Paths Not Taken – the GOSSAMER Roadmap's Other Options

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ABSTRACT

Highly efficient low-thrust propulsion is increasingly applied beyond commercial use, also in mainstream and flagship science missions, in combination with gravity assist propulsion. Another recent development is the growth of small spacecraft solutions, not in size but in numbers and individual capabilities.

Just over ten years ago, the DLR-ESTEC GOSSAMER Roadmap to Solar Sailing was set up to guide technology developments towards a propellant-less and highly efficient class of spacecraft for solar system exploration and applications missions: small spacecraft solar sails designed for carefree handling and equipped with carried application modules.

Soon, in three dedicated GOSSAMER Roadmap Science Working Groups it initiated studies of missions uniquely feasible with solar sails such as Displaced L_1 (DL1) space weather advance warning and monitoring, Solar Polar Orbiter (SPO) delivery to very high inclination heliocentric orbit, and multiple Near-Earth Asteroid (NEA) rendezvous (MNR). Together, they demonstrate the capability of near-term solar sails to achieve at least in the inner solar system almost any kind of heliocentric orbit within 10 years, from the Earth-co-orbital to the extremely inclined, eccentric and even retrograde. Noted as part of the MNR study, sail-propelled head-on retrograde kinetic impactors (RKI) go to this extreme to achieve the highest possible specific kinetic energy for the deflection of hazardous asteroids.
At DLR, the experience gained in the development of deployable membrane structures leading up to the successful ground deployment test of a (20 m)², i.e., 20 m by 20 m square solar sail at DLR Cologne in 1999 was revitalized and directed towards a 3-step small spacecraft development line from as-soon-as-possible sail deployment demonstration (GOSSAMER-1) via in-flight evaluation of sail attitude control actuators (GOSSAMER-2) to an envisaged proving-the-principle flight in the Earth-Moon system (GOSSAMER-3). First, it turned the concept of solar sail deployment on its head by introducing four separable Boom Sail Deployment Units (BSDU) to be discarded after deployment, enabling lightweight 3-axis stabilized sailcraft. By 2015, this effort culminated in the ground-qualified technology of the DLR GOSSAMER-1 deployment demonstrator Engineering Qualification Model (EQM). For mission types using separable payloads, such as SPO, MNR and RKI, design concepts can be derived from the BSDU characteristic of DLR GOSSAMER solar sail technology which share elements with the separation systems of asteroid nanolanders like MASCOT. These nano-spacecraft are an ideal match for solar sails in micro-spacecraft format whose launch configurations are compatible with ESPA and ASAP secondary payload platforms.
Like any roadmap, this one contained much more than the planned route from departure to destination and the much shorter distance actually travelled. It is full of lanes, narrow and wide, detours and shortcuts, options and decision branches. Some became the path taken on which we previously reported. More were explored along the originally planned path or as new sidings in search of better options when circumstance changed and the project had to take another turn. But none were dead ends, they just faced the inevitable changes when roadmaps face realities and they were no longer part of the road ahead. To us, they were valuable lessons learned or options up our sleeves. But for future sailors they may be on their road ahead.

INTRODUCTION

The development of complete concepts and designs of sailcraft which had been discussed as a means of interplanetary propulsion since the beginning of the space age (Garwin, 1958) was occasionally carried through to the stage of hardware production and full-scale ground testing, e.g. Price, 1981; Seboldt et al., 2000; Leipold et al., 2003, 2006; Young and Adams, 2010. Despite the – in terms of technology development research programmes – substantial effort invested in these projects (Agnolon, 2008; Gong and Macdonald, 2019; Jessberger et al., 2010; Leipold et al., 1999), only simplified and/or sub-scale demonstrators were flown before the single exception, IKAROS (Matsuura, 2010; Matunaga et al., 2011; Mori et al., 2009, 2012). Their hardware development however started slowly. First system-level deployment tests only occurred in the 1990’s on the ground, and in suborbital flight in the following decade. (JAXA, 2004) One of these was the successful ground deployment test of a (20 m)² boom-supported sail on December 17th, 1999 at DLR Cologne. (Seboldt et al., 2000; Leipold et al., 2003, 2006)

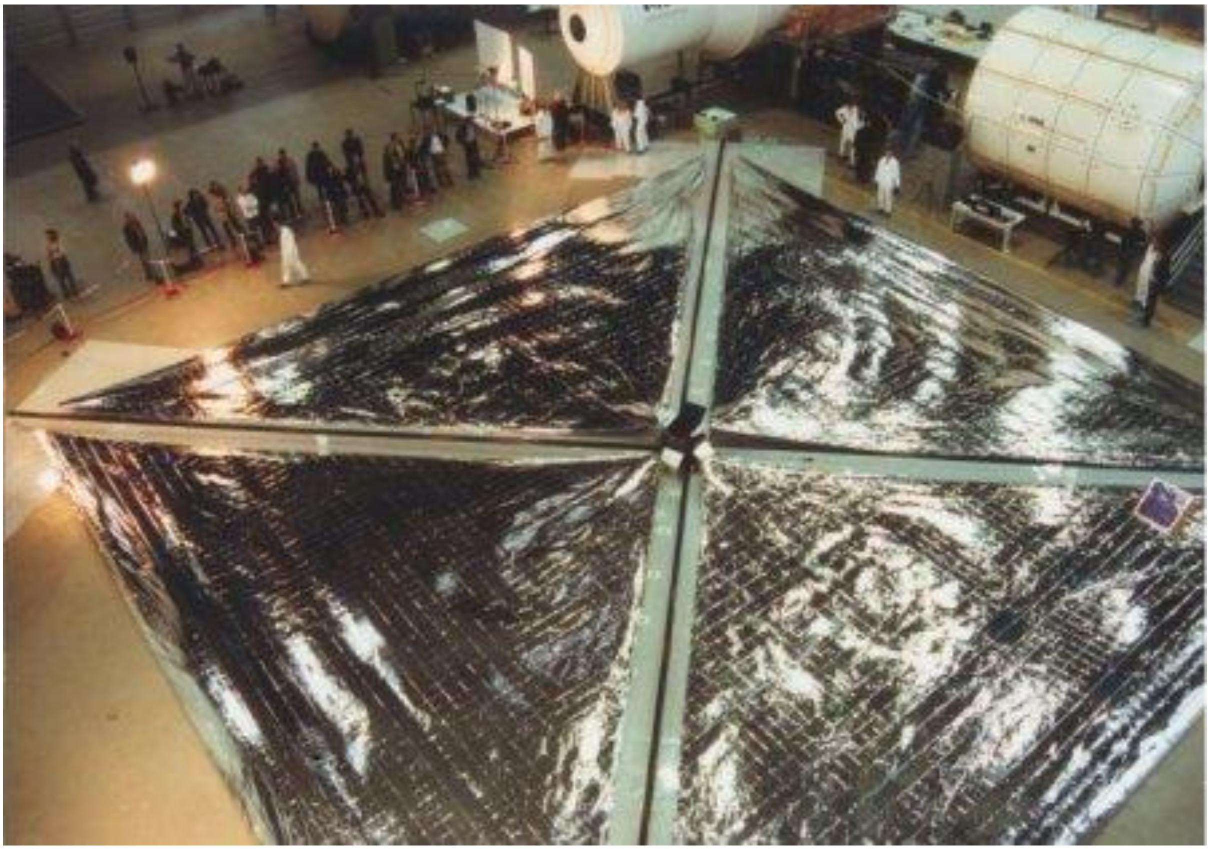

Figure 1 – DLR 1999 solar sail deployment test using membrane quadrants made of 7.5 µm Kapton (left); 4.0 µm PEN (upper right); 12 µm Mylar (lower right) (Seboldt et al., 2000)

It was noted that the performance with respect to sail assembly loading could be substantially improved for larger sails if the hardware components used only once to deploy the sail are jettisoned after the deployment process. This advance over the technology of the 1999 Deployment Demonstration Test would become one key part of the evolution towards the GOSSAMER Roadmap. (Geppert et al., 2011) The successful fabrication and handling of (then) ultra-thin large sail segments (Figure 1) indicated that advanced film materials on the order of 1 µm to 3 µm could be considered.
However, beyond technical factors and features and ultimately chosen designs which we described extensively in our previous publications, another important aspect was also considered in the evolution towards the GOSSAMER Roadmap’s architecture: the way space projects are run and funded in the real world of space agencies. This consideration may appear vague or beside the point to the scientific or technical specialist but it is most tightly intertwined with the managerial and system engineering concept and set-up of the GOSSAMER Roadmap. Often, the paths not taken by its implementation in hardware and studies were left aside when the project’s environment changed. This environment includes the programmatics and policies of

space agencies as well as the political and economical changes of the launch vehicle market, and other such factors. Many of these external factors underwent significant, even radical changes during the time the GOSSAMER Roadmap was actively pursued. One example is the transition of the launch vehicle market from an exclusively government-steered economy via the rise and faltering of conversion launchers towards the emergence of the present ‘new space’ economy. For one, this was completely beyond the control of even the participating space agencies. Another example is the programmatic perspective on and perception of technology development missions versus science-driven missions which is in the realm of space agencies but subject to wider trends in the policies of science and technology as well as national funding strategies.

The demise of the GOSSAMER Roadmap with the cancellation of the GOSSAMER-1 project at DLR left the developed technology in the intermediate state of an Engineering Qualification Model (EQM), a Flight Model (FM) design awaiting its last modifications from the lessons learned in the EQM’s ground qualification, and many studies and reports. These artefacts represent the material form of all the branches and decisions of research and development that led to their completion. In retrospect, all this can appear as a linear, even logical or predictable path which it may have been in an ideal world. This is the path and these are the artefacts which we have described or summarized in all our previous publications.
In this paper, we describe the other paths of development which were abandoned, diverted, descoped or otherwise left behind. Some were part of extensive trades other than the path taken, some followed from disruptions in the project’s environment, for many one led to the other. Here on their own, they may appear as a confusing collection of fragments of varying maturity – which they are. Looking back and including our previous publications, they are the whole picture of the GOSSAMER Roadmap and the GOSSAMER-1 project. Looking ahead, future solar sail projects may choose to or may have no choice but to follow similar routes when their project environment takes different turns than ours did.
For them, we here complete the picture.

Before the Beginning

The perspective of solar sailing is interplanetary spaceflight. Space missions beyond Earth, until very recently were all ‘large’ spacecraft, if not by mass then by funding mechanism and development method. Comprehensive science mission concepts derived from decadal surveys of NASA and similar roadmaps of ESA, are prerequisite for mission selection. Thus, there has always been a tendency to pack every possible instrument onto what is perceived as ‘the only bus out of town’ going ‘there’ in a long time. From the perspective of the funding authorities a mission compiled in this manner becomes a low-risk investment justifiable to the taxpayer.
Not surprisingly, most interplanetary missions strain the limits of their assigned launch vehicles.
Early attempts to find an opportunity to design in detail, build and fly a solar sail followed the same path as any other propulsion module for interplanetary missions. Being new and still experimental, it found itself in competition with well developed, semi-standardized propulsion modules of long-standing flight heritage, co-evolved for decades with gravity-assist plus chemical propulsion oriented planetary science

payloads[1]. As the only available developed method of propulsion, over time, this co-evolution extended into a common gut feeling among all stakeholders on what should be expected of a science payload in terms of performance, mass, volume and power. In the current institutional frameworks for interplanetary missions which are almost exclusively defined by independently set science goals, the traditional perception of 'the payload box' reflects back into mission concept and spacecraft design and stimulates a convergence on a singular mode of realization. But unlike gravity-assists which are essentially spacecraft mass independent, and chemical propulsion which is dependent mainly on fuel to dry mass *ratio* (and not on any absolute mass such as of the fuel carried or of the dry spacecraft), the advantages of solar sails come to bear the better the lower the area loading of the whole spacecraft is. Thus, it immediately becomes obvious that design challenges cannot be resolved by simply scaling up in proportion to the most quickly escalating 'box' in the system. Full overall mission- and system-level optimization of the entire sailcraft is not just a crucial but virtually the sole prerequisite for success.

The very first sail-propelled missions proposed on the science missions tracks for independently defined science payloads consequently would have required a giant technological leap to unrealistically large and/or lightweight sails; indeed, projected sail areas occasionally extended to the km^2 scale. The creation and refinement of all the new basic technologies to reach this performance at a technical readiness level comparable to that of traditional propulsion would have to be done, and done successfully with the commonly acceptable level of assurance, in one, the very first try. Vice-versa, when equipped with a sail of the very largest practical size, a bolt-on combination spacecraft with a traditional science payload would hardly outperform the proven combination of chemical propulsion and gravity-assist fly-bys. Unsurprisingly, solar sail propulsion would then be discarded for the risks inherent in any new technology not justifying the marginal improvement in performance. In this vicious circle the widespread opinion manifested itself that solar sailing never really works in a real mission because it had never been tried and proven in one.

A NEW HOPE – THE GOSSAMER ROADMAP TO SOLAR SAILING

Coupling science missions with main propulsion technology development had proven difficult, prompting a return to pure technology development in the DLR-ESTEC GOSSAMER Roadmap to Solar Sailing, 2009-2014, to be concluded by a full flight qualification campaign. All lessons learned in the preceding efforts were incorporated into the GOSSAMER deployment method which was designed to enable fully controlled membrane deployment, separation of all those mechanisms only required to deploy the membrane, and carefree handling in flight.
On the background of the repeated experience of rejection when it came to 'real missions', the solar sailing community within DLR went back to the drawing board and developed an advanced scalable large lightweight structures deployment technology. The focus was on solar sailing, soon extended to the application of some mass-optimized thin-film photovoltaics to provide a lightweight power source for sailcraft. (Geppert et al., 2011)

[1] M. Macdonald, *Definitions for use by the GOSSAMER Working Groups*: "Payload: The load carried exclusive of what is necessary for operation; especially: the load carried consisting of things necessary to the purpose of the flight, i.e. the science instruments".

Farewell Fantastic Funding

In the years following the 1999 Deployment Demonstration Test these developments on the background of the lessons learned in earlier attempts at getting solar sails into space led to the creation of a bi-institutional framework in the shape of the DLR-ESTEC GOSSAMER Roadmap to Solar Sailing: The key programmatic difference to previous national and European solar sail related studies and projects is its character as a pure technology development undertaking with the explicitly stated complete abandonment of any scientific payload – and thus of any mainstream big mission funding.

Although the resources available to mainstream science missions might initially seem interesting, tempting or of advantage when getting a project like this on the road, overriding scientific objectives at system or project level would later introduce excessive complexity when an entirely new development is forced through as a first-ever mission of its kind. Complexity is perceived as introducing additional risk while the fundamental interest still has to focus on the development of thin film structures deployment methods to a technology readiness level (TRL)[2] that is sufficient for challenging applications. In the end, linking a largely new and first-time technological development to, in principle, unrelated scientific objectives of a conveniently funded space exploration mission would lead to a very difficult project situation. Space agency design standards' requirements for design margins as well as science communities' justified demands for guarantees on the scientific output of the mission would escalate the design into a divergent spiral. Margins added for the uncertainties of a new propulsion technology and (likely also dissimilar) redundancy mechanisms added to cover for not yet long-term flight-proven control methods would recursively leap-frog on top of each other. With the project volume of even a typical agency small class mission at stake, this spiral is virtually certain to continue until either the whole effort of assured development shoots through the cost cap and/or technical feasibility (re)viewed under the layers of margins and redundancies appears to collapse and/or the residual performance of a spacecraft design down-loaded with safety mechanisms and design margins is insufficient to win in a competition selection process. (Grundmann et al., 2013; Montenegro 1999) This vicious cycle was the cause for previous failures of solar sail projects and many other attempts to get game-changing technologies launched, e.g. Findlay et al., 2013.

The GOSSAMER Roadmap Spacecraft

The GOSSAMER Roadmap as originally envisaged consisted of three steps:

- GOSSAMER-1: low cost technology demonstrator, exclusively for membrane deployment technology, with a (5 m)² sail in very low Earth orbit (LEO).
- GOSSAMER-2: validation of solar sail attitude control technologies on a (20 m)² sail at altitudes where photonic pressure becomes dominant.
- GOSSAMER-3: fully functional (50 m)² solar sail to validate the design approach and prove sufficient guidance, navigation and attitude control to conduct planetary science and space weather related missions.

[2] TRL definitions according to Mankins, 2004 were used in GOSSAMER. Other definitions exist.

Note that the size and all other parameters of GOSSAMER-2 and GOSSAMER-3 here and in all of the following are approximates since detailed designs of these spacecraft remain to be completed.

The following table lists the envisaged development and original point-of-departure configurations of the three spacecraft to be developed, at a point in time around the inception of the GOSSAMER Roadmap:

Table 1 – The GOSSAMER Roadmap spacecraft and design goals as envisaged in November 2009 (adapted from Geppert et al., 2010, 2011)

spacecraft	GOSSAMER-1	GOSSAMER-2	GOSSAMER-3
launch in	2013	2014	2015
deployed size	$(5\ m)^2$	$(20\ m)^2$	$(50\ m)^2$
initial orbit altitude	320 km	500 km	>10000 km
mass incl. margin	24 kg	57 kg	≈80 kg
characteristic acceleration requirement	insignificant	unambiguously measurable	$>0.1\ mm/s^2$
performance design goal or expectation based on area & mass	$\geq 0.007 \ldots \approx 0.014\ mm/s^2$	$\geq 0.045 \ldots \approx 0.09\ mm/s^2$	initial operational capability, $\approx 0.2 \ldots 0.4\ mm/s^2$
container volume	$450 \cdot 450 \cdot 500\ mm^3$	$500 \cdot 500 \cdot 600\ mm^3$	$(1000\ mm)^3$
sail foil	7.5 µm	<7.5 µm	<<7.5 µm
reflective layer	2·100 nm Al		
design lifetime	days	≈4 weeks	≈600 days
mission objectives	deployment	limited orbit and attitude control	full orbit and attitude control, Earth-departure capability within ≈100 days, lunar fly-by after ≈600 days
controlled disposal	mandatory, by drag & decay	mandatory, by drag & decay	Earth escape
observation payload	≥2 on-board cameras	≥2 wide-angle cameras, inspector cubesat	≥2 wide-angle cameras, inspector cubesat, narrow-angle camera
launch arrangements	Shtil 2.1, Attached to upper stage on top of QB50 single-block cubesat dispenser		

GOSSAMER-1 – LEARNING THE ROPES

GOSSAMER-1 was intended as a low-cost technology demonstrator for the mechanics and coordination of the membrane deployment process, only. It was to deploy a (5 m)² sail using technology that is already suitable and in part also already scaled for the next step in terms of sail size to be deployed, GOSSAMER-2's (20…25 m)².
We reported on GOSSAMER-1's design and qualification testing in detail in Seefeldt et al., 2014a,b, 2016, 2017a,b,c,d, 2019a,b,c,d; Seefeldt and Spietz, 2016; Seefeldt and Spröwitz, 2016; Spröwitz et al. 2019c, 2017; Straubel et al., 2015; Wolff et al., 2014; Zander et al., 2014, and references therein. In the following, we take a look at all the design options, trades, and operational considerations that ultimately did not come to bear but would have been relevant if the GOSSAMER Roadmap had continued beyond the GOSSAMER-1 EQM.

Raising a New Unit – the BSDU

The key technological advancement from the 1999 Deployment Demonstration Test era to the GOSSAMER Roadmap era was the complete reversal of the entire sail and boom stowage and deployment concept. Both of these key deployables were moved from stowage volumes within the Central SailCraft Unit (CSCU), the center body of the deployed sail, to the boom tips, and all mechanisms and other hardware needed only once for their deployment necessarily went with them, introducing a new modular spacecraft concept: the Boom Sail Deployment Unit (BSDU).
The four BSDUs of a GOSSAMER-type sailcraft are identical self-contained autonomous spacecraft operating synchronously as a self-coordinating deployment flotilla, coupled mechanically only by their temporary attachment to the booms rolled up inside them. Each BSDU unrolls one CFRP boom *end*. For the GOSSAMER-1 EQM, the booms are still very similar in structural technology to the four booms of the 1999 Deployment Demonstration Test (Block et al., 2011) and were on GOSSAMER-1 already scaled for the loads of the (20 m)² class sail of GOSSAMER-2. But now, unlike for the 1999 Test, the four ends are part of only two continuous booms crossing at the centre of the CSCU to form the square sail's diagonals. There, they expand to their full cross-section first, at the location of the highest bending loads in the linear regime (i.e., before it comes to buckling). In addition to half a boom, each BSDU also holds a sail spool on either side carrying one half, each, of the two adjacent sail quadrants.
The deployment process is driven by the boom spool motors pushing the four boom ends out simultaneously and synchronized, thereby gradually unfurling the sails from their slightly brake-retarded sail spools. This gradual and mildly restrained sail release process ensures a minimum amount of circumferential tension already in the deploying sail. This, combined with the simultaneous, but nevertheless sequentially operated unrolling and zigzag-unfolding (compare Fig. 2) guarantees a fully controlled deployment. At any time throughout the deployment process never more than just the needed length of sail is deployed. Thereby the sail is kept from wrapping itself around any other moving parts in a weightless environment as it could if it were released as one package from its container like a parachute commonly is.

This procedure makes the deployment process highly controllable, including the possibility to stop and resume it at any time throughout. Likewise, it enables to use arbitrarily slow deployment speeds and high flexibility regarding speed profiles along

the way. All these are prerequisites also for failure handling and autonomous as well as ground-controlled Fault Detection Isolation and Recovery (FDIR) schemes.
While for GOSSAMER-1 at that time a deployment in one go was planned to accommodate short-duration mission/flight options, the controlled manner of deployment described above furthermore enables other mission operations schemes like e.g. a stepwise deployment scheme. Such a stepwise scheme might be desirable in case, where for whatever reasons deployment progress would be commanded only during passes over ground stations with direct ground control.

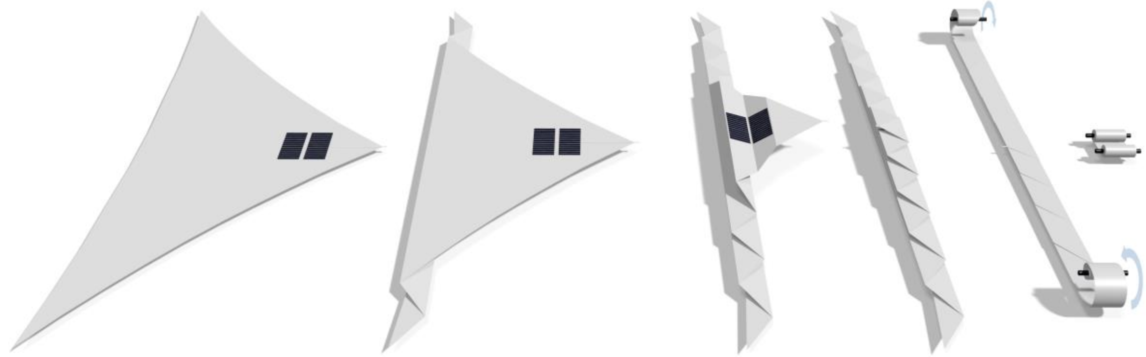

Figure 2 – Solar sail stowing, GOSSAMER style: zig-zag fold and roll up. Deployment is reversing both steps in parallel as the BSDUs at the unrolling boom ends build up distance to their neighbour BSDUs and the CSCU. (Note the dark-blue thin-film photovoltaics section near the central tip of the quadrant to power the deployed sailcraft)

In lieu of any form of strictly synchronized mechanical coupling, as it were possible for a centrally driven push-out deployed boom-sail-structure, synchronized motion is achieved by coordinated autonomous operation of the BSDUs. This reduces the differential loads to a minimum and therefore the net bending moments in the booms during deployment. For this coordination, the BSDUs and the CSCU communicate via radio links, exchanging their parameters of motion. Determination of BSDU position during deployment is achieved by redundant opto-electronic detection of two code patterns laminated onto the booms. (Figs. 3-5) To obtain robustness against count losses due to e.g. straylight entering into the vicinity of the opto-electronics (Fig. 6), recalibration sections were introduced into the code pattern at regular distances along the boom. Likewise, the end section of the deployment was signaled by suitable markers in that code, enabling a retarded approach to the deployment's end position. As an additional back-up, the length of boom deployed was estimated by counting the revolutions within the deployment mechanism. Position information was stored in non-volatile memory to ensure proper recovery even after a full stop and reset of the BSDU's control unit. To enable robust recovery from any type of coordination failure, each BSDU is heavily instrumented for diagnostic telemetry. It also contains its own on-board computer capable of autonomous synchronized deployment control (even in case of CSCU failure) and an independent power subsystem including a photovoltaic generator and a rechargeable battery. GOSSAMER Roadmap spacecraft BSDUs also contain HDTV deployment observation cameras, a feature that will likely be retained by later fully developed GOSSAMER-type sailcraft as it does not impact on sailcraft performance at all, once they are discarded with the BSDU after deployment.

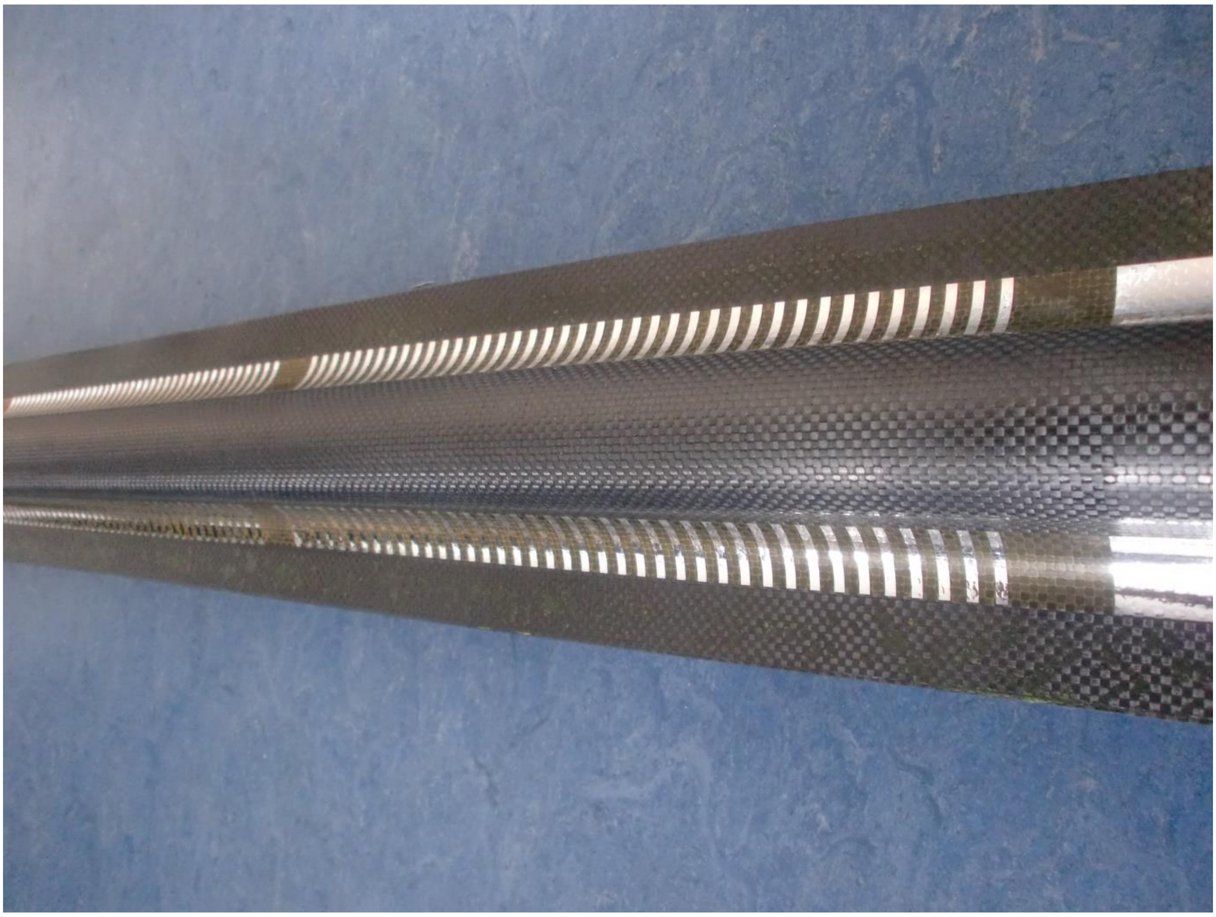

Figure 3 – A breadboard sample of a boom coding strip laminated onto the surface of a boom is shown. Coding strips were fabricated from aluminium coated polyimide foil, which was patterned by laser ablation. The pattern shown was a preliminary test pattern, not yet including recalibration sections.

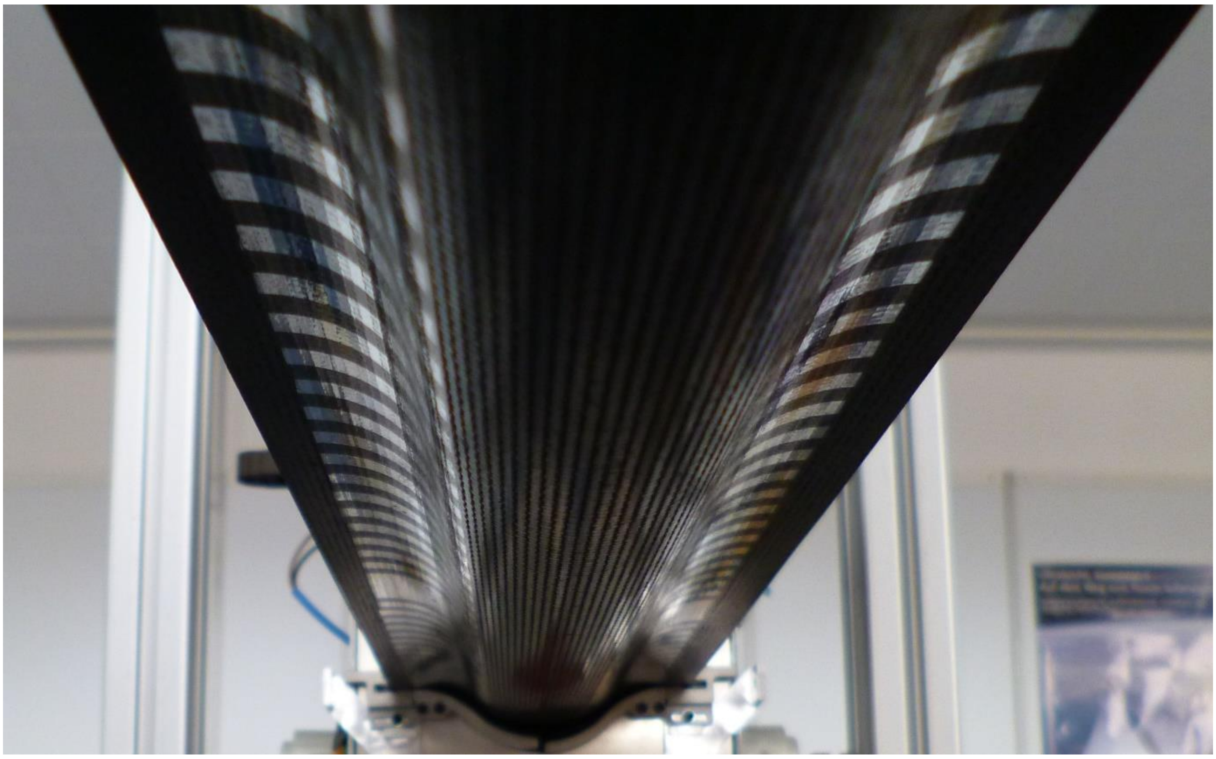

Figure 4 – Engineering Model version of a boom with boom code pattern. In the background the engineering version of a BSDU is visible.

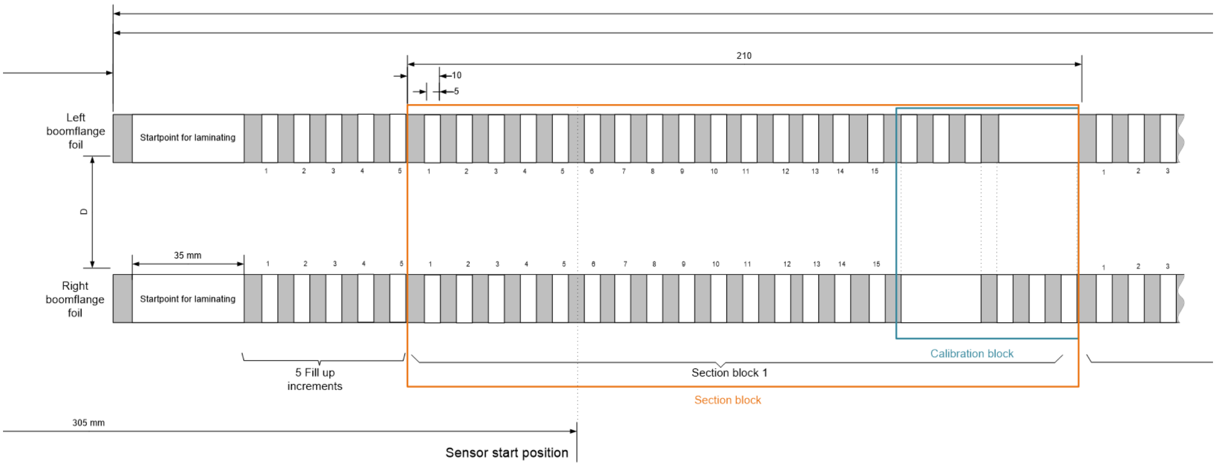

Figure 5 – A section of the boom code pattern is shown, including the first calibration block. A regular pattern of such calibration blocks along the boom enables recalibration at those points along the deployment path.

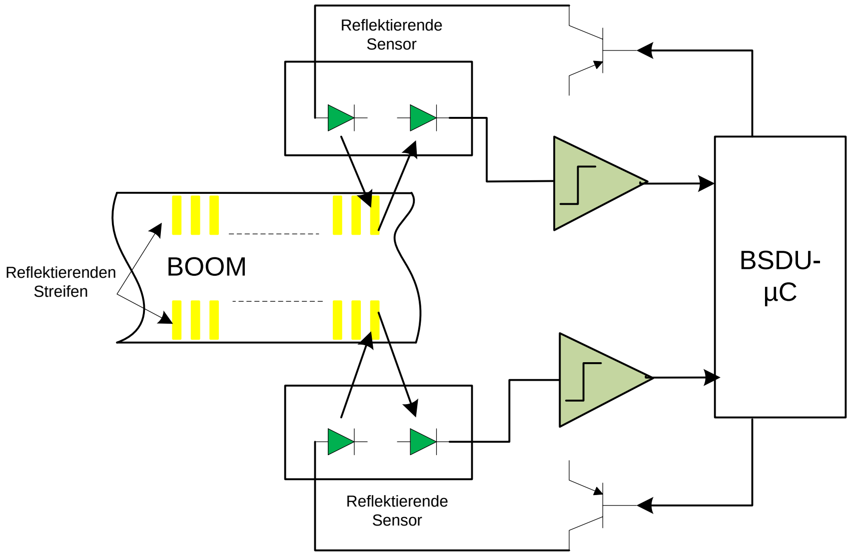

Figure 6 – Concept of BSDU position determination.

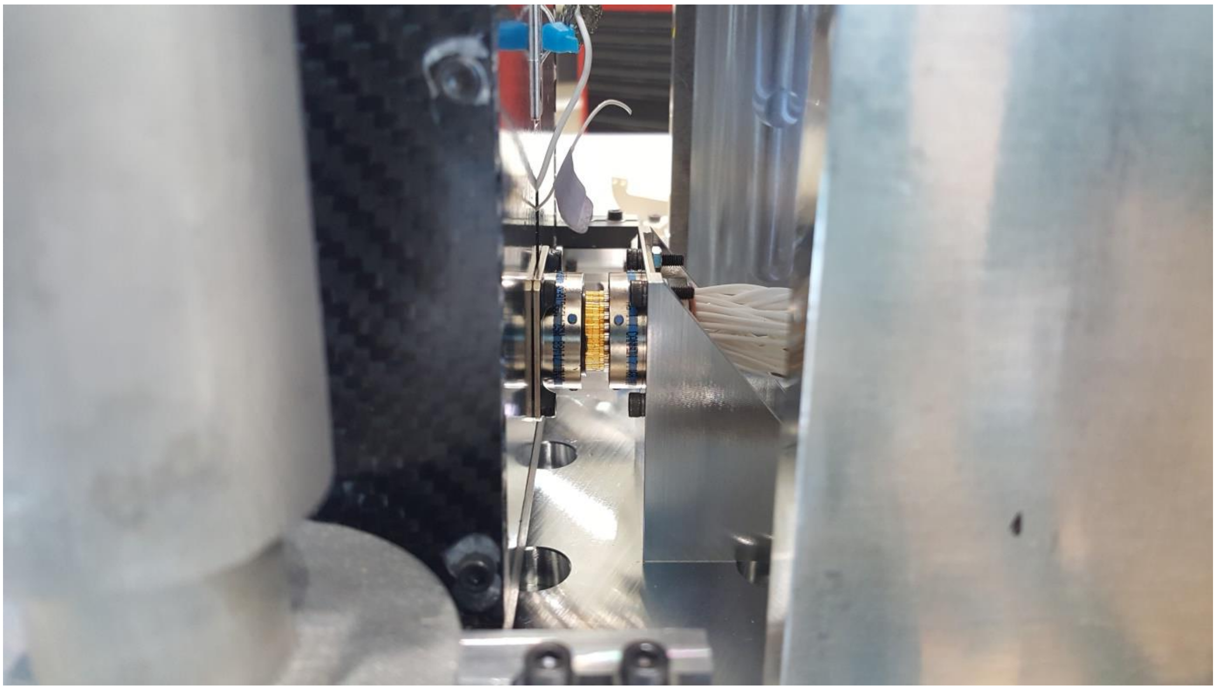

Figure 7 – Umbilical connector between BSDU (left, sail spool in the foreground) and CSCU (right) disconnecting during the separation process

In stowed configuration, each BSDU is electrically connected to a port of the CSCU. This connection makes use of an umbilical connector with individually spring-loaded pins. (Fig. 7) This connector had been previously developed and qualification tested within the PHILAE Lander project but the entire assembly procedure and qualification had to be re-created in a major exercise of experimental archaeology based on the few remaining documents and people with practical experience who had worked on it for PHILAE. In a joint effort of the concurrently running projects, it was adapted to the

needs of the nanolander MASCOT[3] and also GOSSAMER-1. This connector enables mechanically a low force separation and electrically a highly controlled separation. The initial portion of the required separation force is provided by release of the BSDUs pre-loaded launch lock. The spring-loaded pins of the Umbilical Connector provide a significant additional push for the first few mm. Complete separation of the umbilical is then obtained by first activation of the deployment motors.
To ensure safe electrical separation, all high current carrying connections are deactivated before separation. Only low current connections, e.g. for separation indication, remain activated during separation.

At the end of the deployment process, the BSDUs release all rigging elements of the sail, creating the flight configuration of the sailcraft. (Fig. 8) The BSDUs can pause along the way or remain attached at the boom tips indefinitely to enable a post-deployment flight readiness check using all available cameras and sensors still connected. After the deployment of the sail has been completed and verified successfully, the four BSDUs eject themselves from the boom tips using the boom spool motors for the last time, likely at full power. They detach and separate from the boom tips synchronously at a velocity of an order of a few cm/s relative to the tip, not unlike a milli-gravity environment lander's separation from its mothership such as MASCOT from HAYABUSA2. (Grimm et al., 2020; Ho et al., 2017)

[3] A Mobile Asteroid Surface sCOuT, MASCOT, is a specific kind of nanolander developed by DLR. The generic terms are nanolander or nano-scale deployable instruments package, cf. https://www.youtube.com/watch?v=rRi8LptvFZY. MASCOTs and other nanolanders can be made for the exploration of small asteroids or comet nuclei or moons with a milli-gravity environment that makes wheeled rovers impractical and controlled locomotion by thrusters difficult to achieve.

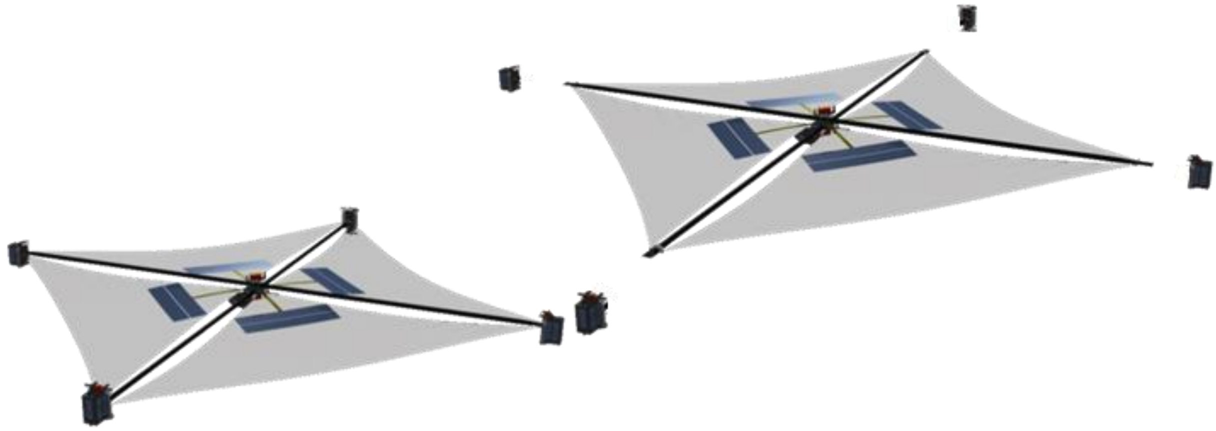

Figure 8 – Solar sail stowing & deployment, GOSSAMER style: the mechanical life cycle of a solar sail

Although likely in almost all cases lacking post-separation attitude control actuators, the BSDUs by the necessities of the GOSSAMER-type sail deployment method are still otherwise complete small spacecraft similar in size and capabilities to 3U or larger 'big-cube'sats. In the GOSSAMER Roadmap flights and likely also in later missions, they would have, after their separation continued to image and monitor the sailcraft receding into the distance as long as they were within useful visual and radio communication range of their cameras and the deployment synchronization wireless network for data relaying, respectively. Depending on the trajectory, direct contact with ground control may be possible if a sufficiently powerful or configurable radio link is foreseen.

Since they remain on the same trajectory or orbit that the combined spacecraft had at sail deployment completion and their mass does not infringe on the performance of the sailcraft in any way, it is quite reasonable to expect that experiments of opportunity and other rideshare payloads will be found that can be carried on one or more of the BSDUs which in any case justify continued communication with ground control.

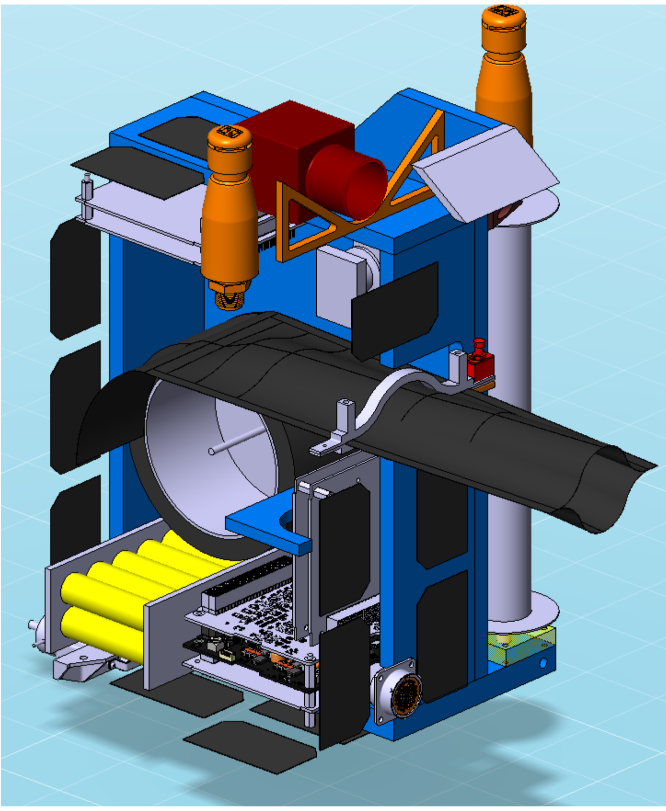

Figure 9 – Inside and outside a BSDU: one complete small spacecraft of four, early version with standard size photovoltaic cells for independent power supply, their carrying structures removed for clarity

It is also unlikely that the BSDU design (Fig. 9) needs to be as optimized in detail as that of the final flight configuration sailcraft part: not having to save all mass possible on the four BSDUs to be jettisoned anyway after deployment may be worth a trade with the envisaged launch concept on the one hand, and on the other with the development effort fraction afforded to the sailcraft proper within the project. Having this trade space, the GOSSAMER solar sail concept lends itself to an integrated Concurrent Design, Engineering and AIV approach where instead of an early mandatory design freeze, the design can be allowed to solidify over time and following the overall project situation and progress, also locally at the level of demand of the individual subsystems and units.

Robustness – cross-training in one spacecraft consisting of 5

BSDU autonomy and mission recovery options

Once started, deployment of the sail quadrants of a GOSSAMER-style solar sail is done and controlled autonomously by the BSDUs placed one at the outer end of each boom. Each BSDU contains:

1. An embedded control Computer which executes the control software.
2. On board radio communication (Bluetooth in GOSSAMER-1) to synchronize the operations with the other BSDUs and to transmit telemetry and pictures to the CSCU.

3. Different sensors to monitor the status of release in general as well as of release mechanisms, forces, speed of deployment and progress of deployment
4. A motor to unroll the booms, including corresponding power electronics to drive it.
5. Local power unit: Batteries, solar cells, power distribution and conditioning.
6. A HDTV cameracovering the BSDU's boom and adjacent sails and the CSCU.

In the nominal case, deployment will be commanded from ground. Ground commands will be received by the CSCU. The CSCU in turn will command the BSDUs to start deployment.
In view of specific mission scenarios, where operation time would be limited to e.g. only a few days by external limitations (short-term battery powered or upper stage attached variants as well as flights into very low orbits), also autonomous deployment scenarios were discussed, initiated by the CSCU. Likewise, in the context of FDIR in case of CSCU failure, start of deployment would be done autonomously by the four BSDUs, after having communicated between them and having confirmed between them that indeed a CSCU failure occurred (and not e.g. just a communication failure between the CSCU and one BSDU).

Once deployment is started, BSDUs communicate their status between them and synchronise deployment speed according to predefined (but ground configurable) speed profiles. In case of non-nominal behaviours these are also communicated and coordinated FDIR measures are taken by the four BSDUs, including e.g. an emergency stop and suitable data acquisition and recovery measures. Speed profiles could also be defined such that halts in the deployment could be foreseen, thereby enabling e.g. the aforementioned option of deploying only during ground contacts.

While deploying, BSDUs acquire images of the deployment process for visual documentation. Images are transmitted via bluetooth to the CSCU, which in the next ground contact will transmit them along with all other housekeeping and monitoring data to ground.
These scenarios might appear to apply only to an initial deployment demonstrator on a brief single-purpose mission like GOSSAMER-1. But the idea can be extended remembering that many solar sail control methods require actuators at the boom tips. Given the size of a future operational fully developed sailcraft, it is much more efficient to power these actuator control units at the boom tips locally and control them via radio link, saving 10’s of m of harness per boom.
In the same way as the CSCU can control the BSDUs’ actions, in principle a BSDU or sail control mechanism unit located at the end of a boom could control the CSCU. Equipped with transceivers able to pass raw command data directly to actuators e.g. in a network-based architecture, this direct delocalized control can be maintained even if a local on-board computer becomes inoperable. For sail control methods without remotely mounted units, accepting a reduction in sail performance, a pair of BSDUs could remain attached to control the sailcraft while keeping it balanced, if and when there was a serious failure in the core spacecraft before deployment is finished by separating the BSDUs.
On a domain-transcending perspective, thermal management by moving around activity dynamically to where and when heating is needed would be a useful addition

to the GOSSAMER integration concept creating one spacecraft out of the CSCU, separable payload modules, and the BSDUs while they are still attached.

Conversely and taking this further, considering the BSDU as being expendable, one can try to limit their avionics to the bare minimum, e.g. implementing just the I/O boards to interface with the actuators and sensors, centralizing then the control software entirely into the CSCU or even an attached separable payload module if the sail structure has no further purpose in a specific mission after its separation upon delivery of the mission payload in its final orbit. This would imply however a robust and responsive interface between the BSDU-integrated I/O boards and the CSCU either via radio link or via extendible cable through the booms.

Thin-film Photovoltaic Experiment (PVX)

As an ultra-lightweight solar sail makes sense only as long as it is also equipped with likewise ultra-lightweight photovoltaics, it came as a natural extension to the originally pure membrane deployment, when thin-film photovoltaics were introduced into the project. Informal cooperation with a DLR Space Agency funded German consortium working on CIGS photovoltaics for space applications (PIPV and PIPV2 “Flexible Dünnschichtsolarzellen für die Raumfahrt” and “Flexible CIGSe Dünnschichtsolarzellen für die Raumfahrt”) opened fruitful ways of knowledge exchange as well as the necessary contacts for this.
As thin-film photovoltaics at that time were (and unfortunately to a large degree today still are) experimental regarding space application, the implementation of thin-film CIGS photovoltaics (Copper-Indium-Gallium-Di-Selenide) was realized as a dedicated experiment.
It is worth noting that the presently frequently mentioned and considered high efficiency thinned GaAs-wafer cells like epitaxial lift-off cells, which are presently still at experimental or early tentative application level, at that time were not yet available at all. Even if they had been available, the larger efficiency of 30% for GaAs versus some 15 to 20% for CIGS need to be traded against the advantages of CIGS thin film photovoltaics in terms of larger radiation hardness as well as lower mass. Membrane stowing and flexibility also need to be considered. The thickness is below 10 µm for CIGS thin-film compared to some ten µm for epitaxial lift-off GaAs plus of the order of 100µm coverglass for radiation protection.
GOSSAMER-1 was to be equipped with standard PV on CSCU and BSDUs as the operational power supply, scaled such that realization of the mission could be guaranteed. In stowed configuration, all operational power (sub)systems of CSCU and the four BSDUs were connected into a common charging network, making use of the aforementioned umbilical connector. Thereby it was ensured that after orbit injection even in case of unfavourable attitude development and resulting unbalanced, “patchy” illumination still all five units would be at the same charge state of their respective batteries at the time of deployment.
Independent of those five operational power systems, the thin-film photovoltaics experiment, PVX, was scaled and implemented such that it would be able to support the mission’s post-deployment operations as well, i.e. sufficient to cover the CSCU’s power requirements. To this end, suitable additional power system electronics were foreseen for the PVX as well as switching capabilities between the PVX and GOSSAMER-1’s standard photovoltaic cell supplied operational power system. The

objective was that after successful deployment and positive system checks, at a given time after deployment and with appropriate orientation towards the Sun, the whole sailcraft, i.e. the CSCU without the then already electrically separated BSDUs, could safely be switched to be powered by the PV experiment. Thereby it would be demonstrated in a credible and realistic way that the thin-film photovoltaics system is able to support and power a mission. This would provide in-orbit demonstration for thin-film photovoltaics as such as well as thin-film photovoltaics as an appropriate power source for solar sailing.

As GOSSAMER-1 was not to be equipped with attitude control capable of orienting the deployed configuration with high agility, a suitable Sun-pointing orientation was to be achieved by acquiring a suitable attitude first in stowed configuration and then spinning up the system to maintain and stabilize that attitude. The maximum spin rate was determined such that it was compatible with resulting loads on booms during deployment. In a second step, deployment would then be started. During deployment, the spin rate would be reduced due to conservation of momentum of inertia and perturbations would be introduced into the system. However, orientation would still be maintained within a sufficiently small solid angle around the original orientation to enable subsequent use of the photovoltaics experiment. This was considered acceptable because even at e.g. 20° deviation from optimal Sun-pointing, incident intensity is only reduced by roughly 6%.

Photovoltaic modules of the PVX were to be realized by two different module-forming interconnect technologies: First, a single cell shingled approach, where single CIGS cells are integrated into a string series connection by shingling them, always placing the back contact of one cell on and connecting it to the front contact of the following cell. This is a standard approach used at that time by companies Solarion (Dresden, Germany, no longer in existence; Figs. 11 and 13) and MiaSolé (U.S.A.). Selected were Solarion cells, as these were manufactured on Polyimide foil providing more light weight material at efficiencies of approximately 10 to 12% at that time (established standard commercial manufacturing line). MiaSolé CIGS photovoltaics are produced on steel foil, which enables higher efficiencies (up to 16 to nowadays 18%), but at a significantly higher mass per area (likewise on established commercial manufacturing line).

As an alternative interconnect technology, so called monolithically structured modules produced on an experimental scientific manufacturing line at ZSW Stuttgart (Zentrum für Sonnenenergie- und Wasserstoffforschung, Stuttgart, Germany – Fig. 12) were foreseen for GOSSAMER-1. Monolithically structured modules are made from semi-fabricated CIGS material obtained in a high throughput roll-to-roll process, which are then structured by suitable laser processes to form series interconnection of cells directly within the CIGS semi-fabricated substrate. The large advantage is that modules come in one piece and no “macroscopic” interconnection is required. This saves effort and mass and reduces risk of interconnect failures.

Modules were arranged on the sail membrane between fold lines so that photovoltaics were not exposed to folding, but only to rolling them up in the second step of the sail stowing process (compare Figure 8). Compatibility with minimum bending radii was checked based on at that time available technical data and was also to be tested within the qualification test campaign. Modules were connected electrically in 4-wire Kelvin manner enabling resistance compensated measurement of current and voltage. Thermistors (also 4-wire Kelvin contacted) were foreseen at each module to enable accurate characterization and temperature compensated maximum power point tracking.

As thermistors two types were foreseen. First, standard off-the-shelf components were selected, which are small enough to be at least kind of compatible with the stowing procedure. Second, ink-jet printed thermistors were foreseen, which had been developed and patented within the GOSSAMER-1 project together with Fraunhofer Institute for Manufacturing Technology and Advanced Materials IFAM (Bremen, Germany). These printed thermistors are realized using sintered platinum patterns on 50µm Polyimide membrane (see Figure 14). Track design and manufacturing is such that contacting with standard FlexPCB technology is possible and a resistance of about 1 kΩ is achieved. Calibration of sensors is done in the course of thermal vacuum cycles by comparison against calibrated test equipment. This type of thermistor enables large area-integrated temperature measurement, which is especially of interest for photovoltaic modules. Thermistors are easily tailored in design and then manufactured according to each desired geometry and size (within ink jet printer dimensions), just as the application requires. In contrast, standard electronics components for temperature measurement usually, while being miniaturized, are still bulky in the context of solar sailing membranes, i.e. not flat, and therefore not ideally compatible with membrane stowing. At the same time due to miniaturisation, they provide information about temperature only at highly localized points. Measurement of temperature across larger areas, as it is of interest in the case of photovoltaics, is next to impossible with such approaches, also because lateral thermal energy flow is negligible in these thin foils.

The electrical harness on the sail membrane was realized using flexible PCB material, i.e. printed circuit boards realized by 17µm Copper on 50µm polyimide laminate. The harness was split into large cross section power harness and low cross section measurement harness. Modules and harness were interconnected by soldered wire jumpers. (Fig. 13) Harness lines follow the fold lines (compare folding scheme in Fig. 8) like branches of a tree, converging towards the triangular quadrant's symmetry axis. At the symmetry axis, the branches are folded inwards and stacked to continue as the harness main trunk along the symmetry axis towards the quadrant's inner corner. There, harness strips are collected and fed into the CSCU. The trunk region along the symmetry axis is not rolled up onto the sail spools. Therefore, it is possible to stow the harness in that region with loose loops, whereby creasing of harness tracks is avoided. (Fig. 10) Thereby the risk of breaks in the copper tracks is avoided.

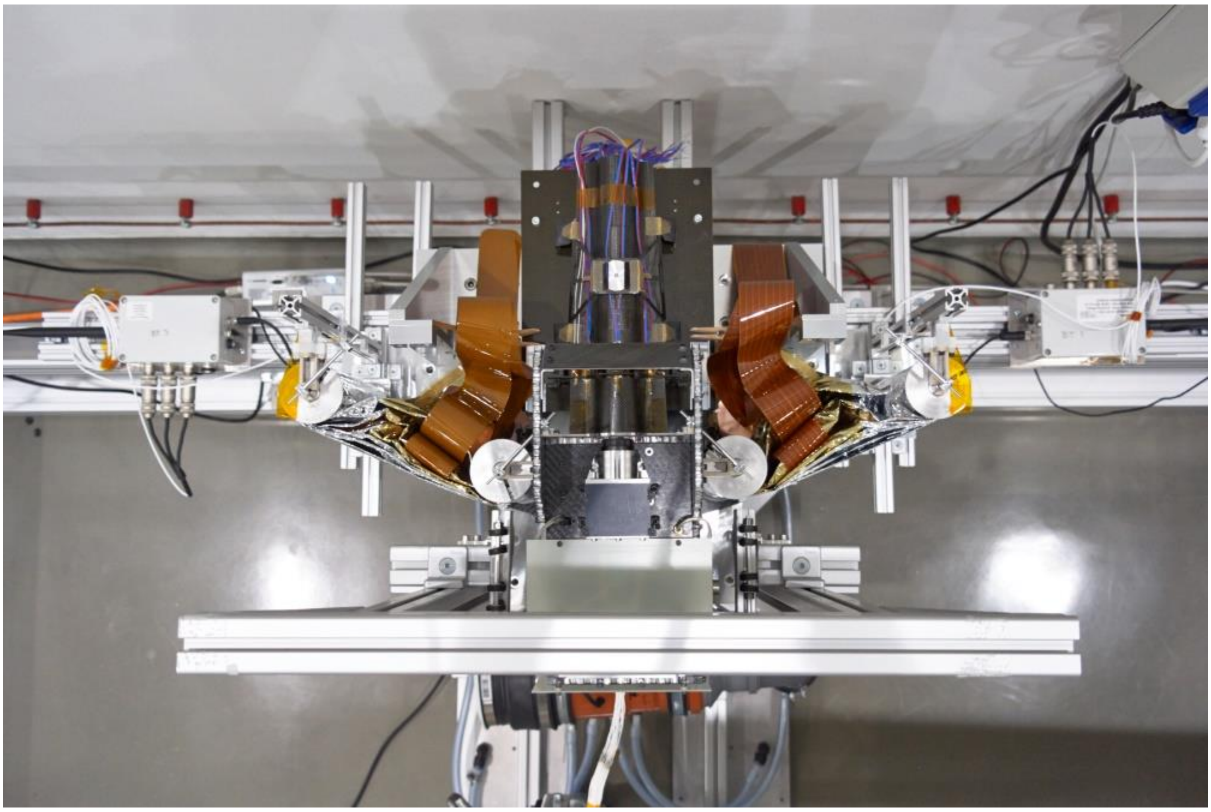

Figure 10 – Top view of the QM BSDU in MGSE with two sail quadrants stowed and attached to it at each side. To the left and right the linear drives are visible, by which two further BSDUs were simulated. Thereby controlled deployment of two quadrants with one fully representative boom and BSDU was possible. Between sail spools the loops of the PVX harness are accommodated.

To reduce cost, not all module positions were equipped with thin-film photovoltaic modules. Instead, a number of mechanically representative dummies were also integrated onto the qualification model sail segment. (Figs. 11-13) Modules were integrated at alternating orientations so that magnetic cleanliness was achieved as well as risk of arcing in the process of opening the folds under sunlight was reduced.
The photovoltaic modules were coated using polysilazane precursor solutions and coating techniques developed at the University of Bayreuth (Germany) in the course of the PIPV and PIPV-2 projects. The procedure provides an approximately 2 µm thin but nevertheless rugged coating with SiO_x. This coating serves several different purposes at the same time: increase of infrared emissivity to reduce heating of modules, thereby increasing photovoltaic efficiency; moisture protection during handling on ground; mechanical protection during handling on ground as well as during stowing, launch and deployment; electrical insulation required during stowage and especially during deployment. This is relevant in the course of opening the folds, when modules lying face to face within that fold will be partially exposed to sunlight while at the same time in lower parts within the fold they are still partially in mechanical contact.

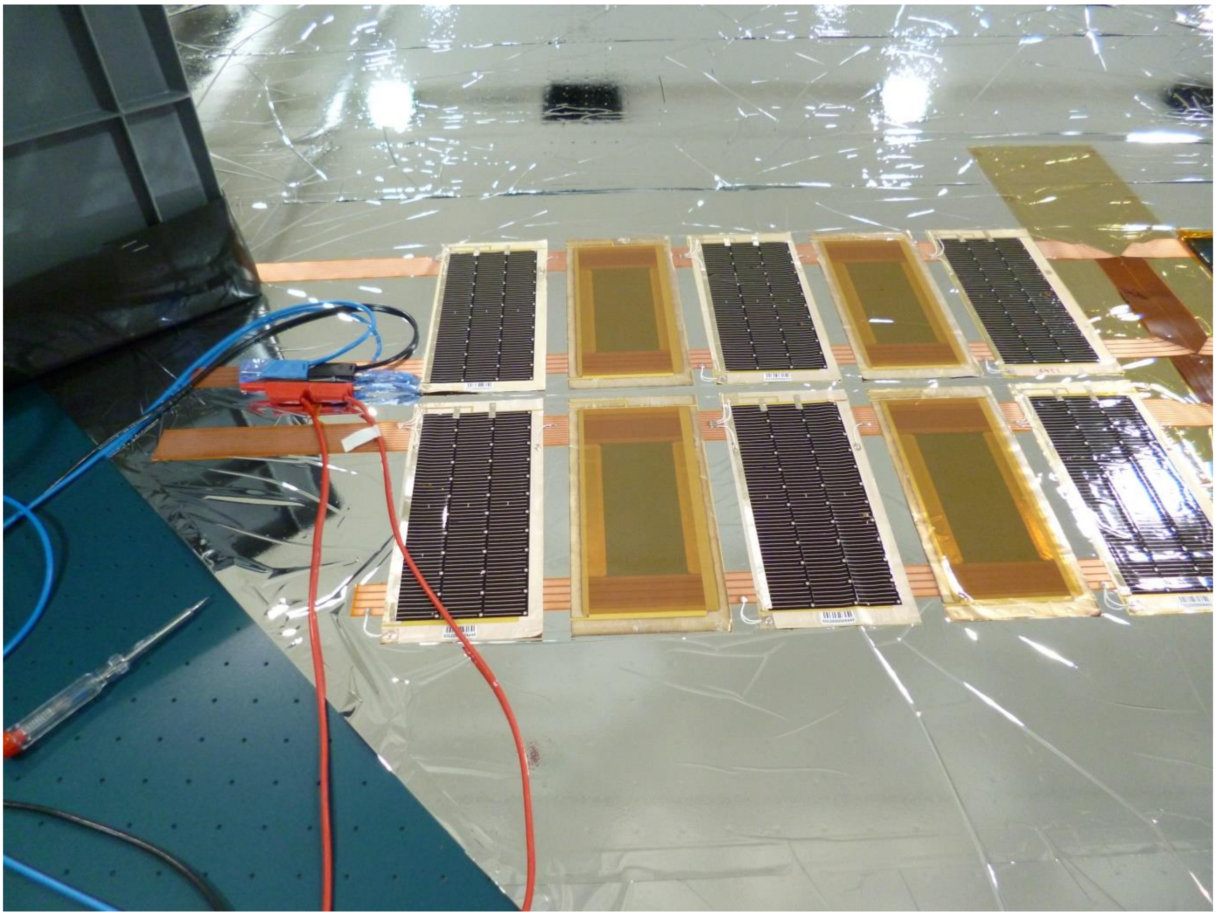

Figure 11 – QM sail quadrant during characterization measurements, showing Solarion mini modules made of three shingled Solarion CIGS cells, each mini module on a dedicated FlexPCB module substrate. Note the soldered wire jumpers between harness and modules.

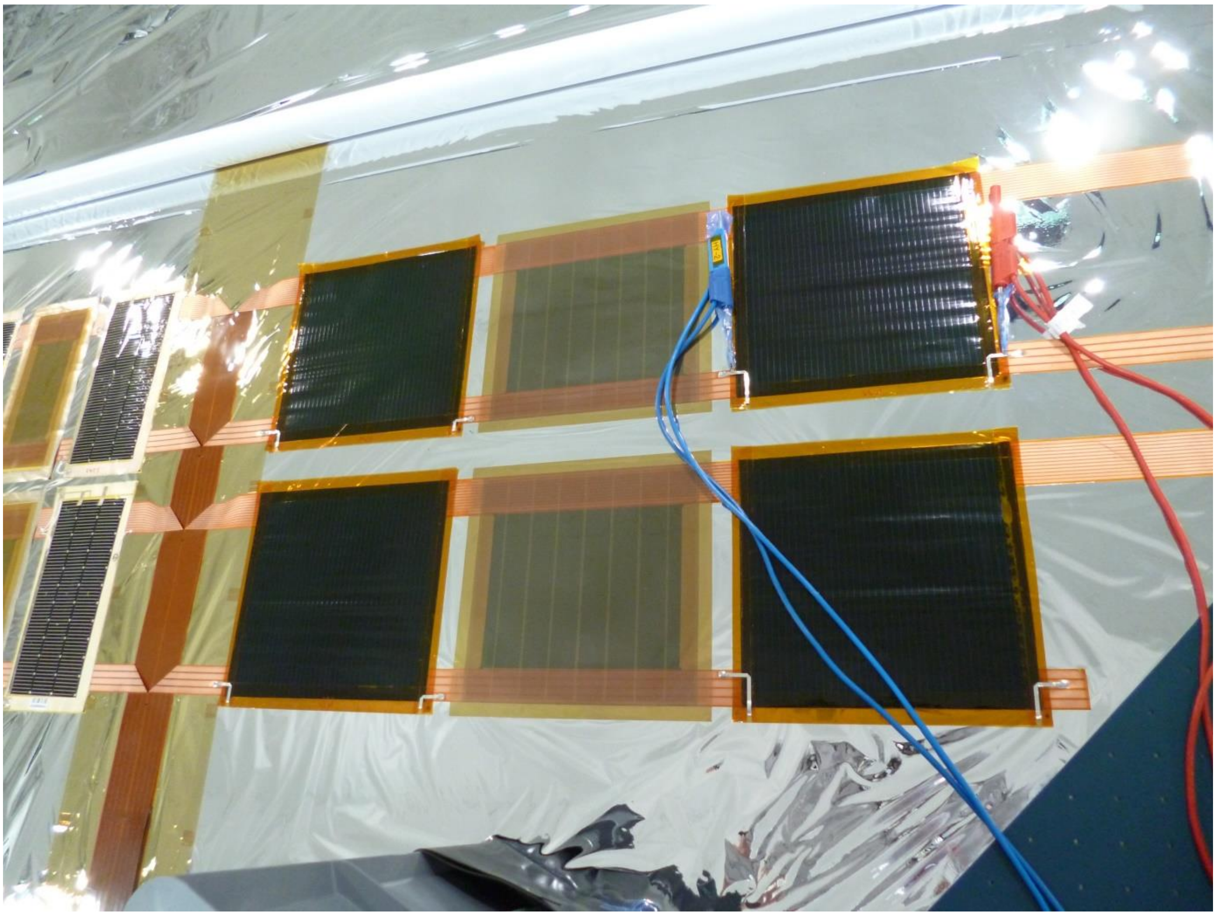

Figure 12 – QM sail quadrant during characterizational measurement, showing ZSW monolithic modules (18 cm)² consisting of 36 cells of 0.47mm, each. To the left the central trunk of the photovoltaics harness with the off-branching sections connecting to the modules is visible. On the other side of the main trunk the previously shown Solarion mini module section is located.

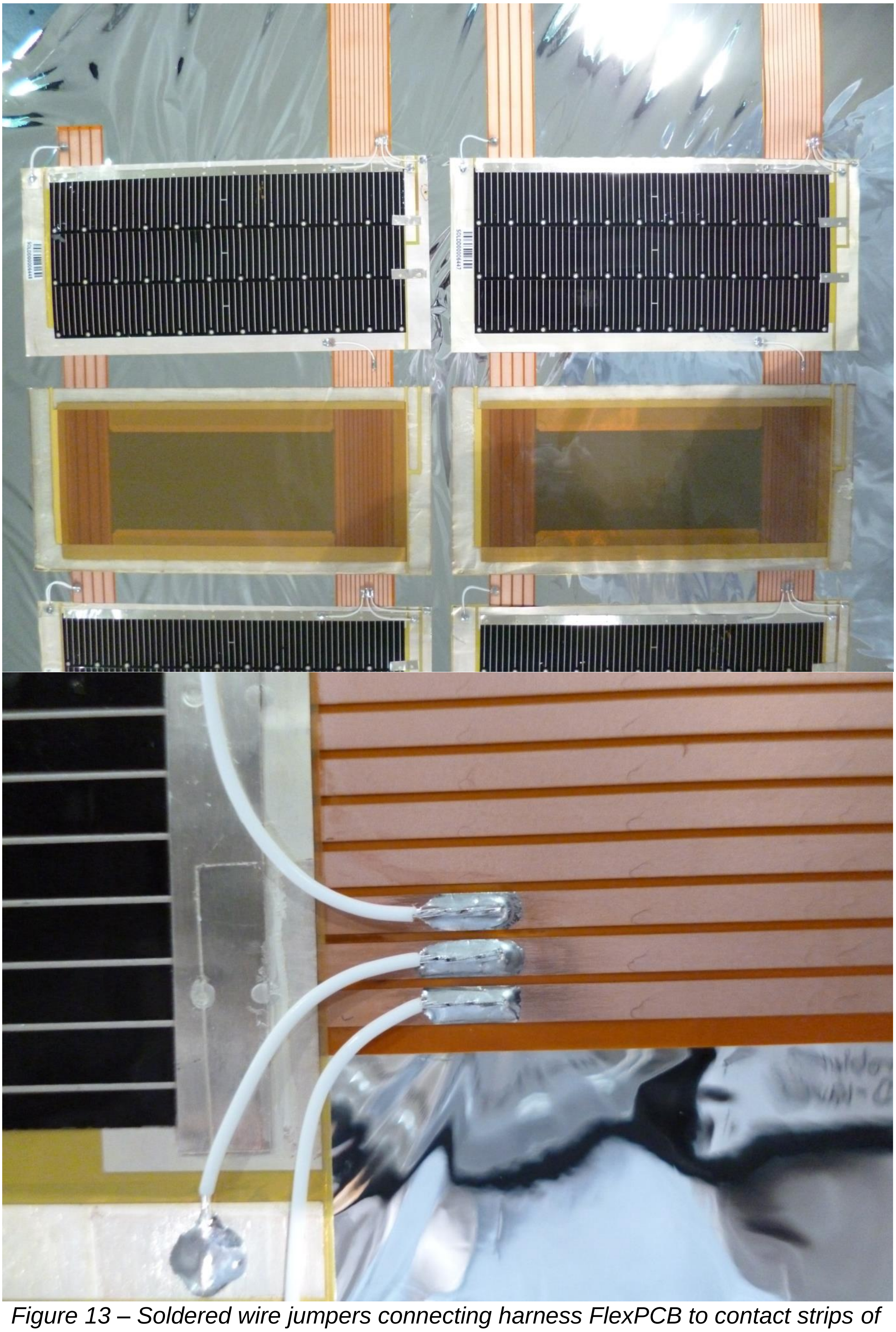

Figure 13 – Soldered wire jumpers connecting harness FlexPCB to contact strips of a mini module.

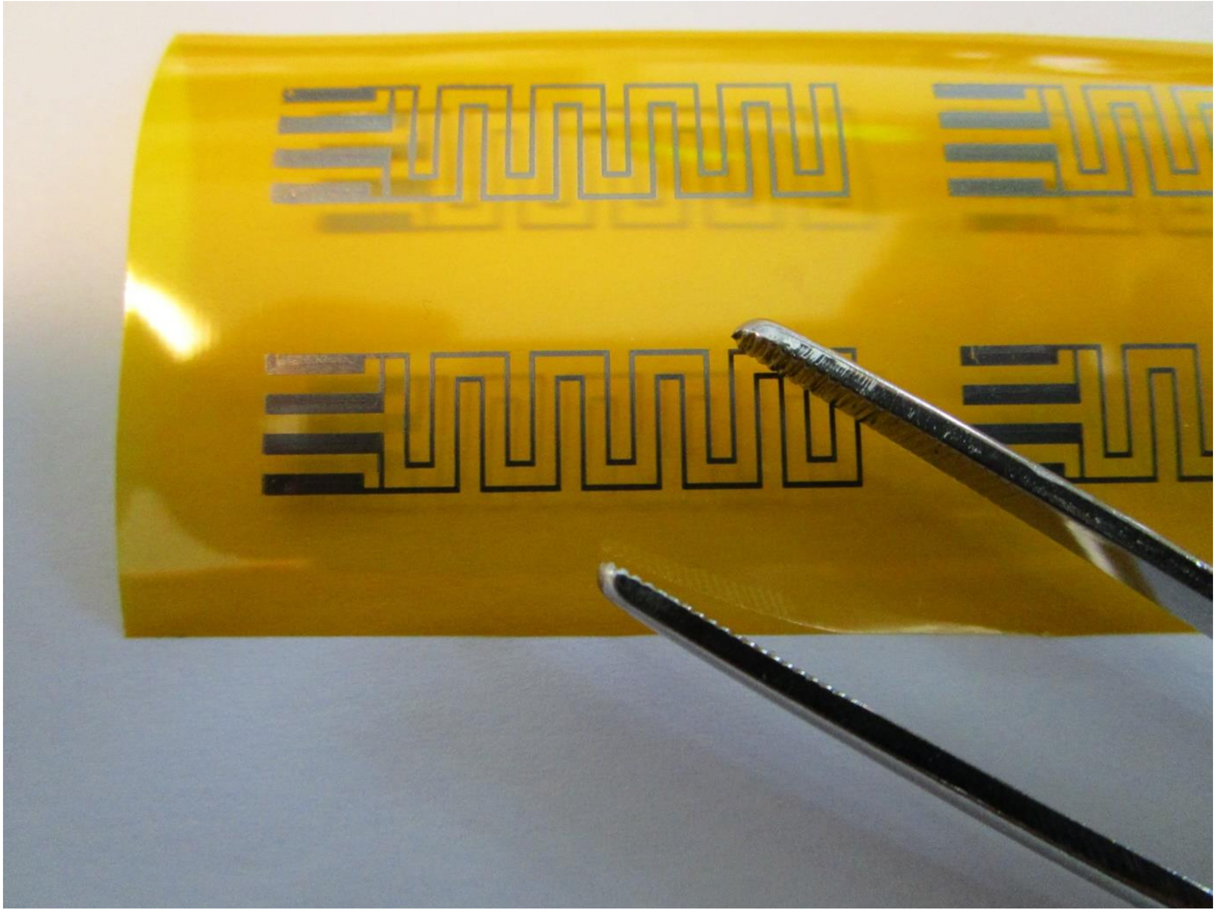

Figure 14 – Ink-jet-printed thermistors with four contact pads for 4-wire Kelvin measurement are shown.

Thoughts for a future that never was – the GOSSAMER-1 configuration options

Entirely focused on the process of deployment, GOSSAMER-1 did not need attitude control. However, a stable orientation with respect to the velocity vector based on atmospheric drag after deployment was not excluded, and could have been tweaked out of the design to enable disposal from relatively high altitude LEO (low Earth orbit) rideshare launches.
GOSSAMER-1 could also have flown in very low LEO, even below the altitude of the ISS (cf. McDowell 2018), but was compatible with all orbits that ensure orbital decay of all spacecraft units within 25 years based on the best estimate of solar effects on the upper atmosphere. (ISO 2011)

The minimum initial orbital altitude of GOSSAMER-1 can be derived from a mission duration just long enough to establish reliable telecommand and telemetry (TC/TM) contact, deploy the sail, and ensure full data return. This would be of the order of a few days before orbital decay of the fully deployed configuration oriented at maximum drag. The resulting minimum deployment altitude is then based on the assumption of flying in a parachute or shuttlecock configuration with the full sail area exposed to atmospheric drag immediately after a first-try successful deployment. The data budget and communication subsystem were adjusted to the brief passes of very low orbits

down to the onset of re-entry as well as the larger distances of the higher orbits commonly available for LEO rideshares.
This assumption was also compatible with an early project stage launch configuration in which GOSSAMER-1 would have stayed attached to the upper stage or any stably orbiting latter stage of the launch vehicle, in its disposal orbit. Due to the large offset of the center of gravity of such a configuration, a damped oscillation transition into a shuttlecock configuration is very likely once drag becomes significant during orbital decay.
The last baseline design of GOSSAMER-1 was an independent free-flyer spacecraft optimized for an as wide as possible spectrum of secondary payload ("piggy-back") launch opportunities. (Figures 15 & 16) We reported on this design in all our publications since 2014.
A variant of this baseline design optimized for an upper-stage-attached mission has been studied in detail, including very short lifetime purely battery powered flights. (Figure 17) This variant also had the option to have no battery of its own, instead using any surplus upper stage battery capacity while acting as passivation load in return. Communication to ground stations may also have used the upper stage's telemetry equipment as far as possible. It is only marginally less complex than the free-flyer, partly because readily off-the-shelf available units are already designed for longer lifetime orbital missions, e.g. off-the-shelf power subsystems always include photovoltaic converters. Thus, the design of GOSSAMER-1 is prepared for all possible launch opportunities and supports very late configuration changes to accommodate even quite radical launch-specific requirements such as mandatory separation or attachment.
A somewhat longer orbital decay lifetime would however have been welcome, to study possible sail degradation effects in space and within the upper atmosphere and to operate the thin film flexible photovoltaics experiment (PVX) to evaluate the effects of the stowing and deployment processes on thin-film solar cells (TFSC).

The maximum launch insertion altitude can be derived from the orbital decay lifetime of the highest density item to be separated. If a full deployment is demonstrated, these are in almost all cases the BSDUs since burnt-out upper stages already have rather low density even if the deployment of the sail foil were unsuccessful. Yet higher altitude launch opportunities were possible but would have required the deployment process to end without BSDU separation, with the option to complete it once the spacecraft had decayed to an altitude that ensures re-entry of the BSDUs within 25 years after their end of operations, if it were still alive and responsive.

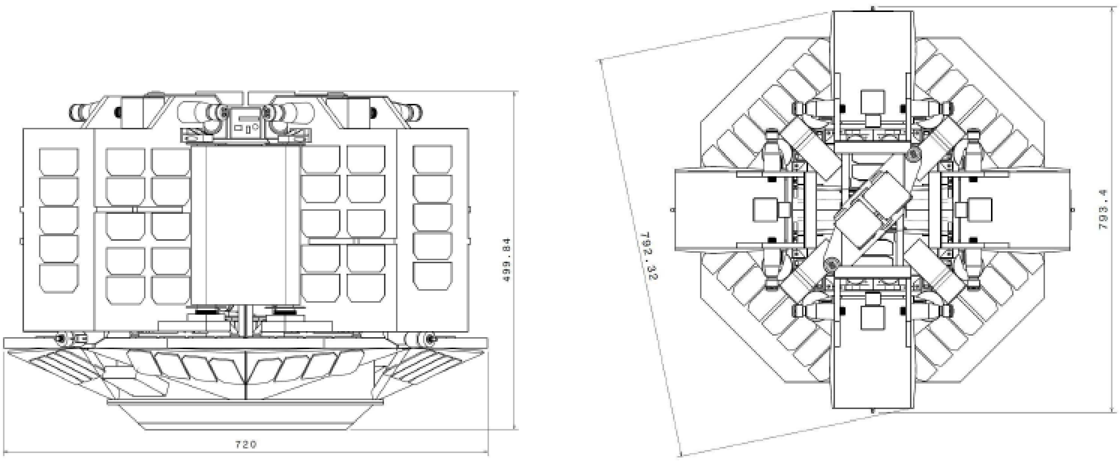

Figure 15 – GOSSAMER-1 latest design, baseline free-flyer variant with optional operational photovoltaics based on conventional triple-junction cells (placeholders shown, only) for independent operation separated from its launch vehicle's upper stage but capable of stage-attached operation in a contingency

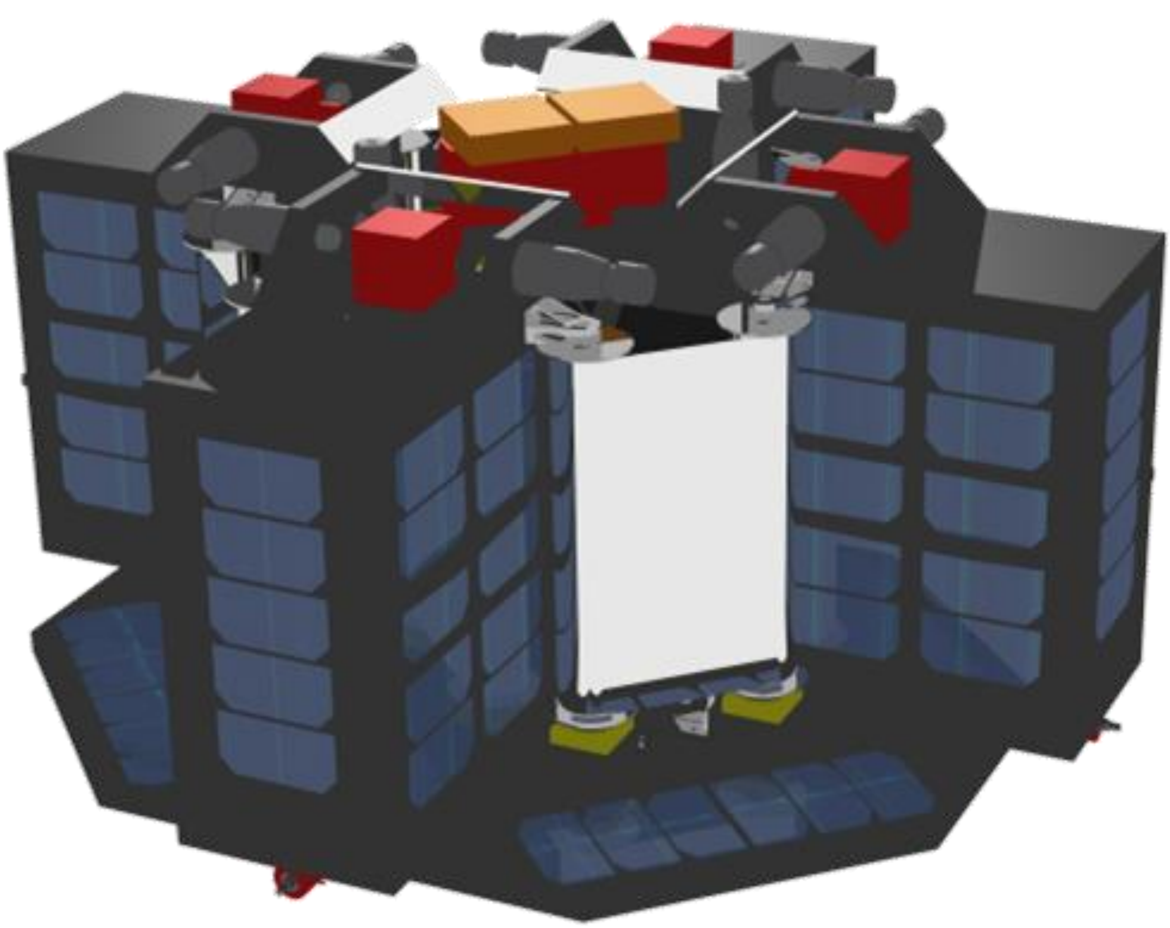

Figure 16 – GOSSAMER-1 independent free-flyer design CAD view. Photovoltaic cells shown are CAD placeholders.

The design was carried forward to the point where hardware integration of an engineering qualification model (EQM) had been completed. The AIV process was

carried out by a very small residual project team, in parts using an adaptation to a model strategy containing only one comprehensive model, of the Concurrent AIV (C-AIV) approach pioneered in parallel by and with MASCOT. (Grimm et al., 2013, 2015a,b, 2019a,b; Grimm and Hendrikse, 2019; Grundmann et al., 2013a,b; Seefeldt et al., 2018) The GOSSAMER-1 EQM consists of one fully functional train of the deployment relevant units and two adjacent membrane quadrants. It subsequently completed qualification-level testing, including a ground deployment test in TVAC space qualification environment only restricted by the size of available facilities. The GOSSAMER-1 EQM is applicable for all possible GOSSAMER-1 launched system configurations, from upper stage attached payload to independent free-flyer.
We report on further detail of GOSSAMER-1 qualification in Seefeldt et al., 2016, 2017d; Seefeldt and Spröwitz, 2016, and references therein.

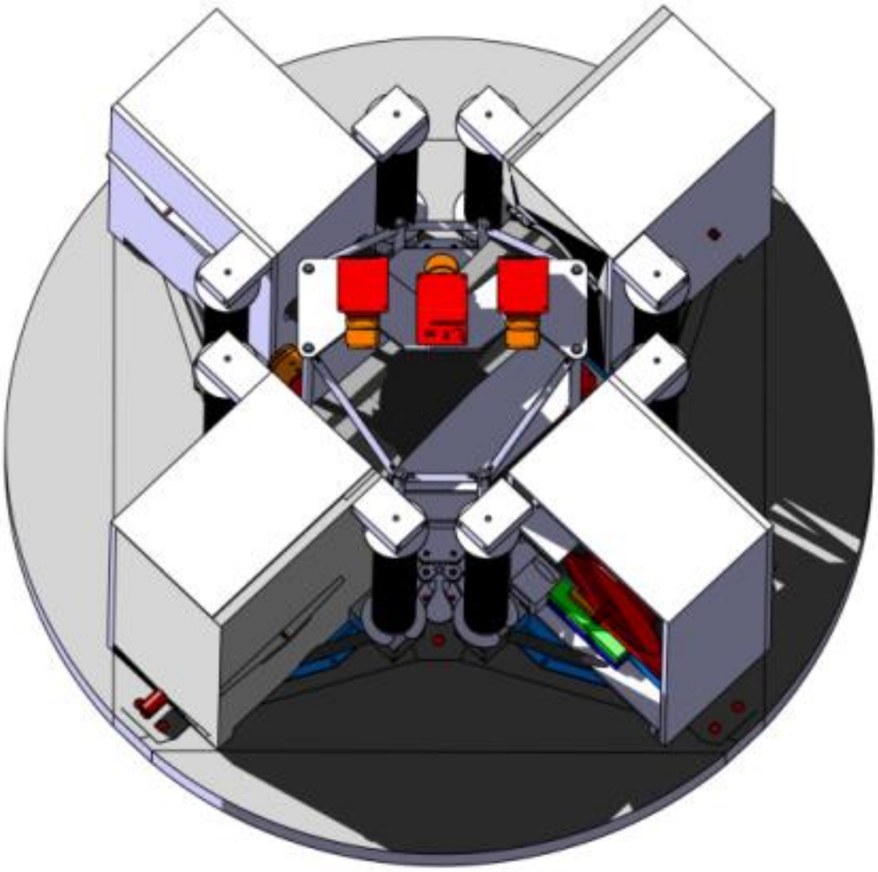

Figure 17 – Alternative, very short lifetime 'one-shot' mission GOSSAMER-1 variant for upper stage attachment; battery-powered, may act as passivation load to use up any surplus upper stage battery capacity; ground communication may also use upper stage telemetry equipment as far as possible (cf. Fig. 26)

Using a proto-flight model (PFM) based approach and C-AIV methods with an AIV team, reinforced by personnel released by other projects such as MASCOT and HP³, a GOSSAMER-1 full 4-quadrant demonstrator spacecraft in free-flyer configuration could have been flight ready within about two years from successful conclusion of the QM campaign. This estimated timeline assumes some concurrent integration preparation work progressing already in parallel to the QM testing campaigns, and depends on continuity of full-time assignment of staff with GOSSAMER project experience, and to a lesser degree also on test facility availability. These in combination had been the three driving schedule constraints for MASCOT. (Grimm et al., 2019a; Grimm and Hendrikse, 2019)

GOSSAMER-2 – UP & DOWN THE LADDERS

Next to being an intermediate step in the development, size and area of the membranes and booms to be deployed, (Block et al., 2011; Hillebrandt et al., 2014; Meyer et al., 2015) GOSSAMER-2 was primarily envisaged as a validation mission for candidate solar sail attitude control technologies using a sail in the (20 m)² size range. As the source of propulsive force of a solar sail cannot be turned off (Dachwald et al., 2007; McInnes 1998; Spencer and Carrol, 2014), trajectory control of a sailcraft crucially depends on effective sail plane attitude control for permanent thrust vectoring. However, compared to most other propulsion methods, quite a large number of fundamental attitude control principles have been proposed for solar sailing, with an even larger variation of suggested methods of realization. (Gong and Macdonald, 2019; Wie, 2004) Thus, an equally important task in the development of GOSSAMER-2 would have been the detailed design, modelling, and down-selection of these attitude control principles to determine the best candidates for the envisaged future sail-based science missions, to test them in design, construction and flight:

- direct sail control
    - pitch & yaw axes
        - reefing – local retraction of a section of each quadrant
        - sliding trim masses – movable masses in the spacecraft central section (bus) and/or within the booms
        - gimballed trim mass boom – 2~3 degrees of freedom (DoF) movable mass at the end of a 5th deployable boom
        - reflectivity control – active surface technologies to locally change the reflectivity on the sail quadrants (e.g. IKAROS)
    - roll axis
        - 'T'-bar booms – roll stabilizer bars at the outer end of each boom tilting the adjacent sail quadrants' planes in a propeller-like fashion
    - all axes
        - 'LL'- or '┐┌' or 'split-T'-bar booms – pitch-yaw-roll stabilizer bars at the outer end of each boom independently tilting each quadrant tip, to tilt the sail plane as a whole and/or quadrants independently in a propeller-like fashion
        - flaps or tip vanes – 2~3 DoF movable solar radiation pressure trim sails, deployed from and mounted at the boom tips
- indirect control
    - bi-axial
        - magnetorquers – wear-free, propellant-less method of momentum generation around two axes perpendicular to the local magnetic field at a time, but only efficient within planetary magnetospheres, i.e. at Earth (acceptable for GOSSAMER-2 or early operation phases of later flights) and the gas giants

- tri-axial
    - reaction wheels – near-instantaneous reaction, can be configured to account for varying moment of inertia (MoI) of the spacecraft main axes and/or multiple redundancy, but require regular and/or continuous desaturation and are subject to wear
        - reaction wheel desaturation may be continuous using control mechanisms of the membrane(s) or magnetorquers, leaving only fine adjustments, brief rapid or/and precision turns to the wheels, preferably in a back-and-forth manner returning the sail to the previous attitude or modulating smallest possible difference angles on a mean turning motion mainly actuated by membrane(s) control
    - thrusters – instantaneous reaction, but limited by propellant supply, and plumes may adversely affect the sails; thrust pulses may induce long-term oscillation of the relatively soft lightweight sailcraft structures; also, they re-introduce a mission-critical limited supply item into the system which contradicts the key advantage of solar sailing

All these methods were investigated and have been studied throughout the work not just leading up to and continued under the GOSSAMER Roadmap, but in the solar sail community in general over many years, e.g. Wie, 2004. With the exception of thrusters on the fully deployed final configuration of the sailcraft after BSDU separation, none of these methods were a-priori excluded from the design of GOSSAMER-2 by the end of Roadmap-related work. With the expansion and development of the rideshare launch market in the past 10 years, carrying a reasonable amount of propellant became acceptable also for secondary payloads, so that a thruster option for attitude control recovery purposes would likely be considered feasible today for a mission similar to GOSSAMER-2.

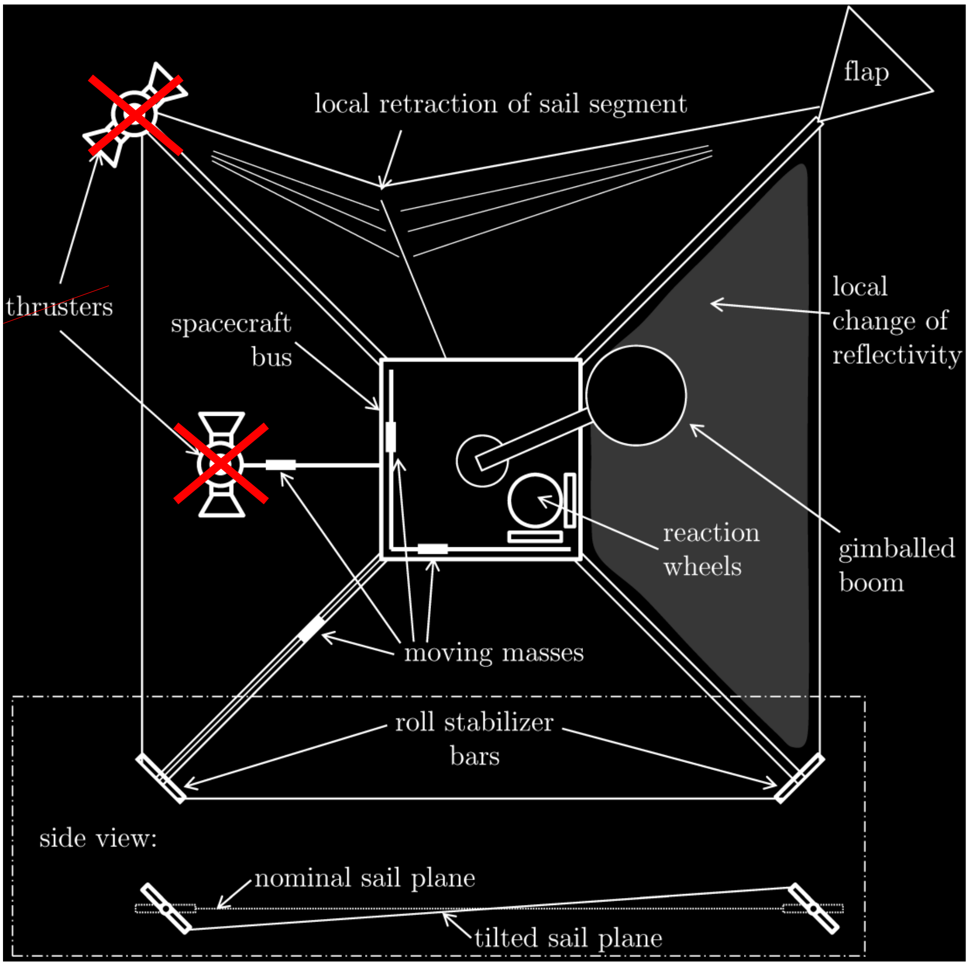

Figure 18 – "Optional solar sail attitude control methods (in a not advisable combination, size of spacecraft bus and sail are not to scale)" adapted from Fig. 10 in Dachwald, 2010: Perhaps not advisable for a developed solar sail spacecraft but, in a much more symmetrical and balanced form, not an unlikely concept of the configuration for GOSSAMER-2

Phasing In Control

The final selection of the control methods to be integrated and test-flown on GOSSAMER-2 would only have been made immediately prior to the beginning of hardware construction and integration when all experiences from GOSSAMER-1 would have been evaluated. This would place some of the mission concept level decisions commonly addressed in Phase 0 or latest A1 at the Phase B to C transition milestone of GOSSAMER-2 in the conventional phased concept of project timelines, or at an equivalent point in time at the subsystem level in a less phase-emphasizing AIV concept. In the staged PFM model construction philosophy that the GOSSAMER Roadmap envisaged for GOSSAMER-1 to -3 and with the originally envisaged biannual launch interval timeline this approach maximizes the time available for a concurrent process of comprehensive review and evaluation of recent progress in solar sailing studies, system-level configuration trade studies, and detailed design.
However, following the MASCOT experience (Grimm et al., 2019a; Grimm and Hendrikse, 2019), a wholly Concurrent AIV/AIT approach has become very likely for spacecraft projects of a comparable effort. Concurrent AIV accepts integration and

testing schedules without pre-set discrete phase boundaries which are based on milestones of common achievement of a uniform maturity status of all units within the spacecraft design. Rather, it embraces vastly different initial maturity levels among the subsystem or functional units and caters for several parallel tracks of test campaigns and corresponding spacecraft models which enable the development process to focus on the lagging elements' growth into the design space defined by the interfaces set up in a partially completed system consisting of the elements leading in maturity. Thus, the detailed design, testing and integration of the full-scale GOSSAMER-1 to -3 PFMs and the respective partial-system- and/or unit-level BBMs, EMs, EQMs and QMs might have progressed semi-independently as knowledge and practical experience had been accumulated, designs had become ready, and hardware had become available. In this way, it is possible that still more time margins had been set free in the schedule to focus on the development of those parts which were likely to still have lagged in maturity.

Since the envisaged staged implementation of GOSSAMER-1 to -3 within the Roadmap created a consistent and – within the procedural constraints of the funding environment – reliable development environment, strategic re-use was not just a desirable option but a likely course of action from the start. It was implemented already in the beginning of the (5 m)² GOSSAMER-1 for the mechanical, membrane-related and mechanisms design, all of which was sized for re-use in the (20 m)² GOSSAMER-2. Extensive re-use of avionics and software became an increasingly more important and likely aspect as the design of GOSSAMER-1 progressed in maturity; many opportunities for re-use were not anticipated in the conceptual stage but became obvious in practical work. Taking into account the overlapping test and integration campaigns of GOSSAMER-1 to -3 in the Roadmap scenario as well as an end-to-end concurrent engineering-and-AIV approach initiated by a team and institute with fresh prior experience in it (see Lange et al., 2015a, 2017, 2018a,b, 2019), ample short-term ad-hoc direct (or 'tactical')[4] re-use could also have been expected which is not possible to the same extent in completely sequential roadmaps and rigidly 'phased' projects, and probably impossible when contractors change from flight to flight or phase to phase.

Up to the Shorelines – taking turns in the harbour

GOSSAMER-2 was also required to demonstrate sail effect with high confidence, i.e. based on a quantitative trajectory prediction and analysis, but in terms of overall mission design this is hardly a mission-driving requirement: Sail effect demonstration is expected to be feasible already in a somewhat higher LEO, actually at any altitude where air drag becomes insignificant for the given sailcraft design after completed deployment. This would have enabled GOSSAMER-2 to take advantage of the bulk of secondary payload ("piggy-back") launch opportunities. These arise in the otherwise unused launch vehicle performance margins of civilian Earth-observation satellite main (primary) payloads. Most of those are launched to approximately polar Sun-synchronous LEO (SSO). Almost the complete altitude range generally used by this class of spacecraft is compatible with all known technical and possible programmatic constraints on the GOSSAMER-2 mission; from approximately 500 km, a typical radar

[4] 'Tactical' as opposed to premeditated or pre-planned 'strategic' re-use which becomes part of a project's system engineering concept very early on based on some form of anticipated continuation.

satellite altitude mostly found in dawn-dusk SSO configuration, to some 800 km, a typical polar weather or wide-area imaging satellite altitude mostly found in morning-pass SSO configurations. Also regarding their illumination conditions, SSO configurations are usually expressed by the Local Time of Ascending/Descending Node (LTAN/LTDN), the local time at the passage of the equator from south to north or vice-versa, respectively, which remains constant to within approximately ±1:00 hours local time over the course of a year.

Dawn-dusk orbits with LTAN or LTDN around 06:00 or 18:00 are constantly exposed to sunlight over most of the year except for two brief seasons of limited orbital eclipses, and thus preferred for power-hungry missions such as radar payloads. Morning pass orbits with LTAN/LTDNs in the range 08:00 to 10:30 or 20:00 to 22:30 are preferred for imaging payloads for a compromise of good illumination and limited convective cloud growth. Atmospheric science missions with an interest in cloud formation or aerosols also use noon-time pass SSOs, e.g. the NASA A-train with a LTDN around 13:15.

Each of these classes of easily rideshare accessible orbits can be addressed by an agile sail which an attitude control test mission can be expected to be. The dawn-dusk SSO requires one rotation per orbit for a tangential thrust to raise or lower the orbit. SSOs closer to the noon-midnight line can achieve this at ½ rotation per orbit, exposing the full sail area mostly on that part of the orbit which is advancing to or receding from the Sun. (Dachwald, 2010; McInnes, 1999) But also quasi-stationary or inertial pointing has predictable effects on most orbit geometries.

Pushing a Tugboat

Since a continued trend towards wider acceptance of small amounts of fuel or pressurized gas on secondary payloads or in shared multiple small spacecraft launches was recently observed, in retrospective beginning perhaps with the PRISMA mission (De Florio et al., 2010; Persson et al., 2010), a simple propulsion system would now be considered for integration on the BSDU side of the spacecraft, beginning at a sail mission level comparable to GOSSAMER-2. The most likely use would be the check-out phase of the various sail control mechanisms while the BSDUs are still attached anyway for continued use of their deployment observation cameras. Thus, a propulsion option would be implemented mainly as a safety or/and secondary (piggy-back) launch flexibility feature. In the latter function it provides the option to perform initial orbit raising e.g. from a low-altitude LEO piggy-back launch to solar radiation pressure (SRP)-dominated LEO altitudes after initial check-out of the still undeployed spacecraft at low altitude, or to lift the very low perigee of a GTO or NavSat transfer orbit (NTO) piggy-back launch quickly by an apogee burn soon after separation, preferably also to get out of the radiation belts quickly. Where this requires a large thruster or/and a larger propellant fraction, the orbit-raising related elements could be discarded with the BSDUs while small attitude control thrusters with a small co-located propellant tank remain attached, either as part of boom-tip mechanisms or the CSCU. Cubesat off-the-shelf micro-thruster systems have become an attractive option for this purpose recently.

The capability to perform limited orbit adjustments could also be useful to widen the spectrum of usable secondary launch opportunities with respect to the regulations regarding space debris avoidance such as applicable Codes of Conduct (CoC, e.g. ISO, 2011). Perigee burns or Hohmann transfer circular orbit raise manoeuvres after

successful commissioning of the stowed and/or deployed configuration could be performed to climb out of relatively low but frequently accessible orbits which are also compatible with the initial undeployed ballistic coefficient of the spacecraft for the 25 years decay lifetime requirement.
After successful deployment and check-out of a sufficient number of control mechanisms to ensure controlled flight of the sailcraft to be separated, residual fuel or unused margins for more demanding launch opportunities can be used to reach still higher orbits that for the deployed sail area and separated BSDUs again comply with the 25 years post-operations orbital lifetime limit. If non-sail propulsion is also available in the CSCU, the orbit raises can be performed in steps:

- after check-out of the undeployed spacecraft in a (very) low orbit to a viable sail deployment altitude
- after deployment of the sail and initial check-out of the membrane control mechanisms to an altitude where the BSDUs can still be safely discarded
- after separation of the BSDUs to an orbit where SRP is dominant

The following table summarizes the configurations, conditions and objectives related to each of these phases and how they relate to orbital altitudes in LEO under the influence of space debris avoidance. In the latter context, as for any spacecraft expecting to use piggy-back launch opportunities, there is a range of altitudes in LEO that complies with nominal mission requirements and proper conduct for safe disposal. This is determined based on space weather predictions when a launch opportunity is selected. The actual altitude then follows from the main passenger's separation orbit, collision avoidance manoeuvres, and possible use of the upper stage's residual capabilities. If propellant is carried aboard the sailcraft – note that the main passenger can require the tanks to remain empty at any time – the best operational orbits for the successive phases of deployment can be achieved based on the latest space weather predictions at the time of the respective flight operations. This enables much better demonstration conditions than even a dedicated launch of the spacecraft to LEO.

Table 2 – Space debris avoidance compliant staged orbit management options for experimental GOSSAMER-style sailcraft in LEO using piggy-back launch opportunities and temporary non-sail propulsion compared to a dedicated launch

mission phase & orbit	spacecraft configuration	minimum altitude condition		maximum altitude condition		conditions drive…	transfer burns done by
		objective	space weather prediction	objective	space weather prediction		
launch to initial checkout	*arrival*: undeployed as launched *departure*: undeployed ready for deployment	orbital decay not before exhaustive but ultimately successful check-out	+3σ high solar activity, as of launch manifesting, possible update if restartable upper stage	orbital decay after exhaustive but ultimately failed check-out +25 years (according to CoC)	best estimate, as of launch manifesting, possible update if restartable upper stage	piggy-back launch opportunity selection & feasibility (also by CoC compliance as a condition of funding)	*arrival*: launch vehicle upper stage *departure*: (main) thruster in BSDU(s)

deployment	*arrival*: undeployed ready for deployment *departure*: deployed ready for BSDU separation	ratio of SRP to atmospheric drag sufficient to climb out by sail propulsion alone in case recovery ACS propellant was exhausted during sail control mechanism check-out and initial performance characterization experiments	+3σ high solar activity, as of completion of initial check-out	orbital decay of the deployed sail with BSDUs attached after exhaustive but ultimately failed check-out +25 years according to CoC	best estimate, as of completion of initial check-out	auxiliary propulsion design, propellant tank capacity	*arrival*: (main) thruster in BSDU(s) *departure*: small ACS or recovery thrusters on CSCU or boom tip mechanism units or BSDUs, sail supported

BSDU separation	*arrival*: deployed ready for BSDU separation *departure*: *sailcraft*: deployed and ready *BSDUs*: separated ready for disposal or extended mission	ratio of SRP to atmospheric drag sufficient to demonstrate sail acceleration unambiguously in case recovery ACS propellant is exhausted during sail control mechanism performance characterization experiments	+3σ high solar activity, as of deployment completion	orbital decay of the separated BSDUs after separation, possible extended mission, and depletion & disposal burn +25 years according to CoC	best estimate, as of the beginning of deployment completion, considering BSDU residual propellant	BSDU attitude control, propellant reserved for BSDU disposal	*arrival*: (main) thruster(s) in BSDU *departure*: *sailcraft*: small ACS or recovery thrusters on CSCU or boom tip mechanism units, sail supported *BSDUs*: (main) thruster(s) in BSDU
initial pure sailing	*arrival*: deployed sailcraft, propellant depleted	ratio of SRP to atmospheric drag sufficient to demonstrate sail acceleration unambiguously and orbital decay not before exhaustive characterization plan completed	+3σ high solar activity, as of BSDU separation	orbital decay of the sailcraft after mission committed to +25 years according to CoC, or perigee >2000 km	best estimate, as of BSDU separation	BDSU orbit raising propellant capacity	*arrival*: small ACS or recovery thrusters on CSCU or boom tip mechanism units *departure*: solar sailing

dedicated launch	*arrival*: undeployed as launched *departure*: deployed sailcraft	ratio of SRP to atmospheric drag sufficient to demonstrate sail acceleration unambiguously and orbital decay not before nominal mission completed	+3σ high solar activity, as of launch manifesting, possible update if restartable upper stage	orbital decay of all sub-spacecraft after nominal mission +25 years according to CoC	best estimate, as of launch manifesting, possible update if restartable upper stage	spacecraft configuration, sailcraft ballasting (cf. IKRAOS), dragsails for BSDUs	*arrival*: launch vehicle upper stage *departure*: solar sailing

It is possible, though maybe not desirable, to leave the applicability region of the decay lifetime limit by raising the orbit to beyond 2000 km perigee altitude. The last of these manoeuvers should also deplete all residual propellant and may place the BSDUs previously carried to a higher altitude back into a compliant disposal orbit, again with less than 25 years expected decay lifetime after their expected end of life, or to perigee altitudes well above 2000 km. At least a one-time spin-up attitude control of the BSDUs affected would then be required for a retro burn.

Tipping a Toe in the Ocean – and then…?

The lower altitude limit for GOSSAMER-2 is based on dominance of solar radiation pressure over air drag, also considering a possible surprise reversal of the fair space weather conditions presently forecast for the next decades. The precise altitude limit depends on these predictions at launch epoch and the actual ballistic coefficient of the deployed sailcraft and the separated BSDUs. The upper altitude limits for GOSSAMER-2 are twofold: In the LEO/SSO range it is the altitude *below* which disposal of the sailcraft and BSDUs within less than 25 years after successful conclusion of the mission by atmospheric drag is still certain, possibly assuming a passively stable shuttlecock orientation mode of the sailcraft once it becomes inactive or inoperable. Significantly higher orbits would also be possible since end-of-life altitudes *above* 2000 km excluding the geostationary belt are presently not subject to spacecraft disposal requirements. Also, the design active mission lifetime of GOSSAMER-2 is fairly limited, enabling the use of LEO/SSO-grade components despite frequent passage of the lower excursions of the radiation belts, possibly with the help of localized stronger than usual shielding. As there is no specific requirement on characteristic acceleration for GOSSAMER-2, a possible mass increase by additional shielding of avionics can be tolerated.

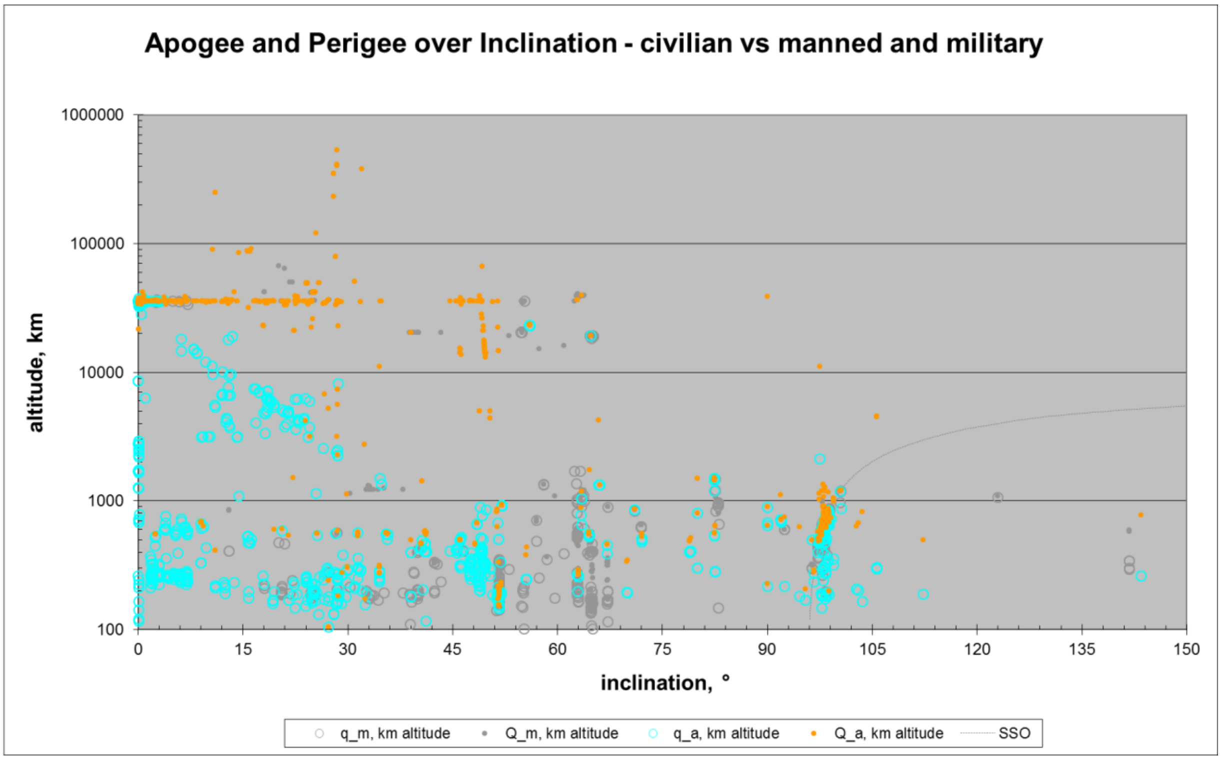

Figure 19 – Civilian payloads launched to Medium to High Earth Orbit, 2004-2011. Blue rings: perigee altitude (q), orange dot: apogee altitude (Q), grey symbols: military & crewed spacecraft considered not accessible for secondary 'piggy-back' payloads; blue-orange or all-grey bulls-eye symbol indicates a circular orbit.

Possible secondary payload launch options beyond LEO/SSO would include build-up or constellation maintenance launches of satellite navigation (NavSat) systems including upper stage disposal orbits in the navigation satellite graveyard or the disposal orbits of upper stages on lifted-perigee geostationary transfer orbits (GTO) or direct geostationary orbit (GEO) insertion. At these altitudes a propellant-less emergency recovery attitude control system using magnetorquers and surplus photovoltaic power would still be feasible.

However, at the time of Roadmap activities, such piggy-back launch opportunities were at best on the horizon or exceedingly rare if they existed at all. In recent years, some have indeed come into existence, culminating in announcements of opportunity for lunar rideshares (e.g. NEA Scout aboard the experimental SLS-1 launch) and even a shared trip to Sun-Earth L_2 (ESA F2018 call with a mainstream science primary payload). These would offer ideal conditions for solar sail development in its proper realm while staying close enough to Earth for easy communication and high enough above its radiation belts for affordable electronic components.

Gossamer-3 – GOING HARDTACK

Gossamer-3 was envisaged as a fully functional solar sail in the $(50\ m)^2$ class, to validate the previously created design approach and prove sufficient guidance, navigation and attitude control capabilities to conduct planetary science and space weather missions. It was to use the sail control methods found most suitable in terms of system-level performance and reliability based on the experience of Gossamer-2.

It was also supposed to incorporate all structural design advances at a sufficient TRL, as expected for a point in time when they might actually be included in the next step towards science missions beyond the GOSSAMER Roadmap. However, a GOSSAMER-3-like project aiming for a final technology demonstration test flight proving the principle would for these key technologies not yet expect the same very high level of TRL as a mainly science driven mission would have to. But it would ensure this TRL range is ultimately reached at the conclusion of its primary mission. The technologies expected to be proof-tested for science missions to come include the thinnest sail foil that can be handled with confidence and reasonable effort for this sail size and the yet again moderately larger next steps as outlined by the GOSSAMER Roadmap Science Working Groups, new light-weight boom designs, and mass-optimized specifically designed mechanisms and bus systems in the CSCU and all other parts of the spacecraft that continue in flight as part of the final deployed sailcraft configuration.

Head Start Welcome

In principle, any launch opportunity viable for GOSSAMER-2 could also have been used for GOSSAMER-3. The lowest theoretical altitude would be defined similarly, by a significant net positive acceleration by solar radiation pressure over residual atmospheric drag. A minimum deployment altitude defined in this manner would likely be somewhat lower for a dawn-dusk SSO, especially outside eclipse seasons, which would allow spiraling out in a quasi-static synchronous rotation mode at constant thrust where one in-sail-plane axis points towards the Earth as if gravity-gradient stabilized and the other at right angle to it is tilted with respect to the velocity vector at an optimal angle for maximum net along-track acceleration, at low altitudes also considering drag and other disturbances. In non-dawn-dusk SSO or other LEO, a likely secondary limiting factor on the minimum launch altitude may be the attitude control agility of the sailcraft if a sail pointing sequence or cycle other than a uniform rotation were required or optimal.
However, such a very low initial orbit would result in a very long initial spiral-up phase through the radiation belts (cf. Di Cara and Estublier, 2005; Koppel, 2005; Milligan et al., 2004, 2005) and – from typical SSO rideshare altitudes of ≈350 to ≈825 km – also through the space debris belt generated by the Chinese ASAT test of 2007 at around 865 km altitude.

As originally envisaged at the inception of the GOSSAMER Roadmap, a high circular orbit insertion would be ideal in terms of sail agility requirements, and also advantageous in terms of eclipse seasons per year and eclipse duration in fractions of the orbital period. There are presently two groups of circular high orbits with regular direct access shortly after the respective launch: the GEO graveyard belt and the NavSat graveyard belts, both accessible soon after delivery of the main payload for their discarded upper stages which mostly have restart capability for civilian launches. The NavSat graveyard belt, though being of lower altitude, has the advantage of rather high inclination which can enable an initial simple sail plane attitude cycle based on constant rotation or a moderate cyclic modification thereof when the β angle is high, and thus fairly constant along-track acceleration for spiral-up in nearly circular orbit. The low-inclination and thus low β angle GEO graveyard belt supports eccentric spiral-up at a ½ rotation per orbit revolution based sail attitude cycle. In such a constant or moderately modulated rotation mode, the sail 'dives through' in sideways minimum cross-section orientation on the sunward velocity vector side of the orbit and exposes the full area for acceleration on the anti-Sun velocity vector side. The GOSSAMER-type structural design concept is flip-over-safe, and thus enables some along-track acceleration at every point of the orbit in this attitude control cycle, except the immediate vicinity of the Sun-pointing velocity vector phase angle where the sail normal to Sun angle (SAN2Sun) is close to 90°.

Although few launches use direct GEO injection or at least a raised perigee disposal orbit for the upper stage, the geostationary market has been the most stable segment of commercial launches, and there still is a trend towards direct GEO injection. However, this market segment of commercial space flight may be challenged in the future by the LEO mega-constellations for digital communication such as SpaceX's Starlink. It can be expected that the NavSat segment will have a steady maintenance cadence even after the initial build-up of constellations, although presently few systems are launched by civilian authorities.

GOSSAMER-3 Mission Design

In the following, a tentative mission concept design for a sailcraft to fly the GOSSAMER-3 mission of the GOSSAMER Roadmap is presented which is based on conceptual work done while the Roadmap was active, in concurrent design spirit ranging from disconnected brainstorming to full-scale numerical-experimental work. Where necessary for the purpose of this paper, this design attempts to fill the many gaps and diffuse uncertainty in the extremely preliminary work on GOSSAMER-3 with reasonable assumptions. No assumptions will be made on the sail attitude control methods to be used on GOSSAMER-3 as these were to be chosen based on experience still to be gained by flying GOSSAMER-2. However, the approximate mass, volume and power demand of the mechanisms and structures required by the sail attitude control methods presently appearing as the most likely can be expected to be within the same order of magnitude, enabling the use of generic parameters. In general, we did *not* follow the common approach to start with a blank sheet to be filled by Phase 0 level precision data. Contemporary MASCOT design work and follow-on studies such as MASCOT2 already applied a mixed maturity approach using Phase D precision level data for units to be re-used 1:1 directly or with minimal or parameter-neutral adaptations in their design, next to pre-Phase 0 best estimates for newly introduced functions. In this context, figures which are presented to appear precise are based on currently available technology which would have been used on the GOSSAMER-1 PFM, or on off-the-shelf units. Moderately precise figures are based on well understood technologies which would likely not have been used on the GOSSAMER-1 PFM but e.g. had been studied already for GOSSAMER-2. One reason not to include them already in the GOSSAMER-1 PFM was that the respective redesign effort would have been in conflict with the last programmatically accepted GOSSAMER Roadmap and GOSSAMER-1 project schedules for a QB50-related launch and/or not feasible with the available personnel.. All other parameters are considered to have only 1…1½ significant digits. In general, best engineering estimates (BEE) will be given here. In project work and related studies, margins were applied following the ESA policy for science assessment studies (ESA, 2012). Additional higher, ±30% and ±50% margin levels were applied on technologies to be selected after or to be sized based on GOSSAMER-2 flight experience, or on highly variable parameters such as CPU load dependent consumption, following local DLR Bremen Concurrent Engineering Facility (CEF) best practice originally introduced for AsteroidFinder/SSB. (Findlay et al., 2011)

Launch

For micro-spacecraft like GOSSAMER-3, a dedicated launch into high orbit is extremely unlikely mainly for cost reasons. Thus, a secondary payload launch opportunity – also frequently called “piggy-back” or previously “hitch-hiker” launch – is the only realistic route available for a project on the likely scales of GOSSAMER-3. Secondary launches to SSO have become commonplace enough so that full commercial rates apply to all but the smallest CubeSat classes in missions of educational use. In any case, the target orbit and available launch vehicle spare capacity in terms of volume and mass usually depend on a prime payload which may be a pair or cluster of separate spacecraft but is generally released into its initial orbit as if it were one.
Next to the choice of orbits constrained by the primary passengers it is also difficult to find a suitable opportunity due to a significant capability gap in the global launch vehicle market in particular when a high altitude (initial) orbit is desired. At the time of the GOSSAMER Roadmap, the only serious attempt at creating a technically and commercially viable small spacecraft launcher was the SpaceX Falcon-1/-1e (SpaceX, 2008a) mainly aiming at the LEO/SSO market. It was terminated after only 4 launches with the backlog moving to piggy-back slots on the larger Falcon-9 (SpaceX, 2009) of the same provider (Delovski et al., 2019; Findlay et al., 2013). Unique in its class until very recently (RocketLab, 2019), the Falcon-1e used environmentally friendly propellants, had very gentle launch loads, and had an announced though modest lunar launch capability with a small added boost stage (SpaceX, 2008b). It would have been ideal for a mission like GOSSAMER-3. More importantly, it would most likely have been affordable at a small bonus for having the best initial orbit possible when compared to the total investment in and total cost of ownership of other then-realistic rideshare or secondary payload launch options which include launcher change tolerant spacecraft design, lifetime margin and operations until the same point of departure orbit is achieved.
For a technology demonstration mission an initial spiral-out phase would likely be acceptable (cf. Di Cara and Estublier, 2005; Koppel, 2005; Milligan et al., 2004, 2005) although it would more likely be avoided for later science missions to minimize degradation of the instruments during passages of the radiation belts. Also, sail deployment in a perigee altitude range between the ISS (some 300 to 400 km) and 2000 km would likely be avoided with some effort due to out of proportion design driving end-of-life disposal requirements (ISO, 2011) on any of the spacecraft parts, in particular the BSDUs which would have to be separated for GOSSAMER-3 to achieve its minimum mission goals.
There are several presently active characteristic orbit families with a perigee significantly above 2000 km and/or a suitably high apogee of an elliptic initial orbit which are compatible with the GOSSAMER-3 mission
Secondary payload launch, “piggyback” options that at the time appeared as in principle accessible for GOSSAMER-3 although rare opportunities included:

Table 3 – Some classes of potential launch opportunities earmarked in the studies and launch market surveys related to GOSSAMER-3

launch vehicle	orbit type	typical altitude or 'perigee x apogee', km	main payload	remarks
Ariane 5	low perigee GTO	200 x 36000	GEO ComSats	space debris avoidance applies near perigee and apogee
Soyuz CSG	Galileo transfer orbit	to 23000	Galileo NavSats	space debris avoidance applies near perigee
Soyuz CSG	Galileo orbit graveyard	>23000	Galileo NavSats	upper stage disposal
Vega	high elliptic	600 x 60530	PROBA-3	space debris avoidance near perigee, PROBA-3 launch unlikely to have sufficient unused capacity, larger similar science missions may follow
Vega	medium to high circular	>2000, <36000	high LEO, MEO	also up-high disposal of AVUM from LEO
Proton-M (mainly)	raised-perigee GTO	~5000 x 36000	GEO ComSats	perigee depends highly on main payload mass

The Δv from a typical GTO to Earth escape c_3 >0 is 749.5 m/s and could be provided by a small kick-stage accommodated with the BSDUs. If a lunar fly-by can be used, less Δv is required.

Initial Trajectory Considerations

In the absence of a single defined reference trajectory for GOSSAMER-3 (cf. next section), in order to get a feeling for durations and possible targets, results from other internal studies on low-thrust lunar, Sun-Earth Lagrange points and interplanetary missions were considered:

1. A solar-electric propulsion (SEP) based heavy spacecraft study assuming a mean acceleration of 0.13 mm/s² was found which is comparable to the expected performance range of GOSSAMER-3. In this study, spiraling out takes:
   - about 13 months to get from a 380 km LEO (65° inclined) to GEO (similar inclination)
   - up to 6 months to get from that GEO into a highly elliptical lunar orbit (using lunar gravity assists)
   - about 5 months to then reach a 100 km LLO (e.g. polar)

2. Having much lower acceleration levels, spiraling out from an equatorial LEO towards GEO can require up to 10 years, as can be seen in Table 4 below summarizing study results for a medium-size GEO spacecraft.

Table 4 – Low-Thrust trajectory calculations from an equatorial Launch Orbit to GEO for a mission study using orbit raising by SEP in two different spacecraft propulsion configurations of lower continuous and higher duty-cycled thrust.

Launch Orbit Altitude [km]	Delta-V [m/s]	lower thrust option near-continuous thrust		higher thrust option duty cycle ≈30% ON-time	
		Acceleration [mm/s²]	Trip Duration [yr]	Acceleration [mm/s²]	Trip Duration [yr]
300	4354	0.0125	11.19	0.0129	3.12
500	4486	0.0125	10.76	0.0129	3.01
1000	4278	0.0126	10.22	0.0130	2.86
5000	2846	0.0131	6.63	0.0134	1.87
10000	1860	0.0135	4.26	0.0137	1.27
20000	815	0.0139	1.83	0.0140	0.52
30000	238	0.0142	0.53	0.0142	0.15
35000	31	0.0143	0.07	0.0143	0.02

It is noteworthy that the acceleration performance level shown in Table 4 is well below the low-end of expectations for GOSSAMER-2 (which had no characteristic acceleration requirement) and at the high-end of early and somewhat optimistic expectations for GOSSAMER-1 (which had no attitude or trajectory control foreseen). An example at the far end of the performance spectrum of SEP technologies at least mature enough to be considered on a space agency mission call is the study of a probe which reaches Jupiter Trojan asteroid (659) Nestor within 4.63 years on a continuous thrust trajectory with a maximum acceleration of 0.24 mm/s² by Maiwald, 2010. Via the 20th CE study at DLR Bremen it became the Trojan Investigation Probe (TRIP) designed for a M-class mission call of ESA. (Maiwald et al., 2012)

Breaking down 'credible demonstration'

The envisaged mission of GOSSAMER-3 was operational conversion, i.e., becoming truly mission capable, in essence learning and experiencing to fly for real, beyond the exercising of the controls that was to be done on GOSSAMER-2. It was to take place relatively close to Earth, essentially within its sphere of influence in the solar system, between the Sun-Earth Lagrange points L_1 and L_2 and around the Moon. This way, the significant effort of implementing a truly interplanetary communication system using a large high-gain antenna could be bypassed although a downscaled implementation using a small steerable antenna was considered. Within Earth's sphere of influence, the capabilities of a fully developed sailcraft were successively and iteratively to be tried, characterized, analyzed, operationally improved and applied 'live' to the next steps in the ongoing mission as it proceeded. A cautious, stepwise approach seemed obvious, leading to a wish-list of 101 small steps for the solar sailor rather than a nominal trajectory.

Minimum Mission

The minimum mission begins with a spiral-up from the launch orbit. Some of the likely launch orbits already provide an opportunity to demonstrate the advantage of the

GOSSAMER design goal of care-free flip-over capability, here enabling a constant-rotation mode that puts the sail edge-on at low perigee for minimum drag (perigee is also usually in Earth shadow in GTO) and generates thrust near apogee to lift the perigee soon. Within this phase, multiple pointing and navigation tests are already possible. Navigation is already required to command sail attitude sequence timelines correctly. Here, mainly the sail's response and performance are analyzed and the prediction updated with this data. For pointing accuracy tests, the cameras used to monitor the sail or to fulfil star tracker functions can be used to image the Earth's limb, the Moon, specific locations on Earth, Venus, Jupiter, or bright stars. A combination of both can be demonstrated by targeted flyby proof tests in preparation of asteroid-related missions such as MNR (Multiple NEA Rendezvous, NEA: Near-Earth Asteroid). The task would be to image from a typical asteroid fly-by distance, i.e. 100's to 1000's km, a Chelyabinsk- to Tunguska-sized object on a well-known trajectory. Most of the target asteroids coming up in MNR trajectory optimization are of this very small size unless special selective criteria are applied, e.g. to only accept trajectories also including PHAs (Potentially Hazardous Asteroids) which are by definition at least 140 m in size based on standard albedo. (Peloni et al., 2016, 2018) The simulated asteroid target can be selected by brightness, i.e., absolute magnitude H, from the population of artificial Earth satellites to ensure that it is visible also to a non-optimized camera. Promising targets would be large spacecraft, e.g. Thuraya-2 (Figures 20 and 21) or any other comsat with a large deployed antenna in GEO which are passed by during the relatively slow initial spiral-up out of Earth's gravity well. Imaging opportunities showing the target with >>1 pixel would be highly welcome if safely possible to unambiguously identify the target and thereby confirm timing and pointing accuracy, in addition to position fixes against the stellar background (Figure 20). The idea of using large spacecraft as target asteroid surrogates was generated serendipitously by the success of near-Earth asteroid (NEA) surveys: the WIND spacecraft was at a point in time registered under the preliminary asteroid designation 2001 DO_{47} (Spacewatch, 2001), ROSETTA briefly misidentified as a tiny H = 26.3 asteroid, 2007 VN_{84}, during its second Earth flyby on November 13th, 2007 (MPEC, 2007), and even JAXA's SEP-propelled NEA explorer HAYABUSA2 was briefly on the NEO Confirmation Page (NEOCP), southbound with an impact probability (Chesley and Farnocchia, 2015), which by the way shows that the NEO warning system works with obviously unidentified targets.

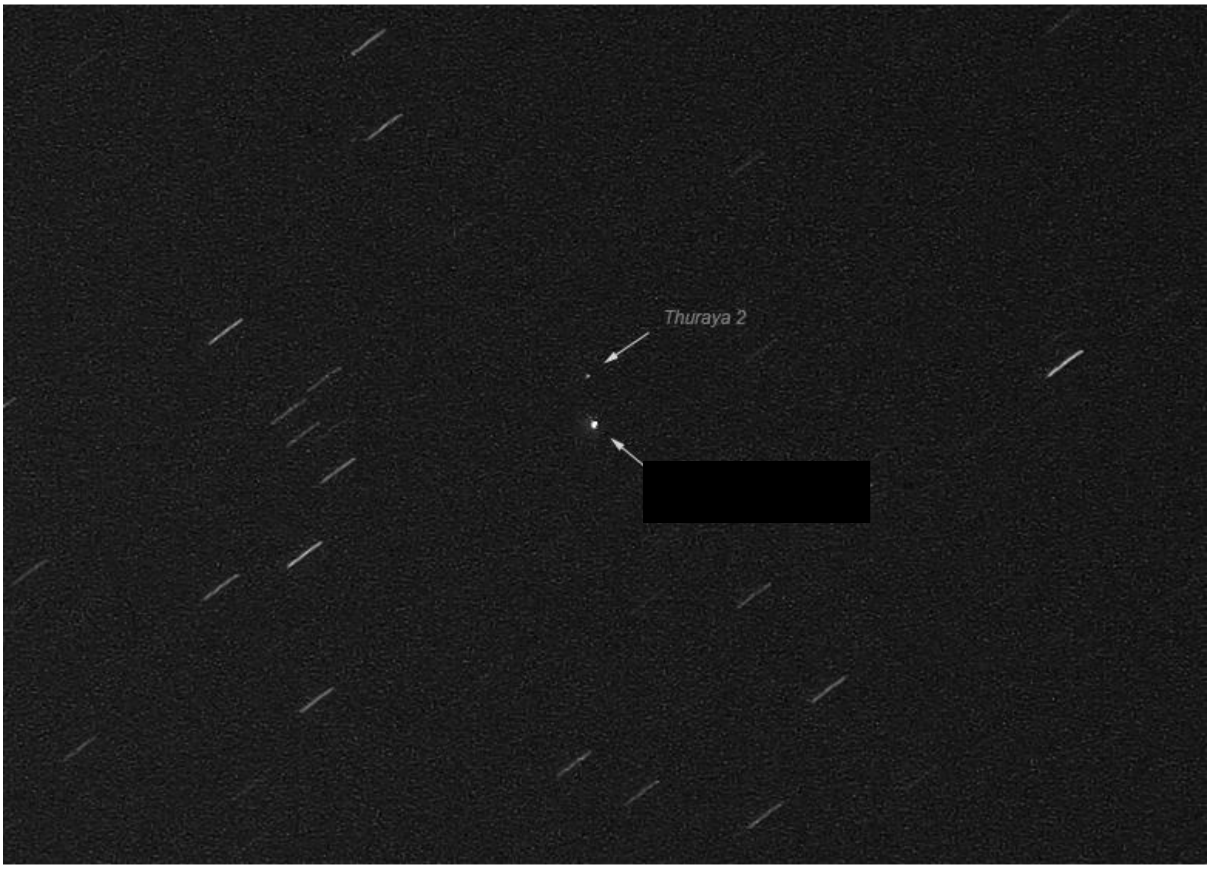

Figure 20 – Thuraya 2 (labeled) and another nearby geostationary satellite, photographed on 8 December 2010 from the Netherlands by Marco Langbroek, adapted from (Langbroek, 2010)

Figure 21 – A fellow large deployable structure in Earth orbit to meet & greet at a safe distance on a GOSSAMER-3 fly-by: Thuraya 2 and 3 design, artists concept, image: Boeing BSS via Gunter's Space Page

The passage through the GEO region most likely would have to be performed on a strongly inclined orbit to minimize risk of collision, i.e., the sailcraft either can keep the inclination of an initial Galileo NavSat related orbit or high LEO SSO, or it has to build up inclination from e.g. an Ariane-5 GTO. For space debris avoidance, the spiral up needs to continue further, at least into GEO upper graveyard for a circular orbit or the perigee of more elliptical orbits.

Expected Mission

The expected or nominal mission would continue the spiral up to nearly lunar distance and then proceed with navigation exercises in the Earth-Moon system, taking advantage of the presence of another large planetary body, for future missions in this environment or among the larger moons and planets of the solar system. The presence of the Moon by gravity, thermo-optical albedo, and as an object of interest would be used for extensive system tests to further expand the envelope of sail characterization and navigation. On the way to the Moon or venturing out again from its vicinity, it was envisaged to explore the Earth-Moon Lagrange points, e.g. look for Kordylewski clouds near the L_4 and L_5 points which appear to consist of interplanetary dust particles and dust ejected by meteoroid and asteroid impacts on the Moon which is captured, temporarily or in a flow equilibrium, in dynamic traps generated by the gravity of the Earth, the Moon and the Sun as well as (mainly solar) radiation pressure. (Slíz-Balogh et al., 2018, 2019) During this phase, lunar flybys are likely to be used to accelerate transfers between the various dynamic realms of the Earth-Moon system, possibly already including pole-sitter trials for Earth observation.

Extended Mission

Also, in the following extended mission, these ‘slingshot intermissions’ at the Moon and possibly Earth (somewhat depending on radiation tolerance) play an important role. The many typical sail operation and mission locations or modes which are desired to be demonstrated require quick transitions of the ‘scoot and shoot’ kind between. Combining lunar and Earth fly-bys for changing missions of the same spacecraft is not without precedence remembering ISEE-3/ICE/ISEE-reboot (Cowing et al., 2014) or the small spacecraft Mars mission, Nozomi. (JAXA, 2004) It was considered to repeat the timely target imaging demonstration again and with much closer approaches to objects less sensitive than geostationary communication satellites. At the Earth-Sun L_2, a targeting flyby test could involve finding, tracking and approaching a recently decommissioned L_2 spacecraft, e.g. Planck, or an upper stage of such a mission if one serendipitously happens at that time as it coasts out, and more detailed imaging at >>1 pixel upon closest approach. This anticipates rendezvous with very small asteroids in the MNR mission context or safe auto-navigation in a more crowded region e.g. passing through the NavSat, GEO and Earth-Sun L_1 regions on the way towards a DL1 mission’s operational location. On the return towards other tasks, passing near Earth again, it may be possible to investigate the geotail or ‘stop over’ for another period of operations in pole-sitter mode perhaps in a different season. (Ceriotti et al., 2012; Heiligers, 2012) Another ‘slingshot intermission’ or entry into a DL1-like transfer trajectory to the Earth-Sun L_1 region for a try-out of Displaced L1 mode operations would likely conclude this phase. This choice was made to enable indefinite ad-hoc operation as an interim DL1 mission there because the required science instruments are very light-weight and prototype or demonstrator versions, e.g. used qualification models (QM) might be relatively easy to obtain from the relevant science communities in exchange for a flight opportunity, and their data would be of the utmost interest to the solar sail hardware community in relation to membrane ageing studies. (Dachwald et al., 2007; Renger et al., 2014; Seefeldt et al. 2018; Seefeldt & Dachwald 2019; Sznajder et al., 2013, 2015, 2017, 2018, 2019a,b)
But like a sail’s thrust which never really stops, mission concept development still found more objectives for an extended-extended mission. (cf. Mori et al., 2012)

…and further on

The division between this and the preceding extended mission phase is that the communication system of GOSSAMER-3 was to be sized only for operations at very moderate data rate *and* within the Earth’s sphere of influence. For the purpose of link budget design, this was defined as being not further out from Earth than the maximum DL1 mode distance from Earth that could be achieved by the actual GOSSAMER-3 design with a factory-fresh sail, based on the best design and characterization data available at the relevant decision point. Also, support by large ground stations could not be expected, neither for an unplanned non-science mission nor from a cost point of view. However, considering the operations of NASA’s Galileo mission on the way to and at Jupiter which had to cope with the deployment failure of its main antenna, it was thought possible that more distant operations could be attempted if they did not drive the spacecraft design (already having a design margin for the extended mission beyond lunar distance!) and there was a worthy objective. Also, there was already encouraging experience of low data rate operation over interplanetary distances from the constant monitoring of the space weather beacons aboard the two STEREO spacecraft which is performed with support of the amateur radio community and dish antennae that fit in a suburban home's backyard (tightly). (AMSAT, 2013; Kucera, 2016; Thompson, 2009) So, should a feasible opportunity present itself and could the sailcraft be released from its DL1 demonstration operations, the final objective would be to fly out to a co-orbital asteroid of Earth's. Prime candidates would be the larger, low inclination, non-fast rotators close to Earth in this very diverse population. The two primary target asteroids considered were 2010 TK_7 and (419624) 2010 SO_{16}, the only two presently known large co-orbital NEAs in Lagrange points librating modes (Lagrange, 1772; Murray & Dermott, 1999).

Table 5 – Presently known large Co-Orbital NEAs (top), considered small co-orbital NEAs (middle) and higher Δv co-orbital NEAs (bottom). ([dataset] Benner, 2018; [dataset] Hahn and Mottola; [dataset] JPL; Christou and Asher, 2011; Connors et al., 2011)

target object	**(419624) 2010 SO_{16}**	**2010 TK_7**
Semi-major axis (a)	1.00336 AU	0.999273 AU
Aphelion (Q)	1.07904 AU	1.189615 AU
Perihelion (q)	0.92656 AU	0.8089311 AU
Inclination (i)	14°.517	20°.87993
Eccentricity (e)	0.075429	0.19048033
condition code	0	0
absolute mag (H)	20.5 mag	20.8 mag
Dimensions	357 m (1σ: 126 m)	379 m (1σ: 123 m)
Albedo	0.084 (1σ: 0.057)	0.059 (1σ: 0.049)
Libration	L_4⬄L_3⬄L_5 horseshoe	L_4 Trojan (tadpole)
near-Earth turn distance	0.15 AU	30…40 · 10^6 km
Δv from Earth	7584 m/s	9392 m/s

target object	**2003 YN_{107}**	**2006 JY_{26}**	**2012 FC_{71}**	**2013 BS_{45}**
Semi-major axis (a)	0.989 AU	1.010 AU	0.988 AU	0.992 AU
Aphelion (Q)	1.002 AU	1.094 AU	1.075 AU	1.075 AU
Perihelion (q)	0.975 AU	0.926 AU	0.901AU	0.909 AU
Inclination (i)	4°.321	1°.439	4°.941	0°.773
Eccentricity (e)	0.0139	0.0831	0.0881	0.0837
condition code	1	3	5	0
absolute mag (H)	26.5 mag	28.4 mag	25.2 mag	25.9 mag
Dimensions	10…30 m	6…13 m	20…40 m	20…40 m
rotation period				0.03 h
observed by radar				☑
Libration	compound horseshoe & quasisatellite	horseshoe	horseshoe (Kozai)	horseshoe
near-Earth turn distance		0.0029 AU	0.0058 AU	0.013 AU
Δv from Earth	4879 m/s	4364 m/s	4696 m/s	4102 m/s

target object	**(3753) Cruithne** (ex 1986 TO)	**(54509) YORP** (ex 2000 PH_5)	**2001 GO_2**	**2002 AA_{29}**

Semi-major axis (a)	0.9977 AU	1.00004 AU	1.0067 AU	0.9925 AU
Aphelion (Q)	1.5113 AU	1.22997 AU	1.1761 AU	1.0055 AU
Perihelion (q)	0.5148 AU	0.77012 AU	0.8373 AU	0.9796 AU
Inclination (i)	19°.8063	1°.8331	4°.6251	10°.7482
Eccentricity (e)	0.5148	0.2299	0.1683	0.0130
condition code	0	0	7	0
absolute mag (H)	15.6 mag	22.7 mag	24.3 mag	24.1 mag
Dimensions	2071 m (1σ: 106 m)	150 · 128 · 93 m	35…85 m	20…100 m
Albedo	0.365 (1σ: 0.082)	0.1		0.05…0.25
rotation period	27.30990 h	0.2029 h		0.55 h
observed by radar		☑		☑
Libration	horseshoe (high-e)	horseshoe	horseshoe (likely) & quasi-satellite	horseshoe & quasi-satellite
near-Earth turn distance	0.08…0.09 AU	0.00526 AU		0.0391 AU
Δv from Earth	13510 m/s	5693 m/s	5364 m/s	6777 m/s

Because low solar elongation asteroid surveys are at best difficult from the ground (cf. Findlay et al., 2013), most co-orbital NEAs are found near their libration turn when lingering closest to Earth, when they briefly enter the parts of the sky accessible to the most prolific asteroid surveys, becoming visible like the tip of an iceberg. This steady trickle of discoveries provides additional targets which are mostly very small fast rotators but also often closer to Earth or astrodynamically much easier to reach than the very few larger ones. Among the objects at the time noted for further studies were the small low-inclination, low-eccentricity horseshoe librators 2013 BS_{45}, 2006 JY_{26}, and 2012 FC_{71}; another small NEA in a low-inclination, low-eccentricity compound orbit alternating between horseshoe libration and quasisatellite states, 2003 YN_{107}; and larger but higher eccentricity or/and inclination co-orbitals requiring higher Δv transfers like (54509) YORP (formerly 2000 PH_5), 2001 GO_2, 2002 AA_{29}, and the largest known, (3753) Cruithne.

The details of the chase and approach phase depend on the target's orbit and when the sail departs from the vicinity of Earth. (see Ceriotti et al. 2021; Pezent et al., 2021) So far, NEA rendezvous orbit optimization has focused on the most efficient ways to rendezvous directly with a NEA. Co-orbital NEAs show a peculiar motion in the heliocentric Earth co-rotating frame: relative to Earth, they loop in the sky around Earth's orbit, always either on the morning side or the evening side. This begs the question whether it is possible to achieve a fast flyby much earlier than an optimal direct rendezvous to catch a first glimpse of the target as soon as possible and then catch up with it and return for rendezvous a little later than in the optimal direct case. Bearing in mind that GOSSAMER-3 would have been a first experimental, fully light-weight and mission-capable sail, as well as a low-cost technology demonstration spacecraft formally designed for a few loops in the Earth-Moon system, made the

option of an earliest possible asteroid encounter particularly attractive. Alternatively, one of the many small objects could have been targeted for a (slow) fly-by before heading out to one of the more attractive options.
Possible solar sailing demonstration operations at the target asteroid very much depend on its mass. With the parallel in-house development of MASCOT, it is obvious that an attempt to drop a small ballistic object, beacon or instrument package on one of the larger possible targets came to mind. It would then be attempted to stay close enough to relay data until the end of life (EOL) of that device. At the time, it was envisaged to carry a few ejectable cameras similar to the DCAMs carried by IKAROS to view the sail from time to time from outside. (Matunaga et al., 2011) Earlier, carrying an inspector cubesat had also been considered. Unused spares of either of these could have been used in this case.

Rest in Regolith

Apart from continuing observations at that NEA indefinitely until a spacecraft failure ends the mission, there are also other options. These include trying to fly to another 'nearby' NEA, i.e., within reach, likely also co-orbital, and on the same, L_4 or L_5, side of Earth, in particular in case the first target was not suitable for a dropped ballistic object landing, e.g. because it is too small and/or a fast rotator. (de la Fuente Marcos and de la Fuente Marcos, 2015) The final choice may be to go out with a bang, at first trying very close passes at the target NEA at successively increasing risk levels. Then it again depends on that asteroid's properties. On a large and slowly rotating target like 2010 TK_7 or (419624) 2010 SO_{16}, it may be possible to beach the sailcraft on that asteroid, if possible so that Sun glints are visible from Earth periodically indicating its continued presence. (cf. Dunham et al., 2002) On small or/and fast-rotating NEAs which are mostly of a similar size or smaller than GOSSAMER-3 would have been, a temporary 'docking' to the summer rotation pole may be possible. As soon as significant illumination from the 'bottom' side begins when the orbit has taken this spin-stabilized assembly to the other side of the Sun, the sail will lift off again. If the sailcraft continues to operate for some time, the radiation pressure change acting on the asteroid's orbit (Micheli et al., 2011, 2013) may be detectable for very small boulders in space.
Many of these ideas were not analyzed in depth beyond a thorough literature research and back-of-the envelope sanity checks. Mostly, they represent the many points of departure for future work.

Into the Distance

Scouting Ahead

Relatively early in the development towards a more specific definition of GOSSAMER-3, and in parallel to the work of the GOSSAMER Roadmap Science Working Groups, we explored the addition of the MASCOT Flight Spare (FS), still being integrated at the time in its previous incarnation as a Qualification Model (QM), in as much 'as-is' as possible (or sensible) form, to the proving-the-principle solar sail mission of GOSSAMER-3. (Grundmann et al., 2015) Although a final trip to an asteroid was never excluded for GOSSAMER-3, by virtue of the choice of its science instruments for the mission on Ryugu, the MASCOT FS was well suited to be primarily considered as a membrane monitoring instrument package[5] to help study ageing processes and the interaction of the sailcraft as a whole with the interplanetary medium. Carried in the shadow of the sail (while it is in its nominal orientation; GOSSAMER-style sails can flip over), every instrument of MASCOT has the capability to support an important aspect of the space weathering study objective of the mission:

MasCAM, the camera of MASCOT, uses a special optical design (Scheimpflug angle) to create images of the asteroid surface in focus from close to the lander at ≈10 cm distance in the lower part of the field of view (FoV) to infinity in the upper part of the FoV. Instead of the all-FoV covering black calibration target carried aboard HAYABUSA2, material samples would be placed close to MasCAM on the carrying structure connecting the nanolander to the CSCU. The LED 4-colour illumination unit of MasCAM can help to track spectral changes caused by ageing of these samples at wavelengths of 465, 523, 633 and 812 nm. Parts of the CSCU would appear in the mid FoV of MasCAM, also partly in the area of illumination. Beyond, the upper part of the FoV would show a boom and parts of the sail membrane to its outer edge. Perhaps the upper quarter to third of the FoV would show empty space, primarily as a margin for extreme distortions of the sail to be analyzed in case of a deployment or boom failure. After a nominal and successful deployment of the membrane in view, this also enables views of the sail, a boom and possible target objects throughout the entire mission. In general, imaging was expected to be performed with emphasis on a particular feature of MasCAM, black-and-white high dynamic range (HDR) photography, because of foreground glints and shadows. (Jaumann et al., 2017)

[5] later lightheartedly backronymed "Membrane Ageing Studies Concurrent Observations Technology package" for Grundmann et al., 2019a and our up-the-sleeves contributions to the PDC 2019 exercise

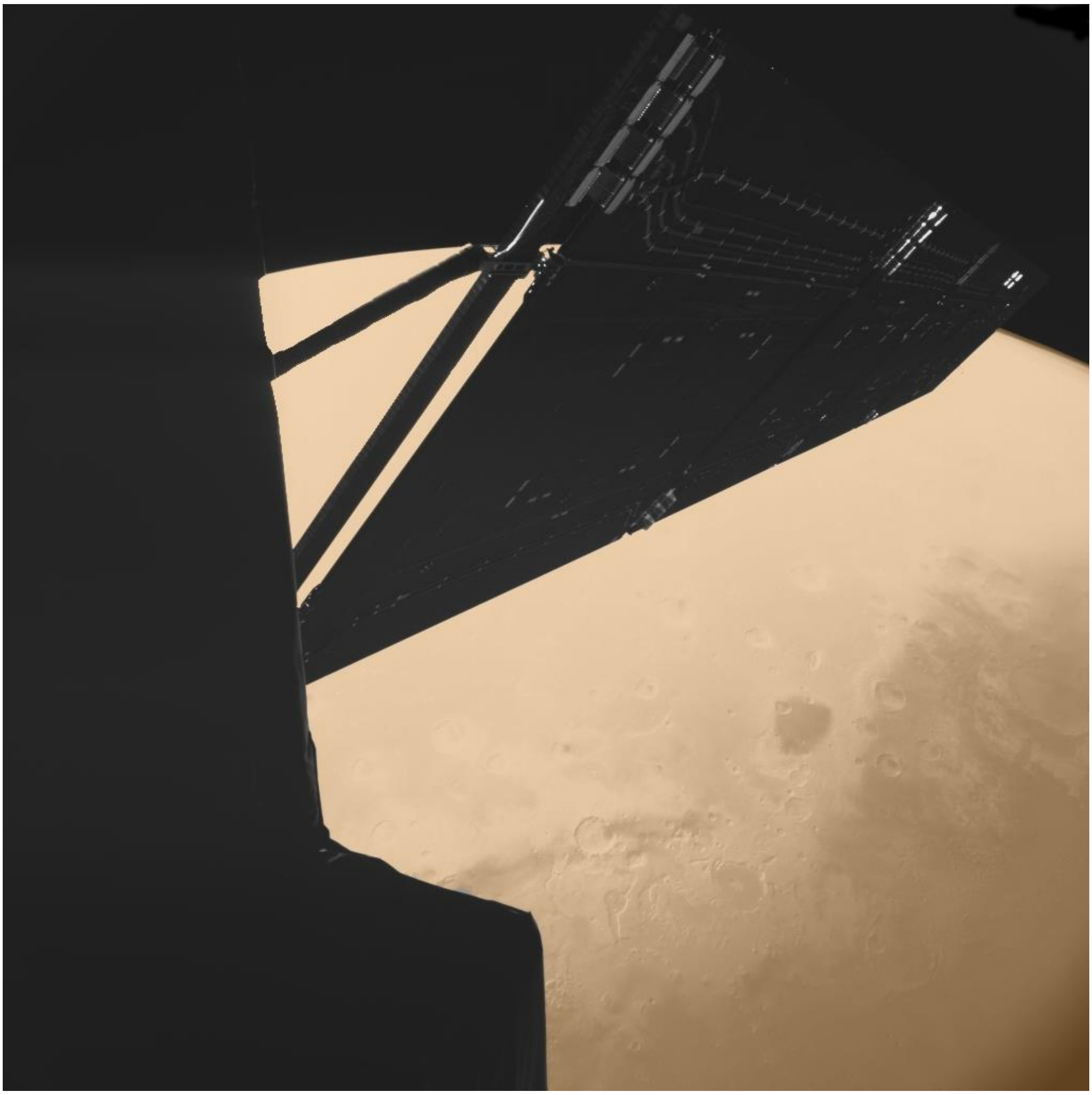

Figure 22 – An example of the use of planetshine for mothersip diagnostic imaging purposes: Mars as seen at ≈1000 km distance from PHILAE along the rear side of the photovoltaic panels of ROSETTA – © ÇIVA/PHILAE/ROSETTA [6]

[6] ESA, Stunning image of Rosetta above Mars taken by the Philae lander camera, http://www.esa.int/ESA_Multimedia/Images/2007/02/Stunning_image_of_Rosetta_above_Mars_taken_by_the_Philae_lander_camera

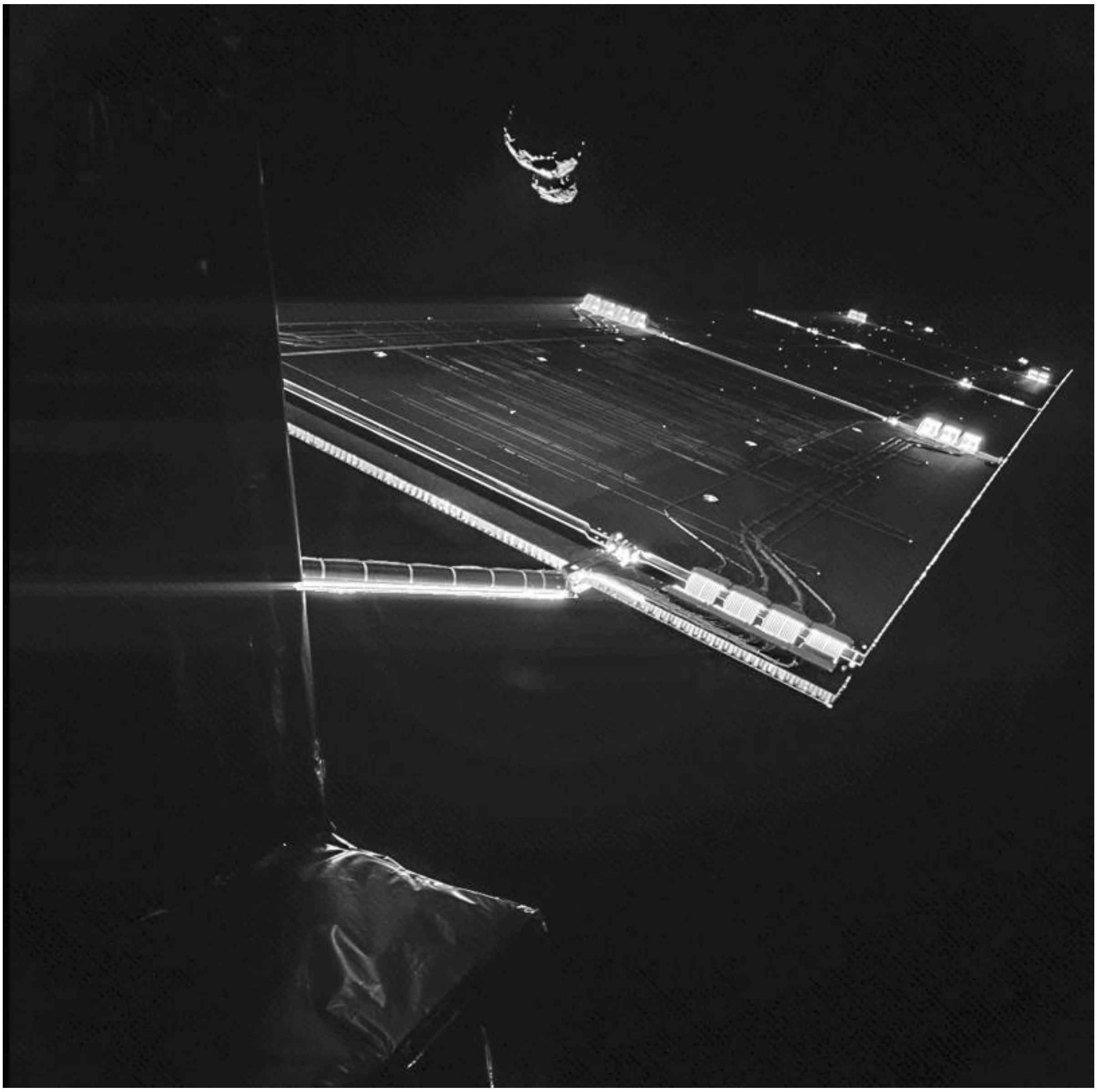

Figure 23 – An example of flyby image targeting practice: 67P seen from PHILAE along the photovoltaic panels of ROSETTA – © ÇIVA/PHILAE/ROSETTA[7]

MARA, the thermal IR radiometer would view a smaller section of the sail foil than MasCAM but within the latter's FoV in the inner region of one sail quadrant. The accommodation of the nanolander in cruise aboard GOSSAMER-3 would be driven by the MARA FoV for this purpose, to record the thermal emission of the membrane as an indicator of ageing. Changes in the spectral response can also be monitored in MARA's thermal IR wavelength bands, 5.5–7, 8–9.5, 9.5–11.5, 13.5–15.5, 8–14, and 5–100 µm. The high sensitivity and the spectral channels are driven by the scientific goals of the MARA instrument, determination of the asteroid's thermal inertia and

[7] ESA, Rosetta mission selfie at comet, http://www.esa.int/ESA_Multimedia/Images/2014/09/Rosetta_mission_selfie_at_comet, also see http://www.esa.int/ESA_Multimedia/Images/2014/10/Rosetta_mission_selfie_at_16_km

characterization of the surface mineralogy. (Grott et al., 2016, 2019; Hamm et al., 2017, 2018) The options for a calibration target (CalTarget) as carried aboard HAYABUSA2 for MASCOT but equipped with a mechanism to move it out of the FoV were under consideration.

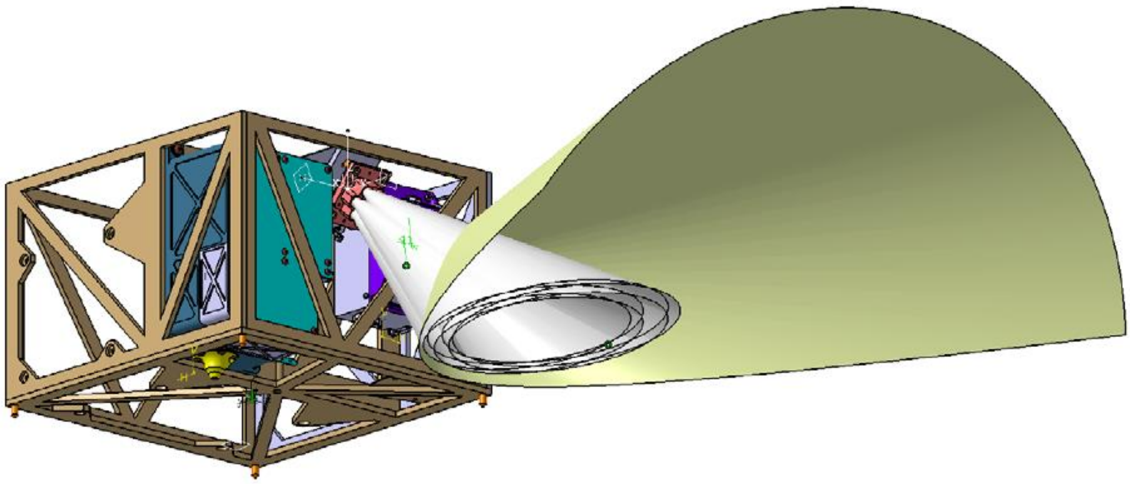

Figure 24 – Overlapping fields of view (FoV) of MARA (light grey cones) and MasCAM (pastel green half-cone). MicrOmega is the large box to the left (turquoise), with the attached optical head (yellow). Its FoV extends out from the sapphire optical window at the tip of the optical head by the depth of focus and is too small to be visible at this scale. MasCAM is the box (two tones of violet) partly hidden by FoV cones. Mounted between them is the MARA sensor head (orange). The MAG Sensor Head is accommodated in the space beneath MasCAM (not shown for clarity).

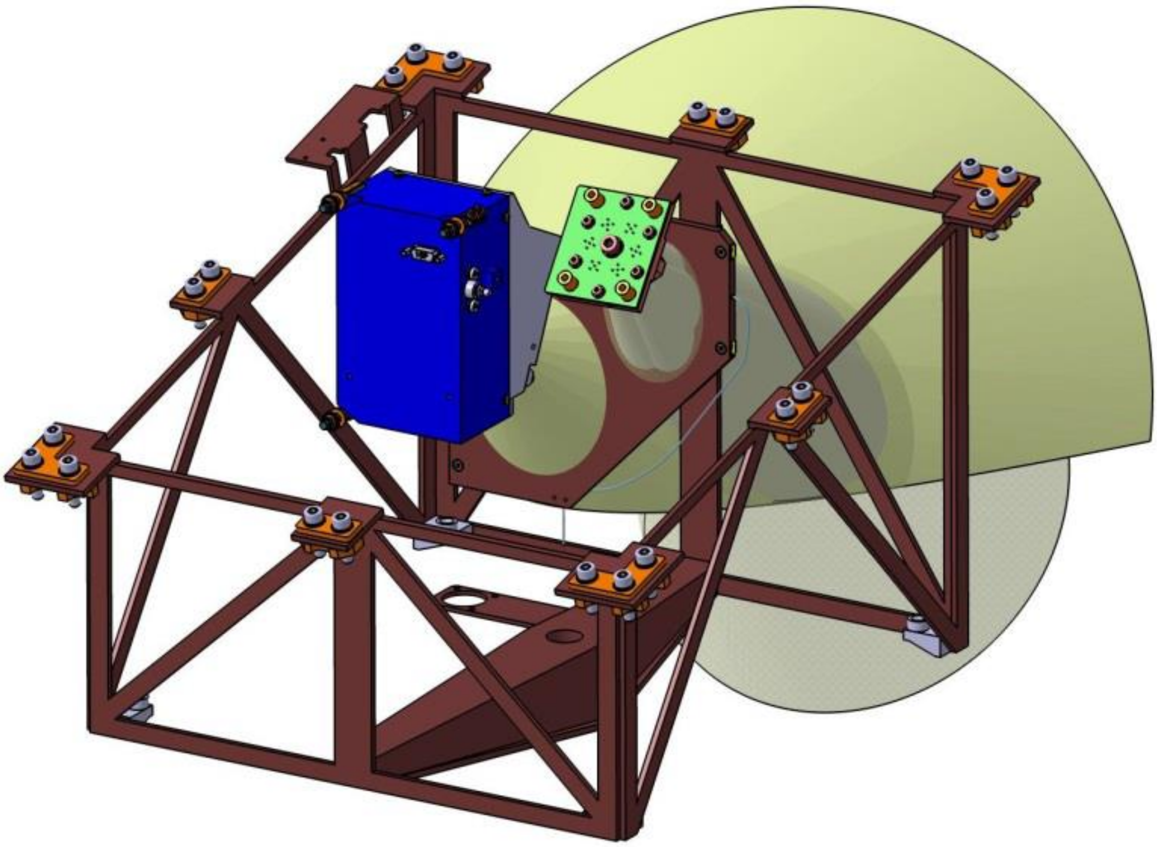

Figure 25 – MARA (light grey) and MasCAM (pastel green) views on the Calibration Target (brown plate) behind MasCAM (blue) and MARA Sensor Head (light green, with screw heads) in MASCOT's cradle, MESS (brown framework), semi-recessed within HAYABUSA2. At close range, MARA and MasCAM FoVs do not overlap.

MasMAG, the magnetometer (Herčík et al.,2017), observes the external and internal magnetic fields on the lander as an instrument providing valuable magnetic field/magnetization measurements, but also as a very good housekeeping monitor (in case of highly integrated MASCOT and the nearby bus of the GOSSAMER-3 CSCU). (Herčík et al.,2020) However, as has been shown by recent observations of ROSETTA and PHILAE, having two magnetometers, one onboard the orbiter and second one on the lander, is extremely useful. The combined measurements help to separate solar wind/plasma external effects from the asteroid's own intrinsic magnetic fields. Therefore, MasMAG could be augmented by two additional identical or a least similar magnetometer sensors near a boom tip.

MicrOmega is the hyperspectral near-IR soil microscope of MASCOT. It acquires 3D (x,y,λ) microscopic image-cubes of samples approximately (3 mm)² in area, with a spatial sampling of 25 µm² in 128² pixel images. For each pixel, a spectrum is acquired in the range 0.99 to 3.55 µm. The spectral sampling is better than 40 cm^{-1} typically with a signal-to-noise ratio of 100, over the entire spectral range. (Bibring et al., 2017) Here it would look at samples of sail foil and other materials exposed to the space environment and direct sunlight. The sail foil sample at the instrument also covers the instrument aperture from incident direct sunlight as part of an instrument protective

cover assembly which also contains a mechanism by which samples can be moved past the optical window from another section of the spacecraft where it is more exposed. The sample strip or tape is of order 1 cm wide and 1 metre long or longer, to reach around the CSCU to be exposed on the sunny side. It can contain foil metallized on both sides, on one side towards the optics or away from it, or not at all. Various coatings can also be applied in sections. A cassette tape drive like mechanism or a simpler electromagnet ratchet drive stepper similar to those used on the Learjet 8-track tape head shift motor were considered, acting on an endless tape circulating in one or both directions or a spring-loaded spool stowed tape being pulled out forward and then rewinding again into a protected container.
Although the MASCOT FS would have been carried on the normally shadowed side of the CSCU, the flip-over capability of GOSSAMER solar sails would have allowed exposing it to the Sun, and the side of the membrane being monitored. Serendipitous Sun glint observation would be performed also outside the MasCAM FoV using the optical GNC sensors on each face of MASCOT (Ho et al., 2017), the PhotoElectric Cells (PEC) and the Optical Proximity Sensors (OPS) in photodiode-only (LED off) mode. Further using the normal attitude changes or dedicated rotations of the sailcraft, the specular reflection of the Sun off the sail foil can provide an improvised flying spot scanner at ~½° spot resolution to monitor surface properties of the membrane in a wider area.

Used in this way during the long cruise to demonstrate solar sailing, MASCOT would provide a unique opportunity to investigate aging of various materials used in space technology, especially aluminized polyimide films. The behavior of these metallic surfaces has a key role in space industry. A failure of a space mission may be the result of a change of metallic structure properties caused by the environmental effects. Therefore, all of the materials planned for space applications have to be evaluated for their behavior under particle and electromagnetic radiation (ASTM, 2010; ECSS, 2008). Thin metallic foils are especially sensitive to the ion irradiation. The free electrons within the metals can in a well-defined energy range recombine with the solar wind protons into neutral hydrogen atoms. Recombination processes and their consequences have direct effects onto the foils' physical and thermo-optical properties. The result of a recombination of metal free electrons and solar protons is a formation of bubbles filled with hydrogen molecular gas (Sznajder, et al., 2015). The thermal IR radiometer MARA's working bands may help to indicate surface changes caused by the ion bombardment. For instance, the typical size of the Hydrogen molecular bubbles caused by the proton bombardment of aluminized surfaces (Sznajder, et al., 2015) lie in the first of MARA's bandwidth ranges, 5.5–7 μm. By use of the MicrOmega instrument and image processing methods we will be able to retrieve distributions of morphological structures which appear during the specimens' exposure to the space environment. By making the images in well specified time steps we will be able to estimate how the distribution evolves with time. The measuring devices of MASCOT will provide an opportunity to track real degradation effects of the mounted specimens. The retrieved data may be then used as a reference for the terrestrial laboratory aging tests.
In an early development mission like GOSSAMER-3, an attached MASCOT could also exercise the previously mentioned design option of sailcraft control by the separable payload module for the purpose of demonstration of new avionics topologies and concepts.

Circumstances changed with the termination of the GOSSAMER-1 project and the return of MASCOT instrument spares after the landing in late 2018. But everything that would be needed to re-assemble the MASCOT FS still exists if anyone wished to do so.
Although the idea of achieving with GOSSAMER-3 the maybe first truly lightweight sailcraft long since followed the GOSSAMER Roadmap into limbo, the idea to fly a Flight Spare lingers. It has led to its own family of MASCOT follow-on-by-re-use studies. Conversely, independent of their re-use approach, most MASCOT follow-on studies involve a rugged but sensitive low-noise MEMS accelerometer as a combined science instrument and bus GNC sensor that was originally developed to sense the structural vibrations of GOSSAMER-1 before, during and after deployment. Also, the concepts developed for the operational photovoltaics of the GOSSAMER-1 BSDUs and resource sharing among CSCU and BSDUs were continued in the MASCOT2 study's (Lange et al., 2018a,b) power subsystem design in combination with features of the efficiency-optimized power subsystem design of AsteroidFinder/SSB, and would have been used to upgrade the MASCOT FS for a ride on GOSSAMER-3 potentially ending with a long-lived landing on any suitable NEA conveniently within reach.
The experience of comprehensive ad-hoc as-is re-use considerations became valuable for our later and in particular the currently ongoing follow-on studies. (Seefeldt et al. 2019a, 2021) The avionics developed originally for the GOSSAMER-1 BSDU is another example of this strategy. It was first modified for use in the ROBEX Lunar Analog Mission project involving a field campaign on top of Mt Etna, Sicily, Italy. There, it became the avionics of the networked Remote Units (RU), a set of MASCOT-sized surface science packages to be autonomously deployed by a rover. (Lange et al., 2015b) It returned in a non-networked single client role in the GOSOLAR photovoltaic (PV) membrane demonstrator payload, as redundant Payload Board Computer (PBC). (Grundmann et al., 2016; Spietz et al. 2017; Spröwitz et al., 2018, 2019a,b; Wippermann et al. 2019)

Dusting Off

Large membranes in space near Earth are by their area much more likely to suffer from impacts of space debris. A mission like GOSSAMER-3 which was envisaged to traverse much of the Earth-Moon system, from GTO or perhaps even high LEO to escape, with a ≈2500 m² exposed area, obviously appears very attractive as a mapping mission for artificial space debris and the natural micrometeoroid environment. It had the potential to become the biggest dust impact detector flown. Also, beyond the technical advantages there already was a connection in the project to Space Situational Awareness (SSA) by all the three chosen mission types to be studied by the Science Working Groups (Dachwald et al., 2014; M. Macdonald, et al., 2014; C.R. McInnes, et al., 2014), and an ongoing in-house development of space debris sensors. Different types of in-situ impact detectors have been utilized in the past or are currently under development. The primary aim of most of those detectors has been the detection of micrometeoroids. However, the knowledge gained with micrometeoroids impact detection can be applied to develop space debris detectors. This has been previously demonstrated by instruments such as GORID (Geostationary ORbit Impact Detector) (Drolshagen et al., 1997), a successor of dust detectors flown on the Galileo, Ulysses and Cassini spacecraft. However, the data gained by detector systems flown until now, has not been implemented into the environmental models. The main reason for that is the high uncertainty of the measured data that makes model validation impossible (Flegel et al. 2010; Horstmann

et al., 2018; Wegener et al., 2004; Wiedemann et al., 2018a,b). To analyze the quantity of space debris and micrometeoroids in space, an innovative in-situ impact detection method has been developed at DLR (German Aerospace Center) in Bremen, Germany (Bauer, 2015; Bauer et al., 2012, 2013, 2015). The method SOLar generator based Impact Detector (SOLID) uses solar panels for impact detection. A layer of thin conducting traces localizes the impact and determines the size of the damage. Transients in the photovoltaic cells electrical output provide the time of impact. Since solar panels provide large detection areas, this method allows the collection of large amounts of data, to be used also for model validation. The SOLID method is a low mass (<200 g/m²) and low-cost detection concept, that offers the potential to realize a large detection area to provide data on the currently non-ascertained objects in-orbit. In this sense, the GOSSAMER spacecraft offers a unique opportunity to realize for the first time in space history a very large in-situ impact detector. A particular advantage of the SOLID method is that impacts can be re-analyzed and confirmed after the event because it enables measurement of the damage in addition to the instantaneous effects. Also, the damage-detecting layer is not exposed except at the immediate edge of the impact damage and thus better protected from space weathering e.g. by erosion or flaking off of conductive traces at the outer surface.
The micrometeoroid detector ALADDIN (Arrayed Large-Area Dust Detectors in INterplanetary space) has been flown on IKAROS. It uses PVDF piezo-electric signal-generating material and has an area of 0.54 m². However, all these impact detection methods add an area-specific mass several orders of magnitude heavier than that of the sail foil except in the relatively small areas used for the sail's power supply. Here, SOLID can be implemented more easily than on rigid panels because of the very similar technology used to make flexible photovoltaic cell connections and provide the flexible panel harness. (Spietz et al., 2017; Spröwitz et al., 2019a,b, 2018; Wippermann et al., 2019) However, other methods to pick up impact signals from the very large membrane of GOSSAMER-3 were considered for study, including the anyway envisaged accelerometers, microphones for sound waves travelling in structural elements attached to the membrane such as the sail rigging, and using the instruments measuring the plasma environment of the sail to monitor ageing. Quite likely, most or all of these methods of debris impact detection would have been prototyped on GOSSAMER-2 and exposed to the debris-laden environment in high LEO resulting from the Chinese ASAT test and the IRIDIUM/COSMOS collision, and SOLID-equipped modules were considered to be flown in the PVX of GOSSAMER-1 and on GOSOLAR.

GOSSAMER ROADMAP – A LAST STATUS

It may be said that like an old soldier, the GOSSAMER Roadmap did not die, it simply faded away. With the cancellation of the GOSSAMER-1 project midways in its qualification campaigns at DLR and the too early loss of key proponents at ESTEC, the Roadmap decayed with the half-life of email traffic without a cause. It may have appeared overly ambitious at its beginning, particularly regarding schedule aspirations, but with hindsight and the experience of designing and building MASCOT hardware within 2 years from PDR to FM delivery, (Grimm et al., 2019a; Grimm and Hendrikse, 2019) this appearance will have to be revisited as far as its technical potential is concerned. In the same manner, when evaluating the achievements in qualified hardware of the GOSSAMER-1 project, it is worth noting that these were achieved with one tenth of the resources made available to MASCOT at the same time and at the same place.

How is the air up there? – QB50 and the GOSSAMER Roadmap

In 2010, the GOSSAMER Roadmap having just been established and presented at the Second International Symposium on Solar Sailing in New York, a team of ESA and The von Kármán Institute for Fluid Dynamics (VKI) people made first advances to initiate an EU project called QB50 to deploy a flotilla of cubesats in very low LEO to study the upper atmosphere. (Masutti et al., 2018; Thoemel et al., 2014) Following numerous spirited visionary diplomatic missions of late Ruedeger Reinhard between ESA, VKI and DLR, GOSSAMER-1 was integrated into the QB50 project concept to fly as an in-orbit demonstrator.

While technically independent from the QB50 dispenser systems, GOSSAMER-1 had been integrated into the overall concept as a flagship payload, and went with the QB50 project through several changes of the baseline launch opportunity as may be expected for rideshare secondary payloads. Depending on these varying launch options, it could have been configured in several variants, adding units incrementally to upgrade from an externally powered short lifetime attached payload as shown in Figures 17 towards a fully independent photovoltaic-powered free-flyer, only limited by its orbital decay lifetime. The latter ultimately became the baseline configuration of the GOSSAMER-1 solar sail deployment demonstrator, also because self-reliance improved launch negotiation flexibility for QB50 which itself moved to more modular small deployers for its sets of cubesats. The upper stage attached variant planned in early stages of the joint QB50 and GOSSAMER-1 concepts picked up an earlier DLR concept, presented at the ISSS 2010 in New York, compare Figure 26.

Finally, most of the QB50 cubesats with atmospheric science payloads were deployed from the ISS and thus stayed safely beneath the altitude of the orbits used by crewed spaceflights and the station itself. A second set consisting mainly of QB50-associated technology demonstration cubesats was launched on June 23rd, 2017, on flight C38 of the Indian PSLV launch vehicle to a ≈505 km circular SSO. Although a launch on ISS traffic with the first QB50 payloads deployment would have been feasible for GOSSAMER-1 as such, deployment of a 40 kg class spacecraft from a platform designed for multiple cubesat delivery may have been unlikely, and in any case the second launch would have been the preferable option.

Figure 26 – Early concept of GOSSAMER-1 coupled to a dedicated launch, using an upper stage attached configuration to share power and telemetry. The later QB50 project involved a mission scenario which turned out to be very similar to this early concept. Note the similarity to the central deployer of the 1999 deployment test (Figure 1) and that BSDUs are not shown in either configuration.

The connection with QB50 put GOSSAMER-1 into a similar timeline and project management situation as another DLR Bremen centered project, the Mobile Asteroid Surface Scout nanolander, MASCOT, in that it had to prepare for a relatively early launch date set by the main mission. (Grimm et al., 2020; Ho et al., 2017) However, in this case the timeline constraint was only driven by planning for a shared launch with a to-be-defined primary payload into Earth orbit, and not by a more rigorously constrained interplanetary launch window. A firmer connection for GOSSAMER-1 would only have become effective with manifesting of a payload combination, 18 months or less ahead to the at that point in time planned launch date of the primary payload. Also, the mission objectives of the primary payload on the respective launch opportunity, of QB50, and of GOSSAMER-1 are not necessarily linked as in the case of MASCOT and HAYABUSA2. Since the free-flyer variant of GOSSAMER-1 had in the meantime through a number of QB50 launch concept changes become the baseline design, both schedules could be decoupled easily at any time, avoiding cross-coupled risks at the programmatic level at all times.

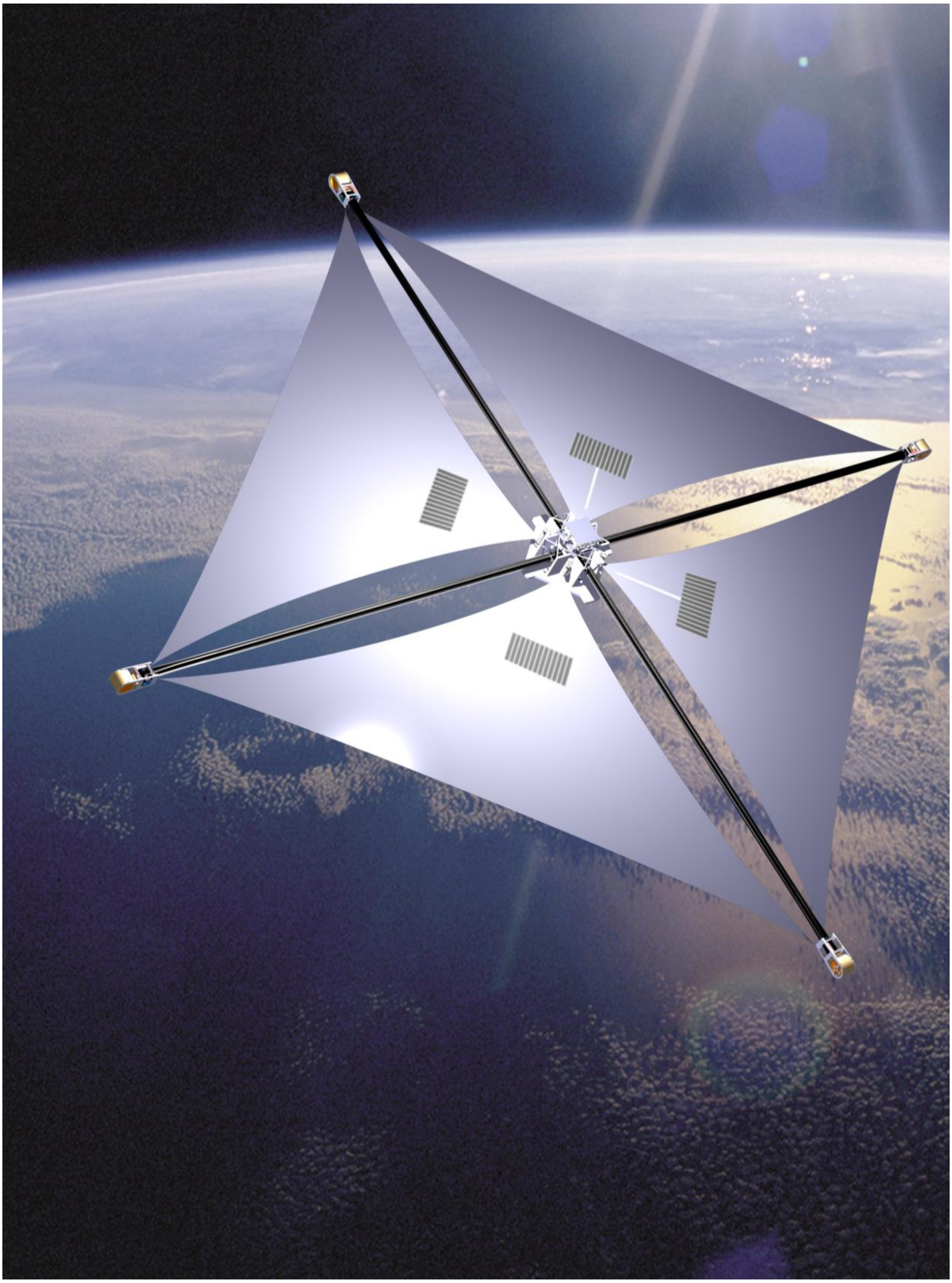

Figure 27 – Matured concept of GOSSAMER-1 in free-flyer configuration. Note that BSDUs are shown, although without outer structure and avionics (cf. Figure 9) to offer a view of the boom spool assemblies.

This was ultimately taken advantage of when interest in solar sailing faded behind a new vigorous programmatic interest in large-scale photovoltaics which ultimately led

to the GoSolAr project (that meanwhile was also recently and suddenly terminated early).
Driven by this programmatic change, a reorientation of the Gossamer project at DLR followed, which effectively suspended the DLR-ESTEC Gossamer Roadmap indefinitely as of August 2014 and paused until further notice all design and integration work on the PFM models of the Gossamer solar sail deployment demonstrator spacecraft. The work already begun with BBM and EM hardware and supporting test stands for Gossamer-1 was guided towards a coordinated conclusion deadline at first set in mid-2014 (i.e., virtually immediately) and then extended to the end of 2015. The scope of work until then included completion of the EQM based on already built and in-production hardware in "¼-QM" configuration with 1 of the 4 BSDUs and thus at least one item, each, of all of the new and critical mechanisms and technologies. A series of pre-qualification tests limited to the key technological components was undertaken prior to integrated testing. Full qualification was done on system level because of the intricate interaction of all key functions in the deployment sequence. The ¼-QM setup consists of one complete set of CSCU-side interfaces, one boom, and one BSDU train, in some tests with two adjacent sail quadrants which are supported at their other ends by controlled linear motion drives replacing the neighbouring BSDUs there. (Seefeldt et al., 2016, 2017d; Seefeldt and Spröwitz, 2016) All key Gossamer deployment technology components in this setup are built to full QM hardware level. Their design is very close to the originally envisaged PFM free-flyer configuration, and in some cases advanced over the last complete state of the PFM design.

Table 6 summarizes the parameters of the three Gossamer Roadmap spacecraft as they had evolved up to this point:

Table 6 – The Gossamer Roadmap spacecraft and design goals as last envisaged at end of July 2014

spacecraft	Gossamer-1	Gossamer-2	Gossamer-3
<u>planned</u> launch in	with QB50 technology demonstrators; (*others were launched on June 23rd, 2017, 03:59 UTC aboard PSLV-XL C38*)	≥ mid-2019 (likely 2021…2023)	≥ mid-2021 (likely ≥2025)
deployed size	(5 m)²	≈ (20…25 m)²	≈ (50 m)²
initial orbit altitude	320·700 km (drag dominant) (*actual QB50 technology demonstrators: 496·518 km*)	SRP dominant (≥ ~700 km)	spiral-up feasible, >2000 km perigee, apogee boost engine not excluded
deployed sailcraft mass, best estimate	19.463 kg	≈55…85 kg *	≈85…180 kg *

effective sail area	22.4 m²	≈330…520 m² excluding tip vanes etc.	≈2050…2250 m²
sail area loading	868.9 g/m²	≈160…260 g/m²	≈40…90 g/m²
characteristic acceleration a_c	≈0.009 mm/s²	≈0.05…0.075 mm/s²	>0.1…≈0.2 mm/s²
undeployed volume	Ø80·50 cm³	≤80·80·100 cm³	≤80·80·100 cm³
launch mass (incl. margins)	37.573 kg	≤200 kg, ASAP & ESPA compatible	≤200 kg, ASAP & ESPA compatible
sail foil	7.5 µm	2.5 .. 7.5 µm	≤2.5 µm
reflective layer	2·100 nm Al		
design lifetime	90 days	~1 year	~1…2 years
sailcraft orbital lifetime	~90 days predicted	≤25 years @ ≤2000 km indefinite @ >2000 km	indefinite (heliocentric)
BSDU orbital lifetime	<10 years predicted	≤25 years @ ≤2000 km indefinite @ >2000 km	indefinite (geo/heliocentric)
mission objectives	deployment	limited orbit control, attitude control, comparison of several attitude control methods	full orbit and attitude control, Earth-departure and lunar fly-by capability
controlled disposal	mandatory, by decay	if <2000 km mandatory, by down-spiraling and/or decay; alternatively up-spiralling to >2000 km	design goal: Earth escape, otherwise long-term safe orbit beyond GEO
observation payload	6 on-board cameras on CSCU & BSDUs	≥2 wide-angle cameras on CSCU, sail & plasma environment (TBC)	≥1 wide-angle camera, 1 narrow-angle camera on CSCU sail monitoring sail environment
launch arrangements	free-flyer, universal piggy-back, *with QB50*	universal piggy-back compatible with all launchers, ASAP & ESPA	universal piggy-back compatible with all launchers, ASAP & ESPA

* see Seefeldt et al., 2021

What next – Science?

Although primarily a technology development project, the purpose of the GOSSAMER Roadmap was to develop a new advanced fuel-free method of propulsion for interplanetary spaceflight and for a whole host of mission scenarios which can be found in sail-based mission analysis literature, e.g. McInnes, 1999 for an overview.

To guide the evolution of the three successive GOSSAMER Roadmap technology development spacecraft to purpose, already during the early phases of the Roadmap development work, three parallel science working groups from different scientific communities were tasked to identify and develop a candidate mission, each, for the first scientific solar sail missions envisaged after solar sail technology had reached the level expected to be attained by GOSSAMER-3.
The guidelines given were that any technology to be assumed for these first science missions was to be very closely based on the expected state of the art to be reached by the end of the Roadmap development, echoing the demand of science missions for very high TRL units. No new technology development was to be expected between whichever would be the first science mission and the final flight of the Roadmap, only modest adaptation to the specific mission. Thus, the first science mission sailcraft was baselined as a lessons-learned refined and only very moderately upscaled variant of the GOSSAMER-3 sailcraft. Its detailed design being far off in the Roadmap's future, a configuration and performance to be expected was postulated at a level which was regarded as a likely outcome at the time of the science working groups' successive sessions. It was only slightly refined in course towards the issued reports and subsequent papers. With hindsight, the relation of these first envisaged science missions' sails to GOSSAMER-3 would have been in a similar way as the HAYABUSA2 spacecraft and its expanded suite of landers, sub-spacecraft and payloads relate to the first HAYABUSA mission spacecraft design – a rebuild with all lessons learned applied and moderate capability upgrades and modernizations. (Kawaguchi et al., 2008; Tsuda et al., 2016) At the time, the maximum sail size to be safely expected as feasible under these assumptions was estimated to about (60…75 m)². Towards the end of the work on GOSSAMER-1, up to (85 m)² were considered to be within reach, sometimes labelled as "being within the comfort zone" of the already existing technology elements.

The following three missions were identified and studied over a period of two years. Each was presented in a comprehensive peer-reviewed paper:

- a space weather early warning mission stationkeeping with Earth ahead of the Sun-Earth Lagrange point L_1 towards the Sun, using the sail thrust to augment Earth's gravity in the balance of orbital forces to generate an artificial Displaced L_1 point (DL1), and carrying a very lightweight suite of plasma instruments. The DL1 position was expected and required to at least double the warning time for oncoming solar storms which can disturb power grids, knock out spacecraft services, hinder radio communication, and increase high altitude radiation on Earth. Sail degradation during the mission would not lead to loss of stationkeeping, merely the displacement distance would recede in proportion back towards the purely ballistic L_1 region of halo orbits. (McInnes et al., 2014)
- a Solar Polar Orbiter for which the solar sail is used to raise the inclination of its heliocentric orbit much further than possible by gravity-assist fly-bys, chemical or electrical propulsion combined. A heavier helioseismic imaging payload could be raised in inclination sufficiently to observe the polar regions of the Sun, and could progress under sail power to somewhat higher latitudes still within the set lifetime, with the potential to reach 90° inclination soon after in extended mission and the option to separate from the sail. A light-weight plasma instruments payload could reach exact solar polar orbit within the required mission duration where the sail would be jettisoned in any case to remove its influence on the plasma environment to be studied. The sail itself

does however not run out of fuel to continue in either case, and could in theory be used for any useful minimal payload mass extended mission purpose progressing to retrograde inclinations. (Macdonald et al., 2014)

- a multiple NEO rendezvous and fly-by mission to visit and rendezvous with, for at least several rotation periods of the respective object, at least three significant NEAs out of a pre-selected population, and to perform faster fly-bys at additional other NEOs within the set lifetime of a decade. (Dachwald et al., 2014)

The requirements of all these missions can *uniquely* be met using solar sail propulsion. Where any other alternative exists at all, the sail-based solution provides a very substantial performance margin over the second-best propulsion solution. Their requirements combined were intended to guide the Roadmap development towards GOSSAMER-3.

Table 7 – The GOSSAMER Roadmap Science Working Group's spacecraft and design goals as envisaged in 2013 (Dachwald et al., 2014; McInnes et al., 2014; Macdonald et al., 2014)

mission	Multiple NEA rendezvous	Space weather Displaced L_1	Solar Polar Orbiter
launch in	2023+	2023+	2023+
deployed size	(54…65 m)²	(65 m)²	(100…125 m)²
initial orbit	$C_3 \geq 0$ km²/s²	$C_3 \geq 0$ km²/s² (GTO + kick stage)	
deployed sailcraft mass, best estimate		110 kg	~250…510 kg
non-sail mass	40…60 kg	15 kg	~45…305 kg
characteristic acceleration	0.3 mm/s²	0.3 mm/s²	0.29…0.54 mm/s²
sail foil		2 µm	2.5 µm
design lifetime	10 years	10 years	10 years
mission objectives	triple pre-selected NEA rendezvous with stationkeeping + fast NEO fly-bys of opportunity	10 years of operation at DL_1 at $2 \cdot L_1$ distance from Earth	solar pole observation for >1 rotation of the Sun at a time
science payload	12 kg	1 kg	5^a / 15^b / 40^c kg

instruments	multispectral imager, Vis-NIR point spectrometer, IR radiometer, 3 drop-probes	MAGIC magnetometer, ChaPS plasma analyzer, SWARM Langmuir probe	[a] total solar irradiance (TSI); [b] TSI, doppler velocity, magnetic field; [c] Doppler & Stokes imager, coronagraph, magnetometer, solar wind analyzer, energetic particle detector
launch arrangements	universal piggy-back compatible with all launchers, ASAP & ESPA, design goal 'micro'	universal piggy-back compatible with all launchers, ASAP & ESPA, design goal 'micro'	universal piggy-back compatible with all launchers, ASAP & ESPA, possible dual launch of 1 light & 1 heavy payload in one 'mini' slot

It was noted at the time that further optimization of the trajectory of the triple NEA rendezvous mission could bring down the requirements on the sailcraft to about 0.2 mm/s² characteristic acceleration, or (39…48 m)² sail size, as a final design goal for a 10-year mission duration. (Dachwald et al., 2014)
Within 3 years, this goal was achieved by the solar sail trajectory development community, and surpassed in the number of rendezvous (up to 5), stay duration (≥100 days, each), and mission options per launch date (10's to 100's). (Peloni et al. 2016, 2018)

Table 8 – Recent progress of near-term solar sail Multiple NEA Rendezvous (MNR) trajectory studies (Dachwald et al., 2014; Dachwald and Seboldt 2011; Grundmann et al. 2017, 2019b; Jessberger et al., 2010; Johnson et al., 2011; Peloni et al. 2016, 2018; Seboldt et al., 2013)

year	study	mission performance: number of targets & stay duration, NEA selection constraints	required characteristic acceleration, a_c
2000	DLR ENEAS study	2 fast flybys & 1 rendezvous in 5 years	0.14 mm/s²
2005	ENEAS-SR	1 sample return & 117 days stay in 10 years	0.10 mm/s²
2005	ENEAS+ / ENEAS+SR	3 rendezvous/sample return in 10 years	0.22 mm/s²
2011	GOSSAMER NEO reference	3 very slow flyby-rendezvous >1 rotation, in 10 years	not specified
2011	Johnson et al.	3 rendezvous of ~30 days, each, in 6 years	0.35 mm/s²

2014	Gossamer NEO reference	3 rendezvous of ~100 days, each, in 10 years	0.20 mm/s^2
2016	Peloni et al.	5 rendezvous of >100 days, each, in 10 years, ≥1 PHA, others only from NHATS	0.20 mm/s^2
2017	Peloni et al., for PDC 2017	4 rendezvous of >120 days, each, and sample-return in 11 years, ≥1 PHA, others only from NHATS	0.20 mm/s^2

For comparison, a set of proposals based on a similar mission profile, with one or three target NEAs and a sample-return option for each rendezvoused object had already been proposed as a small spacecraft mission within the space sciences program of DLR in the 2000's on the conventional science missions track under the designation ENEAS (Exploration of a Near-Earth Asteroid with a Solar Sail). (Dachwald et al., 2014; Dachwald and Seboldt 2011; Jessberger et al., 2010; Seboldt et al., 2013) Except for the triple NEA sample return profile the envisaged spacecraft were generally within the later expected capabilities for a first science mission sailcraft as assumed for the Gossamer Roadmap Science Working Groups' spacecraft, and the single-target missions were close to the properties expected for Gossamer-3. The characteristic parameters of these missions are summarized in Table 9; *note* that these proposals did not yet use the jettisonable BSDU concept, and thus sailcraft mass equals launch mass:

Table 9 – The ENEAS family of NEA rendezvous and sample-return missions, and NASA MSFC ACO multiple NEO rendezvous for comparison with a contemporary study (Jessberger et al., 2010; Dachwald and Seboldt, 2011; Dachwald et al., 2014; Seboldt et al., 2013; Johnson et al., 2011)

mission	ENEAS	ENEAS -SR	ENEAS+	ENEAS +SR	NASA MSFC
target objects	(175706) 1996 FG$_3$		2000 AG$_6$, (65679) 1989 UQ, 1999 AO$_{10}$		1999 AO$_{10}$, (99942) Apohis, 2001 QJ$_{142}$
mission type	single rendezvous	single sample-return	triple rendezvous	triple sample-return	"multiple" (triple) rendezvous
deployed size	(50 m)^2	(70 m)^2	(70 m)^2	(139 m)^2	(80 m)^2
sail area	2500 m^2	4900 m^2	4900 m^2	19321 m^2	6400 m^2
sail area loading	59.2 g/m^2	60.2 g/m^2	38.0 g/m^2	38.0 g/m^2	38.0 g/m^2
characteristic acceleration	0.140 mm/s^2	0.100 mm/s^2	0.218 mm/s^2	0.218 mm/s^2	0.350 mm/s^2
deployed sailcraft = launch mass	148 kg	406 kg	186 kg	734 kg	243 kg
non-sail mass	75 kg	295 kg	75 kg	295 kg	
mission duration	4.2 years <5 years	10.0 years	7.6 years	10.1 years	~6 years

science payload	5 kg		5 kg		
instruments	CCD camera, IR-spectrom., magnetom.		CCD camera, IR-spectrom., magnetom.		

…and the veils fall

Following the sudden termination of GOSSAMER-1 project and after the work using or further developing the already built EQM, deployable membrane development was redirected towards 2D-deployable very large photovoltaic arrays in the GOSOLAR project on which we report in Grundmann et al., 2016, Spietz et al. 2017; Spröwitz et al., 2018, 2019a,b; Wippermann et al. 2019. It was also terminated in early 2020. In parallel, deployable membrane technologies for drag sails to deorbit low Earth orbit satellites were also developed with partners. This work still continues.

Conclusions

The GOSSAMER-1 project created the only form of system hardware achieved by the ambitious DLR-ESTEC GOSSAMER Roadmap for Solar Sailing. But its qualified deployment technology using separable Boom Sail Deployment Units is not the only product of this attempt to make solar sailing technology an interplanetary reality. Not least because of the idea of a roadmap, many ideas and concepts were also explored early on, for the envisaged sailcraft following the demonstration of the unique sail membrane deployment technology created. During these efforts it became apparent that already with the technology applied in and qualified by the GOSSAMER-1 project, the final step in the Roadmap, GOSSAMER-3, would have been capable of first-generation solar sail science and application missions. Also, the association of GOSSAMER-1 with the successful QB50 project deploying many cubesats to explore the upper atmosphere and to demonstrate nanospacecraft technologies provided many important impulses to GOSSAMER-1 leading to a resilient and flexible spacecraft design capable of adapting quickly to launch vehicle changes. In the absence of solar sail projects at DLR we hope to pass these experiences and lessons learned on to those who want to fly by sail.

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Dünnschichtsolarzellen für die Raumfahrt“. The future that never was for GOSSAMER-2 and -3 shared countless seeds of synergetic inspiration with the contemporary development of MASCOT, MASCOT2, the OKEANOS Lander, and their science instruments, in particular from conversations with Uli Auster, Jean-Pierre Bibring, Jens Biele, Ralf Boden, Sebastian Fexer, Matthias Grott, Maximilian Hamm, Vincent Hamm, Lars Hauer, Alain Hérique, Lars Kesseler, Ekkehard Kührt, Roy Lichtenheldt, Stefano Mottola, Ivanka Pelivan, Cedric Pilorget, Dirk Plettemeier, Nicole Schmitz, Elisabet Wejmo, Friederike Wolff, and Christian Ziach. The missions envisaged for GOSSAMER-2 and -3 then and since took on a whole new dynamic also thanks to the work of Jeannette Heiligers, Iain Moore, Alessandro Peloni, and Giulia Viavattene. GOSSAMER-1 and the GOSSAMER PhotoVoltaic eXperiment (PVX) were funded by DLR German Aerospace’s Research and Development program for technology of space systems as ”GOSSAMER-1 Deployment Technology Demonstrator Project”. The studies of the many GOSSAMER-1 launch opportunities were kindly spurred on by the mutual support barter launch agreement with the EC Project ”QB50-An international network of 50 CubeSats for multi-point, in-situ measurements in the lower thermosphere and re-entry research” which has received funding from the European Union’s Seventh Framework Programme for Research and Technological Development under grant agreement № [284427]. The key configuration studies for GOSSAMER-1 and GOSOLAR were conducted at the DLR Bremen Concurrent Engineering Facility (CEF). JTG acknowledges the significant influence of Glimour et al. 1987, Kornfeld and Duboff, 1965, and Peterson et al., 1982, on the philosophy of the GOSSAMER-1/2/3 project and its launch mission analysis exchange with QB50.

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