Performance of Transpiration Cooled Heat Shields for Re-entry Vehicles

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This paper presents results of a system study of transpiration cooled thermal protection systems for Earth re-entry. The cooling performance for sustained hypersonic flight and transient re-entry of a blunt cone geometry is assessed. A simplified numerical model is used to calculate the transient temperature of a transpiration cooled heat shield. The performance of transpiration cooling is assessed by calculating the overall required coolant mass for different steady state and transient flight scenarios. Spatially and temporally optimised mass injection is presented for various flight conditions. The majority of the injection is required on the spherical nose segment of the blunted cone. Carbon/Carbon composite ceramic and the ultra high temperature ceramic Zirconium diboride are considered as wall materials. Both materials require similar amounts of coolant injection. In continuous hypersonic cruise, transpiration cooling is highly effective for flight conditions with velocities below 8 km s^{-1} and altitudes above 40 km. For transient re-entry, transpiration cooling is most viable for trajectories of entry velocities below 8.5 km s^{-1} and ballistic coefficients below 2.1 kg m^{-2} .

Nomenclature

A	Area, m^2
C_D	Drag coefficient
c_p	Specific heat capacity, $\text{J kg}^{-1} \text{K}^{-1}$
F	Blowing ratio $(\rho_f u_f)/(\rho_g u_g)$
h	Heat transfer coefficient, $\text{W m}^{-2} \text{K}^{-1}$
h_v	Volumetric heat transfer coefficient $h \frac{A}{V}$, $\text{W m}^{-3} \text{K}^{-1}$
k	Thermal conductivity, $\text{W m}^{-1} \text{K}^{-1}$
K_D	Darcy coefficient, m^2
K_F	Forchheimer coefficient, m
L	Thickness, m

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m	Mass, kg
$\dot{m}_A$	Mass flux, $\text{kg s}^{-1} \text{m}^{-2}$
$\dot{q}$	Heat flux W m^{-2}
R_N	Nose radius, m
Re	Reynolds number
s	Length along the vehicle's surface, m
t	Time, s
T	Temperature, K
V	Volume, m^3
v	Velocity, m s^{-1}
x	Spatial variable, m
α	Cone half angle, rad
β	Ballistic coefficient $\beta = \frac{m}{AC_D}$, kg m^{-2}
η	Film effectiveness
ϵ	Emissivity
γ	Flight path angle, rad
ϕ	Porosity
ρ	Density, kg m^{-3}
<i>Subscript</i>	
f	Coolant fluid
g	Hot gas at boundary layer edge
r	Recovery
s	Solid

I. Introduction

Hypersonic re-entry vehicles are exposed to very high heat loads that originate from shock heated high temperature gas in front of the vehicle [1]. Therefore, thermal protection systems are required to manage these heat fluxes. Wall temperatures need to be kept below critical values, e. g. oxidation, sublimation or melting temperatures. For shape stable heat shields transpiration cooling is an approach that enables active thermal control [2–8]. It relies on a gaseous or liquid fluid which is fed through a porous solid material [9–14]. The coolant leads to internal convective cooling between the porous material and the fluid, and to the formation of a film that keeps the hot external gas further away

from the surface. This cushion of cold gas reduces aerodynamic heating significantly and forms a protective film that extends further downstream from the injection location.

Transpiration cooling research dates back as far as the 1950s where the potential of this cooling technique was first conceived [15, 16]. In the case of hyperbolic re-entry, the thermal management approach was competing against ablative heat shields that also offer effective cooling and thermal insulation without the drawback of an active control system [17, 18]. The complexity of an active system which adds another point of possible failure needs to be justified. However, transpiration cooling offers a variety of additional advantages like re-usability, shape stability and the ability to optimise mass injection locally and temporally. The large amount of research that has been focused on this technique highlights its potential [11, 13, 19–23].

Significant research has been conducted on the film efficiency in high enthalpy wind tunnels. Experimental studies mainly focused on flat plate, cone and wedge geometries in order to study the development of the coolant film and the associated cooling effectiveness. Figure 1 summarises mostly recent work conducted for a flat plate geometry at hypersonic conditions [13, 21, 24–28]. The Reynolds number refers to the axial flow distance between model leading edge and location of injection, and the blowing ratio is defined as the ratio between coolant mass flux (subscript f) and external boundary layer edge mass flux (subscript g),

$$F = \frac{\rho_f u_f}{\rho_g u_g}, \quad (1)$$

with density ρ and velocity u . Studies are excluded where the Reynolds number or blowing ratio could not be inferred from the published data [8, 29–33], or where other geometries were used [4, 7, 14, 34–39].

In the respective publications, special attention has been paid to the influence of injection on the downstream boundary layer state. Too large blowing ratios can lead to an early onset of transition. The conclusions of the respective test campaigns are indicated in Fig. 1 by a background colour. As expected, the ideal application of transpiration cooling is in low Reynolds number regimes with low blowing ratios. The laminar flow remains mostly undisturbed and a coolant film is formed that does not suffer from strong mixing with the hot external flow. Larger blowing ratios lead to an early triggering of turbulence and increase coolant-external flow mixing which decreases the overall effectiveness of the cooling method. On the other hand, even low blowing ratios lead to transition when the free stream Reynolds number is increased. An especially poor performance is noticed for high Reynolds number flows with large blowing ratios. Even though the effectiveness of the cooling method will be impaired in the regime of turbulent flow, it will still provide a cooling benefit. In re-entry it is likely that a flight vehicle will experience the majority of the identified regimes as external flow conditions range from very high altitude low Reynolds number flows to fully turbulent boundary layers [5, 40–43].

This paper aims to investigate the performance of transpiration cooling for Earth re-entry vehicles. The approach

Fig. 1 Overview of transpiration cooled flat plate experiments [13, 21, 24–28]. The Reynolds number refers to the axial flow distance between model leading edge and location of injection.

is a simulation of the thermal protection system for a large number of flight scenarios. Within the simulation, an optimization of the coolant injection is performed that gives the required mass flux of coolant for each point in time and at each location on the vehicle's surface. The results of this analysis are then used to assess the overall coolant mass that is required for a given trajectory. The altitude-velocity parameter space for hypersonic flow conditions is investigated to assess where transpiration cooling appears to be a promising candidate thermal management technique. Conversely, regions in the parameter space and types of re-entry trajectories are identified where the effectiveness of transpiration cooling is impaired. For this purpose, a simulation tool has been developed that allows for the efficient calculation of the material temperature of a porous wall for a given flight trajectory [44]. Two different materials are considered to assess the general performance of this cooling technique in various hypersonic flow regimes. In addition, the synopsis of many experimental investigations (see Fig. 1) is used to assess practical limitations of the cooling technique.

II. Model overview

The code *Porous Impulse Response Analysis for Transpiration cooling Evaluation* (PIRATE) developed for the analysis of the current work is used to calculate the transient temperature response of a porous transpiration cooled material that is subject to aerodynamic surface heating. Figure 2 presents an outline of the code which is described in detail in Ref. [45]. The code uses mostly analytical or low-cost numerical schemes and hence achieves short computation times which allow for a large amount of calculations in a reasonable time. Aerodynamic correlations are used to determine the surface heat flux for a blunted cone geometry [46, 47]. The cooling effect of local heat flux mitigation

Fig. 2 Flow chart of the computational approach.

and downstream film effectiveness is taken into account using semi-empirical models [20, 48, 49]. The temperature of the fluid and solid phases in the heat shield material are calculated based on an impulse response convolution approach that solves the coupled fluid-solid heat conduction problem assuming an adiabatic back-wall of the plenum structure behind the transpiration cooled segment [44].

The model is validated using data from experiments in the L2K plasma wind tunnel and data from the SHEFEX II re-entry flight test using solid un-cooled C/C-SiC and porous transpiration cooled C/C materials. The different validation cases demonstrate that the numerical model developed matches experimental data closely and agrees with experimental measurements to within 10 %. The relevant effects occurring during transpiration cooled hypersonic flight, i. e. transient heating, local cooling through blowing, and downstream film coverage, are replicated within acceptable accuracy [45]. The validation cases correspond to set-ups where lateral conduction is comparably small and hence a one-dimensional approach captures the relevant effects. Consequently, the geometry studied in the remainder of this work is not expected to suffer from significant two- or three-dimensional heat conduction.

III. Results

A wide parameter space for Earth return flight conditions is assessed in order to identify domains where transpiration cooling offers good performance. Conversely, the analysis also aims to identify possible drawbacks and domains where transpiration cooling is not a suitable cooling technique. The fundamental approach to achieve this goal is the determination of the locally required mass flux along a vehicle's surface to keep the wall temperature below critical values. Hence, the local mass injection is optimised for a certain flight scenario. The so determined mass flux is then integrated over the respective vehicle surface area to yield the total mass flow rate. In the case of a transient calculation, this mass flow rate is integrated temporally resulting in the absolute coolant mass. This allows the estimation of a quantitative mass budget for various hypersonic flight scenarios.

Fig. 3 Geometry considered in this study.

The study presented in this paper considers a single geometry (shown in Fig. 3) based on previous missions and numerical investigations [23, 50]. The vehicle shape is representative of a generic hypersonic re-entry geometry, a 30° half angle blunted cone with a 0.5 m nose radius and a wall thickness of $L = 10$ mm. The total length s from the stagnation point along the body profile is 2 m which results in a rear base radius of 1.17 m.

Two different porous materials are considered, Carbon-Carbon (C/C) ceramic composite, and the ultra high temperature ceramic (UHTC) ZrB_2 [51, 52]. The material parameters used in this study are summarised in Table 1. The high emissivity and low thermal conductivity make C/C a heat shield material that is well suited for radiation cooling and insulation of the backside. The very high volumetric heat transfer coefficient has not been measured and is based on an estimate of Schweikert et al. [54]. The lower emissivity of ZrB_2 makes cooling through re-radiation less effective and the much higher thermal conductivity results in a higher backside temperature of the heat shield. The material lends itself to the transpiration cooling approach as the melting temperature is very high (3505 K) and the manufacturing allows for complex pore and channel arrangements [55, 56]. The extremely fine pore structure (in the order of $1 \mu\text{m}$) also allows for a very even injection with no localised jets from fissures in the material [52]. Furthermore, the high thermal conductivity ($k_s = 41.4 \text{ W m}^{-1} \text{ K}^{-1}$) can be used against hot spots, e. g. at stagnation points or shock impingement locations, in order to cool locally by using lateral conduction to lower temperatures in high heat flux regions of the heat shield. The material has recently been investigated in plasma wind tunnel tests highlighting its applicability as a heat shield candidate in high enthalpy conditions [57–59]. In terms of the coolant fluid, Helium is chosen as it has been identified as one of the best coolant candidates, mainly due to its high heat capacity ($c_{p,f} = 5193 \text{ J kg}^{-1} \text{ K}^{-1}$) [9, 11, 60].

The results of this study are divided into three sections. In section III.A the performance of transpiration cooling for a sustained steady state hypersonic flight is assessed. Section III.B presents the findings for transient re-entry flights.

Table 1 Material properties used in this study.

Material	C/C [22, 51, 54]	UHTC [44, 52, 61, 62]
ϕ	0.105	0.42
ϵ	0.85	0.7
$k_s / \text{W m}^{-1} \text{K}^{-1}$	1.97	41.4
$c_p / \text{J kg}^{-1} \text{K}^{-1}$	1339	437
$\rho / \text{kg m}^{-3}$	1400	6100
K_D / m^2	$7.5 \cdot 10^{-14}$	$3.4 \cdot 10^{-14}$
K_F / m	$8.5 \cdot 10^{-8}$	$15.7 \cdot 10^{-8}$
$h_v / \text{W m}^{-3} \text{K}^{-1}$	$1 \cdot 10^6$	$5.5 \cdot 10^4$

Finally, section III.C provides a comparison between the assessment of steady state flight and transient re-entry.

A. Steady state flight analysis

Figures 4 and 5 show an example of the aforementioned optimisation study and present the mass flux and recovery/film temperature distribution along the vehicle's surface for three flight scenarios with C/C material. The optimisation objective here is to keep the wall temperature below 1800 K everywhere on the vehicle. The analysis assumes a thermal steady state for the transpiration cooled wall segment and hence refers to sustained hypersonic flight at an altitude h with flight velocity v . The highest heat flux always occurs at the stagnation point as no downstream transition is considered in the used aerodynamic heat flux correlations [46, 47]. This results in the stagnation point also always requiring the highest mass flux to keep it below critical temperatures. In the case of point 1 and 3, only part of the spherical segment has to be actively cooled. The smaller downstream heat flux and the film of the upstream injection keep the surface heating low enough for passive cooling to be sufficient. In both cases 1 and 3, the film temperature rapidly falls below the recovery temperature where active cooling is applied. As soon as no more coolant is injected ($s/R_{\text{Nose}} > 0.7$), the heat conduction and diffusion of the hot surrounding gas result in a steadily increasing film temperature that approaches the free stream recovery temperature. In the case of point 2, cooling is required along the entire vehicle length due to the much higher aerodynamic heat flux. Again, most of the mass flux is injected near the stagnation point which features an order of magnitude higher mass fluxes than the conical section of the vehicle. The spike at $s/R_{\text{Nose}} = 1$ results from the transition between spherical and conical segments and the respective change in free stream mass flux. The film temperature of case 2 shows a large decrease near the stagnation point due to the high mass injection. It cools down further to a minimum at $s/R_{\text{Nose}} = 1$. The film is significantly cooler than the post shock temperature which reduces the surface heat flux and therefore necessitates lower mass injection at regions further away from the stagnation point. The low injection rate at the conical segment results in a steady increase of the film temperature, similar to the cases of points 1 and 3. At this stage it should be noted that PIRATE was validated using data from experiments with laminar flows. Turbulent experimental conditions, such as expected for case 2, have not been

Fig. 4 Mass flux along vehicle surface for different flight conditions at steady state (C/C material, $T_{\text{maximum}} = 1800 \text{ K}$).

Fig. 5 Recovery and film temperature along vehicle surface for different flight conditions at steady state (C/C material, $T_{\text{maximum}} = 1800 \text{ K}$).

compared directly to the simulations due to a lack of suitable cases from the open literature. The expected increased mixing between film and external hot gas might lead to different quantitative results. Therefore, the assessments made in this work should be viewed as a lower boundary for transpiration cooling needs under these conditions.

For the next step, the calculated mass flux distribution is integrated over the vehicle area which yields the mass flow rate for the entire vehicle. This optimisation is conducted for many combinations of velocity and altitude resulting in a mass flow rate map for this parameter space. Figure 6 presents the total mass flow rate that would be required to keep the surface temperature everywhere on the vehicle below 1800 K for C/C and UHTC respectively. The three previously investigated flight conditions are indicated as markers. Figure 7 shows the results for the same scenario, but for a maximum surface temperature of 2500 K. In all figures, regions exist that are plotted as white and do not require any cooling, i. e. a mass flow rate of $\dot{m} = 0$. In these cases, the passive cooling of the surface by re-radiation is sufficient to keep the material below the respective maximum temperature. As expected, the velocity limit where active cooling is required is small for low altitudes and large for high altitudes due to the influence of surface pressure on the heat flux. The analysis carried out in this work only considers convective heat transfer and neglects gas radiation. Therefore, the applicability of the results is limited to regions where convective heat transfer dominates. The regime where gas radiation cannot be neglected is indicated by a region bounded by a white line. Within this region the stagnation point radiative heat flux from the post-shock gas exceeds 10 % of the convective heat flux [63]. The flight conditions lying within the bounded region are to be considered only qualitatively but are not treated any further in the discussion.

Both C/C and UHTC materials require a similar coolant mass flow rate, with C/C being slightly more efficient due to the higher volumetric heat transfer coefficient and the higher surface emissivity. As expected from basic aerodynamic

Fig. 6 Required mass flow rate to keep the entire vehicle below 1800K for a steady state flight with C/C (left) and UHTC (right).

Fig. 7 Required mass flow rate to keep the entire vehicle below 2500K for a steady state flight with C/C (left) and UHTC (right).

heating, the required mass flux is higher for lower altitudes due to the higher free stream density. The density increases the heat transfer coefficient to the body and the free stream mass flux. Hence, in order to keep the blowing ratio, one of the key parameters controlling cooling efficiency, at a sufficient level the injected coolant mass flux must increase as well. The required mass flow rate is also higher for higher velocities due to increased free stream enthalpy that leads to a higher heat flux.

At this stage, it is attempted to assess regions in the parameter space where transpiration cooling performs well and where it suffers from drawbacks. The approach to address this is based on findings of previous experimental studies of flat plate models [13, 21, 24–28]. As a first stage approach, these results are used as an indicator to assess the cooling technique for the case of the blunted cone considered in this work. This is due to the flat plate geometry being studied in depth and having provided the most conclusive data for a large variety of flow conditions. As identified in Fig. 1, transpiration cooling works very well in the laminar regime (low Reynolds numbers at injection point) and less

effectively for turbulent flow (high Reynolds numbers at injection point). From the wealth of experimental data available in Fig. 1 and the conclusions of the respective authors we identified an approximate limit Reynolds number of $5 \cdot 10^5$ where injection seem to lead to early transition, regardless of the blowing ratio. This value is purely phenomenological and is based on the independent conclusions of Refs. [13, 21, 24–28]. If the Reynolds number increases even further, the incoming flow will be turbulent before the injection and the effectiveness of the cooling method will be low due to turbulent mixing. In Figs 6 and 7, a shaded region is included for cases where the Reynolds number at an injection point somewhere on the vehicle surface exceeds $5 \cdot 10^5$. Conditions contained in this region will either trigger early transition or be less effective due to an already turbulent incoming flow. As stated above, the quantitative assessment made for this most likely turbulent regime should be viewed as a minimum requirement in terms of needed mass injection. It can be concluded that low altitude conditions all suffer from this restriction. For this criterion C/C shows a slightly better performance than the UHTC material. It should be noted that, although the effectiveness of the cooling method will be impaired in the regime of turbulent flow, it will still provide a cooling benefit.

An additional boundary that has been extracted from Fig. 1 is the maximum blowing ratio for very low Reynolds numbers that results in a lower film effectiveness or even in tripping of the boundary layer. A value of $F = 0.04$ is chosen as the critical parameter. This parameter is again purely phenomenological and based on the observations of the experiments reported in Refs. [13, 21, 24–28]. Another shaded region is included in Figs 6 and 7 that includes the altitude-velocity combinations where this blowing ratio is exceeded somewhere on the vehicle. This region contains conditions at high altitudes and high velocities where even a small coolant mass flow rate results in a large blowing ratio due to the very low free stream density. The high velocity results in a high heat flux necessitating significant coolant injection. However, even small amounts of coolant result in comparably large blowing ratios since the post shock density is small and hence the free stream mass flux is small as well. This large blowing ratio always occurs at the stagnation point due to the high heating rates at this location. Tripping of the boundary layer at this point would be detrimental for the entire downstream region as it would be exposed to a higher heat flux due to early transition. Hence, the $F > 0.04$ shaded regime is a region where transpiration cooling could actually lead to a negative overall effect for the vehicle. Another drawback is that these regions do not benefit significantly from internal convective cooling. This mechanism is dependent on the absolute coolant density in the porous material whereas the external film cooling is dependent on the ratio of coolant to free stream mass flux [11, 44]. The low mass fluxes of this region lead to low coolant densities and hence to a relatively poor internal convective heat transfer. C/C performs significantly better than UHTC mainly due to the very high estimated volumetric heat transfer coefficient that leads to a more effective cooling at low mass fluxes. This means that a larger portion of the cooling occurs through internal convective heat exchange rather than a film effect on the surface.

The conditions that are not included in any of the shaded regions are identified as highly suitable for transpiration cooling. For these altitude-velocity combinations, moderate mass flow rates are required and no early transition as well

Fig. 8 Mass flow rate required to keep the vehicle surface temperature below a specific maximum (C/C material).

as a high film effectiveness is expected. Furthermore, regions of high radiative heat flux could also be considered when the coolant gas is chosen to specifically absorb in wavelength regions where the radiative heat flux occurs.

A further possible benefit of transpiration cooled heat shields is the mitigation of surface oxidation. The practical temperature limit of both C/C and UHTC is determined by the onset of surface oxidation, which starts to occur at higher wall temperatures and prevents the re-usability of heat shields or can even lead to a catastrophic failure [22, 59, 64, 65]. A coolant gas layer can be utilised to prevent the diffusion of oxygen to the surface and hence prevent significant oxidation [58, 59, 66]. If oxidation can be delayed to higher surface temperatures, the effect of passive re-radiation can be exploited and the performance of the heat shield increases. The comparison of Figs. 6 and 7 shows that a significantly lower mass flow rate is required for the higher maximum surface temperature (2500 K). The effect of maximum wall temperature is further explored in Fig. 8 which shows the required mass flow rate for the entire vehicle in the case of three specific steady state flight conditions with C/C material. The highest altitude condition requires the smallest amount of coolant and the lowest altitude condition requires much more mass injection due to the larger free stream mass flux. A non-linear trend is observed where lower wall temperatures need much more coolant. This is due to a number of effects: Re-radiation is very effective at high surface temperatures due to the non-linearity of the Stefan-Boltzmann law. Additionally, the reduction in aerodynamic heat flux due to mass injection is very sensitive for small mass fluxes due to the exponential decay of heat flux reduction with blowing parameter [11]. Low surface temperatures require large coolant mass fluxes where additional blowing does not provide such a significant gain as for low injection cases, i. e. at high surface temperatures. Finally, the internal convective heat transfer scales non-linearly as well. For low injection cases, a thermal equilibrium between solid and fluid phase is likely. In this case, a saturated fluid temperature is reached where the coolant extracted the maximum possible heat. When the coolant mass flux is increased the fluid does not have

enough time to equilibrate with the solid material and consequently a non-equilibrium between the phases becomes more prevalent. In these cases, i. e. at low surface temperatures, the internal cooling becomes less effective.

B. Transient flight analysis

One aim of the system analysis presented in this work is the ability to use it as a means to assess the needed total mass for a given mission. This section presents the required total mass budget for various transient re-entry trajectories and provides a more detailed analysis of an example trajectory. A transient calculation of the heat shield temperature field is performed where for each spatial location s and each time t a mass flux $\dot{m}_A(s, t)$ is calculated that ensures that the surface temperature stays below 1800 K. The mass flux is then integrated over the vehicle area and subsequently integrated over time to obtain the absolute coolant mass. The geometry of the analysed vehicle is again the blunted cone presented in Fig 3 and only UHTC is used as the material choice. The trajectory of a re-entry vehicle is determined by its entry velocity v_{entry} , the entry angle γ , and the ballistic coefficient, which is defined as

$$\beta = \frac{m}{C_D A}, \quad (2)$$

with the vehicle mass m , the drag coefficient C_D , and the cross-sectional area A . Figure 9 shows an example of a ballistic entry with the entry angle of $\gamma = 15^\circ$, a ballistic coefficient of $\beta = 42.1 \text{ kg m}^{-2}$, and an entry velocity of $v_{\text{entry}} = 6.7 \text{ km s}^{-1}$. The maximum stagnation point heat flux peaks at approximately 12 MW m^{-2} 31 s after entry. Figure 9 presents the transient surface temperatures at various locations along the body of the vehicle. Points close to the stagnation point reach the maximum temperature very quickly due to the high heating rates and thus require an earlier start of coolant injection. Locations further downstream profit from the cooling of the upstream film and the lower heat flux and hence reach the maximum surface temperature approximately 15 s later. After roughly 42 s the heating rates have dropped so far that cooling through re-radiation exceeds the incoming aerodynamic heat flux and no need for further coolant injection exists.

The spatially resolved injection mass flux at different times is shown in Fig. 10. It shows again that injection at early times (20 s) is only needed at the nose whereas the whole vehicle has to be cooled at high heating rates at 35 s. Overall, the required spatial distribution is very similar to the results obtained for steady state flight. The same assessment holds true for the film and recovery temperatures shown in Fig. 11. The film temperature distribution at 20 s is strongly influenced by the relatively small amount of injection as the free stream mass flux is small at high altitudes early on the trajectory. However, an almost identical amount of coolant injection at 40 s leads to a very small deviation between film and recovery temperature. This is because of the high free stream mass flux at low altitudes which makes coolant injection much less effective. A complete depiction of the required mass flux at each spatial location and each time is shown in Fig 12. The plot shows again that the majority of the cooling is required on the spherical nose segment.

Fig. 9 Trajectory data and stagnation point heat flux (left), surface temperatures of a UHTC heat shield (right) for the ballistic entry with $\gamma = 15^\circ$ and $\beta = 42.1 \text{ kg m}^{-2}$.

However, during and close to the peak heating time, additional mass injection on the conical segment is required to keep the surface temperature below 1800 K. Injection at the nose segment also leads to a significant reduction in the downstream heat flux due to the coolant film that is established on the surface. The film effectiveness clearly highlights this feature. The growing film along the nose segment leads to a steadily decreasing aerodynamic heat flux and also covers the downstream conical segment where significantly less mass is injected into the boundary layer. The sharp unsteady edges in Fig 12 are due to the local fluid injection temperature. The injected fluid is almost never in thermal equilibrium with the respective surface temperature. However, at locations where no mass is injected, the wall and fluid temperature are identical by definition. Since the effectiveness presented in Fig. 12 is calculated with the local fluid temperature, unsteady edges are present in the plot.

The optimised injection mass fluxes are spatially and temporally integrated to obtain the absolute coolant mass for a given trajectory. For this assessment, twelve generic ballistic entries are analysed where the entry velocity, ballistic coefficient and entry path angle are varied. In order to investigate transpiration cooling for a wide variety of re-entry scenarios, these trajectories range from relatively mild entry conditions, i. e. small entry path angle, ballistic coefficient, and velocity to very harsh conditions, i. e. large entry path angle, ballistic coefficient, and velocity. The trajectory paths are calculated based on these parameters and no lift is considered. Additionally, four real re-entry flight trajectories are considered, the Space Shuttle, IXV, Stardust and Hayabusa. For these real cases, the velocity and altitude as a function of time was taken from Refs. [41, 42, 67–70] and used as input for the PIRATE simulations. Table 2 presents the results for these selected cases. Both lifting body trajectories, i. e. Space Shuttle and IXV both feature low total mass budgets below 0.5 kg and very low maximum blowing ratios below 0.05 which are very advantageous for a transpiration cooled heat shield. Hyperbolic entries like Hayabusa and Stardust feature a larger mass budget and very high maximum blowing ratios. In these cases, an active cooling technique does not perform well. As stated in section III.A, no radiative heat

Fig. 10 Required mass flux for different times along vehicle surface for the ballistic entry with $\gamma = 15^\circ$ and $\beta = 42.1 \text{ kg m}^{-2}$ (UHTC material, $T_{\text{maximum}} = 1800 \text{ K}$).

Fig. 11 Recovery and film temperature for different times along vehicle surface for the ballistic entry with $\gamma = 15^\circ$ and $\beta = 42.1 \text{ kg m}^{-2}$ (UHTC material, $T_{\text{maximum}} = 1800 \text{ K}$).

flux is considered in this analysis which can contribute significantly to the overall experienced heat flux of hyperbolic entries. Hence, the presented values of the required mass flux are only valid for purely convective blocking. The generic ballistic entries show that the total mass budget is smaller for missions with more shallow entry flightpath angles. As expected, the required mass budget increases with increasing ballistic coefficient, since the experienced heat flux rises dramatically with a higher ballistic coefficient. Furthermore, as expected, a higher entry velocity also increases the mass budget significantly.

A further analysis of the ballistic trajectories is performed that investigates the injection characteristics during the re-entry flight. The optimised spatio-temporal injection distribution is compared with the empirical observations from experimental campaigns which are discussed in section I. An example of this analysis is shown in Fig. 13 for the case of $v_{\text{entry}} = 6.7 \text{ km s}^{-1}$, $\gamma = 15^\circ$ and $\beta = 21.1 \text{ kg m}^{-2}$. The left side of the figure shows three curves that correspond to the transient evolution of the blowing ratio $F(s, t)$ and the Reynolds number $Re_s(s, t)$ at their respective locations along the trajectory. For the example of the stagnation point: The first required coolant injection happens at a Reynolds number of approximately 300. As the trajectory continues, the Reynolds number increases and the required blowing ratio increases as well. Eventually, the blowing ratio reaches a maximum and slowly decreases for the later parts of the trajectory. The need for coolant injection stops at a Reynolds number of approximately $1 \cdot 10^5$. A similar plot can be constructed for every discrete location on the vehicle surface. The plot in the top right shows this overlay of all these curves which gives the vehicle's "footprint" in the Reynolds number-blowing ratio parameter space for this trajectory. The same curves are shown in a double-logarithmic plot in the bottom right of Fig. 13 which also includes the conditions of various experiments from Fig. 1. The plot also includes the previously simplified conclusions of the boundaries of $Re_s = 5 \cdot 10^5$

Fig. 12 Required mass flux (left) and film effectiveness (right) for the ballistic entry with $\gamma = 15^\circ$ and $\beta = 42.1 \text{ kg m}^{-2}$ ($T_{\text{maximum}} = 1800 \text{ K}$).

Table 2 Transient flight analysis results.

Trajectory	$\gamma / ^\circ$	$\beta / \text{kg m}^{-2}$	$v_{\text{entry}} / \text{km s}^{-1}$	$m_{\text{total}} / \text{kg}$	Max. $\dot{m}_A / \text{kg m}^{-2} \text{ s}^{-1}$	Max. F
Space Shuttle [67]	-	-	-	0.057	0.0043	0.043
IXV [41, 42, 68]	-	-	-	0.47	0.011	0.043
Stardust [69]	-	-	-	0.64	0.23	0.27
Hayabusa [70]	-	-	-	1.18	0.43	0.18
Ballistic 1	5	2.1	6.7	0.17	0.045	0.026
Ballistic 2	15	2.1	6.7	0.063	0.097	0.0097
Ballistic 3	5	4.2	6.7	0.59	0.11	0.034
Ballistic 4	15	4.2	6.7	0.34	0.21	0.017
Ballistic 5	5	21.1	6.7	7.86	0.49	0.042
Ballistic 6	15	21.1	6.7	5.60	0.99	0.024
Ballistic 7	5	42.1	6.7	20.5	0.82	0.042
Ballistic 8	15	42.1	6.7	14.8	1.61	0.025
Ballistic 9	15	2.1	8.5	0.54	0.23	0.038
Ballistic 10	15	4.2	8.5	1.45	0.42	0.041
Ballistic 11	15	2.1	10.0	1.25	0.48	0.041
Ballistic 12	15	4.2	10.0	3.93	0.78	0.041

Fig. 13 Transient evolution of the blowing ratio and Reynolds number at different points on the surface for a ballistic re-entry with $v_{\text{entry}} = 6.7 \text{ km s}^{-1}$, $\gamma = 15^\circ$ and $\beta = 21.1 \text{ kg m}^{-2}$. Comparison with empirical findings from experiments.

and $F = 0.04$. In the case of the investigated trajectory in Fig. 13, the region of too high Reynolds numbers is crossed at late times of the re-entry.

Figure 14 shows the same $Re_s - F$ -footprint, as described above for a variation in the ballistic coefficient with $\gamma = 15^\circ$ and $v_{\text{entry}} = 6.7 \text{ km s}^{-1}$. The transient evolution of the blowing ratio (left) and mass flux $\dot{m}_A(s, t)$ (right) are shown. In all three cases, the line featuring the highest blowing ratio/mass flux belongs to the stagnation point. It is common to the three footprints that the blowing ratio rises very rapidly at early times of the trajectory and decreases slowly toward the end of the flight. The reason for this is the low internal convective cooling efficiency at low mass flow rates. At early times the internal cooling does not contribute significantly to the overall cooling. Hence, the protective film external to the wall is the main mechanism for the cooling which requires a larger blowing ratio. Later in the trajectory, the free stream density increases and more coolant mass flux is required to sustain a high blowing ratio. This also leads to a much better internal cooling and an improved overall system performance. For all three examples, the maximum blowing ratio occurs in the laminar regime and is below the critical value of $F = 0.04$. This highlights that no premature tripping of the boundary layer is expected at early times of the trajectory. However, the upper Reynolds number limit of $5 \cdot 10^5$ is crossed by the $\beta = 4.2$ and 21.2 kg m^{-2} trajectories. The $\beta = 4.2 \text{ kg m}^{-2}$ trajectory only barely oversteps this boundary whereas the $\beta = 21.1 \text{ kg m}^{-2}$ trajectory clearly features much higher Reynolds numbers at various injection points. Due to these high Reynolds numbers, natural transition would be expected for $\beta = 21.1 \text{ kg m}^{-2}$

Fig. 14 Transient evolution of the blowing ratio (left) and the mass flux (right) at all locations on vehicle surface and the respective Reynolds number along the trajectory of three ballistic coefficients ($v_{\text{entry}} = 6.7 \text{ km s}^{-1}$, $\gamma = 15^\circ$).

Fig. 15 Transient evolution of the blowing ratio (left) and the mass flux (right) at all locations on vehicle surface and the respective Reynolds number along the trajectory for various entry velocities ($\beta = 2.1 \text{ kg m}^{-2}$, $\gamma = 15^\circ$).

where transpiration cooling is not actively harmful but simply less effective. The best performance is clearly seen for the $\beta = 2.1 \text{ kg m}^{-2}$ trajectory where none of the boundaries are crossed.

Figure 15 shows the influences of a varying entry velocity on the required blowing ratio and mass flux, respectively. The figure shows three footprints of trajectories with $\beta = 2.1 \text{ kg m}^{-2}$ and $\gamma = 15^\circ$. A very strong increase of both the required mass flux and blowing ratio is observed when the entry velocity is higher than 6.7 km s^{-1} . For the two high velocity cases, $v_{\text{entry}} = 8.5 \text{ km s}^{-1}$ and $v_{\text{entry}} = 10 \text{ km s}^{-1}$, a larger entry velocity increases the maximum mass flux significantly. However, only small differences in the maximum blowing ratios are predicted. The blowing ratios needed for higher entry speeds cross the region of low Reynolds numbers but high blowing ratios ($F < 0.04$) which would lead to a decreased downstream film effectiveness or even early transition. However, a higher entry velocity only slightly increases the maximum injection Reynolds number and hence, transpiration cooling is still effective throughout most of the later times of the trajectory.

C. Comparison between steady state and transient calculations

Figure 16 provides a comparison between the findings from the transient and steady state analysis. The fully transient analysis incorporates the effects of heat capacity, heat conduction, and transient coupling to the fluid phase of the porous medium. Figure 16 shows an excerpt of Fig. 6 with the previously discussed ballistic trajectory paths plotted on top of the steady state result. The left part of Fig. 16 presents a variation of ballistic coefficients with a constant entry velocity. The steady state and transient findings coincide very well, i. e. the region of too high blowing ratio ($F > 0.04$) is never crossed by the trajectories, but the region of high Reynolds number is barely crossed by $\beta = 4.2 \text{ kg m}^{-2}$ and clearly crossed by $\beta = 21.1 \text{ kg m}^{-2}$.

The right part of Fig. 16 presents a variation of entry velocity with two different ballistic coefficients. It shows again that a high entry velocity results in crossing the regime of a too high blowing ratio that would result in early transition. A high entry velocity furthermore leads to a crossing of the region of too high Reynolds numbers later in the trajectory. In addition, significant radiative heat flux contribution is expected for this type of re-entry. However, for an entry velocity of 8.5 km s^{-1} and a ballistic coefficient of $\beta = 2.1 \text{ kg m}^{-2}$, the encountered injection Reynolds numbers and blowing ratios are within acceptable boundaries and transpiration cooling should be very effective. In conclusion, it is identified that transpiration cooling is very good for vehicles that remain in a hypersonic regime of low Reynolds numbers and lower ballistic coefficients. The close agreement between fully transient calculations and the steady state analysis shows that the findings presented in Figs. 6 and 7 are a suitable way to scope the viability of an actively cooled heat shield.

Figure 17 presents a comparison between the calculation of the total mass budget based on a transient (extracted from Table 2) and a steady state assumption. Steady state calculations are performed by using Fig. 6 as a look-up table. For each instant in time along a trajectory, the required mass flow rate is extracted depending on the current velocity and altitude. This determined mass flow rate is then integrated over time to yield the total required mass budget. The

Fig. 16 Required mass flow rate to keep the entire vehicle (UHTC) below 1800K with ballistic trajectory paths. $v_{\text{entry}} = 6.7 \text{ km s}^{-1}$ (left) and $\gamma = 15^\circ$ (right).

Fig. 17 Comparison between mass budgets calculated through transient and steady state analysis. $v_{\text{entry}} = 6.7 \text{ km s}^{-1}$ (left) and $\gamma = 15^\circ$ (right).

comparison shows that a steady state approach is not suited to assess the absolute mass budget for a certain trajectory. The general trend is replicated for both approaches and good agreement is achieved some cases. However, this is most likely due to different errors cancelling each other out. The discrepancy becomes more apparent when the re-entry becomes more extreme, either due to a higher entry velocity or a higher ballistic coefficient. In those cases, the transient nature of heat conduction becomes more important and hence a solution based on steady state assumptions does not suffice to accurately model the heat transfer processes. Therefore, a fully transient calculation as presented in this work is necessary to obtain total mass budget estimations.

IV. Conclusion

This paper presents the methodology and results of a system study on transpiration cooled heat shields for re-entry vehicles. A numerical model is used that employs semi-empirical and analytical correlations to assess the aerodynamic heat flux for hypersonic vehicles. The cooling performance has been assessed for a blunt cone geometry with a nose radius of 0.5 m and a 30° cone half angle. The analysis is split into a steady state investigation for sustained hypersonic flight and a transient investigation of various ballistic and lifting body re-entry trajectories. The investigated wall materials are a Carbon/Carbon composite ceramic and the Ultra High Temperature Ceramic material Zirconium diboride. Both materials require similar levels of mass injection. Spatially and temporally optimised injected mass flux is presented for various flight conditions. The majority of mass injection is required at the stagnation point and in the spherical segment while the conical part of the vehicle only requires cooling for extreme heating conditions.

It is shown that transpiration cooling features two regions in the altitude-velocity parameter space where a decreased film effectiveness or even premature tripping of the boundary layer is expected. One region is defined by a too high blowing ratio and concerns early times in the trajectory at high altitudes. A critical blowing ratio of approximately 0.04 is identified as the limit of good cooling performance. The overstepping of this boundary occurs for trajectories with high entry velocities above approximately 8.5 km s^{-1} . The second region refers to high Reynolds numbers at the injection locations along the vehicle. A critical Reynolds number of $5 \cdot 10^5$ is identified as the limit of this regime. Here, early transition or turbulent inflow conditions are expected where transpiration cooling would be less effective or even enhance heating. The crossing of this region occurs for vehicles with high ballistic coefficients above approximately 2.1 kg m^{-2} and for entry velocities above approximately 8.5 km s^{-1} .

Steady state analysis has shown that transpiration cooling has its greatest potential for moderate flight conditions with velocities below 8 km s^{-1} and altitudes above 40 km. The transient analysis shows good agreement with this assessment and confirms that transpiration cooling is most viable for entry scenarios where with entry velocities below 8.5 km s^{-1} and ballistic coefficients below 2.1 kg m^{-2} .

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