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THE AURORA SPACE LAUNCHER CONCEPT

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Abstract

Large-scale space transportation cost reductions will require completely new launch vehicle configurations. Various alternative launcher configurations have been investigated in the recent decades including hypersonic single stage or two stage to orbit vehicles. Unfortunately, these alternative concepts were frequently found to be technically infeasible or their economic viability perspectives were considered to be doubtful. However, technological advances can shift the results of earlier trade-offs, and vehicle concepts that so far were found to be uninteresting or even unfeasible can become attractive. Recently, with thin-ply composites a new material technology emerged that offers the potential for major structural weight reductions of launch vehicles. Also in other relevant technology areas improvements have been achieved. Therefore, it is reasonable to re-evaluate alternative launcher concepts and to assess whether with using thin-ply composites and the latest technologies in other areas, novel launcher configurations can be made feasible.

Based on this idea, the Aurora space launcher study was initiated in late-2015/early-2016 with contributions from several European partners. Within the Aurora studies, several spaceplane-like vehicle configurations with different geometries, propulsion systems and mission profiles will be designed, investigated and evaluated with respect to their technical and economic feasibility. The first step of this study is a first order investigation of thin-ply composite mass saving potentials for selected configurations.

This paper consists of two parts. The first part will discuss the study logic and the current status of the Aurora studies and introduces the first three vehicle configurations. In the second part the focus will be shifted to the thin-ply technology and its application on vehicle level. Corresponding results for the first two configurations will be presented and discussed. Although the analysis procedures are still simplified and the findings of preliminary nature, the results indicate that indeed large weight savings are possible when using thin-ply composites, whereas the actual mass saving strongly depends on the particular vehicle configuration, load environment and structural design. This opens very promising perspectives for the realization of advanced launcher configurations and encourages to further investigate alternative space transportation systems.

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 space launchers 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 mass reductions are required.

Research history also tells that research on spaceplanes, no matter if SSTO or not, as well as reusable launch vehicles (RLV) in general often occurred in cycles (FIGURE 1). Advances in technology or revived interest in advanced 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, and that working on such alternative launcher configurations will always yield the same results as older research initiatives did. 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. Therefore it is mandatory to re-evaluate advanced launcher concepts if new technologies appear that have the potential of shifting the results of earlier trade-offs.

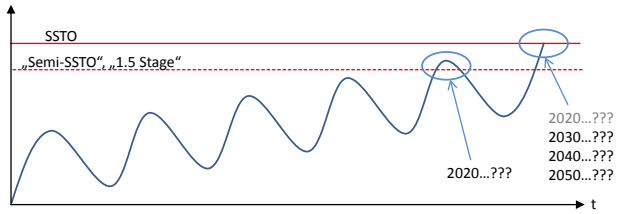

FIGURE 1. Advanced launcher development cycles

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 analyze a series of spaceplane-type launch vehicles using thin-ply based carbon fiber reinforced plastic (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 aircraft like drop tanks. Also designs similar to the FESTIP/Hopper concept that accelerate a payload to high but still suborbital velocity, still requiring a kick-stage for orbit insection, may be considered [2]. These approaches may lead to a design that could be designated as “semi-SSTO” or “1.5 stage”, whatever terminology the reader prefers.

This paper consists of two parts. The first part will discuss the Aurora study logic and introduce the designs of the first three experimental configurations. In the second part, the focus will be shifted to thin-ply technology, its application to the vehicle structure, and the weight saving potentials. Potential advances in other relevant technology areas are not subject of this paper, but will be considered in future Aurora works.

2. AURORA SYSTEM DESIGN OVERVIEW

2.1. 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. The take-off and landing approaches obviously also dictate the vehicle configuration to a large extent. For instance, VTOVL configurations obviously do not require wings, which in turn is an essential feature of HTOHL and VTOHL configurations. Every approach has its advantages and disadvantages. The optimum approach will largely depend on the particular mission, the available technologies, operational matters, robustness and reliability considerations, and of course, costs. For the Aurora vehicle studies the HTOHL approach has been selected as a baseline, whereas VTOHL may be considered 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.

SSTO compared to multi-stage vehicles is another trade-off, whereas the obvious drawback of SSTO is that it is just at or still even below the edge of technical feasibility. Even if SSTO can be realized, payload mass fractions will always be lower compared to multi-stage vehicles. The theoretical advantages on the other hand are impressive. Instead of several vehicles, only one vehicle has to be designed, manufactured, and operated. No stage integration, stage interfaces and stage separation procedures are required, thus reducing costs and failure probability. However, it remains open whether even with the application of thin-ply composites and the latest technologies in general a SSTO can already be realized. Therefore, as noted before, 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 [2], 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 advantages and disadvantages of both concepts for launch vehicles are well known. Most notably, the high energy density of LH2 usually allows for higher payloads. The low mass density on the other hand requires much larger propellant tanks and associated structural mass and volumetric penalties. Furthermore, LH2 needs to be stored at temperatures around 20 K in cryogenic pressure vessels. This offers some advantages for kerosene in particular on the operational and cost side. When considering winged ascent vehicles however, kerosene provides a huge additional advantage that may largely impact the trade-off between LH2 and kerosene. This advantage is the ability to store parts or even the complete kerosene in the otherwise empty wings, just as it is done in conventional aircraft. As a result, the vehicle fuselage size can largely 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 center fuselage to the wings will reduce bending moments in the wings and therefore allow for lower wing masses. 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. For these reasons, a special focus will be placed on fully or partly kerosene fueled configurations.

The Aurora study is planned to consist of two phases. In the current first phase the focus is on first order identification of weight saving potential and tendencies. The vehicle configurations in this first phase including the three examples to be discussed in this paper are therefore of experimental nature, and no optimized actual vehicle proposals. Instead, their main task is to provide representative boundary conditions for the above mentioned first order mass saving studies. Based on the findings of these studies, a second phase will investigate and optimize selected vehicle concepts on a more systematic basis and in a higher level of detail.

The first three configurations will represent three different vehicle design approaches: a LOX/LH2 rocket-propelled vehicle with large drop tanks, a LOX/kerosene flying wing rocket-propelled configuration, and an air-breathing configuration of not yet defined geometric configuration. The current design status of the three configurations will be discussed in the subsequent sub-sections. The structural design with application of thin-ply composites will be presented in Section 4.

2.2. Configuration Aurora-R1

In line with the above mentioned two-phase approach of the Aurora study, 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 for any optimization. This will be left to future Aurora configuration designs.

The vehicle geometry is shown in FIGURE 2 and the basic geometry and mass characteristics are presented in TAB 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 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, for a first order thin-ply mass saving estimation this is completely sufficient. This investigation will be subject of Section 4. Also a 1D TPS sizing study using standard TPS materials has been done, and will be discussed in Section 4 as well.

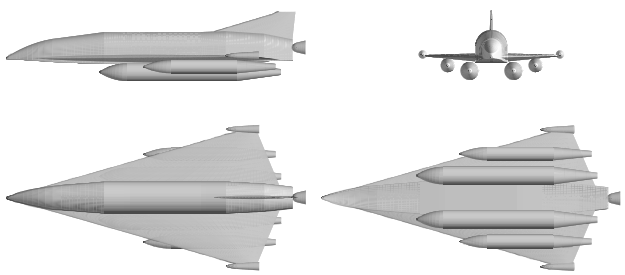

FIGURE 2. External geometry of “experimental/trial” configuration Aurora-R1 (engine geometries not representative)

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 equatorial)	7 t
Total take-off mass	459.2 t

TAB 1. Aurora-R1 main geometry and mass data

2.3. Configuration Aurora-R2

The R2 configuration is a flying wing design that is still in the geometry definition process. R2 will use a LOX/kerosene propellant combination. The LOX will be stored in several parallel arranged cylindrical or conical pressure vessels made of aluminum-lithium alloy. The kerosene instead is planned to be stored directly in the wing similar to conventional aircraft, thus parts of the wing will form an integral tank. Operational issues concerning this approach are however still to be checked, in particular as the tanks cannot be pressurized for weight reasons. The major advantage of this configuration design is that the propellants in the wings (kerosene and the LOX pressure vessels) can be placed such that their inertia forces and the aerodynamic lift forces partly cancel out each other. Thus, the bending moments can drastically be reduced, which may significantly reduce the airframe weight. Also, the vehicle size can become comparatively low due to the omitting of low density LH2 and the utilization of the wing volume for propellant storage. In summary, this enables a very compact and lightweight configuration. FIGURE 3 shows a preliminary albeit not yet finished vehicle geometry that was used for initial mass estimations, trajectory simulations, and structural design sensitivity studies (Section 4). TAB 2 provides tentative dimension and vehicle mass data.

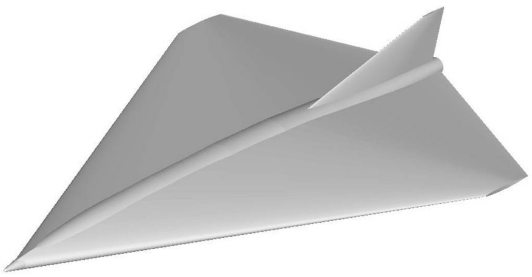

FIGURE 3. Tentative geometry of Aurora-R2

Length (excluding rocket engines)	40.5 m
Wing span (excluding wing tip engines)	22.0 m
Vehicle total mass	450 t

TAB 2. Tentative Aurora-R2 main geometry and mass data

Compared to the relatively simple R1 configuration design, a stronger focus will be placed on system optimization. A special focus will be on TPS and TPS-structure integration, as primary structure and TPS are the primary dry mass drivers. 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 cost. 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 the 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 [3]. 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. 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.

2.4. Aurora-AB1 Design Perspectives

Different propulsion options are available for the air-breathing branch of Aurora, whereas typically combined cycle engines that integrate different propulsion types in one engine are considered in order to save mass and reduce engine dimensions. Such combined cycle engines may be grouped into turbine based combined cycles (TBCC) or rocket based combined cycles (RBCC). For the first air-breathing configuration AB-1 a TBCC cycle will be investigated. This option will consist of two main components: a turbojet 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 air-breathing mode has to be shut down and a rocket motor has to take over. For Aurora-AB1 no vehicle design is available so far, as it is expected that engine geometry, integration and flow path requirements will largely define the vehicle geometry. Thus, this sub-section will focus on the discussion of the propulsion system rather than on the vehicle design itself.

A preliminary sketch of the TBCC based propulsion system for AB-1 is shown in FIGURE 4. To keep the required space as compact as possible for this configuration an “in-line” arrangement has been chosen which requires a single air flow path per engine. For the sake of modularity, each engine should be placed in an individual compartment which houses a separate variable geometry air intake with isolator duct, the turbojet engine, the after burner or Ramjet, and an adjustable thrust nozzle. Critical to the overall engine design are the air intakes since they have to provide a high total pressure recovery in combination with low drag especially during the transonic regime. Further, at high flight Mach numbers the physical stability of the air intake also becomes more important. For this purpose, two-dimensional air intakes have been chosen instead of axisymmetric ones. While an axisymmetric air intake has the advantage of a light weight design in combination with no or minor sealing

problems, it must be doubted that it can provide the required performance (total pressure recovery, drag, stability) along the complete flight path. Especially since its only measure to adapt to the actual flight conditions is to shift a central compression cone backward and forth in order to adjust the oblique compression wave pattern to the flight Mach number.

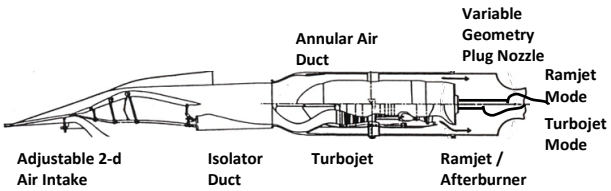

FIGURE 4. Sketch of TBCC engine for Aurora-AB1

A two dimensional air intake with variable ramps can compensate some of the shortcomings of a axisymmetric cone since it consists of two to three compression ramps of which each one`s slope could be varied individually according to the actual flight state. Drawbacks however include higher actuator masses and sealing issues between ramps and side walls. The turbojet will be surrounded by an annular air duct which will act as a bypass system during the turbojet operational regime. With increasing flight Mach number, first an afterburner will be ignited operating concurrently with the turbojet. With transition to pure Ramjet operation, the turbojet will be shrouded by a covering system and the total amount of captured air will be led through the annular air duct into the Ramjet combustion chamber. In order to extract maximum thrust generation from the Ramjet, a nozzle with a variable throat is envisaged. Here, a so called plug or pintle nozzle will be employed that is able to vary the nozzle throat by axial shifting. Bayern Chemie has tested this technology successfully in 2015 for combustion chamber conditions almost identical to the ones relevant here.

In contrast to the wing mounted engine installation approach of SABRE/SKYLON [4], for the AB1 version of Aurora the engine compartments will probably be located on the dorsal or leeward side of the wings, leading to a highly integrated vehicle/propulsion sub system configuration. This choice has been made for limiting lift generation that could otherwise become unnecessary high, associated trimming considerations, as well as volumetric/integration considerations. In fact, this unconventional arrangement might bear some advantages over the common approach. But this has to be investigated more meticulous and in detail in a forthcoming system study.

A second variant of a possible air-breathing combined cycle engine could be seen as a derivative or modification of the SABRE concept of Reaction Engines [4]. Here, the rocket part of the SABRE is used such that a part of the air captured by the air intake is diverted into the heat exchanger/compressor/rocket motor cycle of the SABRE while the main part of the captured air is led through the main duct. During the first trajectory phase, the engine acts as an ejector rocket. After having reached a sufficiently high flight Mach number, the rocket motor is shut down and the engine operates in pure Ramjet mode. This RBCC-concept could be an option for an Aurora AB-2 configuration, with a preliminary sketch being shown in FIGURE 5.

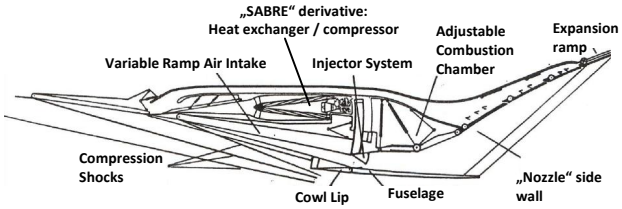

FIGURE 5. Sketch of potential RBCC engine

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 provide low 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 FIGURE 6. The fundamental operating principle is to use a coolant that transports the heat.

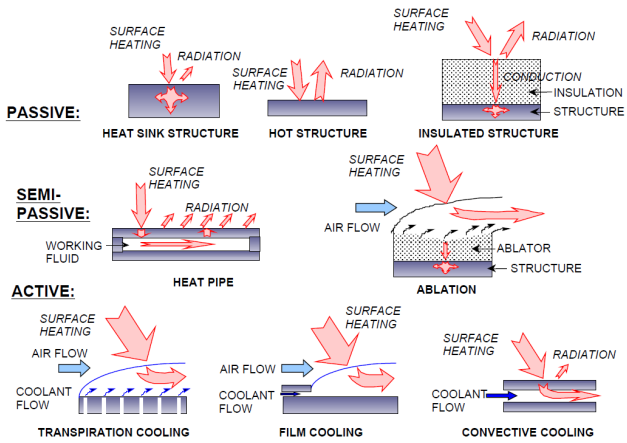

FIGURE 6. Types of thermal protection systems [5]

3. THIN PLY COMPOSITES

After discussing the Aurora study background and the first vehicle configurations, this chapter will now provide a brief overview of the thin-ply technology and associated research results.

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 microcrack initiation strength, typically using altered or added material constituents. Several drawbacks might however occur like the need for specialized material combinations, lowered fiber 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 fiber architecture to thin-ply laminae, while keeping the material constituents unaltered as seen in FIGURE 7.

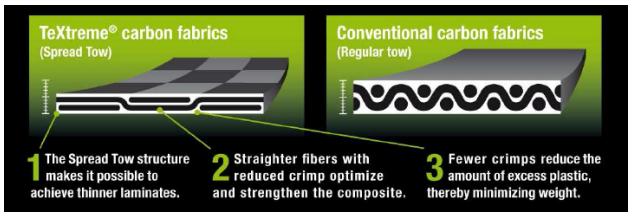

FIGURE 7. Spread tow TeXtreme T700 fabric compared to a conventional fabric [6]

Thin-ply composites are a generic material type which can be expected to give benefits for most fiber- and matrix combinations according to the schematic picture seen in FIGURE 8.

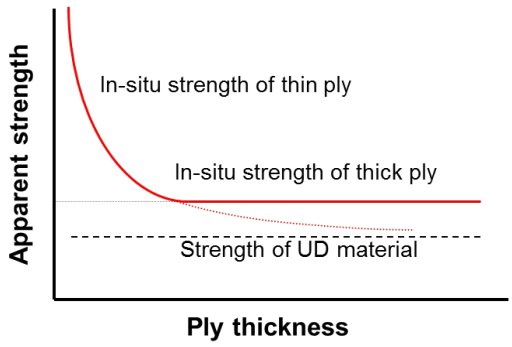

FIGURE 8. General effect for apparent mechanical micro crack strength versus laminae thickness

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 FIGURE 9. 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.

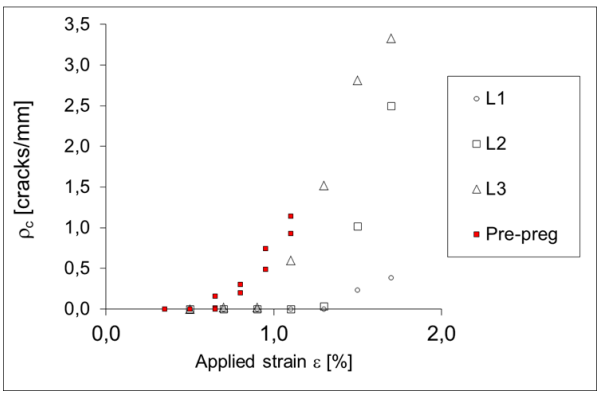

FIGURE 9. Crack density vs applied tensional strain at -50°C for 0°/90° lay-up

The Swedish Oxeon company pioneered the spread tow thin-ply carbon fiber material in 2003 [6]. 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 [7]. It is likely that some of these new applications could not have been realized 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 [8]. 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” [9]. 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 [9], [10]:

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

3.1. Results of the CHATT 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) [11], [12]. 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 FIGURE 10. No cracks could be found in the TeXtreme® laminae while cracks were found in the roving laminae.

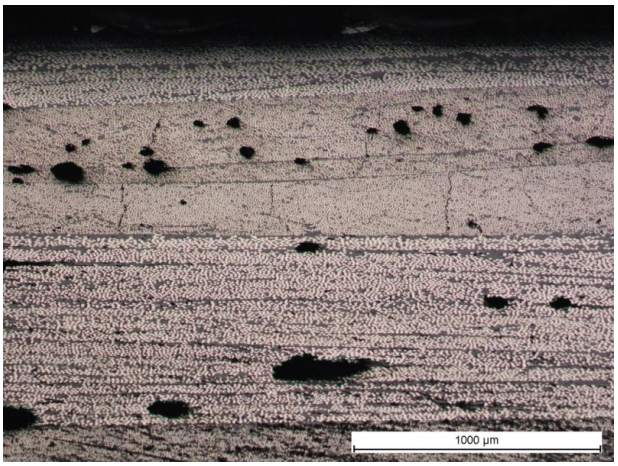

FIGURE 10. Laminate view in tangential direction of the test section

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.

4. VEHICLE STRUCTURAL DESIGN USING THIN-PLY COMPOSITES

In this section thin-ply structural mass calculations will be presented for the Aurora R1 and the R2 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 3, 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.

4.1. Aurora-R1 Structural Analysis

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 FIGURE 11.

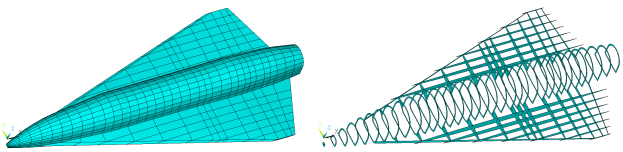

FIGURE 11. Aurora-R1 ANSYS geometry model; full model left), skins removed (right)

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 [13] 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, whereas the actual maximum acceleration in the current trajectory simulation is 4.7 g
- LC2: n_z = 2.0 g normal acceleration maneuver 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 maneuver has been considered and hypersonic pressure distributions may not necessarily be as demanding as subsonic pressure distributions [14]. 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 [15]. 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 analyzed. 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 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.

Vehicle structures made of 4 different materials have been considered: aluminum-lithium 2195 that had also been used for the Space Shuttle Super Lightweight Tank (SLWT) [16], and which is used here as a benchmark, titanium alloy Ti6Al4V in the heat-treated configuration, and two different CFRP composites. IM7/PETI-5 is a polyimide based high temperature composite with material data provided in [17]. 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 [18]. 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. IM7/PETI-5 and titanium have been simulated up to 500 K and 600 K, respectively. 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.

FIGURE 12 shows computed vehicle structural masses as well as the TPS mass as a function of structural skin temperature. Thereby, the structural 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 aluminum and titanium vehicle structures feature a relatively strong increase with increasing temperature in particular due to thermal stress build-up. Titanium is not competitive which is largely a result of the high number of vehicle components that are sized by the 0.5 mm minimum thicknesses criterion, which in turn penalties high density materials. 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 ply number considerations (at least 8 plies per facesheet). 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 fiber orientation as well as transverse tensile strength.

In this analysis an IM7/APC-2 structure is providing the lowest airframe weight, whereas IM7/PETI-5 is coming close at higher temperatures when considering the sum of structure and TPS mass.

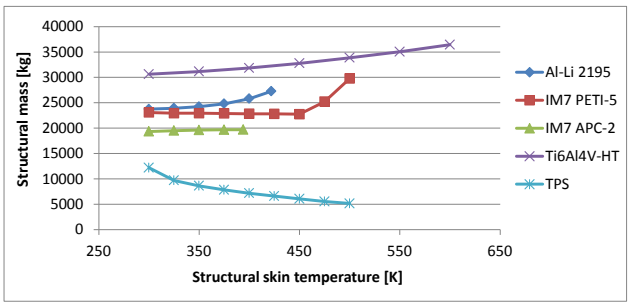

FIGURE 12. Structural masses and TPS masses for different allowed structural skin temperatures

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 FIGURE 13. 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 mm 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 FIGURE 12 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 FIGURE 13 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 0.25 mm 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 FIGURE 13. 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.

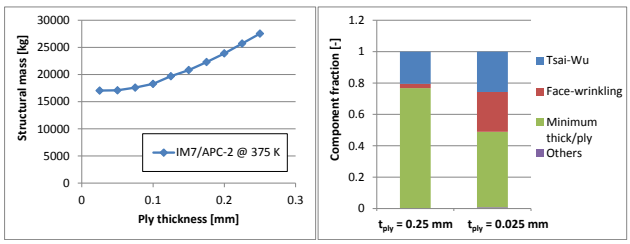

FIGURE 13. Computed structural masses for different ply thicknesses (left); sizing criteria for two selected ply thicknesses (right)

FIGURE 14 investigates the impact of material strength increase. The thin ply effect can lead to an increase in the material transverse and shear strength (see FIGURE 8), while the strength parallel to the fibers 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, FIGURE 14 shows the structural mass savings for the time being for generic strength increases of 10% to 50%, actually 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 FIGURE 14) illustrate that a structural mass reduction of 6.2% could be reached when increasing transverse and shear strengths by 50%. The right part of FIGURE 14 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.

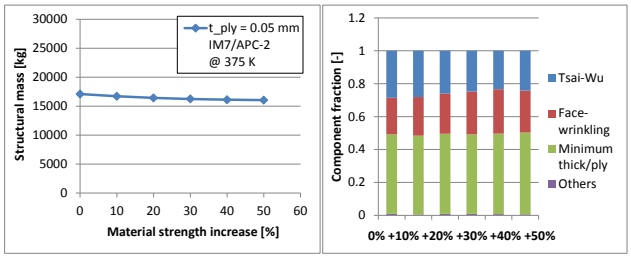

FIGURE 14. Computed structural masses as a function of material transverse and shear strength increase (left); sizing criteria (right)

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 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%.

4.2. Aurora-R2 Structural Analysis

Although the R2 configuration is still in an early vehicle design process, preliminary structural analyses have already been performed to investigate thin-ply mass savings for a second study vehicle. The boundary conditions and load cases are similar to those of the R1 configuration as described previously. Although the

trajectory profile is slightly different, the maximum trajectory loads are comparable. The left part of FIGURE 15 shows the computed structural masses as a function of structural skin temperature for IM7/APC-2 (0.125 mm ply thickness) and aluminum-lithium 2195. Interestingly, both materials yield similar structural masses with 2195 providing even slightly lower masses for low temperatures. This is in contrast to the results for R1 shown in FIGURE 12 before. The reason is that due to the efficient inertia-force/lift-force matching and the compact configuration the stresses in the sandwich facesheets are very low. As a result, the majority of the structural components are sized by minimum thickness/ply-number criteria. In case of the composite facesheet this yields a minimum facesheet thickness of 1 mm for 0.125 mm ply thickness, and only 0.5 mm for aluminum.

The right hand side of FIGURE 15 investigates the effect of ply thickness reduction induced more efficient material utilization. The mass benefit is here even higher than in the case of the R1 configuration. The lowest ply thickness of 0.025 mm allows for a structural mass saving of an impressive 55.3% with respect to 0.25 mm. At this point however the question emerges whether such low structural masses can actually be achieved in a real design, or if secondary load conditions that are not covered in this first order analysis are becoming more prominent for very low mass structures. When comparing a ply thickness of 0.05 mm with the baseline ply thickness of 0.125 mm, the mass saving is still 25.5%. The effect of theoretic transverse and shear strengths increases from 10% to 50% was investigated as well (not shown here), whereas the vehicle structure with 0.05 mm ply thickness was selected. The mass saving however was only 2.7% for the highest considered strength increase of 50%, which is no surprise given that only a few number of facesheets have been sized by strength considerations.

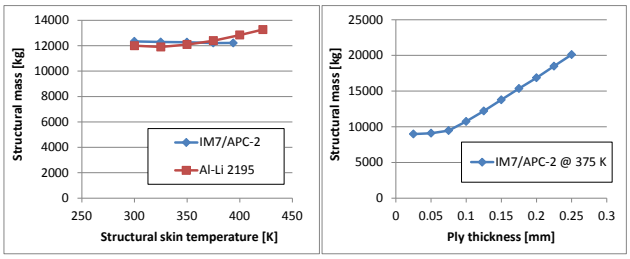

FIGURE 15. Structural masses for different allowed structural skin temperatures (left); computed structural masses for different ply thicknesses (right)

4.3. Structural Analysis Summary

The structural analysis results for Aurora R1 and R2 indicated that large structural mass savings are possible when using thin ply composites. This is mainly due to the fact that a large fraction of the vehicle components are sized by minimum ply number considerations, where thinner plies provide an obvious advantage. The potential mass saving due to material strength increase was found to be of limited magnitude. However, as for these particular vehicle configurations only a comparable small number of facesheets are sized by strength criteria, it is obvious that corresponding mass savings are small.

Care has to be taken when generalizing or extrapolating these results to other hypersonic vehicles. Highly loaded vehicle structures operating under high stresses and with comparatively thick skins will not profit from thin-ply induced more efficient material utilization in the same magnitude as Aurora R1 and R2. In such a case however the benefit of increased material strengths might be much higher than found here. Further mass savings might be possible for many structures if the structure and material architecture would be optimized for thin-plies.

The structural analyses have furthermore demonstrated that compact configurations with high geometric moments of inertia (high fuselage diameters and/or wing thicknesses) and mass distributions for efficient mass/lift matching are a very promising strategy for enabling extremely lightweight vehicle structures. Air-breathing configurations are here slightly penalized as slender and therefore less compact configurations are required for high aerodynamic performance.

5. SUMMARY AND CONCLUSION

This paper introduced the Aurora space launcher system design study and its background. The current design status and/or the design perspectives for three vehicle configurations have been discussed. Afterwards, a brief overview of the thin-ply CFRP technology was provided, and the application of this technology for two Aurora configurations was discussed. 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 indicate that structural mass savings in the order of ~20% compared to conventional CFRP appear to be realistic. Even higher mass savings were found for the R2 flying wing configuration. Future investigations will utilize more sophisticated analysis procedures to quantify the actual mass saving potential with a higher accuracy and reliability. 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 in 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.

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