Thermal Protection Systems

Dec,1,2020

By:

Alexander J Snyder

Table of Contents:

Abstract:

Film cooling, transpiration cooling, and a combination of the two active cooling methods are discussed and compared in detail. Active cooling techniques are the future of hypersonic vehicle's thermal protection systems, due to their ability to decrease thermal and pressure loads acting on the body of a hypersonic vehicle. Film cooling provides adequate cooling at the stagnation point but fails to cool at the sides of a hypersonic nosecone. Transpiration cooling cools effectively at the sides of hypersonic nosecone but does not provide enough cooling and pressure reduction near the stagnation point. The combination of film cooling and transpiration cooling provides the highest cooling efficiency and can protect the hypersonic nosecone at the sides and around the stagnation point. The combination of film cooling and transpiration cooling can cause surface heating of 1800 kelvin to be cooled to 800 kelvins over a 66 percent decrease in surface static temperature.

Definitions:

Hypersonic Flow-Flow Traveling at a Mach number greater than 5
Passive Cooling- Cooling that does not change in flight and does not attempt to affect aerodynamic forces or boundary layers actively.
Active Cooling- Cooling that does change in flight and does attempt to affect aerodynamic forces and boundary layers actively.
Bow Shock – A detached shock that sits generally in the front of a highspeed vehicle that acts normal to vehicle at the center but bends at the ends of the shock to follow the surface of the vehicle.
Reattachment Shock- Shock that occurs when the bow shock is altered, and the shock adjusts and creates another shock wave.
Diffusivity-The process whereby particles of liquids, gases, or solids intermingle as the result of their spontaneous movement caused by thermal agitation and in dissolved substances move from a region of higher to one of lower concentration [1].
Stanton number- Dimensionless number that measures the heat transferred.
Pressure Ratio-The ratio between the free stream total pressure and the opposing jet/transpiration jet total pressure.
Mach Disk- When a bow shocks interaction with a fluid that causes the shockwave to normalize.

Introduction:

Traveling at five times the speed of sound is the definition of hypersonic flow. Due to the high level of kinetic energy, shockwaves, and surface friction aero heating is very intense on the surface of a hypersonic vehicle. This can be seen by the image shown in figure 1 as the velocity of the air is brought to zero, at the stagnation point, both the static temperature and pressure rise causing a risk of thermal and structural failure. Passive cooling methods will not work for sustained hypersonic flight due to the fact that during sustained flight cooling must be continuous to limit adverse effects, instead new techniques were developed to sustain continuous flight, this is what is known as active cooling. Different types of active cooling such as film cooling which pushes fluid off of the nosecone of hypersonic vehicle using either an opposing jet or an array of opposing micro-jets, platelet transpiration cooling which uses slots and offices to pushes fluid across the surface of the hypersonic vehicle, and a combination of the two cooling techniques that were discussed above have been both tested in labs or simulated using Computational Fluid Dynamics (CFD). The figures below show how hypersonic flow is tested in an experimental lab, a shock tube is used that uses a high pressure and low pressure system, two diaphragms, and a diverging nozzle to produce hypersonic flow in the test section. The shock wave placement, intensity, angle, and coolant location are shown using Schlieren imaging that captures a density gradient.

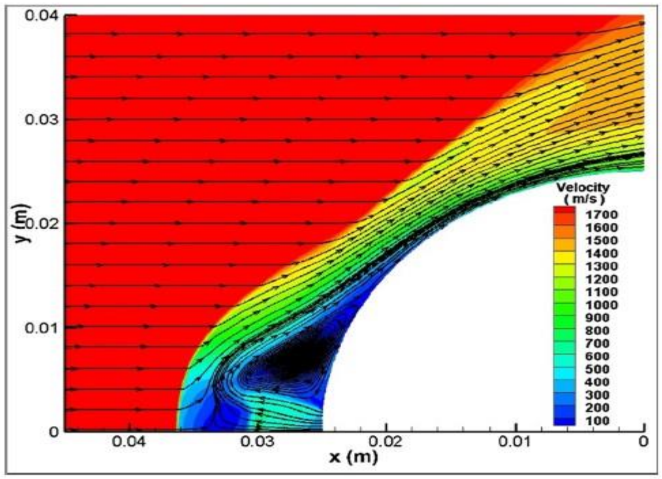


Figure 1 Velocity Flow Field [3]

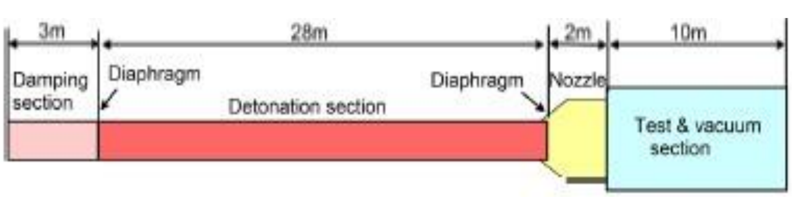


Figure 2 Shock Tube [15]

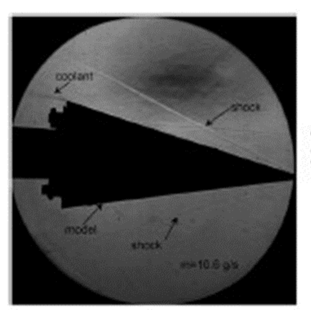


Figure 3 Schlieren Imaging [15]

Due to the control of fluid flow exiting the nosecone or leading edge of the hypersonic vehicle further investigations needed to take place to determine how temperature of the coolant, flow rate of the coolant, type of coolant, distribution of coolant, diffusivity of coolant, film thickness of coolant, and pressure of the coolant effect the surface evaporation in the case of liquids and surface insulation in the case of gases and how the static surface temperature and pressure of a hypersonic vehicle was affected. The image below shows that the shock wave when using the opposing jet or opposing array of microjets does not alter in location, which means desired flight conditions can be maintained.

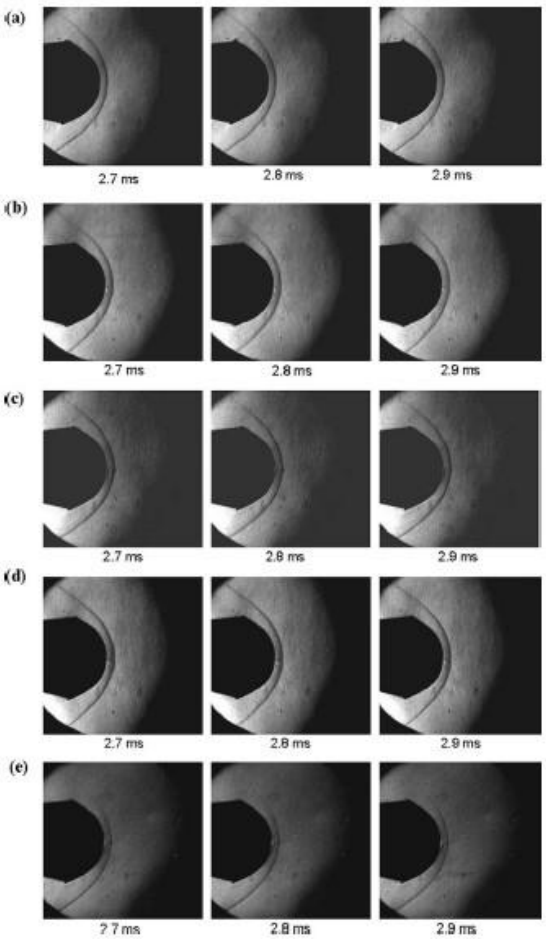


Figure 4 Shock Wave Placement [15]

In this review a deeper look will be shown in to different variables that have been tested to determine what is the most effective techniques that has been produced to date to sustain hypersonic flight, and what research is going to be conducted in the future to make advancements in hypersonic flight.

Passive Cooling with Example:

Figure 5 Re-Entry Vehicle [2]

Hypersonic flow has initially been seen when re-entry vehicles reenter the earth's atmosphere around a Mach number of 23. These types of capsules use a device called an ablative heat shield, the ablative heat shields normally burns off as the re-entry vehicle passes through the Earth's atmosphere and protects the astronauts from intense heat caused by high kinetic energy and surface friction. The use of the ablative heat shield is a device that does not cool continuously, and its trajectory and aerodynamic forces cannot be changed during flight. The ablation heat shield is a great example of passive cooling, in passive cooling the fluid forces around the surface of the capsule are not being attempted to be altered, and the surface is not attempting to be cooled, but instead the main concern for these types of vehicles are to with stand the heat produced so that the heat doesn't melt the personal or electrical components inside [2]**.** In the case of hypersonic flight sustaining vehicles the vehicle must be able to change flight patterns/trajectory and withstand the thermal and pressure loads, therefore an ablation heat shield will not work.

Film Cooling:

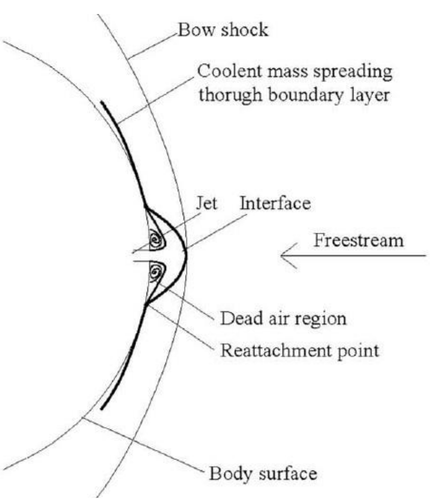


Figure 6 Film Cooling Single Jet [14]

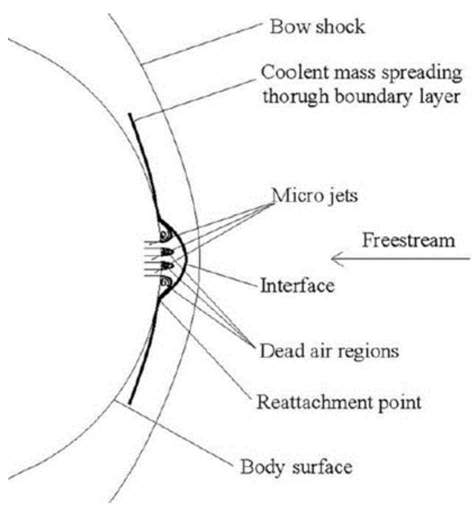
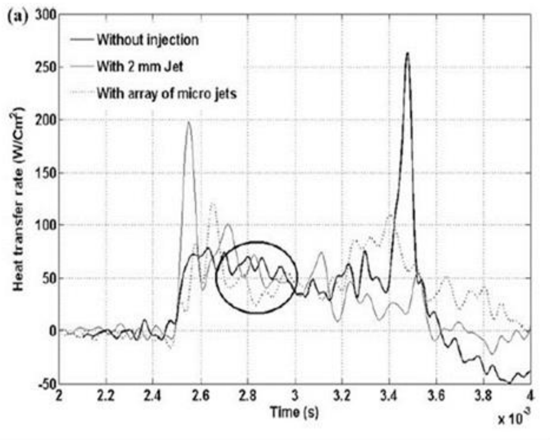


Figure 7 Film Cooling Array of jet [14]

Figure 1 and figure 2 that are shown above are diagrams that demonstrates theoretically how film cooling occurs. Film cooling is an active cooling technique that uses a fluid and an opposing jet or an array of opposing microjets to cool the surface of a hypersonic vehicle, and to control the placement of the bow shock that usually occurs at the stagnation point of the nosecone. The high pressure caused by the free stream velocity being brought close to zero at the stagnation point pushes the fluid flow against the surface of the hypersonic vehicle. Due to the jet or array of microjets contacting the bow shock, a Mach disk is formed which causes a reattachment shock to be created from the original bow shock. The temperature where the reattachment shock contacts the hypersonic vehicle has an extremely high heat flux. R. Sriram and G. Jagadeesh determined that the angle between the reattachment shock and surface of the hypersonic vehicle directly corresponds to the intensity of the heat flux at the surface [14]. Also, the opposing jets or array of opposing microjets causes a recirculation region to occur that provides a mixing section between the free stream flow and fluid from the film cooling. The recirculation region provides an internal boundary layer that holds in low speed flow and a higher molar fraction concentration, this recirculation region causes a barrel shock to occur at a lower Mach number then the original bow shock creating a smaller temperature increase at the surface of the hypersonic vehicle[3]. The mixing section provides a transient section where the active cooling is most effective, as the fluid reaches steady state characteristics the fluids cooling efficiency decreases [14]. An array of jets vs a single jet has been tested experimentally and computationally to determine if the reattachment shock, barrel shock temperature, and pressure across the shock can be lessened by using either a single jet or an array of microjets.

Array of Jets:

The experimental values that were determined by Sriram, R. and Jagadeesh, G are shown below. The figure shown below on the right side (figure 8a) is a heat flux vs time plot where the parameters are a pressure ratio of 1.2, Nitrogen gas as the coolant, and a free stream Mach number of 5.9. It is important to note, the experimental test roughly lasts around 3 microseconds, around 3.4 microseconds a large heat flux occurs this is due to noise after the experiment has been completed and are not included in comparison or results [14]. The figure on the left (8b) is from the zoomed in circled section from figure 8a, from this figure it can be determined that the microjet array out preformed both the without inject and the 2mm jet models. This can be explained theoretically because the jet array provides a smaller angle between the reattachment shock and the hypersonic vehicle [14]. The array of jets also provides a further spaced out and larger mixing region that also decreases the heat flux at the surface of the hypersonic vehicle.



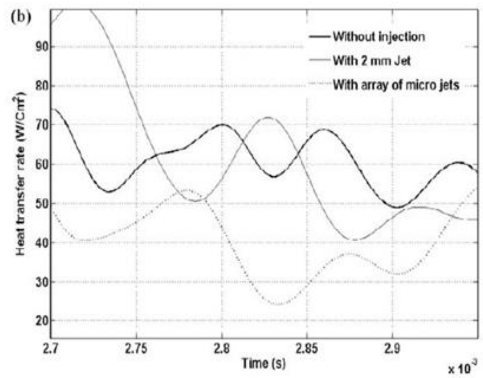


Figure 8a Jet Array Comparison [14]

Figure 8b Zoomed in Jet Array Comparison [14]

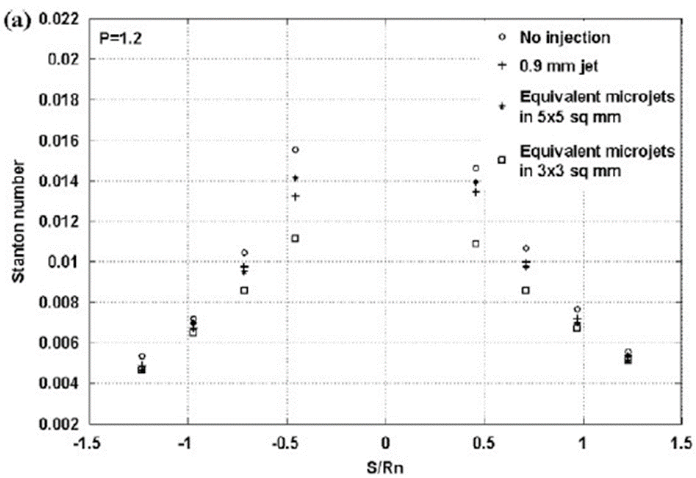


Figure 9 Microjet Spacing [3]

The figure on the left shows a comparison between no injection, a single opposing jet, a 5 by 5 mm area spacing of microjets, and a 3 by 3 mm area spacing of microjets [3]. The results showed that the 3 by 3 mm area array of microjets performed better than any of the other tested models. It was determined that if the area between the microjets becomes to large the microjets begin to act more independently, like a smaller pressure ratio set of single jets. Interestingly, at a larger spacing, when the microjets were acting most independent, the experimental Stanton number was most like no active cooling at all. The figure on the right shows a photo representation of physical shockwave and its

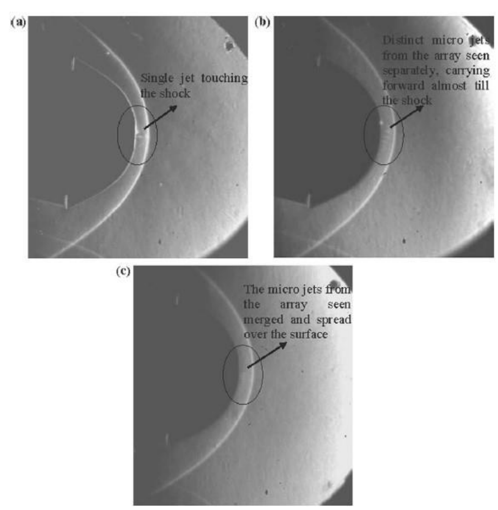


Figure 10 Photo of Experimental jets [3]

interaction with the opposing jet and microjets. It can be observed that the single jet provides a steeper bow shock then the array of microjets meaning a higher heat flux on the surface.

Changing Fluid Type:

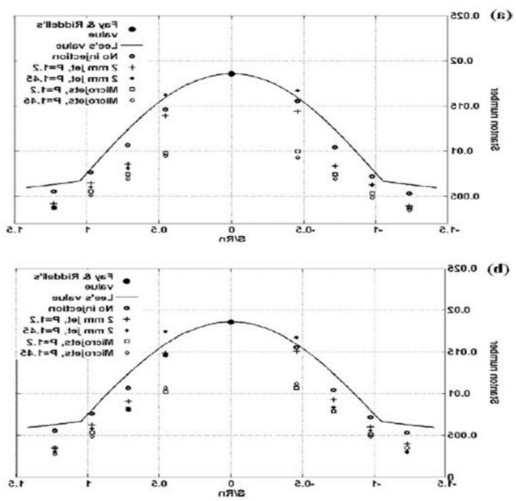


Figure 11 Nitrogen (top) and Helium (bottom) Stanton number Vs. Location from Stagnation Point [14]

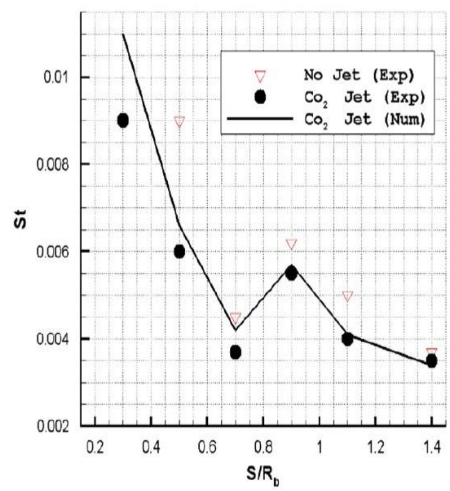


Figure 12 CO2 Stanton number Vs. Location from Stagnation Point [3]

The figures above show a comparison of the Stanton number vs a dimensionless location on the hypersonic vehicle using Nitrogen (Top Left), Helium (Bottom Left) and CO2 (right) as a coolant. The image on the left will be discussed in greater detail in a future section when changing the jet(s) pressure ratio is considered. The images above show that all three fluids protect the surface of the hypersonic vehicle similarly, this leads to the understanding that other fluid properties such as diffusivity needed to be considered. In hypersonic film cooling, generally a gas is chosen to create a film over the hypersonic vehicle, but Yuan, C., Li, J., Jiang, Z. and Yu, H have proven that water can also be used in film cooling as seen in figure 13. Liquid Film cooling is not generally considered due to the weight requirement that would need to be placed inside of the hypersonic vehicle.

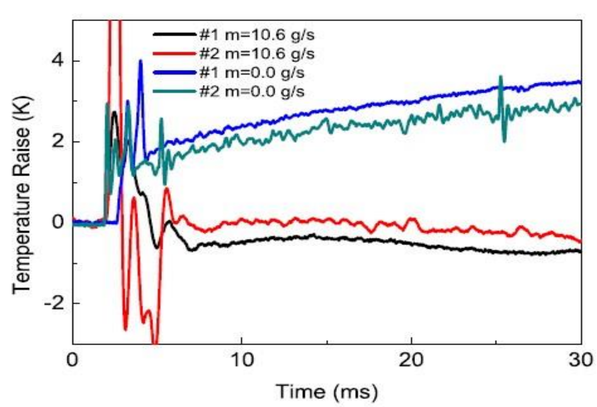


Figure 13 Liquid Film Cooling [15]

Film layer Width and Thickness:

The thickness and the width of the fluid was determined to have a positive correlation with the effectiveness of active cooling [14]. Yuan, C., Li, J., Jiang, Z. and Yu, H completed further testing to determine which factors increased the thickness and width of the surface film. Figure 14 a and b show that as the mass flow rate of the fluid was increased that the film thickness did not increase, but the rate at which the fluid reached probe 1,2, and 3 did increase. However, figure 14 c shows that as the mass flow rate increased the overall width increases which does increase the overall cooling efficiency [15].

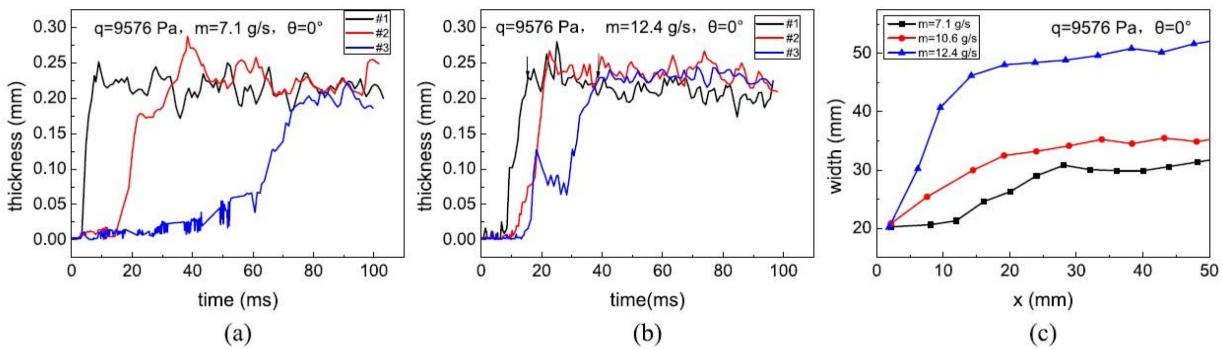


Figure 14 Liquid Film Cooling Thickness [15]

Figure 15 shows that as the pressure from the opposing jet is increased the overall thickness of the film also increased. This provides the understanding that as the pressure increases the cooling efficiency also increases [15]. It is important to note that the mass flow rate and angle of attack were held constant throughout the pressure effect experiment. This result makes sense theoretically because as the pressure of the jet is increased it provides a higher pressure to push against the highest free stream pressure acting at the stagnation point.

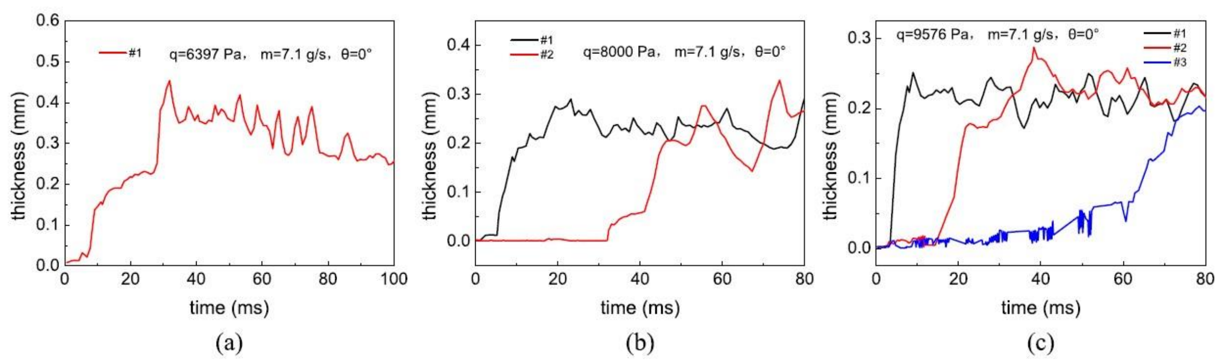


Figure 15 Liquid Film Cooling Thickness [15]

<u>Change in Pressure Ratio</u>:

The figure, from Barzegar Gerdroodbary, M., Imani, M. and Ganji, to the right compares the flow field of both a single jet and an array of microjet setup using helium as a coolant and a free stream mach number of 6. From figure 16 it can be seen that at lower Pressure ratios (PR) that a single jet provides more coverage on the nosecone then the array of microjets [4]. The figure also shows that the angle steepness between the nosecone and flow field increase at the edges as the distance from the nosecone increases. At the larger PR the microjets and single jet show similar flow fields. This leads to the understanding that at high pressure ratios microjets can be used in place of single jets with lesser reattachment shocks as determined from before.

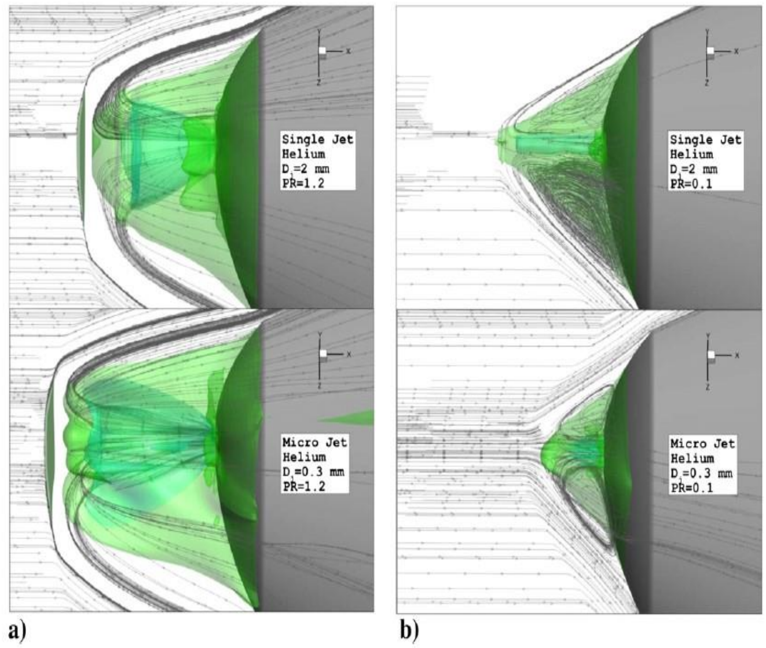


Figure 16 Flow Field Microjet Vs jet [4]

Lu, H. and Liu, W. determined that the Stanton number decreases as the measured point moves further away from the stagnation point. It was also determined as the pressure ratio of the microjets increased the Stanton number decreased. Also, that as the pressure ratio was increased the rate of change of the Stanton number decreased [8].

The effects of a higher-pressure ratio causing an increase in the free stream flow field penetration will be discussed further in the LPM and SPM section.

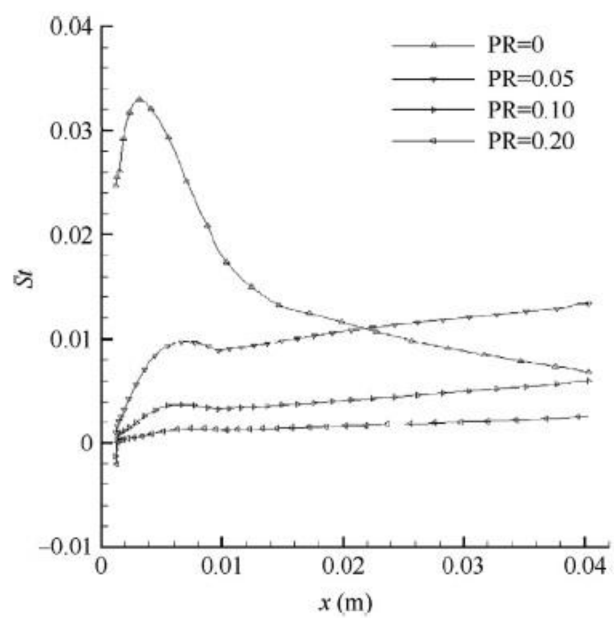


Figure 17 Stanton Number Vs Positions [7]

The experimental effects of changing total pressure ratio can be seen in figure 17 [14]. The figure shows that as, discussed previously, the pressure ratio increases the heat transfer that occurs on the surface of the hypersonic vehicle decreases.

SPM, LPM, and Diffusivity:

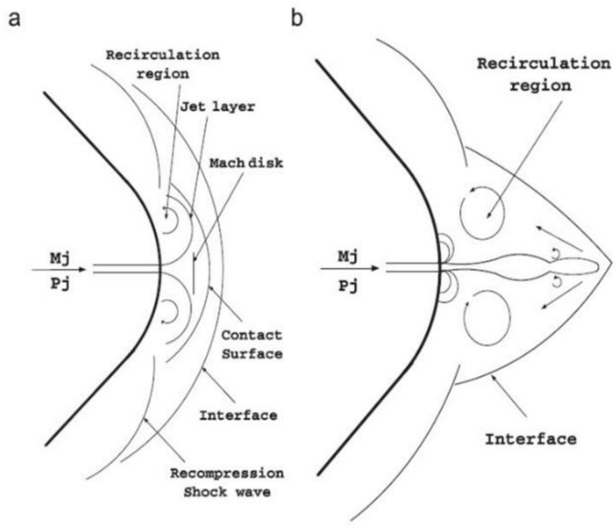


Figure 18 Short Penetration Mode and Long Penetration mode [3]

A detailed investigation between short penetration mode (SPM) and long penetration mode (LPM) was researched heavily by Barzegar Gerdroodbary, M., Bishehsari, S., Hosseinalipour, S. and Sedighi, K to determine the difference between the two methods. It was determined that the penetration type was dominantly controlled by the pressure ratio of the opposing jet and the diffusivity of the gas that is being used for film cooling [3].

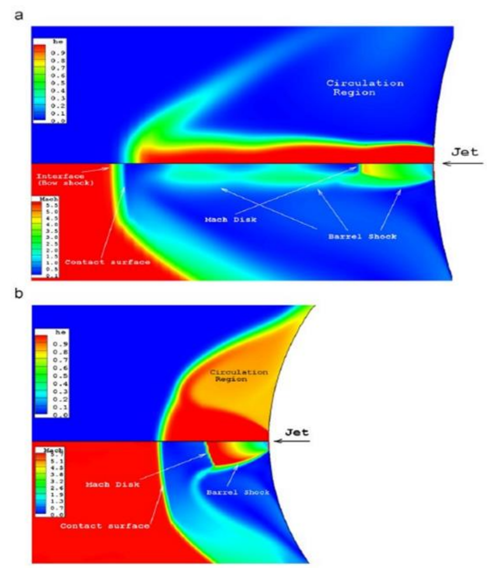


Figure 19 Mixing Region and Mach Number [3]

It was discussed earlier that the further away the penetration mode moves away from the nosecone the greater the reattachment shock and heat flux at the connection point between the reattachment shock and hypersonic vehicle surface. The figure to the right shows how helium, which is a highly diffusive gas ,in long penetration mode has no amount of helium in the recirculation region, where in contrast in the short penetration mode the helium is found in high amounts in the recirculation regions. The cooling efficiency is highest when a higher concentration of gas is found within the mixing region of the opposing jet(s) [3]. This leads to the conclusion that if a highly diffusive gas is used in a short penetration mode with an array of microjets that high cooling efficiency can be achieved. Figure 20 shows a comparison between He, which is a highly diffusive gas, and

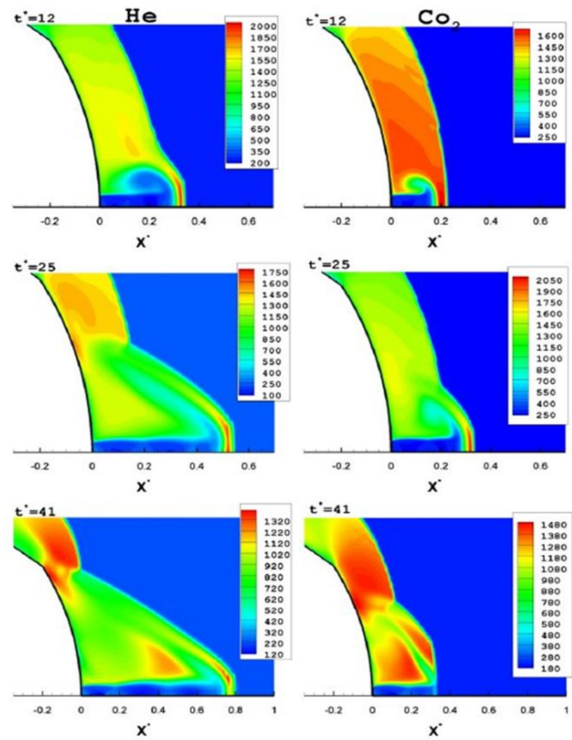


Figure 20 Example of Diffusivity [3]

CO2, which is a less diffusive gas, at three different nondimensional time steps [3]. It can be observed from the figure to the right that as the time steps increase the He moves quickly to the right, where the CO2 moves very little when compared to its original penetration. The shorter penetration in He is effective because even though it is not initially pushed far off the nosecone by a high-pressure ratio it diffuses easily into the free stream. The high diffusivity allows for a large amount of mixing to take place which can easily be seen in the figure to the right, from earlier discussion it was determined that a larger mixing region provided better cooling efficiency. The results of the higher cooling efficiency are shown by the temperature decreasing faster using He (left) as a film cooling fluid then when using CO2 (right).

From the research conducted on film cooling many positive take away can be observed, including a reduction in static temperature, an understanding of the effects of a gases diffusive characteristics, how an increase in pressure ratio positively effects the cooling efficiency, how microjets can limit reattachment shocks heat flux at the stagnation point, and how a long penetration mode and short penetration mode effect the free stream flow field.

Issues with Film Cooling:

Film cooling has two main issues that limits its ability to actively cool.

1) Film cooling does a sufficient job of stopping the aero heating that takes place at the stagnation point, but even with an array of microjets as the heat flux is measured further away form the film cooling the heat flux rises because of the lack fluid film distribution on the sides of the nosecone.

2) The number of microjets that protrude from the nosecone are limited on the sides of the nosecone, the nosecone aerodynamic forces/boundary layers start to get effected if the microjets veer too far away from the stagnation point.

Transpiration Cooling:

Figure 21 Transpiration cooling [9]

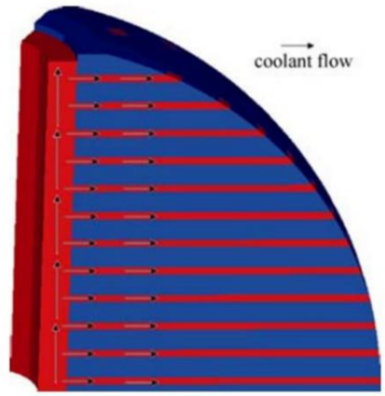


Figure 22 Channels of Transpiration cooling [9]

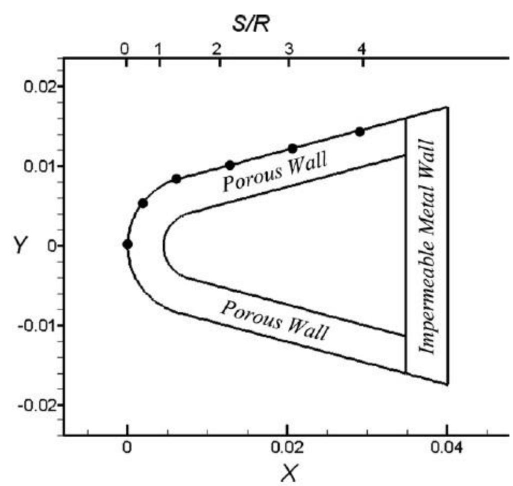


Figure 23 initial transpiration cooling [9]

Transpiration cooling, unlike film cooling does not use an opposing jet at the center of the nosecone to push the bow shock of the surface of the hypersonic vehicle. Instead, it uses a combination of channels to provide available cooling on the surface of nosecone/leading edge. Similarly, once the fluid is placed on the surface of the nosecone/leading edge the pressure force caused by the high speed flow being brought theoretically to rest pushes the fluid against the surface of the hypersonic vehicle. This causes a thermal protection boundary layer to be formed that protects against the aero heating that is caused by the shock wave, surface friction, and high levels of kinetic energy. Initially, Transpiration cooling used porous like materials such as a sponge which can be seen in figure 23 [9]. Due to the high pressure loads placed on hypersonic vehicles this type of transpiration cooling could not be used, instead platelet transpiration cooling was designed so that fluid could flow through channels, and allowed for the nosecone/leading edge to be made out of strong materials that could withstand the pressure loads and can be actively cooled to withstand the thermal loads. The facts from film cooling about single versus array, fluid type, “film” thickness and width, change in pressure ratio, SPM, LPM, and diffusivity remain true as transpiration cooling is being discussed.

Thermal Boundary Layers:

The figure below demonstrates a computationally calculated flow field from the channels of platelet transpiration cooling and its interaction with a bow shock. The transpiration cooling creates a boundary layer around the nosecone/leading edge to protect it from the aero heating that would naturally occur at the surface. The image shows a contour of static temperature from orange being the hottest temperature around 1300k and the blue being the coolest temperature around 300k which is at the boundary layer [5]. This area is expected to be the coolest because it is located at the furthest point from the stagnation point. Similarly, to film cooling, it can be observed that as the cooling layer subsides there is an increase in static temperature where this acts like a much weaker comparison to the phenomena of a local heat flux increasing at the location where the reattachment shock meets the surface of a hypersonic vehicle. Unlike one of the main issues of film cooling, platelets can be placed completely along the side of the nosecone/leading edge making it much easier to place than an array of microjets without the worry of them acting independently.

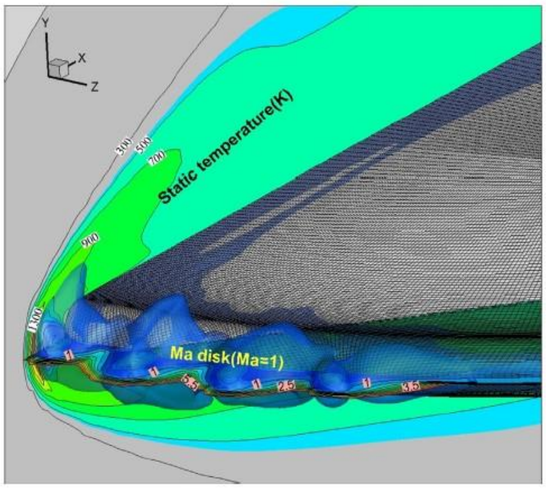


Figure 24 thermal boundary layer [5]

Shibin Li, Wei Huang, Jing Lei, Zhenguo Wang determined the same results in another study conducted from Li, Shibin. The figure to the right (25) shows the flow

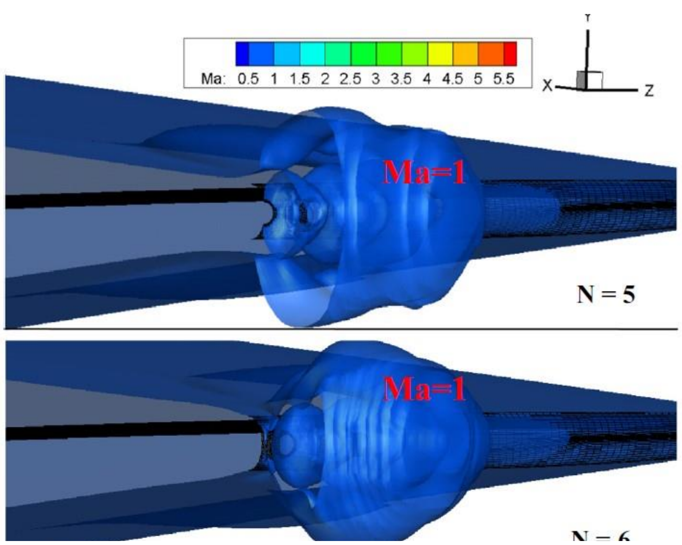


Figure 25 distribution of flow around microjet [6]

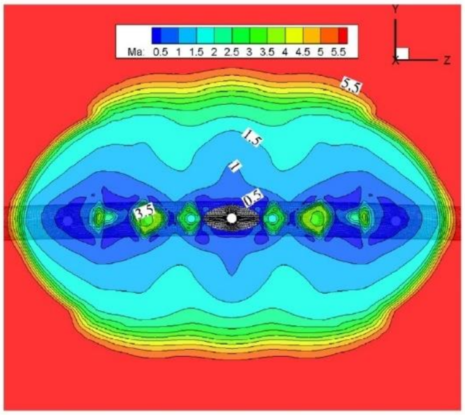


Figure 26 Mach Number Distribution [6]

field at an individual platelet [6]. Figure 25 is a computationally simulated platelet providing a Mach number of one to push against the free stream Mach number. Figure 26 shows how the free stream Mach number is affected by the platelet jets that are shown above in figure 25. The figure to the left shows that as the distance from the platelet increases the Mach number in the freestream also increases [6]. Li, Shibin also shows that the set of platelet jets acting on a leading edge/nosecone work as a group to minimize the kinetic energy that gets transferred to aero heating [6]. These results are like an array of microjets being spaced appropriately to work together to reduce the aero heating at the stagnation point of a nosecone.

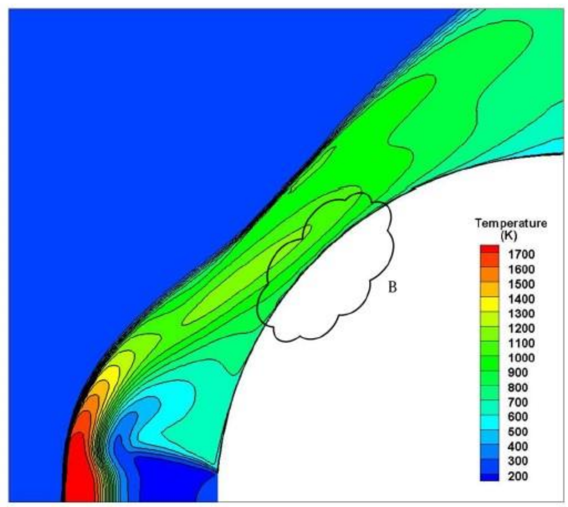


Figure 27 transpiration cooling flow field [12]

Shen, Yin, Zhang, and Liu took a deeper look into the boundary layer formed at the surface of the hypersonic vehicle. From figure 27 one of the fatal flaws of transpiration cooling can be observed, the cooling is adequate at the sides of the nosecone/leading edge but at the tip of the nosecone the surface temperature is 1700 kelvin. Figure 28 shows the interaction between the flow field around the surface of the hypersonic vehicle and a thin thermally protective boundary layer being created by the platelet jet’s transpiration cooling. Figure 28 also shows that the film layer is continuous and follows the shape of the hypersonic vehicle [12]. Dissimilarly, to film cooling specifically at the nosecone, a film boundary layer is not produced along the nosecone when using an array of microjets due to the intense pressure force and the protrusive nature of the active cooling,

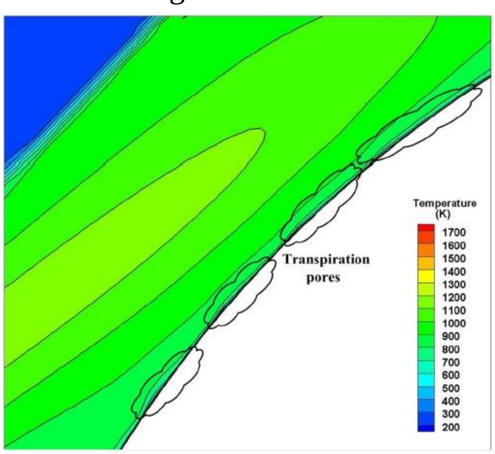


Figure 28 zoomed in transpiration cooling flow field [12] making platelet transpiration cooling a more beneficial form of active cooling for this specific task.

Static Temperature and Pressure effects at Different Pressure Ratios:

The images below show how the static pressure and temperature are affected by changing the total pressure ratio between the orifice jet and the free stream [5]. Similarly, to film cooling, as the pressure ratio is increased the static temperature and pressure at on the surface of the hypersonic vehicle are reduced. Unlike film cooling, as the total pressure ratio is increased in transpiration cooling a long penetration mode is not created and therefore minimizing any type of reattachment shock to occur that would increase the local heat flux. This fact can be observed in figure 29, as the pressure ratio increases only a slight increase in distance interaction with the flow field is observed. The figure below and to the left (30) Shows that as the pressure ratio is increased the pressure acting on the surface of the hypersonic vehicle is decreased, thus limiting the structural forces that are acting on the body of vehicle. These two observations that were observed and completed by Fan, W., Li, S., Zhou, J., Huang, W., Ou, M. and Zhang, R provide the result that transpiration cooling effectively provided an essential amount of active cooling at the sides of a nosecone, as well as provided enough structural resistance to avoid structural failure at hypersonic speeds [5].

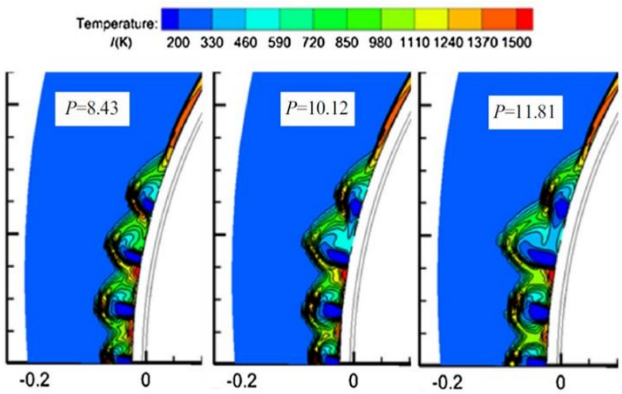


Figure 29 Static Temperature at different pressure ratio [5]

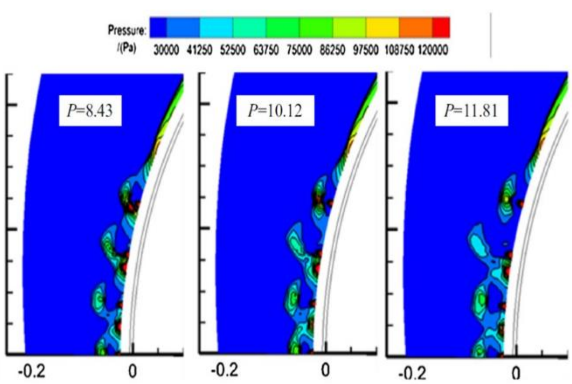


Figure 30 Static Pressure at different pressure ratios [5]

Similarly, an independent study of the previously discussed pressure reduction observations was conducted by Li, Shibin who also determined that an array of orifice jets could reduce the static pressure acting on the surface of the hypersonic vehicle. He also determined that as the number of orifices increased so did the pressure reduction [6]. Unlike the microjet array because the orifices did not protrude into the free stream flow no affect was noted about a decrease in aerodynamic performance. Shibin also determined, similarly to microjets, that as the number of orifices increased that radii of the orifices could be decreased [6].

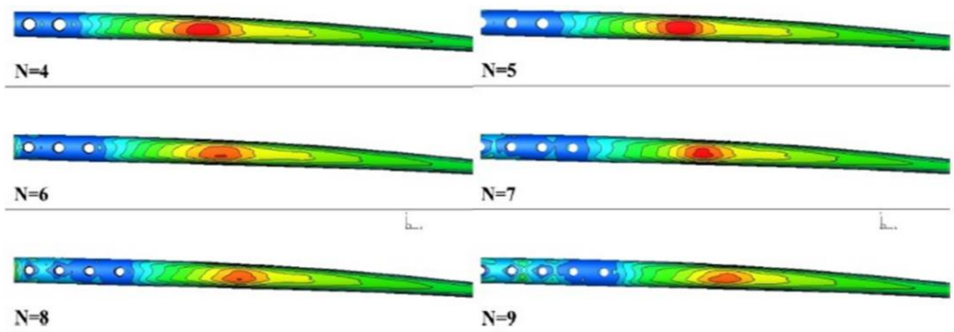


Figure 31 pressure reduction with number of orifices [6]

From the research conducted on transpiration cooling many positive attributes can be associated with transpiration platelet cooling including lack of large reattachment shock, an adequate pressure distribution, temperature reduction at the sides of a nosecone, a decrease in Mach number acting at the surface of the hypersonic vehicle, and thin thermally protective insulating boundary layer being produced.

<u>Issues with Transpiration Cooling:</u>

Transpiration cooling has many positive characteristics but does have a one main issue that needs to be addressed.

1) Transpiration cooling reduces pressures and temperature successfully at the sides of the nosecone, but due to the fact that it does not produce any form of long penetration mode flow the shock sit almost directly on the surface at the stagnation point which allows an extremely high static temperature and pressure at the stagnation point.

Combinational Cooling:

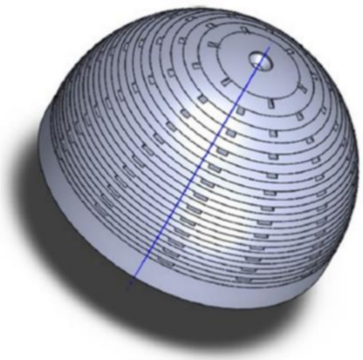


Figure 32 Combination Film and Transpiration Cooling [10]

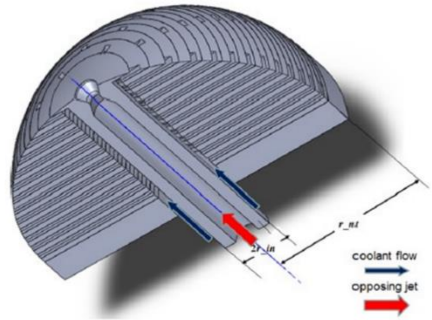


Figure 33 Cut to see internal opposing jet and channels in combination cooling [10]

In similarity to most things, if you have two methods that are working well but struggle in the area that the other one is exceeding in then it makes complete sense to combine them. Combinational film/platelet transpiration cooling is just that, it uses transpiration cooling to effectively cool the sides of the leading edge/nosecone of the hypersonic vehicle and an array of opposing microjets/film cooling to push the bow shock away at the stagnation point. Due to the reattachment shock, platelets are put close to the tip of the cone to begin forming a thermal protective boundary layer to help the microjets in reducing the local heat flux at the reattachment point. All facts that have been discussed about both film cooling and transpiration cooling remain true while analyzing the effects of combinational cooling.

Temperature Distribution:

The figure to the right depicts a nosecone with no type of active cooling shown with temperature scale in kelvin. As expected, extreme temperatures occur at the nosecone due to the high amount of kinetic energy, surface friction, and bow/reattachment/barrel shocks [10]. It also shows that as the contour got further away from the stagnation point the temperature drops, this is because the velocity begins to increase again which can be seen in the velocity profile shown in figure 1 [3]. The second image on the left shows just transpiration cooling taking place on the nosecone with the temperature scale also being in kelvin. From

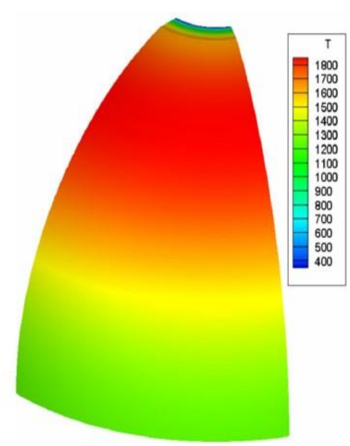


Figure 34 no type of active cooling [10]

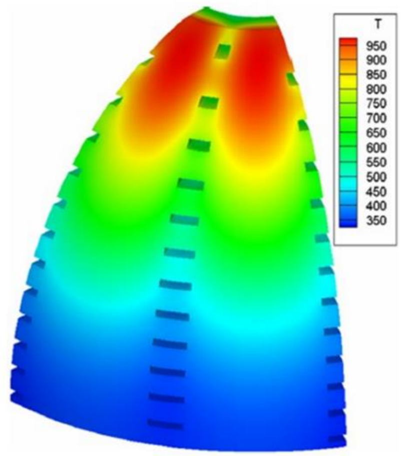


Figure 35 platelet cooling [10]

previous observations the transpiration cooling does an excellent job of cooling where the orifices are located and further down the sides of the nosecone. High temperatures occur at the nosecone because without the opposing microjets no force is there to push the bow shock off the nosecone. The third image, to the right, that was produced by Rong, Y. and Liu, W. shows the temperature contour, in kelvin, of a nosecone that is experiencing combinational film/platelet transpiration cooling. It is important to note, that all the contours are not on the same scale where the red portion in figure 33 is 1800 Kelvin but the red portion in figure 35 is only 800 Kelvin. Figure 35 shows that the temperature is even further reduced when using combinational cooling then instead of just using transpiration cooling. When comparing figure 34 and 35 it can be shown that the highest temperature region actually shifts further down the nosecone, this is because the opposing jets provides thermal relief dealing with the bow shock and as discussed before the transpiration cooling effectively cools the reattachment shock region. The results show a reduction in temperature from a max temperature of 1800 kelvin to 800 kelvin which is much

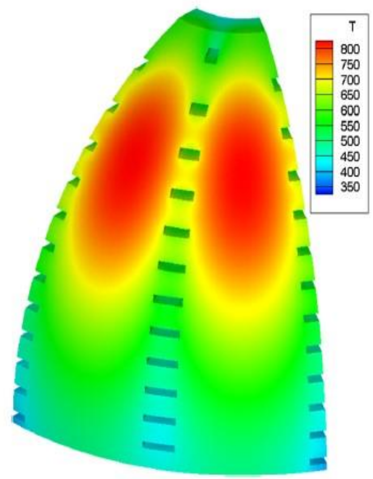


Figure 36 combinational cooling [10]

more manageable in terms of available materials. A secondary study was performed by Shen, B. and Liu, W. who produced similar results to that of Rong, Y. and Liu, W. One major difference between their computational predictions is that Shen, B. and Liu, W. looked at film cooling(opposing jet) only while of Rong, Y. and Liu, W. looked at transpiration cooling only, where it can observed as expected that the opposing jet provided better cooling at the tip of the nosecone and transpiration cooling provided better cooling to the sides of nosecone. Similarly, both studies found that the combinational cooling to be the most successful form of active cooling. A second computational study completed by Shen, B., and Liu, W. shows how the pressure ratio and comparison between the mass flow rate of the opposing to the jet to the transpiration jet mass flow rate effected the temperature distribution on the nosecone [11]. Figure 38 shows that effective cooling can be produced by reducing the overall pressure ratio and by increasing the mass flow rate of transpiration platelet jet [11].

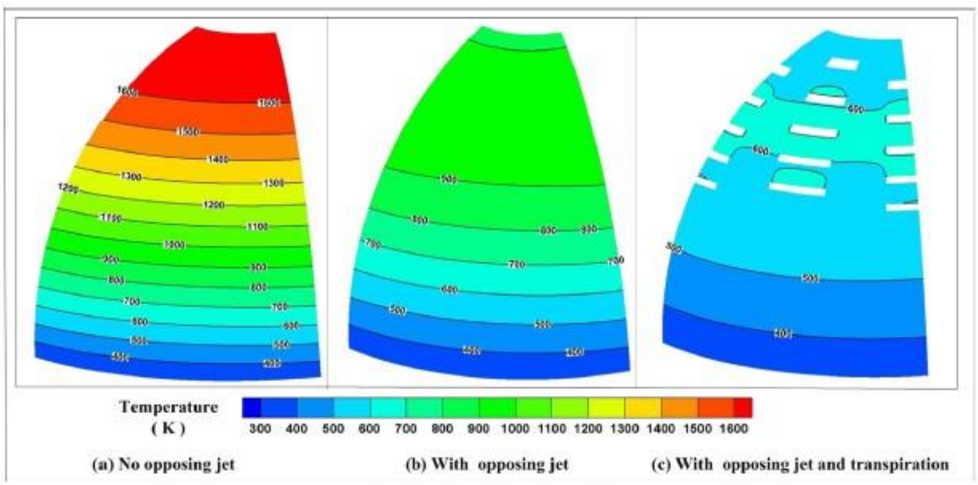


Fig. 18. Temperature contour of different TPS.

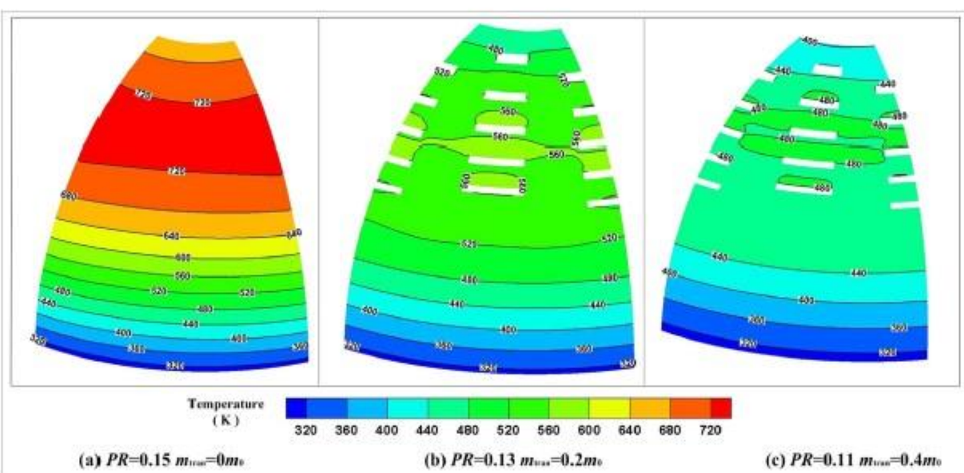


Figure 37 Further investigation between different forms of active cooling [11]

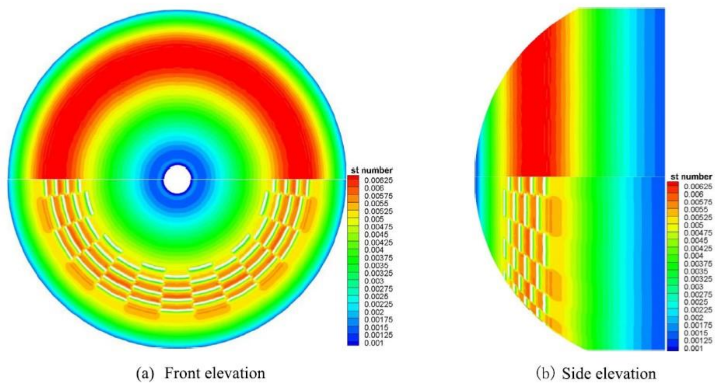


Figure 38 transpiration cooling vs no transpiration cooling [12]

In a secondary study completed by Shen, B., Yin, L., Liu, H. and Liu, W. c they determined that a surface heat flux decreased when using transpiration cooling then if transpiration cooling was not used [12]. Which follows the trend found by Shen's first study and Rong's study.

Pressure Ratio Comparison between Opposing jet and Transpiration jet:

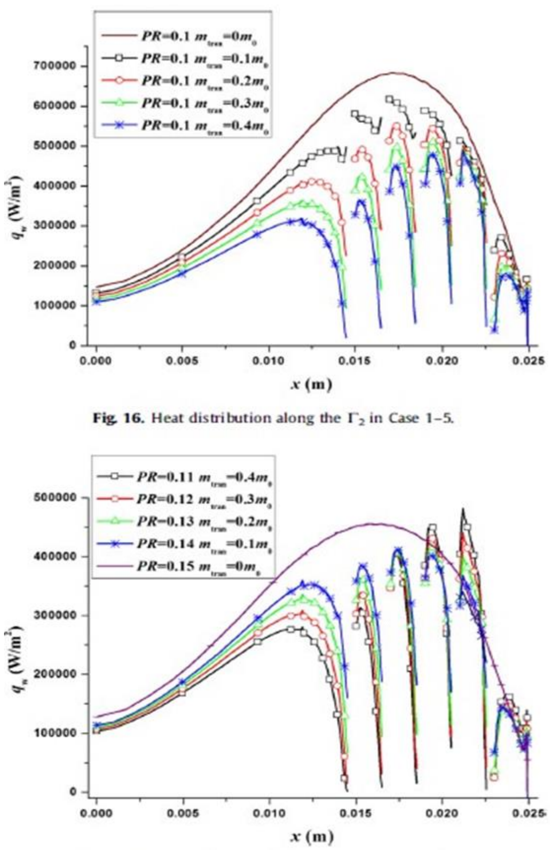


Fig. 16. Heat distribution along the Γ_2 in Case 1–5.

Fig. 17. Heat distribution along the Γ_2 in Case 6–10.

Figure 39 heat flux Vs. Pressure Ratio [11]

Similarly, to the results discussed in figure 38, figure 40 continues the discussion on how pressure ratio and mass flow rate between the transpiration and opposing jet effects the heat transfer at the surface of the hypersonic vehicle. The top figure shows that as the pressure ratio is held constant but the mass flow rate in the transpiration platelet jet is increased the overall heat flux decreased, similarly to the temperature decrease that was found from the research completed by Shen, B., and Liu, W. The bottom image from figure 40 shows that as pressure ratio is increased and the transpiration platelet jet mass flowrate is decreased that the surface heat flux increased. This leads to the understanding that increasing the transpiration platelet jet mass flow rate has a positive correlation to the decrease in heat flux, while increasing the pressure ratio does not guarantee that the heat flux will decrease [11]. This can be explained by a larger pressure ratio causing a larger reattachment shock, which will cause a larger local heat flux at the reattachment point.

Orifices on Hypersonic Nosecone:

Figure 41 shows how the number of orifices effect the heat transfer on the surface of the hypersonic nosecone. The research from Shen, B., Yin, L., Liu, H. and Liu, W. shows that as the number of orifices increases the heat transfer decreases [12]. The heat transfer specifically decreases in between the spacing of orifices [12].

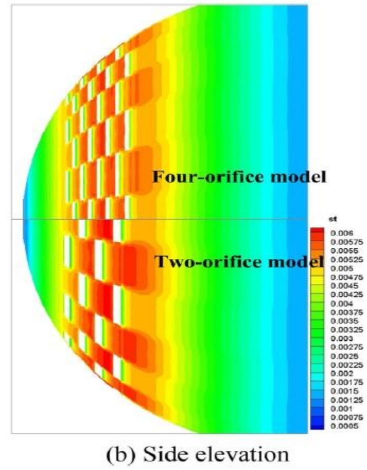


Figure 40 Comparison between number of Orifices [12]

Conclusions:

The following conclusions can be made from the information discussed above.

1) Film cooling provided adequate cooling at the stagnation point. As the pressure ratio increased the heat transfer rate on the surface of the hypersonic vehicle decreased. The pressure ratio has a direct correlation to a long penetration method and short penetration method, as the penetration method pushes further into free stream flow field the reattachment shock strengthens which leads to a stronger local heat flux at the reattachment point. An array of jets can be used in place of a single opposing jet to decrease the heat flux at the reattachment point. Diffusivity of fluid must be considered because as the fluid becomes more diffusive then pressure ratio can be decreased because the fluid spreads more easily into the freestream. Film cooling has two major setbacks, as heat transfer is measured furthered from the stagnation point film cooling doesn't cool the surface of hypersonic vehicle as adequately, and microjets cannot be placed all around the surface of a nosecone because the protruding jets would begin to cause undesired aerodynamic forces.

2) Transpiration cooling when being discussed with hypersonic design is specifically referring to platelet transpiration cooling. This type of film cooling creates a thin protective thermal boundary layer specifically at the sides of a hypersonic nosecone. Similarly, to standard film cooling, transpiration cooling provides better cooling and pressure force resistant as the pressure ratio is increased. Dissimilarly, transpiration film

cooling does not cause a large reattachment shock like film cooling does. The one major issue with transpiration cooling is that because it doesn't produce any type of long penetration method, which explains why there is not a reattachment shock, the bow shock sits directly at the surface of the hypersonic nosecone, this causes extremely high local heat flux and pressure forces at the stagnation point.

3) Combinational film cooling is the combination of opposing jet (film cooling) at the stagnation point to control the bow shock and transpiration cooling on the sides of the nosecone to deal with reattachment point as well as to form thin protective thermal boundary layer along the nosecone. This method out preformed both transpiration cooling and film cooling by themselves when it came to temperature distribution along the hypersonic nosecone. This has to do with the fact that both methods succeeded where the other method failed. This method also showed that pressure ratio can be fixed, and that as the mass flow rate of the transpiration platelet jets were increased the overall surface heat flux, temperature, and Stanton number were all decreased. The number of orifices were also explored for this method and it was demonstrated that as the amount orifices increased the temperature between the orifices decreased

All three methods are forms of active cooling; each cooling method was discussed in detail. A look to the future will be conducted about combinational cooling and how I believe it is the future of thermal protection systems in hypersonic flight.

Conflicts of Interest:

The author declares that there is not a conflict of interest in the paper if it were to be published.

Acknowledgements:

The author would like to thank Dr. Chuck Margraves for supporting, guiding, and mentoring the student to write an acceptable literature review about the information on thermal protection system that was described above. The author also thanks his coworkers for guidance and listening, questioning, and preparing the author for what will hopefully be a successful future.

Look to the future:

As thermal protection systems advance, it is generally decided that a combination style active cooling is the future of sustained hypersonic flight. It has also been generally determined that a highly diffusive gas such as helium will cause a lot of mixing at the stagnation point which will allow for a reduction in thermal and structural loads. On the sides of a hypersonic nosecone a film boundary layer must be formed through transpiration cooling to create a thin protective thermal boundary layer that will protect the leading edge from reattachment shocks and areas of high surface friction. It has been shown in limited studies that a non-diffusive gas is best used to hold a thin boundary layer because it doesn't drift from the platelet jet, this can be useful for such things as a the sides of a hypersonic nosecone. A combination cooling method generally only uses one type of fluid such as nitrogen or helium, but never a mixture of the two gases.

I believe that the future of hypersonic flight will use a microjet array and a combination of non-diffusive and diffusive gases such as helium to interact with the bow shock, but will use a gas such as nitrogen on the sides of the nosecone for transpiration cooling to create a protective boundary layer and to protect against the reattachment shock. Further calculations would need to be conducted about the combination of diffusive and non-diffusive gases to verify that as they interact with the free stream and each other that the two cooling gas do not have any adverse chemical effects at either high temperature or pressure such as combustion. A low-pressure ratio will be used with a higher mass flow for a transpiration cooling to reach the most effective cooling efficiency. Instead of using a platelet transpiration cooling channels, I believe it would be more beneficial to create a nosecone that has continuous circles grooved around the surface of the hypersonic nosecone to avoid areas that would not be reached by the transpiration cooling.

This new design with a combination of diffusive and non-diffusive gases will be tested first by using computational fluid dynamics by simulating hypersonic flow across the nosecone by using k-ω turbulence modeling around the stagnation point to accurately depict the recirculation around the nosecone. After an AutoCAD geometry, mesh refinements, and simulations have been finalized a physical nozzle will be manufactured and tested in a shock tunnel. Pressure ratios, static temperatures, static pressures, and heat flux rates will be recorded on the surface of the hypersonic nosecone. A journal article will be written about the comparison between the simulated and experimental results and the device's cooling efficiency will be written in terms of Stanton number.

References

[1] Definition of DIFFUSION." Merriam-Webster.com, 2019, www.merriam-webster.com/dictionary/diffusion.

[2] "NASA Completes Orion Heat Shield Milestone for Crewed Artemis-2 Mission." AmericaSpace, 9 July 2020, www.americaspace.com/2020/07/09/nasa-completes-heat-shield-milestone-for-artemis-2/. Accessed 16 Nov. 2020.

[3] Barzegar Gerdroodbary, M., Bishehsari, S., Hosseinalipour, S. and Sedighi, K., 2012. Transient analysis of counterflowing jet over highly blunt cone in hypersonic flow. Acta Astronautica, 73, pp.38-48.

[4] Barzegar Gerdroodbary, M., Imani, M. and Ganji, D., 2015. Investigation of film cooling on nosecone by a forward-facing array of micro-jets in Hypersonic flow. International Communications in Heat and Mass Transfer, 64, pp.42-49.

[5] Fan, W., Li, S., Zhou, J., Huang, W., Ou, M. and Zhang, R., 2019. Study on the drag and heat reduction performance of porous opposing jet in hypersonic flow. International Journal of Heat and Mass Transfer, 139, pp.351-361.

[6] Li, Shibin, et al. "Drag and Heat Reduction Mechanism of the Porous Opposing Jet for Variable Blunt Hypersonic Vehicles." International Journal of Heat and Mass Transfer, vol. 126, Nov. 2018, pp. 1087–1098, 10.1016/j.ijheatmasstransfer.2018.06.054. Accessed 30 Nov. 2019.

[7] Liu, Y., Jiang, P., Jin, S. and Sun, J., 2010. Transpiration cooling of a nosecone by various foreign gases. International Journal of Heat and Mass Transfer, 53(23-24), pp.5364-5372.

[8] Lu, H. and Liu, W., 2013. Research on thermal protection mechanism of forward-facing cavity and opposing jet combinatorial thermal protection system. Heat and Mass Transfer, 50(4), pp.449-456.

[9] Rong, Y. and Liu, W., 2010. Research on cooling effect with coolant groove structure parameters in platelet transpiration cooled nosetip. Journal of Thermal Science, 19(5), pp.438-444.

[10] Rong, Y., Wei, Y. and Zhan, R., 2016. Research on thermal protection by opposing jet and transpiration for high-speed vehicle. Aerospace Science and Technology, 48, pp.322-327.

[11] Shen, B. and Liu, W., 2019. Insulating and absorbing heat of transpiration in a combinational opposing jet and platelet transpiration blunt body for hypersonic vehicle. International Journal of Heat and Mass Transfer, 138, pp.314-325.

[12] Shen, B., Yin, L., Liu, H. and Liu, W., 2019. Thermal protection characteristics for a combinational opposing jet and platelet transpiration cooling nose-tip. Acta Astronautica, 155, pp.143-152.

[13] Shen, B., Yin, L., Zhang, X. and Liu, W., 2019. Investigation on cooling effect with a combinational opposing jet and platelet transpiration concept in hypersonic flow. Aerospace Science and Technology, 85, pp.399-408.

[14] Sriram, R. and Jagadeesh, G., 2009. Film cooling at hypersonic Mach numbers using forward facing array of micro-jets. International Journal of Heat and Mass Transfer, 52(15-16), pp.3654-3664.

[15] Yuan, C., Li, J., Jiang, Z. and Yu, H., 2019. Experimental investigation of liquid film cooling in hypersonic flow. Physics of Fluids, 31(4), p.046101.

[16] Zhou, Y., Wu, H., Li, Y. and Cai, Y., 2018. Two-dimensional film cooling over a flat plate in hypersonic flow. Propulsion and Power Research, 7(3), pp.205-217.