SEMARAK ILMU
PUBLISHING
202103268166(003316878-P)

CFD Letters

Journal homepage:
https://semarakilmu.com.my/journals/index.php/CFD_Letters/index
ISSN: 2180-1363

Numerical Investigation of the Hypersonic Inlet under Throttling with Heat Source

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ARTICLE INFO

Article history:
Received 13 August 2022
Received in revised form 31 August 2022
Accepted 16 September 2022
Available online 31 October 2022

Keywords:
Numerical Simulation; Hypersonic; Shock Wave Boundary Layer; Heat Source; Active Flow Control

ABSTRACT

The influence of using a heat source to manage the shock wave boundary layer interactions (SWBLI) at the hypersonic inlet under throttling were studied numerically. This hypersonic inlet was created for a fluid flow Mach number of 5. The throttling was induced by a plug placed near the intake isolator's outlet. The study's parameters included the heat source power and size. The intake performance indicators were the total pressure recovery and the flow distortion. The position of the heat source was determined by studying the interplay of the shock waves from the compression ramp. The results demonstrated the existence of the shock waves at the heat source, and its influences on the SWBLI inside the isolator. This behaviour, led to an increase in the total pressure recovery and reduction of the flow distortion.

1. Introduction

Flow control is a great alternative for addressing the Shock-wave Boundary Layer Interaction (SWBLI) problem. Several studies have proposed the implementation of a porous bleed system to mitigate the negative impact of the SWBLI and lower the risk of boundary-layer detachment. The traditional issues of shock wave boundary layer interferences have been studied by researchers, who have conducted extensive studies utilizing shock wave simulators and synthesized essential empirical formulations [1]. Over the last four decades, several studies have been conducted to evaluate the capacity of Computational Fluid Dynamics (CFD) to efficiently anticipate aerothermodynamic loads on basic designs which produce SWBLI equal to aircraft conditions [2]. SWBLI are complicated flow characteristics connected to a wide range of flows, such as response control jets, supersonic inlets, overworked Nozzles, missile base flows, and high-speed flight control surfaces [3]. SWBLI generates boundary layer thickening, separation of bubbles/regions, and increased turbulence effects, all of which slow down total the pressure recovery. Several techniques have been employed to reduce the negative effects of SWBLI [4-7].

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https://doi.org/10.37934/cfdl.14.10.7986

Flow control is regarded as a major challenge in upcoming automotive designs, necessitating a thorough physical knowledge underlying the phenomena of the mechanics. Flow control is a key solution to the SWBLI challenge. Wang *et al.*, [8] and Mengxiao *et al.*, [9] used a transverse plasma jet to experiment and computationally regulate the SWBLI on a ramp in a supersonic flow. They observed that a high turbulence and temperature flow can worsen the separation barrier, and that the influence of plasma regulation on the boundary layer detachment is dependent on the Lorentz force vector.

One of the first researchers to investigate the effects of placing a heat source inside a high-speed aircraft intake was Macheret *et al.*, [10]. The term "virtual cowl" was coined by the researchers as the flow was pushed to streamline thru the hypersonic intake by the heat source region. They discovered that the flow within scramjet intake moved faster once the Mach value was lower than what was anticipated. The best position of the heat source was then identified by the researchers in order to minimise air leakage. By placing an energy source ahead upon its intake, Kremeyer *et al.,* [11] research's team found a way to improve the efficiency of high-speed vehicle intake. Comparable to this, Russell *et al.,* [12] showed how heat source may be employed to improve an internal combustion engine's efficiency while also reducing dispersion in a high-speed aircraft.

To change the frequency of the shock wave oscillation in the SWBLI, Pham *et al.,* [13] and Russell *et al.*, [12] employed laser energy deposition to successfully manage flow separation and disturbances in the supersonic duct flows. Their findings showed that utilizing a laser might enhance the flow behaviour of an inlet model using a central conical compression surface. In addition, the discharge plasma energy deposition was examined in the SWBLI area on a ramp. It was demonstrated that when the discharge is activated, the force acting on the model's surface may be greatly decreased [14, 15].

Numerous studies have looked at the aerodynamics of subsonic, supersonic, and hypersonic flow employing numerical modelling [16-19]. Several computational studies dealing with hypersonic intake flow management revealed that the development of turbulence was a result of the interaction of the low-density zone and the bow shock, resulting in a change in the drag coefficient and flow detachment in the high-speed flow. According to current computational calculations, lasers may result in drag reduction, positive adjustment of the aerodynamic forces, and shock control [20]. Jiang *et al.*, [7] numerically investigated the effect of the MHD plasma actuator's location on the SWBLI and classified them into four categories based on the distinct processes. They discovered that placing the actuator in the isobaric zone had the biggest impact on minimizing the divergence distances.

This study seeks to investigate the numerical heat effect to control the SWBLI inside the isolator, to reduce the ensuing pressure loss.

2. Methodology

The geometry of an inlet-isolator and boundary conditions adopted from Refs. [21-24] are shown in Figures 1 and Table 1, respectively. The effect of a heat source location was studied by Sepahi-Younsi & Esmaeili [25]. They found that the ideal location for the source of heat is challenging, because of the dimensionless radius ratio used in the study. In the present study, location of the heat source is shown in Figure 1. A pressure far-field, pressure inlet, two pressure outlets, fixed temperature walls, and symmetry define the computing domain, as shown in Figure 2.

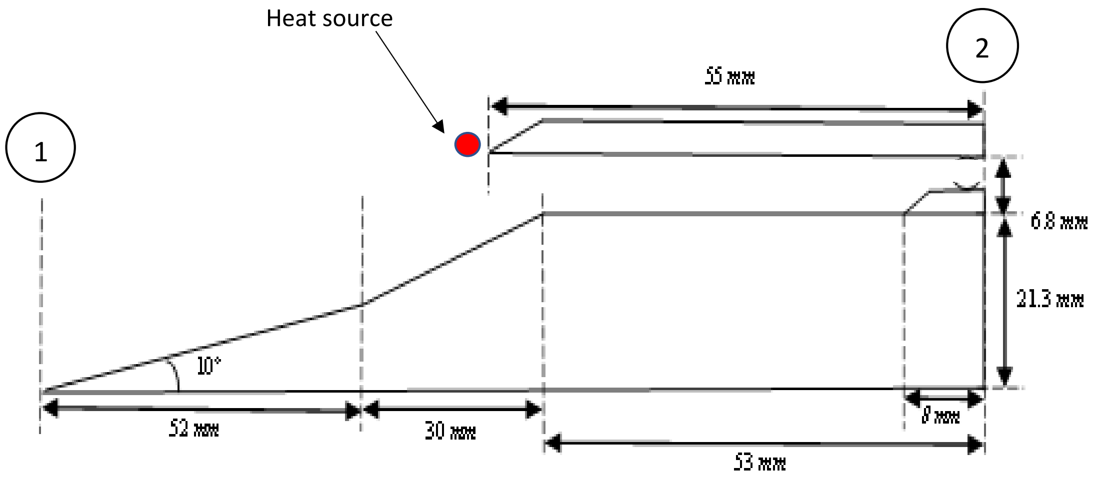

Fig. 1. Generic scramjet inlet-isolator dimension

This study was made using ANSYS Fluent. The turbulence model of the k-omega (k–ω) (SST) was employed in this work as the viscous model for turbulence. The turbulence's viscosity ratio was set to one. To preserve stability, the Courant–Friedrichs–Levy (CFL) ratio was first adjusted to 0.5, and then increased by a similar amount per 1,000 iterations. The initialization settings for the turbulence calculation were inviscid solutions for each parametric case.

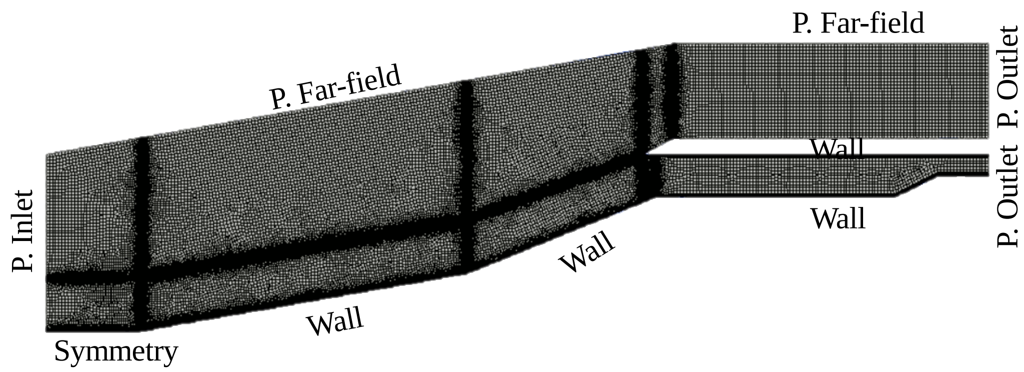

Fig. 2. Mesh for the baseline case

Table 1
Boundary condition

Properties	Value
Mach number	5
Stagnation pressure (MPa)	0.65
Stagnation inlet temperatures (K)	375
Reynolds number (m^{-1})	13.2×10^{6}
Angle of attack (°)	6
Throttling ratio	0.4
Diameter of the heat source (mm)	0.25, 0.5, 1
Heat source energy (W/m^3)	$1x10^{13}$, $2x10^{13}$, $5x10^{13}$

The mesh size optimization was done to reduce the computational costs. The data is gathered from the leading edge of the compression ramp until the outlet of the isolator. Figure 3 illustrates the element number of 55133 elements, 57010 elements, 64097 elements, 107009 elements and

141922 elements was utilized for the mesh independent study. To reduce the computational costs, an element size of 64097 was selected for all simulation instances.

Figure 4 shows the distributions of the mean wall pressure, P_{amb} along the floor surface from point 0 until point 3 (Figure 1) . The present numerical results are consistent with the experimental data reported by [21]. Nevertheless, there was a difference at the compression ramp due to boundary layer impacts, but not inside the isolator. Since the present study is focusing inside the isolator, thus the present numerical method is suitable for the analysis.

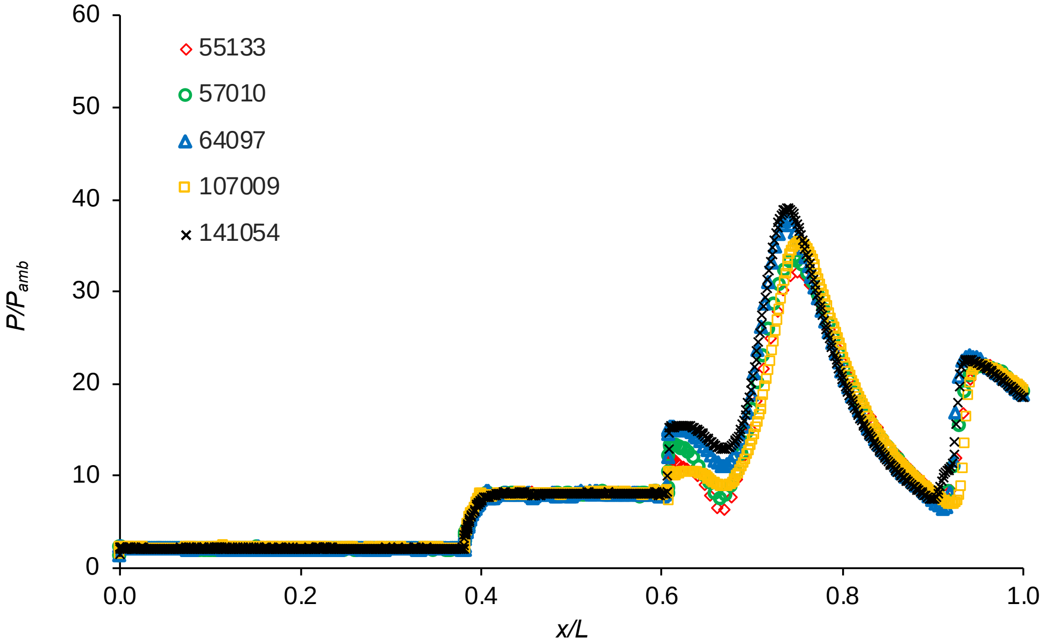

Fig. 3. Grid independent test

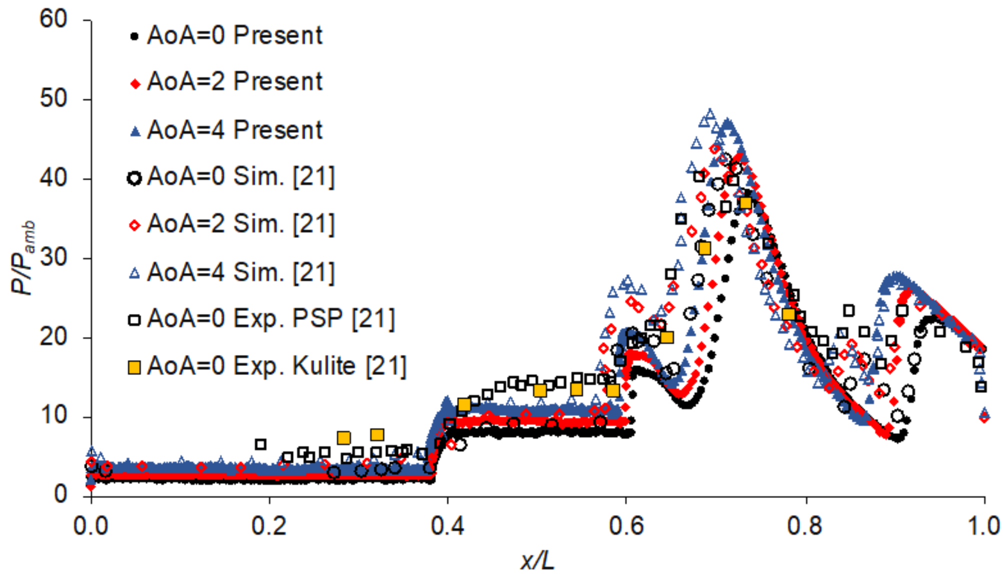

Fig. 4. Validation of the present simulation results with published data [21]

3. Results and Discussion

This section summarises the findings of the hypersonic inlet investigation. The influence of the heat source on the Flow Distortion (FD) and Total Pressure Recovery (TPR) are addressed. This is an essential characteristic, since it influences the aircraft's friction forces. The boundary layer separation causes pressure drops within the inlet. The magnitude of this pressure decrease is in relation to the desired constant flow, and is monitored and quantified using flow distortions [25]:

$$FD = \frac{P_{t,max} - P_{t,min}}{P_{t,avg}} \tag{1}$$

where, $P_{t,max}$ and $P_{t,min}$ = represent the maximum and minimum total pressures at the exit, and $P_{t,avg}$ = represents the average total pressure at the exit lip. Total pressure recovery (TPR) is the ratio of the total pressure at the exit plane to that of the total pressure in the fluid flow [25]:

$$TPR = \frac{P_{t,f}}{P_{t,\infty}} \tag{2}$$

The results of FD and TPR are shown in Tables 2 and 3 respectively. Table 2 demonstrates the variations in the FD at an Angle of Attack (AoA) of 6, and a Throttling Ratio (TR) of 0.4 before the introduction of the heat source. The value of FD decreases as the heat source is added. It seems that the FD's lowest value was at a diameter of 0.25mm, and an energy heat source of $5\text{x}10^{13}$ W/m^3. When the heat source was not used at the intake, the FD value was the greatest.

Table 2
The values of FD without and with the heat source

Heat source diameter (mm)	FD (without heat source)	FD (with heat source)		
		$1\text{x}10^{13}$ W/m^3	$2\text{x}10^{13}$ W/m^3	$5\text{x}10^{13}$ W/m^3
0.25		0.54	0.85	0.48
0.50	1.14	0.52	0.54	0.56
1.00		0.54	0.53	0.64

Table 3 demonstrates that the variations in TPR before and after adding a heat source, which had a lower value. It demonstrated the greatest value TPR value for a diameter of 0.25mm with a $2\text{x}10^{13}$ W/m^3 energy heat source. As the energy of the heat source changed, the FD and TPR influences rose significantly. As a result, the TPR rose but the FD fell significantly. The results revealed that by varying the chosen heat source size and energy, it considerably changed the intake of the FD and TPR. D = 0.25mm looked to be a better value for its source diameter, and $5\text{x}10^{13}$ W/m^3 appeared to be a better amount for the energy heat source.

Table 3
The values of TPR without and with the heat source

Heat source diameter (mm)	TPR (without heat source)	TPR (with heat source)		
		$1\text{x}10^{13}$ W/m^3	$2\text{x}10^{13}$ W/m^3	$5\text{x}10^{13}$ W/m^3
0.25		136.49	227.46	120.89
0.50	104.06	145.48	146.61	127.29
1.00		110.14	126.18	107.77

For bettering the performance of the supersonic air intake, the TPR should be kept high. Similarly, the lower the FD, the better the performance of the hypersonic air intake. The TPR must be kept as high as possible due to the compressor's magnifying effect. Otherwise, the separated flow becomes intrinsically unsteady and lacks consistency as it reaches the compressor, limiting its performance and operating accuracy. The FD has a negligible influence on the thermodynamic process. An applied load and specific fuel consumption are two examples of setbacks attributed to the overall performance induced by the FD in the compression component's efficiency in a gas turbine engine. Inadequate engine control system reactions are the primary cause of a gas turbine engine's performance loss. As a result, the magnitude of FD must be kept low [26].

The flow was analysed without the heat source in the first part, and the performance characteristics were then assessed. The heat source was then introduced, and the performance characteristics were assessed. Several working circumstances were created by varying the diameter and the energy heat source, as well as evaluating the performance characteristics for each parameter. For an efficient hypersonic intake, larger TPR values and smaller FD values are preferred.

The flow arrangements based on the Figure 5 can address the flow characteristics of a common oscillation phase. The density contour behaves similarly to that of the Schlieren imaging technique developed by Li *et al.,* [22]. The differences in flow parameters in the isolator were varied between that without a heat source (Fig. 5a), and that with a heat source (Fig. 5b) at Mach 5, AoA 6, and TR 0.4. The results showed that the shock wave formed around the heat source affected the SWBLI inside the isolator. Thus, it affected the flow characteristics in Figure 5b.

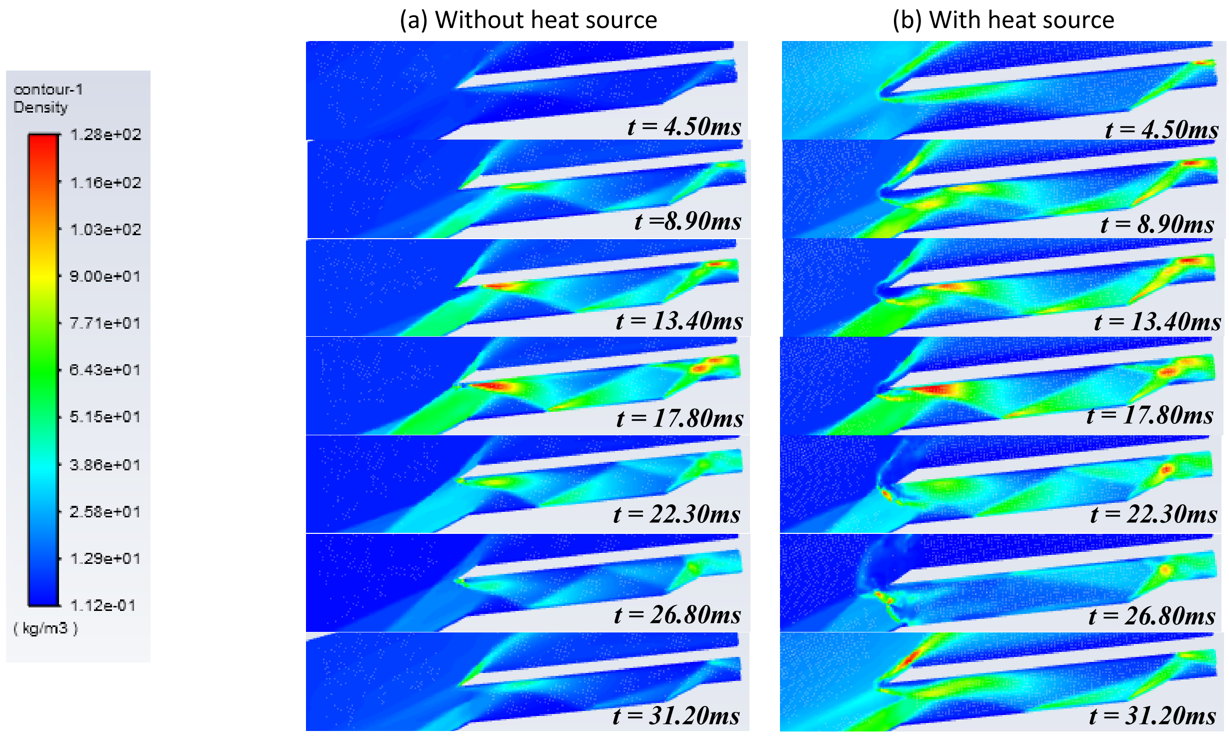

Fig. 5. Density contour of Mach 5 AoA 6 TR 0.4 (a) without and (b) with heat source (dia.=0.25mm, $5x10^{13}$ W/m^3)

The separation shock impinges on the cowl tip at t = 4.50 ms, as depicted in Figure 5a, and the intake entering the flow rate then increases. The reattachment shock reaches the shoulder at t = 8.90 ms, and the unfavourable pressure difference reduces rapidly. The detachment flow at the intake entrance subsequently reduces. At t = 13.40 ms, the detachment shock reaches the intake, and the separation on the isolator converges at the shoulder and moves downstream. At t = 17.80 ms, a substantial barrier forms at the inlet's entrance, and the separation impinges on the cowl lip right before overflowing. At t = 26.80 ms, the whole shock train through the isolator is almost gone. The incoming flow level increases at t = 31.20 ms. A subsequent upstream-moving shock occurs in the duct's back portion. After this phase, a new oscillation cycle begins when the shock starts to develop inside the isolator. As seen in the Figure 5, the flow characteristics at t = 31.20 ms are almost comparable to those at t = 8.90 ms. Previous studies indicated that the development and evolution of the shock train in the inlet has a substantial impact throughout the cycle [26].

4. Conclusions

The flow characteristics of the hypersonic isolator with a back pressure and heat source were investigated numerically. The results agreed well with the experimental results from previous studies. The investigation demonstrated that a shock wave originated around the heat source and influenced the SWBLI inside the isolator. The SWBLI influenced the flow distortion and total pressure recovery. The addition of the heat source helped to reduce the flow distortion and increased the total pressure recovery. It was found that a heat source diameter of 0.25mm was a preferable option for the source diameter, alongside an energy heat source of $5\text{x}10^{13}$ W/m^3 which is a preferable number. These values produced a significant effect on the flow distortion and total pressure recovery.

Acknowledgment
The authors would like to thank Universiti Pertahanan Nasional Malaysia (UPNM) for providing the support required to complete this research work (UPNM/2021/GPPP/TK/3). This work was supported by the Air Force Office of Scientific Research under the award number FA2386-21-1-4016.

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