

MiniCAS - A Compact Constellation Acquisition System

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ABSTRACT

Satellite gravimetry missions such as GRACE-FO and NGGM require attitude control to establish interferometric laser links between two spacecraft, yet unknown pointing errors caused by manufacturing tolerances, launch vibrations, and thermo-elastic deformations necessitate acquisition scans lasting up to several hours. To enable near-instantaneous acquisition with microradian sensitivity, a novel approach is the Miniaturized Constellation Acquisition System (MiniCAS), a compact optical system for relative attitude determination. Identical Emitter-Receiver pairs are installed on each spacecraft. The Emitter generates a divergent beacon beam that fully covers the pointing uncertainty cone. On the receiving spacecraft, relative attitude offsets induce beam tilts that are optically mapped onto a focal plane array (FPA) as measurable spot displacements, directly yielding the relative line-of-sight orientation. Integrated within a compact, approximately 2U form factor, MiniCAS features a mechanically decoupled optical bench, a modular architecture, and external thermal referencing to minimize thermo-mechanical misalignments. The Receiver combines a baffle, precision lens assembly, and an InGaAs FPA, paired with a dedicated Emitter optimized for beam coverage. MiniCAS significantly reduces acquisition time without compromising pointing accuracy, making it suitable for rapid-deployment satellite constellations. The hardware is currently under development and targeted for Technology Readiness Level 5 (TRL 5) by late 2026.

Keywords: Constellation Acquisition, Laser Ranging Interferometer, Attitude Control

1. INTRODUCTION

Since 2018 the Gravity Recovery and Climate Experiment – Follow-On (GRACE-FO) mission monitors temporal variations in Earth’s gravity field, using satellite-to-satellite tracking in low-Earth orbit (LEO).^{1,2} By establishing a stable laser link between the spacecraft that are separated by more than 200 km, GRACE-FO is continuously mapping the redistribution of planetary mass, providing critical insights for instance into changes in terrestrial water storage, ice sheet dynamics, and ocean circulation.¹⁻³ GRACE-FO will be replaced by the GRACE-Continuity (GRACE C) mission in 2028, for a seamless transition to the previous mission. Additionally, in the early 2030s, the Next Generation Gravity Mission (NGGM) will deploy two identical satellites pairs in an inclined orbit, complementing GRACE-C’s polar-orbiting satellite pair to improve the spatial and temporal resolution of global mass-change measurements.⁴ NGGM will employ the same technique of satellite-to-satellite tracking between a satellite pair. Precise relative attitude control is critical for satellite gravimetry missions such as GRACE-FO, NGGM, and Mass Change and Geoscience International Constellation (MAGIC), which require highly accurate initial alignment to establish interferometric laser links. During the commissioning phase unknown errors arise in estimating the true line of sight (LOS). These errors can emerge from manufacturing and integration tolerances, which are inevitable when constructing and assembling precise optical systems. Additionally, transitioning from Earth’s gravity to micro-gravity causes the spacecraft to de-bend in space. Vibrations during the rocket launch and setting effects will affect the alignment of the optical sub-systems on the spacecraft.⁵ A major step in the commissioning phase is the acquisition and maintenance of a stable laser link between the satellites. This important procedure might be carried out solely by the laser interferometer itself or it might be supported by a link acquisition system, which would provide essential pointing information to the spacecraft and its sub-systems.

On GRACE-FO this calibration procedure is carried out without dedicated hardware.^{6,7} Instead the fast steering mirrors (FSMs), which are installed on both spacecraft, are used to carry out simultaneous spatial scans of the laser beams. The FSM on the master spacecraft carries out a hexagonal scan, while the transponder executes a Lissajous scan while simultaneously ramping its frequency. For each scan step that the master performs the transponder carries out the Lissajous scan. The entire procedure takes about nine hours, the data then needs to be downloaded and processed. Afterwards, the roles are switched and the procedure is repeated to minimize the effect of systematic errors which could arise due to the different scans carried out by the master and the transponder.⁷

To ease the process of link acquisition, significantly decrease the acquisition time and to minimize the scan complexity a new approach is the use of a dedicated hardware like Miniaturized Constellation Acquisition System (MiniCAS). This hardware resembles one implementation option of such a link acquisition system and is especially relevant to missions without an integrated FSM, as this implies that no fast spatial scans of the laser beams can be carried out.

MiniCAS will consist of a modular system that pairs an optical Receiver with a dedicated light source (Emitter). Identical systems will be installed on two spacecraft (see Figure 1). By detecting the light emitted by one spacecraft, the Receiver on the other spacecraft can determine its misalignment relative to the LOS. This measurement is directly linked to the relative tilt of the spacecraft with respect to the LOS. A classic field of application for MiniCAS are satellite gravimetry missions. In principle, however, the MiniCAS technology offers added value wherever the relative alignment of any two spatially separated objects is of interest.

This paper first introduces the current GRACE-FO link acquisition method and outlines its underlying operating principles. Particular attention is given to the method's main limitations and their impact on the reliability and efficiency of the acquisition process. This analysis establishes the technical baseline for the MiniCAS system, which is presented in the subsequent sections as an approach designed to overcome these limitations and improve link acquisition performance.

2. ACQUISITION OF INTER-SATELLITE LASER LINKS

GRACE-FO allows users to determine temporal variations in Earth's gravity field by precisely measuring changes in the distance between its two satellites. These distance variations result from differential accelerations caused by spatial and temporal fluctuations in Earth's gravitational field. The inter-satellite distance is measured with a laser ranging interferometer (LRI), which employs heterodyne laser interferometry.^{3,7}

The heterodyne signal is generated by transmitting a stabilized laser beam from Satellite A to Satellite B. At Satellite B, the local laser is phase-locked to the received light with a fixed frequency offset. The phase-locked local laser generates a coherent beam that is transmitted back to Satellite A. This produces a heterodyne beat signal on Satellite A whose phase is measured. The offset to the phase lock shifts the signal away from zero, which enables the distinction between the satellites approaching and moving away from each other. Under the assumption that the phase tracking on the transponder is perfect, any observed changes in the phase correspond to changes in the distance between the two satellites.

During commissioning of the laser ranging instrument, the actual LOS will deviate from the on-ground predictions. These deviation errors may include manufacturing and integration tolerances, structural relaxation in microgravity, launch vibrations, and settling effects that may alter the alignment of the optical subsystems which cannot be fully characterized on ground. In total five degrees of freedom have to be corrected, in order to establish the optical link. These degrees of freedom include two pointing angles on each spacecraft and the frequency difference between the two independent lasers.^{6,7} For the LRI on GRACE-FO the uncertainty spaces were estimated to be $\Delta\gamma = \pm 3$ mrad for the four angles and $\Delta f = 320$ MHz for the frequency difference between the lasers on the two satellites. As the LRI itself is used on GRACE-FO to carry out this initial acquisition sequence, only interferometric measurements can be done. This implies that these five degrees of freedom have to simultaneously match their correct values at least once during the procedure, leading to a complex acquisition strategy. During this search the two spacecraft have individual roles, called the master and transponder spacecraft. Since both LRI units are functionally identical, either spacecraft can in principle operate as master or transponder. GRACE-FO uses a hierarchical scan involving the FSMs on both spacecraft and a frequency sweep of the transponder laser. The transponder FSM performs a fast two-dimensional Lissajous scan. Simultaneous sinusoidal commands are applied to its two axes f_α and f_β , with frequencies of $f_\alpha = 100$ Hz and $f_\beta \approx 2$ Hz.⁷

In parallel, the master FSM follows a slower discrete hexagonal scan pattern. Hereby, the master scan contains approximately 1200 discrete positions with a dwell time of about 0.56 s per position, resulting in a duration of approximately 11.7 min for one complete master angular scan.^{6,7} For every angular position of the master, the transponder completes an entire Lissajous scan. The detected heterodyne peaks are recorded together with the corresponding FSM positions, from which the four pointing offsets can be estimated. The remaining degree of freedom is covered by sweeping the transponder-laser frequency across the expected uncertainty range. With an approximate duration of

$$T_{\text{acq}} \approx \frac{\Delta f_{\text{scan}}}{\dot{f}_{\text{scan}}} = \frac{320 \text{ MHz}}{10.5 \text{ kHz s}^{-1}} \approx 3.05 \times 10^4 \text{ s} \approx 8.5 \text{ h}, \quad (1)$$

the frequency scan is the slowest scan dimension.⁷ After this initial acquisition, a reduced-range re-acquisition procedure is carried out. The angular scan range is decreased to approximately ± 300 μrad because the dominant static pointing offsets have already been identified. Once a sufficiently strong heterodyne peak is detected, the scan is terminated and the instrument transitions to phase tracking. The transponder laser is then offset phase-locked to the received light, while the FSM control loops minimize the differential wavefront signal (DWS) and maintain the optical alignment. The nominal reduced-range scan sequence was designed to last up to 80 s.^{6,7} Although the concept was successfully demonstrated, this does not mean that the initial acquisition procedure is completely free of risks. A missing beat note e.g. does not indicate which search coordinate is incorrect. Either spacecraft may be mispointed, the frequency difference may be outside the Receiver bandwidth, or the received optical power may be insufficient.

Its success strongly depends simultaneously on both FSMs, both lasers, the photoreceivers, the phasemeters, and the spacecraft attitude-control systems. Changes in attitude, temperature, alignment, or laser frequency during the multi-hour scan can additionally shift the optimum parameters. The procedure is therefore not only lengthy but also operationally complex and difficult to diagnose in the event of an unsuccessful acquisition. Although GRACE-FO was the first in-orbit mission to demonstrate this type of inter-satellite laser-link acquisition, future missions such as GRACE-C, LISA, NGGM, and laser-communication CubeSat missions face comparable acquisition challenges. The GRACE-FO concept, however, relies on FSMs to scan the angular search space and is therefore not directly applicable to missions without such systems. A more generally applicable acquisition concept must be capable of operating independently of a FSM. A dedicated acquisition sensor such as MiniCAS addresses this limitation by enabling line-of-sight determination for such missions, while also reducing the operational complexity and duration time of the acquisition process to almost instantaneous measurements.

3. THE MINICAS SYSTEM

The development of MiniCAS is closely aligned with the requirements of satellite gravimetry missions. GRACE-FO serves as a technological reference, as it successfully demonstrated both laser ranging interferometry and laser link acquisition in orbit. Many of the technologies and concepts established on GRACE-FO are also being used and further developed for future gravimetry missions.

To ensure that MiniCAS is applicable beyond the GRACE-FO architecture, the requirements of upcoming missions such as NGGM and GRACE-C are also considered, resulting in the preliminary requirements for the MiniCAS system stated in Table 1. It should be noted that MiniCAS is designed as a self-contained system, that provides its own beacon beam for detection. However, the Receiver FPA was deliberately chosen to also be compatible with the wavelength of the lasers that will be used on GRACE-C and NGGM ($\lambda = 1064 \text{ nm}$).⁷

Table 1. Requirements and reference parameters for the MiniCAS system.^{2, 4-7}

Parameter	Requirement / Reference Value
Detector technology	InGaAs sensor / camera
Wavelength	$\lambda \approx 1064 - 1550$ nm
Minimum received power	Reliable beam detection and position estimation at $P_{\text{sens}} \geq 1$ pW
Field of view	$\geq \pm 3$ mrad (6 mrad full FoV)
Angular accuracy	≤ 10 μ rad
Inter-satellite distance	100–300 km
Output Data Rate	1 – 10 Hz
Operating Temperature	20 °C

3.1 Measurement Principle

MiniCAS is a specialized system developed for high-precision angular measurements of an incoming beacon beam. It is designed specifically for deployment in LEO. The system’s architecture is modular, featuring separate components for signal emission and reception, ensuring flexibility, and adaptability to the demands of specific satellite missions.

MiniCAS involves three subsystems with very distinct tasks. It consist of the Emitter Unit, the Receiver Unit and the Instrument Control Unit (ICU). The Emitter Unit features a laser source combined with an optical lens system designed to generate the beacon beam. The lens configuration is tailored to meet the requirements in Table 1, that employ a satellite separation of over 200 km, creating a beacon beam expansion that sufficiently covers the pointing uncertainty cone at the receiving interface. On the receiving end, the light enters the MiniCAS system, impinges on a lens configuration that focuses the light onto the chosen camera.

The following Figure 1 describes the measurement principle of MiniCAS, where a) indicates the system with perfect alignment of the two separated spacecrafts. The receiving optical system is aligned with the laser beam propagating through the center of the lens and being focused onto the nominal detector center position. A tilt of the emitting spacecraft, owing to the spherical wavefront of the emitted beam, causes the local propagation direction at the Receiver to remain unchanged, such that the beam does not enter the receiving optics at an additional angle (see sub-figure b)). However, a tilt of the receiving spacecraft as demonstrated in c) is causing a misalignment between the incoming beam and the optical axis. The resulting angle of incidence is converted by the lens into a lateral displacement of the beam spot on the detector. MiniCAS can be operated in both directions simultaneously just as illustrated in d).

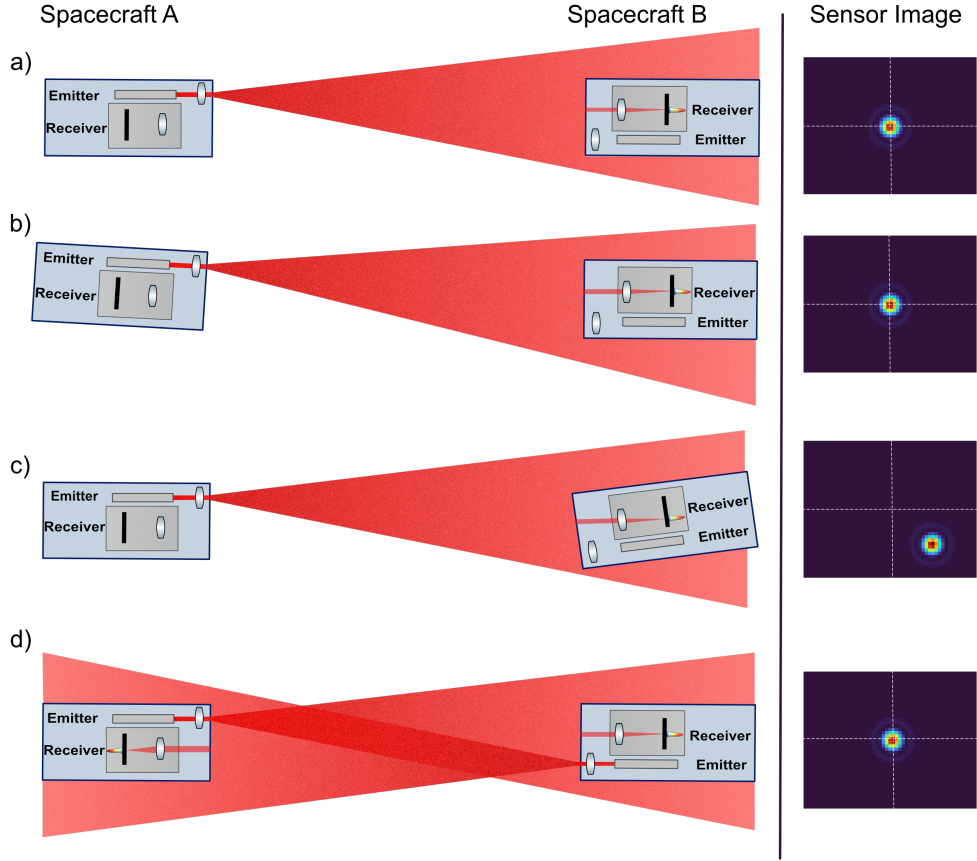


Figure 1. (a) Shows a perfectly aligned sensor where the laser beam passes directly through the lens to the sensor. (b) Illustrates a tilt of the emitting spacecraft. Due to the spherical wavefront, the beam does not enter the Receiver at an angle. (c) If the receiving spacecraft is tilted the light enters the Receiver at an angle which results in a displacement of the beam on the detector plane. The last image d) shows simultaneous bidirectional measurement, enabling the relative alignment to be measured in both directions at the same time.

The target tracking is a function that consists of two sub functions, the localizing of an impinging laser beam on the Receiver (LILB: Localize Impinging Laser Beam) and the laser localization process information (LLPI), such as the signal-to-noise ratio (SNR) and fit residuals.

The purpose of the LILB process is to determine the angle under which an impinging laser beam was received. Therefore, the raw camera data is pre-processed, meaning that the non-uniformity is corrected, the defective pixels are identified and masked, noise and the relative gain of each pixel is corrected. The position of the impinging beam on the sensor is determined by calculating the correlation between the corrected camera frame and a pre-computed template of the expected point spread function (PSF). This yields the position of the impinging beam in the detector plane. A least-squares fit of a Gaussian function is applied to the area around the obtained maximum of the correlation function. This yields the position of the impinging beam in the detector plane to sub-pixel resolution. A scaling factor is applied to convert from the pixel domain into physical angles in the optical reference frame of the MiniCAS system. The beam's angle of incidence is determined and output by the MiniCAS system. The LLPI function provides the necessary information which allows the MiniCAS system to judge on the quality of the obtained results. This may include the maximum and further information on the correlation function between sensor data and template, the SNR of the correlation algorithm. Besides this, the residuals and quality of the Gaussian fit, and the RMS sensor noise are provided. More information on the spot localization can be found in subsection [3.2.1](#).

3.2 MiniCAS Architecture

MiniCAS is a compact and modular system that integrates all functions. The product tree shown in Figure 2 provides an overview of the main functions and units of MiniCAS. At the highest level, MiniCAS is divided into the acquisition system optical bench (ASOB), the housing, and the electronics. This allows MiniCAS to operate independently of the host satellite's mission-specific components.

MiniCAS separates this high-precision optical bench (ASOB) from the external housing via a specialized interface that provides both mechanical and thermal decoupling, shielding the sensitive optics from less stringent housing requirements. The primary interface is thermally insulated, and the system's thermal management is referenced to a stable temperature reference point. The ASOB comprises the Emitter that generates the beacon beam and Receiver Units that collect the information about the local tilt of the spacecraft. The units are further subdivided according to their respective functions such as the lens systems.

The housing provides the mechanical structure for mounting and integrating all MiniCAS units and components. The housing consists of a base frame and removable cover plates. The base frame provides the primary structural interface to the satellite platform, while the cover plates provide protection against radiation and mechanical influences and enable the attachment of external thermal insulation. Measures were taken to minimize the thermal interaction between the active components and to limit heat transfer to the optical bench.

The electrical interface provides the MiniCAS subsystems with the required supply voltages and currents. The ICU acts as the central electrical control and data-processing unit and interfaces with the active components of the system. It controls the laser subsystem of the Emitter Unit and receives the image data generated by the camera for further processing. In addition, the ICU can acquire housekeeping data such as temperature measurements from the integrated sensors.

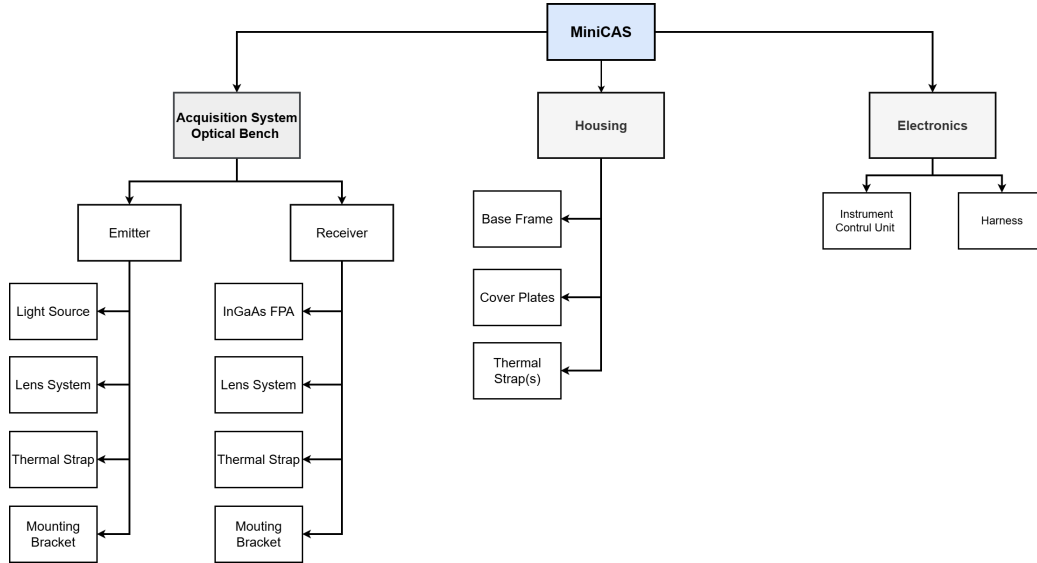


Figure 2. This chart presents the MiniCAS product tree, providing a hierarchical breakdown of the system. MiniCAS is structured into three main functional elements: the acquisition system optical bench (ASOB), the housing, and the electronics. Each of these elements is further decomposed into its corresponding sub-functions and units.

3.2.1 Receiver Unit

The Receiver is one of the three sub-units of MiniCAS. Its primary function is to detect beam displacement and convert it into an angular measurement, enabling the determination of the local satellite tilt relative to the LOS. The Receiver comprises an optical lens system that collects incoming light from the Emitter and focuses it onto an InGaAs FPA camera. This optical design must satisfy specific requirements for FOV and sensitivity, while remaining CubeSat-compatible with a volume of about 2U, as mentioned in Table 1. The lens configuration generates a stable spot on the FPA that is robust against pixel defects and noise. Additionally, it maintains a uniform spot shape across the entire sensor plane to optimize template-matching accuracy.

The first step comprises the characterization and calibration of the camera, which will be conducted in collaboration with the DLR Institute of Space Research in Berlin. The calibration campaign includes the determination of the read noise and dark noise as functions of the FPA temperature, although the read noise is expected to remain largely temperature-independent. Furthermore, the effects of the high- and low-gain settings will be characterized in detail. Additional measurements include the photon transfer curve (PTC), detector linearity, photo-response non-uniformity (PRNU), bad-pixel identification based on anomalous light response, quantum efficiency (QE), and the relative spectral response over the wavelength range from 800 to 1700 nm. For this purpose, a radiometric test bench will be established, comprising a controlled radiation source, suitable optical components, and calibrated reference instrumentation. By comparing the camera response with the corresponding reference measurements under well-defined illumination conditions, the response of the sensor to incident radiation can be quantified. The resulting characterization data will subsequently be used to derive the required camera calibration parameters.

Various optical sensor simulations were carried out using FPA parameters such as pixel count, pixel size, noise characteristics and bit depth. The aim was to reproduce realistic sensor behaviour, investigate the influence of sensor-related disturbances on signal detection, and determine the spot size that the lens configuration within the baffle should generate. A digital sensor model was created in which a defined proportion of pixels was randomly assigned as defective, including dead, hot and randomly responding pixels. Their positions remained fixed throughout all simulations to ensure comparable conditions. Relevant noise sources, including dark current noise, readout noise and white noise, were also included based on the sensor characteristics and datasheet information. These contributions, together with the bad-pixel structure, were used to generate a representative dark frame. For the signal simulations, an optical signal was placed at random positions on the sensor and scaled according to the sensor's bit depth and dynamic range. Noise and defective pixels were then applied, resulting in the image in Fig. 3 a). The dark frame was subsequently subtracted to reduce reproducible disturbances and improve signal visibility, as shown in Fig. 3 b). Finally, the signal position was determined using a pre-computed template that matches the size and shape of the expected PSF. By comparing the sensor and the template, a correlation function identifies the correct position of the signal on pixel resolution. By fitting a Gaussian function on the region of interest sub-pixel accuracy was achieved, enabling an assessment of the achievable position-detection accuracy under realistic sensor conditions.

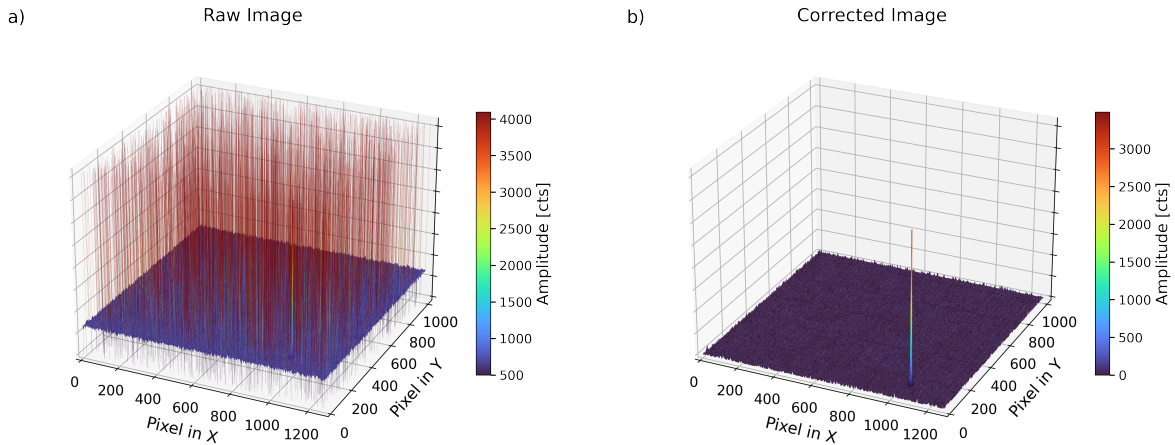


Figure 3. The sensor image is simulated by adding noise, defective pixels, and the signal to the sensor plane, as shown in a). To correct the image, a dark image is acquired and subtracted from the raw image. This dark-frame correction reduces the influence of sensor noise and defective pixels, making the signal more clearly visible, as visible in b).

Building on these simulations and the requirements of the gravimetry missions, an initial lens system was designed using Ray transfer matrix analysis calculations. The optical design was then further refined and evaluated in Ansys Zemax OpticStudio, with particular attention to spot shape and optical aberrations, before being experimentally verified in the laboratory.

3.2.2 Emitter Unit

The primary function of the Emitter Unit is to generate a divergent beacon beam. Accordingly, its optical layout is relatively simple, consisting of a laser source and a baffle incorporating a specifically selected lens. The overall design of the Emitter is driven by the requirements of the Receiver Unit and the properties of the satellite constellation (e.g. the inter-satellite distance). The Emitter wavelength was selected based on an analysis of the QE of the FPA. A high QE is achieved at an operating wavelength of 1550 nm. This wavelength also offers advantages in terms of component availability, as 1550 nm is widely used in telecommunications. The laser is combined with a carefully selected lens configuration, which is mechanically integrated with the laser via a baffle. The lens is designed to produce the required beam divergence such that, even at the maximum operating distance of 300 km and under the maximum expected misalignment of at least ± 3 mrad, sufficient optical power reaches the Receiver of $P_{\text{sens}} \geq 1$ pW, see Table 1.

3.2.3 ICU

The ICU comprises both the main payload data processing (from image to pointing) as well as telecommand/telemetry (TC/TM). To achieve the former, it provides a Camera Link Medium interface for the camera, which is directly connected to a fast Frame Grabber logic on the FPGA fabric of an system-on-chip (SoC). On that SoC, the main data processing is implemented solely in fabric, extracting from the image a beam position via cross-correlation with a template. The resulting data can then be converted into angular data in software running on the SoCs CPU system. Intermediate and final data transfer between the aforementioned components is implemented via external DDR memory. The latter ultimately transmit this data to the spacecraft on-board computer (OBC) via UART. The same interface is used to receive TCs and to transmit TM. In particular, the ICU is responsible of executing the TC for itself, the Emitter and the camera. Also, it gathers housekeeping data of those units and funnels it for TM interfacing with the OBC. The ICU requires a supply voltage of 12 V.

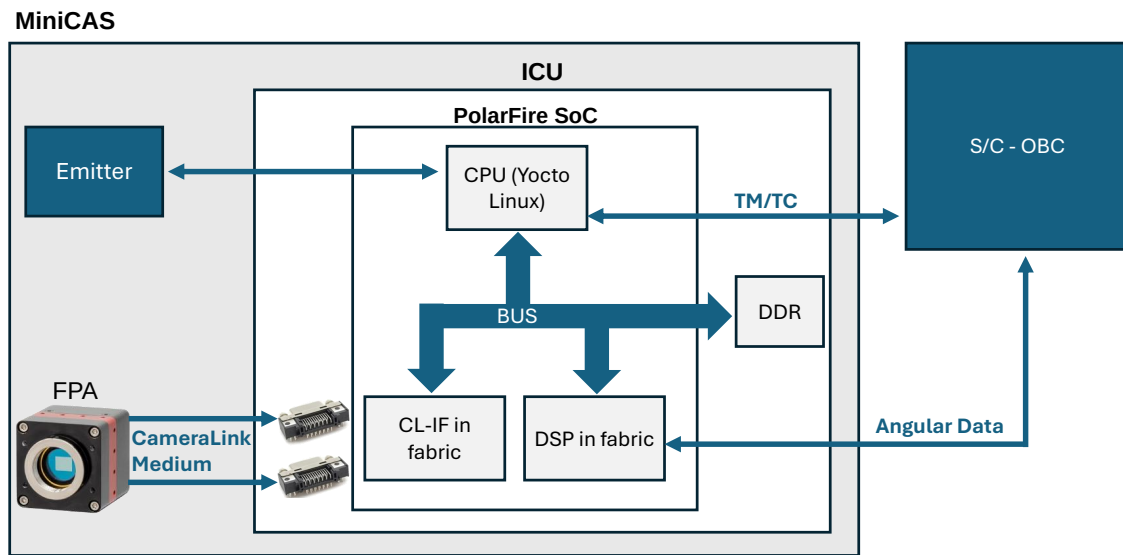


Figure 4. Schematic representation of data and power interfaces of the Emitter, FPA, data processing unit and spacecraft OBC.

4. MINICAS APPLICATION

MiniCAS is currently under active development. All individual components are scheduled to be integrated over the course of Q3 of 2026. The mechanical design, shown in Figure 5, features a compact and modular architecture with all functional units integrated into a single assembly. The Receiver and Emitter are optically isolated on a precision optical bench, while the ICU is mounted directly behind it. The compact and self-contained architecture of MiniCAS enables flexible integration into a wide range of mission concepts, largely independent of other optical and electronic subsystems on the satellite. Its modular design allows the system to be efficiently adapted to mission-specific interfaces and platform requirements.

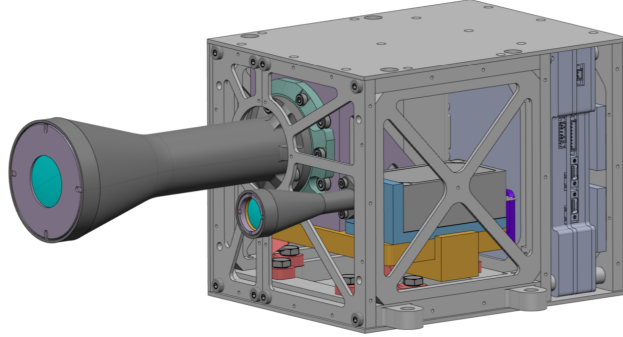


Figure 5. The mechanical design of MiniCAS integrates all required units within a compact assembly. The Receiver Unit is enclosed together with an attached baffle containing the corresponding lens configuration. The Emitter Unit is positioned directly adjacent to the Receiver and comprises the laser source and an integrated baffle with its associated lens configuration. Both the Emitter and Receiver Units are mounted on an optical bench (here in yellow) to provide improved mechanical stability and isolation, while the ICU (in light blue) is located on the rear side of MiniCAS.

Following integration, the system characteristics will be verified, with particular emphasis on determining the alignment between the Emitter and Receiver. The MiniCAS calibration and performance demonstration will be carried out using a far-field beam generator to produce the Receiver input beam. MiniCAS will be mounted on a six-axis hexapod, enabling precise positional and angular adjustment. The angles measured by MiniCAS will be compared against a high-resolution angular reference. The targeted angular sensitivity is below $10 \mu\text{rad}$ at a three-sigma confidence level. To qualify MiniCAS for space-based gravimetry missions and demonstrate Technology Readiness Level 5 (TRL 5), the system will undergo an environmental test campaign. Thermal-vacuum testing will be performed under ultra-high-vacuum conditions at pressures below 1×10^{-7} mbar. During these tests, MiniCAS will be subjected to temperature cycles ranging from -10°C to 50°C in the non-operating state and from 15°C to 30°C during operation. Mechanical qualification will comprise three-axis vibration testing as well as shock testing. The applied qualification loads will be selected to reflect the environmental conditions and qualification requirements representative of LEO gravity-recovery missions.

MiniCAS can be relevant to any application in which the alignment of two or more separate objects is important. Functional and performance laboratory tests with MiniCAS will be carried out at short-range separations of only a few meters, which do not constitute a realistic separation between Emitter and Receiver Units. Hence, to verify the system functionality in field-relevant configurations, a test campaign is scheduled for Q3/Q4 in 2026 at DLR's National Experimental Test Center for Unmanned Aircraft Systems involving moving platforms at large distances. The test comprises a fixed terrestrial optical Receiver and Emitters integrated onto two unmanned aerial vehicles (UAVs). Measurements will be conducted at multiple distances, with the UAVs operating at separations ranging from 1 km to 3 km from the Receiver, for more realistic scenarios.

5. CONCLUSION

MiniCAS is a compact, modular system developed specifically for space missions such as gravimetry missions. It provides a compact solution for missions that do not include dedicated hardware such as an FSM, which is used to steer the beam during acquisition scans. Due to its simplified design, MiniCAS reduces the complexity of the scanning process and enables measurements to be performed almost instantaneously instead of requiring several hours. The Emitter Unit generates a divergent beacon beam, which can be detected by the Receiver on a satellite more than 200 km away. The Receiver collects the incoming light using its specially developed lens configuration and focuses it onto the FPA sensor. Custom software then measures the displacement on the sensor and determines the corresponding local angular misalignments with respect to the LOS. MiniCAS is designed as a self-contained system, that provides its own beacon beam for detection. At the same time, the Receiver FPA was deliberately chosen to also be compatible with the wavelength of the lasers that will be used on GRACE-C and NGGM. The MiniCAS system is currently under active development and is planned to be tested to TRL 5 standards by the end of 2026. In addition to space applications, MiniCAS is also intended for use in aircraft and UAV scenarios. For this reason, the system functionality will also be tested in a UAV environment in Q3/Q4 2026.

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