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Testing combined cryogenic insulation and thermal protection systems for reusable stages

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Abstract

Reusable launchers experience a large range of thermal extremes, reaching from the cryogenic temperatures within the propellant tanks to the high external heat flux encountered during reentry. During all operational phases (filling, hold, ascent, ballistic, reentry) every component of the launcher has to be kept within the allowed temperature domains to assure reusability of the system. For this purpose, not only is cryogenic insulation needed but also an external thermal protection system. While usually treated as separate domains, for winged reusable launch vehicles cryogenic insulation and the thermal protection system have to be designed together since they are both integrated onto the propellant tanks external surface. Additional complexity arises from the fact that multiple cycles of thermal and mechanical loads have to be survived without substantial refurbishment in order to assure a cost-effective design.

Within the DLR this topic was first explored in the AKIRA project, which investigated selected technologies that were deemed critical for operational reusable booster stages but which are not being covered within the flight demonstrator projects of the DLR. An overview over the AKIRA experimental campaign with representative test objects, which verified the thermal functionality of the derived design, is given. Within a currently ongoing follow-on project (TRANSIENT) the design is further refined and applied to both metallic and composite tank materials. For both cases segments meant to represent a slice of an RLV propellant tank will be manufactured and equipped with the cryogenic insulation and thermal protection systems and will be tested under representative loads for up to 50 cycles. In addition to the thermal loads investigated previously, mechanical loads will also be applied to the test object.

Keywords: RLV, TPS, Cryogenic insulation, cryogenic propellant, Test

Acronyms/Abbreviations

AOA	Angle of Attack
CAD	Computer Aided Design
CMC	Ceramic Matrix Composite
DIC	Digital Image Correlation
ELV	Expendable Launch Vehicle
GLOW	Gross Lift-Off Mass
ITO	Integrated Test Object
LFBB	Liquid Fly-Back Booster
LH2	Liquid Hydrogen
LN2	Liquid Nitrogen
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
RLV	Reusable Launch Vehicle
RP-1	Rocket Propellant (Kerosene)
SLB	SpaceLiner Booster
Ti	Titanium
TPS	Thermal Protection System
TRL	Technology Readiness Level
TSTO	Two-Stage-To-Orbit

1. Introduction

Reusable launch vehicles experience a large range of thermal loads during their mission, ranging from the temperatures of the propellant over the heat loads encountered during reentry to the vacuum of space during ballistic phases. This range of temperatures is extended substantially when using cryogenic propellants, especially hydrogen.

The number of operational reusable launch vehicles (RLV) in the history of spaceflight is limited. None of them have made use of a cryogenic tank insulation on their reusable stages. The Space Shuttle and the Soviet Buran were orbital stages without any large cryogenic tanks. The Falcon 9 is a booster stage with integral tanks but without cryogenic insulation. Therefore, no practical experience with an operational RLV with reusable cryogenic tank insulation exists. For future RLV's however, the use of cryogenic fuels has large benefits with regard to vehicle mass, size and environmental impact [3]. For a winged system the cryogenic insulation has to be integrated with the thermal protection system on the propellant tank, imposing new design requirements on both.

Within this paper two DLR projects are presented in which this issue has been investigated. First the AKIRA project is discussed, which has been presented at the IAC previously [1]. However, the experiments had not been completed at that point, so no final conclusions could be included. The experiments and their conclusions are shown in the first half of this paper. The second half concerns the TRANSIENT project in which the work from AKIRA is continued and expanded.

1.1. Technical challenge

The understanding of the behavior of reusable tank insulation is of crucial importance for an RLV with cryogenic propellants. While the spacecraft is fueled, its outer tank shell is cooled down to very low temperatures, when not insulated properly. Not only is the propellant loss reduced by tank insulation, but there are safety issues as well. If icing occurs at the outside of the launcher it can cause serious damage to the vehicle structure or the thermal protection system (TPS).

As a winged RLV is subject to elevated temperatures during re-entry, the tank insulation becomes a complex system considering the high temperature gradients between TPS and cold propellant tank wall. Typical cryogenic insulations are usually limited to around 100°C to less than 200°C maximum temperature. On the other hand, high-temperature insulations of the TPS should be kept above the dew point of air during ground operations to prevent internal moisture condensation or even icing.

This thermal coupling as well as the joint integration onto a vehicle propellant tank surface means that these two previously separate subsystems have to be designed and tested together. Further complexity is introduced by the fact that these systems are expected to have a major impact on the refurbishment cost of the vehicle and thus a robust and low maintenance design is highly desirable.

2. DLR project AKIRA

Within the DLR project AKIRA[1] technologies deemed critical for reusable booster stages but not covered by current flight demonstrator projects were investigated. One of the technologies was reusable cryogenic insulation and its integration with the TPS. The focus was to develop a design and test its thermal performance with differently sized Integrated Test Objects (ITO) in the test facilities available within the DLR.

During the design evaluation phase, it was quickly found that the lowest allowable temperature for the TPS on the launch pad would be a significant design boundary. Initially, the goal was to achieve a temperature above 0°C in the TPS in order to avoid ice formation in the polycrystaline wool chosen as high-temperature thermal insulation. While it is not expected that the ice will directly cause damage, it is unknown how it will affect the performance of the TPS during re-entry, where it will quickly melt and evaporate and fill the segment with hot steam. Additionally, when considered for the entire surface of the vehicle it could lead to a noticeable increase in mass.

Ice on the outer surface of the TPS is also undesirable, as, depending on its location, it can break off during ascent and potentially damage the thermal protection systems of lower protruding elements, such as wings.

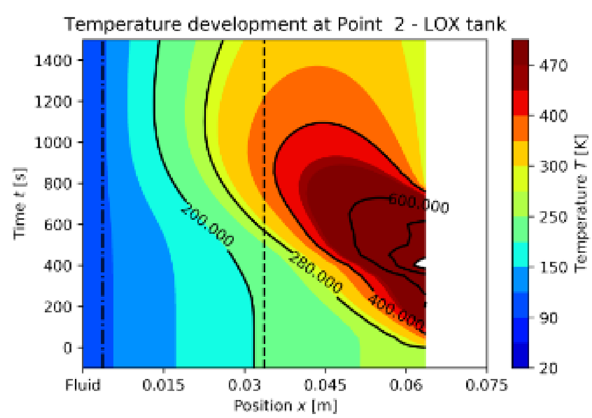

Figure 1: Exemplary evolution of temperature over time for an architecture consisting of an external insulation of PU-foam (30 mm) and a TPS layer of 30 mm

Avoiding sub-zero temperatures at the outer surface of the cryogenic insulation requires thicker insulation than for expendable systems, because the thermal protection layers effectively insulate the interface from “warm” external conditions that would usually heat up the external side of the cryogenic insulation.

First, 1-D heat conduction tools were used to efficiently assess different layer architectures. An exemplary result is shown in Figure 1. It was quickly found that for low external temperatures expected at European launch sites unfeasibly thick cryogenic insulation would be needed to keep the external thermal protection above the freezing point. In order to avoid thick and heavy insulation layers and still assure an ice-free system, different options of heating the TPS were evaluated. Finally, it was decided to include a purge gap into the design.

Through the purge gap, an inert gas (e.g. nitrogen or dry air) could be used to either heat the interface sufficiently to prevent any ice formation or simply displace any moist air from the system and thus prevent ice formation.

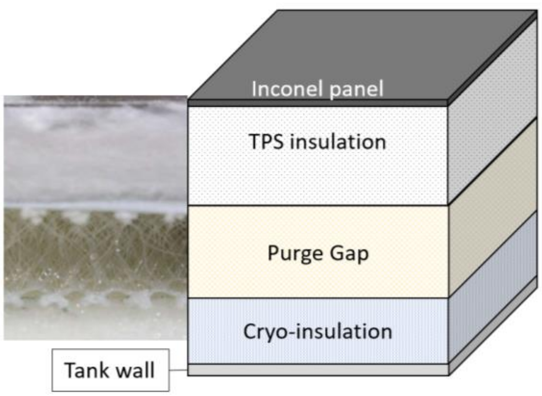

Figure 2: Layer architecture chosen for ITO

The main elements of the whole system were the cryogenic insulation directly attached to the tank wall (aluminium plate), a flexible spacer-mesh fabric on top of the cryogenic insulation, which provides for the purge gap, and the high-temperature insulation on top of the mesh fabric. The surface of the test articles consisted of a metallic Inconel panel. The surface panel was connected to the tank wall via a structural standoff connection which also served as the fixation for the purge gas supply tube. A more detailed description of the design process can be found in [2], a sketch of such a standoff can be seen in Figure 4.

In order to validate the thermal performance of this design, representative segments were prepared as ITO’s and tested under the thermal conditions expected for the different phases of the launch trajectory. In total, three ITOs were prepared for three different experimental facilities: A climate chamber to simulate various launch pad conditions, an arc-heated wind tunnel as well as the thermo-mechanical test facility INDUTHERM to apply reentry heat loads. Multiple mission’s thermal loads were applied to each test object in order to verify their thermal functionality and identify potential changes over the number of cycles.

2.1. Tests under launch pad conditions

The tests representing various launch pad conditions were performed with a climate chamber at the cryogenic laboratory at DLR Bremen. The core goal was validating the performance of the purge gap system and thus showing the ability to control the system at the interface and keep the high-temperature insulation ice free.

The ITO used in these tests was the largest, representing an 800x800mm slice of the tank structure with the different layers shown in Figure 2: Cryogenic insulation, purge gap, high-temperature insulation and finally a surface panel. On one side a container filled with liquid nitrogen provided the cryogenic temperature boundary condition while a climate chamber was used to achieve the desired “ambient” conditions. The experimental apparatus and its integration have been described in more detail in [1]. In order to account for multiple possible launch sites, different environmental profiles were applied to the test chamber that represented lab conditions, typical conditions at Kourou and Kiruna as well as an “ice condition” that was used to provoke ice formation.

Table 1: Ambient conditions used for launch-pad condition tests

	Lab	Kourou	Kiruna	Icing test
Ambient temperature [°C]	20	30	0	0
Rel. humidity (target) [%]	35	90	80	80
Rel. humidity (actual) [%]	35	52-78	17-50	63-65

Different purge gas mass flows and temperatures were applied in order to assess their effect on the thermal state. During the experimental campaign it became apparent that the nitrogen introduced via the purge gap quickly lowered the humidity in the entire chamber. This could not be entirely compensated by the climate control system. Thus, the humidity actually achieved was not always the target value. The actual and target values for the different scenarios are shown in Table 1. However, since the external temperature was not affected by this, the experiments can still show that the purge gap can control the interface temperature sufficiently to preclude any ice formation even at higher humidity values.

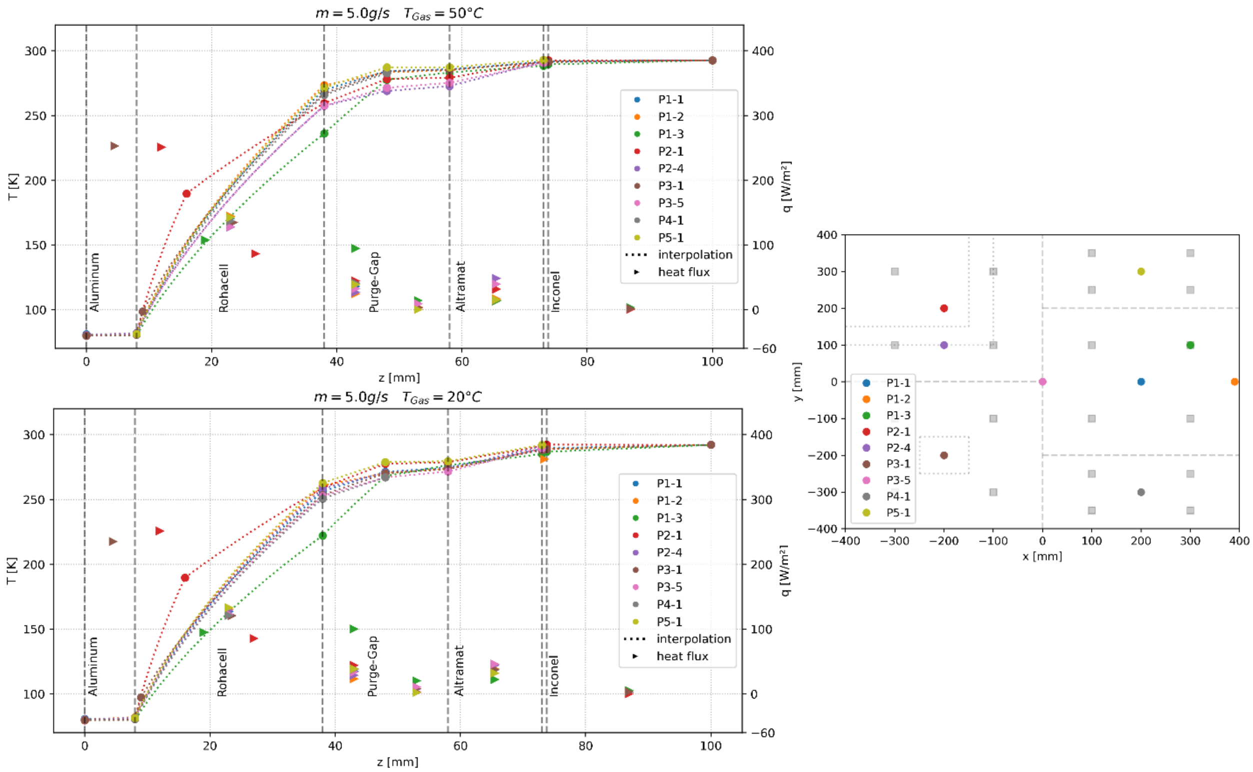

Figure 3: Selected temperature measurements and their positions within the ITO for launch pad condition test for two different purge gas temperatures

Figure 3 shows the temperature profiles of selected measurement points for different purge gas temperatures. It can be clearly seen that the standoff (P1-3) has much lower temperatures than the surrounding insulation and thus functions as a heat bridge. However, the effect of the purge gas streaming through the standoff is also visible: above the purge gap the temperature of the standoff is similar to the surrounding thermal protection layers.

With the gas temperatures and mass flows shown above, the temperature within the thermal insulation can still be slightly below 0°C but the dry gas displaces any humidity and thus no ice formation was found. For reference, the mass flow of 5 g/s is sufficient to cycle the gas within the system every ~6 s. It seems likely that lower mass flow rates would also be sufficient to prevent icing even if the temperatures in certain areas (especially around the standoffs) drop below the freezing point.

At the locations of the purposefully introduced damages of the cryogenic insulation, deviations in the temperature profile could be measured. However, this necessitates a thermocouple fairly close to the damage site. This is an unrealistic scenario for a full-sized vehicle with large surface areas, where the large number of sensors and cables would add substantial complexity and mass.

In conclusion, it was shown, that the ITO design performed as expected with regard to its thermal characteristics at launch pad conditions and that the purge gas system was successfully able to prevent conditions favorable to ice formation.

2.2. Test under re-entry thermal loads in INDUTHERM

During the re-entry leg of the flight, the vehicle's outer surface is exposed to high thermal loads caused by aerodynamic heating. Obviously, this leads to high thermal gradients between the cryogenic propellant tank and the hot surface panel which must be tolerated for multiple missions. Hence, it was intended to test the entire insulation system also under the highly-transient thermal re-entry conditions in the thermo-mechanical test facility INDUTHERM of DLR. For this purpose, an additional ITO was built to prove both the functionality and the reusability of the chosen insulation design concept including the purge gap system. Every test included both the steady-state cold soak condition representative of the pre-launch phase including the purging, as well as the subsequent transient high-temperature re-entry conditions.

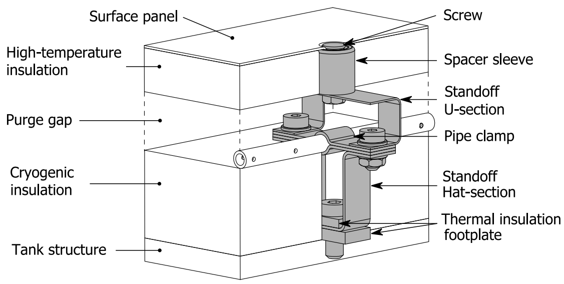

Figure 4: Section of the INDUTHERM-ITO with standoff and purge pipe.

The INDUTHERM-ITO was a representative section of the above described architecture of 400x400mm size including a single centered standoff, shown schematically in Figure 4. In the test chamber, the ITO was placed with the aluminum plate facing up, as can be seen in Figure 5. Insulation plates placed between ITO and chamber walls formed an enclosed basin into which liquid nitrogen was filled as propellant substitute. Pre-heated, gaseous nitrogen was used for purging during the steady-state case. The gas pipe was guided through a temperature-regulated water bath, and then entered the ITO where the gas escaped the pipe through small holes into the purge gap. For the transient case, an inductively heated graphite coupling plate located underneath the surface panel provided the re-entry heat load.

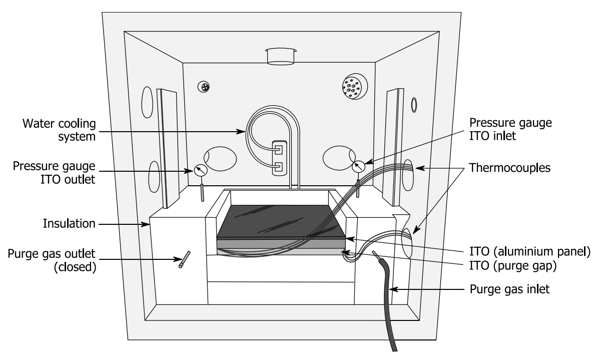

Figure 5: Sketch of the experimental setup for the steady-state case under ambient conditions with active purging.

The interface temperatures and the temperatures at the standoff components were monitored using sheathed type K thermocouples of 0.5mm diameter. Additionally, the fluid temperature inside the purge gap was measured in-plane by 4 thermocouples each at the interfaces of the high temperature insulation and the purge gap (P-Altra-x) and the cryogenic insulation and the purge gap (P-Cryo-x) to assess the thermal area of influence of the purge gas escaping the pipe. The positions of the measuring points are depicted in Figure 6.

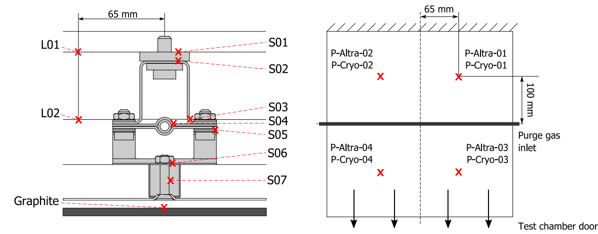

Figure 6: Thermocouple instrumentation on the standoff (left) and on the purge gap-insulation interfaces (right).

With the described experimental setup, the ITO was first subjected to the steady-state cooling under ambient conditions to simulate the pre-launch tank filling phase. The test chamber remained open, while the purge flow was active and the basin was constantly refilled with LN2 until all signals of the temperature inside the ITO reached steady-state with no significant changes. A total of 18 test runs were carried out to assess a mass flow and gas temperature combination reliably preventing negative temperatures inside the high-temperature insulation.

After the steady-state cold condition test runs were successfully completed, the transient re-entry case was tested under vacuum conditions. In order to do so, the steps described for the steady-state cooling tests were repeated with a proven flow parameter combination, until the steady-state condition was reached. After that, the purge gas was turned off, the test chamber was closed and the chamber pressure was reduced to vacuum conditions. Almost simultaneously, the inductive heating device was turned on for heating up the graphite plate which delivered in turn the heat load to the ITO surface panel. Five transient test runs were carried out with selected initial steady-state conditions.

The requirement was not to exceed 100°C on the cryogenic insulation, hence making the sensors located on the interface cryogenic insulation-purge gap the critical measuring points. Here, the highest temperature was measured at L02 with 101°C. However, the integral heat load acting on the ITO was significantly larger than the heat load which is assumed for the reference booster, due to the lack of convective cooling within the vacuum chamber. The requirement therefore is considered fulfilled within the context of the measurement accuracy. Figure 7 shows selected transient temperature profiles for one representative test run, including the corresponding temperature profile of the heating plate. The graph starts right before the heating phase, hence the temperatures of the previous LN2-cooling phase are not shown.

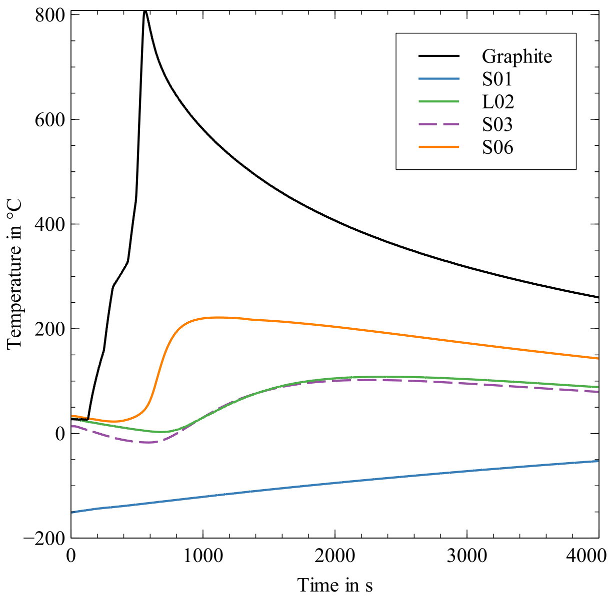

Figure 7: Selected temperature profiles for a transient test run.

2.3. *Tests in arc-heated wind tunnel*

The third ITO was tested in the arc-heated wind tunnel L2K at DLR Cologne and because of the constraints imposed by the experimental facility was the smallest, with a surface area of 400x200mm, thus representing half a panel. The design of this ITO is also shown in more detail in [1].

While initial test runs were successful, two issues were encountered during the tests: First, the adhesion of the cryogenic insulation to the tank wall failed which was caused by a manufacturing default of the tank. A new tank was fabricated and successfully integrated. Then, in the following experimental runs under reentry heat loads another issue appeared: Under certain conditions, the liquid nitrogen interfered with the water cooling of the support structure. Without effective cooling, the support structure was subsequently damaged by the heat loads experienced within the wind tunnel. The lessons learned were incorporated in a follow-on test campaign in the TRANSIENT project, depicted in section 3.3.1.2.

2.4. *Conclusions from AKIRA*

Overall it was found that the combined cryogenic insulation and external thermal protection system behaved as expected and predicted with the numerical simulation for both launch pad and re-entry thermal environments. The next step, evaluating the design under combined thermal and mechanical loads is being assessed in the currently active DLR project TRANSIENT.

3. DLR project TRANSIENT

The DLR project TRANSIENT (**T**he**r**m**a**lko**n**troll**s**ystem für w**i**ederverw**en**dbare **T**räger) was initiated in the beginning of 2020 and will last until the end of 2022. The main goal is the further advancement of the technologies necessary to integrate a reusable cryogenic insulation with external thermal protection systems through thermo-mechanical experiments with representative test objects. Building on the work done in AKIRA, the designs are further refined and applied to both metallic and composite base structures. Within the experimental campaigns both versions will be put through up to 50 cycles of thermal and mechanical loads representative of a first stage RLV mission.

3.1. *System aspects*

In support of the experimental campaigns at the core of the project, system aspects are also being evaluated in order to assure a better understanding of the full-sized vehicle and provide representative boundary conditions for the sizing of the relevant subsystems.

3.1.1. *Reference configurations*

In order to assure that the boundary conditions of the following development efforts are applicable for future reusable first stages, reference configurations are needed. For this purpose, two consolidated DLR-studies were chosen, the SpaceLiner 7 Booster [5] and the ENTRAIN 2 HL [4] stage. The main requirement was the existence of an aerothermal database for the descent phase in order to accurately assess the heat loads. Both stages are also deemed sufficiently representative for future winged reusable stages. Their separation Mach numbers are 9 and 12 and thus cover the expected range for future stages. Both stages use hydrogen as fuel. While DLR system studies[3] indicate that hydrogen is the most attractive option for a European reusable first stage, the lessons learned remain valid for other fuels such as methane, since its storing temperature is similar to oxygen and thus the design for the oxygen tank can be applied to possible methane-fueled vehicles.

In addition to the thermal loads, mechanical loads for the reference points had to be generated. These points are situated at the front and back of each tank, with an additional reference point being placed at the middle of the large hydrogen tank (5 points in total) where bending moments are expected to be highest. In contrast to standard structural launcher design, the loads are not only analyzed at the critical trajectory points but instead over the whole flight. The structural analysis is performed with an in-house tool which simplifies the rocket to a bending beam which is why the results are separated into ‘tension side’ and ‘compression side’. While this methodology does not deliver the degree of detail of for example a FEM-tool, it enables the efficient generation of hundreds of computations along a trajectory and delivers useful upper and lower boundaries for the expected mechanical loads.

The diagram in Figure 8 shows the Von-Mises stress on both sides of the simulated bending beam along the ascent and descent trajectory of the SpaceLiner Booster.

In general, the largest stress is the circumferential stress created by the internal tank pressure which is present throughout the whole flight.
During the ascent phase, this is overlaid with other stresses. First of all, the propulsion system creates a significant force in axial direction. Secondly, profound bending moments are apparent during the phases with high dynamic pressures. Additionally, wind and gust loads lead to high effective angle of attacks, in combination with high dynamic pressures resulting in even higher bending moments.
Because of a low re-entry flight-path-angle of the SpaceLiner booster, the aerodynamic loads in the descent phase do not have a large impact on the overall stress experienced by the vehicle. The maximum dynamic pressure is also significantly lower than during ascent (40 kPa vs 5 kPa).

The compressed side of the stage experiences higher stress levels because the thrust from the propulsion system leads to a universal compression of the rocket structure which is why the stage is not stressed symmetrically.
Overall, these loads are deemed to be representative for the experimental campaigns. For this early development stage, without a specific vehicle development driving the design, it is more critical to understand the general order of magnitude and distribution of the loads than to get exact values. For instance, the maximum stress shown in Figure 8 results from the high strength assumed for the aluminium alloy at cryogenic temperatures [6]. If a different alloy is used, the structure will be sized differently and exhibit different stresses.

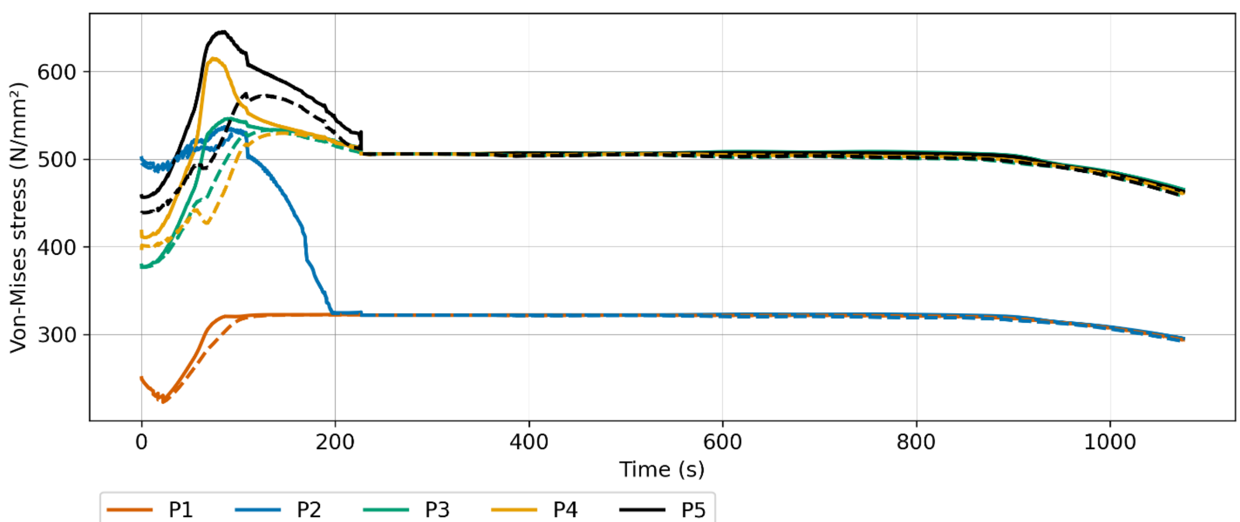

Figure 8: Von-Mises stress of the SpaceLiner booster on tension (dashed) and compression side (solid) at points P1 (front of oxygen tank) to P5 (back of hydrogen tank)

3.1.2. Improvement of numerical tools to analyse entire vehicles

While sophisticated tools and methods exist to analyse segments of future reusable stages, applying them to entire vehicles quickly leads to unfeasible computational requirements, especially if different design variations shall be evaluated or optimized. In addition to the complexity of the involved physical phenomena, this is caused by the sheer size of the vehicle as well as the long simulation time to be covered in order to account for an entire mission.

In order to gain a better understanding of the full system and the interaction of its subsystems with each other, numerical tools are being developed with the aim of assessing the thermal behaviour of the full vehicle throughout the entire trajectory while still keeping the computation effort in a magnitude that allows parametric studies and optimization at an early design stage.

3.1.2.1. CFD

Reusability introduces new challenges in the propellant management. For ELV the mission ends with the last engine cut-off, for RLV-stages the propellant and the tank pressures have to be controlled until the stage is successfully landed and all propellants can be safely vented without compromising the structural integrity. In the case of the Vertical Take-off Vertical Landing (VTVL), the spacecraft needs to perform a 180-degree rotation with a subsequent burn to reduce the loads for the ascent phase. For both RLV types it is essential to accurately predict the pressurant gas mass requirements with CFD solvers.

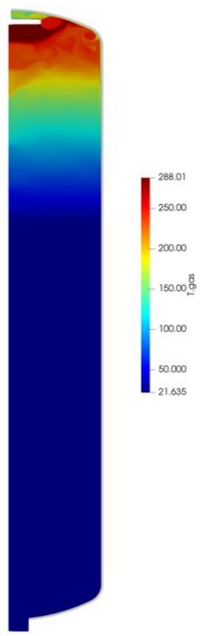

Figure 9: Thermal stratification during propellant pressurization

The inclusion of CFD solvers in early design stages requires fast, cost-effective and easy-to-use tools. Meshless methods such as the Smooth Particle Methods and scaled 2D-CFD sloshing simulations are investigated in order to evaluate if they fulfil the above requirements to deliver sufficient accuracy for the first trade-offs. Furthermore, heat and mass transfer models of CFD solvers are improved for stability, accuracy and for the use of unstructured grids. This approach significantly reduces the time-spent by costly engineers as the geometry can meshed automatically with minimal input by the engineers.

3.1.2.2. FEM

In order to correctly assess the thermal state of a reusable launch vehicle, a variety of physical phenomena have to be considered:

- The external thermal loads, ranging from ambient conditions on the launch pad to the aerothermodynamic loads encountered during ascent, reentry and recovery
- The heat transfer within the vehicle subsystems, mostly by heat conduction
- The heat transfer into the cryogenic propellants.

While these phenomena are usually treated as separate problems with their own engineering tools, they have to be considered together in order to analyze and ultimately size a combination of cryogenic insulation and thermal protection system. In order to achieve this for a full vehicle surface, a tool suite is being created that allows the coupling of aerothermodynamic databases and propellant management software with an FEM Solver in order to size the various layers of cryogenic insulation and the TPS. A critical goal is the ability to quickly size the system (ideally in less than 30 minutes wall time) in order to be able to incorporate these advanced methods in pre-design studies and parametric variations.

3.2. Integration of reusable cryogenic insulation with external TPS

The core of the TRANSIENT project is the further development of the reusable cryogenic insulation integrated jointly with an external TPS onto a propellant tank structure. In addition to the metallic base structure considered in AKIRA, composite base structures are also investigated. For both base structures a system is designed and will be verified in an experimental campaign under thermal and mechanical loads. The expected loads are taken from the system level work discussed in section 3.1, and the test objects are manufactured in a size that allows these loads to be recreated in the available test facilities.

3.2.1. Aluminium base structure

Starting from the design derived in AKIRA, which included a metallic base structure, a refined design was created. Some elements were simplified in order to ease production and integration of the test objects. These refinements would also positively benefit the cost of a full-scale vehicle. One simplification is the omission of a dedicated purge gap. Instead, if needed, a purge gas can be introduced through channels milled into the cryogenic insulation in order to prevent icing in the high temperature insulation. This allows a significant simplification of the stand-off design and its manufacture. The overall architecture and material layer choice however will remain similar to the architecture derived in AKIRA.

3.2.2. Carbon fibre base structure

A large fraction of a launcher's dry mass is composed by the propellant tank structure. Due to their low specific weight combined with high strength and stiffness, the use of carbon fibre composite materials as tank structure material promises a mass reduction by up to 30%, compared to conventional tank designs made of aluminium-lithium alloy [7]. However, demanding requirements need to be satisfied for the application of reusable cryogenic fuel tanks, which so far have precluded commercial use. On the one hand, the manufacturing of large composite components is still a challenge, and on the other hand, the anisotropic material properties call for sophisticated component designs. As one of the key issues, transverse microcracks occur in the resin as a consequence of the large temperature differences, which is due to the difference between the axial and the transverse coefficients of thermal expansion.

These microcracks eventually lead to permeation leak paths and reduced mechanical strength.

Aiming to increase the TRL of composite cryogenic tanks, NASA and Boeing started the Composite Cryotank Technologies and Demonstration project (CCTD, 2011-2013), building on the experience of over 20 years of composite cryogenic tank development. This project involved the development of suitable material systems, corresponding structural concepts and associated manufacturing techniques. A tank with a diameter of 2.4m was built and cyclically tested successfully with LH2, reaching pressures of up to 9.3bar. [8]

To further advance the application of carbon composite materials for RLV, the integration of a thermal protection system onto a composite cryogenic tank structure was selected to be assessed in detail within TRANSIENT. Starting from the identification of a suitable carbon composite system for large cryogenic structures, the work includes the simulation-aided design of an ITO comprising of the composite structure, a cryogenic insulation, a thermal protection system and a fastening system. The same reference configuration and design-defining thermal requirements were applied as in AKIRA. Finally, the composite ITO will be subjected to thermo-mechanical tests in order to verify the insulation system.

Based on the achievements of the CCTD-project, a combination of conventional carbon-epoxy prepreg plies and carbon thin plies were chosen as material system for the tank structure. The selected prepreg system CYCOM®5320-1 is designed for out of autoclave manufacturing and thus is particularly suitable for large structures [9]. Micro-crack resistant thin plies of IM7-fibre tapes (TeXtreme®) were inserted in the center of the laminate to minimise permeation. Test plates of different numbers of plies were produced for material characterisation. The laminate quality was verified via scanning electron microscopy, which proved a homogeneous quality with very low porosity.

The different laminates are furthermore tested for their gas permeation and leakage rate uncycled and thermally cycled with LN2. To do so, a sample plate of 125x125mm in size is placed into a sample holder. As schematically shown in Figure 10, one side of the plate is pressurised with a constant predefined helium test pressure. On the other side, the sample holder is connected to a helium leak detector which creates a vacuum in the system. If helium passes through the composite sample, it diffuses into the mass spectrometer and the leak detector outputs the leakage rate. First comparative tests proved a superior permeation property of cycled laminates containing a small number of thin plies compared to cycled conventional laminates without thin plies. These tests are still ongoing and the final evaluation will be presented after the test campaign is completed.

The detailed structural design will be supported by thermal simulations. Following the example of the CCTD structural concept, a double-shell construction with internal longitudinal stiffening is envisaged as tank structure. Based on the AKIRA results, the potential benefits and the feasibility of a purged fluted core will be assessed in detail.

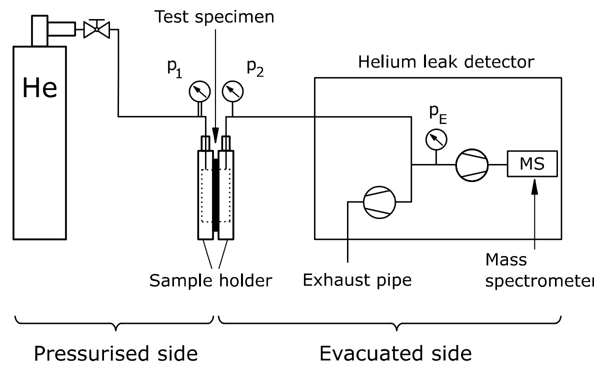

Figure 10: Scheme of the permeation test rig.

3.3. *Outlook on thermo-mechanical experiments*

The updated designs for both possible base structures will be integrated into ITOs meant to represent slices of an RLV propellant tank. In total, three test campaigns with three different ITO's are planned for 2022.

3.3.1. Thermo-mechanical test facility THERMEX

The THERMEX facility at DLR Brunswick allows the investigation of coupled thermal and mechanical loading on test objects with up to 190 kW of heat and 400 kN of force. It has been used in the past for similar investigations[17]. Test objects with sizes up to 800x800mm can be accommodated within the facility. In Figure 11 the facility is shown including its infrared heating elements.

While the facility can be used to test the structural performance of the base structure itself, here the focus is on the cryogenic insulation and thermal protection system mounted onto the base structure and their integration. Thus, the main goal of introducing the mechanical loads is to achieve comparable displacements of the base structure to the displacements expected during the mission of a reusable first stage.

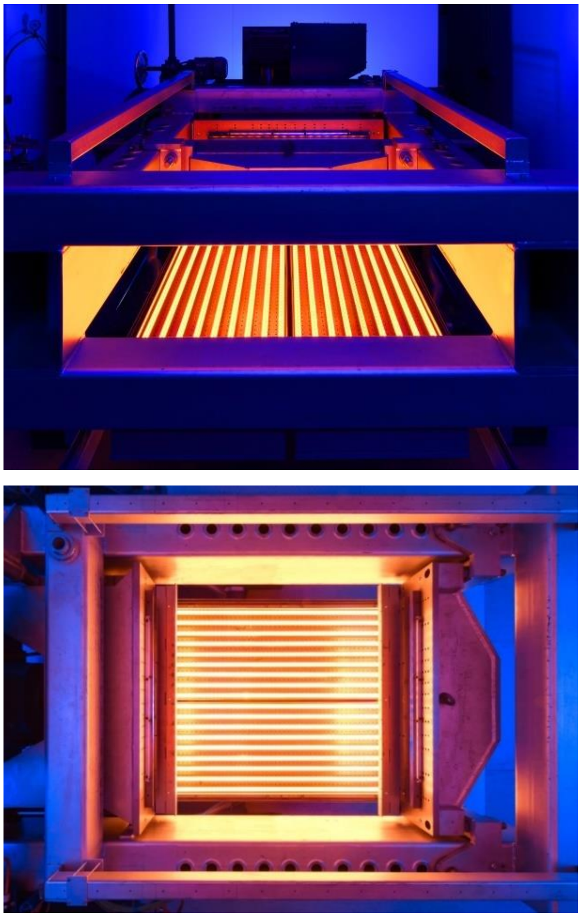

Figure 11: Front and top view of thermo-mechanical test facility THERMEX

3.3.1.1. Planned tests

The two ITOs, one with metallic base structure and one with carbon fibre composite material base structure will be integrated into the experimental facility and subjected to representative thermal and mechanical loads.

A container filled with LN2 will be used to impose cryogenic temperatures onto the base structure representing the tank wall. After allowing the initial cooling of the cryogenic insulation, the thermal and mechanical loads are introduced by the facility and the thermal response of the ITO is carefully monitored. While damage in the outer layers can be assessed by inspection, cracks within the cryogenic insulation might not be immediately visible. However, by monitoring the change in thermal response over the large number of cycles, large damages can be identified. Damage within the cryogenic insulation that, even after many load cycles, does not have a noticeable impact on the thermal function of the ITO, will probably also be acceptable for a full-scale vehicle.

The goal is to apply up to 50 representative load cycles. The functionality of the system and possible degradations in performance will be monitored in order to ensure the derived designs will retain their functionality throughout multiple missions.

3.3.1.2. Arc-Heated Wind Tunnel L3K

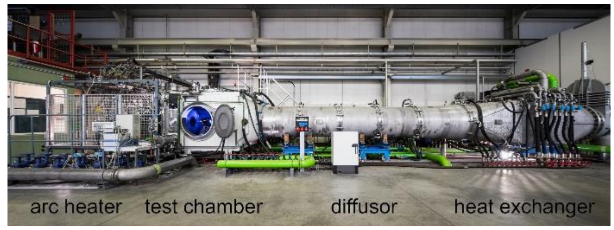

Figure 12: Arc-Heated Wind Tunnel L3K [11]

A smaller ITO with a metallic base structure will be tested in the arc-heated wind tunnel L3K at DLR Cologne (Figure 12, see also [10] for a full description of the facility). In this facility, air is heated to total temperatures between 4000 and 7000 K by an arc-heater, and then expanded and accelerated in a conical nozzle. A free jet forms in the test chamber that is evacuated before the wind tunnel run. The model is moved into the free jet after steady flow conditions are established. During the wind tunnel run, optical measurements of the surface temperature by infrared camera and pyrometers as well as surface deformation by a digital image correlation (DIC) system will be conducted in addition to temperature measurements inside the wind tunnel model (see [11] for an example of the DIC system at the L3K facility).

3.3.1.3. Planned tests

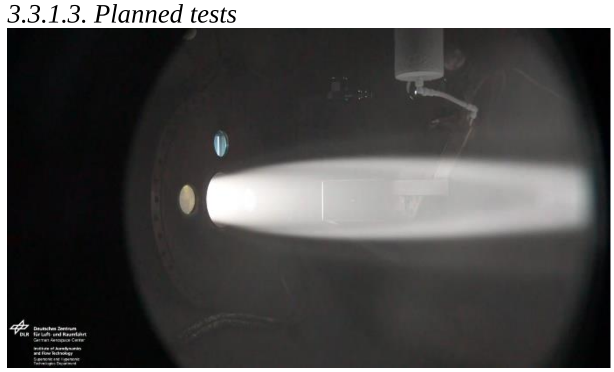

Figure 13: Wind tunnel model during preparatory test run in the arc-heated wind tunnel L2K

In the planned tests, an ITO with the same thermal control system design as in the THERMEX facility will be used including an internal tank filled with liquid nitrogen. However, the wind tunnel model will be adapted in size to the requirements of the L3K facility. Figure 13 shows a wind tunnel run conducted in preparation of the final model design.

The main objectives of the wind tunnel experiments are:

- Investigation of the thermal control system performance under repeated aerothermal load cycles
- Investigation of the effect of icing (as it may occur on the launch vehicle prior to launch) on the behaviour of the thermal control system
- Observation of potential deformations of the metallic TPS surface

The latter point is relevant as it has been shown in various studies on generic configurations that structural deformations can lead to significant increases in local heat flux and temperature [11][12][13][14]. Regarding actual vehicle structures, such behaviour was for example considered regarding X-33 TPS panels [15] or observed on the SR-71 [16]. In the present case, it would thus both be useful to show the absence of such effects or to observe to what extend they occur.

4. Conclusion

The DLR continues to investigate the combined integration of reusable cryogenic insulation with external insulation onto a cryogenic propellant tank for future reusable first stages. The thermal functionality of the derived designs has been verified in a first set of experiments within the AKIRA project. Within the subsequent TRANSIENT project experimental campaigns are currently being prepared where thermal and mechanical loads will be applied for up to 50 representative load cycles. The experiments will take place in 2022. This is a critical step for preparing this technology for the next generation of in-flight demonstrators of reusable winged first stages.

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