INVESTIGATING NEW SPACE STRUCTURES WITH THE FOCUS EXPERIMENT

Philipp Reiss[(1)], Elias Breunig[(2)], Philipp Zimmerhakl[(3)], Nora Newie[(4)] Andreas Zeiner[(5)]

[(1)] *Technische Universität München, Boltzmannstr. 15, 85748 Garching, Germany, philippreiss@mytum.de*
[(2)] *Technische Universität München, Boltzmannstr. 15, 85748 Garching, Germany, elias.breunig@googlemail.com*
[(3)] *Technische Universität München, Boltzmannstr. 15, 85748 Garching, Germany, philipp.zimmerhakl@mytum.de*
[(4)] *Technische Universität München, Boltzmannstr. 15, 85748 Garching, Germany, Nora_Newie@mytum.de*
[(5)] *Technische Universität München, Boltzmannstr. 15, 85748 Garching, Germany, andreas.zeiner@mytum.de*

ABSTRACT

FOCUS (First Orbital Curing Experiment of University Students) is a space experiment which was launched in February 2011 within the European REXUS (Rocket-borne Experiments for University Students) programme. The experiment, realized by a group of aerospace students from Technische Universität München, was performed in order to investigate the feasibility of a new technology concerning the manufacturing of ultra large space structures. The approach to build these structures by using a composite material, respectively a construction with glass fibre strings which is in situ curable, has been tested and verified on a parabolic zero gravity flight on-board a sounding rocket. This paper describes the scientific approach as well as the technological development of the composite structure and the design of the experiment. Furthermore the realization of the experiment including the results of the flight campaign is presented. Finally the potential next steps in developing this technology and its possible future applications are discussed.

1. INTRODUCTION

The idea of In-Orbit-Rigidization (IOR) as an enabling technology component for ultra large space structures and solar sailing spacecraft is not new. In fact, its origins can be dated back as far as the beginning of the space age, now more than half a century ago. So-called gossamer structures have clearly been identified as key to achieving the packaging efficiencies required to go from very large deployed sizes down to the launcher payload envelope [2]. Furthermore the variety of conceivable structural architectures is increased through greatly simplifying the deployment mechanisms [7].
Whilst there have been several flight demonstration of purely inflatable membrane structures, the means of rigidizing these structures for prolonged application periods is still in the early stages of development. Considerable effort was undertaken in recent years, but little of the technology has made it to flight demonstration yet. Most projects completed so far have been only laboratory breadboards or theoretical assessments.
The FOCUS experiment, in difference, was aimed to gather new knowledge and heritage by going for real flight testing quickly, without spending too much time in analysis.
The goal of the experiment was declared to qualify and test a deployable, in-orbit-rigidizable truss structure on the µ-gravity flight offered by the *REXUS* rocket programme. The central questions that have been posed at the beginning of the 16 months long project were:

- How must an in-orbit-rigidizable composite truss be designed to function in space and withstand space environment?
- How can the raw structure withstand ground handling, prolonged storage and launch loads?
- Which curing method and materials are most promising?
- How does an uncured structure behave during deployment and curing?
- How can such a structure be manufactured cost- and time-efficiently enough to test it in a student experiment on a sounding rocket?
- What are the main problems in terms of craftsmanship when building and flying the real hardware?

2. EXPERIMENT COMPONENTS & DESIGN

2.1. DEPLOYABLE RIGIDIZABLE STRUCTURE

The primary component of the FOCUS experiment was the deployable and rigidizable composite structure to be tested during flight. This structure was designed completely new from scratch to meet the conditions and requirements of the REXUS launcher.
In principle a rigidizable structure can be realized in several ways and can be divided into 3 interdependent technology regimes – in the method of rigidization, the material and design of the load bearing component and last but not least the stowing and deployment concept.
Considering the first component, there are a number of available technologies to choose from. Those include thermally cured composites, light initiated curing, inflation gas catalysed reaction, shape memory polymers, plasticizer or solvent boil-off composites, foam rigidization and some more.
With an application prospect in the regime of not only ultra-large but also ultra-light structures – as required

Proc. '20th ESA Symposium on European Rocket and Balloon Programmes and Related Research'
Hyère, France, 22–26 May 2011 (ESA SP-700, October 2011)

for instance by solar sailing missions – it is clear that a compliance of the rigidization method with the use of high modulus fibre materials is very desirable. Therefore the rigidization methods to be chosen from where clearly reduced to the curing of resins, usable as composite matrix material.
A further study of this specific technologies led to the conclusion that light initiated curing is the most promising method, because it can be carried out using the solar radiation flux available on the respective orbit in space. The other two methods mentioned both require special equipment on the spacecraft, which might not be compatible with many mission concepts, either in terms of mass or system complexity.
The drawback that comes with the selection of light initiated curing is a limitation in choice of fibre material and thickness of the composite, as it must stay penetrable for the used wavelength, which are of soft UV in most cases.
All in all this led to the choice of glass fibre as reinforcement material. Obviously glass fibre is not the most capable fibre material available. Mass to stiffness ratios of carbon fibre materials can be better by almost an order of magnitude. Nevertheless polymer fibres like aramid, dyneema or spectra etc. can be regarded as promising substitutes for the glass fibre in the future and will then provide also much better structural performance alongside the requirement for partial transparency. More research is clearly required in this matter.
The actual resin that was used on FOCUS is a modified acrylic adhesive which was delivered by the company Cyberbond [8] and specifically tailored for the application as a composite matrix. It was mainly chosen because of its fast curing time of less then 60s when exposed to standard solar flux. The glass fibre used was a standard industry grade UD E-Glass with a bundle diameter of 3.5mm.
The major task to be solved when integrating these components to form an in-orbit-rigidizable structure is to contain both, matrix and fiber, during launch and pre-curing phases and keep the resin from boiling off into the vacuum environment.
A protective membrane is needed to surround the raw composite for these reasons.
Several possibilities have been investigated and finally a 70µm polyethylene membrane was selected because of its ruggedness and good chemical compatibility as well as easy availability. More favourable in terms of low modulus, which facilitates the folding of the whole structure, would have been silicones, but those turned out to be incapable of containing the acrylate properly for chemical reasons. A cross-sectional view of the final design of a single deployable beam is depicted in *Fig. 1*.
The completely assembled structure is a hexapod style conical truss. Its uses six of the individual beams that are connected at a radius of 70mm at the bottom and 50mm at the top and are arranged equally on each radius in a distance of 120° *(Fig. 2)*. Each beam reaches from the bottom radius to the top one, thus forming a slightly conical shape. Folding of the whole structure into the storage container is achieved through zigzag folding of all individual beams. The folded package is dense enough to restrain itself without additional means.
The manufacturing process for the composite beams was based on a resin infusion process, which served the double function of efficiently evacuating the containment membrane of any gaseous components. The residual pressure inside the beams was therefore only slightly above the vapour pressure of the resin at room temperature.
Prior to the resin infusion, the glass fibre component has been inserted into the polyethylene membrane, together with additional elements at the ends of the membrane tube that served as connection to the payload casing and sealing of the beam after resin infusion.
The rigidization by UV light at a wavelength of approximately 400nm was decided to be achieved with a dedicated UV light source. This approach had to be chosen because the limited control of rocket attitude and launch time would make it nearly impossible to properly illuminate the experiment module through any kind of window. Deploying the structure to the outside was also rejected because of the strong desire to recover the structure for evaluation after re-entry.

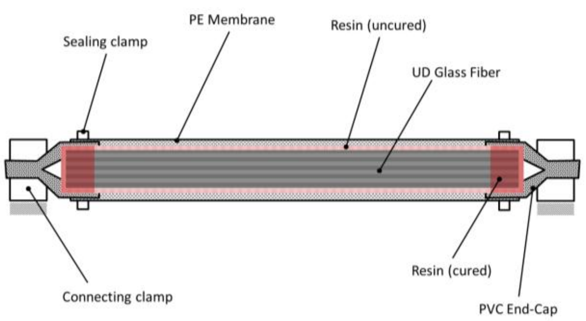

Figure 1. Detailed design of a FOCUS composite beam

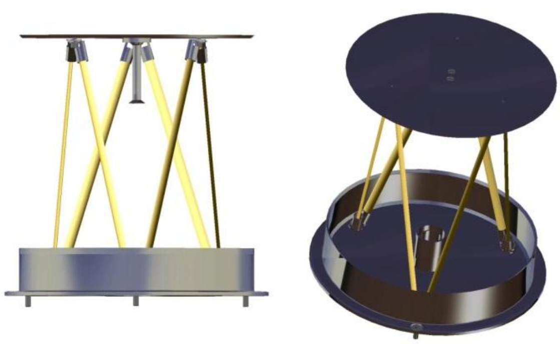

Figure 2. Deployable composite structure integrated with the Storage Container

2.2. RELEASE AND CATCH MECHANISM

In order to release the composite structure from its stowed position and subsequently deploy it, a “hold down and release mechanism” (HDRM) was used. This device has been developed within a previous work at TUM [1] and was qualified on a former REXUS experiment.
Its functional principle is based on burning a dyneema wire by heating a coil of resistive wire wrapped around it. Cutting the dyneema will release two brackets, which are used to fix a cone connected to the upper cover plate of the payload structure (*Fig. 3*). Once these brackets release the cone, the cover plate and subsequently the structure is pushed away and deployed with the help of a spring (*Fig. 4*).

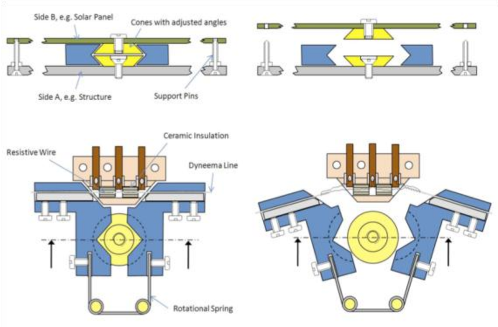

Figure 3. Sketch of HDRM and Functional Principle

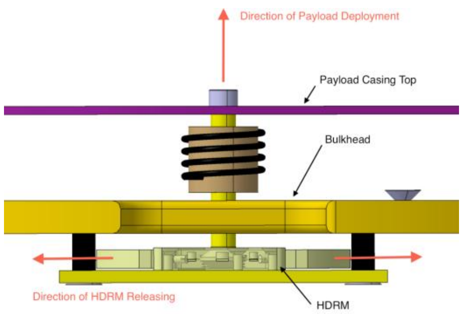

Figure 4. Arrangement of Release- and Push-Hardware

Due to the high uncertainty in the deployment behaviour of the raw composite structure, it was felt neccessary to integrate means of soft-latching the structure, once it has reached its full legth.. The solution was to catch the deployed structure by using magnets installed at the final position.
The magnetic force decreases with the square of the distance, so that with decreasing distance, the structure is pulled with higher force. At a distance of 50mm the magnets still exert a force of 0,09N, which is sufficient to catch the deploying structure even when tilting or not deploying entirely.
Of course such an installation is not desired to be needed in a mission application of such a rigidizable boom structure, but in this case, the time available for deployment would be much longer and can possibly be aided by gravity gradient, inflation or an additional servicing platform.

2.3. EXPERIMENT MODULE

The whole FOCUS module can be divided into two sections: the experiment section above the bulkhead and the electronics section below it. The experiment section consists of a large aluminium cylinder which houses the deployable structure as well as the UV LED panels, sensors and a video camera. The electronics section contains the battery, the HDRM assembly and the PCB stack.
The physical dimensions of the different elements, including the length of the deployed structure were tailored to fit into the standard REXUS payload module with a diameter of 346mm and a height of 300mm. From outside, the experiment could be accessed through a hatch in the module. *Fig. 5* and *6* illustrate the experiment design.

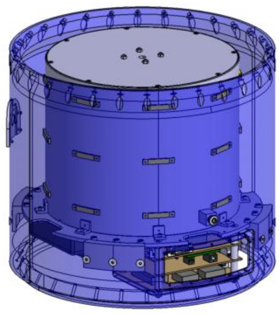

Figure 5. FOCUS Experiment with REXUS Module

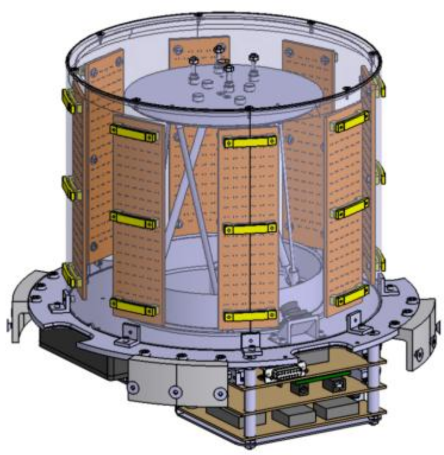

Figure 6. FOCUS Experiment without REXUS Module

2.4. ELECTRONICS & SENSORS

The experiment's on board computer was a Arduino Duemilanove microcontroller [9], which is a very low cost, yet powerful evaluation board for ATMEGA 8bit processors. The OBC communicates with the rocket telemetry to receive event signals and transmit sensor data to ground station. The microcontroller is mounted on a separate PCB, which also contains a SD-card storage device backing up the real time telemetry link. The PAL-formatted camera image is directly transmitted via a separate telemetry channel by the rocket service module.

Two additional PCBs are used to control and distribute the external power coming from the rocket service module (28-32V) and the internal experiment batteries (29.7V). This PCDU unit converts this voltage into 3.3V, 5V, 12V and 24V for the respective electric components of the experiment.

For safety purpose, the PCDU contains a safe and arm device, which can be accessed via a "remove before flight" switch. This functionality was required specifically for test countdowns which should not lead to a deployment of the composite structure but still checkout all other functionality of the experiment. Furthermore the dead payload functionality is maintained through a redundantly designed relay, which cuts the internal battery power from the experiment, if the external rocket power supply is switched off.

The experiment was equipped with four temperature sensors, one pressure sensor and one radiation sensor. The temperature was measured inside the experiment chamber and on the electronic components mounted on the printed circuit boards. Pressure and radiation were measured inside the experiment housing.

The sensors inside the experiment chamber were installed to evaluate the environmental condition during the curing process. Especially the temperature and ambient pressure were of interest, the latter for instance to indicate positive venting of the module.

3. FLIGHT RESULTS

3.1. ROCKET FLIGHT DATA

The REXUS 10 rocket was launched on Wednesday, February 23 2011 at 10:00:00 UTC from Esrange Space Center. During its 13 minutes parabolic flight, it reached an apogee altitude of 82445m at T+140s. The maximum vertical speed of 1131m/s was reached after motor burnout at T+25s at an altitude of 19027m. The maximum acceleration has not been measured, but it is assumed that it was near the predicted maximum of 17g at T+4s. The roll-, pitch- and yaw rates varied between 0Hz after parachute opening and 2.73Hz (roll), 1.19Hz (pitch) and 1.25Hz (yaw) at their maximum. During microgravity phase, all rotational rates were in the range of 3°/s (roll), 16°/s (pitch) and 15°/s (yaw). *Tab. 1* summarizes the most significant flight data.

Name	Unit	Value	Relative Time
Flight Duration	[s]	≈ 780	-
Altitude max	[m]	82445.23	T+140s
Vertical Speed max	[m/s]	1130.84	T+25s
Ground Speed max	[m/s]	170…180	T+20…230s
Acceleration max	[g]	17.00	T+4s
Roll Rate max	[Hz]	2.73	T+5…75s
Pitch Rate max	[Hz]	1.19	T+332s
Yaw Rate max	[Hz]	1.25	T+332s
Roll Rate in µg	[°/s]	≈ 3	T+80…210s
Pitch Rate in µg	[°/s]	≈ 16	T+80…210s
Yaw Rate in µg	[°/s]	≈ 15	T+80…210s

Table 1. Most Significant Flight Data

3.2. EXPERIMENT FLIGHT DATA

During flight the experiment data was stored onboard on a SD card and transmitted to ground station via telemetry link. The sensor data displayed in *Fig. 7* represents the environmental conditions inside the experiment chamber. In the first seconds after liftoff, the temperature dropped from 6°C down below 5°C, due to the airstream cooling the rocket and its interior. In higher altitudes the temperature started to increase again (no convective cooling) to a maximum of 25°C after completion of the experiment at atmospheric re-entry.

The pressure profile represents the flight profile in terms of altitude, as pressure is directly related to a certain barometric altitude. Thus the pressure falls after liftoff and rises again with re-entry. Due to the limited measurement range of the pressure sensor, values below 0.1bar cannot be displayed, which corresponds to an altitude of about 24km.

The measured radiation indicates the current mode of the experiment, as it distinguishes between radiation from white LEDs (before experiment is started) and UV LEDs (while experiment is active). In *Fig. 7* two significant jumps in the radiation profile can be recognized: the first before liftoff when the experiment is switched on (white LEDs on) and the second between T+90s and T-210s when the experiment is activated and the UV LEDs are switched on. The maximum radiation according to the sensor data is approximately 1.5mW/cm^2.

3.3. DEPLOYMENT AND RIGIDIZATION RESULTS

The deployment process of the composite structure worked without major problems.

The comparison between the late deployment tests under influence of gravity (using a compensation device) and the flight video showed, that the deployment time decreased from an average of 7 seconds to only about 2 seconds in microgravity. It could also be observed that the deployment process was more uniform than under gravity, where the structure had sometimes been deployed very quickly to about 90%, but then finished deploying very slowly.

This leads to the conclusion that the gravity compensation device introduced friction forces and a residual imbalance in the whole system, which were not present during the actual flight deployment. Therefore the force of the deployment spring could have been lower than it has been actually chosen.

The soft latching of the structure with the help of the magnetic catch-system worked flawlessly.

The overall alignment of the structure could not achieve the best results of ground-test units, but was clearly within the envelope of statistical manufacturing errors. The position accuracy of the upper cover plate after deployment was within 3mm of the theoretical position. This translates into an accuracy of about 2/100 for the whole structure. One of the beams showed a significant dent, which most probably was caused by a combination of manufacturing error and handling damage during integration.

The rigidization of the composite was fully successful, no weak regions could be found throughout the structure. No degradation of the protective PE membrane from vacuum, temperatures or launch loads could be found. Long-term evaluation however showed a slow loss of containment in all beams over a period of several weeks, indicating problems with the longevity of the implemented sealing concept.

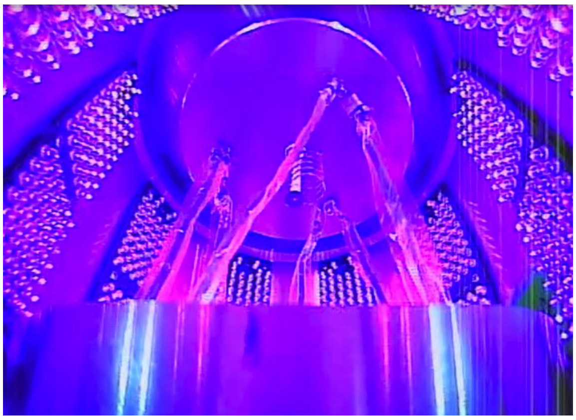

Figure 8. Deployed composite structure during flight

Figure 9. Deployed and rigidized Structure after Recovery

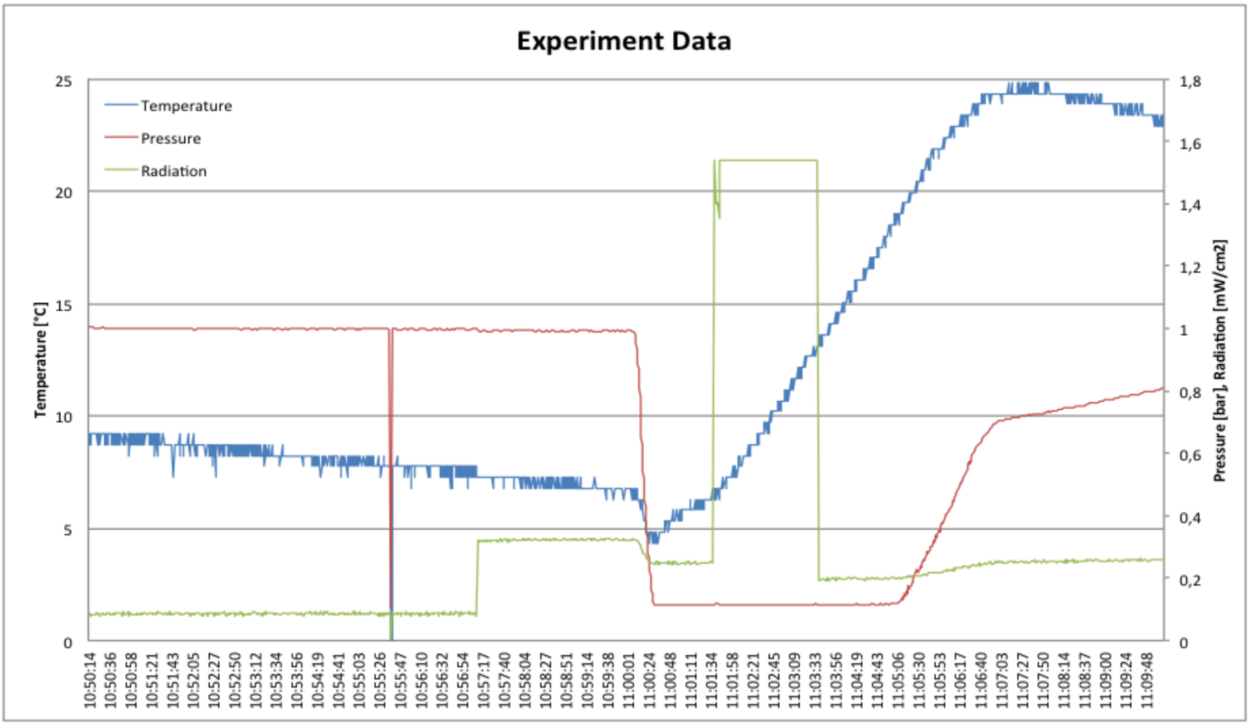

Figure 7. Temperature, Pressure and Radiation as measured by the experiment sensors

4. CONCLUSIONS AND OUTLOOK

With the success of the FOCUS experiment the feasibility of in-orbit rigidization using UV-light curing of fiber composites was demonstrated. It could be shown that this method for achieving a small stowage volume and low deployment complexity is even realizable without the massive use of high fidelity technologies or budgets. Furthermore, the relatively simple FOCUS design was able to withstand the rather severe launch environment of a sounding rocket without any noticeable damage.

Of course, the FOCUS composite structure is not optimized for any specific application, nor generally competitive with currently available coilable or collapsible masts. Many items for optimization and in-depth investigation have been willingly disregarded in order to develop a generally working concept during the short project time. However, a lot of hands-on experience could be gained as well as lessons learned from this pathfinder mission, so that the next steps of developing an application-ready technology can be taken.

The most important of these steps clearly is a reiteration on the materials selection including the resin. As the resin used on FOCUS was a modified adhesive and not a real structural resin, it does not possess all of the properties required for a high-end ultra-light structure. High curing shrinkage is one example. Promising technologies are cationic epoxies for instance, which couldn't be used on FOCUS because of their generally longer curing time.

Additional to the selection of a new resin, also the containment membrane material must be re-evaluated as it can be improved to lower modulus and thickness.

As a second important field of development, the overall structural concept must be optimized. Clearly one of the most promising fields of application are solar sails. As those require booms of giant length, and yet lowest mass, the new deployable composite structure to be developed should resemble such a slender boom design. Investigations in this matter have revealed the IsoGrid Boom concept, developed by Cadogan et al. for NASA sponsored research, in the field of deployable rigidizable structures. Such an Isogrid Boom is composed of several longerons and spiralling battens forming a sparse but very efficient structure. A parametric model of such a structure is shown in *Fig. 10*.

The last of the three main points of further development is the optimization of the manufacturing process. Up to now the manufacturing of the FOCUS deployable rigidizable beams was conducted by almost pure handcrafting, with the resin infusion being the only automated process step. The design of the structure itself of course also has implications on the manufacturing process. But with the Isogrid Boom design in mind, a manufacturing process can be conceived, that utilizes automated fiber deposition on a mandrell. As depicted in *Fig. 11*, in a first step a solvable membrane material is sprayed onto the mandrell to form the lower, respectively inner part of the containment membrane, then pre-impragnated UD-fibers are placed onto the mandrell according to the IsoGrid scheme. In a third step, the outer layer of the containment membrane is sprayed on top of the fibers, and finally the mandrell can be removed. The so constructed raw boom can readily be deployed via inflation. If alternate deployment methods are desired, the higher mass of the completely closed containment membrane can be improved by laser cutting of the triangles between the fiber network. The entire manufacturing process can be highly automated and does not yet require any special equipment, which isn't already utilized as part of today's composite manufacturing technologies.

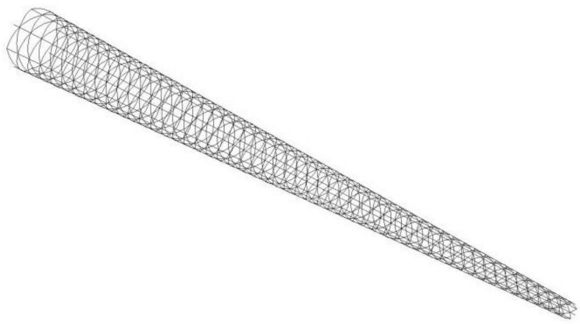

Figure 10. IsoGrid Boom Structure

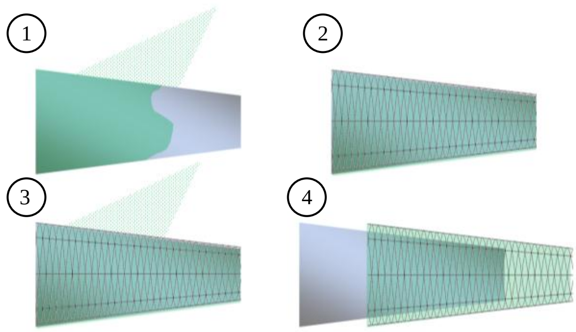

Figure 11. Manufacturing concept utilising mandrells and sprayable membrane technology

5. ACKNOWLEDGEMENTS

The authors would like to thank all the institutions and companies that supported the FOCUS project, specifically:

- ABUS SECURITY
- CYBERBOND

- DLR
- DLR-MORABA
- ESA
- HEINRICH UND LOTTE MÜHLFENZL STIFTUNG
- LEITON
- LUMITRONIX
- SNSB
- SSC
- TUM/LRT

6. REFERENCES

[1] Rackl, W.: *Multi-Purpose Deployment Mechanism for the MOVE Satellite*, Semesterthesis, RT-SA 2009/01, Technische Universität München, January 2009

[2] Jenkins, *Gossamer Spacecraft: Membrane and Inflatable Structures Technology for Space Applications*, Progress in Astronautics and Aeronautics Vol. 191, AIAA, 2002

[3] Jenkins, *Recent Advanced in Gossamer Spacecraft* Progress in Astronautics and Aeronautics Vol. 212, AIAA, 2006

[4] John M. Hedgepeth, *Influence of fabrication tolerances on the surface accuracy of large antenna structures*, AIAA Journal Vol.20, pp. 680-686, 1982

[5] Cadogan, D. P., Lin, J. K-, Saapna, G. H., and Scarborough, S. E., *Space Inflatable Technology Development for Solar Sails and Other Gossamer Applications: GR/SMP Isogrid Boom Development Final Report* NASA Task Order 10442, ILC Dover,Inc., October 2001

[6] http://www.cyberbond.de

[7] Freeland, R. E., Bilyeu, G. D. Veal, G. R., and Mikulas, M. M., *Inflatable Deployable Space Structures Summary*, 49th Congress of the International Astronautical Federation, IAF-98-1.5.01. Sept. 1998

[9] http://www.arduino.cc