The 8th International Conference on Jets, Wakes and Separated Flows, ICJWSF-2024
September 23-25, 2024, Firenze, Italy

Investigation of the Flow Field Morphology of Film Cooling in Supersonic Flow

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Abstract

Film cooling is widely implemented in highly thermally stressed gas turbine components. Its performance has been extensively investigated for several decades and many results are available in the literature. In conventional gas turbines, cooling in regions of supersonic flow is not prevalent and should generally be avoided. For this reason, results relative to film cooling in supersonic flow are limited. Nevertheless, a new interest related to rotating detonation combustors (RDC) and supersonic turbines is growing. The implementation of those engine components in a gas turbine is likely to need film cooling for thermal protection. In this context it becomes crucial to gain an understanding of how the film interacts with the freestream when operated in a supersonic flow. This paper investigates the effect caused by the injection of film cooling on the morphology of the supersonic flow field. Results obtained by means of schlieren imaging indicated that the coolant injection acted as a wedge inside the flow, determining the local formation of an oblique bow shock around each film cooling hole. The shape, inclination, and strength of the oblique shock showed a dependency on the fundamental dimensionless parameters considered for the characterization of the operating conditions of film cooling. Furthermore, as the amount of mass injected was increased, the inclination of the generated shocks increased and the impingement location of the reflected shock moves upstream along the injection plate. The fluid dynamics of this interaction affected the local pressure distribution on the injection plate, measured by means of Pressure Sensitive Paint. Different film cooling geometries and main flow conditions were tested at multiple operating conditions. The relative impact of the different parameters is presented, providing useful information for the design of a film cooled engine component exposed to a supersonic flow.

Keywords: *Film cooling, Shock Waves, Pressure Field, Schlieren, Pressure Sensitive Paint*

1. Introduction

Gas turbine components require both internal and external cooling to maintain their temperatures at acceptable levels. While for internal cooling several approaches can be investigated, external cooling is referred to as film cooling. This cooling technique consists of the injection of cooling air through the surface to be protected, covering it with an air film at a lower temperature. Conventional film cooling in subsonic flow has been extensively studied by the scientific community for many decades, resulting in a comprehensive understanding of its performance. For example, Bogard and Thole [1] provided an extensive review of film cooling performance, comparing many operating conditions such as blowing ratio (BR) as well as different geometries. They also reviewed the role of the main flow Mach number on film cooling performance.[1]

Early studies in this regime were performed by Wittig et al.[2] who first introduced the blockage effect and associated shock generation due to the coolant injection in a supersonic flow. Later, Gritsch et al.[3] limited the investigation to the characterization of the discharge coefficient. Then, the same authors published another more extensive work investigating the adiabatic effectiveness [4]. They analyzed three different cooling geometries, several operating conditions, and considered both subsonic and supersonic main flow conditions. This study thus represents the first time that the effect of a supersonic main flow on discrete film cooling holes was investigated. The results showed that as long as the Mach number remains well below one, the performance was independent of the Mach number. However, for the supersonic case an improvement to the film cooling was induced by the alteration of the flow field caused by the jet injection. The phenomenon is described as the formation of a bow shock due to the obstacle in the supersonic flow induced by the jet injection. Then, according to the authors, a re-compression shock is considered to be responsible for turning the ejected jet into a flow parallel to the surface and thus improving the measured adiabatic effectiveness. The improvement is thus more relevant for high blowing ratio cases where the liftoff is impeded by the shock formations. Nevertheless, this description of the flowfield is simply inferred considering the findings of Spaid and Zukiski [5], but was not verified by Gritsch et al. [4]

Outside of these investigations, there is a noticeable lack of attention from the scientific community regarding the performance of film cooling in supersonic flow. With a new interest in rotating detonation combustors (RDC), which explicitly contains supersonic flow in the detonation channel, there is a need for experimental investigations of film cooling with a supersonic Mach number. In this context, a test rig was developed at the University of Florence to allow the characterization of the morphology of the flow field of film cooling injected in a supersonic flow.

2. Experimental facility

The test rig consists of a suction type supersonic wind tunnel delivering supersonic flow to a film cooled test section. The test section was established downstream of a 2D convergent divergent nozzle, shaped on one side only, capable of delivering a uniform flow at Mach 1.65. Optical access was enabled on both the transversal direction through highly refined quartz windows, and through the top surface, with a polished PMMA window. The plenum is installed in the lower part of the tunnel. The coolant temperature and pressure are monitored by a thermocouple and a pressure probe installed in the plenum. The mass flow is controlled by valves and is measured with a Coriolis mass flow meter, allowing mass flows up to 12 g/s with an accuracy of 1% of the measured value.

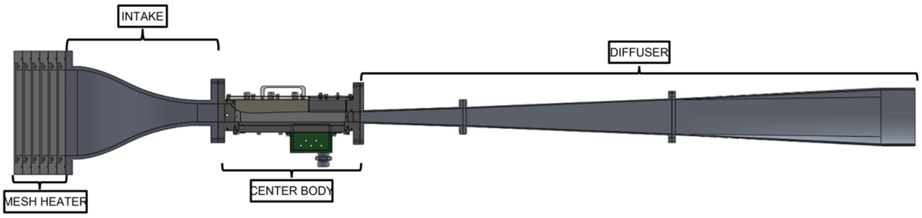


Fig. 1: Schematic of the test rig and its main components.

Additionally, the replaceable injection plate allows investigation of different film cooling geometries. For the present study, three film cooling hole geometries were selected. Specifically, a single row of cylindrical film cooling holes with a 30-degree angle incline, similar cylindrical holes but with an additional 30 degrees compound angle, and the 777 fan shaped holes of Schroeder and Thole [6] were investigated. The diameter D of each hole was 2 mm and the pitch between them was 5D for every considered geometry. The ratio of the length of the hole over the diameter was equal to six for the straight cylindrical holes. The plate thickness was kept the same in the design of the other geometries. A schlieren set up and a PSP set up were implemented to allow the investigation of the morphology of the flow field.

2.1. Schlieren Imaging

The schlieren technique is an optical imaging method used to visualize density changes in a transparent media. In this setup, the beam comes from a "point" source generated by a system consisting of a LED UV illuminator (IL-106UV), a condensing lens with a focal length of 30 mm, and an orifice. The beam is then collimated by a lens with a focal length of 300mm and then refocused downstream of a second field lens, by an achromatic lens with a diameter of 76.6 mm and a focal length of 849.9 mm. Behind a horizontal knife edge the vertical density gradients were captured by a CCD PCO camera equipped with a CANON EF 75-300mm f/4 lens. The images acquired by the schlieren technique were post-processed via a MATLAB routine. This analysis aimed to identify the shock waves present within the test section generated by the injection of film cooling.

2.2. PSP technique

Pressure sensitive paint is an experimental technique employing a paint that emits light in response to the oxygen concentration. The higher the oxygen concentration, the lower is the emission. Since the concentration (molar fraction) of a gas in a mixture is equivalent to its partial pressure, and the concentration of oxygen in air air is well known, this technique allows the calculation of pressure field on the surface investigated, as described by Han et al. [7]. The fundamental components for PSP measurement are the paint itself, to be applied on the test article, a UV illuminator, and a camera, equipped with a filter with a passing band between 500 and 600 nm. Both illuminator and camera used for the schlieren imaging technique are also used for the PSP technique. To evaluate the PSP signal, three images are required. First, a *dark* reference image was obtained with the illuminator turned off. Then, for each operating condition, *wind on* images were acquired with the illuminator turned on and the main flow and coolant gas activated. Then, with the illuminator still turned on, a *wind off* image was acquired with no flow in the test rig. To minimize errors introduced by temperature variations, the wind off image was taken shortly after turning off the wind tunnel, allowing the PSP to be at the same temperature as it was during the acquisition of the *wind on* images. The images are combined to evaluate the emission intensity ratio and calculate the P/P_{ref} distribution, where P_{ref} represents the pressure of the test rig when the acquisition of the *wind off* image is acquired, i.e. the ambient pressure.

3. Results analysis

The present study focuses on the experimental investigation of the flowfield determined by film cooling in a supersonic flow. In this experiment, air and CO_2 were used as coolant gases for the jets. The two gases, have different molar masses, namely 29 g/mole and 44 g/mole, respectively. It can be shown that the ratio of the density ratios between operating conditions using CO_2 and air as coolant is equal to the ratio of the densities of the two coolants, i.e. $DR_{CO_2}/DR_{air} = \rho_{CO_2}/\rho_{air}$, where the subscripts air and CO_2 refer to the coolant being used. When subsonic main flows are considered, the density ratio between coolant and main flow is only a function of the ratio between the molar masses of the two gases. Thus when air is used as coolant, DR is equal to 1 and when CO_2 is used as coolant DR is equal to 1.52. However, when a supersonic main flow is studied, its low pressure results in the film cooling holes being choked. In this case, the DR becomes a function also of the ratio of specific heats of the two gases γ_{air} and γ_{CO_2}. For both coolant gases, air and CO_2, the ideal gas law equation applies for the gases in the plenum $P_0 = \rho_0 \frac{R}{W} T_0$. Here, P, T, and ρ, correspond to the total quantities, as the velocity is zero. Subsequently, the mass flow discharged into the test section can be described with the assumption of choked flow, indicating that the mass flow is independent from the operating condition inside the wind tunnel test section: $\dot{m} = C_d A \sqrt{\rho_0 P_0 f(\gamma)}$ where $f(\gamma)$ is equal to $\gamma \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}$. Algebraic manipulations of the following equations with the assumptions of equivalence between air and CO_2 for the mass flow, the total temperature in the plenum, the discharge coefficient, and the passage area, leads to the following equation:

$$DR_{CO_2}/DR_{air} = \rho_{CO_2}/\rho_{air} = \sqrt{\frac{\left(\frac{2\gamma}{\gamma+1}\right)_{air}}{\left(\frac{2\gamma}{\gamma+1}\right)_{CO_2}}} \cdot \sqrt{\left(\frac{W_{CO_2}}{W_{air}}\right)} \tag{1}$$

Eq1 defines the value of the ratio of the density ratios, which remains equal to 1.26 for any blowing ratio value above 0.4, i.e. when the flow in the holes is choked.

For the schlieren images, BR values of 0.5, 0.8, 1.0, 1.2 and 1.5 were acquired with both air and CO_2, allowing for a comparison of the morphology of the flow field obtained between the two gases. Figure 2 shows the schlieren images obtained for the three considered geometries for blowing ratios ranging between two and four. The origin of the horizontal axis is placed at the exit of the film cooling holes. The results obtained are similar for the cylindrical holes aligned with the main flow and those with a compound angle. This is probably linked to both having the same exit velocity at the hole exit, and thus the same jet momentum for the jets. At these BR values, the cylindrical hole jet penetrated three diameters into the main flow at an axial location of five diameters downstream of the exit. The situation is different for the fan shaped holes, where the outlet expansion reduces the jet penetration, causing the jet to only reach a height of 1.5 diameters at the same location downstream of the exit. This expansion at the outlet caused a velocity reduction and thus a reduction in momentum and a lower variation of the angle of the generated oblique shock.

For all the images the inclination of the oblique shock waves were determined. Figure 3b shows a nearly linear variation of shock wave angle for increasing BR values for the cylindrical holes. Looking at the shock polar for oblique shocks, the film injection process acts as a wedge to the incoming flow. Equation 2 enabled computation of an equivalent wedge angle. Fig.3b also shows this wedge angle on the second Y axis. As the blowing ratio increases, the shock angle linearly increases from 42° where the wedge angle is 4° up to 53° with a corresponding wedge angle of 11°. Furthermore, the bow shock measured angle for the three cooling hole patterns were plotted against BR. It is clear that the larger outlet area of the fan shaped holes causes a reduction of the incidence angle of the generated shock waves. Such phenomena is associated with a lower momentum of the coolant gases, causing a lower penetration as shown in Fig.3c.

$$tan\theta = 2cot\beta \left[\frac{M_1^2 sin^2\beta - 1}{M_1^2(\gamma + cos2\beta) - 2}\right] \tag{2}$$

4. Pressure distributions on injection plate

For PSP measurements only air was used, as CO_2 injection does not allow for the measurement of the pressure field. Also, PSP measurements were only performed on the baseline geometry of straight cylindrical film cooling holes. Additional tests were performed with only a single cooling hole at blowing ratios up to seven. Figure 4 shows the pressure distribution over the complete array of five cylindrical film cooling holes with air injection. As the BR increased, the shock structure increased in intensity, generating areas of progressively higher pressure on the sides of each film hole and progressively lower pressure downstream of them. This high pressure zone is caused by the compressing action of the shock itself which gains strength as the blowing ratio increases. As the BR increases, the equivalent wedge angle increased and thus the inclination of the shock. More inclined shocks result in stronger reduction of the Mach number across them and a greater pressure rise. For the higher blowing ratios the adjacent cooling holes start to merge into a 2D planar oblique shock structure.

In addition, the zone of interaction of the reflected shock with the film injection plate moves upstream. The impingement location of the reflected shock is visible as the dark yellow region in Fig. 4. The growth in intensity of the

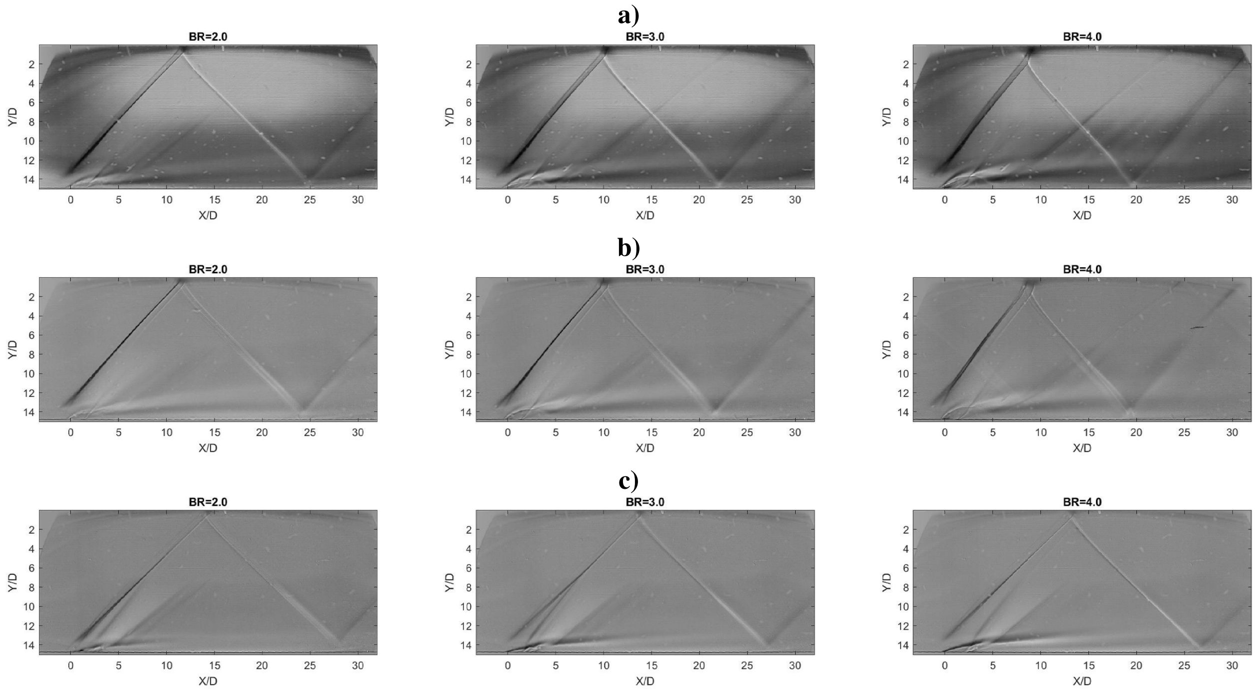
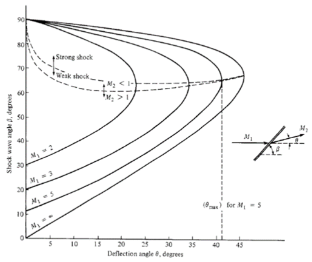
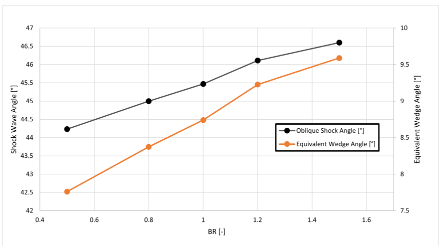


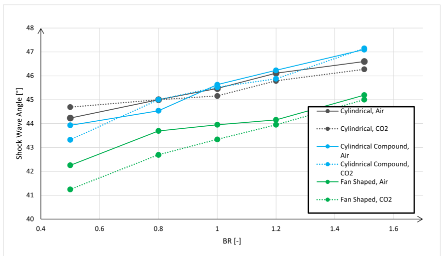
Fig. 2: Schlieren visualization of air injection at high blowing ratios, a) Cylindrical holes, b) Cylindrical holes with compound angle, c) Fan-shaped holes.



(a) shock polar generated from Equation2, [8]



(b) Variation of shock wave angle and equivalent wedge angle for cylindrical holes and air.



(c) Shock wave angle versus BR for different cooling holes and different coolant gases.

Fig. 3: Oblique shock measurements

pressure rise at the reflected shock impingement location is consistent with a gain in strength of the generated oblique shocks. The location of shock reflection is at x/D = 30 when the BR = 0.5 and moved upstream towards x/D = 20 as the BR increases to 4, as shown in Fig.5a. Fig. 5b then indicates that the peak pressure location lines up directly with the location of the reflected shock from the schlieren images.

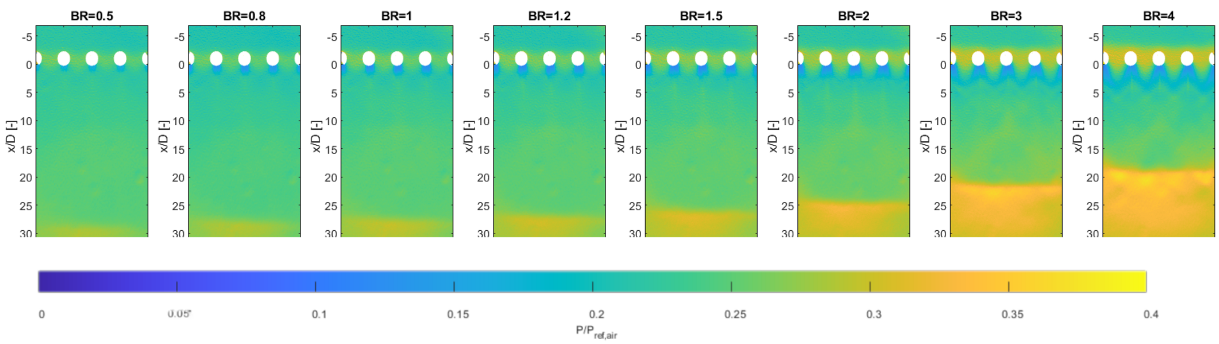
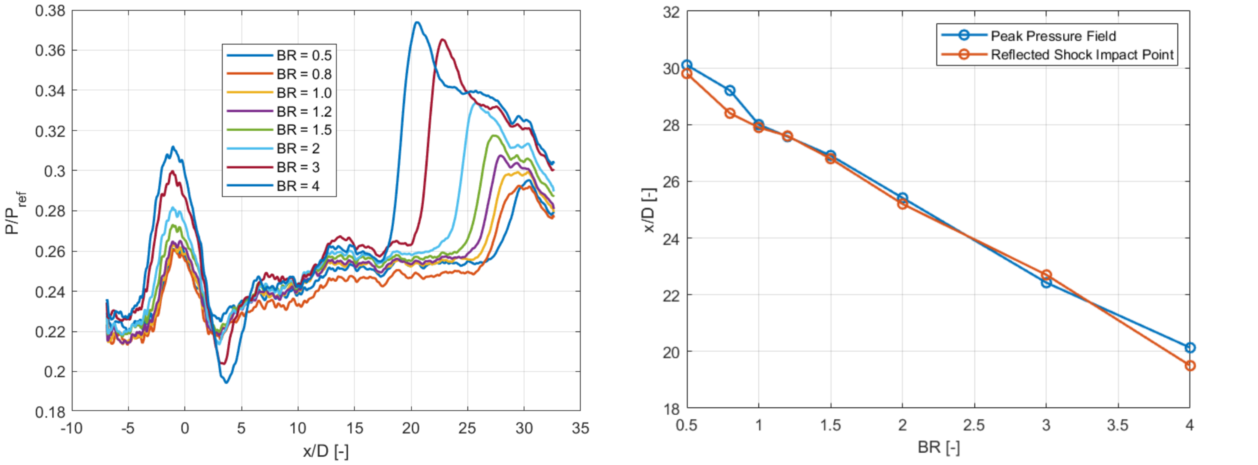


Fig. 4: Pressure distribution for increasing BR for 5 film cooling holes

To gain further insight into the morphology of the flowfield, tests using only the center cooling hole were performed. The other holes were closed from below with duct tape. With only a single hole the bow shock is clearly visible in



(a) Pressure profiles along the centerline between film cooling holes

(b) Pressure profile peak location and point of incidence of the reflected shock obtained from the schlieren images.

Fig. 5: Scaling of Shock wave angle for different cooling holes geometries and different

Fig.6 as it wraps around the single central jet. The results are consistent with the numerical work of Pudsey and Russel [9]. The area of high pressure generated by the bow shock surrounds the whole film cooling hole and protrudes in the direction of the flow with a curvilinear shape. The peak intensity was adjacent to the inlet of the film cooling hole for all the BR considered. In addition, as the BR increases, the intensity of the high pressure peak increases both in magnitude and spacial extension.

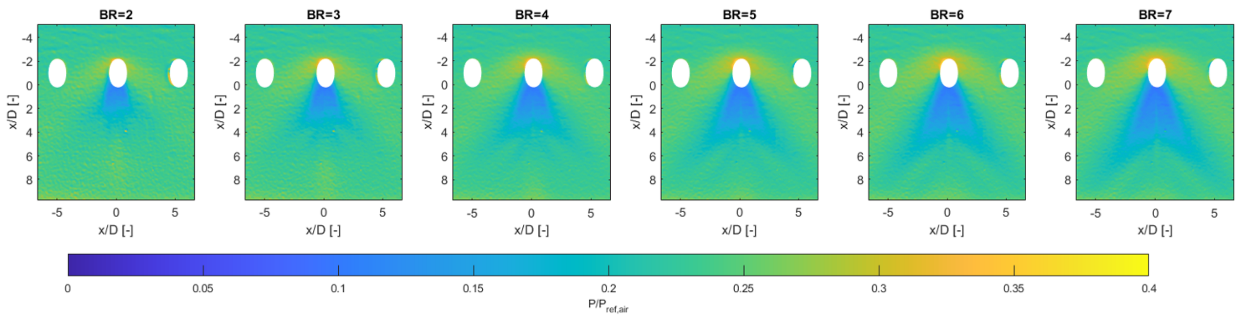


Fig. 6: Pressure distribution for increasing BR for one film cooling hole at high BR values.

Those features support respectively the variation of inclination associated with an increasing BR and also the bow shock shape. In addition, according to Gruber et al.[10], the three-dimensional bow shock interacts with the boundary layer causing it to separate and the subsequent formation of a horseshoe vortex around the jet. A comparison between the jet formation described by Gruber et al.[10] and the pressure distribution on the injection plate is shown in Fig.7. Consistent with their findings, a peak in pressure was identified upstream of the vortex, while a low pressure zone, associated with the formation of a recirculation region was formed downstream causing a downward pitching moment.

5. Conclusions

This work investigated the morphology of the flowfield associated with the implementation of film cooling in a supersonic flow. Schlieren imaging results represent the first visualization of this type of interaction available in the literature. The flow injection was shown to act as wedge in the supersonic main flow, causing the formation of oblique shock waves. The inclination of the generated shocks revealed a strong correlation between the blowing ratio and the angle of the shock induced by the jet. Additionally, results indicated that fan shaped holes causes the formation of weaker oblique shock due to the smaller penetration of this coolant jet. The structures previously presented in the literature for vertical injection have been visualized with the use of Pressure Sensitive Paints. Since the holes are discrete, the wedge-like structure does not extend throughout the entire width of the test section. A configuration with a single film cooling hole showed the bow-like shock structure. When the whole array of film cooling holes was considered, PSP results showed that the bow shocks interact with each other, causing the shock structures to merge, causing them globally to act as a 2D planar oblique shock. The impact of the reflection of the generated shock back

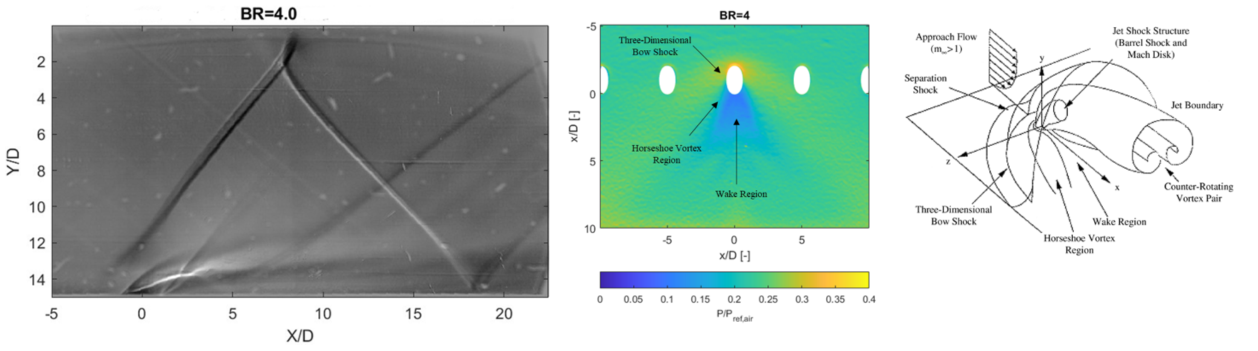


Fig. 7: Comparison between Schlieren image, measured pressure distribution, and theoretical shock system morphology [9]

onto the injection plate has been also investigated. Peaks in pressure profiles captured with PSP are well aligned with the impact location identified by schlieren images. The results have indicated a pressure rise associated with the impact of the reflected shock on the already disrupted boundary layer developing on the injection plate. The present study has provided additional understanding of the dominant features of film cooling in supersonic flow providing support for the optimal design of film cooling arrays for supersonic main flow applications.

Acknowledgments

This project has received funding from the European Union's Horizon 2020 research and innovation program under the Marie Skłodowska-Curie grant agreement No 956803.

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