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Technical Background and Challenges of the SpaceLiner Concept

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At the Space Launcher System Analysis (SART) department of DLR-Cologne, a hypersonic spaceplane for passenger transportation is being investigated. The spaceplane is called the "SpaceLiner". The vehicle performs its rocket powered, intercontinental flight via a suborbital trajectory. The paper describes the latest developments and improvements on the design of the SpaceLiner. The aerodynamic heating of the vehicle is discussed, and a possible solution for handling the extreme heatloads will be presented. The solution involves an innovative new way of transpiration cooling, using liquid water.

1. Introduction

For hypersonic aircraft, the airbreathing SCRAM jet is usually seen as a promising option. Although it may be promising, practical implementation is still far from feasible. An alternative is the use of a rocket powered vehicle. An example of such a rocket powered vehicle is the SpaceLiner [1,2,3,7,8]. The SpaceLiner design is made taking into account two main requirements. First of all, it should be able to fly the distance from Sydney to Western Europe, carrying 50 passengers. Secondly, the complete vehicle should be reusable [8]. Other requirements are that acceleration should not exceed 2.5 g in axial direction during ascent and acceleration should not exceed 1.5 g in normal direction during descent and re-enrty.

It consists of two stages, a winged booster stage and a second stage, called the orbiter. The SpaceLiner is designed for vertical take off, much like the Space Shuttle does. There are no solid boosters present, the booster stage and orbiter both use LH2-LOX powered staged combustion engines with moderate chamber pressure. The same engines are used for both stages. With 8 engines for the booster and 2 for the orbiter, the vehicle is able to perform its mission. As long as the orbiter is attached to the booster, cross feed fuelling is foreseen. After separation of the two stages occurs, the booster makes a controlled re-entry and returns to the launch site.

The orbiter then accelerates further and after all the fuel has been used and the remaining part of the flight is powerless. By using a so called 'skip' trajectory, the range covered by powerless flight is greatly improved as compared to a ballistic trajectory. A downside of such a trajectory is the high heat load encountered during a skip. This paper will describe the SpaceLiner concept in more detail and identify the technological challenges of the concept. It will be shown that the high heat load is thought to be the greatest challenge. As a potential solution to this problem a new and innovative transpiration cooling method using liquid water is presented in [1,2,3]. This cooling method has been successfully tested in the L2K arc heated windtunnel at DLR-Cologne [1].

2. Evolution of the SpaceLiner

Since the first introduction of the SpaceLiner [2,7], the design has been subject to adaptations and improvements.

A picture of the first SpaceLiner design, from now on called SpaceLiner 1, is given in Figure 1. Takeoff weight of the complete system of SpaceLiner 1 is estimated at 900 tons [2].

The latest design updates have included a mass estimation of the transpiration cooling subsystem [1]. In addition, a more realistic mass estimation of the passive TPS is made together with an updated estimation of wing structure mass. This resulted in a mass increase of about 10 tons for the dry mass of the orbiter. Tank volume in the orbiter has been decreased, whereas tank volume of the booster has been increased. This was done to achieve more

optimal staging. Aerodynamic performance of the orbiter is increased by changing the geometry of the fin (higher sweep angle) and making the wing somewhat thinner. Finally, nozzle expansion ratios of the engines were optimized for both the booster and the orbiter. The staged combustion engine cycle data is presented in Table 1.
The updated SpaceLiner will be addressed with the number 2 from now on. A picture can be seen in Figure 2, characteristic data can be found in Table 2. A velocity at burnout of 6.55 km/s at an altitude of 75 km would suffice for SpaceLiner 2 to perform the mission, instead of a velocity of 6.7 km/s at 100 km altitude for SpaceLiner 1. At the expense of some additional fuel, the ascent trajectory of SpaceLiner 2 could be made such that the 100 km boundary is passed. This would allow for the passengers to become official astronauts.
The net result of these changes is that the orbiter has become shorter, but nevertheless has a higher dry mass due to increased subsystem mass. The booster has become more voluminous and has a higher takeoff weight and somewhat higher dry mass. The takeoff weight of the complete SpaceLiner configuration has increased from 900 tons to about 1094 tons. A mass breakdown of SpaceLiner 2 is given in Table 2, together with some characteristic dimensions.
Aerodynamic performance of the SpaceLiner is very important. Maximum range depends largely on the glide ratio. The lifting parameter has a big impact on the aerodynamic heating. The lower the lifting parameter is, the lower the aerodynamic heating will be. This is because of the fact that in this case C_L will be relatively high and the vehicle will therefore generate enough lift at higher altitudes where air density is low.
Aerodynamic data is presented in Table 3. Because of the fact that during its flight the SpaceLiner will use cooling water, mass will change. It is estimated that about 9 tons of cooling water will be needed [1]. The aerodynamic properties such as wing load, ballistic coefficient and lifting parameter will therefore change during flight. The table shows these properties in case of completely filled water tanks and empty water tanks.

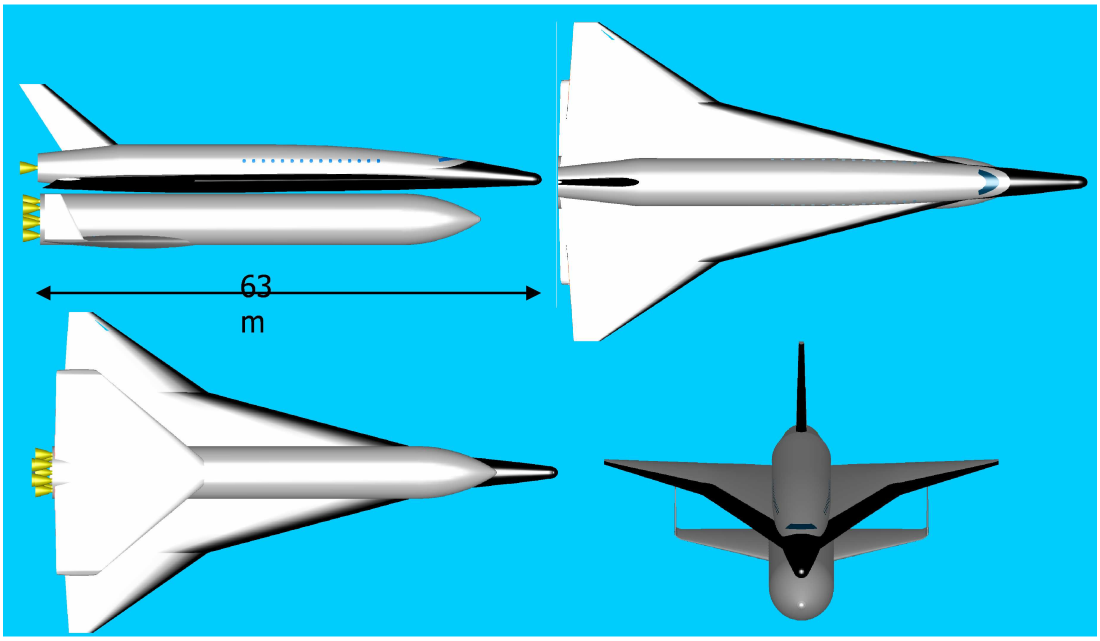

Figure 1. SpaceLiner 1

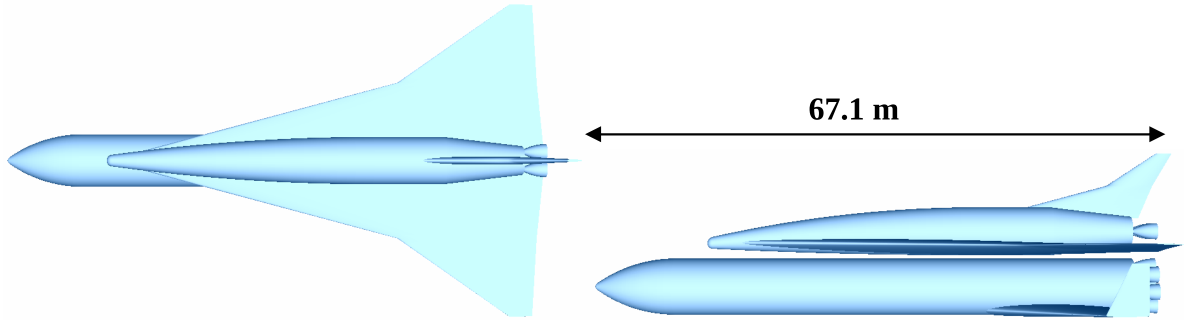

Figure 2. Latest SpaceLiner 2 Configuration

	Booster	Orbiter
Number of engines	8	2
Mixture ratio	6:1	6:1
Chamber pressure [MPa]	16	16
Mass flow per engine [kg/s]	384.5	384.5
Specific impulse in vacuum [s]	437.6	448
Specific impulse at sea level [s]	388.4	360.4
Thrust in vacuum per engine [kN]	1650.6	1689.8
Thrust at sea level per engine[kN]	1465.0	1359.4

Table 1. Engine data

	GLOW Mass [kg]	Mass at burnout [kg]	Propellant mass [kg]	Fuselage length [m]	Max. fuselage diameter [m]	Wing span [m]	Projected wing surface area [m^2]
Orbiter	275,200	120,200	155,000	53	6	40	955
Booster	818,534	114,534	704,000	67.1	7	25.5	325
Total	1093,734	234,734	859,000	-	-	-	-

Table 2. SpaceLiner 2 Characteristics

	Water Tanks Filled	Water Tanks Empty
Wing load $\frac{m}{S}$ [kg/m^2]	125.9	116.3
Glide ratio at Mach 20 [-]	4.08	4.08
Ballistic coefficient $\frac{m}{C_D S}$ [kg/m^2] at max. glide ratio and Mach 20	8167	7818.5
Lifting parameter $\frac{m}{C_L S}$ [kg/m^2] at maximum glide ratio and Mach 20	2075.6	1918.3

Table 3. Aerodynamic Characteristics of the Orbiter of SpaceLiner 2

3. Trajectory

As explained, the SpaceLiner flies a suborbital trajectory. Generally speaking, a suborbital trajectory implies a ballistic trajectory. However, another option for suborbital flight exists. This is a so called 'skip' trajectory. During such a skip trajectory, the vehicle flies a ballistic arc, after which it enters the atmosphere. During its atmospheric flight phase, lift is created and the vehicle leaves the atmosphere again. This process is repeated until the skipping converges into a steady, gliding flight. As compared to a ballistic trajectory, skipping greatly increases the range of the vehicle. This can be seen in Figure 3. Here, the red line represents the ballistic trajectory and the blue line the skip trajectory. Initial speed and altitude are equal in both cases. Only the initial flight path angles differ. In case of a ballistic trajectory, the optimal initial flight path angle for maximum range was determined via parametric variation and was found to be 30°. To obtain the skip trajectory, flight path angle was set to 1°. Note that the ballistic trajectory shown here could in reality never be used for passenger flight, due to the extremely high deceleration and thermal heat loads when re-entering the atmosphere.

Apart from this, it can be seen that the range of the optimal ballistic trajectory is about 10000 km, whereas the range for the skip trajectory is more than 15500km. This shows the huge benefit of using a skip trajectory. As stated in the previous chapter, for a skip trajectory aerodynamical performance of the vehicle is of big importance. The SpaceLiner is designed to have a high glide ratio at hypersonic speeds. At Mach 20 the glide ratio is about 4 (see Table 3).

The trajectory flown by the SpaceLiner starts at Sydney and ends in Western Europe. The powerless skipping phase is presented in more detail in Figure 4 till Figure 101. As can be seen, the vehicle begins its skip trajectory at an altitude of 75 km and with a velocity of 6550 m/s. When an altitude of about 50 km is reached, enough lift is created to leave the atmosphere again. After about 3500 seconds, the skip trajectory has converged into a steady, gliding flight. After only 4500 seconds the SpaceLiner 2 has flown almost 16000 km and reached its destination.

Figure 7 shows that during its first dip in the atmosphere, SpaceLiner 2 flies Mach 19 at an altitude of 48 km. As a result, very high thermal loads will be experienced during flight. Stagnation point heat loads reach 1.9 MW/m^2 at this point. For comparison, the maximum heat load on the Space Shuttle is 0.5 MW/m^2.

G-load in normal direction is presented in Figure 10. As can be seen is does not exceed 1.3g, staying well within the requirement of 1.5g. Angle of attack, presented in Figure 10, varies slightly during to flight to make sure optimal glide ratio is achieved at every instance.

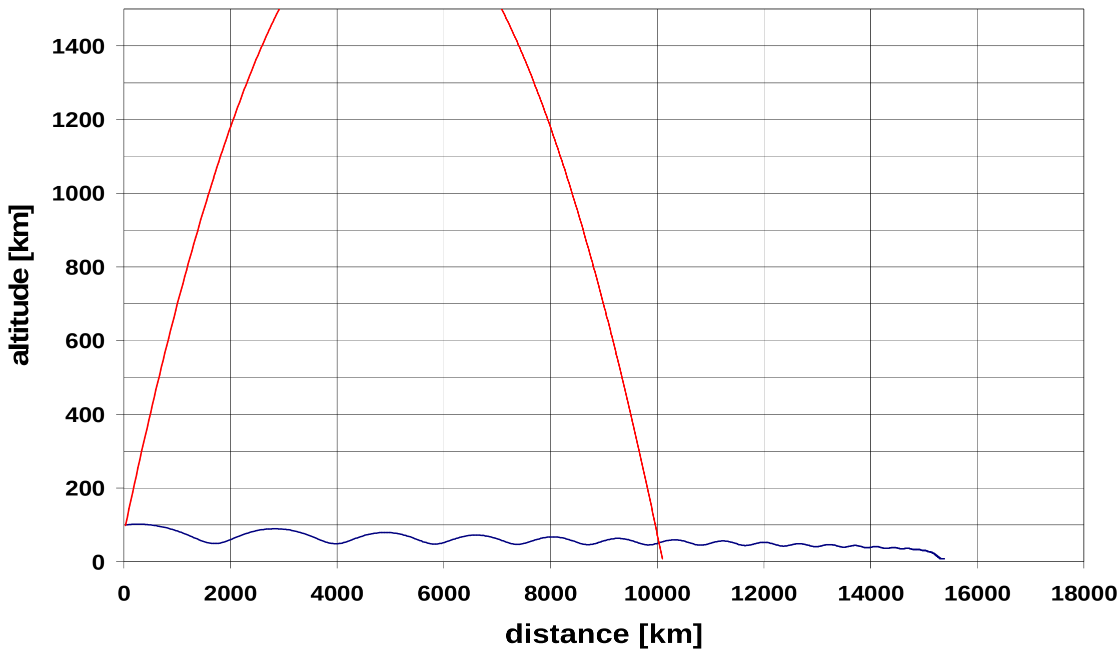

Figure 3. Ballistic versus Skip Trajectory [2]

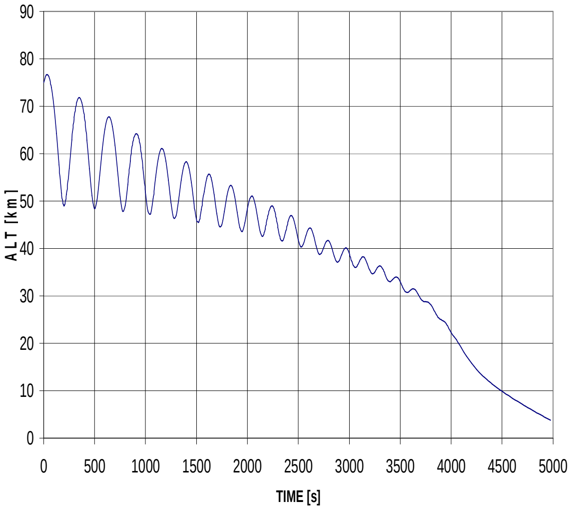

Figure 4. Time History of the Altitude

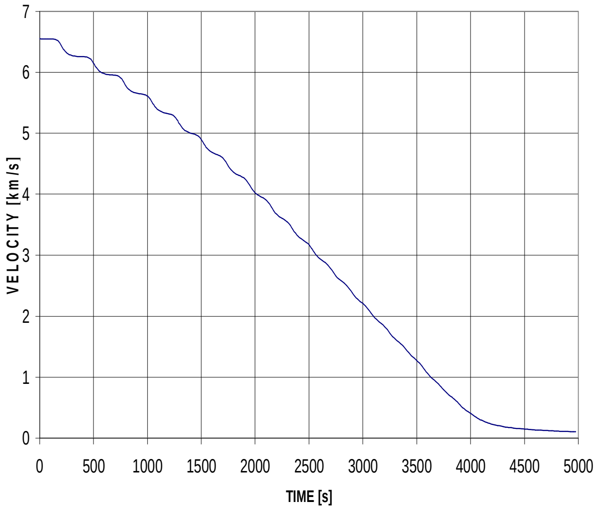

Figure 5. Time History of Velocity

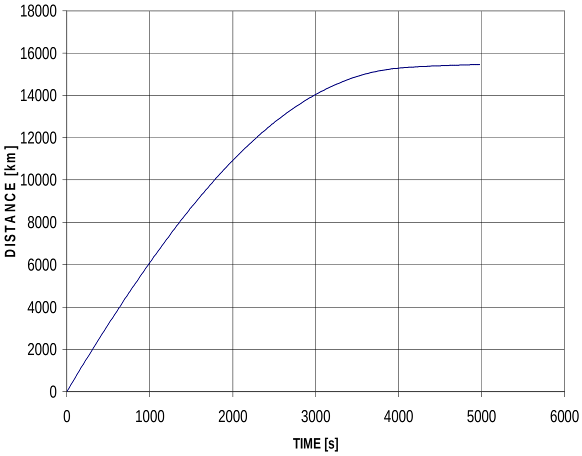

Figure 6. Distance Flown

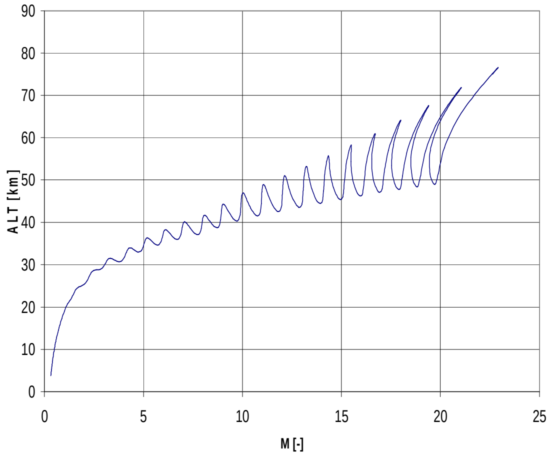

Figure 7. Altitude versus Mach Number

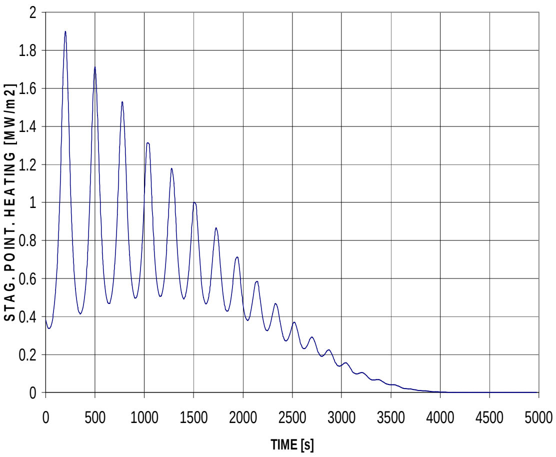

Figure 8. Time History of Stagnation Point Heating

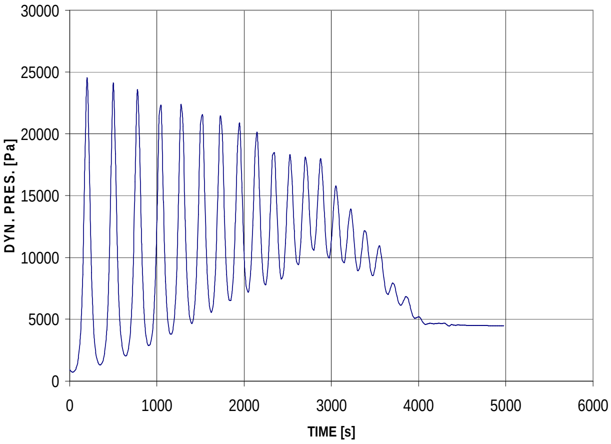

Figure 9. Time History of Dynamic Pressure

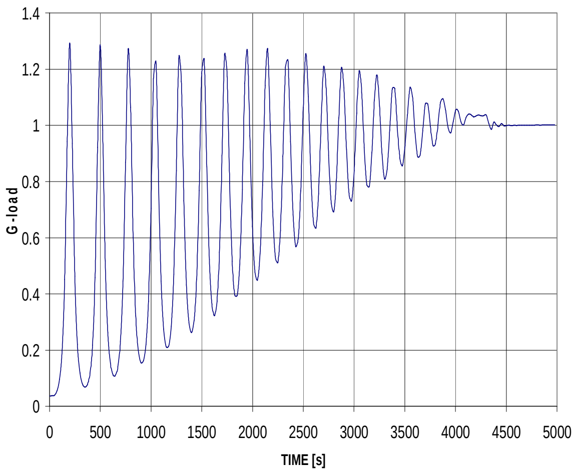

Figure 10. G-Load in Normal Direction

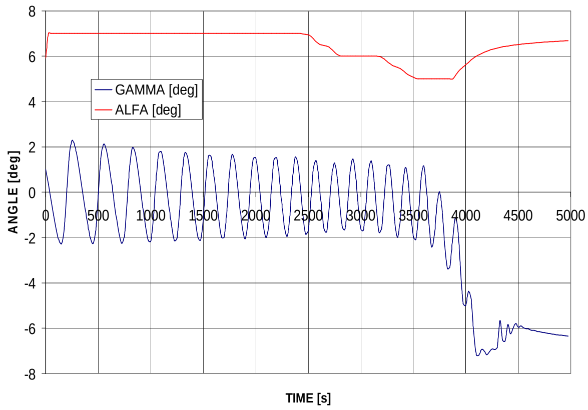

Figure 11. Time History of Flight Path Angle and Angle of Attack

4. Flight Environment and Aerodynamic Heating

To get a better idea of the flight environment of the SpaceLiner, its trajectory is compared to that of the Space Shuttle. In Figure 12 it can be seen that the SpaceLiner travels in approximately the same speed regime, but at lower altitude. This off course means a more denser atmosphere and therefore more extreme heating. This is the main reason why heating of the SpaceLiner is higher than for the Space Shuttle.

Hypersonic flight introduces flow phenomena which are absent in case of lower speed flight. Because of the high air temperatures behind the shock, air cannot be modeled anymore as a perfect gas. Which flow phenomena are present during the flight of the SpaceLiner, can also be seen in Figure 12. Vibration and excitation energies are introduced, as well as dissociation of oxygen and nitrogen. When doing a numerical analysis of the heating, these effects have to be taken into account.

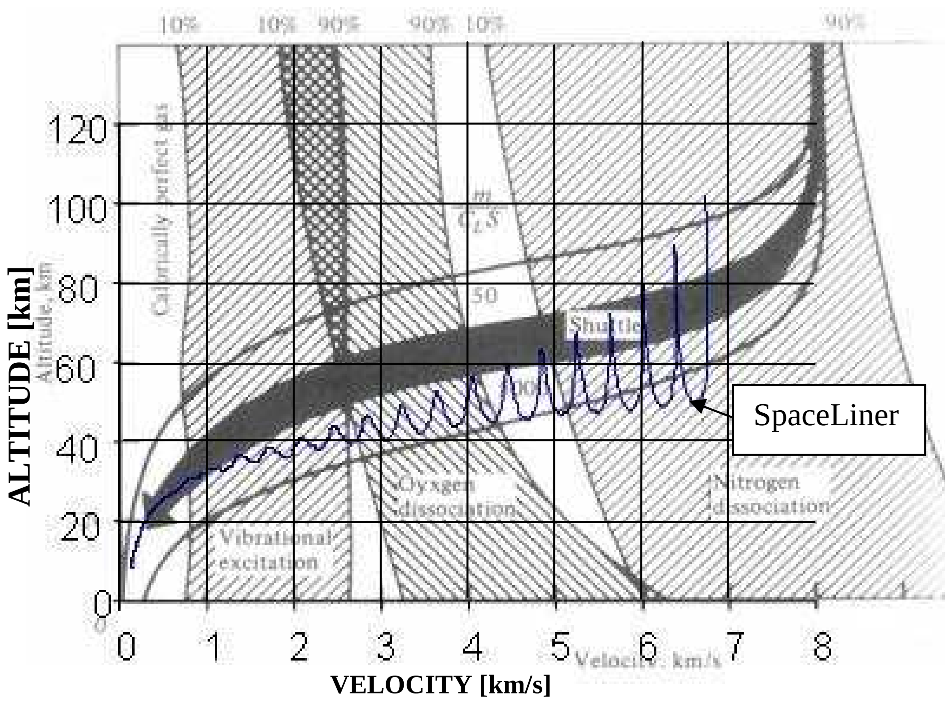

Figure 12. Re-entry of Space Shuttle Compared to SpaceLiner 1 [1]

At the body surface of the vehicle, temperature will generally speaking be lower than the temperature directly behind the shock. The dissociated molecules will start to recombine. These dissociation and recombination reactions take a certain amount of time. If one assumes that the velocity of the air molecules behind the shock is low enough to allow for enough time for the reactions taking place, the equilibrium gas model can be used for numerical analysis.
In case of the SpaceLiner maximum heating is experienced at an altitude of 48 km and a Mach number of 18.8. Heating analysis using the equilibrium gas model results in Figure 13. The left part of the figure assumes a laminar boundary layer, whereas the right part assumes a turbulent boundary layer. As can be seen a laminar boundary layer greatly reduces overall temperature. Temperatures on the leading edges and nose are about equal in both cases and reach about 2900 K and 2400 K, respectively. Such temperatures exceed the limitations of all current thermal protection materials. Therefore, some way to reduce these temperatures has to be found.

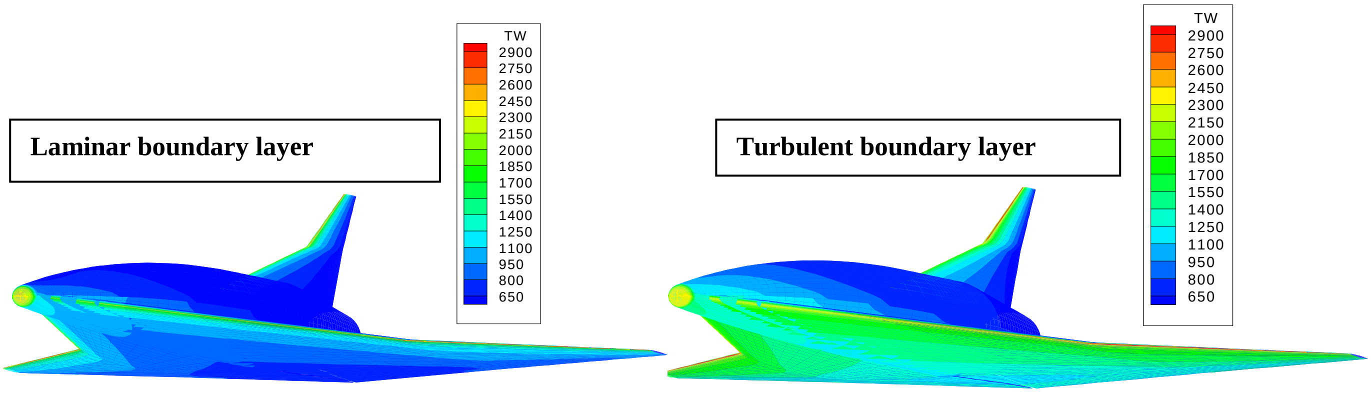

Figure 13. SpaceLiner 2 Equilibrium Temperatures, for an emissive coefficient of 0.83, M=18.8, H=48 km, alpha=7°

5. Transpiration Cooling

To limit the temperatures experienced by the SpaceLiner, a number of options exist. The first option is to adapt the trajectory such that heatloads decrease. Analysis shows that the initial velocity of the powerless flight phase then has to be increased to 7.5 km/s to limit heating to 1MW/m^2 [2]. This results in a big increase in the total mass at lift off. In [2] it is stated that increase in weight would be at least 300 tons, probably even much more than this.
The second option is to change the geometry of the vehicle. For example the nose and leading edges radii could be increased. However, this would lead to a decrease in aerodynamic performance. To make up for this loss, initial speed should again be increased with the result that the weight increases by the same amount as before.
The third option is to actively cool the material down. This can be done by transpiration cooling. By making the heated surface out of a porous material, a cooling fluid can run through this material. The cool fluid absorbs heat by convection and thus cools the material down. Usually, a gas is used as a coolant. Transpiration cooling using a gas has been tested at DLR [4]. To make the cooling system as light as possible, a coolant with high cooling capacity per kg has to be used. In [1,2] it is therefore proposed to use liquid water as a coolant. Together with the wind tunnel department at DLR Cologne, a test campaign in the arc heated wind tunnel L2K has been set up to investigate the feasibility of liquid water as a coolant. In order to verify the advantage of water compared to the gas, additional tests were carried out using nitrogen gas as coolant.

Liquids will not become hotter than their boiling temperature. In case of water this boiling temperature is 100°C at 1 bar and increases proportional to the pressure. If water remains in its liquid state during the transportation through the porous material, the convective cooling will be very efficient due to the large temperature difference of liquid water and the uncooled material. When a material with a very high porosity is used, it will be cooled down to approximately the boiling temperature of the water. To prevent water from evaporating within the porous material, new water has to be supplied at a sufficiently high mass flow rate. The amount of heat, which is necessary to evaporate one kg of water, is called ‘heat of vaporization’. The higher the heat of vaporization is the lower the coolant mass flow can be. Water has the highest heat of vaporization (2260 kJ/kg at 1 bar) of all liquids. A liquid in a porous material will introduce a capillary pressure. This pressure will cause water to flow into regions where no water is present. This capillary action will therefore automatically distribute the liquid over the porous material. A simplified model of capillary action in a porous material can be made by assuming a porous material is made up of a bundle of tubes with a certain radius [5]. As soon as a capillary tube has completely filled itself with water, there will be no capillary action anymore. In case of the cooling method using liquid water, this means that when water evaporates at the surface of the material, the liquid water level in the material will drop. Capillary tubes are not completely filled with water anymore and this then causes capillary action. New water is automatically supplied to the surface at exactly the required mass flow rate.

The cooling concept was tested in the L2K arc heated wind tunnel at DLR-Cologne [1,6]. Three different nose cone models were made out of a porous material called Procelit 170. This material consists of 91% Al_2O_3 and 9% SiO_2. This material was chosen because of its high porosity and its ability to withstand temperatures of up to 2000 K. The models have a varying nose radius, the smallest radius being 1 cm, the middle radius being 1.75 cm and the largest radius being 2.5 cm. The nose radius was varied to be able to investigate the influence of model geometry on the cooling efficiency. The models are shown in Figure 14. Inside the models, a reservoir has been drilled out. The models were connected to a stagnation probe holder of L2K. A copper tube enters the reservoir for water supply. Water mass flow could be adjusted using a valve.

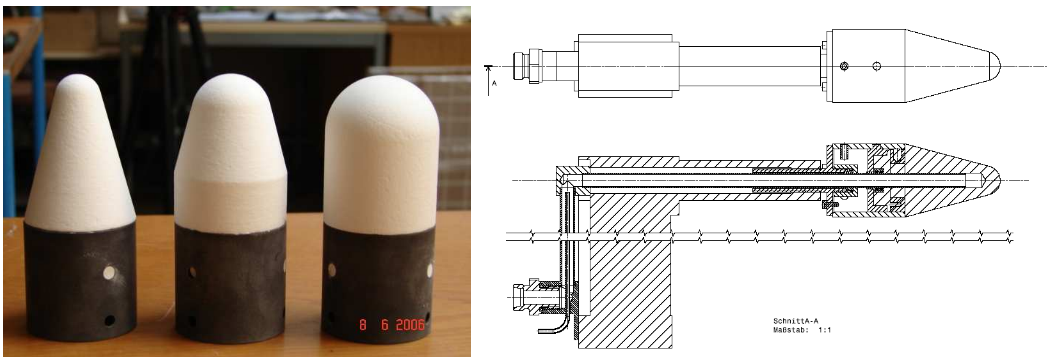

Figure 14. Windtunnel Models [1]

Tests were done using all the models. First, liquid water was used as a coolant. Temperature drops were observed for a certain water mass flow. After these tests had been completed, Nitrogen gas was used as a coolant. All the conditions were chosen identical to the other tests. The same coolant mass flow rate was used as well as the same wind tunnel flow conditions. The surface temperature was measured using an infrared camera. The test procedure was to first insert the models in the flow, without transpiration cooling switched on. Following this procedure radiation adiabatic temperatures could thus be measured. Next, cooling was switched on and the temperature drop could be observed.

Test results of cooling using the model with nose radius of 2.5 cm are presented here. Figure 15 shows an infrared image of the temperatures in the radiation adiabatic case. As can be seen temperatures in the stagnation point reach over 2040 K. The right part of the image represents the behavior of the temperature on certain spots on the model with water cooling over time. The water mass flow rate was 0.2 g/s. Time is presented in minutes. What can be seen is that the whole model is eventually cooled to temperatures below 500 K. The infrared camera is not able to measure temperatures lower than this value, but as explained before it is expected the temperature will be equal to the boiling temperature of the water (which is about 290 K at wind tunnel conditions).

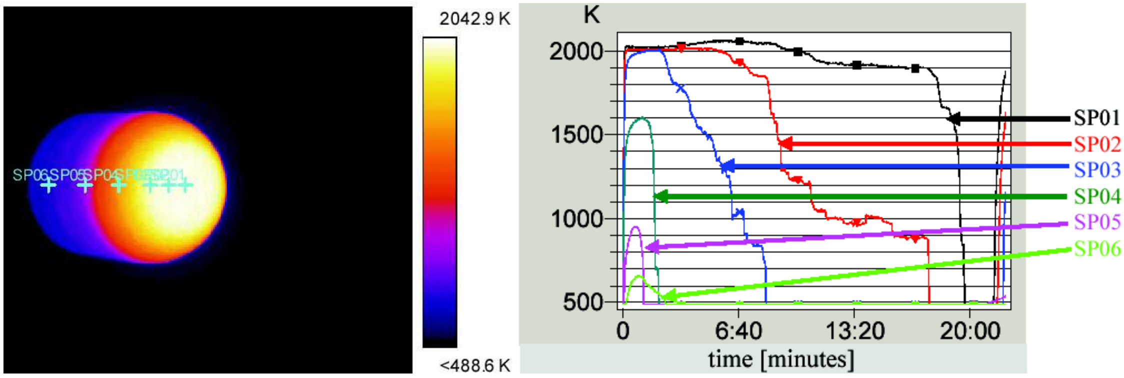

Figure 15. Test Results Using 0.2 g/s Liquid Water [1]

The surface temperature development of the same spots using 1 g/s of Nitrogen can be seen in Figure 16. In this case the stagnation point cooled down to about 1500 K. So even for 5 times higher gas mass flow as water, the temperature drop is still much smaller. In the right part of the figure it can be seen that for the same mass flow rate of the gas as the water (0.2 g/s), temperature drops are extremely small, especially in stagnation point regions.

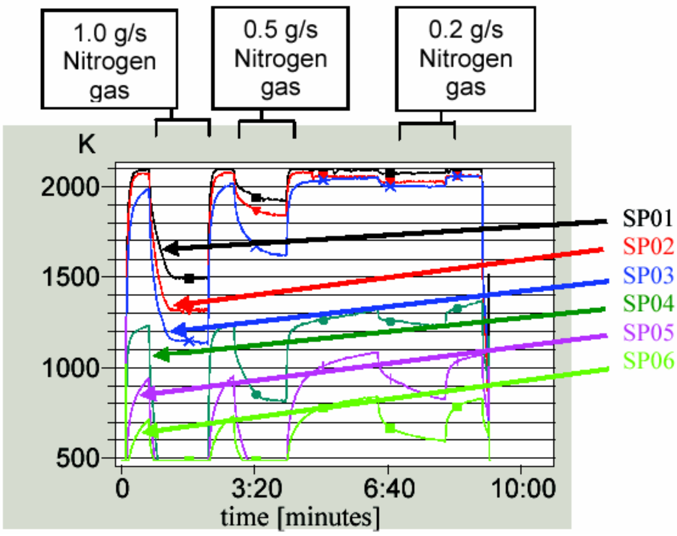

Figure 16. Test Results Using Nitrogen Gas [1]

Transpiration cooling using liquid water has been proven to be much more efficient compared to gas cooling. To be able to make predictions of the required water mass flow for cooling, the results have to be quantified. The first step is to determine the heat flux into the model. The heat flux then determines the evaporation rate of the water and therefore the required water mass flow. Numerical calculations for heat fluxes at wind tunnel conditions result in Figure 17. Here the x axis represents the distance along the centerline of the model and the vertical axis represents the heat flux in W/m^2 at the surface of the model. Note that in case of radiation adiabatic conditions (cooling switched off), heat flux is much smaller than in case of a cooled wall. As explained, during the tests the model is cooled down to about 300 K. So this line is representative for the test conditions. By integrating the heat flux over the surface of the model, the total heat flow into the model can be obtained. In case of water cooling this results in 578 W. Dividing this value through the heat of vaporization of water (2460 kJ/kg at wind tunnel conditions), a required water mass flow of 0.235 g/s is calculated. This is close to the 0.2 g/s of water flow rate, which was measured during the test. The difference is due to not considering the blocking effect in calculations [1]. Further experiments and calculations showed that analysis without blocking overestimate water mass flow rate by about 30%. This then implies that even 0.2 g/s water mass flow rate is too much for this test condition.

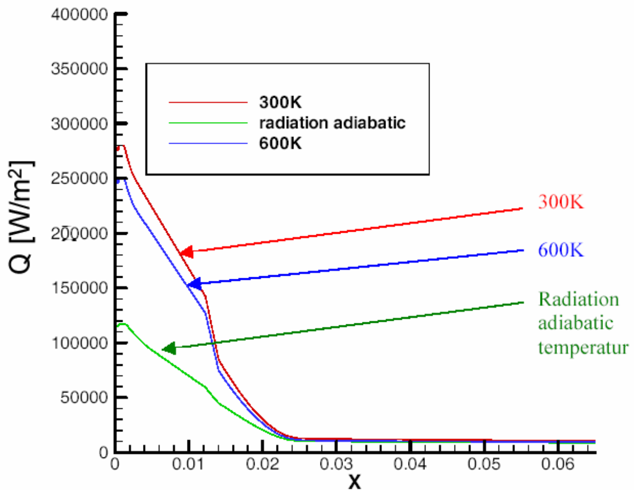

Figure 17. Heat Flux Along the Surface of the Model [1]

An overview of the test results is presented in Table 4. It can be seen clearly that using liquid water as a coolant can save coolant mass compared to using Nitrogen gas as a coolant. Therefore, this new way of cooling is considered very promising and further test are planned.

	Temperature drop using 0.2 g/s water	Temperature drop using 0.2 g/s nitrogen gas	Temperature drop using 0.5 g/s nitrogen gas	Temperature drop using 1 g/s nitrogen gas
SP01	>1500K	0K	200K	600K
SP02	>1500K	50K	250K	800K
SP03	>1500K	100K	400K	850K
SP04	>1100K	100K	400K	>700K
SP05	>450K	300K	>450K	>400K
SP06	>160K	250K	>200K	>200K

Table 4. Comparison between Gas and Liquid Water Coolants

6. Application of Transpiration Cooling to the SpaceLiner

The test results show that the water cooling method is a promising solution for the extreme heating of the SpaceLiner. The application of the new cooling method is investigated further, to determine how much water is needed to cool the vehicle down during its flight. To be on the safe side, the TPS is designed for the case of a turbulent boundary layer. Furthermore, it is assumed that a TPS material is used that can withstand temperatures of up to 1800 K. In this case, only the nose and the leading edge radii have to be cooled down actively. In [1] the water usage is estimated at 9.11 tons.

It is noted that he Procelit 170 material used during the tests is not suitable for application in real flight. The material is extremely brittle and breaks easily. Because of its high porosity, easy manufacturing characteristics and high temperature resistance it is ideal for wind tunnel experiments. In real flight CMC (Ceramic Matrix Composites) such as C/C and C-SiC are more interesting. These materials are very strong. During manufacturing, porosity can be adapted and the required porosity can be obtained. Temperature resistance of C/C is fairly low in oxidizing atmospheres (720K). C-SiC has a temperature resistance of up to 2020K and is therefore the more promising of the two for application on the SpaceLiner.

During testing, the model was cooled down to below 500 K. If a material such as C-SiC is used on the SpaceLiner, such a temperature decrease is off course not necessary. By choosing a lower value of porosity, less water can flow through the material and temperature will not decrease as much. This would save coolant mass and so the 9.11 tons of water calculated is a conservative value.

Another option to decrease water usage could be decreasing the nose and leading edge radii. This can be seen by taking a look at the following equation:

$$\dot{q}_{stag} = C\,\frac{\rho^{0.5}V^{3}}{R_N^{0.5}} \tag{1}$$

where

$\dot{q}_{stag}$	is the stagnation point heat flux
C	is a constant
ρ	is the air density
V	is the airspeed
R_N	is the nose radius

As can be seen, for a smaller nose radius the heat flux in the stagnation point increases, proportional to $\frac{1}{\sqrt{R_N}}$.

According to [1], the total heat flow into a half sphere is given by:

$$\dot{Q}_{tot} = -\frac{4}{5}\pi R_N^2\,\dot{q}_{stag}\cos^{\frac{5}{2}}\theta\Big|_0^{\theta}\,,\ 0 \le \theta \le 70^{\circ} \tag{2}$$

Inserting (1) in (2) yields:

$$\dot{Q}_{tot} \triangleq R_N^{1.5} \tag{3}$$

This shows that decreasing the nose radius will lead to a higher heat flux in the stagnation point, but less heat flow into the complete nose. For leading edges a similar procedure can be used which according to [1] results in:

$$\dot{Q}_{tot} \triangleq \sqrt{R_N} \tag{4}$$

7. Conclusions

To perform a flight from Sydney to Western Europe, the SpaceLiner needs to be accelerated to 6.55 km/s and an altitude of 75 km. The biggest challenge seems to be the aerodynamic heating. A promising new way of transpiration cooling, using liquid water as a coolant, is introduced and first test results are presented. A huge increase of cooling efficiency is observed when using water instead of the option of using a gas as a coolant.

Evolution of the design of the SpaceLiner is described. Aerodynamic performance is improved but total mass has increased due to increased subsystem masses.

Preliminary analysis of the water usage of the SpaceLiner during its flight shows that about 9 tons is necessary to cool the vehicle down during its flight. Other options to reduce the heatload are adapting the trajectory or geometry of the vehicle. This would increase total takeoff weight by more than 300 tons. A number of ways may exist to reduce water usage, such as reducing the nose and leading edge radii. However, more tests are needed to confirm these ideas.

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