NOVEL HYPERSONIC LAUNCHER CONCEPT USING THIN-PLY CFRP COMPOSITES

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ABSTRACT

Thin-ply carbon fibre reinforced plastic (CFRP) composites may allow for significant structural mass reductions of space launch vehicles, and could even become a break-through technology for space transportation. To evaluate the potential mass savings and their implications for the feasibility of novel launcher configurations, the Aurora hypersonic launcher studies were initiated in late-2015/early-2016. The task of these studies is to define a series of spaceplane type launchers using thin-ply CFRP composites, to quantify the weight saving potential on vehicle level when using thin-ply composites as well as the latest technologies in other areas, and eventually to evaluate whether new types of launch vehicles can be realized that are infeasible with conventional technology.
This paper will provide a brief overview of the thin-ply technology and will discuss a first evaluation of vehicle level mass savings for a rocket propelled Aurora configuration. The vehicle system design and structural analysis approaches are still simplified, but do nevertheless allow for a first order assessment of the relative mass saving potentials compared to conventional structures.
The preliminary results presented in this paper indicate that thin-ply composites could indeed lead to large scale mass savings on vehicle level, and encourage to further advance this promising technology. Finally, the paper provides an outlook on the future development line of Aurora and associated technologies.

1. INTRODUCTION

It is well known that the high costs for space transportation have been and still are the limiting factor for large scale human exploration and exploitation of space. One of the main reasons for these high costs is the widely missing reusability of space launch vehicles. Additionally, the staging approach that requires the design, manufacturing and integration of several vehicles rather than just one vehicle, as well as limited flexibility, comparatively poor reliability and relatively high infrastructure costs of today's launch vehicles pose further cost drivers. Current activities on partly reusable launch vehicles (RLV) aim on significant reduction of space transportation costs. Different approaches are envisaged, with the toss-back and vertical landing of the SpaceX Falcon 9 first stage surely being the most famous one. Other approaches include reusable winged fly-back boosters or return and reusability of the most expensive launcher parts, such as the engines.
Currently, it is not known which cost reductions can actually be reached with the proposed approaches. However, it is likely that relative cost reductions will remain below 50%, if not even

far below 50%. Although relative cost reductions in the order of, say, 20-40% are impressive, they are hardly sufficient to revolutionize space transportation. This would probably require cost savings of at least an order of magnitude. Order of magnitude cost savings in turn will however require completely new vehicle concepts. This logic made various aerospace companies and research institutions in the recent decades work on alternative launcher concepts, whereas many hopes were counting on the "holy grail" of space transportation, single stage to orbit vehicles (SSTO). As we know today, none of these activities has ever led to an operational system. Frequently, the technical hurdles turned out to be too high to be mastered with the available technology or the available budget. Either technological breakthroughs in propulsion technology or large-scale vehicle weight reductions are required.

In the recent years a new material technology emerged, thin-ply composites that actually promises large weight reductions for launch vehicles. Whether this is already sufficient for enabling novel categories of space launch vehicles, is not known as of today, and needs to be investigated.

With this idea in mind, the Aurora space launcher studies were initiated at DLR in late-2015/early-2016, quickly joint by Swerea SICOMP (S), Bayern Chemie (D), and Delft University of Technology (NL). Initial preparatory studies for possible configurations had been done in late 2015 at DLR, and indicated useful starting points for the Aurora studies [1]. The objective is to develop and analyse a series of spaceplane-type launch vehicles using thin-ply based CFRP technology as well as the latest technological advances in other areas, and eventually to evaluate their technical and economic feasibility. Ideally, the result would be a technical feasible and fully reusable SSTO configuration able to provide large scale cost savings and flexibility increases with respect to state-of-the-art launch vehicles. However, this is a very ambitious aim, and experience from history advises to be cautious. Thus, some deviations of the fully reusable SSTO approach may be allowed if necessary. This may include launch assist systems or drop tanks. Such a design may be designated as "semi-SSTO" or "1.5 stage", whatever terminology the reader prefers.

Research history also tells that research on space planes, no matter if SSTO or not, as well as RLV in general, occurred in cycles (Fig. 1). Advances in technology or revived interest in advance launchers regularly led to larger research and development initiatives over several years. Typically, after some years the activities decline when mastering the technical challenges turns out to be still too ambitious. Pessimistic views might conclude that this is a never-ending cycle; however this is actually not true as long as there are objective technological advances. If this is the case, then with every cycle the gap towards a feasible system is getting smaller. Thin-ply composites provide such a technology that could make the gap getting smaller, ideally even disappear. The evaluation of this subject is the principal objective of the Aurora system studies. First results are very promising, but it is by far too early to come to a final conclusion. Research in the coming years will provide more insights.

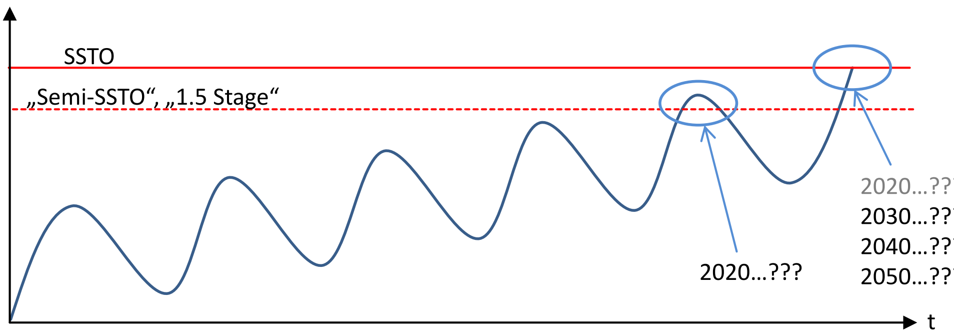

Fig. 1. Advanced launcher development cycles

This paper will focus on thin-ply CFRP technology, its application to the vehicle structure, and the weight saving potentials. The Aurora system study logic and overall vehicle system design is not the primary subject of this paper and will only briefly be discussed. More information on Aurora can be found in [2].

2. THIN PLY COMPOSITES

The achievable linear elastic strain level, when the material is essentially undamaged, is an important material characteristic for the dimensioning of many composite material structures. The first significant damage is commonly the development of micro-cracks. There are several ways to increase the micro-crack initiation strength, typically using altered or added material constituents. Several drawbacks might however occur like the need for specialised material combinations, lowered fibre content, lowered Tg, complex interactions between constituents, complicated manufacture, quality control during and after manufacture, cost, etc. Another approach is to instead change the local fibre architecture to thin-ply laminae, while keeping the material constituents unaltered as seen in Fig. 2.

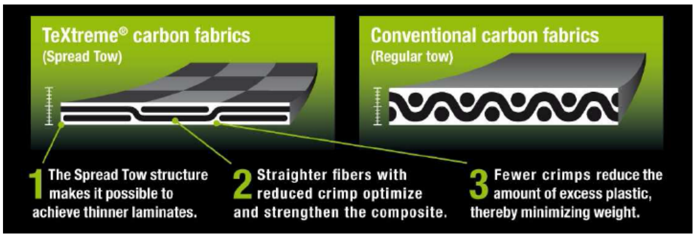

Fig. 2. Spread tow TeXtreme T700 fabric compared to a conventional fabric [3]

Thin-ply composites are a generic material type which can be expected to give benefits for most fibre- and matrix combinations according to the schematic picture seen in Fig. 3.

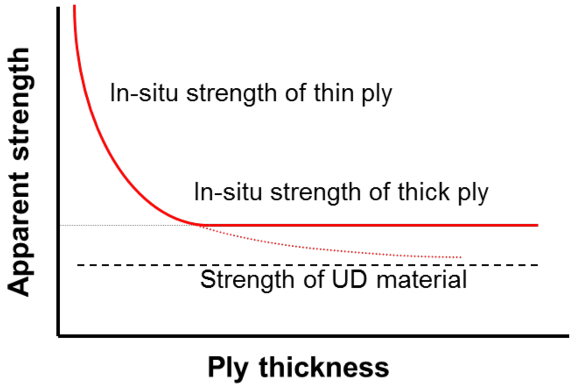

Fig. 3. General effect for apparent mechanical micro crack strength versus laminae thickness

The thin-ply effect alters the stress state in the laminae. Minute cracks still initiate but cannot propagate due to a larger crack-propagation energy needed. This effect cannot be seen in a standard FE-analysis, since the crack propagation needs to be studied. The effect of reduced laminae thickness for carbon/epoxy specimens with 0°/90° lay-up tension tested at -50°C can be seen in Fig. 4. The laminae thickness is 300 µm for pre-preg, L3 is 150 µm, L2 is 100 µm and L1 is 50 µm. Laminae thicknesses < 100 µm commonly give significant improvements, with doubled strain performance here for 50 µm laminae thickness. The fully developed crack in a thin-ply material is furthermore geometrically much smaller than for traditional roving laminae.

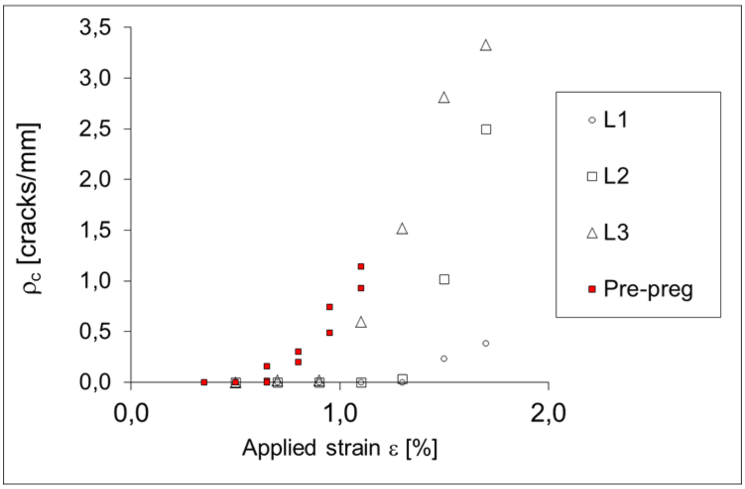

Fig. 4. Crack density vs applied tensional strain at -50°C for 0°/90° lay-up

The Swedish Oxeon company pioneered the spread tow thin-ply carbon fiber material in 2003 [3]. Several other material suppliers have in recent years introduced similar material types. The first applications were mainly sporting goods, car parts, boats, light aircraft etc. The use has however spread to advanced applications like aircraft and space, where Solar Impulse 2 (first solar driven round-the-world flight) is a good example from the aircraft industry [4]. It is likely that some of these new applications could not have been realised using traditional composite materials. Indeed, the new material type might be worthy of the name micrometer composites, since the laminae thickness and fiber architecture is defined in µm- instead of the usual mm scale. Thin-ply composites commonly enable weight savings of 10-30% compared to a traditional roving based material with identical material constituents, depending on the specification for the studied structure. A prime example for space applications is recent work by NASA where a 5.5-m diameter cryogenic demonstrator test tank was developed in cooperation with the Boeing Company. This liner-less tank is using thin-plies for permeation barrier, ventable and purgeable sandwich structures, and structural health monitoring to support damage tolerance [5]. The tank passed a series of fill-and-drain tests, containing cryogenic liquid hydrogen with acceptable seepage. Weight savings over aluminum tanks approached the 35% target set by NASA. NASA describes extended thin-ply composites applications like this in their recent development call “Game changing development program, thin-ply composites for space exploration applications” [6]. According to this, thin-ply composites are those with cured ply thicknesses ranging from 64 µm to 25 µm or less. Their potential is described as: “Thin-ply composites hold the potential for reducing structural mass and increasing performance due to their unique structural characteristics”. This may include [6], [7]:

- Improved damage tolerance,
- Resistance to micro-cracking (including cryogenic-effects),
- Improved aging and fatigue resistance,
- Reduced minimum laminate thickness,
- Increased scalability,
- Increased bearing strength.

Results of the CHATT Research Project

Thin-ply materials have shown radical improvements in critical material properties during use in the recent EU project CHATT (Cryogenic Hypersonic Advanced Tank Technologies) [8], [9]. On plate level, tensile tests of Textreme® thin-ply laminates have been performed at -50°C and -150°C and the evolution of damage has been analyzed. Very high strain levels of 1.7% have been applied to the test samples and the obtained results proved that formation of micro-cracks is significantly delayed in the thinnest laminae. Thermal fatigue tests of Textreme® thin-ply laminates were performed to study the micro-cracking in samples representing a liner-less tank concept subjected to a high number of thermal loading cycles. The results showed only a few micro-cracks in the thickest laminae after 100 cycles and no micro-cracks were found in the thinnest laminae (50-100 μm). These results show that the use of thin-ply laminae is promising in liner-less tanks as a gas barrier to prevent gas leakage.

The hybrid laminate concept that was chosen for the final subscale demonstrator tube contains both traditional roving- and thin-ply materials in the laminate. In this case, the traditional roving laminae will fail due to thermal and mechanical loads during service life whereas the thin-ply laminae are effectively damage free. Importantly, a crack in roving laminae is assumed to not progress through the adjacent thin-ply laminae. The final subscale demonstrator tube is 2 mm thick and has 3 integrated Textreme® thin-ply laminae. The function is hence similar to having 3 compliant (similar material properties as the roving laminae) load carrying liners in the structure, with predicted benefits regarding progressive damage distribution needed to achieve a leakage path through the tank wall, resulting in leakage redundancy for large tank structures. The selected liner concept is hence potentially superior to the use of one non-load carrying liner (polymeric or metallic) with its sensitivity to defects for large tanks and differing coefficient of thermal expansion (CTE). Swerea SICOMP developed manufacturing methods suitable for liquid composite manufacture (wet filament winding, RTM) of both thin-ply laminates and hybrid laminates that can be up-scaled to larger structures. The manufacturing challenge has been to achieve high quality and short cycle times. The processing issues have been solved using a combination of process simulation and manufacturing equipment modifications. The manufactured demonstrator tubes can be considered as having high quality, with < 0.5% voids in the critical thin-ply laminae. The manufactured subscale demonstrator tubes have successfully been tested in CHATT towards the demanding loading conditions specified in the project, indicating that the TeXtreme® material performs well as a load carrying liner material.

The results from the testing showed that the selected winding angle of ±25° for the Textreme® laminae effectively stopped the microcracks from growing through the whole thickness of the demonstrator. Hence, no leakage channel and Helium gas permeability leakage was produced through the laminate during testing although the axial tension load reached close to 1000 kN, corresponding to 1.6% axial applied strain, combined with -150°C and an inner pressure of 3 bar. The fractography evaluation after testing showed that the void content in the TeXtreme® laminae is < 0.5% while the void content in the roving laminae is 3%, see Fig. 5. No cracks could be found in the TeXtreme® laminae while cracks were found in the roving laminae.

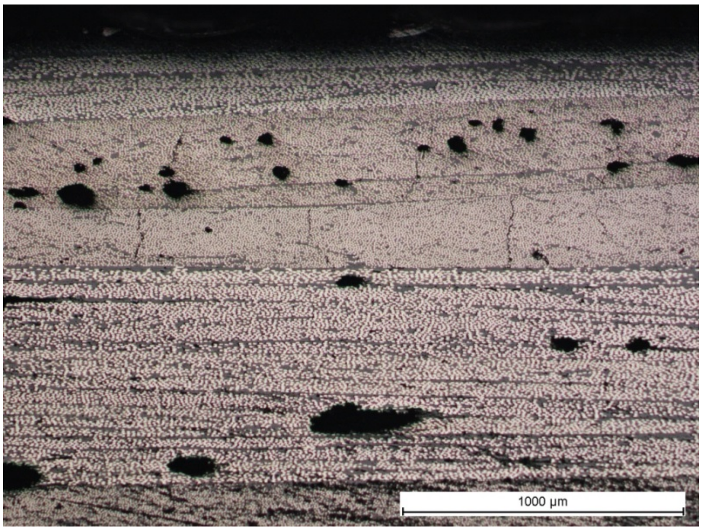

Fig. 5. Laminate view in tangential direction of the test section

The CHATT results are well in line with the NASA results regarding cryogenic tank development. The use of a fully load carrying liner (TeXtreme®) compliant with the rest of the laminate, three integrated liners, much higher dimensioning strains and out-of-autoclave manufacture, enable a predicted 30% structural weight reduction. The introduction of thin-ply materials thus generally enable 10-30% lighter other structures to be manufactured, which might enable new space vehicle designs.

3. AURORA SYSTEM DESIGN OVERVIEW

Vehicle Design Rationales and Study Logic

The Aurora study will design, assess and compare a series of different vehicle configurations based on a common basic vehicle and mission architecture. Preliminary assumptions and requirements include:

- Transport of a payload mass of at least 5 t into LEO
- Vehicle payload mass ratio of at least 1%
- Horizontal Take-Off Horizontal Landing (HTOHL) preferred; Vertical Take-Off Horizontal Landing (VTOHL) may however be considered as well
- Ideally fully reusable SSTO, but limited non-reusability or limited deviation from a pure SSTO approach may be allowed if necessary

Reusable launch vehicles may utilize different launch and landing methods, including the above mentioned HTOHL, Vertical Take-Off Vertical Landing (VTOVL) such as Falcon 9, or a combination in the form of VTOHL, as it is envisaged for many reusable booster concepts. For the Aurora vehicle studies the HTOHL approach has been selected as a baseline, whereas VTOHL may be considered as second option as well. Main reasons for the HTOHL preference include advantages on the operational and robustness side, which in turn may contribute to cost reductions and flexibility increases. Most notably, HTOHL configurations may at least in principle operate from any airfield and may provide abort capability at any point of the mission.

As noted before, in the ideal case Aurora would be a SSTO configuration. However, it remains open whether even with the application of thin-ply composites and latest technologies in other areas a SSTO can already be realized. Therefore, deviations from a fully reusable SSTO line in the form of, for example, launch support systems or fighter-aircraft like drop tanks (expendable or reusable) are options to be considered for Aurora. In particular a trolley like launch support system is currently assumed to be used for all Aurora HTOHL configurations. Although rail-guided acceleration such as envisaged for the FESTIP-Hopper offers large advantages [10], rail-launch simultaneously disables one of the fundamental advantages associated with HTO, namely the operation flexibility of being able to operate from arbitrary airfields/locations.

Within the previously discussed boundary conditions, large freedom exists concerning vehicle configuration design. Thereby, the optimum solution is far from being obvious, which in turn requires the investigation and assessment of different configurations. Fundamental trade-offs include the selection of the propulsion concept, whereas pure rocket configurations as well as combinations of rocket and air-breathing propelled vehicles will be investigated. This trade-off led to the creation of two branches within the Aurora studies, a pure rocket based branch (Aurora-R), and a combined air-breathing/rocket branch (Aurora-AB). Other trade-offs include the propellant selection, which is of course connected to the propulsion selection. Currently considered options include LOX/LH2 or LOX/kerosene combinations. The first “experimental” Aurora configuration is a LOX/LH2 fuelled configuration with rocket propulsion only, denominated Aurora-R1. For this R1 configuration a preliminary design has been completed. Currently, the first combined cycle concept, Aurora-AB1, as well as a rocket based configuration with kerosene fuel and optimized LOX/LH2 configurations are under definition (see Section 5 and [2]).

Aurora-R1 Preliminary System Design

The first vehicle configuration R1 is not an actual vehicle proposal, but rather an "experimental/trial" configuration that serves as a study vehicle for a first order estimation of thin ply-based mass savings and for identification of vehicle design sensitivities. Therefore, the focus was on designing a vehicle that provides representative boundary conditions, while no efforts were undertaken to optimize the vehicle. This will be left to future Aurora configuration designs.
The vehicle geometry is shown in Fig. 6 and the basic geometry and mass characteristics are presented in Table 1. The vehicle is equipped with four large LOX and LH2 drop tanks, as well as with wing tip and aft mounted rocket engines of yet generic nature. The utilization of wing-mounted drop tanks enables a better span-wise matching of mass forces and aerodynamic lift, thus reducing bending moments in the wings. The fuselage houses a payload bay of 10 m length, and another two non-integral LOX and 2 non-integral LH2 tanks. Future trade-offs will investigate integral tanks as well, as one of the potential main advantages of thin-plies is to enable lightweight CFRP cryo-tanks. The drop tanks are pressure stabilized and do not have to carry any vehicle loads. The vehicle dry mass includes a 15% mass margin for structure, TPS and subsystems group, and 10% for the propulsion group. The payload mass into a generic low inclination LEO transfer orbit of 80 x 450 km is 7 t when launching from an equatorial position in eastern direction. The corresponding payload mass fraction is 1.52%, while circularization of the orbit would cost approximately 50% of the payload mass.
The current design is relatively inefficient with the fuselage propellant volume fraction being just around 35%, resulting in a largely oversized fuselage. Also, the drop tanks are very large, resulting in high aerodynamic drag and cost penalties in case of non-reusability. Trajectory scheme and aerodynamic configuration are initial guesses rather than optimized design solutions. However, as noted before R1 is primarily a study vehicle for first order thin-ply mass saving estimations (see Section 4).

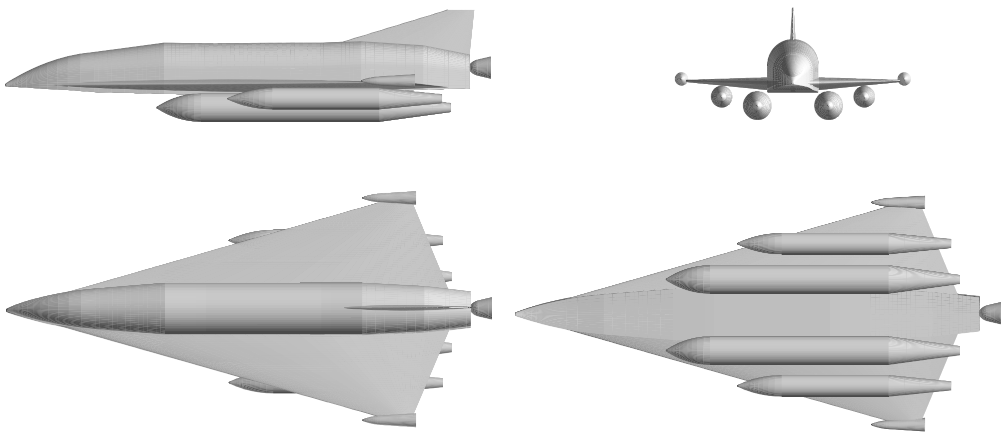

Fig. 6. External geometry of "experimental/trial" configuration Aurora-R1 (engine geometries not representative)

Table 1. Aurora-R1 main geometry and mass data

Length (excluding aft mounted rocket engines)	52.7 m
Wing span (excluding wing tip engines)	24.0 m
Maximum fuselage diameter	5.75 m
Fuselage stored propellant mass	150 t
Drop tank stored propellant mass	240 t
Dry mass (incl. residuals, reserves, RCS, drop tanks)	62.2 t
Payload mass (80 x 450 km LEO transfer orbit)	7 t
Total take-off mass	459.2 t

4. VEHICLE STRUCTURAL DESIGN USING THIN-PLY COMPOSITES

In this section thin-ply structural mass calculations will be presented for the Aurora R1 configuration. The focus will be on two effects. Firstly, thin-ply composites may allow for more efficient material utilization. In particular, structures that are sized according to minimum ply number or panel symmetry considerations may benefit from lower ply thicknesses. Secondly, the increased material strength will be evaluated on vehicle level. Another major advantage, the potential application for liner-less and very lightweight cryogenic tanks as discussed in Section 2, will be investigated later within the Aurora studies.
Structural mass estimations at preliminary design level are no simple task when designing a vehicle of a category that never has been built in history and with challenges that are unmatched by today's launch vehicles. It is even questionable whether at preliminary design level accurate mass predictions for such a vehicle are possible at all. The applications of large safety factors and mass margins as well as worst-case assumptions in cases where problems have to be simplified are reasonable strategies. This is particularly important for SSTO-like vehicles, where the payloads mass fractions are low and even small vehicle dry mass increases can result in the unfeasibility of a launcher concept. As structural and TPS design for Aurora are being done on a preliminary level with typical preliminary system analysis tools, it is appropriate to consider relatively high safety factors and margins as well.

Aurora-R1 Structural Analysis Overview

The structural analysis for Aurora-R1 has been done using a parametric ANSYS-based vehicle modelling and analysis tool named HySAP (Hypersonic vehicle Structural Analysis Program). HySAP iteratively adapts structural member thicknesses in an automated loop until convergence has been reached. A converged design is assumed as soon as the vehicle structural mass changes by not more than 1.5% in 4 successive iterations.
The vehicle is completely modelled with shell elements and honeycomb sandwich design is utilized for all structural components. The ANSYS geometry model is shown in Fig. 7.

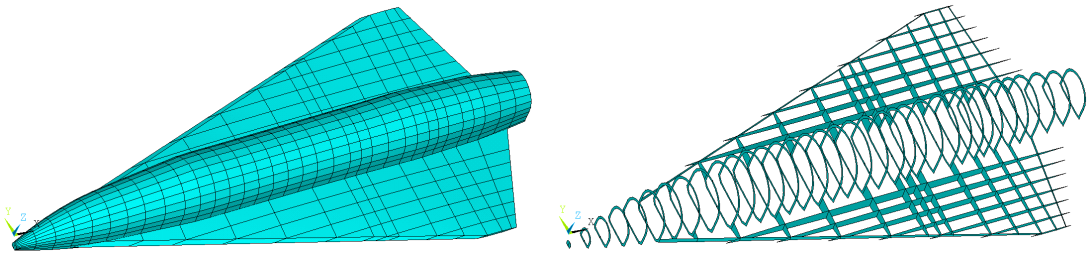

Fig. 7. ANSYS geometry model; full model (left), skins removed (right)

Optimization of facesheet and core thicknesses is done iteratively on a local panel basis. Sizing of the facesheets includes Von Mises (metal) or Tsai-Wu (CFRP) for strength, as well as facesheet wrinkling, shear crimping, and intracellular buckling. The sandwich core heights are sized to prevent global buckling of the panels. Furthermore, the Shanley criterion according to [11] is applied for sizing of the fuselage frames against global fuselage buckling. CFRP facesheets are symmetric and balanced and consist of 0°/90°/45°/-45° plies with at least 2 plies per orientation, yielding a minimum of 8 plies per facesheet. Furthermore, a minimum thickness of 0.5 mm per facesheet has been considered for metallic and CFRP facesheets. The analysis is currently limited to 3 load cases (LC):

- LC1: Maximum n_x during rocket ascent; an acceleration of 6.0 g is applied here which is higher than the actual maximum acceleration of 4.7 g in the trajectory simulation

- LC2: n_z = 2.0 g normal acceleration manoeuvre during ascent at hypersonic speed and with full tanks and flap deflection loads for trimming; this is conservative as the maximum normal acceleration found in the trajectory simulation is 1.45 g; the pressure distribution was generated using an inclination based analysis code that also provides heat flux and temperature loads over the vehicle surface
- LC3: Landing with main gear touch-down and a normal acceleration of n_z = 2.5 g

The higher acceleration levels in LC1 and LC2 provide some contingency margins for covering dynamic effects and other secondary loadings that are not considered so far. Also, in LC2 so far only a hypersonic manoeuvre has been considered and hypersonic pressure distributions may not necessarily be as demanding as subsonic pressure distributions [12]. Future investigations will include more sophisticated loads analyses. Subsystems are modelled via mass point elements, while the propellant masses of the non-integral- and drop-tanks are introduced at the corresponding structural member positions.
A 1D TPS sizing code has been applied for computing the TPS masses for the complete vehicle surface. For the current configuration no active cooling is required, with the maximum temperature at nose and leading edges approaching 1700 K. The vehicle surface is segmented into 12 temperature areas with an individual insulation thickness computed for each temperature area. Five different TPS material concepts are being used, including FRSI, AFRSI, TABI, AETB-TUFI, and CMC according to [13]. This is based on the re-entry trajectory only, as during ascent the heat loads are comparably small. This will change as soon as air-breathing trajectories will be analysed. The insulation thicknesses are sized such that a user-defined maximum temperature at the primary structure under the TPS is not exceeded. This maximum allowed structural temperature, as assured by the TPS in turn will be applied to the wing and fuselage structure skins in the HySAP structural analysis. So far, no vehicle internal heat distribution analysis is available. Therefore, an assumption is made that the internal members ribs, spars and frames are at room temperature. This may present a worst case scenario as the temperature differences between the warm/hot skins and the cold internal members may create strong thermal stresses.
A safety factor of 1.5 has been applied to all strength and buckling/stability allowables. For strength sizing of metallic structures, this applies to the yield rather than to the ultimate material strength. Furthermore, the computed structural masses will be increased by a non-optimum factor of 1.67 for the wings and 1.58 for the fuselage. This covers various structural details and unknowns that are not considered in the idealized “optimum” vehicle structural analysis, such as fasteners, bolts, attachments, local reinforcements, cut-outs, etc. When adding the previously mentioned 15% mass margin, the safety factor of 1.5, and the non-optimum factor, the structural mass exceeds the computed theoretic minimum structural mass required to resist the 3 considered load cases by a factor of 2.88 for the wings, and 2.73 for the fuselage. This margin together with the higher accelerations levels applied in the loads analysis is considered to be sufficient to cover the various simplifications and uncertainties at the current design level.

Results

Vehicle structures made of 3 different materials have been considered: aluminium-lithium 2195 that had also been used for the Space Shuttle Super Lightweight Tank (SLWT) [14], and which is used here as a benchmark, and two different CFRP composites. IM7/PETI-5 is a polyimide based high temperature composite with material data provided in [15]. Unfortunately, in the reference only a few data points are available and it is not yet clear whether the material properties provided already represent consolidated data. Nevertheless a high-temperature CFRP like the latter one is interesting for comparison. The second composite is a PEEK based material, with material data taken from [16]. For the composite materials an initial ply thickness of 0.125 mm has been used. The structural skin temperature levels considered start at 300 K with a step size of 25 K. For IM7/APC-2, the maximum temperature considered is 394 K, and 422 K for Al-Li. For comparative purposes always

the whole vehicle structure is made of the particular material, although in practice of course different materials will be utilized for different structural components.
The left part of Fig. 8 shows computed vehicle structural masses as a function of structural skin temperature. Thereby, the masses as shown represent wing and fuselage mass, while other structural mass items such as non-integral tanks, fin or thrust-frame are considered in the mass budget as subsystems with empirical/statistical mass estimation. The aluminium vehicle structure features a relatively strong increase with increasing temperature in particular due to thermal stress build-up. The CFRP composite structures instead show only small mass changes with increasing temperature. This is a result of the low CTE on the one hand, but also strongly results from the fact that a large number of panel facesheets are effectively "oversized" due to minimum thickness/minimum ply number considerations. If then the thermomechanical loads are increased, they can to a large extent be covered by the existing material without the need of increasing facesheet thickness. The striking structural mass increase for the IM7/PETI-5 structure beyond 450 K results from a relatively sharp degradation of material properties, in particular loss of compressive strength parallel to the ply orientation as well as transverse tensile strength.
The right hand side of Fig. 8 shows computed TPS masses as a function of allowed structural temperature. When using the APC-2 based vehicle structure, it is obvious that the maximum considered structural temperature should be used as the structural weight increase with increasing temperature is low, while simultaneously the TPS mass decrease is significant. When considering structure and TPS on an integrated basis, a PETI-5 based airframe at 450 K is almost as lightweight as an APC-2 based airframe at lower temperatures.

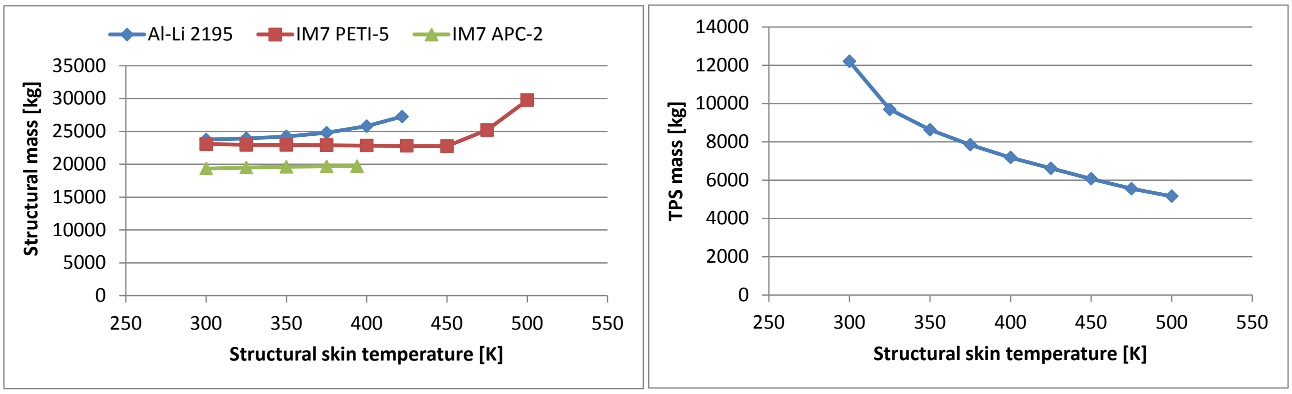

Fig. 8. Structural masses for different structural skin temperatures and three different materials (left); TPS mass as a function of allowed structural skin temperature (right)

Based on these results, the effect of ply thickness variation shall be demonstrated. For this, the IM7/APC-2 vehicle structure at 375 K structural skin temperature has been selected. Vehicle structural analyses have been done with varying ply thicknesses between 0.25 mm and 0.025 mm, with the results being shown in the left part of Fig. 9. Note that no material property changes have been considered. Thus, the change in structural mass is solely a result of the more efficient material utilization, most notably minimum ply number effects. The results reveal an impressive structural mass saving potential. The lowest ply thickness of 0.025 mm allows for mass reduction of 38.1% compared to the highest ply thickness of 0.25 mm. Between 0.05 and 0.025 mm ply thickness no significant mass saving can be achieved anymore, implying that in this case 0.05 mm is a reasonable target value. When compared to the baseline ply thickness of 0.125 mm as used for the results shown in Fig. 8 before, 0.05 mm still allows for a mass saving of 13.2%. It is explicitly to be noted that mass savings of this order are to a large extent a result of the generally low thicknesses of the facesheets of the vehicle that are in many cases sized by minimum ply number considerations rather than mechanical loads. In case of highly loaded structures with high wall thicknesses lower mass savings are to be expected.
The right part of Fig. 9 further investigates the effect of the reduction of ply thickness. Shown here is the fraction of vehicle facesheets that are sized according to different sizing criteria. As can be

seen, in case of the high ply thicknesses the majority of the facesheets are sized according to minimum ply number / minimum thickness considerations. If the ply thickness is reduced, the number of components sized by actual strength and stability criteria increases. Note that no discrimination between minimum thickness and minimum ply number is made in Fig. 9. Especially for the thin ply example (0.025 mm) many facesheets are at the minimum allowed thickness of 0.5 mm and can therefore not further be reduced in thickness.

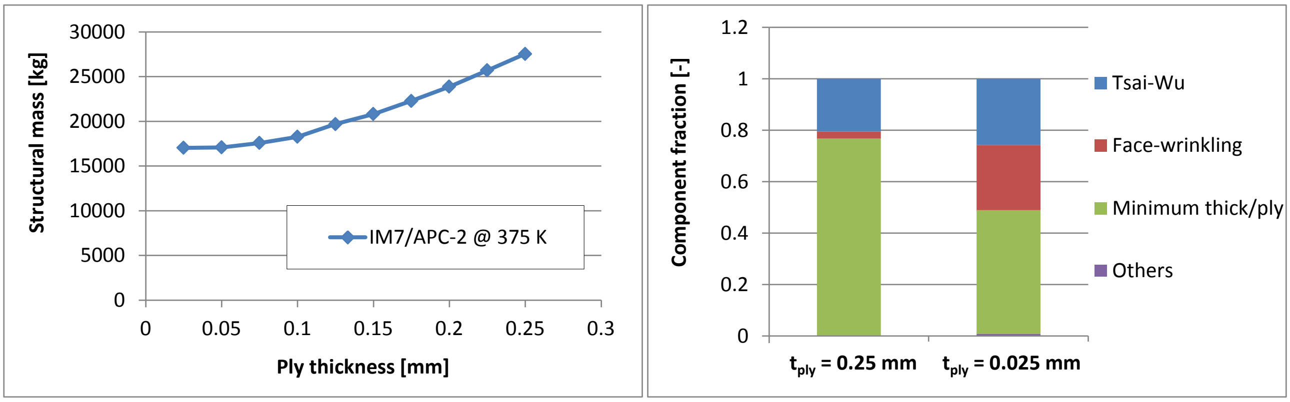

Fig. 9. Computed structural masses for different ply thicknesses (left); sizing criteria for two selected ply thicknesses (right); IM7/APC-2 at 375 K structural skin temperature

Fig. 10 investigates the impact of material strength increase. The thin ply effect can lead to an increase in the material transverse strength as well as shear strength (see Fig. 3), while the strength parallel to the fibres remains unchanged. Generic preliminary calculations for IM7/PETI-5 and IM7/977-2 UD-plies performed as part of this study indicate a strength increase potential of up to 60%. These results however still need to be confirmed by more detailed analysis with considering the vehicle level relevant boundary conditions. Thus, Fig. 9 shows the structural mass savings for generic strength increases of 10% to 50%, actually being lower than the predicted 60%. Computations have been done for an IM7/APC-2 vehicle structure at 375 K skin temperature using thin-plies with 0.050 mm ply thickness. The resulting structural masses (left part of Fig. 10) illustrate that a structural mass reduction of 6.2% could be reached when increasing the material transverse and shear strengths by 50%. The right part of Fig. 10 shows the fraction of vehicle component facesheets sized according to different sizing criteria. As can be seen, with increasing material strength the number of components sized by strength reduces, while the number fraction for the other sizing criteria increases.

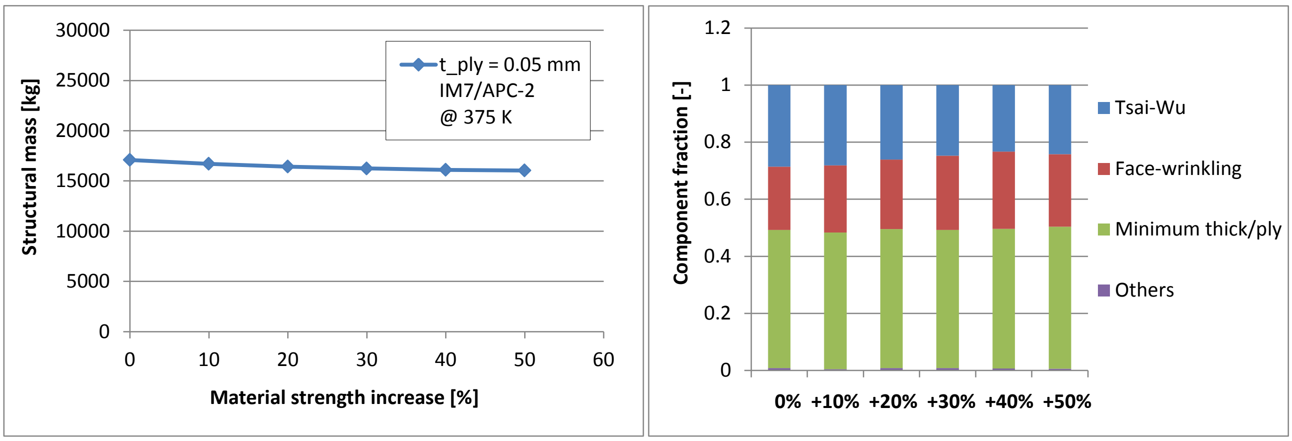

Fig. 10. Computed structural masses as a function of material transverse and shear strength increase (left); sizing criteria (right); IM7/APC-2 at 375 K structural skin temperature

Fig. 11 provides a structural component group mass breakdown for the strength increase investigation. The highest structural mass saving of up to 8.2% could be achieved for the wings skins. The lowest mass benefit was found for the wing ribs, where the maximum weight saving was only 3.2%. Note that the relatively high mass of the frame group is a result of a minimum sandwich core thickness of 50 mm, which was applied only for the frames.

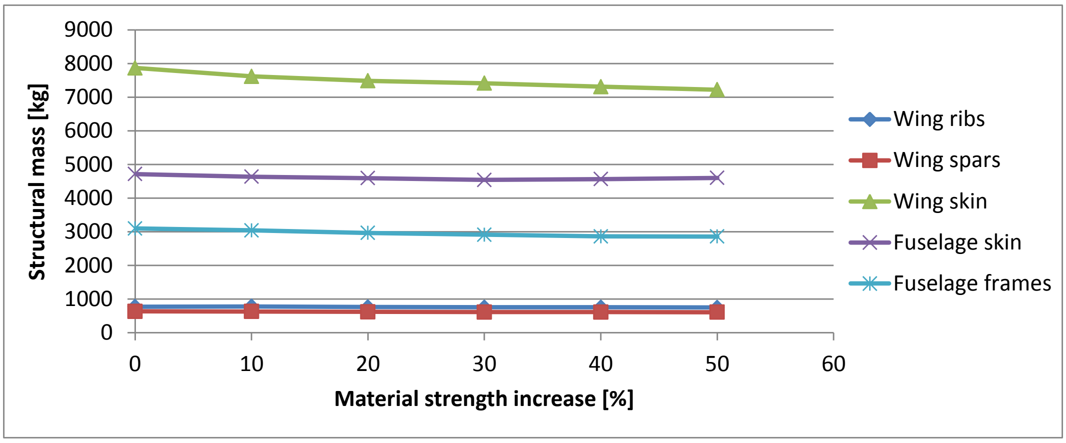

Fig. 11. Component breakdown for computed structural masses as a function of material transverse and shear strength increase; IM7/APC-2 at 375 K structural skin temperature

Considering the IM7/APC-2 vehicle structure with 375 K structural skin temperature, a structural mass saving of 18.5% can be reached when switching from the baseline 0.125 mm plies to 0.05 mm plies and assuming a generic, but probably not unrealistic transverse and shear strength increase of 50%. Compared to a vehicle structure with relatively thick plies of 0.25 mm, the mass saving would even be 41.8%. Although the preliminary nature of these results has to be highlighted, they indicate the very promising potential of thin-plies for launch vehicles. Early calculations done for an Aurora-R2 flying wing configuration led to even higher mass savings than found for R1 [2].
Further mass savings might be possible if the material architecture would be optimized for thin-plies, or if structures are considered where dimensioning for strength failure rather than other sizing criteria dominates. Furthermore, the large mass saving potential that thin-plies may allow for by enabling CFRP cryo-tanks has not yet been included in the vehicle analysis.

5. DESIGN PERSPECTIVES FOR FURTHER AURORA CONFIGURATIONS

Based on the promising experiences made with the R1 experimental configuration and the associated thin-ply utilization results, next steps will include the definition of an efficient LOX/LH2 configuration, a LOX/kerosene configuration, as well as the first air-breathing configuration. First considerations for a flying wing configuration R2 with wing-stored kerosene fuel indicate the potential for an extremely lightweight vehicle structure [2]. When utilizing the otherwise empty wings for kerosene storage similar to conventional aircraft, the vehicle size can be decreased with corresponding benefits on the mass and aerodynamic side. Furthermore, wing-stored kerosene offers additional gains. One of them is that the redistribution of mass from the centre fuselage to the wings will reduce bending moments in the wings and therefore allow for lower wing masses, an advantage that could already be exploited for R1 by using wing-mounted drop tanks. Also, the inherent rib/spar segmentation of the wings into compartments will eliminate the sloshing problem at least for the fuel, which otherwise could become a critical design issue for horizontal launchers. Moreover, the cooling capacity of the kerosene in the wings can be utilized for reducing wing TPS mass in case that the ascent thermal loads are dimensioning. Operational issues concerning the wing-stored kerosene approach are however still to be checked, in particular as the tanks cannot be pressurized for structural weight reasons.
Compared to the relatively simple R1 configuration design, future analyses will place a stronger focus on system optimization and more sophisticated modelling. The simplified CFRP analysis models in particular for thin-plies will be complemented and simultaneously validated by high fidelity numerical models. This will allow for a deepened understanding and more reliable thin-ply

based mass saving potentials on vehicle level. For future LH2 fuelled configurations, furthermore the utilization of thin-ply based integral cryo-tanks is to be studies, as here one of the major advantages of thin-plies for space launcher may be found.

A special focus will be placed on TPS and TPS-structure integration. The reusable Space Shuttle used an intricate net of ceramic TPS tiles, which could withstand very high temperatures, but were very fragile and many tiles needed to be replaced after flight, leading to very high maintenance costs. The current Aurora-R1 TPS is based on these types of materials, and is therefore not necessarily the optimum solution. For an RLV, apart from fulfilling the thermal requirements, the main requirements would be related to reusability and reliability. Lessons learned from the Space Shuttle taught us that the TPS should be more robust and less sensitive to damage, which would exclude ceramic tiles. NASA concluded that thermal protection tiles with a metallic outer protecting casing would be very promising and this new technology was applied in the conceptual design of the X-33 [17].

This class of TPS, either ceramic or metallic, is also known as a cold-structure solution, where the thermal-protection function is separated from the load carrying function. The latter is taken care of by the underlying structure that is to be kept at a low temperature. The alternative is that of a hot structure, where both functions are combined. As Aurora is to be equipped with a lightweight CFRP airframe, a hot structure is no option due to the limited temperature carrying capability of CFRP. Nevertheless the structure should operate under elevated temperatures in order to reduce TPS mass, as it was done for the R1 configuration. Thereby, the optimum may strongly depend on details such as thermal bridging and local hot spot generation, and is therefore not easily to be determined at preliminary system analysis level.

The TPS of A rocket based Aurora configuration is dimensioned by the re-entry loads. An Aurora-type RLV with air-breathing propulsion however will experience high thermal loading both during ascent and descent. Critical areas are the nose region, wing leading edges, (air-breathing) engine inlets, and control surfaces, to name a few, since nose and leading edge radii have to be small in order to minimize aerodynamic drag. However, when the surface area is small, e.g., a small nose or a leading edge, one is faced with two problems: the surface area to radiate heat is too small to matter, and the heat load is extremely high, as it is inversely proportional with the radius. Alternative solutions can be found in semi-passive and active TPS, of which an overview is presented in Fig. 11. The fundamental operating principle is to use a coolant that transports the heat.

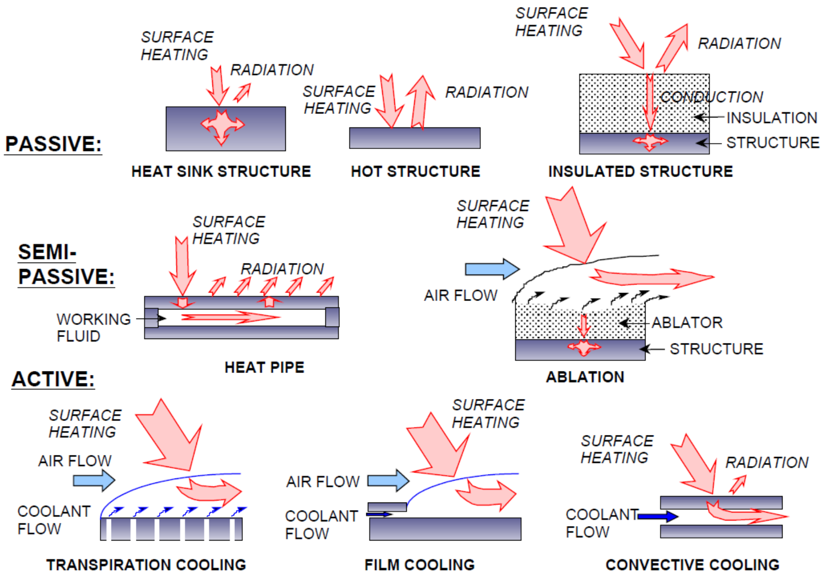

Fig. 12. Types of thermal protection systems [18]

The air-breathing propulsion system selection for Aurora-AB configurations will strongly impact the thermomechanical loads environment. Most probably, integrated air breathing engine concepts will be considered, combining at least two different types of air-breathing engines or an air-breathing engine with a rocket engine. The option to be considered for the first AB-configuration can be described as a set of combined cycle engines with two main components: a turbo jet needed to accelerate the vehicle up to a flight Mach number of 2.1, approximately, and a Ramjet engine which will take over afterwards and cover the flight trajectory up to a flight Mach number of 5(+). Afterwards the airbreathing mode has to be shut down and an integrated rocket motor has to take over. Critical for both, the system performance as well as the thermomechanical loads, is the engine installation approach. Different installation approaches are possible and have individual advantages and disadvantages. For the AB1 version of Aurora the engine compartments will most probably be located on the dorsal or leeward side of the wings, leading to a highly integrated vehicle/propulsion sub system configuration. The thermo-structural design of such a configuration will be very different to that of rocket propelled configurations and requires different design solutions. But also in this case thin-ply composites are expected to provide vehicle weight decreases, a question that will be addressed in the future of the Aurora study.

6. SUMMARY AND CONCLUSION

This paper provided a brief overview of the thin-ply CFRP technology and investigated its application on vehicle level for the hypersonic launcher configuration Aurora-R1defined within the Aurora system studies. Although the vehicle system design and structural analysis procedures are simplified, the principal mass saving potential of thin-ply composites could be demonstrated. The investigations for the Aurora-R1 configuration show that structural mass savings in the order of ~20% compared to conventional CFRP appear to be realistic. Future investigations will utilize more sophisticated analysis procedures to quantify the actual mass saving potential with a higher accuracy. Thereby it is important to always consider the vehicle level since theoretical improvements on material level cannot directly be extrapolated to vehicle level weight savings without a representative vehicle design. The actual mass saving potential strongly depends on the particular structural and material concepts, as well as on the vehicle and mission design and the corresponding loading environment.
Based on the promising results for the first Aurora-R1 study configuration, further Aurora configurations will be defined with a higher level of detail, including pure rocket as well as rocket/air-breathing combined cycle concepts. Thereby, not only thin-ply composites, but also latest technological improvements in areas such as thermal protection and propulsion technology will be included. The ultimate aim is to evaluate whether novel vehicle configurations are possible now and how they compare to conventional launch vehicles.

7. REFERENCES

[1] Roorda, E.: “Configuration and Preliminary Design Studies for the “LEO Fracht Transporter” (LFT) Concept”, DLR SART TN-016/2015, Bremen, June 2016

[2] Kopp, A., Mattsson, D., Olofsson, K., Kurth, G., Mooij, E., Roorda, E.: “The Aurora Space Launcher Concept”, Deutscher Luft- und Raumfahrtkongress 2016, Braunschweig, Germany, September 2016

[3] www.oxeon.se

[4] http://www.compositesworld.com/articles/spread-tow-technology-takes-off-

[5] Vickers, J.: “NASA Composite Cryotank Technology Project Game Changing Program”, NASA Marshall Space Flight Center, December 2015.

[6] NASA: "Game Changing Development Program, Thin-Ply Composites for Space Exploration Applications – Request for Information", Number NNH15ZOA007L, NASA, June 2015.

[7] Amacher, R., Cugnoni, J., Botsis, J., Sorensen, L., Smith, W., Dransfeld, C.: "Thin ply composites: Experimental characterization and modeling of size effects". Composite Science and Technology 101 (2014) 121-132.

[8] Mattsson, D.: "Liner-less tanks for cryogenic applications – Final report", CHATT Deliverable D3.4.7, Swerea SICOMP, June 2015.

[9] Sippel, M., Kopp, A., Mattsson, D., Freund, J., Tapeinos, I., Koussios, S.: "Final Results of Advanced Cryo-Tanks Research Project CHATT", 6th European Conference for Aeronautics and Space Sciences (EUCASS), 2015

[10] Spies, J., Kuczera, H.: "The Sub-Orbital Hopper – one of FESTIP's Preferred Concepts", AIAA-99-4945, 9th International Space Planes and Hypersonic Systems and Technologies Conference, Norfolk, Virginia, November 1999

[11] Shanley, F. R.: "Weight-Strength Analysis of Aircraft Structures", Second Edition, Dover Publications Inc., New York, 1960

[12] Bardenhagen, A.: "Massenabschätzung und Gesamtauslegung der Unterstufe von Hyperschall-Raumtransportern", Dissertation, Institute of Aircraft Design and Lightweight Structures, Technical University of Braunschweig, 1998

[13] Garbers, N.: "Overall Preliminary Design of the Thermal Protection System for a long Range Hypersonic Rocket-Powered Passenger Vehicle (SpaceLiner)", 7th European Workshop on Thermal Protection Systems and Hot Structures, Noordwijk, The Netherlands, April 2013

[14] Jenkins, D. R.: "Space Shuttle – The History of the National Space Transportation System – The First 100 Missions", Sixth Printing, Crecy Publishing Ltd, Manchester, UK, June 2010

[15] Johnson, T. F., Gates, T. S.: "High Temperature Polyimide Materials in Extreme Temperature Environments", AIAA-2001-1214, 42nd AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference and Exhibit, Seattle, Washington, April 2001

[16] APC-2 PEEK Thermoplastic Polymer, Cytec technical data sheet

[17] Cook, S.: "X-33 Reusable Launch Vehicle Structural Technologies", AIAA-96-4563, AIAA 7th International Space Planes and Hypersonic Systems and Technologies Conference, Norfolk, Virginia, November 1996

[18] Blosser, M. L.: "Advanced Metallic Thermal Protection Systems for Reusable Launch Vehicles", PhD thesis, Graduate School of Engineering and Applied Science, University of Virginia, May 2000