CONCEPTUAL DESIGN OF THE SPACELINER THERMAL PROTECTION SYSTEM

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ABSTRACT

With the SpaceLiner the DLR has proposed a visionary concept for hypersonic suborbital passenger transport over extremely long distances [1]. Depending on the configuration or mission type, the maximum flight Mach numbers of the vehicle can exceed Mach 20, hence the consideration of aerodynamic heating becomes mandatory during the design process.
The paper addresses the aerothermal challenges of the SpaceLiner flight and the latest updates in the conceptual design of the thermal protection system (TPS). Due to the very high heat loads, an active TPS is required for the nose and the leading edge whereas a passive, radiatively cooled TPS is sufficient for the downstream surface regions. Potential solutions for both, active and passive thermal protection are developed and presented. Mass and systems requirements as well as an economically justifiable effort for manufacturing and maintenance are constraints which are also taken into account during the whole design process.

1. INTRODUCTION

Aerothermodynamic design drivers are crucial for the development of future RLV and hypersonic suborbital systems.
Different options exist to protect the vehicle structure against the very high heat flows. The outer shape of the vehicle, particularly in the stagnation regions, has a significant impact on the heat loads. Hence a reduction of heat loads could be achieved by shape geometry modifications. Another option is the use of high temperature materials for the vehicle structure (i.e. hot structures) which can be applied up to a maximum temperature, dependent on the material. An alternative solution is the cold structure approach. A passive thermal protection system is combined with a back structure which is usually made of low temperature materials (e.g. aluminium alloys). If the heat flows and the surface temperatures exceed the capacity of hot structure materials or even of passive thermal protection materials, ablative heat shields or active cooling systems can be applied. Ablative systems are certainly not the optimum choice in terms of the mass balance, the continuous shape change, the contamination of the boundary layer and the radiant heat of the ablation products. If reusability is a fundamental requirement, active cooling remains the only option for high temperature applications, since an ablative system must be replaced after each flight. Finally the choice of an adequate approach strongly depends on the systems requirements of the vehicle.
Future hypersonic transport concepts such as the SpaceLiner must be designed with maximum aerodynamic performance to be operationally efficient [2]. This means a high lift-to-drag ratio (L/D) and therefore low drag which necessitates rather sharp and slender geometries with small nose and leading edge radii. On the other hand the maximum flight Mach numbers of the SpaceLiner orbiter stage could reach 24, depending on the configuration or mission type. The high Mach numbers during the nominal mission from Australia to Europe in combination with the sharp and slender geometry can result in maximum radiation adiabatic stagnation temperatures of almost 2600K and heat fluxes of more than 2MW/m^2 (Figure 1).

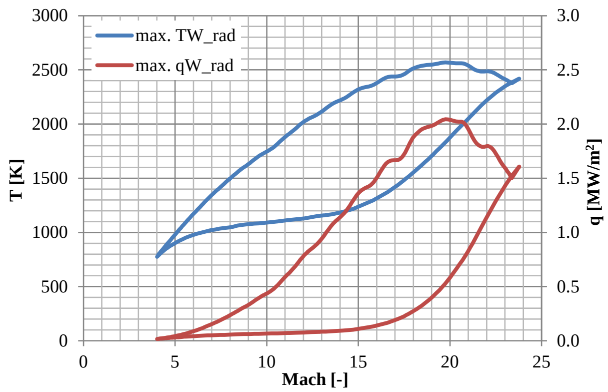

Figure 1. Maximum radiation adiabatic wall temperature and heat flux as a function of the Mach number for the SpaceLiner reference mission.

Because these thermal loads exceed the range of use of passive TPS materials, an additional active cooling system is required to reduce the maximum surface temperatures in the stagnation regions such as the nose and the wing leading edges. Different cooling system architectures, materials and coolants have already been investigated in the past and this knowledge can be transferred to the SpaceLiner reference vehicle. Each method has its specific advantages and drawbacks and must be assessed carefully with respect to the requirements of the SpaceLiner, which are mainly related to safety, reliability and low system mass. Therefore preliminary system designs of the most promising concepts are elaborated.
During the FAST20XX project an innovative method of

transpiration cooling using liquid water as a coolant was investigated for the SpaceLiner at the DLR arc-heated wind tunnels in Cologne [3]. A very high cooling efficiency, caused by the high evaporation enthalpy of water, was experimentally proven. Hence this option was chosen as a first reference concept for the SpaceLiner active cooling system.

However, besides the evident advantages of transpiration cooling with liquids, fundamental issues and challenges have already been identified and must be assessed carefully with respect to the specific requirements of the SpaceLiner.

Downwards the stagnation regions the temperatures are lower and a passive TPS is sufficient. The chosen approach is a trade-off between cold and hot structure (“warm structure approach”).

The latest results of the SpaceLiner active and passive TPS studies are presented within this paper.

2. METHODS AND TOOLS

The SpaceLiner TPS has to withstand the heat loads according to nominal flight and different abort cases. To be able to determine the heat loads for a full vehicle along different trajectories, fast engineering methods are used. The external heat flux is calculated by HOTSOSE, a fast code for preliminary flow analyses in hypersonic flow. Dependent on the particular geometry, Newtonian, modified Newtonian or Shock-Expansion Theory is applied to approximately determine the pressure distribution along the vehicle surface. In addition, HOTSOSE provides the option of approximately considering the influence of viscous effects either for ideal gas or in case of thermodynamic equilibrium flow assuming an isothermal or radiation adiabatic wall. The corresponding parameters such as radiation adiabatic wall temperature, surface heat flow and skin friction coefficients are calculated by established engineering methods. The implemented methods are well proven for a variety of vehicle shapes in hypersonic flow conditions [4,5].

For the passive TPS, different reusable materials were considered, depending on the maximum temperature for subdivided surface regions: CMC (Ceramic Matrix Composite), TABI (Tailorable Advanced Blanket Insulation), AETB (Alumina Enhanced Thermal Barrier Tiles with TUFI Coating), AFRSI (Advanced Flexible Reusable Surface), FRS (Felt Reusable Surface Insulation) and a metallic TPS, which could be an alternative to TABI in the temperature range of 900-1400K. Detailed material information can be found in [6].

The thicknesses of the different materials were optimized by using a 1D thermal conduction model [7].

The determination of the required coolant mass for the active TPS regions is conducted by a simple tool which is based on the HOTSOSE results. The tool identifies critical temperature areas on the vehicle surface and calculates the coolant massflow which is required to cool down this area to a certain predefined objective temperature, dependent on the local heat flow and the enthalpy difference of the coolant.

3. PASSIVE THERMAL PROTECTION

3.1. Passenger Stage

The passenger stage TPS is required to be dimensioned for the heat loads during the nominal mission, as well as for different types of flight abort scenarios in case of emergency.

The capsule’s upper half is part of the orbiter’s outer shell. Hence, it has a requirement of a maximum inner temperature of less than 303K in order to ensure the passenger’s comfort. This part is considered in section 3.3. The rest of the orbiter is dimensioned for a higher maximum structure temperature, which depends on the chosen structure material. If an aluminium (e.g. Al 2024-T3) structure is used, the structure could heat up to 400 K. By changing to titanium or the polymer PEEK (Polyether Ether Ketone), the maximum allowed temperature could be increased to 530 K [8].

The total TPS mass of 22.4t, which would result from an aluminium structure (T_{max}=400K), can be reduced by 22.3% to 17.4t for a structure temperature of 480K and by 34.4 % to 14.7t for a structure temperature of 530K.

Hence the challenge is to find an optimum trade-off between structure mass and TPS mass. Kopp and Garbers already addressed this issue on a preliminary level in [9].

Additionally, different flight durations were simulated to consider the impact of the amount of flight time on the TPS mass. Here, flight time may also include the time of de-boarding and additional buffer times, in which the maximum structure temperature should also not be exceeded. Three different scenarios were simulated: flight time until landing, until landing plus 300s and until landing plus 600s.

Table 1 gives an overview of the results of the different simulations.

Table 1. Total TPS mass (passenger stage w/o fin & w/o passenger cabin, using TABI, AFRSI, AETB and CMC)

Temp. [K]	Mass [t]		
	until landing	until landing +300s	until landing +600s
400 K	22.4	22.8	24.6
480 K	17.4	17.6	17.8
530 K	14.7	15.3	15.5

In the case of a structure temperature of 400K, the total mass of 22.4t (simulation until landing) increases to 22.8t (+1.79%) for simulation until landing plus 300s and to 24.6t (+9.8%) for simulation until landing plus 600s.

A detailed mass budget for the TPS layers in the different temperature ranges can be found in Table 2. The data is given for the “worst” case in terms of mass

(T_{max}=400K, simulation until landing plus 600s).

Table 2. TPS mass break down (passenger stage w/o part of passenger rescue cabin, structure temperature 400K, simulation time: landing plus 600s)

Temp. [K]	Material	Area [m²]	Total Thickness [m]	Surface Density [kg/m2]	Mass [kg]
<400	No TPS required				
401 – 600	FRSI	851	0.0472	5.6	4789
601 – 700	AFRSI	250	0.0547	6.5	1625
701 – 800	AFRSI	409	0.0607	7.0	2863
801 – 900	AFRSI	109	0.0653	7.4	807
901 – 1000	TABI	85	0.0794	8.8	748
1001 – 1100	TABI	43	0.0826	9.1	391
1101 – 1200	TABI	117	0.0857	9.4	1104
1201 –1300	TABI	164	0.0888	9.7	1594
1301 – 1400	TABI	550	0.0910	9.9	5469
1401 – 1500	AETB_8	227	0.1131	15.2	3448
1501 – 1600	AETB_8	68	0.1150	15.4	1049
1601 – 1700	CMC	17	0.3098	22.0	378
1701 – 1850	CMC	10	0.3173	22.3	226
1850 – 1950	CMC	6.3	0.2604	20.3	129
Sum					**24620**

The high thickness and mass of the SpaceLiner TPS in comparison to the Space Shuttle TPS is not surprising since the thermal loads are even beyond those of the Space Shuttle orbiter at re-entry.
As an alternative to TABI, a metallic TPS was considered. It consists of mainly two layers, a metallic skin with stand-offs and a thermal insulation and has been developed for a reusable re-entry vehicle by Fatemi et al. [10]. Assuming a maximum inner temperature of 400K and no additional time for de-boarding etc., a metallic TPS would have an additional mass of approx. 5.9t in comparison to the use of TABI, even under optimistic assumtions. But metallic TPS is more damage tolerant and needs less maintenance in comparison to ceramic materials. This may be more important than mass aspects to achieve the rapid turnaround and low life-cycle costs required for the SpaceLiner concept.

3.2. Booster

For the booster stage the nominal ascent and re-entry trajectory has to be considered. Abort cases and off-nominal manoeuvers have not been taken into account yet. The maximum heat loads can be found at the nose area and the underside of the wing leading edges. The TPS material has to be fully reusable. It is to be dimensioned in a way that the internal structure temperature does not exceed 400K because an aluminium Al2219 substructure is assumed in the current design.
There is no need of TPS in the rear of the upper wing surface (Figure 2). Most of the remaining part of the upper half is exposed to a maximum temperature of 600K. For this area, FRSI is chosen which has a thickness between 1.16 and 1.94cm, depending on the location at the surface. The lower half becomes much hotter: the underside of the wing edges reaches up to 1930K. From 1600K to 1930K, CMC was chosen. With a thickness of 13.16 and 13.41cm, it is much thicker than the AETB-8, which is used for 1400 – 1600K (4.84cm).
For the average temperature range of 900 - 1400 K TABI was selected (3.35 -3.79 cm).

Table 3. TPS mass break down (booster stage, structure temperature 400K, simulation time: landing plus 600s)

Temp. [K]	Material	Area [m²]	Total Thickness [m]	Surface Density [kg/m2]	Mass [kg]
<400	No TPS required				
401 – 500	FRSI	1167	0.0116	2.20	2571
501 – 600	FRSI	425	0.0194	2.95	1254
601 – 700	AFRSI	352	0.0228	3.40	1197
701 – 800	AFRSI	254	0.0249	3.59	913
801 – 900	AFRSI	213	0.0267	3.77	804
901 – 1000	TABI	253	0.0335	4.41	1118
1001 – 1100	TABI	356	0.0351	4.56	1625
1101 – 1200	TABI	68	0.0363	4.68	319
1201 –1300	TABI	238	0.0376	4.80	1141
1301 – 1400	TABI	149	0.0379	4.83	722
1401 – 1500	AETB_8	43	0.0475	6.80	292
1501 – 1600	AETB_8	17	0.0484	6.92	116
1601 – 1700	CMC	114	0.1316	15.80	1797
1701 – 1930	CMC	0.48	0.1341	15.88	7
Sum					**13878**

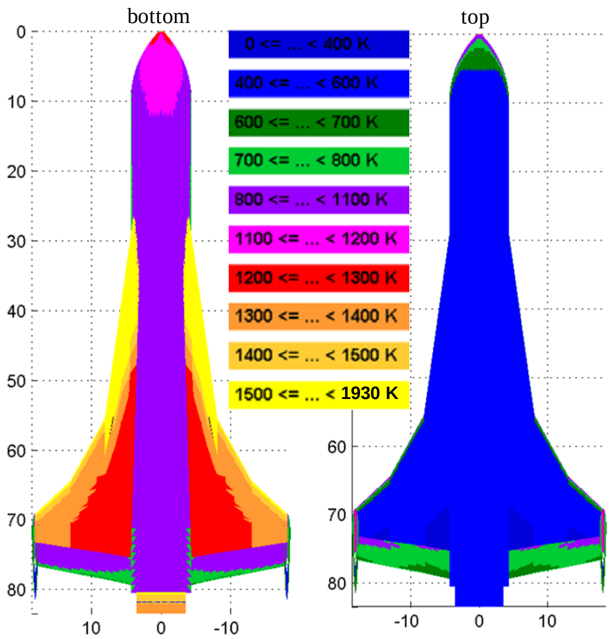

Figure 2. Temperature regions for SpaceLiner booster

3.3. Capsule

To guarantee passenger safety and comfort, the temperature inside the passenger capsule should not exceed habitable room temperature.
The results of the passive capsule TPS optimization are shown in Table 4 and the surface temperature regions are shown in Figure 3.

The Capsule's lower half and nose are protected by the orbiter structure and TPS during the nominal mission. They are therefore not subjected to the external heat load until the capsule is separated in case of life-threatening emergency.
In contrast, the capsule's upper half is part of the orbiter's outer shell and so is heated up during nominal flight. These differences lead to different starting conditions regarding initial temperature after abort separation. The upper half of the TPS is required to be dimensioned for the nominal mission Australia to Western Europe.
The high heat flux and ability to use a non-reusable TPS for the nose (as it is only required once during an emergency abort scenario) allows for the use of an ablative thermal protection system. The low system complexity of an ablative TPS helps to guarantee high safety and reliability. The chosen material is Avcoat 5026-39/HC-6, which has already been used on other spacecraft such as the Apollo capsule [11]. This TPS has a mass of 1347 kg with thickness of 13.3cm.

*Table 4. TPS mass break down, orbiter parts are denoted with * (Passenger capsule, inner temperature: 303K, simulation time: landing plus 300s)*

Temp. [K]	Material	Area [m²]	Total Thickness [m]	Surface Density [kg/m2]	Mass [kg]
700 – 800*	AETB	63.21	0.1419	18.9	1194
800 – 1000*	AETB	58.58	0.1469	19.5	1144
1001 – 1100	AETB	10.00	0.0694	9.6	96
1101 – 1200	AETB	24.22	0.0706	9.8	236
1201 – 1300	AETB	18.11	0.0719	9.9	179
1301 – 1400	AETB	12.01	0.0719	9.9	119
1401 – 1500	AETB	8.45	0.0719	9.9	84
1501 – 1600	CMC	42.03	0.1910	17.8	751
1601 – 1850	CMC	10.95	0.1910	17.8	195
Sum					**3998**

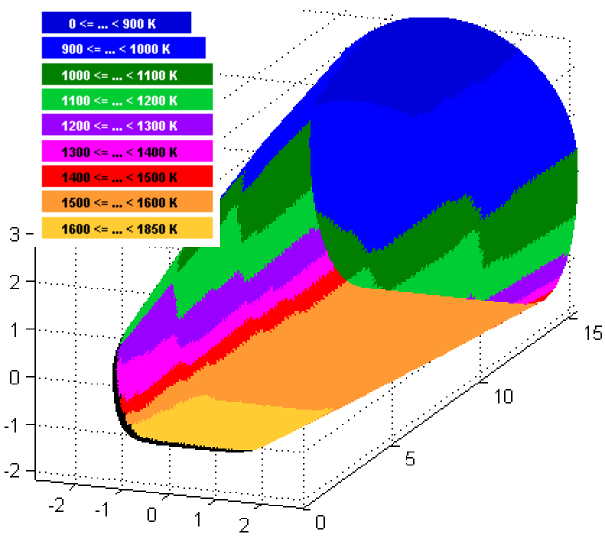

Figure 3. Temperature areas, passenger cabin

For the rest of the capsule, reusable TPS is used. The bottom side has to be only dimensioned for abort cases including a capsule separation. Therefore, it has a thinner dimension in comparison to areas exposed to the heat load during nominal flight – although it experiences higher temperatures. Details can be found in [12].
Note that however, due to limited space in the orbiter or capsule, the lightest TPS version might not fit within the available space and a thinner but heavier TPS material combination might be necessary. This issue must be addressed in the future.

4. ACTIVE COOLING

Active cooling is only required for the SpaceLiner passenger stage as it suffers the highest heat loads during the mission. The requirements and boundary conditions for the active cooling system were already described in [2]. This section focuses on the latest mission updates in terms of heat loads and required coolant masses.
For the conceptual design of the SpaceLiner active TPS the regions above a certain limiting temperature T_{lim} must be identified. T_{lim} is derived from the upper temperature limit of the material. In a previous approach CMC materials were chosen up to T_{lim}=1850K. All surface regions above this temperature must be actively cooled. They are usually limited to the stagnation areas such as the nose and the wing leading edge as exemplarily represented in Figure 4.

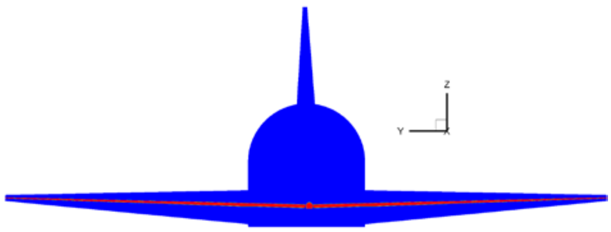

Figure 4. Visualization of thermally critical surface regions at the vehicle nose and leading edge

For a fixed geometry the size of these areas depends on the flown trajectory and mission parameters. The most ambitious and critical mission is the reference mission from Australia to Europe. Hence, the active cooling system must be designed to manage the heat loads for this reference mission. Figure 5 shows the vehicle velocity and flight altitude as a function of the flight time. Active cooling is required between t=376s and t=2918s because surface temperatures above 1850K are detected within this critical period.
The maximum occurring surface temperatures T_{max} and the area A_{cool} which must be actively cooled within the critical period are plotted as a function of the time in Figure 6. It must be noted that the flown angle of attack (AoA) has a significant influence on the size and the position of A_{cool}. E.g. the rapid drop in A_{cool} and the drop in T_{max} around t=435s are caused by a sudden reduction of the angle of attack of the passenger stage after MECO.

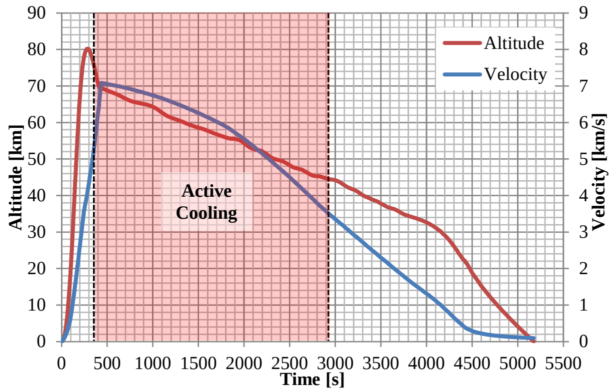

Figure 5. Altitude, velocity and active cooling period as a function of the flight time for the reference mission

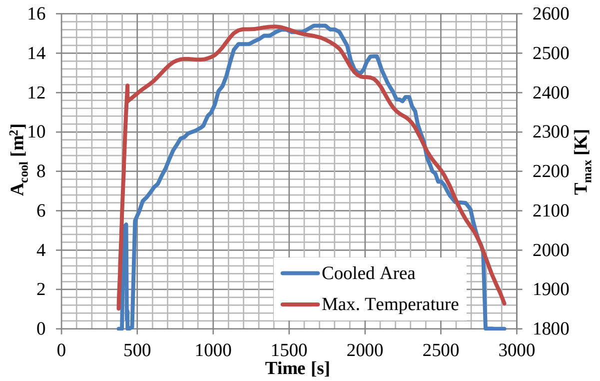

Figure 6. Actively cooled area A_{cool} and maximum surface temperature T_{max} as a function of the flight time

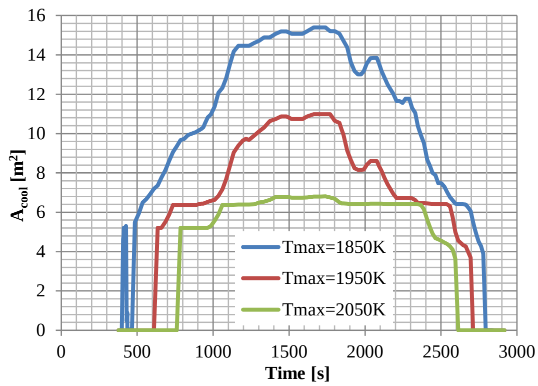

Figure 7. Actively cooled area as a function of the flight time and T_{max}

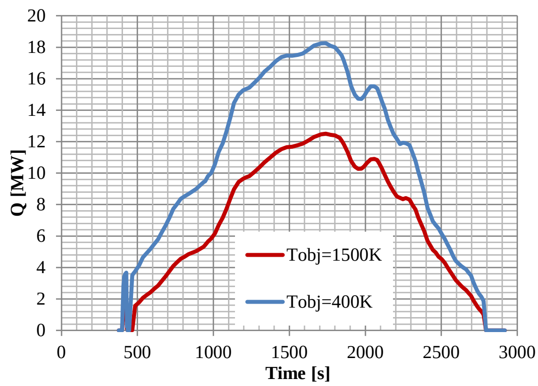

Figure 8. Heat flow Q in the critical region A_{cool} as a function of the flight time and T_{obj}

For this calculation the surface area A_{cool} which eventually requires active cooling during the full mission is 15.4m^2. The area could be reduced by accepting higher temperatures for the passive TPS material. The impact of the maximum allowed passive TPS temperature on A_{cool} was already investigated in the framework of the DLR internal THERMAS project [2] and is presented here for the latest SpaceLiner trajectory updates (Figure 7). A_{cool} could be reduced to 11m^2 for T_{max}=1950K and to 6.8m^2 for T_{max}=2050K.

The total amount of heat which must be absorbed respectively carried away by the coolant in the active cooling system does not only depend on the surface area A_{cool} but also on the surface temperature difference which should be achieved by the cooling system. The objective cooling temperature T_{obj} to which the surface should be cooled down has a major impact on the amount of heat which must be managed by the system. Decreasing values of T_{obj} result in increasing heat flows. This correlation is shown in Figure 8 for T_{max}=1850K and two different values of T_{obj}. The total amount of heat Q_{tot} which must be managed by the cooling system can be calculated by integration of the heat flow along the mission time. Q_{tot} is more than 27GJ for T_{obj}=400K, for T_{obj}=1500K it is only 17.6GJ.

Preliminary investigations have already proven that a closed loop cooling system cannot be applied to the SpaceLiner since there is no viable option to release the large amount of heat from the coolant during the flight. Hence, the most reliable solution is to dump the coolant. Therefore Q_{tot} is also relevant for the determination of total required coolant mass whereas the performance and the power of the system are affected by the maximum heat flux peaks on the vehicle surface during the mission.

The current reference design for the SpaceLiner active cooling system is based on active transpiration cooling with liquid water as a coolant. This approach was investigated at the DLR Cologne [3,13].

A liquid water mass flow is piped through cooling channels below a porous wall. Forced by capillary action, a certain amount of water passes through the porous material, absorbs the heat from the wall material and vaporizes (Figure 9a). The vapour film along the surface also reduces the incoming heat from the external flow into the wall (blocking effect, Figure 9b).

Different sample geometries and materials were tested in the arc heated wind tunnel facility at DLR Cologne.

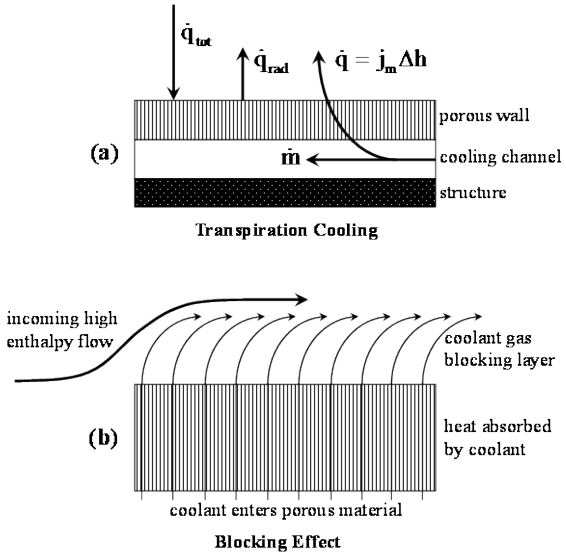

Figure 9. Transpiration cooling and blocking effect [2]

In preparation of FAST20XX, three different nose cone models, made of Procelit 170 (91% Al_2O_3, 9% SiO_2) were tested (Figure 10). This material was chosen because of its high porosity and its ability to withstand temperatures of up to 2000 K.

Figure 10. Procelit 170 wind tunnel models [14]

First, liquid water was used as a coolant and the wall temperature drops were observed for a water mass flow of 0.2g/s. A comparison with gaseous N_2 as a coolant was conducted under the same test conditions. A huge increase of cooling efficiency was observed when using liquid water instead of N_2 gas, which is caused by the high evaporation enthalpy of water. Stagnation temperature drops of more than 1500K were achieved. Even for a five times higher N_2 gas mass flow of 1g/s the maximum reduction was only 850K.

Due to these promising results the studies were continued in FAST20XX. A weak point of Al_2O_3 is the low tensile and bending strength which makes it not well qualified for a structural material. Therefore, three other materials were tested in FAST20XX, C/C, C/C-SiC and AVA-Z-P50 (Al_2O_3 fiber/matrix composite). Flat-faced cylinder samples are shown in Figure 11. Additionally, blunt cone samples were tested.

In contrast to the high cooling efficiency potential, problems were also detected during the test campaign. Dependent on the material and the fibre orientation, steady state conditions could not be reached under the given conditions.

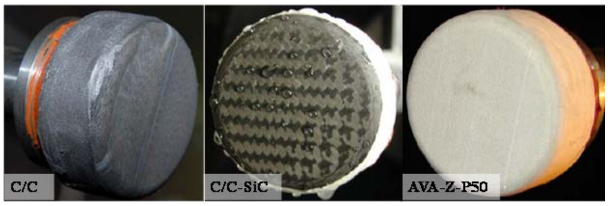

Figure 11. Sample wind tunnel models, FAST20XX [3]

This resulted either in an onset of boiling in the coolant reservoir behind the sample and therefore in a rapid decrease of cooling performance on the one hand or in the formation of ice along the sample surface on the other hand (Figure 12).

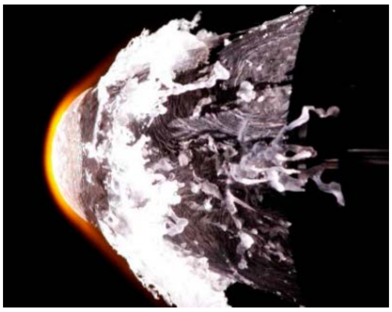

Figure 12. Ice formation during wind tunnel testing [3]

Another, general issue for transpiration cooling is the triggering of the laminar-turbulent boundary layer transition [15]. In the worst case this could cause local hot spots downwards the stagnation regions.

These issues challenge the integration of an active transpiration cooling system with liquid water into an operating system like the SpaceLiner and must therefore be handled in the future. In addition, potentially promising alternatives like internal spray cooling or convective cooling are currently under investigation at DLR [2].

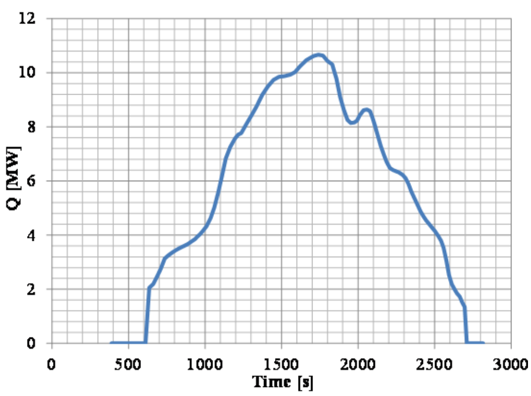

Figure 13. Heat flow Q in the critical region A_{cool} as a function of the flight time (T_{max}=1950K, T_{obj}=1500K)

Applying the transpiration cooling with liquid water to the latest SpaceLiner reference trajectory, an estimation of the required water mass can be performed. The latest system updates in the TERMAS project considered a maximum acceptable wall temperature of 1950K for the passive TPS as a trade-off, which could be achieved e.g. by improved material properties.

Theoretically it would be sufficient to cool down the stagnation regions to 1950K, however, a safety margin is considered and the objective cooling temperature is set to T_{obj}=1500K. The resulting heat flow which must be managed by the cooling system is shown in Figure 13. The water mass flow which must be evaporated to absorb the heat can be estimated by the following equation and is shown in Figure 14.

$$\dot{m}_{H2O} = \frac{\dot{Q}}{h_{H2O,vap} - h_{H2O,liq}}$$

The total water mass can be calculated to m=5572.2kg by integration of the mass flow along the mission time. The fraction of water which is required in total for the vehicle nose is calculated by 31.2kg and is therefore negligibly small in comparison the 5541kg which are required to cool down the wing leading edges.

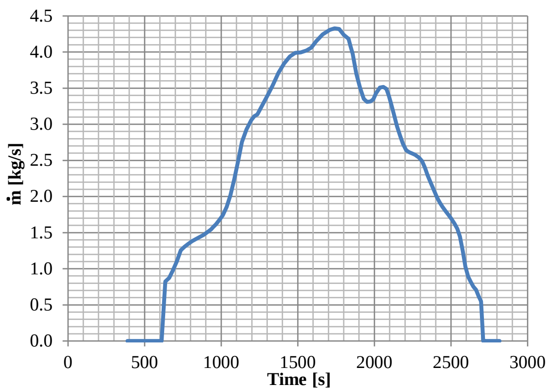

Figure 14. Transpiration cooling water mass flow as a function of the flight time

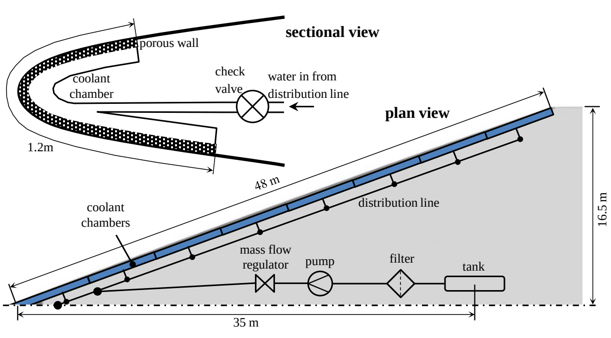

Figure 15. Preliminary design of a transpiration cooling system for Space Liner nose and leading edge [2]

It can be noted that the increase of T_{max} from 1850K to 1950K resulted in an additional passive TPS mass of 129kg. In contrast, the required water mass is reduced by 1572.5kg from 7144.7kg (T_{max}=1850K) to 5572.2kg. These calculations are based on the simplified assumption of an isothermal phase change from liquid to vapor. A schematic systems design approach for active cooling is conducted in [2] (Figure 15).

The updated masses of the cooling system components are estimated via simple engineering methods and given in Table 5. It must be noted that a margin of 20% is included in the mass of the system components and the coolant.

Table 5. Updated estimation of active cooling system components mass, based on [2]

component	Mass [kg]
pipes	146
tanks	200
pumps	212
valves	14
mass flow regulators	10
filters	12
porous wall (C/C-SiC)	880
coolant	6688
Total	**8203**

5. CONCLUSION

The present paper gives an overview of the latest updates of the SpaceLiner thermal protection system.

Passive TPS can be applied up to a maximum radiation adiabatic temperature of 1950K whereas an active cooling system is envisaged for surface areas above this temperature.

The SpaceLiner booster stage does not suffer temperatures above 1950K and can therefore be passively protected. A preliminary design of the passive booster TPS is presented, including a proper choice of materials for the different temperature zones as well as an estimation of the mass budget.

In case of emergency ejection the SpaceLiner capsule can be subjected to temperatures beyond 1950K. However, since reusability is no major design aspect for this emergency system, ablative TPS can be used within the critical regions. A preliminary design and mass budget update is also presented for the capsule.

The SpaceLiner passenger stage is designed for maximum aerodynamic performance and due to the rather sharp and slender shape, radiation adiabatic stagnation temperatures of 2600K can occur. Since the stage should be fully reusable, an active cooling system is the most viable solution to manage these heat loads. A preliminary design update on an active transpiration cooling system with liquid water as a coolant is presented. The total mass of the system is estimated. Furthermore the passive TPS for the passenger stage is optimized. The results in terms of mass, materials and surface zones are presented in this paper.

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