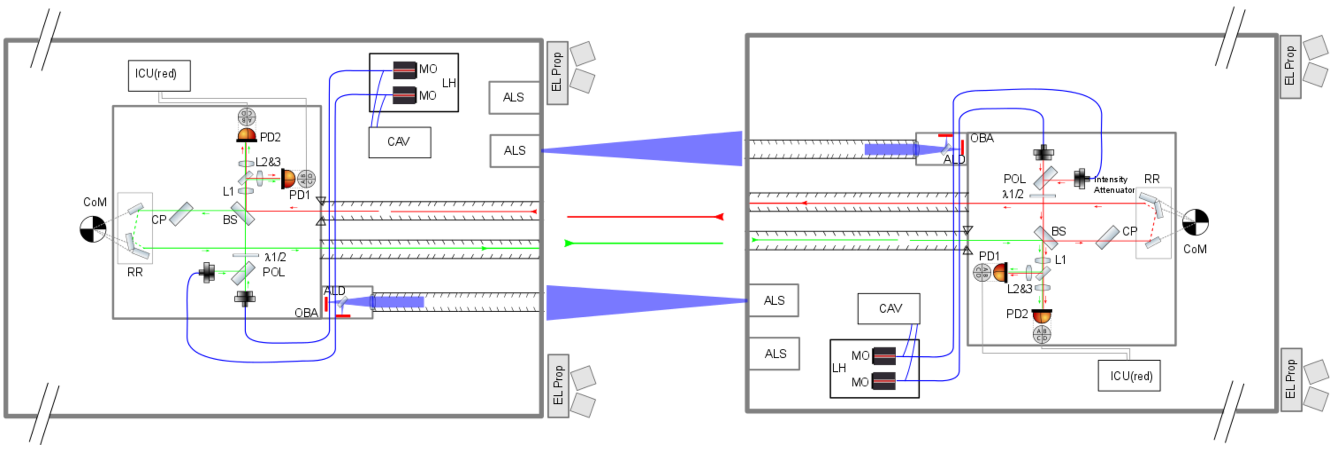

Figure 3: Schematic overview of the LMI transponder scheme of two satellites.

The cavity is not redundant on one spacecraft; redundancy is achieved by flying one cavity on each spacecraft while needing only one on one spacecraft at a time and the possibility to exchange the Master and Slave operational modes. The fibre connection of the LH to the OBA is foreseen to be cold redundant, with a beam superposition achieved via polarisation and a motorized halfwaveplate for polarisation control. The optical setup on the OBA routes the beam from the fibre collimator (via the retroreflector) to the other spacecraft and to the photoreceivers (using a beamsplitter). The imaging optics in front of the photoreceivers image the exit of the fibre collimator and the OBA entrance aperture onto the photoreceivers, thereby minimizing the effect of beam walk due to beam angle changes as well as phase errors due to diffraction effects of the baffles and entrance aperture. The compensation plate (CP) minimizes the ranging noise introduced by the beam splitter (BS) under pointing noise. Only the BS and CP are in the direct measurements path, in which any pathlength noise (thermally or pointing driven) directly couples into the ranging performance. The noise of all other optical elements (from fibers to BS and BS to photoreceivers) is strongly suppressed due to common-mode effects.

The received beam from the other spacecraft with a power of about 1-3 nW (derived in optical link budget calculations for high and low case) enters the optical bench and is reflected at the beam splitter and imaged onto the photoreceivers.

The quadrant photoreceivers on the optical bench can operate in hot or cold redundancy. The heterodyne signal (the beat between the local oscillator and the received beam from the other spacecraft) is read out from the 4 elements of the photoreceivers, preamplified and processed in the phasemeter of the Instrument Control Unit (ICU). The beat frequency/phase change delivers the relative distance change of the spacecraft to each other, which is the main science signal. The phase difference of the individual quadrant delivers the pointing information to the other spacecraft, used to drive the attitude control system of the satellite to µrad accuracy.

The off-axis retroreflector (RR) with its vertex in the Center of Mass(CoM) of the spacecraft routes the beam around the CoM, thereby enabling the distance measurement from CoM to CoM of the two spacecraft. It needs to provide a beam coalignment (incoming to outgoing beam) of less than 40 µrad and a low dependency of the vertex position with temperature to enable the required ranging performance.

At the second spacecraft (with the laser in slave mode), the identical instrument configuration is implemented, but differing on the two S/Cs with respect to the flight direction. The received signal from the first spacecraft is used to offset lock the local laser by some MHz and also to point the second spacecraft to the first spacecraft. Apart from the mode of operation of the LH, the operation principle is as on spacecraft 1.

3.2 Retroreflector Configuration

In the retroreflector LMI, the laser head on S1 is locked to the cavity ("Master Laser"), providing 500 mW of single mode single frequency signal to the optical bench on two separate fibres. The Laser head internally consists of a master oscillator (either the TESAT NPRO or an ECDL, partially or non-redundant), a fibre amplifier with redundant pump diodes, a fibre coupled frequency shifter (AOM), and the laser head control electronics.

The frequency shifter in the laser head provides the required offset frequency between the local oscillator and the transmit beam.

Figure 4 shows an overview of the two satellite configuration of the LMI in retroreflector scheme.

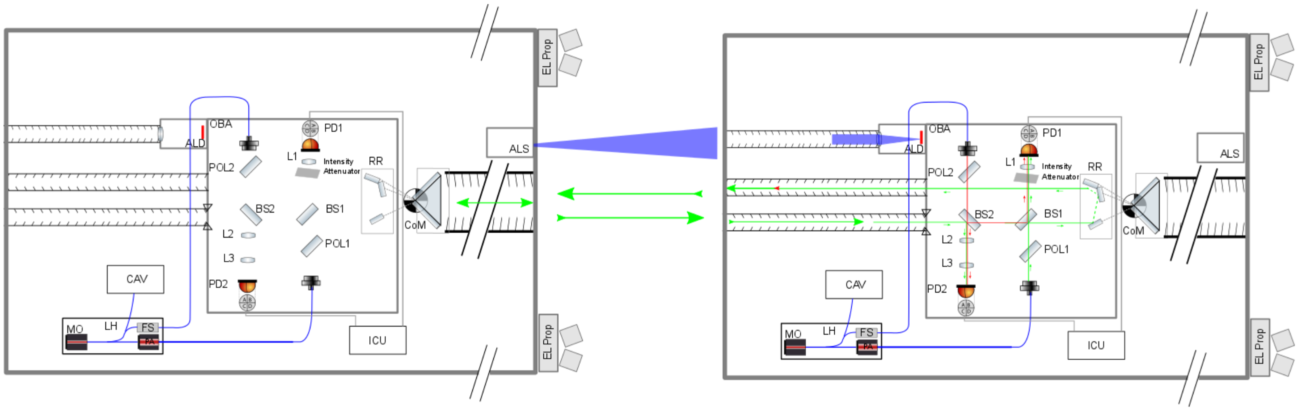

Figure 4: Schematic overview of the LMI retroreflector scheme of two satellites.

The two laser signals (the base frequency and the frequency shifted signal) are delivered to the optical bench assembly (OBA) via two separate fibres.

The frequency shifted beam with a power of some mW is delivered to the optical bench and split up at beam splitter 2 (BS2). The transmitted part serves as local oscillator on photodiode 2 (PD2). The imaging optics in front of the photoreceivers image the exit of the fibre collimator and the OBA entrance aperture onto the photoreceivers, thereby minimizing the effect of beam walk due to beam angle changes as well as phase errors due to diffraction effects of the baffles and entrance aperture. The reflected part is routed to PD1, which serves as reference diode to cancel out any phase fluctuation between the two laser signals introduced by the fibre delivery.

The base frequency laser signal of about 500 mW is delivered via the second fibre to the OBA and routed via the retroreflector (RR) towards the second spacecraft. A small portion of the signal transmits through BS1 towards PD1 to enable the fibre phase fluctuation subtraction.

Only the BS1 and BS2 are in the direct measurements path, in which any pathlength noise (thermally or pointing driven) directly couples into the ranging performance. The noise of all other optical elements (from fibres to BS and BS to photoreceivers) is strongly suppressed due to common-mode effects or cancelled out by the fibre phase fluctuation subtraction.

The received beam from the other spacecraft with a power of about 10-35 pW enters the optical bench and is reflected at the beam splitter and imaged onto PD2.

The heterodyne signals on the photodiodes (the beat between the local oscillator and the received beam from the other spacecraft on PD2, respectively the small part of the transmit beam on PD1) is read out from the 4 elements of the photoreceivers, preamplified and processed in the phasemeter of the ICU. On PD2 the beat frequency/phase change delivers the relative distance change of the spacecraft to each other, which is the main science signal. On PD1 the heterodyne signal is used to subtract the phase fluctuations caused by the delivery fibres from the science signal. The phase difference of the individual quadrants of PD2 delivers the pointing information to the other spacecraft, used to drive the attitude control system of the satellite to µrad accuracy.

The off-axis retroreflector (RR) with its vertex in the Center of Mass (CoM) of the spacecraft routes the beam around the CoM, thereby enabling the distance measurement from CoM to CoM of the two spacecraft. It needs to provide a beam coalignment (incoming to outgoing beam) of less than 40 µrad and a low dependency of the vertex position with temperature to enable the required ranging performance.

Unlike the transponder case, the instrument on the second spacecraft only consist of a three mirror retroreflector, reflecting the signal back to the first spacecraft. For redundancy reasons, the identical instrument configuration is implemented on both spacecraft to provide instrument redundancy (by exchanging the spacecraft positions).

3.3 Basic LMI Scheme comparison and TRL assessment

Table 1 summarizes key requirements and parameters of the two interferometer concepts described above for an inter-satellite-distance of 100 km.

Table 1 Comparison of the two LMI concepts.

Criterion	**Retroreflector Concept (Racetrack)** **[5][6]**	**Transponder Concept (Racetrack) [implemented similar to LRI on Grace FO]**
Laser source stabilisation	Ultra-stable Reference cavity as baseline. The laser noise of 40 Hz/√Hz x NSF(f) yields a ranging noise of 14.2 nm/√Hz x NSF(f) at 100 km inter satellite distance alone. Results from HSL activity show at least a factor of 2 better performance of the laser stabilized to the reference cavity.	
Minimum required received power	5 pW (for *d* = 100 km), cycle slips tbd	100 pW (see SW2), limited by cycle slips
Free Space Loss (for d=100 km)	Approx. 1.2×10^{10} (101 dB)	Approx. 6.3×10^{6} (86 dB)
Minimum Laser Power	500 mW (at Laser Head output)	> 8 mW
Laser source	Master oscillator power amplifier concept (NPRO/ECDL + Fibre amplifier), frequency shifter (AOM)	Single frequency laser (NPRO/ECDL)
Detection system	Evaluation of CW beam, Quadrant photo receivers, science mode beam pointing based on DWS signal, second quadrant photoreceiver as reference to calibrate out phaseshifts in delivery fibre.	Evaluation of CW beam, Quadrant photo receivers, science mode beam pointing based on DWS signal. Two photoreceivers in redundancy. No relevant phaseshifts in beam delivery due to common mode suppression
Optical system	Only plane optics in performance critical path	
Redundancy approach	Two identical S/Cs, Redundancy on Instrument level by exchanging S/C position and operation between active and passive function.	Partial or full redundancy foreseen, requires redundant units per S/C. S/C configuration not identical unless fully redundant interferometer design is implemented.
Acquisition	'Coarse' Pointing of one SC (from some mrad down to 100 µrad) with dedicated acquisition light source (ALS) detected by an wide-FOV acquisition light detector (ALD), Science mode co-alignment actively controlled via DWS (on active S/C only), heterodyne frequency ensured by single laser source plus frequency shifter.	'Coarse' Pointing of both SCs (from some mrad down to 100 µrad) with dedicated acquisition light source (ALS) detected by a wide-FOV acquisition light detector (ALD), frequency sweep of slave laser to heterodyne frequency. Science mode co-alignment actively controlled via DWS, heterodyne frequency ensured by offset frequency lock of slave laser
AOCS-Requirements for the laser beam pointing.	Absolute pointing error of transmitting satellite along the satellite-satellite line $\leq 2 \cdot 10^{-5}$ rad. Passive satellite pointing to mrad sufficient Pointing stability of both satellites along the satellite-satellite line $\leq 2 \cdot 10^{-6}$ rad/√Hz.	Absolute pointing error of both satellites along the satellite-satellite line $\leq 4 \cdot 10^{-5}$ rad. Pointing stability of both satellites along the satellite-satellite line $\leq 2 \cdot 10^{-6}$ rad/√Hz.
Ranging Performance	< 20 nm / √Hz (for *d* = 100 km), limited by laser frequency stability and signal detection noise	< 20 nm/ √Hz (for *d* = 100 km), limited by laser frequency stability
Mass estimate	Approx. 45 kg	Approx. 43 kg (65 for fully redundant scheme)
Power estimate (science)	Approx. 96 W	Approx. 65 W

Analysing the individual key units with respect to their technology readiness level, including the heritage frim GRACE FO, former NGGM studies and LISA activities a TRL assessment of the LMI units has been performed and is summarised in Table 2. Please note that the TRL levels for the transponder concept assume the GFO design. For NGGM design modifications are expected (but not strictly necessary) due to different spacecraft interfaces and redundancy approach.

Table 2 TRL Assessment of key LMI units (in europe)

(Key) Unit	**TRL estimate 2018 (EU)**	**Comment**
ICU(Phasemeter)	3-4	NGGM & LISA Phasemeter developments (EBB of AEI/DTU)
LH (Seeder/amplifier)	NPRO:9 ECDL:4 Amplifier:5	Seeder: Tesat ECDL: FBH Amplifier: HSL 2
CAV	5	HSL2
OBA (transp. / Retro)	9/3	Transponder TRL from GFO
RR	9	Taken from GFO

4. PERFORMANCE ANALYSIS

The LMI performance has been analysed including laser interferometer noise contributors such as laser frequency noise, heterodyne signal detection noise (such as shot noise, electronic noise, power noise, RIN & dark current in relation to the heterodyne signal amplitude), thermal noise contribution of the retroreflector and the optical bench, as well as 'satellite induced' noises due to limited center of mass stability of the satellite and pointing noise of the satellite. Yet to be analysed is the influence of straylight for the retroreflector configuration as well as potential cycle slip issues due to the low received signal.

Figure 5 displays the resulting breakdown of the current laser metrology system ranging noise analysis for both interferometer schemes: optical transponder, retroreflector. The error breakdown refers to the threshold level and to a baseline length of 100 km. The figure shows the individual contributions from each unit and scheme in terms of allocation, and current best estimate. The individual contributions are the added up via RSS, as they are considered independent from each other. In addition a worst case (WC) based on the individual CBEs is given to get a 'realistic worst case'. For this the individual contributions are all added up linearly. The contributions of the two spacecraft differ from each other due to the master/slave (in the case of the transponder) and active/passive side (in the case of the retroreflector) instrument setup. To come from the single spacecraft noise to the single link noise, the two SCs contributions are added up (RSS, LIN) and divided by two. The different allocation breakdown reflects the different sensitivity of the two interferometer schemes to the individual error contributors.

In summary, according to the current analysis, the two LMI schemes show very similar ranging noise performance, below the goal of 10 nm/√Hz x NSF(f) for the CBE and below the 20 nm/√Hz x NSF(f) for the "worst case CBE". Key contributor to the ranging noise for both schemes is the laser frequency noise.

5. LINK ACQUSITION

For the heterodyne phase measurement of the LMI it is required that the laser beams of the satellites point to the other spacecraft better than the beam divergence (approx.). For the planned LMI schemes this means a pointing accuracy of about 100 µrad. Above this value, the received signals are too low to detect a beat note due to received laser power and heterodyne efficiency for two superimposed beams with different propagation angles. In addition the laser frequency offset needs to be in the detection bandwidth of the photoreceivers and phasemeter, which is in the range of some to some ten MHz (on GFO it is ~~s~~ 4-18 MHz). The initial offset frequency accuracy for the transponder concept is assumed with up to 1 GHz. For the retroreflector concept, the offset frequency is given by the frequency shifter in the needed range.

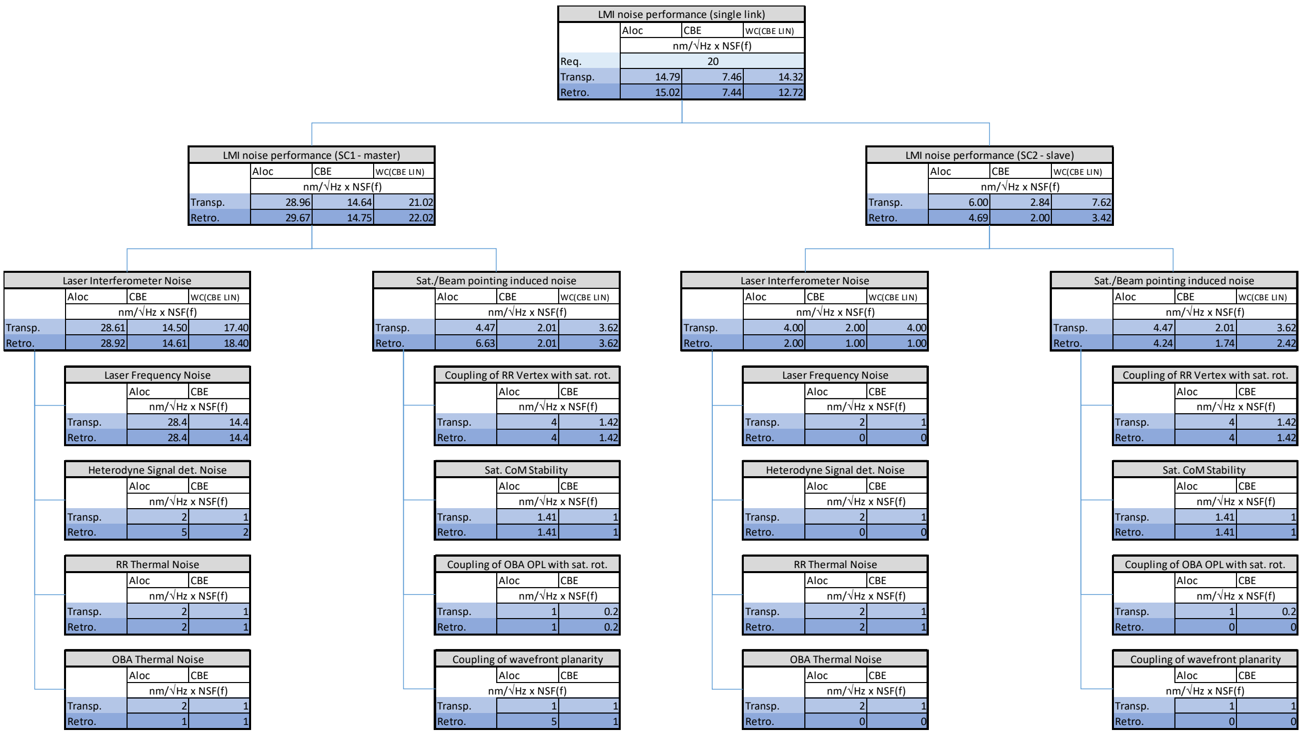

Figure 5: LMI ranging noise breakdown.

The AOCS based on star camera signals are expected to control the spacecraft to 100 µrad accuracy, but the initial alignment of the Optical benches with respect to the satellite-to-satellite direction is expected to be limited to about 3 mrad due to on-ground alignments, one-g to zero-g effects, launch setting effects and to the knowledge of the satellite relative position.

To achieve the initial link acquisition (and potentially later re-acquisitions) it is therefore needed to reduce the pointing uncertainty. In GFO a fine steering mirror is used to scan the 3 mrad field of view and to simultaneously scan one of the laser frequencies until both the pointing and the laser offset frequency are the same. This approach was decided, as the spacecraft pointing control is rather course with approx. 2 mrad absolute pointing accuracy and 300 µrad pointing cone (on/off thruster control). It is a potentially rather lengthy and complex 5-dimensional scan (spacecraft orientations (2x2) and laser frequency(1))

For NGGM a much better pointing control will be achieved by use of linearly controlled electric propulsion, allowing absolute pointing of some single µrad (after initial calibration and using the LMI DWS signal as sensor). For the initial calibration, aligning the star camera accuracy with the LMI, a dedicated acquisition light system is proposed, consisting of an "acquisition light source" (ALS) and an "acquisition light detector" (ALD).

The ALD consists of a laser source with 1 W of output power, a beam divergence of 3 mrad and a wavelength optimized for the ALD spectral sensitivity, e.g. a single emitter fiber coupled laser diode. The ALD acts as an 'artificial guide' star for the ALD. The ALD consist of an angle detector based on a position sensing detector (already breadboarded by Thales Alenia Space) or on a pixel array sensor, allowing for higher accuracy at lower signal levels.

In the following the link acquisition approach is described for the NGGM transponder scheme, the NGGM retroflector scheme and – for reference - the GFO transponder scheme.

5.1 NGGM Transponder Scheme

For the transponder scheme the initial pointing uncertainty is assumed with +/- 3 mrad and frequency offsets with +/- 1 GHz. To reduce the pointing uncertainty the ALS/ALD systems are turned on simultaneously on both spacecraft and the attitude control system is using the ALDs to turn the optical axes of the spacecraft to each other to less than 100 µrad accuracy (in pitch and yaw, roll remains controlled by the star cameras). The received power in the ALD at 100 km and 3 mrad pointing error is: 0.67 nW (assuming 30 mm aperture).

Once the 100 µrad are reached, the slave frequency is sweeped until the photoreceivers see a heterodyne signals (which should happen on both S/C at the same time), then the slave laser frequency is offset locked to the master laser frequency. The ALS/ALD system can then be turned off and the spacecraft attitudes are controlled via the DWS signal for the whole mission in science mode. Figure 6 left shows an illustration of the acquisition procedure.

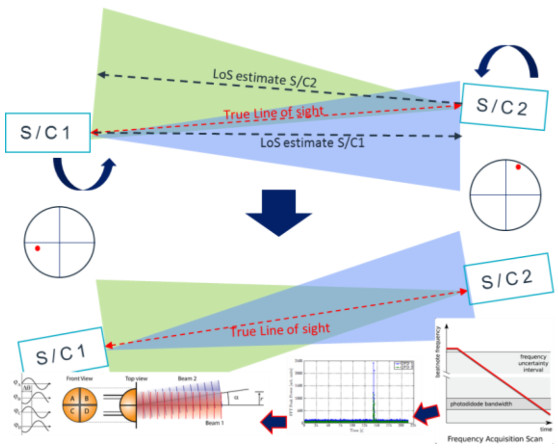

Figure 6: left: NGGM Transponder Scheme Initial link acquisition; right: NGGM retroreflector Scheme Initials link acquisition

5.2 NGGM Retroflector Scheme

For the transponder scheme the initial pointing uncertainty is assumed with +/- 3 mrad. As only on one SC the LMI is active, while on the second SCs, only the passive retroreflector is used, and the offset frequency is set by the frequency shifter on the active SC, the acquisition procedure is more simple.

To reduce the pointing initial uncertainty the ALS on the passive spacecraft is turned on as well as the ALD on the active side and the attitude control system on the active SC is using the ALD to turn the optical axes of the spacecraft to the second one to less than 100 µrad accuracy (in pitch and yaw, roll remains controlled by the star cameras). The received power in the ALD at 100 km and 3 mrad pointing error is: 0.67 nW (assuming 30 mm aperture). On the passive side, the 3 mrad pointing accuracy are sufficient when the baffles diameter are designed accordingly. However, also on the passive spacecraft, the pointing noise in the measurement band needs to be as good as on the active side. It is currently assumed the normal AOCS is able to achieve this noise level.

Once the 100 µrad are reached, the photoreceiver on the active SC sees a heterodyne signal and the DWS signal can be used. The ALS/ALD system can then be turned off and the spacecraft attitudes are controlled via the DWS signal for the whole mission in science mode. Figure 6 right shows an illustration of the acquisition procedure.

6. RELIABILITY ASSESSMENT

A first reliability assessment of the LMI schemes has been performed base on available or assessed Failures In Time (FIT) values for individual components and assembly. Given the long mission duration stated with 11 years, long-term instrument reliability is a design driver.

6.1 Reliability logics

As described above, the two LMI schemes are foreseen to follow different redundancy approaches. For the transponder scheme, a partial redundancy of critical equipment is proposed as baseline. For the retroreflector scheme instrument level redundancy is implemented by having the identical instrument configuration on both spacecraft, giving instrument level cold redundancy by exchanging the spacecraft positions and switching the active/passive role in case of a failure of one LMI component. The setup on each spacecraft is (to a large extend) single-string. Both approaches can be modified (e.g. adding full redundancy for the transponder scheme or adding additional partial redundancy for the retroreflector scheme, both having impact on mass, envelope and finally cost).

Table 3 states the assessed FIT values for key LMI components. Please note that these assumptions are currently rough order values taken from other developments or best guesses. Purely mechanical and simple optomechanical elements (e.g. optical plates in mount) are considered to have a low FIT value (1-10) achieved by proper design.

Table 3: FIT value assumptions for key LMI components

Component	UNIT	FIT	Component's Reliability for mission life time	Expected failure probability during mission (11 years)	Redundancy	FIT* (11 years)	Reliability for mission life time	Expected total failure probability	FIT Source
Power Electronics	ICU, LH	482	0.955	0.045	2	21.4	0.998	0.002	MERLIN FRU
FPGA Board	ICU, LH	182	0.983	0.017	2	3.1	1.000	0.000	MERLIN FRU -> Processor board?
Backplane	ICU	47	0.995	0.005	2	0.2	1.000	0.000	MERLIN FRU
ECDL Seeder	LH	1000	0.908	0.092	2	88.0	0.992	0.008	first guess, generic
Seeder Current drivers (2xLD 2XTEC)	LH	392	0.963	0.037	2	14.3	0.999	0.001	MERLIN FRU
Fiber Amplifier (Optical part)	LH	200	0.981	0.019	2	3.8	1.000	0.000	first guess, generic
Amplifier Pump diodes (incl. Drivers)	LH	921	0.915	0.085	2	75.2	0.993	0.007	MERLIN FRU (DFB Diodes)
AOM incl. Driver	LH	500	0.953	0.047	2	23.0	0.998	0.002	first guess, generic
EOM Driver (Cavity)	LH	250	0.976	0.024	2	5.9	0.999	0.001	first guess, generic
Optical Cavity alignment	CAV	50	0.995	0.005	2	0.2	1.000	0.000	first guess, generic
Photoreceiver	CAV	643	0.940	0.060	2	37.6	0.996	0.004	MERLIN FRU
EOM	CAV	250	0.976	0.024	2	5.9	0.999	0.001	first guess, generic
Optical System	CAV	50	0.995	0.005	2	0.2	1.000	0.000	first guess, generic
Fiber splices	CAV	50	0.995	0.005	2	0.2	1.000	0.000	MERLIN FRU, 10 per splice
Fiber collimator	OBA	10	0.999	0.001	2	0.0	1.000	0.000	estimate, GFO design
Optics	OBA	100	0.990	0.010	2	1.0	1.000	0.000	first guess, generic
PD1 Quadrant-Photoreceiver + Preamp	OBA	800	0.926	0.074	2	57.3	0.994	0.006	MERLIN FRU (non quadrant)
RR Optomechanical Setup	RR	50	0.995	0.005	2	0.2	1.000	0.000	first guess, generic

6.2 Transponder scheme – partial redundancy

Based on the FIT value assumption the transponder concept partial redundancy approach is to make all high FIT valued components (such as: Electronics, Laser sources & Photoreceivers) redundant. Redundant units (and in some case components in units) can be cross linked in case of a failure. Figure 7 shows the foreseen redundancy logic for the transponder scheme.

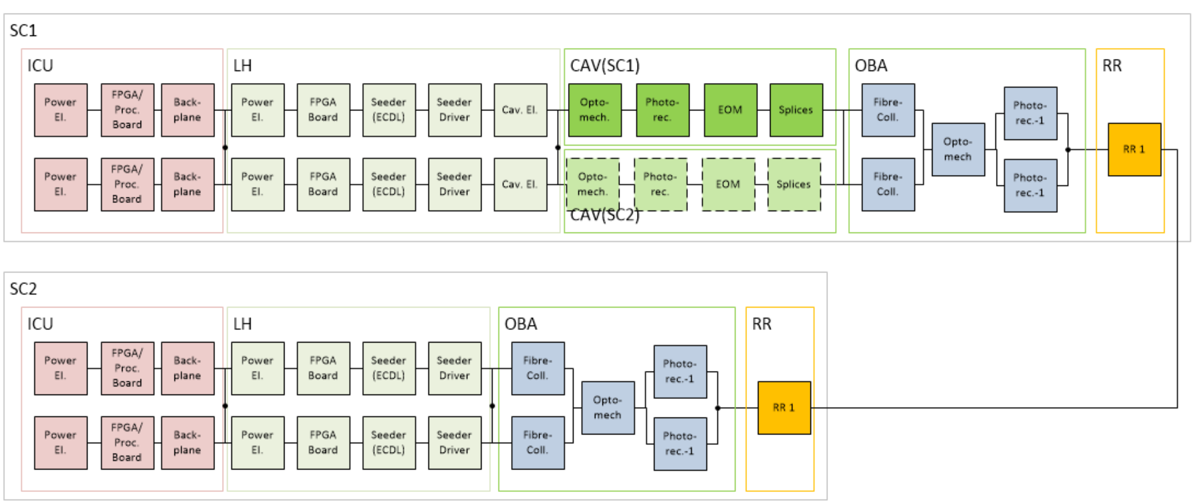

Figure 7: Transponder scheme Reliability Logic – Partial Redundancy

6.3 Retroreflector scheme – instrument level redundancy

Based on the FIT value assumption the retroreflector concept considers instrument redundancy (due to identical spacecraft, but with each spacecraft LMI being single-string). In case of a component failure the S/C positions and active/passive roles can be exchanged with each other. In addition partial redundancy may be added if it seems appropriate.

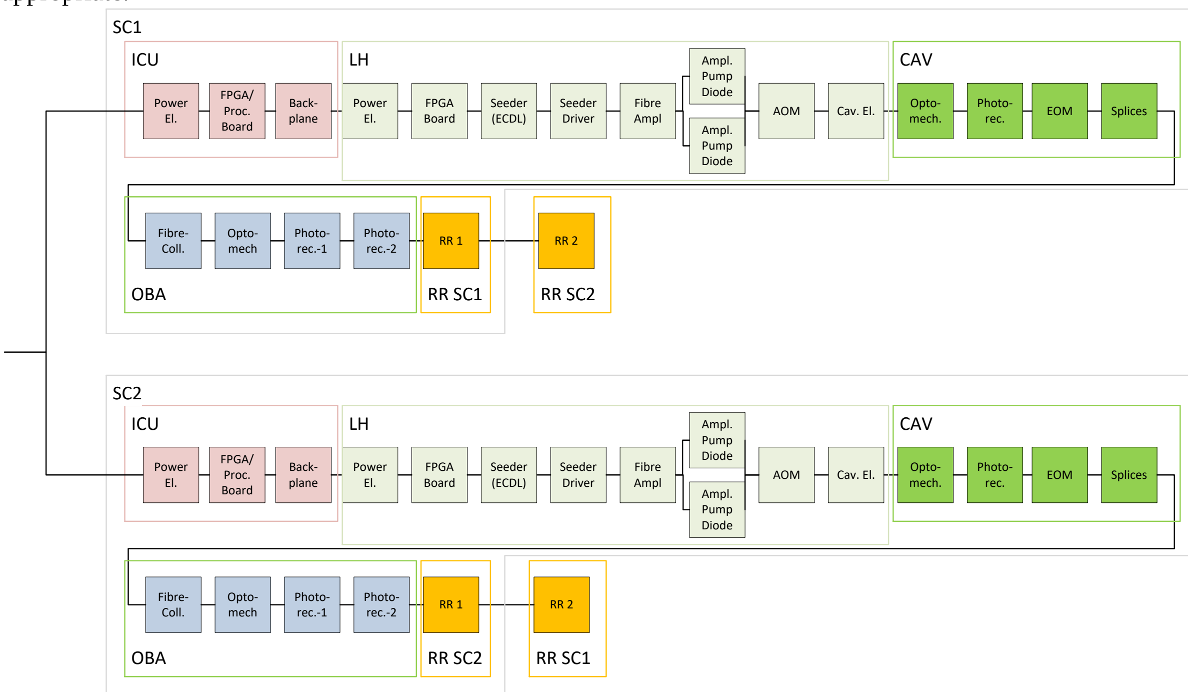

Figure 8 shows the foreseen redundancy logic for the retroreflector scheme.

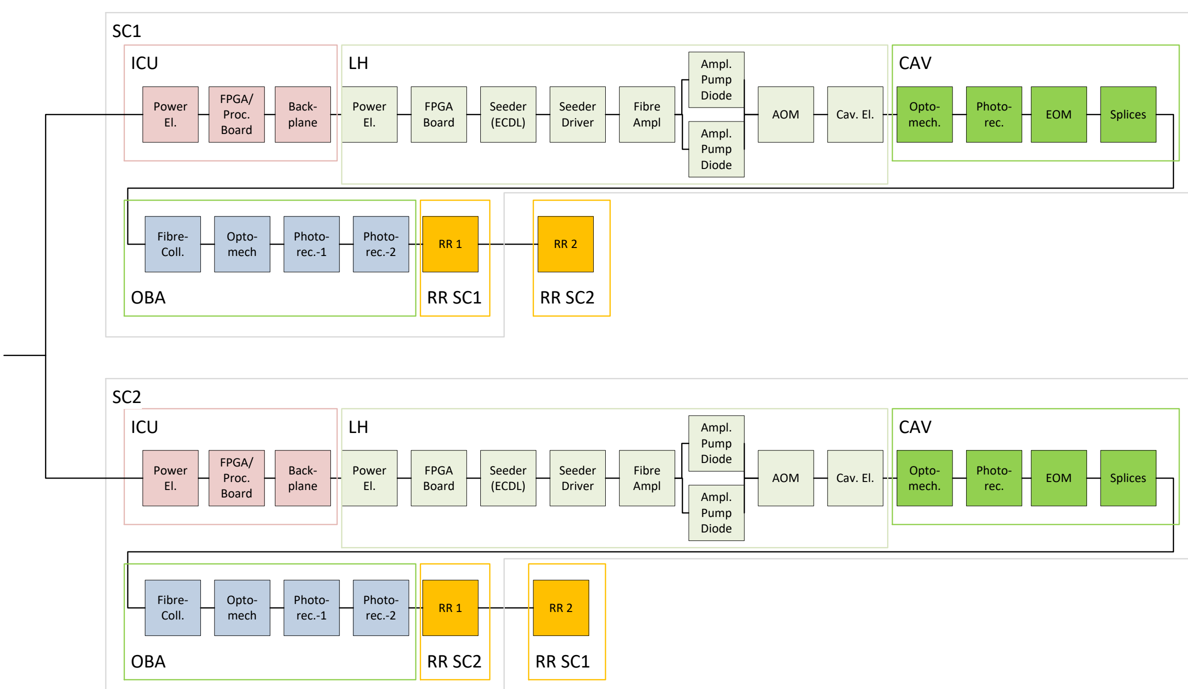

Figure 8: Retroreflector scheme Reliability Logic – instrument level redundancy

6.4 Results

The reliability assessment performed here should be seen as a first indication of qualitative behaviour. The exact numbers are expected to change given the input data which need to be refined in later stages. However, the qualitative results show that a partial redundancy with cross-linked units may have better reliability than a fully redundant instrument where the individual units are not cross-linkable.

shows the assessed reliabilities for the two baseline schemes for different mission duration, clearly showing the influence this has on reliability.
The redundancy approach will be reconsidered in the next step of the design exercise. For this a definition of the required mission reliability on ESA side is recommended. Also a reconsideration of the requested mission lifetime is encouraged.

Table 4: Reliability assessment of baseline LMI schemes for different mission durations

LMI Scheme	11 Years	6 Years	2 Years (for comparison to GOCE)
Retroreflector – instrument level redundancy	80.8 %	92.8 %	99.0 %
Transponder – partial redundancy	90 %	96.2 %	99.2 %

7. SUMMARY

Both schemes, the transponder and the retroreflector, are in principle realisable as on-axis and off-axis variants. For both the off-axis variants are considered of advantage due to less criticality with respect to polarisation and straylight and resolution complexity of operation. Therefore the off-axis variants are selected for the potential LMI design for NGGM.
At 100 km inter-spacecraft-distance (considered adequate for the mission scientific objectives) the transponder and the retroreflector scheme are considered to provide the required ranging performance with significant margin the ranging noise requirement and very likely also the ranging noise goal. While for the transponder scheme in GFO the achievability of the assumed lower level performance allocations has been verified during the instrument qualification, for the retroreflector scheme more verification is needed to be able to state compliance.
The laser link acquisition strategy is similar for both schemes with somewhat increased complexity for the transponder concept.
Accommodation for both schemes is very similar. A main difference consists in the possibility of implementing an identical configuration of the two spacecraft (leader and follower in the formation) with the retroreflector scheme of the laser interferometer, while the symmetry between the two spacecraft is broken with the optical transponder scheme of the laser interferometer with respect to the flight direction.
The reliability assessment of the two schemes indicate a higher reliability for the partially redundant transponder scheme due to the possibility to crosslink individual units, compared to the need to switch to the redundant instrument (and exchange the spacecraft positions) in case of a unit failure in case of the retroreflector scheme. Depending on the ultimately required mission lifetime and reliability, both schemes offer the possibility of additional redundancy (at increased mass, envelope and cost).
The technology for transponder scheme is mostly available as flight model from GRACE follow on, modifications will be necessary to adapt to other mission requirements and to implement redundancy.
For the retroreflector scheme, the laser amplifier is under development. The baseline laser seed source is the fully qualified TESAT NPRO, which is also used on GFO. For both alternatives ECDLs as alternative seed source are considered attractive with respect to power, mass and cost. Currently these sources have EBB status and need to be developed further to be used for NGGM. As these sources are also attractive for GWO and atomic clock applications, it is expected that the development towards qualification for NGGM would provide synergies with other mission scenarios.

Summarizing the different aspects of the interferometer concept trade off, we conclude that both schemes, the retroreflector and the transponder schemes, have their benefits and drawbacks. The transponder scheme is a ‘safe’ option, with the LRI configuration from GRACE FO being fully qualified and operation in orbit. The retroreflector scheme offers the possibility to implement identical spacecraft, a less complex optical link acquisition. The low received power and the resulting high requirement on straylight, multipath suppression and potential cycle slipping needs to be investigated further.

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