CHARACTERIZATION OF HIGH SPEED INLETS USING GLOBAL MEASUREMENT TECHNIQUES

A THESIS SUBMITTED TO THE UNIVERSITY OF MANCHESTER FOR THE DEGREE OF DOCTOR OF PHILOSOPHY IN THE FACULTY OF ENGINEERING AND PHYSICAL SCIENCES

2014

By
Azam Che Idris

School of Mechanical, Aerospace and Civil Engineering
<table>
<thead>
<tr>
<th>Section</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>Table of Contents</td>
<td>1</td>
</tr>
<tr>
<td>List of Figures</td>
<td>8</td>
</tr>
<tr>
<td>List of Tables</td>
<td>17</td>
</tr>
<tr>
<td>Nomenclature</td>
<td>19</td>
</tr>
<tr>
<td>Abstract</td>
<td>24</td>
</tr>
<tr>
<td>Declaration</td>
<td>26</td>
</tr>
<tr>
<td>Copyright</td>
<td>27</td>
</tr>
<tr>
<td>Acknowledgements</td>
<td>28</td>
</tr>
<tr>
<td>Chapter 1 Introduction</td>
<td>30</td>
</tr>
<tr>
<td>1.1 Introduction and Motivation</td>
<td>30</td>
</tr>
<tr>
<td>1.2 Ramjet and Scramjet Basic Principle</td>
<td>31</td>
</tr>
<tr>
<td>1.3 Historical Overview</td>
<td>34</td>
</tr>
<tr>
<td>1.4 Aim and Objectives</td>
<td>38</td>
</tr>
<tr>
<td>1.5 Structure of Thesis</td>
<td>39</td>
</tr>
<tr>
<td>Chapter 2 Background Study on Scramjet Inlet</td>
<td>41</td>
</tr>
<tr>
<td>2.1 Scramjet Inlet Design</td>
<td>41</td>
</tr>
<tr>
<td>2.1.1 Two-dimensional Planar</td>
<td>41</td>
</tr>
<tr>
<td>2.1.2 Two-dimensional Axisymmetric</td>
<td>42</td>
</tr>
<tr>
<td>2.1.3 Three-dimensional Inlet</td>
<td>43</td>
</tr>
</tbody>
</table>
3.1 University of Manchester’s High Supersonic Tunnel (HSST)........................................82

3.2 Generic Scramjet Inlet-isolator Model Design.............................................................85
   3.2.1 Baseline Design......................................................................................................85
   3.2.2 Three-dimensional Design View ...........................................................................95

3.3 Schlieren Flow Visualization .......................................................................................98

3.4 Basic Pressure and Temperature Measurement .......................................................100
   3.4.1 Facilities................................................................................................................100
   3.4.2 Experimental Procedures....................................................................................100

3.5 Pressure Sensitive Paint .............................................................................................101
   3.5.1 PSP Technique in Scramjet Inlet Investigation .................................................101
   3.5.2 PSP System Facilities.........................................................................................102
   3.5.3 Luminophore and Substrate Selection...............................................................103
   3.5.4 PSP System Setup...............................................................................................107
   3.5.5 Error and Uncertainty Analysis ..........................................................................111

3.6 Infrared Thermography .............................................................................................113
   3.6.1 Background...........................................................................................................113
   3.6.2 Experimental Setup.............................................................................................115

3.7 Numerical Analysis .....................................................................................................117
   3.7.1 Background...........................................................................................................117
   3.7.2 Numerical Code Available ..................................................................................117
3.7.3 Turbulence Model Selection

3.7.4 Computational Fluid Dynamics Theory

3.7.5 Numerical Setup

Chapter 4 Scramjet Inlet-isolator characteristics at design conditions

4.1 External Compression Flow field

4.1.1 Flow Properties

4.1.2 Static Pressure Measurement on Compression Ramp for Baseline Case

4.1.3 Compression Ramp Streamwise Vortices

4.2 Internal Isolator Flowfield

4.2.1 Isolator Flow Features

4.2.2 Isolator Surface Pressure Profile

4.2.3 Isolator Surface Pressure Map

4.2.4 Sidewall Plane Pressure Map

4.2.5 Inlet-isolator Exit Static Pressure and Distortion Index

4.2.6 Inlet-isolator Exit Mach Number and Performance Estimation

4.3 Conclusions

Chapter 5 Scramjet Inlet-isolator Characteristics at Off-design Conditions

5.1 Effects of Changes in AoA

5.1.1 AoA Effects on Compression Corner Separation Size

5.1.2 AoA Effects on Compression Ramp Streamwise Vortices
5.1.3 AoA Effects on Isolator Shock-structures .......................................................... 163

5.1.4 Comparison of Scramjet Inlet-isolator Pressure Profile at Different AoA ...... 165

5.1.5 Comparison of Scramjet Inlet-isolator Pressure Map at Different AoA ........ 167

5.1.6 Comparison of Scramjet Inlet-isolator Sidewall Pressure Map and Performance at Different AoA ................................................................................................................................. 170

5.2 Effects of Changes in Cowl Length ..................................................................... 173

5.2.1 Cowl-length Effects on Isolator Shock-structures ........................................... 175

5.2.2 Comparison of Scramjet Inlet-isolator Pressure Profile with Different Cowl Length 177

5.2.3 Comparison of Scramjet Inlet-isolator Pressure Map with Different Cowl Length 179

5.2.4 Comparison of Scramjet Inlet-isolator Sidewall Pressure Map and Performance at Different Cowl Length .................................................................................................................................................. 181

5.3 Conclusions ........................................................................................................ 184

Chapter 6 Scramjet inlet-isolator optimization .......................................................... 186

6.1 Effectiveness of Cowl Deflection to Control Shoulder Separation ..................... 187

6.1.1 Cowl Deflection Effects on Isolator Shock-structures .................................... 188

6.1.2 Comparison of Scramjet Inlet-isolator Pressure Profile with Different Cowl Deflection .............................................................................................................................................................................. 190

6.1.3 Comparison of Scramjet Inlet-isolator Pressure Map with Different Cowl Deflection Angle ............................................................................................................................................................................. 192
6.1.4 Comparison of Scramjet Inlet-isolator Sidewall Pressure Map and Performance at Different Cowl Angle ................................................................. 193

6.2 Effectiveness of MVG for Scramjet Inlet-isolator Flow Control ....................... 196

6.2.1 Boundary Layer Trip Development for Compression Ramp ......................... 196

6.2.2 MVG in Scramjet Inlet-isolator Flow Control Application .......................... 205

6.3 Effectiveness of Variable-geometry Cowl to Optimize Flow at Off-design Conditions 210

6.3.1 Variable-geometry Effects on Scramjet Inlet-isolator Flow Structures at Off-design Conditions ................................................................. 212

6.3.2 Variable-geometry Effects on Scramjet Inlet-isolator Pressure Profile at Off-design Conditions ................................................................. 214

6.3.3 Variable-geometry Effects on Scramjet Inlet-isolator Pressure Map at Off-design Conditions ................................................................. 216

6.3.4 Variable-geometry Effects on Scramjet Inlet-isolator Pressure Map at Off-design Conditions ................................................................. 217

6.4 Strategy for Scramjet Inlet-isolator Performance Optimization ....................... 219

6.5 Conclusions ........................................................................................................ 221

Chapter 7 Conclusions and Future Recommendations .......................................... 223

7.1 General Conclusions .......................................................................................... 223

7.2 Future Recommendations ................................................................................... 226

References ............................................................................................................... 229
Appendix ......................................................................................................................... 245

A. High Supersonic Tunnel Calibration ........................................................................ 245

B. Pressure Sensitive Paint (PSP) Theory ..................................................................... 247
   B.1 Photoluminescent Kinetics Theory .................................................................... 247
   B.2 Derivation of Stern-Volmer Equation for PSP .................................................... 251

C. Infrared Thermography ............................................................................................. 255
   C.1 Theory ................................................................................................................ 255

D. Background-Oriented Schlieren (BOS) in Scramjet Inlet-isolator Investigation ...... 257
   D.1 Introduction ........................................................................................................ 257
   D.2 Basic BOS Theory ............................................................................................. 259
   D.3 Cross-correlation Theory .................................................................................. 261
   D.4 Experimental Setup ............................................................................................ 262
   D.5 Results and Discussions ..................................................................................... 266
   D.6 Conclusions ........................................................................................................ 270
LIST OF FIGURES

Figure 1-1 Schematic of air-breathing propulsion device concept ........................................ 31
Figure 1-2 Working schematic of typical ramjet engine (figure taken from Heiser and Pratt\textsuperscript{5}) .................................................................................................................. 33
Figure 1-3 Working schematic of typical scramjet engine (figure taken from Heiser and Pratt\textsuperscript{5}) .................................................................................................................. 34
Figure 1-4 Leduc 010 ramjet aircraft (figure taken from Fry\textsuperscript{6}) ........................................ 35
Figure 1-5 Griffon II turbo-ramjet aircraft (figure taken from Gozlan\textsuperscript{7}) ...................... 36
Figure 1-6 First freejet tested ramjet combustion engine done by Ferri in 1963 (figure taken from Fry\textsuperscript{6}) .................................................................................................................. 36
Figure 1-7 Artist impression of X-30 (figure taken from Kazmar\textsuperscript{13}) .......................... 37
Figure 1-8 Artist impression of X-43 aircraft (figure taken from Kazmar\textsuperscript{13}) .................. 38
Figure 1-9 Artist impression of X-51 (figure taken from Kazmar\textsuperscript{13}) .......................... 38
Figure 2-1 Sketch of two-dimensional planar scramjet inlet with double ramp as external compression surface ...................................................................................................................................... 42
Figure 2-2 Oswatisch inlet (figure taken from Molder et al.\textsuperscript{19}) ................................. 42
Figure 2-3 30 degree sweep sidewall compression inlet (figure taken from Holland and Perkin\textsuperscript{20}) .............................................................................................................. 43
Figure 2-4 Scramjet inlet station definitions ............................................................................. 44
Figure 2-5 Brayton cycle temperature-entropy (T-s) diagram for typical airbreathing engine 45
Figure 2-6 Typical shocks structure of (a) ideal shock-on-shoulder inlet and (b) non-ideal, re-expansion inlet ...................................................................................................................................... 55
Figure 2-7 Supersonic inlet starting mechanism (figure taken from Heiser and Pratt\textsuperscript{21}) .... 59
Figure 2-8 Scramjet inlet designed for Mach 5.5 operating in Mach number (a) M = 4, (b) M = 5.5 and (c) M = 6 (figure taken from Bachchan and Hillier\textsuperscript{48}).

Figure 2-9 Axisymmetric scramjet inlet operating at 5 deg AoA, (a) windward (b) leeward (figure taken from Boon and Hillier\textsuperscript{50}).

Figure 2-10 Trailblazer propulsion system with translating centrebody studied by NASA (figure taken from Steffen and DeBonis\textsuperscript{56}).

Figure 2-11 TBCC engine with variable cowl studied by NASA (figure taken from Slater and Saunders\textsuperscript{57}).

Figure 2-12 PREPHA engine studied by MBDA France (figure taken from Falempin\textsuperscript{4}).

Figure 2-13 PROMETHEE engine studied by MBDA France (figure taken from Falempin\textsuperscript{4}).

Figure 2-14 PIAF engine studied by MBDA and Moscow Aviation Institute (figure taken from Falempin\textsuperscript{4}).

Figure 2-15 Ground Demonstrator Engine tested with moveable cowl flap designed and tested by Pratt and Whitney (figure taken from Kazmar\textsuperscript{13}).

Figure 2-16 Movement of internal shock system with change in cowl height (figure taken from Wie and Ault\textsuperscript{31}).

Figure 2-17 Change in throat Mach number, compression ratio, heat loss and kinetic energy efficiency with change in cowl position (figure taken from Wie and Ault\textsuperscript{31}).

Figure 2-18 Edney's Type II\textsuperscript{64} shock-shock interaction forming at throat entrance of the inlet studied by Mahapatra and Jagadeesh\textsuperscript{63}. In this figure, C1 and C2 are interacting oblique shocks, C3 and C4 are resultant oblique shocks, C5 is Mach stem, and T1 and T2 are triple points.

Figure 3-1 High supersonic tunnel in University of Manchester.
Figure 3-2: Variation of unit Reynolds number envelope in relations to Mach number, stagnation pressure and temperature (figure taken from Erdem). 

Figure 3-3: Schematic of two-dimensional planar with double compression ramp where shock-on-lip and shock-on-shoulder conditions are satisfied. $M$ = Mach number, $S$ = shock wave angle, $d$ = flow turning angle, $x$ = geometrical length, $h$ = geometrical height, subscript 0 = freestream, 1 = downstream of first shock, 2 = downstream of second shock and 3 = downstream of third shock. 

Figure 3-4: Schematic of cowl shock and shoulder expansion wave interaction typical for re-expansion inlet. 

Figure 3-5: Interaction between expansion wave and shock wave, where combination of upstream Mach number and flow turning angle could result in regular reflection, Mach reflection or dual-solution. Here, $M_1$ represents the Mach number of flow upstream of shock wave, and $\delta_1$ is the flow turning angle imposed by the shock wave (figure taken from Hillier). 

Figure 3-6: Two-dimensional design sketch of baseline scramjet inlet-isolator used in this experimental campaign. 

Figure 3-7: Three-dimensional schematic of baseline scramjet inlet-isolator used in this experimental campaign. 

Figure 3-8: Collections of different interchangeable top cowl designs for parametric studies: (a) Base-cowl (b) Short-cowl (c) Long-cowl (d) 3deg-cowl and (e) 5deg-cowl. All lengths are quoted in mm. 

Figure 3-9: Combined cowl-sidewall component made from quartz. 

Figure 3-10: Schlieren setup used in this experiment.
Figure 3-11 Pressure sensitivity of different combination of luminophores and substrates (figure taken from Quinn et al.\textsuperscript{109}) ................................................................. 106

Figure 3-12 Temperature sensitivity of different combinations of luminophores and substrates (figure taken from Quinn et al.\textsuperscript{109}) ................................................................. 106

Figure 3-13 Side view of PSP system Setup-1 for compression and isolator surface pressure measurement .................................................................................................................. 107

Figure 3-14 Top view of PSP system Setup-2 for isolator sidewall pressure measurement ............................................................... 107

Figure 3-15 Wavelength spectra of illumination source and luminophores (figure taken from Quinn et al.\textsuperscript{109}) .................................................................................. 110

Figure 3-16 Schematic of IR thermography experimental setup .......................................................................................................................... 116

Figure 3-17 Medium mesh ........................................................................................................................................................................ 126

Figure 3-18 Normalized static pressure of centreline of model scramjet inlet-isolator with different mesh sizes .................................................................................................................. 127

Figure 4-1 Colour schlieren image of scramjet inlet-isolator for baseline case ................................................................................................................................. 128

Figure 4-2 (a) Experimental and (b) numerical schlieren images of scramjet inlet for baseline case focusing only around compression corner ................................................................................................................................. 129

Figure 4-3 Schematic of flow structures on double ramp for baseline case where SW1 = leading edge shock wave, SW2 = separation shock, SW3 = re-attachment shock, \( \Theta_{SW1} \) = angle SW1 to horizontal, \( \Theta_{SW2} \) = angle SW2 to horizontal, \( \Theta_{SW3} \) = angle SW3 to horizontal, \( \delta_1 \) = first ramp angle, \( \delta_2 \) = boundary layer separation wedge angle and \( \delta_3 \) = second ramp angle .......... 129

Figure 4-4 Normalised static wall pressure on compression ramp for baseline case ................................................................................................................................. 131

Figure 4-5 Normalised pressure map of scramjet inlet-isolator focusing only on second ramp using (a) PSP-Ru(II) and (b) PSP-PtTFPP ................................................................................................................................. 135
Figure 4-6 Increase of surface temperature on scramjet inlet for baseline case after 3 seconds of wind tunnel flow start. Note that the cowl component was not added.

Figure 4-7 Cross sectional view of streamwise/Goertler vortices flow pattern.

Figure 4-8 (a) Experimental and (b) numerical schlieren images of the scramjet inlet-isolator for baseline case.

Figure 4-9 Schematic of shock interactions around throat area for baseline case.

Figure 4-10 Comparison of centreline pressure profile estimation using numerical and experimental methods.

Figure 4-11 Comparison between linear and quadratic calibration curves of Stern-Volmer plot.

Figure 4-12 Sketch of oblique shock structure as it interacts with sidewall boundary layer (figure taken from Kubota and Stollery).

Figure 4-13 Surface pressure map for baseline case.

Figure 4-14 Spanwise pressure profile at entrance ($x/L = 0.59$), middle ($x/L = 0.78$) and exit of isolator ($x/L = 0.99$).

Figure 4-15 Sidewall plane pressure contour layered on top of schlieren image for baseline case.

Figure 4-16 Schematic of typical internal shock structure resulted from interactions between cowl shock and expansion wave.

Figure 4-17 Empirical relations between normalized throat Mach number and adiabatic kinetic energy efficiency. Solid triangle is the point plot of current inlet-isolator model in baseline test conditions.

Figure 5-1 Experimental and numerical schlieren images comparison of compression ramp separation at different AoA.
Figure 5-2 Comparison of surface temperature increase for (a) baseline, (b) AoA-2 and (c) AoA-4 case after 3 s of wind tunnel flow start. Note that the cowl component was not added.

.................................................................

Figure 5-3 Comparison of experimental and numerical schlieren for baseline [(a) & (b)], AoA-2 [(c) & (d)] and AoA-4 [(e) & (f)] (angle-of-attack at windward).

.................................................................

Figure 5-4 Isolator surface centreline pressure profile of scramjet inlet-isolator for baseline, AoA-2 and AoA-4 case (angle-of-attack at windward)

.................................................................

Figure 5-5 Inlet-isolator surface pressure map of (a) baseline, (b) AoA-2 and (c) AoA-4 case (angle-of-attack at windward)

.................................................................

Figure 5-6 Spanwise pressure profile at entrance, middle and exit of isolator for AoA-2 case (angle-of-attack at windward)

.................................................................

Figure 5-7 Spanwise pressure profile at entrance, middle and exit of isolator for AoA-4 case (angle-of-attack at windward)

.................................................................

Figure 5-8 Comparison of sidewall pressure contours for (a) baseline, (b) AoA-2 and (c) AoA-4 case (angle-of-attack at windward)

.................................................................

Figure 5-9 Comparison of isolator experimental and numerical schlieren for baseline [(a) & (b)], long-cowl [(c) & (d)] and short-cowl [(e) & (f)] cases

.................................................................

Figure 5-10 Isolator surface centreline pressure profile for baseline, long-cowl and short-cowl cases

.................................................................

Figure 5-11 Comparison of isolator surface pressure map for (a) baseline, (b) short-cowl and (c) long-cowl case

.................................................................

Figure 5-12 Comparison of sidewall pressure contour for (a) baseline, (b) long-cowl and (c) short-cowl cases
Figure 6-1 Comparison of experimental and numerical schlieren for baseline [(a) and (b)],
Cowl-3 [(c) and (d)] and Cowl-5 [(e) and (f)] cases.................................................................188
Figure 6-2 Distance of separation onset point upstream of shoulder according to
experimental and numerical schlieren..................................................................................189
Figure 6-3 Isolator surface pressure profile for baseline, Cowl-3 and Cowl-5 cases........190
Figure 6-4 Comparison of isolator surface pressure map for (a) baseline, (b) Cowl-3 and (c)
Cowl-5 cases.........................................................................................................................192
Figure 6-5 Comparison of sidewall pressure contour for (a) baseline, (b) Cowl-3 and (c) Cowl-
5 cases.....................................................................................................................................193
Figure 6-6 Position of sand paper strip from leading edge ................................................197
Figure 6-7 Comparison of sand paper thickness effectiveness for eliminating compression
corner separation: (a) P150-strip, (b) P100-strip, (c) P60-strip, and (d) MVG-array cases for
boundary layer trip.....................................................................................................................199
Figure 6-8 Temperature increase after 3 s of flow for (a) P150-strip, (b) P100-strip, (c) P60-
strip, and (d) MVG-array........................................................................................................201
Figure 6-9 Dimension of a single MVG..................................................................................203
Figure 6-10 Distance between MVGs in an array .................................................................203
Figure 6-11 The MVG array was positioned with distance, x/L = 0.11 from leading edge and
its middle apex coincides with inlet centreline .....................................................................204
Figure 6-12 The MVG array was positioned with distance, x/L = 0.48 from leading edge and
its middle apex coincide with inlet centreline........................................................................206
Figure 6-13 Experimental schlieren image of isolator flow for (a) baseline (no MVG) and (b)
with MVG-array......................................................................................................................206
Figure 6-14 Normalized static pressure profile without and with-MVG case....................207
Figure 6-15 Inlet-isolator surface pressure comparison of without and with-MVG .............208
Figure 6-16 Spanwise pressure for case with MVG ..........................................................209
Figure 6-17 Typical pressure contour of flow on flat plate with a single MVG (flow from left to right) (figure taken from Li and Liu\textsuperscript{198}) .................................................210
Figure 6-18 Experimental and numerical schlieren for AoA-4 [(a) & (b)], AoA4-Height [(c) & (d)] and AoA4-Length [(e) & (f)] cases (angle-of-attack at windward) .................................................212
Figure 6-19 Isolator surface static pressure for AoA-4, AoA4-Height and AoA4-Length cases (angle-of-attack at windward) ........................................................................214
Figure 6-20 Isolator pressure map for (a) AoA-4, (b) AoA4-Height and (c) AoA4-Length cases (angle-of-attack at windward) ........................................................................216
Figure 6-21 Comparison of sidewall pressure contour for (a) AoA-4 and (b) AoA4-Length cases ..........................................................................................................................217
Figure 6-22 Plot of kinetic energy efficiency vs. throat Mach number ratio .......................219
Figure 0-1 Time histories of stagnation pressure, pitot pressure and stagnation temperature. The temperature scale is shown on the right (figure taken from Erdem\textsuperscript{71}) .........................245
Figure 0-2 Fast-Fourier transformed plot of stagnation and pitot pressure histories (figure taken from Erdem\textsuperscript{71}) ..............................................................................................................246
Figure 0-3 Schlieren image of pitot rake at 2 mm from nozzle exit from start to end of tunnel duration (figure taken from Erdem\textsuperscript{71}) ..............................................................................................................247
Figure 0-4 Schematic of PSP working principle (figure taken from Liu and Sullivan\textsuperscript{132}) ..................249
Figure 0-5 Jablonski energy level diagram (figure taken from Liu and Sullivan\textsuperscript{132}) ..............249
Figure 0-6 schematic of one-dimensional semi-infinite wall model for thin-film sensor ......255
Figure 0-7 Deflection of light path from background image due to change in density gradient (figure taken from Vekatakrishtnan and Meier\textsuperscript{208}) ..............................................................................................................259
Figure 0-8 Movement of elements in an interrogation window for cross-correlation analysis
..........................................................................................................................................................261

Figure 0-9 Schematic of scramjet inlet-isolator with image of suitable pattern glued on the inside of sidewall..................................................................................................................................................263

Figure 0-10 Top view schematic of BOS setup..................................................................................................................264

Figure 0-11 Pre-processed results of (a) wind-off and (b) wind-on (both figures are at 25% of original size) .................................................................................................................................................................266

Figure 0-12 Schlieren images of baseline scramjet inlet-isolator model; (a) colour-schlieren (b) rendered from density gradient calculated using Direct Cross Correlation (DCC) and (c) rendered from density gradient calculated using Fast-Fourier Transform Cross Correlation (FFT-CC)...........................................................................................................................................268

Figure 0-13 BOS setup with reflective surface (figure taken from Tokgoz et al.\textsuperscript{224})..............269
LIST OF TABLES

Table 3-1 Geometrical dimension of scramjet inlet after iterative process ..................................91
Table 3-2 Thermal properties of the material used for infrared experiments.................................116
Table 4-1 Parameters for baseline case conditions ......................................................................128
Table 4-2 Measured shock and flow turning angle relative to horizontal axis from schlieren image ........................................................................................................................................................................................................130
Table 4-3 Flow properties on compression ramp surface using measured shock and flow angle from schlieren image ........................................................................................................................................................................................................130
Table 4-4 Flow properties on compression surface from computational analysis.....................130
Table 4-5 Scramjet inlet-isolator flow properties and performance indicators in baseline test conditions ........................................................................................................................................................................................................155
Table 5-1 Parameters for AoA-2 and AoA-4 test conditions ........................................................159
Table 5-2 Comparison of spanwise average and standard deviation of normalized pressure at entrance, middle and exit of isolator section ........................................................................................................................................................................................................169
Table 5-3 Comparison of calculated isolator exit flow properties and performance indicators ........................................................................................................................................................................................................172
Table 5-4 Parameters for long-cowl and short-cowl test conditions ............................................174
Table 5-5 Comparison of calculated flow parameters and performance indicators for baseline, long-cowl and short-cowl cases ........................................................................................................................................................................................................182
Table 6-1 Case definition .................................................................................................................187
Table 6-2 Comparison of calculated flow properties for baseline, Cowl-3 and Cowl-5 cases195
Table 6-3 Test definitions for different sand paper thickness........................................................197
Table 6-4 Variable geometry case definitions .................................................................................211
Table 6-5 Comparison of flow parameters and performance of inlet at AoA = 4 without and with cowl optimization ................................................................. 218

Table 0-1 Typical freestream flow properties at Mach 5 (table taken from Erdem\textsuperscript{71}) ........ 247
# NOMENCLATURE

## Roman Symbols

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>$A$</td>
<td>Area, $m^2$</td>
</tr>
<tr>
<td>$A$</td>
<td>Stern-Volmer constant</td>
</tr>
<tr>
<td>$a$</td>
<td>Speed of sound, $m/s$</td>
</tr>
<tr>
<td>$\left(A_{\text{th}} \right)_k$</td>
<td>Kantrowitz area limit for self-starting inlet</td>
</tr>
<tr>
<td>$B$</td>
<td>Constant for linear term in Stern-Volmer curve</td>
</tr>
<tr>
<td>$C$</td>
<td>Constant for quadratic term in Stern-Volmer curve</td>
</tr>
<tr>
<td>$C_f$</td>
<td>Schlichting's skin friction coefficient</td>
</tr>
<tr>
<td>$c_p$</td>
<td>Constant pressure specific heat</td>
</tr>
<tr>
<td>$\tilde{C}$</td>
<td>Compression ratio</td>
</tr>
<tr>
<td>$D$</td>
<td>Constant for cubic term in Stern-Volmer plot</td>
</tr>
<tr>
<td>$d$</td>
<td>Flow turning angle relative to initial flow direction, degree</td>
</tr>
<tr>
<td>$E_{\lambda}$</td>
<td>Radiation intensity detected by infrared thermography camera</td>
</tr>
<tr>
<td>$E_a$</td>
<td>Rate of excitation of molecules, molecules population/s</td>
</tr>
<tr>
<td>$e$</td>
<td>Euler’s number</td>
</tr>
<tr>
<td>$F$</td>
<td>Force, $kg \cdot m/s^2$</td>
</tr>
<tr>
<td>$f$</td>
<td>Short notation of oblique shock relations function</td>
</tr>
<tr>
<td>$H$</td>
<td>Total height of isolator section, mm</td>
</tr>
<tr>
<td>$h$</td>
<td>Specific enthalpy, $J/kg$ or height, mm</td>
</tr>
<tr>
<td>$\hbar$</td>
<td>Planck constant</td>
</tr>
<tr>
<td>$I$</td>
<td>Intensity</td>
</tr>
<tr>
<td>$k$</td>
<td>Turbulent kinetic energy</td>
</tr>
<tr>
<td>$L$</td>
<td>Total length of scramjet inlet-isolator, mm</td>
</tr>
</tbody>
</table>
\( M \)  \( \) Mach number

\( \dot{m} \)  \( \) Mass flow rate, kg/s

\( O_2 \)  \( \) Symbol of oxygen

\( p \)  \( \) Pressure, Pa or bar

\( Q \)  \( \) Heat transfer, W/m\(^2\)

\( R \)  \( \) Molar gas constant, m\(^3\)kg/s\(^2\)K\(^1\)mol\(^1\)

\( R^2 \)  \( \) Coefficient of determination for curve fitting

\( Re \)  \( \) Reynolds number

\( R_{ij} \)  \( \) Reynolds stress tensor

\( S \)  \( \) Shock angle relative to initial flow direction, degree or singlet excited state of molecules

\( s \)  \( \) Entropy, J/K

\( \Delta s/c_p \)  \( \) Dimensionless entropy increase

\( T \)  \( \) Temperature

\( T^* \)  \( \) Triplet excited state of molecules

\( u_i \)  \( \) Component of velocity (u in x-direction, v in y-direction and w in z-direction)

\( \bar{u}_i \)  \( \) Mean component of velocity, m/s

\( \hat{u}_i \)  \( \) Fluctuating components of velocity, m/s

\( u^* \)  \( \) Frictional velocity, m/s

\( V \)  \( \) Velocity, m/s

\( \nu_f \)  \( \) Excitation light frequency, Hz

\( W \)  \( \) Total spanwise of scramjet inlet-isolator, mm

\( x \)  \( \) Horizontal axis direction

\( y \)  \( \) Vertical axis direction or first grid size on CFD mesh, mm

\( Y_c \)  \( \) Cowl Height, mm (or in)

\( y^+ \)  \( \) Dimensionless y-distance to wall surface
\( Z_B \) Distance between background plane to lens for BOS setup, m

\( Z_D \) Distance between plane of measurement in density field to background plane for BOS setup, m

\( Z_I \) Distance between image plane and lens for BOS setup, m

**Greek Symbols**

\( \beta \) Oblique shock wave angle relative to initial flow direction, degree

\( \gamma \) Ratio of specific heat

\( \delta \) Flow turning angle relative to horizontal, degree

\( \varepsilon \) Error in measurement or turbulence dissipation rate

\( \varepsilon_y \) Angle deflection in vertical direction as light pass through density field, degree

\( \varepsilon_\lambda \) Spectral hemispherical emittance

\( \eta_c \) Compression process efficiency

\( \eta_{c(ad)} \) Adiabatic compression process efficiency

\( \eta_{KE} \) Kinetic energy efficiency

\( \eta_{KE(ad)} \) Adiabatic kinetic energy efficiency

\( \eta_o \) Overall scramjet engine efficiency

\( \theta \) Flow turning angle induced by a shockwave, degree

\( \mu \) Dynamic viscosity, kg/m·s or Mach angle

\( \mu_T \) Turbulent viscosity, kg/m·s

\( \pi_c \) Total pressure efficiency (also called total pressure ratio)

\( \rho \) Density, kg/m³

\( \sigma_p \) Static pressure distortion index at isolator exit

\( \tau_w \) Wall shear stress, kg/ms²
\( \omega \) Specific dissipation rate

Subscript

0 Station 0 (upstream of first compression shock)
1 Station 1 (downstream of first compression shock)
2 Station 2 (downstream of external compression shocks)
3 Station 3 (downstream of internal compression shocks)
\( \infty \) Freestream
\( d \) Downstream of an oblique shock

\textit{pitot} Pitot

\( t \) Total (stagnation)

\( th \) Throat section

\( u \) Upstream of an oblique shock

Acronyms

AoA Angle-of-attack
BOS Background-oriented schlieren
CFD Computational fluid dynamics
CFL Courant-Friedrichs-Levy
CR Contraction Ratio
DCC Direct cross-correlation
FFT Fast Fourier transform
<table>
<thead>
<tr>
<th>Acronym</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>HCl</td>
<td>Hydrochloric acid</td>
</tr>
<tr>
<td>HSST</td>
<td>High Supersonic Tunnel</td>
</tr>
<tr>
<td>IR</td>
<td>Infrared</td>
</tr>
<tr>
<td>LED</td>
<td>Light Emitting Diode</td>
</tr>
<tr>
<td>MTEOS</td>
<td>Methyl triethoxysilane</td>
</tr>
<tr>
<td>MVG</td>
<td>Micro-vortex generator</td>
</tr>
<tr>
<td>PIV</td>
<td>Particle image velocimetry</td>
</tr>
<tr>
<td>PSP</td>
<td>Pressure sensitive paint</td>
</tr>
<tr>
<td>PtTFPP</td>
<td>Platinum-tetrakis (pentafluorophenyl)porphyrin</td>
</tr>
<tr>
<td>Ru(II)</td>
<td>Ruthenium bathophenanthroline</td>
</tr>
<tr>
<td>SoL</td>
<td>Shock-on-lip</td>
</tr>
<tr>
<td>SoS</td>
<td>Shock-on-shoulder</td>
</tr>
<tr>
<td>SOU</td>
<td>Spatially accurate upwind scheme</td>
</tr>
<tr>
<td>SWBLI</td>
<td>Shock wave boundary layer interactions</td>
</tr>
<tr>
<td>TSP</td>
<td>Temperature sensitive paint</td>
</tr>
<tr>
<td>TR</td>
<td>Throttling ratio</td>
</tr>
<tr>
<td>TLC</td>
<td>Thin layer chromatography</td>
</tr>
<tr>
<td>UV</td>
<td>Ultraviolet</td>
</tr>
</tbody>
</table>
ABSTRACT

After the end of the NASA space shuttle programme, there has been resurgence of interest in developing a single stage-to-orbit spacecraft. The key technology to realize this dream is the airbreathing scramjet engine. The scramjet concept has been around for decades, but much work is still needed in order to eliminate the remaining obstacles to develop a practical working prototype of the engine. Many such obstacles are related to the inlet which functions as the main compression unit for the engine.

Typically, a high speed inlet is designed to function properly in a single flight condition. Such an inlet would experience adverse flow conditions related to various shock-shock interactions, viscous effects, shock-boundary layer interactions, and many other flow phenomena at off-design conditions. The traditional mechanism to mitigate the adverse flow conditions is by varying the inlet geometry at off-design conditions. There are still gaps in understanding the behaviour of inlets at off-design conditions and the effectiveness of variable geometry as inlet flow control. This is partly due to complex flow diagnostics setup, which limits the type, quantity and quality of information that can be extracted from the inlet flow.

The first objective of this thesis was to develop a global inlet measurement system that can provide an abundance of information on inlet flow. The pressure sensitive paint method was employed together with other methods to provide comprehensive understanding on inlet flow characteristics. Calculation of Mach number at the isolator exit using the isolator sidewall pressure map was successfully demonstrated. The measurement of Mach number at the isolator exit has allowed for performance of the inlet to be calculated without the need for intrusive flow diagnostics tools used by previous researchers.
The global measurement system was then employed to investigate the characteristics of the scramjet inlet operating at various off-design conditions. Complex shock structures were observed at the inlet cowl entrance as the angle-of-attack was increased. The relationship of flow quality and inlet performance was examined and discussed. General improvements on the inlet performance were obtained if the size of separation on the compression ramp was reduced. The inlet was also observed to perform poorly when compression shocks impinged on the inner cowl surface.

Cowl deflections were demonstrated to be effective in controlling the internal flow of the inlet and improving its performance. An exploratory study on the role of micro-vortex generators to control boundary layer separation on scramjet inlets has been included as well. Strategies for optimizing an inlet at off-design conditions were analysed, and it was found that any variable geometry combination must maintain high throat-to-freestream Mach number ratio in order to preserve high inlet performance.
DECLARATION

No portion of the work referred to in this thesis has been submitted in support of an application for another degree or qualification of this or any other university or other institute of learning.
COPYRIGHT

I. The author of this thesis (including any appendices and/or schedules to this thesis) owns any copyright to it (the “Copyright”) and s/he has given The University of Manchester the right to use such Copyright for any administrative, promotional, educational and/or teaching purposes.

II. Copies of this thesis, either in full or in extracts, may be made only in accordance with the regulations of the John Rylands University Library of Manchester. Details of these regulations may be obtained from the Librarian. This page must form part of any such copies made.

III. The ownership of any patents, designs, trademarks and any and all other intellectual property rights except for the Copyright (the “Intellectual Property Rights”) and any reproductions of copyright works, for example graphs and tables (“Reproductions”), which may be described in this thesis, may not be owned by the author and may be owned by third parties. Such Intellectual Property Rights and Reproductions cannot and must not be made available for use without the prior written permission of the owner(s) of the relevant Intellectual Property Rights and/or Reproductions.

IV. Further information on the conditions under which disclosure, publication and exploitation of this thesis, the Copyright and any Intellectual Property Rights and/or Reproductions described in it may take place is available from the Head of School of Mechanical, Aerospace and Civil Engineering (or the Vice-President).
ACKNOWLEDGEMENTS

In the name of Allah, the Most Beneficent, the Most Merciful.

All acknowledgements, praises and appreciations are due only to Allah. It is only with his infinite mercy and wisdom; this thesis could ever be concluded. It is for answering His call, a raison d’etre for this thesis ever came into being; for He said in the final testament:

(96:1) Read: In the name of thy Lord who createth.

It is of great fortune that a man is given the ability to read; where a thinking mind could decipher the meaning of everything that surrounds him, giving those names and transcribe them back into symbols and words for other thinking mind to comprehend. The cognitive ability to read the different signs in natural courses, to understand them, to critically reflects upon them with both admiration and sense of wonder, to achieve the elevated status of enlightenment and awakening, and finally to manipulate those natural occurrences for the common benefits of people; has not been bestowed upon other creation of the Almighty, the Most Wise, the Most Benevolent. Indeed, the Almighty said:

(3:190) Behold! In the creation of the heavens and the earth and (in) the difference of night and day are tokens (of His sovereignty) for men-of-wisdom (Ulul Albab),
(3:191) (Men-of-wisdom are the ones who) remember Allah, standing, sitting, and reclining, and consider the creation of the heavens and the earth, (and say): Our Lord! Thou createth not, this in vain. Glory be to Thee! Preserve us from the doom of Fire.

This thesis is then, O Lord, a dedication to You; a small achievement in the long path to attain wisdom and understanding of the surrounding world.

I am indebted to Prof. K. Kontis for his valuable advice and guidance. I am also very grateful for the assistance, technical or otherwise, provided by School of MACE and its dedicated staff. I would also like to thank all my friends that have been there for me for the small chat, coffee, and camaraderie.

I could not have thanks enough for the moral support and unconditional love given by my mother, Zainab Ismail. I could not even repay my father, Che Idris, for all the financial backings that he has given me.

To my daughter, Aneesa, and my son Ahmad Yassin; your laughter was a soothing music to my ears when I was finishing up my writing.

To my wife Azyyati; your passionate love has left me perplexed and mystified. Your companionship has left me challenged and enhanced, priming me in my pursuit of knowledge.
CHAPTER 1 INTRODUCTION

1.1 Introduction and Motivation

Since the birth of first powered, heavier-than-air aircraft, human has endeavoured to fly faster and higher. From the first flight in 1902 that flew over only 37 m in 12 seconds, aviators have been able to break the previously impenetrable sound barrier in 1947. The aftermath of World War II has presented the Western world with rocket power and enabled man to conquer the last frontier of outer space. In 1961, sky is no longer the limit as proven by the first astronaut Yuri Gagarin in his orbit around planet Earth before re-entering Earth’s atmosphere with the speed in excess of Mach 25. Since then, high speed flight of hypersonic regime has found applications in many parts of human life, especially space access and defence.

Even though rocket propulsion system is capable of high speed flight, but heavy oxidizer needed on-board presents a limit on useful payload it could carry. Small payload with heavy aircraft translate into expensive cost; hence a solution is needed. Air-breathing propulsion system would eliminate the requirement to carry an oxidizer, and the scramjet engine is the most suitable type of air breathing engine for hypersonic flight regime.

Although scramjet was first developed in 1950s as a derivative of ramjet engine, the field is still maturing and many more issues need to be solved before practical application of scramjet engine in flight\(^1\). Most of the main scramjet engine problems in urgent need of attention centred on the inlet compression and pre-combustion components\(^2\). Typical inlet flow phenomena in need of improvements are [1] boundary layer transition, [2] shock-boundary layer interactions (SWBLI), [3] boundary layer separation and re-attachment, [4]
shock-shock interactions, [5] glancing shock interactions, and many more. Compression component efficiency is very important for high velocity flight as large proportion of the energy needed for propulsive thrust is already contained in the high kinetic energy of the flow itself, and conserving this energy during compression would become as important as maintaining efficient combustion. Hence, novel methods of flow control for scramjet’s compression component (i.e. the inlet) need to be investigated.

1.2 Ramjet and Scramjet Basic Principle

The basic principle in any jet engine, be it turbojet, ramjet, scramjet or even shcramjet, is that propulsive thrust is a by-product of reaction from accelerating a mass-volume of gas. Newton’s Third Law of Motion states that mutual action and reaction between two bodies must be equal in magnitude but opposite in direction. In this case, a body, a propulsion device, accelerates a mass-volume of working fluid; thus, thrust, equals to the magnitude of change of momentum of the working fluid, is exerted on the body. To illustrate, consider Figure 1-1 below:

![Figure 1-1 Schematic of air-breathing propulsion device concept](image)

A streamtube of gas with velocity, $V_0$, and mass flow rate, $\dot{m}_0$ enters the generic propulsion device before being accelerated to exit velocity $V_e$, and mass flow rate, $\dot{m}_e$. Thrust exerted on the propulsion device is:
Equation 1-1

\[ \text{Thrust} = F = (\dot{m}_e V_e) - (\dot{m}_0 V_0) \]

From the equation, assuming negligible fuel mass flow rate addition \((\dot{m}_e \approx \dot{m}_0)\), one can see that thrust can be maximized if the velocity difference of \((V_e - V_0)\) is maximized. That is the essence of every type of jet engine, where a streamtube of air is compressed to high pressure before being injected with fuel and 'burned' to further increase speed.

In turbojet or turbofan engine, air is compressed by a mechanical compressor before being injected with fuel for combustion. The combustion process will increase the flow velocity, pressure and temperature to be exhausted through nozzle for thrust production. Part of exhaust energy is captured by turbine component to drive the mechanical compressor.

At sufficiently high speed, compression effect by flow deceleration is enough to increase pressure for combustion process. This is the premise behind ramjet and scramjet engine, where high dynamic pressure of the incoming flow is converted into high static pressure rise by decelerating the flow into small volume without the need for mechanical compressor. In conventional ramjet, incoming air speed is decelerated subsonically\(^4\) to about Mach 0.3, thereby increasing its pressure and temperature to a condition best for injecting, mixing and burning fuel.

The compression could be achieved by subjecting the incoming flow through one or more oblique shock wave before the flow pass through converging duct to decelerate the supersonic flow into subsonic, and diverging duct to further decelerate the flow to lower subsonic speed. Fuel is then injected and the high flow temperature due to deceleration vaporizes the fuel, enhancing air-fuel mixing. The hot gas produced from combustion then
accelerates to supersonic in the converging part of the nozzle before the diverging part further accelerates it to Mach number higher than inlet speed (see Figure 1-2).

Figure 1-2 Working schematic of typical ramjet engine (figure taken from Heiser and Pratt\(^5\))

The problem with ramjet propulsion is obviously the high speed needed for its operation, thus necessitating an aircraft to have a separate engine for take-off before turning the ramjet engine on. That is the reason why in historical example, ramjet engine is usually coupled with turbine-based engine or rocket propulsion to reach its cruise speed.

At around Mach 3, the efficiency of thrust generation for ramjets would surpass those of turbojets due to its simple design, low weight and higher temperature tolerance\(^4\). However, around Mach 5, it is more beneficial for combustion to be done supersonically instead of subsonically. The reason is because the total pressure loss and thermal load produced by decelerating flow to subsonic speed becomes too large at high Mach number\(^4\). Scramjet circumvents the problems by maintaining airflow speed for supersonic combustion (see Figure 1-3).
As the combustion is done supersonically, scramjet engine design does not need a converging-diverging inlet and nozzle, thus simplifying its design process. Scramjet inlet is usually much longer than a ramjet since the former utilises only oblique shock waves for compression. This makes the engine and airframe much more integrated, where the aircraft forebody is typically utilised as the first compression ramp for the engine.

1.3 Historical Overview

The concept of ramjet propulsion actually has seniority than the concept of turbojet engine. The first patent was awarded to Lake in 1909, and the first treatise on subsonic ramjet was published by Lorin\(^6\) in 1913. However, ramjet engine did not found much practical application other than theoretical studies due to its requirement of high speed flight necessary for compression, which cannot be satisfied in the early days of aviation. With the advent of turbojet and modern rocket engines around 1930s, aviators have found that ramjet could be used as a secondary engine to propel aircraft at higher speed after it has attained the initial ramjet starting speed.
Ramjet received renewed attentions during post-WWII in 1950s, with many countries envisioned its applications in missiles technology. The impetus of the sudden interests came from a new type of weapon that the Germans had unleashed upon the world in WWII; the V1 two-stage, rocket-ramjet powered, subsonic cruise missiles. Without the need for the missile to carry its own oxidizer, heavier warhead could be carried, thus making it a key technology in national defence. Ramjet continued its dominant firm footings in missiles application throughout history and its detailed historical overview can be found in a paper by Fry⁶.

The first ramjet application in aircraft has been demonstrated by Leduc in France. His prototype, Leduc 010 (Figure 1-4), became the first ramjet powered aircraft in history, as it flew using only its ramjet engine after detachment from a ‘carrier’ aircraft. In 1953, Griffon II (Figure 1-5) became the first aircraft to fly on turbo-ramjet engine, which combines conventional turbojet and ramjet engine cycle. The aircraft managed to achieve quite an impressive speed of Mach 2.1. In 1966, the famous SR-71 blackbird was introduced and became the fastest air-breathing manned aircraft ever, thanks to its ingenious variable geometry designed turbo-ramjet engine.

![Figure 1-4 Leduc 010 ramjet aircraft (figure taken from Fry⁶)](image)
Realizing the superior efficiency of scramjet engine at very high Mach number, researchers began pioneering work on scramjet propulsion in late 1950s. The first test on scramjet engine was done by Ferri in 1960s. This gave birth to many scramjet projects, with the most significant is Hypersonic Research Engine (HRE) programme launched in 1964. A demonstrator scramjet engine was designed and tested using X-15 aircraft in Mach 4 – 7 flow. Even though it failed to demonstrate a safe manned hypersonic aircraft, it provided a comprehensive database of inlet and combustor performance that is valuable for future researchers before the programme was cancelled in 1968.

Another significant milestone in scramjet history is the National Aero-space Plane (NASP) program launched in 1980s in the USA to develop a single stage-to-orbit aircraft. Officially
named as X-30 (see Figure 1-7), it was heralded as the aircraft that can shorten the Washington to Tokyo flight duration to less than 2 hours with velocity of up to Mach 25, but it was cancelled prematurely in 1993 due to unbearable cost. Despite its short-lived hype, the programme would prove to be a valuable starting point for resurgent in hypersonic flight research in later years.

![Figure 1-7 Artist impression of X-30 (figure taken from Kazmar)](image)

German also launched a similar hypersonic space-plane research campaign around 1980s. Known as Sanger II program\(^\text{14}\), it focused on developing a two-stage orbital aircraft that can carry crew and payload to space before meeting similar fate as NASP in America.

In recent years, Hyper-X program\(^\text{15}\) has become the centre of technological breakthrough in scramjet engine research. X-43 aircraft (see Figure 1-8), a part of the Hyper-X program, set the fastest airbreathing engine speed record of Mach 9.8 flight for 12 seconds after it was launched from B-52 aircraft in 2004\(^\text{16}\). Then in May 26\(^{\text{th}}\), 2010, another aircraft of the same program, X-51 (see Figure 1-9), has achieved the longest scramjet combustion duration\(^\text{17}\) of 140 seconds at Mach 5.
There are a lot more scramjet development programs being pursued in other countries such as Russia, France, Australia, and others, but this thesis could merely note but a few. The main findings of studying the history of ramjet and scramjet is that it has much applications in space access and defence since its inception in early days, and will continue to dominate the airspace in the future.

1.4 Aim and Objectives

Traditional method of scramjet inlet flow control uses hydraulically powered variable geometry inlet. The inlet geometry would be varied at off-design condition for inlet unstart mitigation and performance maximization. Testing the inlet in ground based facilities typically require complex setup of intrusive flow diagnostics techniques. Thus, the ultimate
aim of this thesis is to develop a simpler method of analysing scramjet inlet-isolator characteristics using a global measurement system, and then to utilise the method to study the behaviours of a generic inlet in off-design conditions.

The research campaign consists of different significant milestones. The first is to design a simple generic inlet to be used with variable geometry cowl. The inlet must be designed as simple as possible so as to avoid observation of the variable geometry effect being obscured by complex inlet flow features.

The second is to apply a global non-intrusive flow measurement technique to investigate the scramjet inlet-isolator with the intention of finding the relations between internal flow structures and overall inlet-isolator performance. The inlet will be subjected to various off-design flow conditions and their efficiency will be calculated. Validated numerical methods will help in interpreting experimental data and fill the gap in observation.

The inlet will be optimized and investigated in the final stage using variable cowl geometry method. A novel SWBLI control via micro-vortex generator in scramjet inlet-isolator will be investigated as well.

1.5 Structure of Thesis

Chapter 1 covers the basic introduction of scramjet engine and its contribution to overall aerospace technological advantage. Chapter 2 contains detailed discussions on the fundamentals of scramjet inlet principle, which must be considered in designing hypersonic aircraft. Topics in this chapter include inlet design, performance study, start-unstart behaviour and inlet off-design characteristics. Typical experimental techniques used to
characterize an inlet found in published literatures are listed and new inlet flow diagnostics concepts are presented. Reviews of the studies related to variable geometry inlet flow control are also presented in this chapter. Next, the facilities and principle of experimental works done are explained in Chapter 3. Detailed methodology utilised in designing a generic scramjet inlet-isolator for this research campaign is also included in Chapter 3. Then, the practicality of using the new flow diagnostic methods to characterize the scramjet inlet-isolator at design conditions is discussed in Chapter 4, whereas the behaviours of the inlet-isolator at off-design conditions are presented in Chapter 5. Finally, new methods of optimizing the inlet performance are explored in Chapter 6. Overall conclusions of the current research campaign are summarized in chapter 7. Room for improvements and future studies potentials are added in the final chapter to conclude the thesis.
CHAPTER 2 BACKGROUND STUDY ON SCRAMJET INLET

2.1 Scramjet Inlet Design

The role of an inlet (sometimes also referred as intake) is to capture air for propulsion and conditioning (radiator) purposes. It must be able to achieve the objective without incurring too much weight and drag. Thrust will be maximized when the efficiency of process, in which air flow kinetic energy is transformed into pressure potential energy, is maximized. In supersonic and hypersonic flight regime, high efficiency compression could be achieved without the need of mechanical compressor such as that in turbojet. The inlet itself could be constructed to compress the incoming air. Thus, for supersonic and hypersonic aircrafts, the inlet must provide an efficient compression process.

For scramjet engine, the goal of its inlet component is to provide efficient air flow compression and deliver supersonic flow to the combustor. Many designs have been produced in literature to achieve this task. It could be subdivided into two-dimensional (axisymmetric or planar) and three-dimensional types. A good review of different inlet types is given in the book chapter written by Wie and summarized below.

2.1.1 Two-dimensional Planar

This type of inlet utilised a set number of oblique shock produced by compression ramps (see Figure 2-1). Using contoured surface to produce Mach fan for isentropic compression is possible but not favourable, as this will complicate the design process, and the inlet would end up with excessive length (and weight). Therefore, an inlet designer will often settle with
a number of discreet oblique shock generator ramps. As the type is very simple to design and analyse, it will be the basis of research in this thesis.

2.1.2 Two-dimensional Axisymmetric

Two-dimensional axisymmetric inlet has a lot of similarity with two-dimensional planar except that in the former, the flow requires more turning to achieve the same compression ratio with the latter. Two of the most popular inlet of this type are the Oswatisch inlet and Busemann inlet.

Oswatisch inlet (see Figure 2-2) utilised a combination of a conical shock, compression fan and single cowl shock that is cancelled at inlet shoulder. This inlet suffers from disadvantages such as high cowl lip drag and high sensitivity to angle-of-attack.
Busemann inlet is the exact opposite of Oswatisch inlet, where the former is inward turning while the latter is outward turning. This means that the compression is done internally and all compression shock meet at a single point leading to free standing conical shock wave. This inlet offers high performance compared to other types but usually suffers with unstart problem in steady flow conditions.

2.1.3 Three-dimensional Inlet

Flow in a three-dimensional inlet is forced to turn in an extra dimension consequently, resulting in a reduced length inlet. This is very beneficial since shorter inlet will incur less weight.

One of the most popular three-dimensional inlets is the sidewall compression inlet. Compression is provided by the sidewalls in addition to ramp compression. The leading edges are usually swept to assist in inlet starting process (see Figure 2-3). The disadvantage of the inlet is that the inlet has more complex flow structures, making it harder to analyse.

Figure 2-3 30 degree sweep sidewall compression inlet (figure taken from Holland and Perkin²⁰)
2.2 Scramjet Inlet-isolator Performance

Figure 2-4 Scramjet inlet station definitions

Figure 2-4 shows typical classification of different components in two-dimensional scramjet inlet-isolator. Station 0 is the freestream condition and prior to external compression. Station 1 is downstream of the first compression shock, which is typically provided by an aircraft forebody. The end of external compression or the start of internal section is designated as Station 2. This station is also called the inlet throat, which is defined as the smallest cross section area of internal section of inlet. Station 3 marks the end of isolator section or downstream of any internal compression shock. Inlet engineers would mostly concern themselves with flow phenomena and performance of components between only station 0 and 3.

In Heiser and Pratt\(^{21}\), it was established that the performance of scramjet engine, measured by overall efficiency, \(\eta_o\), is influenced heavily on compression efficiency \(\eta_c\). Consequently, compression efficiency is related\(^{18}\) to total pressure efficiency, \(\pi_c\), kinetic energy efficiency, \(\eta_{KE}\), and dimensionless entropy increase, \(\Delta s/c_p\).
2.2.1 Adiabatic Compression Process Efficiency

Consider the diagram in Figure 2-5 depicting typical thermodynamic cycle of fluid operating in an airbreathing engine. Station 0 to 3 describes the adiabatic compression process where the process will increase static temperature and entropy. Station 3 to 4 is the combustion process where heat is added at constant pressure, while station 4 to 5 is the re-expansion of the hot exhaust flow to the freestream pressure. The station labelled x is the imaginary station if the compressed fluid at station 3 is expanded isentropically to freestream pressure.

The efficiency of the compression process is defined as the ratio of total energy contained at station 3 to the energy initially in the control volume of flow at freestream (station 0). In other words, compression efficiency shows how much energy is spent by the compression process. This can be described by:
Equation 2-1

\[ \eta_c = \frac{h_3 - h_x}{h_3 - h_0} = \frac{c_p(T_3 - T_x)}{c_p(T_3 - T_0)} \]

Here, \( h \) denotes total energy, \( c_p \) is the constant pressure specific heat and \( T \) denotes static temperature.

2.2.2 Total Pressure Efficiency

Total pressure efficiency is the ratio of total pressure at isolator/combustion interface to total pressure at freestream. The compression in scramjet inlet is achieved in a series of discreet compression shock, where total stagnation pressure drops after each shock. Total pressure efficiency is a measure that demonstrates how much total pressure is wasted through the compression process. It has been demonstrated as the most important performance indicator in subsonic and supersonic compression system, but for hypersonic engine, its importance is less obvious. This is because stagnating hypersonic flow would result in gas molecules ionization, and calculating total pressure would not be straightforward. Total pressure recovery in hypersonic inlet is usually very small in comparison to turbojet and ramjet engine, which makes judging the inlet performance from quoted value quite non-trivial. The equation for total pressure efficiency is given below:

Equation 2-2

\[ \pi_c = \frac{p_{t3}}{p_{t0}} = \frac{p_3}{p_0} \left\{ \frac{1 + \frac{\gamma - 1}{2} M_3^2}{1 + \frac{\gamma - 1}{2} M_0^2} \right\}^{\gamma/(\gamma-1)} \]

Here, \( p_t \) denotes the total pressure, \( p \) is the static pressure, \( \gamma \) is the specific heat ratio and \( M \) is the Mach number. Total pressure ratio could be linked to compression efficiency by:
2.2.3 Kinetic Energy Efficiency

Kinetic energy efficiency is defined as the ratio of kinetic energy of compressed flow at station 3, if it is to be expanded isentropically to freestream pressure, to the initial kinetic energy available at station 0. This measure the effectiveness of the compression process in terms of energy management. In hypersonic application, kinetic energy efficiency is more useful than total pressure efficiency, as the former relates to momentum preservation, which is an important factor for a propulsion unit. Kinetic energy efficiency can be written in terms of velocity ratio squared.

Equation 2-4

\[ \eta_{KE} = \frac{V_x^2}{V_0^2} \]

Where \( V_x \) is the velocity of compressed flow at throat if it is expanded isentropically to freestream pressure. Denote this station by subscript \( x \). By relating velocity to Mach number and speed of sound, \( a \), we have:

Equation 2-5

\[ \eta_{KE} = \left( \frac{M_x a_x}{M_0 a_0} \right)^2 \]

Local speed of sound can be related with local temperature by:
Equation 2-6

\[ a^2 = \gamma RT \]

Isentropic equation can give the relation between local Mach number and local temperature.

Equation 2-7

\[ \frac{T_t}{T} = \left( 1 + \frac{\gamma - 1}{2} M^2 \right) \]

From equations above, we can have:

Equation 2-8

\[ \eta_{KE} = \left( \frac{1}{M_0^2} \right) \left( \frac{T_{tx}}{T_x} - 1 \right) \left( \frac{2}{\gamma - 1} \right) \left( \frac{T_x}{T_0} \right) \]

Equation 2-9

\[ \eta_{KE} = \left( \frac{1}{M_0^2} \right) \left( \frac{2}{\gamma - 1} \right) \left( \frac{T_{tx}}{T_0} - \frac{T_x}{T_0} \right) \]

If we assume compression is done adiabatically, total temperature will be conserved from station 0 to station 3 at throat. Total temperature is also conserved from throat to station x:

Equation 2-10

\[ \frac{T_{tx}}{T_0} = \frac{T_{t3}}{T_0} = \frac{T_{t0}}{T_0} \]

Thus, we would obtain equation below similar to the one given by Heiser and Pratt\textsuperscript{21}.

Equation 2-11

\[ \eta_{KE(ad)} = 1 - \left( \frac{2}{\gamma - 1} \right) \left( \frac{1}{M_0^2} \right) \left[ \left( \frac{T_x}{T_0} \right) - 1 \right] \]

Where, \( \eta_{KE(ad)} \) is the adiabatic kinetic energy efficiency, and \( T_x \) is given by:

Equation 2-12

\[ T_x = T_{t3} \left( \frac{p_0}{p_{t3}} \right)^{\left( \frac{\gamma - 1}{\gamma} \right)} \]
Kinetic energy efficiency is related to compression efficiency by:

Equation 2-13

\[ \eta_{c(ad)} = 1 - \frac{(\gamma - 1) M_0^2}{2} \left( \frac{1 - \eta_{KE(ad)}}{\frac{T_3}{T_0} - 1} \right) \]

If the flow is assumed adiabatic, then compression efficiency is labelled as \( \eta_{c(ad)} \).

### 2.2.4 Dimensionless Entropy Increase

Another useful performance indicator is dimensionless entropy increase. The indicator shows the irreversibility of the flow as it goes from station 0 to station 3. By integrating Gibbs entropy equation, we get:

Equation 2-14

\[ \frac{s_3 - s_0}{c_p} = \ln \left( \frac{T_3}{T_0} \right) - \left( \frac{R}{c_p} \right) \ln \left( \frac{p_3}{p_0} \right) \]

The relation between adiabatic compression efficiency and entropy rise is shown in the equation below:

Equation 2-15

\[ \eta_c = \frac{T_3}{T_0} - \frac{e^{(s_3 - s_0)/c_p}}{\frac{T_3}{T_0} - 1} \]
2.2.5 Scramjet Inlet-isolator Performance Measurement

As shown by all equations above, it is obvious that aerodynamic performance indicators of total pressure efficiency, $\pi_c$, kinetic energy efficiency, $\eta_{KE}$, and dimensionless entropy increase, $\Delta s/c_p$, are easily interchangeable between each other. In fact, it is acceptable for an inlet to be quoted with just one performance indicator. However, each one represents different aspect of the performance, and an inlet will usually be quoted with at least two indicators as pair. Another glaring conclusion to be made is that to calculate all the aerodynamic performance indicators and inlet adiabatic compression efficiency, an inlet engineer must be able to measure the Mach number $M_3$, static pressure, $p_3$ and static temperature, $T_3$.

It could be argued that according to station definition of Figure 2-4, station 3 is not the geometric throat of the inlet, and the flow properties of station 2 should be used instead (see for example the performance calculation methodologies shown in Wie\textsuperscript{18} and Heiser and Pratt\textsuperscript{21}). However, this thesis considers the inlet and isolator part to be inseparable, as the isolator only provides “background” shock waves and does not contribute significantly to overall compression ratio\textsuperscript{22}. Treating inlet and isolator as one component also makes more sense, since the combustor component will be “fed” with flow from isolator and thus, the performance of the inlet calculated using flow properties measured at geometric throat does not represent the flow encountered by the combustor. This approach is also logical since the inlet-isolator is always treated as a single compression device for Brayton cycle analysis of an overall scramjet engine.
2.2.5.1 Stream Thrust Concept

The most common method to measure the set of flow parameters $M_3$, $p_3$, and $T_3$ is by utilising the stream thrust concept. First proposed by Curran and Craig\textsuperscript{23} as a method to predict the performance of hypersonic ramjet engine, this concept has become the standard experimental investigation method found in popular scramjet engine textbook\textsuperscript{18,21,24}. An almost similar method of inlet performance prediction of using thrust-drag measurement has been shown by Balent and Kutschenreuter\textsuperscript{25}. Some practical examples of this method can be found in works by Matthews\textit{ et al.}\textsuperscript{26}, Matthews and Jones\textsuperscript{27}, Roberts\textsuperscript{28}, Roberts and Wilson\textsuperscript{29}, and Goon’ko\textsuperscript{30}.

The premise of the idea is simple. It starts with the conservation of mass, momentum and energy of a control volume similar to sketch in Figure 2-4. Considering the continuity equation:

\textbf{Equation 2-16}

$$\dot{m}_0 = \dot{m}_3 = \rho_0 A_0 v_0 = \rho_3 (A_i - A_r) v_3$$

Here, $\dot{m}$ is the mass flow rate, $\rho$ is the density, and $v$ is the velocity. $A_i$ and $A_r$ are the cowl lip height and throat height from horizontal datum, respectively. The momentum equation can be notated as:

\textbf{Equation 2-17}

$$p_0 A_i - p_3 (A_i - A_r) - F_x = \dot{m}_0 (v_3 - v_0)$$

Where $F_x$ is the net axial force acting on a sting holding the scramjet inlet-isolator model inside the wind tunnel test section.

Considering the energy conservation for the control volume yields:
Heat transfer $Q$ is considered in the calculation since the flow will have lower energy by transferring it upon contact with the compression surface. Overall surface heat transfer can be measured by discrete temperature gauge, optical thermal mapping or measuring heat transfer to water cooling under the inlet surface. Mass flow rate is measured by attaching a mass flow meter at the isolator exit\textsuperscript{18}. Net force on the sting is typically measured using a force balance. Using the three measurement techniques and known freestream properties, iterative procedures can be applied to solve for $M_3$, $p_3$, and $T_3$. The set of data is hard to obtain experimentally but can be simplified, depending on the level of accuracy required. For example, Wie and Ault\textsuperscript{31} proposed a simplified stream-thrust analysis by combining experimental and numerical analysis. They measured drag force and heat transfer rate experimentally and obtained the capture mass flow rate from numerical simulations of the inlet. Mitani et al.\textsuperscript{32} reported that internal drag can be estimated from high spatial resolution pressure map of the whole inlet. Emami et al.\textsuperscript{33} suggested that mass flow rate can be estimated empirically without the need to install mass flow meter. However, it appears that their empirical equations are specific to their inlet geometry. The flow at throat can be assumed to be adiabatic and $Q$ could be eliminated from Equation 2-18, with the final compression efficiency would be notated as $\eta_{c(ad)}$. 

Equation 2-18

$$h_0 + \frac{v_0^2}{2} - \frac{Q}{\dot{m}_0} = h_2 + \frac{v_2^2}{2}$$
2.2.5.2 Pitot Rake Measurement

Another method to measure the set of flow properties is by placing pitot pressure rake at the inlet throat or at isolator exit. It was actually the method of choice prior to development of stream thrust analysis described in Section 2.2.5.1.

Practical description of pitot rake measurement for hypersonic inlet can be found in early pioneering work by Berstein and Haefeli\textsuperscript{34–36} in NACA during early 1950s. They used pitot rake to find the average pitot pressure, where the combination with throat static pressure (average value of top and bottom wall static pressure) can be used to calculate throat Mach number. The flow encountered by each pitot probe is subsonic due to the appearance of bow shock and requires the use of Rayleigh pitot formula given below:

Equation 2-19

\[
\frac{p_{t,pitot}}{p_3} = \left[ \frac{(\gamma + 1)M_3^2}{4\gamma M^2 - 2(\gamma - 1)} \right]^{\gamma-1} \left[ \frac{1 - \gamma + 2\gamma M_3^2}{\gamma + 1} \right]
\]

The total pressure measured by each pitot, \( p_{t,pitot} \) is not the real total pressure of the flow at throat due to the loss induced by bow shock. The real stagnation pressure must be found using:

Equation 2-20

\[
\frac{p_{t,3}}{p_3} = \left( 1 + \frac{\gamma - 1}{2} M_3^2 \right)^{\gamma-1}
\]

Usually, there is a number set of pitot used together at throat which for which the final throat pressure and Mach number is taken from the average of all pitot probes. The total pressure recovery, \( \pi_c \), of the inlet can then be calculated by using Equation 2-2. Thus, kinetic energy efficiency can be calculated by a simplified equation of:
Extensive scramjet inlet experimental analyses done in German Aerospace Center (DLR)\(^{37-39}\) also utilised pitot pressure rake for calculating inlet performance but with slightly different set of equations. The initial strategy is similar, where pitot rake is used to measure average \(p_{t,pitot}\). The total pressure recovery is then calculated directly without correction such as in equation below:

\[
\eta_{KE} = 1 - \left(\frac{1}{\pi_c}\right)^\frac{\gamma-1}{\gamma} - 1 \frac{\gamma-1}{2} M_0^2
\]

2.2.5.3 Potential for New Performance Measurement Concept

Based on the discussions in Sections 2.2.5.1 and 2.2.5.2, it can be understood that the stream thrust concept is more complicated and involves more experimental techniques compared to pitot rake measurement method. The former also incorporates the effect of heat transfer but its final uncertainty is higher than the latter since it depends on at least three measurement devices (i.e. mass flow meter, temperature/heat transfer transducer,
force balance). Both stream-thrust and pitot rake methods require complex setups that are intrusive to the inlet flow and produce discrete data that leave significant uncertainties between each measurement points.

This thesis recognizes a gap in inlet flow diagnostics technology and proposes a simpler method of obtaining flow parameters at isolator exit applicable to planar two-dimensional scramjet inlet. The figure below is considered:

![Figure 2-6 Typical shocks structure of (a) ideal shock-on-shoulder inlet and (b) non-ideal, re-expansion inlet](image)

Figure 2-6 (a) shows that the flow from freestream will be turned a few times by the external compression surface before being turned by a single cowl shock to enter the throat and isolator. As the cowl shock impinges directly on the sharp expansion corner (shoulder), the shock will cancel the expansion fan effect, and there will be no shock reflections progressing downstream towards the isolator exit. This condition is called the shock-on-shoulder (SoS) condition and is considered ideal as the flow downstream of shoulder is uniform. However,
the inlet with SoS usually has very high contraction ratio, and a designer might deliberately design for the cowl shock to impinge downstream of shoulder. This inlet is classified as re-expansion inlet by Matthews et al.\textsuperscript{26,27} and they explained that the shock will interact with shoulder expansion fan producing reflective shocks structure downstream. The oblique shock reflections that follow the cowl shock impingements are called “background shockwaves”\textsuperscript{22} and they exist solely as a mechanism for the flow to gradually change its direction to be parallel with top and bottom wall surface. The background waves are indeed the precursor for shock-train\textsuperscript{22}. With enough isolator length, the background waves will gradually diminish in strength and the final oblique shock at isolator exit is usually very weak and possesses almost similar behaviour to a Mach wave. Thus, the flow properties at isolator exit can be simply calculated by analysing the final oblique shock such as shown in Figure 2-6 (b).

The flow properties such as Mach number $M$, static pressure $p$, static temperature $T$, density $\rho$ and others at downstream (subscript $d$) could be described by simple oblique shock relations to the properties at upstream of the shock (subscript $u$). The equations are:

**Equation 2-23**

$$\frac{p_d}{p_u} = 1 + \frac{2\gamma}{\gamma + 1} [M_u^2 \sin^2 \beta - 1]$$

**Equation 2-24**

$$\frac{\rho_d}{\rho_u} = \frac{(\gamma + 1)M_u^2 \sin^2 \beta}{(\gamma + 1)M_u^2 \sin^2 \beta + 2}$$

**Equation 2-25**

$$\frac{T_d}{T_u} = \frac{p_d \rho_u}{p_u \rho_d}$$
Here, $\beta$ is the shock angle relative to direction of flow upstream of the shock. From Figure 2-6 (b):

**Equation 2-26**

$$\beta = \phi_s - \phi_u$$

Here, $\phi_s$ and $\phi_u$ are the shock angle and initial flow direction angle relative to horizontal axis, respectively.

The final objective is to find the Mach number downstream of the final shock and it is given by:

**Equation 2-27**

$$M_a = \frac{1}{\sin (\beta - \theta)} \sqrt{1 + \left(\frac{\gamma - 1}{2}\right) M_u^2 \sin^2 \beta}
\frac{1}{\sqrt{(\gamma M_u^2 \sin^2 \beta) - \left(\frac{\gamma - 1}{2}\right)}}$$

Where $\theta$ is the flow turning angle given by:

**Equation 2-28**

$$\theta = \phi_d - \phi_u = \tan^{-1} \left\{ 2 \cot \beta \frac{M_u^2 \sin^2 \beta - 1}{M_u^2 (\gamma + \cos 2\beta) + 2} \right\}$$

If the isolator is assumed to be long enough and has constant shape from throat section to exit, the final shock is very weak and approaches the characteristics of a Mach wave. This is because the flow upstream of the final shock is almost parallel to the wall and the shock will produce infinitesimal amount of flow turning. Thus, in this case:

**Equation 2-29**

$$\beta \approx \phi_s$$
With this simplifying assumption, Mach number at isolator exit could be determined with high accuracy using global map of pressure, density or temperature. For example, consider that the sidewall of the isolator section is painted with pressure sensitive paint, and three-dimensional effects of the flow at sidewall are negligible. With optical access on the other side, pressure rise ratio across the final shock $\frac{p_f}{p_u}$ and shock angle $\phi_s$, could be measured easily. In this case, isolator exit Mach number $M_3$ can be calculated by solving Equation 2-23, Equation 2-27 and Equation 2-28. If, let say that density field could be determined, then Equation 2-24, Equation 2-27 and Equation 2-28 could be solved to find $M_3$.

With proper optical access of the isolator section, global measurement methods such as pressure sensitive paint (PSP), temperature sensitive paint (TSP), background-oriented schlieren (BOS), quantitative colour schlieren and many others could be used to map the final shock structures. Global map of the flow properties are always superior as they give a higher accuracy of the average one-dimensional value compared to using the input from a set of discrete measurements points. This performance measurement concept also relies on non-intrusive method that could be set up easily.
2.3 Inlet Start/Unstart Characteristics

2.3.1 Inlet Starting Behaviour

An inlet designed to achieve high performance is useless if it cannot be started. Inlet is considered started when a desired state of flow is established inside the inlet. In a paper by

Figure 2-7 Supersonic inlet starting mechanism (figure taken from Heiser and Pratt\textsuperscript{21})
Wie\textsuperscript{18}, it is defined as the inlet operation under conditions where flow phenomena in the internal portion of the inlet do not alter the capture characteristics of the inlet. Unstart flow is identified when the shock system inside the inlet is expelled outside, thus producing high spillage and total pressure loss. The inlet would also be subjected to high aerodynamic and thermal load.

Figure 2-7 above depicts a simple representative of a pitot inlet starting process as it accelerates to its operating condition. In State 1, the freestream is subsonic and the flow is not established inside the inlet. As it accelerates to a higher subsonic speed, flow at minimum area throat would reach sonic speed such as in State 2. This condition is called choked flow. In State 3, as the freestream is supersonic, a normal shock formed just upstream of the inlet interface to warn the incoming flow of the flow restriction formed by the choked throat. This inlet is said to be operating subcritically and has very large mass flow and total pressure loss. As the inlet continue to accelerate, the normal shock moves downstream and “swallowed” by the inlet. The flow is said to be established as no spillage occurs and shock are contained inside the inlet. This is represented as State 4 and the inlet is said to be operating supercritically or started. For scramjet inlet, it is indeed the desired condition to have supersonic flow into the combustor, but for ramjet inlet, a subsonic flow is required. Thus, for the inlet above, after the inlet has started, variable geometry method could be used to decrease captured area, so that the flow is sonic at throat (choked condition) and subsonic beyond, such as depicted in State 5.
2.3.2 Contraction Limit

Figure 2-7 above demonstrates that after the inlet has started, variable geometry could be used to increase contraction ratio up to certain value without unstarting the inlet. Here, contraction ratio is defined as the ratio of maximum internal area (at cowl lip plane) to minimum internal area (at throat). Typically, increasing the contraction ratio will result in overall increase of compression ratio and lower total pressure loss\(^{40,41}\). There is, however, limit on contraction ratio for inlet, at which beyond the limit, the inlet will unstart.

For pitot-inlet such as in Figure 2-7 above, its contraction limit in the function of freestream Mach number was calculated by Kantrowitz and Donaldson\(^ {42}\) and known as the Kantrowitz limit. It was calculated by assuming a normal shock at the start of internal contraction.

Equation for Kantrowitz Limit given as:

\[
\frac{A_1}{A_t} = \frac{1}{M_1} \left[ \frac{(\gamma + 1)M_1^2}{(\gamma - 1)M_1^2 + 2} \right]^{\frac{\gamma}{\gamma - 1}} \left[ \frac{\gamma + 1}{2\gamma M_1^2 - (\gamma - 1)} \right]^{\frac{1}{\gamma - 1}} \left[ \frac{1 + \gamma - 1/2M_1^2}{\gamma + 1/2} \right]^{\frac{\gamma + 1}{2(\gamma - 1)}}
\]

The subscript in the equation above follows the labelling convention of pitot-inlet shown in Figure 2-7.

The equation describes that for pitot inlet operating at \(M_1\), the contraction ratio \(\frac{A_1}{A_t}\) must not be higher than the limit given by \(\frac{A_1}{A_t} \) to ensure inlet self-start. It is possible to design an inlet which has contraction ratio higher than Kantrowitz limit and start the inlet by overspeeding or variable geometry inlet. In the case of overspeeding, the inlet can be started by first accelerating it to achieve higher freestream Mach number, thus increasing apparent contraction ratio above Kantrowitz limit before slowing down back to desired
operating Mach number. A variable geometry inlet could avoid overspeeding by temporarily reducing contraction ratio to start the engine. Contraction ratio could be reduced by reducing inlet area or by increasing throat area. The underlying idea is to reduce mass capture so that “blockage” inside the inlet is relieved. That is also the principle behind perforations (or bleeds), where flow spillage is deliberately allowed to ensure inlet starting.\(^{43}\)

Impulsive starting method occurs when the inlet is fully closed before being opened fully at design condition. The impulsive force of sudden inlet opening will force start the inlet.\(^{44,45}\)

### 2.3.3 Inlet Unstart Behaviour

According to Heiser and Pratt\(^{21}\), once the inlet has started, the inlet could be unstarted through three types of disturbances. First, the inlet would unstart if the aircraft reduce its flight Mach number significantly. Second, increase of backpressure will push shock system upstream and out of the inlet. This backpressure can be increased by amplifying the combustion rate or by decreasing the nozzle area. Third, unwanted distortion introduced into the flow by unexpected angle of attack, boundary layer separation, or ingestion of unusual gas could present the flow with “blockage” effect and lead to unstart.

One of the most comprehensive study outlining the mechanisms of hypersonic inlet unstart was presented by Tan et al.\(^{46,47}\). The authors argued that in the past, inlet unstart phenomena has been traditionally studied by assuming steady phenomena, hiding its true oscillatory nature. In their study, they designed an inlet for operation in Mach 6 and quasi-steadily changed its throttling ratio (TR) by means of conical plug to induce unstart. The conical plug was moved gradually to block the isolator exit until the inlet-isolator was unstarted. It was found that the shock system inside the diffuser section moved upstream.
with increase of TR. At TR = 89%, the inlet was found to be unstarted with surface static pressure fluctuation in the margin of 176 kPa. The fluctuation was caused by rapid disgorging and swallowing of shock system, and its value demonstrated extreme loading on inlet surface during unstart.

The authors then explained the unsteady unstart mechanism in terms of three different stages. The first stage is the Mass Filling Up stage. This is the longest stage of the three and at this stage, the flow is still quite stable, with shocks from compression ramp can be observed impinging on the cowl lip. At certain throttling ratio, the rate of mass captured would overtake the rate of mass going out through isolator exit. Hence, mass would accumulate around the conical plug, increasing the backpressure and pushes the shock system slowly and continually.

The second stage is called the Shock System Disgorging and Swallowing stage. In this stage, as the name implies, the shock is disgorged before being swallowed back again rapidly. When the shock is disgorged to become detached bow shock, the Mach number inside the inlet would decrease and the flow would reverse it directions. Large detached shock at cowl lip could prevent the inlet to capture more air while allowing the captured air to exit easily. This will cause rate of mass exiting the inlet to be higher than rate of mass captured into the inlet. Flow spillage would allow for drop in internal duct pressure and with overall pressure drop, shock is swallowed again. In third stage, started flow pattern near throat would re-establish and the cycle will be repeated.

It was found that as the TR increased quasi-steadily beyond the start limit of the inlet, the cycle repeats but with first and third stages shortened.
2.4 Scramjet Inlet-isolator Off-design Conditions

There are many different inlet off-design scenarios. The first attempt to classify it is done by Bachchan and Hillier\textsuperscript{48}. They classify the scenarios into five distinct cases that are differentiated according to different combinations of external compression shock interactions. Basically, off-design occurs when the inlet is not operating the way it is designed to be. It could be caused by inlet operation at off-design Mach number, off-design angle-of-attack (AoA), off-design yaw angle, off-design gas temperature and off-design gas properties. Various factors causing off-design condition will be discussed in the following sections.

2.4.1 Off-design Caused by Mach Number Change

When the inlet operated at higher Mach number, compression ramp shocks would impinge inside the cowl surface and interacts with shock system from cowl inducing higher drag\textsuperscript{49}. For operation at lower Mach number, the compression shock would miss the cowl and produce high mass spillage and high drag. It is very important to address the issue of off-design Mach number because the aircraft needs to accelerate to cruise condition and also decelerate as it lands.

Bachchan and Hillier\textsuperscript{48} subjected an inlet which has been designed for Mach 5.5 to different flow Mach number to study its flow features at off-design conditions. The different shock structures formed at Mach number 4, 5.5 and 6 are shown in the figure below:
Figure 2-8 Scramjet inlet designed for Mach 5.5 operating in Mach number (a) $M = 4$, (b) $M = 5.5$ and (c) $M = 6$ (figure taken from Bachchan and Hillier\cite{48})

Bachchan and Hillier\cite{48} observed that at lower than the designed Mach number, the shoulder separation bubble propagated upstream onto the compression ramp (see Figure 2-8 (a)). The strong separation shock then impinged onto the cowl surface and induced cowl separation, which are later detached from the cowl lip. Detached cowl shock such as this has been shown to induce high stagnation pressure loss and drag\cite{49}.

Figure 2-8 (b) shows the inlet at Mach 5.5 for which it is designed for. The compression fan impinged exactly at cowl tip before being deflected as cowl shock and impinged downstream of expansion shoulder. The cowl shock interacted with separation shock and induced flow non-uniformity downstream. Even though this inlet was designed for Mach 5.5, but it can only satisfy shock-on-lip condition and not shock-on-shoulder condition. The impingement of cowl shock on isolator surface has induced boundary layer separation around shoulder region.

Bachchan and Hillier\cite{48} noted that the boundary layer thickness on the compression ramp increase with Mach number. This partly explains the reason shoulder separation in higher than design Mach number (Figure 2-8 (c)) to be bigger than at design (Figure 2-8 (b)). Thus,
shoulder separation shock became strong enough to induce cowl boundary layer separation such as shown in Figure 2-8 (c).

2.4.2 Off-design Caused by Angle-of-Attack (AoA) Change

For high manoeuvrability aircraft, inlet designer must also consider the effect of high AoA. For basic understanding of inlet operating in high AoA, much can be learned from Boon and Hillier\textsuperscript{50}. In their study, a Mach 5 inlet was subjected to Mach 6 flow with 5° AoA.

Large separation bubble was observed to occur at expansion shoulder in the windward side. The separation bubble induced an extra shock that impinged on the cowl inner surface. The separation shock interacted with cowl impingement shock, thus resulting in high total pressure loss. The leeward side had more complex and unsteady flow structures. The compression shocks impinged on cowl inner surface and interacted with cowl expansion fan. This unstable interaction formed a detached bow shock wave, which interacted with boundary layer on the compression ramp. The flow at leeward side was then reversed by the large separation on the compression ramp, and the inlet finally unstarted. The shock structures for windward and leeward side are given in Figure 2-9.
Besides flight Mach number and angle-of-attack, change in gas temperature beyond the capability of the inlet could also result in a lot of off-design issues. For example, it has been shown by McRae and Neaves\textsuperscript{51} that 10\% increase in freestream temperature is enough to unstart an inlet. In another study, Mayer and Paynter\textsuperscript{52} investigated the operation of inlet designed for Mach 2.35 flight, and it has been shown that 3.5\% increase in freestream temperature is enough to unstart an inlet.

With increase in temperature, boundary layer becomes thicker, thus making shock boundary layer interaction effects more pronounced\textsuperscript{53,54}. In Chang \textit{et al.}\textsuperscript{53}, an inlet was tested with and without backpressure to study the effect of wall cooling. In the cases without backpressure, it has been shown that with wall cooling, significant gains were observed in mass capture ratio, total pressure ratio and maximum sustainable backpressure ratio. Flow uniformity improved as the shock boundary layer weakened and separation onset point was delayed.
downstream. Only static pressure ratio was observed to reduce in small value. All observed findings could be related to the reduction in boundary layer thickness in wall-cooled case.

From Lin et al.\textsuperscript{54}, it can be shown that with increase in flow temperature, isolator needs to be made longer to fully contain the shock train. In extreme temperature increase, the flow could be choked thermally, which lead to inlet unstart. With heat addition, the study found that boundary layer became thicker and Mach number decreased.

Krause et al.\textsuperscript{55} demonstrated that at certain flow temperature, the separation bubble at throat could become so big that flow would choke leading to unstart.

\textbf{2.4.4 Off-design Caused by Gas Properties Change}

In the paper by Holland and Perkins\textsuperscript{20}, a three-dimensional inlet was subjected to operation in perfect gas and tetrafluoromethane CF\textsubscript{4}. Specific heat ratio of perfect gas was assumed 1.4, while CF\textsubscript{4} has $\gamma = 1.22$. It was found that compression shock was weaker in lower specific heat ratio gas. The inlet has been shown to self-start in CF\textsubscript{4} but would unstart in a perfect gas. The authors then concluded that if an inlet could self-start in a type of gas, then using another gas with lower specific heat ratio would not affect the startability of the inlet.

\textbf{2.5 The Needs for Variable Geometry Inlet at Off-design}

Scramjet inlet need robust mechanism to address all the problems at off-design conditions discussed in the previous section, as a hypersonic aircraft would normally experience a large flight envelope. For example, an accelerators/cruisers type aircraft, such as orbital hypersonic plane, reconnaissance aircraft and strategic missiles, typically climb through large
altitude before cruises at constant hypersonic speeds. In this case, the inlet must be able to adapt to rapid changes in atmospheric air conditions. For access to space vehicle that is a part of multi-stage to orbit systems, acceleration to high Mach number is more important than altitude climb. In this case, the inlet must be able to cope with changes in air flow speed.

Obviously operating at such large flight envelopes would require the engine to adapt suitably for maximum performance. The problem is that a scramjet inlet is usually designed and optimized for operation at single flight condition. The problems are addressed by designing a variable geometry inlet which could be adapted to various off-design conditions. For example, NASA has designed axisymmetric inlet with translating centrebody for Trailblazer single stage-to-orbit hypersonic plane. It is a Rocket Based Combined Cycle (RBCC) engine (see Figure 2-10) with dual mode of subsonic and supersonic combustion. For the airbreathing mode of operation, the translating cowl will adjust according to flight Mach number to achieve maximum performance.56
NASA also studied the possibility of using Turbine Base Combined Cycle (TBCC) engine for two-stage-to-orbit propulsion system (see Figure 2-11). The first stage would have both turbine and ramjet/scramjet engine, which can carry the rocket powered second-stage vehicle for access to space. Variable geometry inlet has been developed for the first-stage vehicle to allow the engine to adapt to different Mach number. For take-off and low Mach number operation, a splitter is used to allow maximum mass flow into the turbine. As the Mach number increase, splitter would split mass flow for mixed turbojet and ramjet operation. For hypersonic speed, the turbojet would be fully closed by the splitter cowl to allow full scramjet operation\textsuperscript{57}. Henry and Anderson\textsuperscript{58} did an analytical study on an engine concept similar to Figure 2-11 and found that with variable geometric inlet, specific impulse of the engine increased to a maximum of 16% compared to fixed inlet.
MBDA-France also developed a few variable geometry inlets for dual mode ram-scramjet applications. One of the early designs is the PREPHA\textsuperscript{4} engine, which uses a rotating cowl (see Figure 2-12). The cowl would deflect or rotate inwards to maintain maximum performance at various Mach numbers. Another design by MBDA-France is the PROMETHEE\textsuperscript{4} engine (see Figure 2-13), where the inlet could alter both diffuser and combustion area size at off-design Mach number. MBDA-France also cooperated with Moscow Aviation Institute to investigate a translating cowl variable geometry scramjet engine. Similar to PROMETHEE engine, this engine, called PIAF\textsuperscript{4} (see Figure 2-14), would also have overall component area change at different Mach number.

Pratt and Whitney, a major propulsion device supplier, designed a scramjet inlet for Mach 7 flow with variable flap cowl that can maximize mass flow and aerodynamic contraction (see Figure 2-15). The temperature seals that connect the fixed and movable components have been shown to operate flawlessly\textsuperscript{13}. Clearly, variable geometry inlet is the preferred method of controlling flow to ensure engine startability and to achieve maximum scramjet engine performance.
Figure 2-12 PREPHA engine studied by MBDA France (figure taken from Falempin⁴)

Figure 2-13 PROMETHEE engine studied by MBDA France (figure taken from Falempin⁴)

Figure 2-14 PIAF engine studied by MBDA and Moscow Aviation Institute (figure taken from Falempin⁴)
2.6 Study on the use of Variable Geometry to Increase Inlet Performance

From the background study, we could conclude that inlet performance depends on various flow phenomena that are caused by off-design flight condition. Various phenomena such as mass spillage, shock boundary-layer interactions, separation bubble, shock-shock interactions, shock-expansion wave interactions, and Mach reflection have been shown to reduce performance, increase distortion and induce flow instabilities. Inlet variable geometry method has been developed to mitigate such problems.

Wie and Ault\textsuperscript{31} discussed the different flow phenomena as cowl height is changed and its relation to performance. An inlet was designed for Mach 20 operation condition but subjected to Mach 10.4 condition, with total pressure and total temp of 4600 psia and 3300 R\textsuperscript{0} respectively in shock tunnel. CFD study was performed using Parabolized Navier Stokes equation to complement the experimental work and use as a basis for performance study. Cowl position was varied vertically between 6.43 to 5.69 inches in height. With ramp and
isolator section geometry fixed, changing the height will change the contraction ratio. With decrease in cowl height the shock wave was found to move upstream (see Figure 2-16). It can be seen on the figure that with cowl height $Y_c = 6.04$ inches, the shock-on-shoulder condition was almost satisfied, and the flow was relatively uniform compared to others. Kinetic energy efficiency of the inlet is at the highest if the cowl position allows the cowl shock to impinge close to shoulder region (see Figure 2-17).

![Internal shock diagram](image)

*Figure 2-16 Movement of internal shock system with change in cowl height (figure taken from Wie and Ault)*

Page 74 of 270
Influence of cowl height on flow uniformity was also investigated. It has been observed that the flow became more uniform with decrease in cowl height. Distortion index reached a minimum at around $Y_c = 6.04$ inches, and this was identified as the shock-on-shoulder condition. When the condition was satisfied, there would not be any shock-expansion wave interaction around shoulder region and the flow should be uniform. The inlet flow became more distorted as the cowl shock impinged on the compression ramp with further reduction in cowl height.

Das and Prasad\textsuperscript{59} analysed a generic supersonic inlet with the aim of comparing the performance gain from inwards cowl deflection and those from using boundary layer bleed. Both of the methods have been demonstrated to improve the inlet performance by “softening” the impact of boundary layer separation. In their numerical study, they set the cowl to deflect inward between $1^\circ$ and $5^\circ$. In all cases, it was found that cowl deflection improves the pressure recovery of the inlet compared to no deflection case. The separation
bubble moved downstream with deflection angle and reduced the blockage at throat. The shoulder separation bubble was also observed to diminish in size.

One of the most comprehensive studies of variable geometry scramjet inlet was done by Emami et al.\textsuperscript{33}. An inlet with isolator was designed and tested in supersonic wind tunnel at Mach 4. They were interested in investigating the relationship between isolator flow structures and its performance. Four parameters of the inlet geometry were varied and they were cowl angle, cowl length, isolator length and boundary layer thickness. Cowl angle was varied to control throat Mach number, captured mass ratio and inlet contraction ratio. Cowl length was varied to vary the distortion level, as most of its compressions were done internally. Variations of both cowl angle and length were combined to study the effect of shock strength and internal compression shock numbers onto the flow. Boundary layer thickness was varied by attaching a flat plate before the ramp so that a fuller boundary layer was developed upstream of the compression ramp. With combination of all variables, there were 250 geometric considerations. Backpressure produced from combustion was simulated using variable area throttling device at the back of isolator.

It was observed that contraction ratio increase would result in total pressure ratio and throat Mach number decrement. The influence of contraction ratio to total pressure ratio and Mach number was reduced with thick boundary layer. They also confirmed the findings of Wie and Ault\textsuperscript{31}, which states that with shock-on-shoulder condition, the distortion would be minimized but not entirely eliminated. Shock and expansion wave interaction at shoulder increased the distortion level and reduced the maximum backpressure the inlet could withstand. It was also found that maximum backpressure the inlet could withstand increased
with isolator length and contraction ratio. Thin boundary layer was found to be beneficial to start the inlet.

A study on the relationship between isolator lengths to inlet performance was shown by Reinartz et al.\textsuperscript{60}. It was found that increasing the isolator length will increase the maximum backpressure that the inlet could sustain. In their study, there was a maximum isolator length, where any increment beyond that value would not help to increase the maximum backpressure that the inlet could sustain.

Another method of varying inlet geometry for performance improvement is by varying the ramp shape instead of moving the cowl segment. The effect of ramp shoulder shape was studied by Krause et al.\textsuperscript{55}. Their numerical code analysed a double ramp inlet combined with gradual convex curve in one case and sharp convex corner in another. Expansion fan at shoulder was more subtle if the shoulder had a convex curve shape, but became more abrupt with sharp convex corner. In their experiments, the strength of shoulder expansion wave influenced the shoulder separation directly. This was related to the fact that expansion wave would increase the strength of oblique shock that interacted with it; thus bigger separation bubble was induced.

2.7 Study on the use of Variable Geometry to Start an Inlet

Mahapatra and Jagadeesh\textsuperscript{61–63} have published a few papers recently discussing the mechanism behind the unstart phenomena caused by high contraction ratio and how variable geometry methods can help to mitigate the situation. In all studies, they utilised a single ramp inlet designed for Mach 8 flow. In Mahapatra and Jagadeesh\textsuperscript{63}, they studied the
effect of changing contraction ratio (CR) by changing the height of the inlet. With CR = 8.4, the shock was shown to impinged on the cowl lip before the inlet eventually unstarted. Large separation bubble appeared on the compression ramp but was not considered to be the main cause of unstart because the contraction ratio was too high to start with. As cowl height was increased, the ramp shock impinged on cowl inner surface, where it interacted with cowl shock and produced a Mach reflection at CR = 5 and an Edney’s Type II\textsuperscript{64} shock interaction at CR = 4.3 (see Figure 2-18). Both flow phenomena were found to be unsteady with Mach stem in CR = 5 as the experiment progressed with time and the Edney’s Type II can be seen moving about a fixed point. This proved that employing cowl height increment to lower CR for inlet starting mechanism would not be very efficient as it could induce flow instabilities.

Figure 2-18 Edney’s Type II\textsuperscript{64} shock-shock interaction forming at throat entrance of the inlet studied by Mahapatra and Jagadeesh\textsuperscript{63}. In this figure, C1 and C2 are interacting oblique shocks, C3 and C4 are resultant oblique shocks, C5 is Mach stem, and T1 and T2 are triple points.

In Mahapatra and Jagadeesh\textsuperscript{62}, they expanded their studies further by changing the cowl deflection angle and length. For inlet with increased length, none was able to start and many flow interactions such as Edney’s Type I\textsuperscript{64} and Type II\textsuperscript{64} occurred around the cowl tip. This
was due to the fact that increasing the cowl length will increase the contraction ratio beyond the Kantrowitz limit. For cowl deflection angle case, flow was established and inlet was able to start. Cowl inward rotation reduced the contraction ratio and made the cowl shock weaker. As can be noted from their studies\textsuperscript{61-63}, it seems like the best among the three methods of suppressing hard unstart is by using cowl deflection.

Donde \textit{et al.}\textsuperscript{65,66} numerically investigated the minimum cowl deflection angle needed to avoid inlet unstart. A single ramp of two-dimensional inlet was designed to operate at Mach 6.2. It was shown that at the designed conditions, there were equal possibilities of inlet start or unstart depending on how the aircraft arrive to that point. To illustrate, in one case, it was shown that as the boundary condition was initialized at zero to simulate acceleration to design velocity, large spillage occurred from strong shock on ramp, resulting in low pressure recovery and high pressure drag. In another case, if the boundary layer was initialized at the design point (which indicates an impulse starting), then the shock system satisfied both shock-on-lip and shock-on-shoulder conditions. Then, dynamic mesh was used to allow cowl deflection/rotation during numerical simulation to help start the engine.

Another question that demands answer is whether rate of inward cowl rotation plays an important role in starting characteristics of the inlet. In the paper by Throckmorton \textit{et al.}\textsuperscript{67}, investigation was made to answer the question. The study concentrated on a single-ramp inlet model with a rotating pneumatically actuated door placed on the ramp to shut or partially open the inlet. The rate of door rotation was measured. With partially closing the inlet before opening it, the inlet achieved 89\% greater pressure ratio, which proved that inlet has started. It was found that opening rate has no effect.
Another mechanism of unstart that can be solved by variable geometry is soft unstart phenomena, which is caused not by too much air captured, but rather by separation bubble so large that air cannot passed through. A separation bubble at throat would cause the thickening of or separates the boundary layer on ramp and isolator. This would cause temporary increase of contraction ratio that can lead to shock expulsion out of the duct. Inevitably, this will cause inlet unstart. In the paper by Das and Prasad\textsuperscript{68}, it has been shown that using cowl deflection has a positive role in mitigating shoulder separation induced unstart. The authors then expanded the study\textsuperscript{69} and found that cowl deflection in combination with bleed was better than using cowl deflection or bleed alone.

In the article by Lanson and Stollery\textsuperscript{70}, a two-dimensional air intake has been designed with wedge of $11^\circ$, followed by isentropic compression surface with total flow turning of $30^\circ$. It has been subjected to Mach 8.2 in Gun Tunnel in Cranfield University. The authors wanted to study the relations between separation bubble sizes with cowl height. They observed that separation in throat vicinity decreased in size as the cowl height was extended.

### 2.8 Key Facts and Conclusions from Literature Survey

Scramjet engine is the key component for military hypersonic missions and/or affordable access to space. Since scramjet does not carry its oxidizer and no mechanical compressor available on board to control high compression efficiency, scramjet performance is very dependant on its inlet performance.
Decrease in inlet performance or even worse, inlet unstart, could occur at off-design conditions. There are various causes of off-design conditions and all of them must be considered for good inlet design.

Traditional method of mitigating the issues is by using variable geometry inlet. Even though the idea of variable geometry inlet has been around since the early days of scramjet technological development, there are still considerable gap in understanding various flow phenomena involved, especially in the isolator part where high-quality optical flow-diagnostics could not always be provided. Novel flow-diagnostics methods that measure globally would add further understanding in the relations between inlet flow quality and its performance. Computational fluid dynamics code could also benefit and improve by having solid validation data from global measurement of inlet flow properties.
CHAPTER 3 METHODOLOGIES

This chapter will discuss, in the order listed below, three main topics:

- **High Supersonic Tunnel (HSST):** This tunnel is the experimental facility available where all experimental works are done. Discussions will include the characteristics and performance of the tunnel along with its calibration process.
- **Design of Scramjet Inlet-isolator:** The philosophies behind the design process of the scramjet inlet-isolator model will be explored.
- **Experimental Techniques and Flow Diagnostics:** All experimental techniques involved will be explained and their applicability with regards of the current test environment will be discussed.

### 3.1 University of Manchester’s High Supersonic Tunnel (HSST)

![HSST Diagram](image)

Figure 3-1 High supersonic tunnel in University of Manchester

High supersonic tunnel in University of Manchester is one of the few available tunnels in UK that is capable of producing hypersonic flow with ample testing duration. Details of HSST are described at length by Erdem\(^7\) as the first researcher who worked on this facility in Manchester. The blowdown supersonic tunnel, with its schematic given in Figure 3-1, depends on the pressure and vacuum tank for its operation. Compressed high pressure air is supplied through the adsorption dryer unit (Ecoair Series AT 15) and stored at 16 Bar in the pressure tank. The total pressure for tunnel operation is controlled by dome valve Type C4 Model 208/3. The firing valve unit is a pneumatically actuated quick-acting valve of model...
Worcester Type 45-4466-TT. The firing valve is connected to a 24 kW direct current electric resistive heater, which regulates the desired flow total temperature. The current for the heater is supplied by a remote controlled Lincoln Electric Type SAE 600 welding motor generator. Insulation is important for reducing heat loss from the heater. Hence, the heater is housed in a container packed with vermiculite granules. The temperature is raised to between the minimum temperature just above liquefaction point and 700 K for maximum enthalpy flow. The settling chamber downstream of heater is connected to a contoured axisymmetric nozzle, which is interchangeable for setting up Mach number of 4, 5 and 6. The nozzle exit is enclosed in a test section with dimension of 325 mm square cross section and length of 900 mm. A pair of 195 mm diameter quartz windows is located on either side of the test section to allow for flow diagnostics. The top part of the test section has 120 mm diameter quartz window for optical access and can be fitted with Germanium window for infrared thermography. A diffuser entrance of 250 mm diameter is also enclosed inside the test section. The diffuser ensures the startability of the wind tunnel by swallowing the starting shock. It could be moved upstream or downstream in the range of 140 – 400 mm distance to nozzle exit plane in order to accommodate various model size and geometry. A slide valve downstream of the diffuser allows for isolation of the vacuum tank from test section in order to avoid pressuring the tank during test section opening. Vacuum condition with pressure of below 1.5 mbar in the tank is maintained by a set of vacuum pump. The set consists of Edwards EH2600 Roots mechanical booster pump, General Engineering Kinney Size GKD220 and Edwards Model 412J rotary piston vacuum pump.
Figure 3-2 Variation of unit Reynolds number envelope in relations to Mach number, stagnation pressure and temperature (figure taken from Erdem\textsuperscript{71})

Reynolds number of the flow can be set by fixing stagnation temperature and pressure. The dependence unit Reynolds number with other independent factors are given in Figure 3-2. In the plot, the minimum and maximum total temperature of 375 K and 700 K respectively are considered with stagnation pressure of 5, 6, 7 and 8 Bar. The plot is valid only in the Mach number range of 3 – 7. Solid vertical lines crossing the blue and red lines depict the range of unit Reynolds number at three possible Mach number. It seems that for a given Mach number and stagnation temperature, increasing the total pressure will increase the unit Reynolds number. On the other hand, increasing total temperature will lower unit Reynolds number. At Mach 5, where all the current experiments were conducted, unit Reynolds number could be set between $4 \times 10^6$ m$^{-1}$ and $16 \times 10^6$ m$^{-1}$. Wind tunnel calibration details are included in the appendix.
3.2 Generic Scramjet Inlet-isolator Model Design

3.2.1 Baseline Design

3.2.1.1 Selection of Scramjet Inlet Type

The function of a scramjet inlet is to compress high speed air efficiently with highly uniform flow towards the combustor without incurring high drag and weight\(^\text{18}\). There are many scramjet inlet proposed and being used in various projects, where one could have quite a hard time determining the most suited for one’s flight condition. Among them are two-dimensional planar inlet, Oswatitsch inlet, Busemann inlet, sidewall-compression inlet and modular Busemann inlet\(^\text{18}\). The simplest among them is the two-dimensional planar design, which uses discreet oblique shock waves or continuous Mach wave compression regions to compress air.

As the focus of this thesis is the development of scramjet inlet characterization methods, it would be best if the design of the inlet is kept as simple as possible. Using a three-dimensional inlet would complicate the flow structures inside the inlet and obscure the focus and objective of the investigation. On the other hand, using an axisymmetric inlet would make it quite challenging for optical flow diagnostics. Hence, a two-dimensional planar design was selected. The flow on two-dimensional planar inlet could form a basis to more complex flow on axisymmetric and three-dimensional inlet.

This thesis also chose a two-dimensional planar inlet design as it will be easier to design and manufacture. Using Oswatisch, Busemann or Modular Busemann inlet will incorporate a smooth contoured surface for isentropic compression. Small surface defects or manufacturing irregularities on the contoured surface could severely affect performance.
Therefore, with all the reasons stated above, two-dimensional planar inlet design seems the best design to choose.

Having established the reasons for selecting two-dimensional planar inlet, decision must be made on the number of oblique shocks the design should have. Mahapatra and Jagadeesh\textsuperscript{61–63} in their study of variable geometry inlet utilised two oblique shocks (one external and one internal shock) inlet simply for convenience in manufacturing. With one external oblique shock, the shock and its reflection or interactions would be very strong and could be visualized easily during experiment. Emami \textit{et al.}\textsuperscript{33} also used single-ramp compression surface but they combined the ramp with a flat plate at upstream to simulate thick incoming boundary layer. Donde \textit{et al.}\textsuperscript{66} also depended on a single ramp in their numerical investigation of inlet unstart and restart characteristics. While it may be beneficial to use single ramp as it would make design process easier and faster, this thesis aims to simulate a working practical inlet which is used in real flight application. Single-ramp compression inlet would incur too large pressure loss and generate excessive heat to be used in real flight hypersonic condition that it is rarely used. This is partly the reason why experimental group in DLR Germany employed double-ramp for their scramjet intake studies\textsuperscript{39,72–74}. There are also a few other experimental test campaigns that applied three compression ramp surfaces for their scramjet inlet\textsuperscript{75–78}. However, this thesis considers that with extra shock introduced into the compression systems, the unwanted shock-shock interactions during off-design test could be too complex. Also, Heiser and Pratt\textsuperscript{21} stated that the adiabatic compression efficiency of two oblique shocks (i.e. single-ramp inlet) could only achieve the maximum of 0.83, while three oblique shocks (i.e. double-ramp inlet) system could increase the value to
almost 0.9. Hence, the current experimental campaign will focus only on two-dimensional planar inlet with double compression ramp surfaces.

### 3.2.1.2 Iterative Design Process

![Schematic of two-dimensional planar with double compression ramp where shock-on-lip and shock-on-shoulder conditions are satisfied.](image)

A perfect operating scenario for a planar inlet is when the compression shocks impinge exactly at the cowl tip and then reflects as a single shock on the shoulder such as shown in Figure 3-3 above. With shock-on-lip, no spillage will occur and satisfying shock-on-shoulder will ensure uniform flow inside the isolator. The schematic in Figure 3-3 could be divided into three separate regions bounded by three discreet shock waves and can be described by a set of equations such as below:

**Equation 3-1**

\[ (M_1, S_1) = f(M_0, d_1) \]

**Equation 3-2**

\[ (M_2, S_2) = f(M_1, d_2) \]
Equation 3-3

\[(M_3, S_3) = f(M_2, d_3)\]

Equation 3-4

\[d_3 = d_1 + d_2\]

Equation 3-5

\[\frac{h_1}{x_1} = \tan(d_1)\]

Equation 3-6

\[\frac{h_2}{x_1 + x_2} = \tan S_1\]

Equation 3-7

\[\frac{h_2 - h_1}{x_2} = \tan(S_2 + d_1)\]

Equation 3-8

\[\frac{(h_2 - h_3)}{x_3} = \tan(S_3 - d_3)\]

The \( f \) function in the first three equations denotes the simplified notation of oblique shock relations. From the set of equations, only value of \( x_1 \), \( x_2 \) and \( x_3 \) are known. They are set to follow from previous scramjet inlet designed\(^7\) for use in HSST in order to avoid tunnel unstart if the model size is too big. Another important design criterion states that for maximum performance, the ramp shocks must be of equal strength so that pressure loss can be minimized\(^2\). This criterion adds another set of equations:

Equation 3-9

\[\frac{P_{t_2}}{P_{t_1}} = f(M_1, S_2)\]
Equation 3-10
\[
\frac{P_{t_1}}{P_{t_0}} = f(M_0, S_1)
\]

Equation 3-11
\[
\frac{P_{t_2}}{P_{t_1}} = \frac{P_{t_1}}{P_{t_0}}
\]

Also, it would be beneficial to set \( S_2 \) larger than \( S_1 \) so that the total length could be minimized as the two shocks will meet at smaller distance from the leading edge.

Equation 3-12
\[ S_2 > S_1 \]

All shock angles cannot be larger than \( 62^\circ \) as that is the limit of weak shock. With bigger angle, the shock is considered as strong shock, and subsonic flow is produced downstream of the shock.

Equation 3-13
\[ S_1 < 62^\circ \]

Equation 3-14
\[ S_2 < 62^\circ \]

Equation 3-15
\[ S_3 < 62^\circ \]

For the first ramp, \( S_1 \) cannot be lower than \( 12^\circ \) as that is the Mach angle limit for Mach 5 flow where all experiments will be conducted.

Equation 3-16
\[ S_1 > 12^\circ \]
Good rules of thumb used in real life design practice are that the ratio of $\frac{M_3}{M_0}$ must be below 0.38 to avoid gas disassociation\textsuperscript{21}.

Equation 3-17

$$\frac{M_3}{M_0} \leq 0.38$$

Kantrowitz limit must be satisfied to ensure that the inlet could self-start\textsuperscript{42}. Kantrowitz limit for double ramp inlet type was formulated using algebraic simplification from Veillard et al.\textsuperscript{80}:

Equation 3-18

$$\left(\frac{A_e}{A_i}\right)_k = \left[\frac{\gamma - 1}{\gamma + 1} + \frac{2}{(\gamma + 1)M_2^2}\right]^{\frac{1}{\gamma - 1}} \left[\frac{2\gamma}{\gamma + 1} - \frac{\gamma - 1}{(\gamma + 1)M_2^2}\right]^{\frac{1}{\gamma - 1}} \frac{\sin(S_1 - d_1) \sin(S_2 - d_2)}{\sin S_1 \sin d_1}$$

Equation 3-19

$$\frac{A_e}{A_i} = \frac{h_2 - h_3}{h_2}$$

Iterations were performed by considering all equations above, and it was found that no solution existed. The problem was that to obey the Kantrowitz limit, shock-on-shoulder condition could not be satisfied. Satisfying the shock-on-shoulder condition would require the inlet to have smaller than throat height allowed. Matthews et al.\textsuperscript{26} proposed that to have lower contraction for self-starting, it is better to have the shock impinge downstream of the shoulder to become what is known as re-expansion inlet.

Hence, another iterations were conducted but with Equation 3-8 ignored. The final geometrical dimensions are summarized in table below:
Table 3-1 Geometrical dimension of scramjet inlet after iterative process

<table>
<thead>
<tr>
<th>$d_1$</th>
<th>$d_2$</th>
<th>$d_3$</th>
<th>$x_1$</th>
<th>$x_2$</th>
<th>$x_3$</th>
<th>$h_1$</th>
<th>$h_2$</th>
<th>$h_3$</th>
</tr>
</thead>
<tbody>
<tr>
<td>10°</td>
<td>12°</td>
<td>22°</td>
<td>52 mm</td>
<td>28 mm</td>
<td>2 mm</td>
<td>9.2 mm</td>
<td>28.1 mm</td>
<td>21.3 mm</td>
</tr>
</tbody>
</table>

3.2.1.3 Shock-Expansion Wave Interaction at Cowl

![Figure 3-4 Schematic of cowl shock and shoulder expansion wave interaction typical for re-expansion inlet](image)

Since this inlet was deliberately designed for the cowl shock to impinge downstream of shoulder, the shock would interact with the expansion wave such as given in Figure 3-4. The shock would curve and gain in strength as it interacts with the expansion wave. In inviscid consideration, the reflection of cowl shock as it impinges downstream of the expansion corner could be in the form of Mach reflection or normal reflection. The deciding criterion is the total flow turning imposed by the shock and the initial Mach number upstream of the cowl shock. Mach reflection would induce more total pressure loss due to its normal shock component compared to normal oblique reflection. It is of important
interest to check whether the shock-expansion wave interaction will result in unwanted Mach reflection.

![Interaction between expansion wave and shock wave](image)

**Figure 3-5** Interaction between expansion wave and shock wave, where combination of upstream Mach number and flow turning angle could result in regular reflection, Mach reflection or dual-solution. Here, $M_1$ represents the Mach number of flow upstream of shock wave, and $\delta_1$ is the flow turning angle imposed by the shock wave (figure taken from Hillier\textsuperscript{83}).

The Mach number downstream of second ramp shock $M_2$ was calculated as 3.14. If it is to be turned by $22^\circ$ prior entering the isolator, it would lie inside the dual solution domain of the plot shown by Hillier\textsuperscript{83} (see Figure 3-5). This means that a regular reflection is expected, but any disturbance in the flow could induce a Mach reflection and make it stay even after the elimination of disturbance\textsuperscript{84}.

### 3.2.1.4 Isolator Length

Current experimental campaign considers the coupling effect of inlet and isolator. Many papers have studied the performance of ramjet/scramjet isolator-diffuser extensively, but most of them have been conducted in isolation from the inlet segment\textsuperscript{54,85–95}. By doing this,
distortion effects due to inlet operating at off-design condition are ignored\textsuperscript{33}. Varying inlet geometry could also distort flow into the isolator section and reduce its performance\textsuperscript{33}. A recent paper by Tan et al.\textsuperscript{22} argued that most inlet would have “background waves”, where the cowl shock is not cancelled at shoulder (shock-on-shoulder condition) and interacts with expansion wave before being reflected all the way through the isolator section. The “background waves” will occur at zero isolator back-pressure and will turn into shock train when backpressure is introduced\textsuperscript{22}. Thus, it makes more sense to study the coupled effect of inlet with isolator, especially for this current re-expansion inlet model where the cowl shock is deliberately positioned to impinge downstream of shoulder. The performance of the inlet-isolator will be inferred by considering the properties of flow at isolator exit instead of at the inlet throat (shoulder).

The isolator design was dictated by the need to contain full shock train leading to combustor entrance face. Fully contained shock train is defined as the length of constant area isolator section, where further increase in its value would not lead to rise in pressure\textsuperscript{21}. Shorter isolator will lead to inadequate static pressure rise for combustor, while longer length will lead to viscous loss and weight penalty\textsuperscript{33}. Using the method shown by Heiser and Pratt\textsuperscript{21}, the isolator length was calculated as approximately 53 mm.

It must be noted that up to this point, any boundary layer displacement due to strong viscous effects is not considered. This is to allow for investigation of vortex generator and variable inlet geometry mechanism for viscous flow control. The inlet in its baseline configuration will not achieve shock-on-lip condition in real life viscous flow, but variable geometry inlet will be utilised for performance optimization. Indeed, this is another focus of
this current experimental campaign, which is to characterize the performance of a variable geometry inlet. The final design dimensions are summarized in the sketch below:

From Figure 3-6, the compression corner is positioned at 52 mm (on horizontal axis) from the first ramp leading edge coordinate. The first ramp makes an angle 10° with horizontal axis. The second ramp is bounded between compression ramp and shoulder (throat), and has total angle of 22° and length of 30 mm. The isolator is of constant cross sectional area type and its length from throat to exit is 53 mm. The height of the isolator exit is 6.8 mm. Cowl leading edge is located 55 mm upstream of isolator exit plane.
3.2.2 Three-dimensional Design View

The final CAD model for the design must consider provisions for flow diagnostics. Thus, it was decided to build the model into three main components; main body, cowl and sidewalls. The modular design concept allows for rapid parametric studies. The main body was made from aluminium and would contain twelve pressure tappings along the middle-plane of the model, with three on first ramp, five on second ramp and the rest downstream of shoulder. Three-dimensional effects on pressure can be studied by positioning pair of off-centre pressure tappings, each on first ramp, second ramp and inside the isolator section. There were no pressure tappings on sidewall and cowl piece. Both sidewall and cowl have Perspex windows inside aluminium frame to allow for optical based flow diagnostics. Sidewall was deliberately designed longer than isolator section, with its leading edge more upstream than cowl leading edge. This is to avoid the obstruction of view of flow interactions at throat (shoulder) region. Sidewall window size permits optical access to almost full isolator length except near isolator exit, where 2 mm of sidewall frame blocked the viewing area. Full isolator height would be available for optical access. On the top cowl component, the
Perspex window was limited only in middle span of 17 mm out of 30 mm isolator width. This means that flow data near sidewall-isolator surface intersection was not available for analysis.

![Diagram of top cowl designs](image)

*Figure 3-8 Collections of different interchangeable top cowl designs for parametric studies: (a) Base-cowl (b) Short-cowl (c) Long-cowl (d) 3deg-cowl and (e) 5deg-cowl. All lengths are quoted in mm.*

Provisions for parametric studies of variable inlet geometry have been considered as well.

Five different shapes of top cowl component have been designed and shown in Figure 3-8. Baseline cowl shape is shown in Figure 3-8 (a), with the length of 55 mm with thickness of 5.5 mm. Testing the effect of horizontal cowl position relative to fixed shoulder location can be done using Long-cowl and Short-cowl shown in (b) and (c), respectively. Inward cowl
deflection angle (relative to horizontal) of 3 and 5 degree has been fixed on cowl (d) and (e), respectively. The cowl vertical location can be varied by changing the sidewall component (not shown). There were three sidewall set components with effective isolator height of 6.8 mm, 9 mm and 13 mm. The modular top cowl and sidewall components could be mixed to create many derivatives inlet-isolator shape. To thoroughly study the effect of three-dimensional flow effects at sidewall-isolator surface intersection, a single cowl-sidewall component was made from a solid see through quartz. It will give effective isolator height of 6.8 mm and 55 mm top cowl length from isolator exit. Due to expensive price of the quartz cowl piece, parametric modification involving cowl length, deflection angle and sidewall height were been considered. The sketch of this component is given below:

![Combined cowl-sidewall component made from quartz](image-url)
3.3 Schlieren Flow Visualization

Schlieren is one of the most important flow visualization techniques needed to characterize scramjet inlet-isolator as the associated flowfield is fully compressible, and large density gradient can be detected easily. The available schlieren setup in University of Manchester is of Toepler’s z-type Schlieren system\textsuperscript{96}, and its schematic is given in Figure 3-10 above. For this particular experimental campaign, the setup was modified into “colour-schlieren” setup, with colour filter replacing the typical knife edge in the cut-off plane in normal schlieren. There are strong reasons why colour schlieren technique was adapted. Settles\textsuperscript{97} did an extensive review of colour schlieren technique and found that they have been used primarily in high speed flow regime. He reported that colour schlieren system is useful for distinguishing complex flow features, where several phenomena may occur at the same time. This is particularly valuable advantage of colour schlieren method where strong SWBLI, separated flow and shock-shock interactions can be explicitly differentiated from each other\textsuperscript{97}. Howes\textsuperscript{98} commented that colour schlieren provides more detail non-uniformities in flow. This is very much applicable to current scramjet inlet-isolator case, where re-expansion

\begin{figure}[h]
  \centering
  \includegraphics[width=\textwidth]{schlieren_setup.png}
  \caption{Schlieren setup used in this experiment}
  \label{fig:schlieren_setup}
\end{figure}
at shoulder would induce large flow non-uniformities at isolator exit. Colour schlieren system, with the right setup, could also be used as a non-intrusive quantitative method and not just for visualizing flow\textsuperscript{99,100}. Thus, for this current investigation, colour-schlieren system was adopted. However, quantitative aspect of colour schlieren system being used was not considered.

The light source for the setup shown in Figure 3-10 consists of Oriel Research Arc Lamp system model 66924 with Xenon bulb lamp of 450 W. Its high power collimated output ensures good quality images with high colour intensity. In this current setup, the light passed through pinhole before being reflected by 8 inches parabolic mirror that produced parallel light shining to test section. Density gradients contained in the flowfield distorted the light direction, and this information was reflected towards light sensor by another parabolic mirror with similar size. The focal length of both mirrors was similar at 6 ft. Offset angle of $5^\circ$ was fixed onto the mirrors relative to their axis to eliminate astigmatism and coma\textsuperscript{71}. The density gradient information contained in the reflected light was selectively filtered by tri-coloured wheel before being focused and shone directly onto CMOS sensor of a SLR camera. The camera model used was Canon DSLR model EOS-450D with 12 MP resolutions. Images were recorded at 3.5 frames per second with full resolution. Some quantitative data from schlieren images, such as oblique shock angle and boundary layer thickness, could be obtained using image processing software ImageJ.
3.4 Basic Pressure and Temperature Measurement

3.4.1 Facilities

Flow quality can be diagnosed by using conventional discreet pressure and temperature measurement. Stagnation pressure was measured by a pitot device connected to a Kulite XTE-190M with 0 – 100 psi range housed inside the settling chamber. The settling chamber also housed a K-type thermocouple probe with 1.0 mm wire diameter that measures stagnation temperature. To measure model static pressure distribution, six Kulite XTE-190M with range of 0 – 3.5 bar were housed together and connected to 1.0 mm diameter pressure tappings drilled on scramjet-inlet model via heat resistance flexible tubing.

Data acquisition card of model National Instruments PCI-6251 digitised and collected the data from pressure and temperature probes. Conditioning unit model SCXI-1520 and SCXI-1112 were used to condition the signal from pressure and temperature measurements, respectively. The system was capable of collecting high frequency measurement data at 333 kHz with 16 bit digitisation. LABVIEW v 8.5 was used as the main data processing unit.

Kulite pressure measurements were given in volt, which were calibrated linearly with readings from 0 – 3 bar gauge positioned on top of the test section. The accuracy was typically within ± 6 mbar. However, the accuracy would vary with flow unsteadiness and thus must be calculated individually for every transducer locations at every parametric case.

3.4.2 Experimental Procedures

Since the scramjet-inlet isolator model has 18 pressure tappings in total while there are only six pressure transducers, three run for every unique case must be done to cover all tappings.
For repeatability analysis, pressure at every tapping was recorded twice, which results in a total of six run per unique test case. Average of transducer readings was taken from the beginning to the end of stable plateau such as shown in Figure 0-1, page 245. All static pressure was normalised to the freestream static pressure calculated from total pressure.

3.5 Pressure Sensitive Paint

3.5.1 PSP Technique in Scramjet Inlet Investigation

Currently, there is a gap in applying PSP for scramjet inlet investigation. There have been a few inlet studies that utilise PSP, but their scope are limited to only external flowfield on compression ramp. For example, the paper by Kuriki et al.\textsuperscript{101} developed a novel temperature cancelled PSP system to investigate flow on hypersonic compression corner in Mach 7. They applied different luminophore with low temperature sensitivity onto anodized-aluminium porous surface and obtained good comparison with transducer reading. However, discrepancies between PSP and transducer reading associated with temperature dependence still exist, which shows that their system could not handle the high temperature rise effect in hypersonic inlet application. More recently, Yang et al.\textsuperscript{102–104} applied similar anodize aluminium PSP method to a double compression ramp in Mach 5 and succeeded to get a very good dynamic response similar to Kulite transducer readings. Also, a recent paper by Manisankar et al.\textsuperscript{105,106} tried to investigate the effect of trip and vortex generator on double wedge surface by applying binary PSP method.

PSP could be applied as global measurement method to complement discreet static pressure reading by transducer. For example, the scramjet inlet-isolator streamwise static pressure
distribution usually measured by transducer is limited by how many pressure taps could be drilled, and valuable pressure data between adjacent taps could be loss. Similarly, three-dimensional effects of inlet-isolator flow could be visualized and investigated clearly with PSP pressure map. Moreover, patterns on sidewall surface made by glancing shock train inside isolator could give hints of flow Mach number. Thus, this thesis will rely mostly on PSP methods to fully characterize a generic scramjet inlet-isolator. Detailed PSP theory can be found in the appendix.

3.5.2 PSP System Facilities

Aero-physics Laboratory in The University of Manchester has developed in-house PSP systems suitable for many applications. Different combinations of luminophores and polymer binders have been tested with their characteristics reported in Zare-Behtash et al.\textsuperscript{107,108} and Quinn et al.\textsuperscript{109}. Typically, two types of luminophore are available in the lab, Ruthenium bathophenanthroline [Ru(II)] and Platinum-tetrakis (pentafluorophenyl) Porphyrin (PtTFPP), and they can be applied with polymer-HCl, polymer-Ace, thin layer chromatography (TLC) plate, and anodized-aluminium porous binder. Polymer-HCl and polymer-Ace used similar sol-gel precursor, methyl triethoxysilane (MTEOS), as binder but with different solvents; hydrochloric acid (HCl) for the former and acetone for the latter. Typically, polymer binder with HCl is more sensitive than with acetone. TLC and anodized aluminium provide rapid response to change in pressure. PtTFPP is generally more sensitive to pressure compared to Ru(II), but the latter has greater quantum yield that translates into greater signal level. PtTFPP-based PSP also has lower temperature sensitivity compared to Ru(II) for similar binder used.
Ruthenium-based PSP methods have been applied to jet and nozzle interaction flow and reported by Zare-Behtash et al.\textsuperscript{108,110–113}. PSP painted on the sidewall of the two-dimensional nozzle was able to visualise shear layer, shock cell structures and expansion fan. Glancing shock phenomena at transonic speed have also been studied by Zare-Behtash et al.\textsuperscript{110,111,113} and Kontis et al.\textsuperscript{114,115}. Shock traces can be identified easily on the sidewall pressure map. The application of in-house developed PSP systems in hypersonic speed has been reported by Yang et al.\textsuperscript{102,104,116,117}. Moving and glancing shock wave in shock tube application has been investigated using PSP by Gongora-Orozco et al.\textsuperscript{118}. The suitability of in-house developed PSP system for low speed flow has also been demonstrated by Quinn et al.\textsuperscript{118,119}.

In conclusion, the lab has necessary expertise and high quality facilities in applying PSP for investigating complex flow phenomena in diverse flow speed regime.

### 3.5.3 Luminophore and Substrate Selection

The scramjet inlet-isolator model was made from aluminium alloy, but few attempts at anodizing the model were not successful. The alloy was similar to the material of the anodized-aluminium double-ramp model used by Yang et al.\textsuperscript{102,104,116,117}, hence this failure came as a surprise. The exact reason for this unsuccessful anodization is currently unknown. Fast dynamic response polymer-ceramic PSP (PC-PSP) method\textsuperscript{120–127} was also developed in-house by mixing DuPont R900 TiO\textsubscript{2} particles into the available polymer binder (MTEOS) to act as porous structure similar to anodized-aluminium. The resultant binder was found to be too brittle and not robust enough for extreme flow environment in HSST. Commercially available thin layer chromatography (TLC) plate, which has porous structure suitable for high speed PSP, was available as well but the complex geometry of the inlet was quite a challenge
to fix TLC plate onto it. TLC plate also has quite significant thickness relative to the isolator height, thus applying the TLC layer onto isolator surface would change the characteristics of the inlet-isolator. Thus, only low frequency response polymer substrate remained as an option. Nevertheless, since the scramjet inlet-isolator was designed for self-starting and no backpressure will be imposed at the isolator exit, the unsteadiness of the internal shock systems was not expected. The flow could be safely assumed to be steady and polymer-based PSP will be good enough for pressure measurement.

Quinn et al.\textsuperscript{109} did a survey on different combinations of luminophores and substrates, and they concluded that PtTFPP with polymer binder has more pressure sensitivity compared to Ru(II) with similar polymer binder (see Figure 3-11). It was also found that PtTFPP-polymer has lower temperature sensitivity than Ru(II)-polymer (see Figure 3-12). PtTFPP-polymer PSP will experience only 0.75% drop in intensity with a unit increase in temperature. Ru(II)-polymer has much higher sensitivity value at -1.07% intensity per °C. Puklin et al.\textsuperscript{128} states that PtTFPP in fluoroacrylic polymer approaches “ideal” PSP characteristics, where temperature and pressure sensitivity are independent. Thus, for the current experimental campaign, PtTFPP was chosen as the main luminophore as it has high pressure sensitivity, but most importantly its low temperature sensitivity quality will lower the measurement error due to large temperature rise in high supersonic speed.

Preparation of PSP starts by mixing polymer and MTEOS with solvent, ethanol and HCl. PtTFPP molecules were then added to the mixture with concentration of 4 mM ± 0.2 mM. The optimum ratio for the mixture was 13.7 mg of PtTFPP powder: 13.9 g of MTEOS (98%): 8.34 g of ethanol (99.5%): 5.56 g of HCl (0.1 M). The mixture was then introduced with ultrasonic bath for typically 30 minutes to ensure all PtTFPP particles were dissolved. The
scramjet model was polished with sand paper and sprayed with 3 – 5 layers of Ambersil Matt white acrylic prior to paint application. The white base coat serves as a reflector to direct all emitted photons to required direction for detection. The luminophore-polymer paint mixture was applied only after the base coat has fully dried. Clas Ohlson Airbrush Kit (30-3224) was used to spray the paint mixture. The supply pressure for the airbrush was set typically to 2 bar and normally about 15 – 20 layer of thin coats are applied onto the model. Thin and light coat is ensured by applying a fast sweeping action during the spraying. The model was left to dry for at least a minute (2 minutes is best) after every 2 light coats. This is very important as it has been found that applying successive coats without time gap between each other will combine the overall layers into single layer that will easily peel off from the model surface. Since the PSP mixture have been reported to have significant photodegradation rate of -8.5% per hour, the mixing and spraying process must be done in dimmed environment. The model with PSP layers was left inside an oven for at least 7 hours at 343 K. Test samples that have been cured in less time were found to suffer from easily cracked paint layers.

Ru(II)-based PSP mixture was also developed for comparison purpose. It followed similar mixing, spraying and drying procedures as explained above. The amounts for each components in the paint were in the ratio of 7.36 mg of Ru(II) : 7.36 g of MTEOS (98%) : 5.52 g of ethanol (99.5%) : 2.94 g of HCl (0.1M). For cost comparison, it must be noted that PtTFPP cost as much as USD 689.26 per gram, while Ru(II) was a lot cheaper at USD 142.60 per gram.
**Figure 3-11** Pressure sensitivity of different combination of luminophores and substrates (figure taken from Quinn et al.\textsuperscript{109})

**Figure 3-12** Temperature sensitivity of different combinations of luminophores and substrates (figure taken from Quinn et al.\textsuperscript{109})
3.5.4 PSP System Setup

Figure 3-13 Side view of PSP system Setup-1 for compression and isolator surface pressure measurement

Figure 3-14 Top view of PSP system Setup-2 for isolator sidewall pressure measurement
Figure 3-13 shows the side view of PSP system Setup-1 optimized for measuring pressure intensity on compression and body-side isolator surface. The camera was placed on top of the test section, with its lens focused onto the model through the top window at an angle. The camera setup angle was not important because model length was calibrated by markings on the model. The angle of camera placement was adjusted as necessary to accommodate the full model length. The scramjet inlet-isolator model was fixed on sting holder, which can be rotated on horizontal-vertical plane to vary the model angle-of-attack. For this setup, LED plates were placed on both side windows for even illumination on the model. LED lamps were preferred than other light source due to their monochromatic nature. Filter was used for the camera to minimize recording of unwanted photon from environment. For in-situ calibration, static pressure distribution was measured using discreet transducer at the same time as pressure intensities were recorded. Intensity measurements were done in the darkest possible condition to reduce light contamination.

PSP Setup-2 in Figure 3-14 has the camera and both LED panels placed on the same side window. PSP were sprayed onto inlet model sidewall farthest from the camera whilst nearest sidewall provided a window for optical access. The camera viewing plane was parallel to side window plane. In this particular case, the farthest sidewall component sprayed with PSP was made from aluminium without Perspex inserted. The LED panels were fixed to the lowest angle to side window plane as possible to minimize shadows by aluminium frame of model sidewall. Sidewall window on the other side of the test section and the top window were covered with blinds to avoid illumination contamination. In addition, all lights in the lab were turned off as well. Similar camera filter utilised in Setup 1
was also used for this setup. For this setup, no pressure taps were drilled on sidewall surface for calibration. Thus, intensity will be calibrated with pressure measured using PSP Setup-1.

For both setup, each LED panels had 192 units of UVSTZ-395-30 model LED. The peak wavelength from LED illumination was 395 nm. Testing on the illumination of the LED lamps were conducted previously, and it was found that no signal can be recorded at wavelength higher than 420 nm or lower than 330 nm\textsuperscript{109}. Ultraviolet LED used in the panels was chosen specifically because its peak wavelength coincided with peak excitation wavelength of PtTFPP-based PSP (see Figure 3-15). Emission peak wavelength for this luminophore was about 650 nm as shown in Figure 3-15, thus a combination of 530 nm long-pass and 700 nm cut-off filter were placed together in front of camera lens. This will ensure the camera to record only photons with wavelength in 530 - 700 nm range. The filter was fixed with adhesive seal on the lens to avoid any gaps where unwanted photon could enter. Scientific grade 12 bit CCD camera model LaVision Imager Intense was used as the main photon detector device for both setups. The camera was set to optimum exposure time of 7.5 ms with frame rate of 10 Hz. Sufficient exposure time was needed to allow the camera to capture the maximum photons possible and make use of its full-well capacity. Recorded images were pre-processed and viewed in real time by using Davis 7.0 software installed on Microsoft Windows-based workstation connected to the camera. MATLAB image processing toolbox was then utilised for further data processing.

With capture duration set as 10 seconds, there were in total 100 intensity images per wind tunnel run. Only the final 30 images before the end of tunnel run were selected and summed together to improve signal to noise (SNR) ratio\textsuperscript{129}. Obviously, increase in number (N) of summed images will lead to higher SNR, which is proportional\textsuperscript{130} to square root of N.
However, the pressure intensity changed significantly during tunnel start and took some times to reach a stable plateau due to high response time of PtTFPP-polymer PSP. Thus, the number of image (N) was limited to just 30. Reference wind-off images were taken immediately after each wind-on test run. The reason is to have the closest temperature distribution on the model as possible to those exist during wind-on\textsuperscript{116}. Similarly, the thirty wind-off images for each case were summed together. Dark images were captured with no wind tunnel flow and excitation source was turned off. Illumination contaminations were eliminated from measured intensity by subtracting the dark images\textsuperscript{113}. Temporal variations in intensity measured must also be considered. With 30 images captured, the intensity value at the same pressure level has been degraded by \(0.71 \times 10^{-2}\)% after 3 seconds. Photodegradation between each wind-tunnel run was minimized by turning on the LED panels only during recording of images. The model was removed from the test section and kept inside oven to prevent from moisture/dust/oil contamination on the surface.

![Figure 3-15 wavelength spectra of illumination source and luminophores (figure taken from Quinn et al.\textsuperscript{109})](image)

Page 110 of 270
3.5.5 Error and Uncertainty Analysis

Since the system depends on many components and photokinetics mechanism, there will be many associated error in pressure measurements as well. Liu et al.\textsuperscript{131,132} listed all possible source of errors involved in PSP measurement. Bell et al.\textsuperscript{130} categorized the source of errors into three main groups. Group 1 lists the source of errors related to the response of the paint with change in pressure. For example, a-priori calibration might still contain error due to imperfect fitting between pressure and intensities, given other factors such as temperature, illumination intensity, moisture and others are in control conditions.

The second group is related to errors inherent in the system setup and/or its components. The unavoidable error from this group comes from unwanted noise measured by the photodetector. The noise can be limited by subtracting from measurement, the image captured by the same setup but without light source\textsuperscript{132} called a dark image. Zare-Behtash et al.\textsuperscript{108} stated that photon shot noise is inversely proportional to square root of the number of photon captured, $\Delta_n = 1/\sqrt{\langle N \rangle}$. If the camera exposure is set to capture the maximum photons allowed, then this type of error will be minimized. Photodegradation of the paint will occur every time the paint is exposed to illumination source whether intentionally or otherwise. If the images are recorded in series, the last image in the series would have lower intensity compared to the first image even if pressure level is similar. With steady flow, the series of images could be summed together to increase the SNR, and the photodegradation effect during each wind tunnel run is negligible. This will not be the case with unsteady high response PSP system, where each image is calibrated to pressure individually. Another type of error in this second group is related to filter leakage, where unsuitable filter allows detection of unexpected photon from ambient or any other source not related to test
sample emission. Most of error belonging to this group has been minimized by proper setup, as explained in Section 3.5.4.

The third group of error sources relates uncertainty during signal analysis. Temperature compensation must be considered in applying a-priori calibration curve for intensity-pressure conversion. Deformation or movement of model could also introduce errors by misaligning measured intensity at supposed pressure tap location.

For this current experimental campaign, only in-situ calibration process was used. Unlike a-priori method, this technique correlates intensity to pressure measured in the exact similar testing conditions; thus all source of errors are combined into calibration error\textsuperscript{132,133}. Even though the calibration error absorbed all possible errors, it was dominantly affected by temperature and model deformation error\textsuperscript{133}. Calibration error for current investigation was calculated simply by finding the standard deviation of curve fitted pressure ($p_{\text{in-situ}}$) from transducer measured pressure ($p_{\text{transducer}}$)\textsuperscript{132,133}. If the pressure data set have N number of observed discreet pressure points, then the overall calibration error is given by:

\begin{equation}
\varepsilon_{\text{in-situ}} = \sqrt{\frac{1}{N} \sum_{n} (p_{\text{predicted},n} - p_{\text{observed},n})^2}
\end{equation}

In-situ calibration error calculation assumes that the discreet pressure measured from transducer is the true pressure. However, problem arises in this particular experimental investigation when the transducer registered significant amplitude of pressure oscillations with time. To take this into account, in-situ predicted pressure must be related to the true pressure by:
Equation 3-21

\[ p_{\text{in-situ}} \pm (\varepsilon_{\text{in-situ}} + \varepsilon_{\text{transducer}}) = p_{\text{true}} \]

Where \( \varepsilon_{\text{in-situ}} \) is the standard deviation of curve fitted pressure from measured pressure, and \( \varepsilon_{\text{transducer}} \) is the standard deviation of transducer pressure oscillations in time. Another issue that must be considered is that there was no single value of \( \varepsilon_{\text{transducer}} \). As will be shown later in results chapter, Kulite transducer registered different amplitude of oscillations depending on its location on the scramjet inlet-isolator model. Thus, for each parametric test case, \( \varepsilon_{\text{transducer}} \) will be taken as average of all transducers on the model.

For PSP analysis of sidewall surface pressure, the intensity is calibrated in-situ by using predicted pressure from PSP analysis done on isolator surface. This is due to lack of discreet pressure tappings on sidewall surface. Thus, the uncertainty of PSP-derived pressure on sidewall is higher compared to uncertainty of pressure on isolator, such as shown in:

Equation 3-22

\[ p_{\text{in-situ2}} \pm (\varepsilon_{\text{in-situ2}} + \varepsilon_{\text{in-situ}} + \varepsilon_{\text{transducer}}) = p_{\text{true}} \]

Where \( p_{\text{in-situ2}} \) is the predicted sidewall pressure from PSP and \( \varepsilon_{\text{in-situ2}} \) is the standard deviation of \( p_{\text{in-situ2}} \) from isolator pressure \( p_{\text{in-situ}} \).

### 3.6 Infrared Thermography

#### 3.6.1 Background

Infrared (IR) thermal imaging technique has been gaining popularity in hypersonic experimental investigations due to its simple and non-intrusive nature. In fact, IR measurement technique was first demonstrated in hypersonic flow regime by Thomann and
Frisk in 1968. They measured surface heat flux distribution of an elastomeric paraboloid shape model in Mach 7 flow. Most of IR experiments in hypersonic or high supersonic speed regime were related to boundary layer transition, SWBLI characteristics, surface heat flux determination and Goertler vortices investigations.

IR measurement system usually consists of thermal imager camera that detects the radiated electromagnetic energy from surface of subject and numerical software for image processing. Photons radiated in IR wavelength by measured object are detected by the camera sensor and turned into electrical signal. The radiation intensity is dictated by Planck’s Law:

\[
E_\lambda = \varepsilon_\lambda E_{\lambda 0} = \frac{\varepsilon_\lambda C_1}{\lambda^5(e^{C_2/\lambda T} - 1)}
\]

Where \( \varepsilon_\lambda \) is the spectral hemispherical emittance, \( C_1 = 3.7415 \times 10^{-16} \text{ Wm}^{-2} \) and \( C_2 = 1.4388 \times 10^{-2} \text{ mK} \) are the radiation constants, \( \lambda \) is the radiation wavelength and \( E_{\lambda 0} \) is the blackbody monochromatic radiation intensity. However, in real practice, no object can be truly considered to have black body radiation properties. Radiation from object is transmitted through medium (test gas or atmosphere) and specially designed lens (typically germanium, silicone or sapphire) before reaching Focal Plane Array (FPA) photon sensor. Filters for the lens are available to exclusively allow only photons of certain wavelength.

Fully computerized IR system allows for recording and manipulation of image to provide for real-time temperature mapping. The video recording of temporal temperature variation enables the determination of heat transfer rate. There are three methods of calculating heat flux from IR measured temperature field with the one-dimensional “thin-film” model is the
most popular method used\textsuperscript{148}. Spatial temperature variation and heat flux map provide aerodynamicist the opportunity to qualitatively and quantitatively analyzed different flow phenomena. A concise review of different applications of IR in fluid dynamics research has been done recently by Carlomagno and Cardone\textsuperscript{148}.

### 3.6.2 Experimental Setup

FLIR ThermaCAM SC 3000 Infrared camera was available for current experiments. The measuring range of the camera is from -20 °C to 1,500 °C, with maximum estimated error of ± 2%. Its onboard cooling system is capable of lowering the temperature from ambient to 70 K in less than 6 minutes. The images can be captured with frequency of up to 900 Hz in 14-bit format. For current experiments, the camera was set to capture only 100 Hz. Its thermal sensitivity was 20 mK at standard temperature of 30 °C. It can automatically correct for different atmospheric conditions based on distance, temperature and relative humidity. The camera was accompanied by ThermaCAM Researcher Software for real-time and post processing. The schematic of the experimental setup is shown below:
The camera was placed on top of test section similar to PSP system Setup-1. This was because only top plate can be fitted with germanium window required for IR measurement. The model was painted black to achieve emissivity of about 0.95. As the model was made from aluminium alloy with high heat conductivity, the flow visualization shown by temperature increase will be less visible compared to using insulator material. The properties of the material are given below:

Table 3-2 Thermal properties of the material used for infrared experiments

<table>
<thead>
<tr>
<th>Material</th>
<th>Density</th>
<th>Thermal Heat Capacity</th>
<th>Thermal Conductivity</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aluminium Alloy (6082)</td>
<td>2,700 kg/m³</td>
<td>896 J/kg.K</td>
<td>167 W/m.K</td>
</tr>
</tbody>
</table>

The current experiments concentrated mostly on flow on compression ramp surface; thus the cowl was not fitted. As the sidewall and cowl top window was not made from IR transmittable material, no analysis of isolator flow characteristics can be done with IR.
technique. However, sidewall was put in place to study its influence on the flow on compression ramp.

3.7 Numerical Analysis

3.7.1 Background

Numerical simulation has been relied upon equally as much as experimental works in regards of scramjet inlet-isolator investigations. More often than not, experimental works done in characterizing scramjet inlet are accompanied by computational fluid dynamics (CFD) analysis to help in interpreting experimental observations or to fill in any gap in measurements, where geometry or any other factors prevented from possible experimental flow-diagnostics\textsuperscript{22,60,73,149–155}. Numerical methods could also provide good insight prior to inlet designs process or predict parametric inlet performance before actually spending valuable resources in conducting experiments in costly wind tunnel\textsuperscript{156}.

3.7.2 Numerical Code Available

In this current investigation, commercially available computational fluid dynamics package FLUENT was chosen as the main numerical solver. FLUENT has been tested and validated against many flow phenomena related to scramjet inlet-isolator flow. For example, Guan\textsuperscript{157} has managed to replicate to a good degree, using FLUENT as numerical solver, the inlet-isolator flow parameters found experimentally by Reinartz \textit{et al.}\textsuperscript{60}. Das and Prasad\textsuperscript{68,69} has done both experimental and computational investigations of mixed compression supersonic inlet, and the solver was shown to be capable of matching experimental observations.
Internal flow of scramjet intake with bleed has been investigated by Häberle and Gülhan\textsuperscript{158}. They concluded that FLUENT was able to reproduce their experimental flow field data. Dong \textit{et al.}\textsuperscript{159} has designed a hypersonic jaw inlet and subjected it to both experimental and numerical testing. Data produced by FLUENT was very encouraging for their study. Recently, Abdel-Salam and Micklow\textsuperscript{160} has shown that FLUENT was capable of simulating interaction between scramjet isolator and combustor with fuel injection that fits very closely to experimental data. Recently, Ahmed\textsuperscript{161}, in his thesis, conducted a validation exercise of FLUENT solver with three experimental data on spiked-blunt body at hypersonic problem done by Crawford\textsuperscript{162}, Kalimuthu \textit{et al.}\textsuperscript{163} and Gnemmi \textit{et al.}\textsuperscript{164}. The exercise showed that FLUENT was capable of predicting severe aerothermal heating, flow oscillations, shock-shock interactions, flow separation and many other phenomena that typically occurs in scramjet inlet-isolator. Erdem\textsuperscript{71} and Erdem and Kontis\textsuperscript{165} validated FLUENT using data of transverse jet injection in Mach 5 flow. The flow phenomena that characterized such problems include shock-shock interactions, Mach disk, strong viscous interactions and many more flow features similar to scramjet inlet-isolator flow. Thus, it is believed that FLUENT could predict flow data close to the current experimental investigation and help in data interpretation where necessary.

### 3.7.3 Turbulence Model Selection

For the unit Reynolds number where current scramjet inlet-isolator experiments were conducted, the boundary layer was expected to be laminar unless tripped\textsuperscript{71}. Nevertheless, boundary layer separation at compression corner was expected, and it has been shown previously\textsuperscript{166} that transition to turbulence could be induced by re-attachment. Thus, a
suitable turbulence model must be chosen for the numerical solver. FLUENT has the following turbulence models: [1] Spalart-Allmaras\textsuperscript{167}, [2] Standard \( k-\epsilon \) model\textsuperscript{168}, [3] Renormalization group \( k-\epsilon \) model, [4] Realizable \( k-\epsilon \) model, [5] Standard \( k-\omega \) model\textsuperscript{169}, [6] Shear Stress Transport (SST) \( k-\omega \) model\textsuperscript{170}, [7] Reynolds Stress Model (RSM), and [8] Large Eddy Simulation (LES). Model number [1] – [7] are all related to Reynolds Averaged Navier-Stokes (RANS) equation, which use statistical method of turbulence fluctuation simplification. Spalart-Allmaras is the simplest RANS turbulence model which employs only one equation to model the transport equation. RSM model is more complex, with five equations describing the Reynolds stresses. All other listed RANS models use two-equations and are more widely used than RSM. On the other hand, LES tries to solve the Navier stoke equations of large scale eddies directly, and use model simplification at smaller energy scales. Typically, more computational cost and poor convergence is associated with using more complex turbulence model (from [1] – [8] of the listed model).

A survey of fundamental shock wave turbulent boundary layer interactions simulation validated by solid experimental data has been made by Knight et al.\textsuperscript{171}. Flow phenomena surveyed includes two-dimensional compression corner, two-dimensional shock-wave boundary layer interactions, and three-dimensional fin problem. The reviewed literatures included in the survey have diverse turbulence model, such as Direct Numerical Simulations (DNS) that directly solve for Navier-Stokes equation without attempting to model it. LES and many RANS type equations are also employed. The conclusion given in that survey was that DNS produced the best results similar to experiment. However, DNS was very costly and according to the survey, DNS was limited to only low Reynolds number flow. LES was capable to closely perform up to par with DNS with up to 5% difference, but with significant saving in
computing power. The survey also commented that DNS, LES and RANS differed considerably with experimental measurement of heat flux. RANS generally has quite satisfactory fit with experimental data.

Another recent review of turbulence model validation has been presented by Roy and Blottner\textsuperscript{172}. They focused only on the experimental and numerical works done on two-dimensional and axisymmetric cases in hypersonic flow. They limited their survey to only one- and two-equation RANS model. One of the conclusions was that no turbulence model could claim superiority and consistently fit experimental data in every flow situations.

Gnoffo \textit{et al.}\textsuperscript{173} conducted numerical validation of a few turbulence models using published experimental data, focusing only on compression corner (two-dimensional and axisymmetric) problems in Mach 7 and above. The turbulence models tested were Baldwin-Lomax, Cebeci-Smith, Spalart-Allmaras, Menter’s SST $k$-$\omega$, and two version of Wilcox’s $k$-$\omega$. For all the cases surveyed, each turbulence model seems to have equal chance of performing better than the others.

Doctorate thesis by Bosco\textsuperscript{151} published recently has shed more light to finding the most suitable turbulence model for scramjet inlet-isolator flow. She compared SST-$k$-$\omega$ with RSM turbulence model in hypersonic intake flow situations (external compression corner and internal inlet-isolator flow), and found that RSM was superior in simulating experimental data. This was expected since RSM was more costly and took longer to solve. However, SST $k$-$\omega$ data was remarkably close to RSM results and consumed much less computing power.

The comparison of accuracy between SST $k$-$\omega$ and RSM in scramjet inlet flow prediction has been done by Reinartz\textsuperscript{174} as well, and her results also show similar conclusion. The suitability
of using SST $k$-$\omega$ turbulence model for scramjet inlet investigation was demonstrated by Krause in his doctorate thesis. Hence, the current study will utilise Menter’s SST $k$-$\omega$ turbulence model embedded in FLUENT to accompany the experimental observations.

3.7.4 Computational Fluid Dynamics Theory

The following discussion on the equations behind the solver is based on FLUENT user guide lecture series. Navier-Stokes equation is derived from Newton’s second law on a control volume of fluid flow. The conservation of momentum of such problem is given by:

\begin{equation}
\rho \left( \frac{\partial u_i}{\partial t} + u_j \frac{\partial u_i}{\partial x_j} \right) = -\frac{\partial p}{\partial x_i} + \frac{\partial}{\partial x_j} \left( \mu \frac{\partial u_i}{\partial x_j} \right)
\end{equation}

Where $\rho$ is the density, $x_i$ is the spatial axis ($x$, $y$, and $z$), $u_i$ is the component of velocity ($u$, $v$, and $w$), $p$ is the static pressure, and $\mu$ is the dynamic viscosity.

In turbulence, any variable could be separated into mean and fluctuating components, for example:

\begin{equation}
u_i = \bar{u}_i + \dot{u}_i
\end{equation}

Substituting into the original equation yields:

\begin{equation}
\rho \left( \frac{\partial \bar{u}_i}{\partial t} + \bar{u}_j \frac{\partial \bar{u}_i}{\partial x_j} \right) = -\frac{\partial \bar{p}}{\partial x_i} + \frac{\partial}{\partial x_j} \left( \mu \frac{\partial \bar{u}_i}{\partial x_j} \right) + \frac{\partial R_{ij}}{\partial x_j}
\end{equation}
Where \( R_{ij} \) is called the Reynolds stress tensor and can be related to turbulent viscosity \( \mu_T \) via Boussinesq hypothesis:

**Equation 3-27**

\[
R_{ij} = -\rho \bar{u}_i \bar{u}_j = \mu_T \left( \frac{\partial \bar{u}_i}{\partial x_j} + \frac{\partial \bar{u}_j}{\partial x_i} \right) - \frac{2}{3} \mu_T \frac{\partial \bar{\omega}}{\partial x_k} \delta_{ij} - \frac{2}{3} \rho k \delta_{ij}
\]

Turbulent viscosity \( \mu_T \) is determined differently depending on the turbulence model used.

The \( k - \varepsilon \) model calculates the viscosity by:

**Equation 3-28**

\[
\mu_T = f \left( \frac{\rho k^2}{\varepsilon} \right)
\]

On the other hand, standard \( k - \omega \) uses the function below:

**Equation 3-29**

\[
\mu_T = f \left( \frac{\rho k}{\omega} \right)
\]

Where \( k \) is the turbulent kinetic energy, \( \varepsilon \) is the turbulence dissipation rate and \( \omega \) is the specific dissipation rate.

Menter’s SST \( k - \omega \) uses a blending equation to mix \( k - \omega \) set of functions near the wall with \( k - \varepsilon \) model outside the boundary layer. Thus, this method combines the best from both widely used turbulence models.

### 3.7.5 Numerical Setup

Density-based solver was selected for solving the equations with FLUENT. Second-order accuracy was achieved by utilising Second Order Spatially Accurate Upwind Scheme (SOU)
with Roe’s Flux-Difference Splitting. The turbulence intensity at freestream was set to 0.5 % to allow for correction for transition. The CFD package used in this study utilised low-Re modifications of the original turbulence model to model the transition behavior of the flow correctly. Low-Re model depends on damping functions that can predict viscous sub-layer near the wall. This damping function eliminates the need for a wall function. Wilcox\textsuperscript{177} proposed changing the value of $k$ when approaching the wall and adding two more transition specific coefficients in his $k-\omega$ turbulence model.

Turbulence viscosity ratio was fixed at 1. Stability was ensured by setting the initial Courant-Friedrichs-Levy (CFL) number to 0.5, and it was gradually increased by the same value every 1,000 iterations. Inviscid solutions for every parametric case were used as inputs for initializations for the turbulence solver.

Computational domain was bounded by pressure inlet, symmetry, far-field, constant temperature walls and two pressure outlet (see Figure 3-17). The properties for pressure inlet were taken from HSST flow conditions (Table 0-1, page 247). Symmetry region between the pressure inlet boundary and first compression ramp wall assisted in maintaining stable iterations. Properties at the two pressure outlets were computed by assuming free flow, where the flow would exit the isolator expanding to freestream conditions. Meshes were made using quadrilateral cells with higher density grid concentrated around large flow turning region. The first grid spacing suitable for low-Re turbulence model used in this thesis was calculated by first setting\textsuperscript{178} the $y^+$ value to 1. This variable is the dimensionless $y$-distance to wall surface and can be related to the size of the first grid size, $y$, by following the calculation method found in White\textsuperscript{179}. 
From Sutherland law, viscosity can be calculated as:

**Equation 3-30**

\[
\mu = \mu_0 \frac{T_o + c (T / T_o)}{T + c (T / T_o)}^{3/2} = 4.13 \times 10^{-6} \text{ kg/ms}
\]

Where \(T_o = 273.15 \text{ K}\), \(\mu_0 = 1.72 \times 10^{-5} \text{ kg/ms}\), \(c = 110.4 \text{ K}\), and \(T = 62 \text{ K}\), which is the freestream temperature.

Reynolds number can be calculated by:

**Equation 3-31**

\[
Re = \frac{\rho U L}{\mu} = 1.76 \times 10^6
\]

Where \(\rho = 0.0685 \text{ kg/m}^3\), \(L = 0.135 \text{ mm}\), and \(U = 787 \text{ m/s}\).

Inserting the calculated \(Re\) into formula for Schlichting\textsuperscript{180} skin friction coefficients:

**Equation 3-32**

\[
C_f = [2 \log_{10}(Re) - 0.65]^{-2.3} = 3.39 \times 10^{-3}
\]

Wall shear stress is given by the formula below:

**Equation 3-33**

\[
\tau_w = \frac{1}{2} C_f \rho U^2 = 72.09 \text{ kg/ms}^2
\]

According to Law of the Wall, frictional velocity can be calculated by:

**Equation 3-34**

\[
u^* = \sqrt{\frac{\tau_w}{\rho}} = 32.43 \text{ m/s}
\]

Inserting \(y^+ = 1\) resulted in estimation of suitable first cell height:
Equation 3-35

\[ y = \frac{y^+ \mu}{\rho u^+} = 1.85 \times 10^{-6} \ m \]

For the current mesh, \( y \) was calculated to be around 2 micron. Thus, for all three different mesh sizes, the first grid size was always set to be \( y = 2 \) micron. It is important to note that the actual value of \( y^+ \) depends on the wall shear stress which is part of the solution from the numerical analysis. The calculation using Equation 3-30 to Equation 3-35 merely estimates the needed grid size in order to ensure suitability with the selected turbulence model.

The convergence for each case was considered established if the residuals of x-velocity, y-velocity and energy are lower than \( 10^{-4} \). This follows the precedent shown by Guan\textsuperscript{157} in his paper, which validated FLUENT software against available high speed inlet experimental data. It was lower than the default residuals convergence criterion set by the software developer (ANSYS), which is \( 10^{-3} \). This is to ensure high confidence in the simulation. Further check for convergence was provided by ensuring that the difference between mass flux entering and leaving the computational flow domain was less than 1% relative to the smallest mass flux.

Sensitivity of results to grid density was analysed by using three different grid refinement levels. It is concluded from Figure 3-18 that medium grid of 52,250 had very close fit with pressure readings from fine grid of 73,840 cells. Coarse grid of 34,485 cells also showed close approximation with the other two grid levels, except around region \( x/L = 0.3 – 0.5 \) where insufficient grid was provided to resolve compression corner separation. Therefore, medium grid was adopted for all parametric case studies.
The computer used to solve the equations was Dell Optiplex 760 workstation with Intel Core2Quad Q9650 (3.00 GHz) and RAM size of 4 GB. It took about 8 to 10 hours (solver parallelized with 3 core) to solve each case, considering it needs to solve for inviscid flow prior to turbulence modelling.

Mach number at isolator exit was taken by mass averaging across the isolator height. This will be compared with experimentally predicted Mach number. Numerical schlieren images were produced by Tecplot 360 software by computing the density in the y-direction for the whole flowfield. The density gradients were then visualized using colour mask to mimic the experimental colour schlieren.

![Figure 3-17 Medium mesh](image)
CHAPTER 4 SCRAMJET INLET-ISOLATOR CHARACTERISTICS AT DESIGN CONDITIONS

This chapter discusses the characteristics of the generic inlet-isolator flow field at supposed design conditions, which are summarized in Table 4-1 below. The characteristics were considered as the baseline, which will be compared with off-design conditions. It must be noted that the simple design process of the generic inlet-isolator disregarded the viscous effects; thus rendering the real characteristics of the aforementioned inlet to be slightly different from inviscid expectations. Boundary layer displacement, SWBLI, shock-shock and shock-expansion wave interactions occurred in the external and internal section of the inlet-

Figure 3-18 Normalized static pressure of centreline of model scramjet inlet-isolator with different mesh sizes
isolator as depicted in Figure 4-1 below. The external flowfield region was considered to be from the first compression ramp leading edge up to minimum throat area, whilst the internal section was from the throat up to combustor face. In this chapter, the following Section 4.1 focuses on the external flow field, whilst Section 4.2 concerns only on the internal flowfield.

Table 4-1 Parameters for baseline case conditions

<table>
<thead>
<tr>
<th>Case Name</th>
<th>Mach Number</th>
<th>Reynolds Number</th>
<th>Stagnation Pressure</th>
<th>Stagnation Temperature</th>
<th>Cowl Geometry</th>
<th>AoA</th>
</tr>
</thead>
<tbody>
<tr>
<td>Baseline</td>
<td>5 ± 0.4%</td>
<td>13.2 × 10^6 m^-1</td>
<td>6.5 ± 0.05 (bar)</td>
<td>375 ± 5 (K)</td>
<td>Base-cowl</td>
<td>0°</td>
</tr>
</tbody>
</table>

Figure 4-1 Colour schlieren image of scramjet inlet-isolator for baseline case
4.1 External Compression Flow Field

4.1.1 Flow Properties

Figure 4-2 (a) Experimental and (b) numerical schlieren images of scramjet inlet for baseline case focusing only around compression corner.

Figure 4-3 Schematic of flow structures on double ramp for baseline case where SW1 = leading edge shock wave, SW2 = separation shock, SW3 = re-attachment shock, $\theta_{SW1} = \text{angle SW1 to horizontal}$, $\theta_{SW2} = \text{angle SW2 to horizontal}$, $\theta_{SW3} = \text{angle SW3 to horizontal}$, $\delta_1 = \text{first ramp angle}$, $\delta_2 = \text{boundary layer separation wedge angle}$ and $\delta_3 = \text{second ramp angle}$.

Figure 4-2 (a) shows colour schlieren image of the inlet-isolator for baseline case focusing only on the flow around the compression corner between the first and second ramps. The colour has been enhanced by increasing brightness, contrast and saturation to enable easier identification of flow phenomenon. Corresponding numerical schlieren image using similar colour scheme is provided in Figure 4-2 (b). Figure 4-3 shows the simplified schematic of the...
flow structures that occurred on the compression ramp for the current baseline case. Typically, boundary layer separation at compression corner will divide the flow into five zones (i.e. A, B, C, D and E) bounded by shock waves, an expansion wave and a slip line. Using the oblique shock relations, detailed flow properties in each zone are determined and compared with the numerical predictions (see Table 4-2, Table 4-3 and Table 4-4).

Table 4-2 Measured shock and flow turning angle relative to horizontal axis from schlieren image

<table>
<thead>
<tr>
<th>$\theta_{SW1}$</th>
<th>$\theta_{SW2}$</th>
<th>$\theta_{SW3}$</th>
<th>$\delta_1$</th>
<th>$\delta_2$</th>
<th>$\delta_3$</th>
</tr>
</thead>
<tbody>
<tr>
<td>19.59°</td>
<td>26.57°</td>
<td>34.50°</td>
<td>10.23°</td>
<td>12.85°</td>
<td>21.23°</td>
</tr>
</tbody>
</table>

Table 4-3 Flow properties on compression ramp surface using measured shock and flow angle from schlieren image

<table>
<thead>
<tr>
<th>Zone</th>
<th>Mach</th>
<th>$p$ (kPa)</th>
<th>$p_f$ (kPa)</th>
<th>$T$ (K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>5.00</td>
<td>1.22</td>
<td>644.92</td>
<td>62.5</td>
</tr>
<tr>
<td>B</td>
<td>3.98</td>
<td>3.79</td>
<td>558.11</td>
<td>90.09</td>
</tr>
<tr>
<td>C</td>
<td>3.79</td>
<td>4.91</td>
<td>557.13</td>
<td>97.02</td>
</tr>
<tr>
<td>D</td>
<td>3.23</td>
<td>10.34</td>
<td>534.37</td>
<td>121.49</td>
</tr>
<tr>
<td>E</td>
<td>3.20</td>
<td>10.27</td>
<td>505.89</td>
<td>123.14</td>
</tr>
<tr>
<td>F</td>
<td>3.24</td>
<td>10.27</td>
<td>534.37</td>
<td>121.23</td>
</tr>
</tbody>
</table>

Table 4-4 Flow properties on compression surface from computational analysis

<table>
<thead>
<tr>
<th>Zone</th>
<th>Mach</th>
<th>$p$ (kPa)</th>
<th>$p_f$ (kPa)</th>
<th>$T$ (K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>5.00</td>
<td>1.22</td>
<td>644.92</td>
<td>62.5</td>
</tr>
<tr>
<td>B</td>
<td>3.93</td>
<td>3.83</td>
<td>529.48</td>
<td>89.54</td>
</tr>
<tr>
<td>C</td>
<td>3.71</td>
<td>5.14</td>
<td>526.27</td>
<td>97.69</td>
</tr>
<tr>
<td>D</td>
<td>3.16</td>
<td>11.18</td>
<td>517.59</td>
<td>122.74</td>
</tr>
</tbody>
</table>
It was discovered that there were slight differences between the value predicted by the numerical solver and the value calculated analytically. These discrepancies could be attributed to the weakness of the numerical model to resolve the exact separation size in addition to the human error in measuring the shock angle from the schlieren image. Nevertheless, the differences can be assumed to be negligible as evidence from qualitative schlieren image in Figure 4-2 for both experimental and numerical suggest strong resemblance between each other.

### 4.1.2 Static Pressure Measurement on Compression Ramp for Baseline Case

![Figure 4-4 Normalised static wall pressure on compression ramp for baseline case](image-url)
The difficulty in determining the accurate static pressure in region under the influence of the corner separation is demonstrated in Figure 4-4. The PSP measured pressure using two different luminophores of Ru(II) and PtTFPP were compared with reading from Kulite, simulation and theoretical inviscid value. The analytical pressure predictions across the separation region taken from Table 4-3 are also included for comparison purpose. All Kulite readings were taken along the middle line of the model. There were two pressure transducers upstream of separation point, and one upstream and two downstream of the re-attachment shock. The x-coordinate of the compression ramp was normalized to the total length of the inlet L. All pressure was normalized to the static freestream pressure $p_{\infty}$. The compression corner was located at $x/L = 0.385$.

The CFD pressure profile shows that pressure increased well upstream of the corner due to the separation shock, until it reached a plateau. The pressure then increased again just upstream of $x/L = 0.4$ due to the re-attachment shock. The CFD predicted a slightly lower pressure profile on the second ramp in comparison with Kulite pressure.

The analytical calculation shows two pressure jumps at $x/L = 0.32$ and $x/L = 0.4$, where the separation and re-attachment shock were located. A strong viscous effect associated with hypersonic flow means that the observed shock angle on the first ramp was larger than inviscid solution. However, the static pressure calculated using the shock angle was similar to the inviscid. This phenomenon demonstrates that the small boundary layer displacement on the first ramp was negligible. The static pressure calculated analytically using re-attachment shock angle was significantly lower than Kulite’s, indicating that the re-attachment shock angle should be larger than the one measured from schlieren image.
It could be observed that both PSP measurements had similar pressure increase trend with Kulite readings. On the first ramp, PSP-PtTFPP over predicted the static pressure but had a nice fit on the second ramp. There was no noticeable pressure kink in the separation region detected using PSP-PtTFPP. On the other hand, the pressure measured with PSP-Ru(II) fits the Kulite only around the upstream part of the second ramp where the re-attachment shock occurred. However, it is believed that if more pressure transducer was placed across the separation region, more disagreement between PSP and Kulite could be found. Significant errors in PSP readings were found inside the region affected by sidewall shadows.

The output signal of PSP-Ru(II) had a lot more noise compared to that of PSP-PtTFPP, especially around the separation and re-attachment point. A contributing factor to this observation is the unsteadiness effect inside the separation region. It has been reported that PSP using Ru(II) as luminophore has significantly lower response time\textsuperscript{109} compared to that of PtFPP; hence indicating that the former is better in terms of responding to high frequency pressure change in unsteady flow. The unsteadiness effect, movement of the separation and re-attachment shock on double ramp case are well known\textsuperscript{147,181–183}. Nonetheless, since PSP-Ru(II) method is a polymer-based PSP with response time\textsuperscript{109} of only about 0.5 s, the unsteadiness effect could not be fully captured. Summation of the images with partial detection of unsteady spatial pressure fluctuations finally resulted in the large spatial noise measured by PSP-Ru(II) around separation and re-attachment point (Figure 4-4). The PtTFPP-based PSP has also been reported to possess lower spatial noise in comparison with Ru(II)-based PSP\textsuperscript{109}. The slight difference in readings between both PSP techniques and Kulite could be attributed to the inherent error in calibration process. The calibrations were done using the full scramjet (double ramp and isolator) intensity map where flow temperature varies.
considerably in streamwise direction. As the PSP response to temperature as well as pressure changes\textsuperscript{132}, a multiple calibration curve should be used to convert intensity into pressure at different temperature region.

4.1.3 Compression Ramp Streamwise Vortices

In Figure 4-2 (a), a slight density change in the boundary layer could be observed at the re-attachment point (identified by the colour change from red-orange to bright yellow). Incidentally, streamwise vortices were detected to originate from the re-attachment shock such as depicted in Figure 4-5 below. The figure shows global pressure distribution on the scramjet inlet focusing only on the second ramp using (a) Ru(II) and (b) PtTFPP-based PSP. In Figure 4-5 (a), Goertler vortices can be clearly identified as streak line originating from re-attachment line quite far downstream of compression corner. PtTFPP-based PSP was not able to detect any distinctive feature of the vortices. This vortices contributed to the noise in pressure recorded by PSP-Ru(II) shown in Figure 4-4. Nevertheless, the fact that PSP method could visualize streamwise vortices clearly is quite an exciting development; hence offering a new method as an alternative to the heat transfer mapping technique usually employed by many studies\textsuperscript{75–78,117,147,184,185}. The streamwise vortices can be better visualised in Figure 4-6, where streaks of temperature increase mark their locations, size and length. According to Ishiguro \textit{et al.}\textsuperscript{186} and Yang \textit{et al.}\textsuperscript{117}, each striation of pressure (or temperature) is located between two adjacent vortices (see Figure 4-7). If the combined effects lift up the flow from the surface, lower pressure/temperature area will be created. Similarly, if the two adjacent vortices induce downward flow towards the surface, higher pressure/temperature spot can be observed.
Figure 4-5 Normalised pressure map of scramjet inlet-isolator focusing only on second ramp using (a) PSP-Ru(II) and (b) PSP-PtTFPP

Figure 4-6 Increase of surface temperature on scramjet inlet for baseline case after 3 seconds of wind tunnel flow start. Note that the cowl component was not added

Figure 4-7 Cross sectional view of streamwise/Goertler vortices flow pattern
The streamwise vortices are most probably Goertler vortices since it only appeared at re-attachment line of compression corner separation\textsuperscript{77}. However, as shown by Matsumura \textit{et al.}\textsuperscript{77}, the detected streamwise vortices might have also originated from the leading edge instabilities, which would then merged or reacted with Goertler vortices at the compression corner\textsuperscript{77}. To isolate the cause of the instabilities, vortex generators could be introduced in the flow upstream of the separation and the change in pressure or heat transfer streaks, if any, could then be observed.

Goertler vortices on the second ramp are shown to retain their structure up to the inlet expansion corner as shown in Figure 4-6. This is consistent with the observation made by Yang \textit{et al.}\textsuperscript{117} and Neuenhahn\textsuperscript{147}, where the Goertler vortices occur at the re-attachment point and decay or lift-off prior to arriving at the expansion corner for many flow conditions. Therefore, it allows the assumption that Goertler vortices on the forebody ramp would not have any effect on the flow inside the isolator. Three-dimensional effect on the flow was noted, where the intensity of streak lines decreased dramatically in spanwise direction in both Figure 4-5 and Figure 4-6. Typically, Goertler vortices are usually associated with transition to turbulence\textsuperscript{18}. However, Neuenhahn\textsuperscript{147} refuted this misconception and explained that there have been many examples of Goertler vortices formation in laminar, transitional and turbulence boundary layer.
4.2 Internal Isolator Flowfield

4.2.1 Isolator Flow Features

Figure 4-8 (a) Experimental and (b) numerical schlieren images of the scramjet inlet-isolator for baseline case

Figure 4-8 shows the close-up view of flow structures in isolator section for the baseline case. Generally, the flow structures between experimental and numerical schlieren are closely similar. The shocks from the compression ramp impinged very close to cowl tip, indicating that the boundary layer displacement on the compression surface was minimal. Near maximum mass capture ratio was achieved by the inlet. Shock wave originating from cowl tip impinged downstream of shoulder (expansion corner) and interacted with the expansion wave, forming a Type 5 off-design wave interactions\(^{48}\). Three boundary layer separation bubbles appeared in the entire isolator section, and each separation was identified by the presence of separation and re-attachment shocks. These separations acted as obstructions to the core flow and caused losses due to the irreversible mixing and unwanted shocks. Strictly speaking, the shock structures surveyed in the figure did not exactly form a shock-train since there was no backpressure. They are more suitably named
as “background waves”\textsuperscript{22}, since their presence were all due to the impingement of cowl shock downstream of expansion corner at shoulder.

In both experimental and numerical observations, the largest separation bubble was located at shoulder region, with its separation point extending just upstream of the expansion corner. The sheer size of this separation indicated the severity of the cowl shock impingement on the isolator surface. The cowl shock was very strong, as the compressed flow from the double ramp had to turn 22° to enter the isolator section. The strength of the cowl shock could be decreased by deflecting it inwards, thereby reducing the turning angle required for the flow to enter the isolator, effectively reducing the impact of the separation. It must be noted that if the flow is inviscid, there should be an expansion wave located at shoulder and its interaction with cowl tip shock will strengthen the shock. This is another reason for the large separation at shoulder\textsuperscript{83}.

The inlet could be easily unstarted since the separation bubble originated upstream of shoulder. Tan \textit{et al.}\textsuperscript{47} has shown that an inlet with unsustainable backpressure would first enter into unstart mode by the formation of large separation at throat entrance, where it will oscillate by expanding and shrinking periodically (known a “little-buzz”). If more backpressure is introduced at the isolator exit, then the inlet would go into “big-buzz”, where the inner shock system inside the isolator would be dispelled, affecting the external shock system on the compression surface\textsuperscript{47}. In our case, since a large separation appeared at duct entrance with no backpressure involved, the introduction of backpressure would only increase the probability of unstart. Wagner \textit{et al.}\textsuperscript{187} explained that backpressure would induce separation at isolator exit and will intensify with the help of pre-existing thick boundary layer or separation bubble, known as “path of least resistance”.
The shock from the shoulder separation impinged on the cowl surface and caused the cowl tip separation. This cowl separation predicted numerically is much smaller in size than that observed experimentally. The separation extended upstream to cowl tip in experimental as opposed to that from the numerical observation where it is localized. This small difference was suspected to be related to the degree of sharpness of the cowl leading edge. A small radius of bluntness obviously existed in the experimental cowl, which gave rise to an entropy layer with strong vorticity. Boon and Hillier\textsuperscript{188} explained that as shock interacts with this entropy layer, a region of circulated flow would appear. The mechanism involved is purely inviscid but could manifest itself as flow separation in viscous conditions\textsuperscript{188}. Formation of both cowl tip and shoulder separation has reduced the effective isolator entrance height by about 70%.

Figure 4-9 shows the simplified schematic of the complex shock interactions around the isolator duct entrance observed experimentally. The cowl tip shock and shoulder separation shock crossed with each other to form Edney Type I interaction\textsuperscript{64}. The shock-shock interaction divided the flow of the isolator entrance into five distinct zones with different

![Figure 4-9 Schematic of shock interactions around throat area for baseline case](image-url)
flow velocity and direction. Zone G had non-uniform flow properties, inherited from zone E and F (see Figure 4-3). On the other hand, zone H, I, J and K had different properties from each other as they were located downstream of different shocks. A slip line was formed at intersection of cowl and shoulder separation shock, thus dividing the flow of zone J and K. This slip line was the product of viscous interactions between different flow velocities in zone J and K. The simple schematic demonstrates the complexity of flow around the throat area of the inlet-isolator and the anticipated difficulty in predicting satisfactory one-dimensional flow properties at this region. It makes more sense to utilise one-dimensional value of flow properties at isolator exit (with no backpressure) to represent the inlet-isolator, since it had higher level of uniformity compared to throat. The proof of this idea could be observed in the experimental schlieren of the inlet in Figure 4-8 (a), where the final oblique shock upstream of exit was faintly visible. This finding suggests that the shock was very weak and could be assumed to behave like a Mach wave, where all flow properties change very slightly across the shock. This assumption is proven via analysis of the pressure ratio across the final shock, which is discussed in Section 4.2.6 (page 151).
4.2.2 Isolator Surface Pressure Profile

It is particularly important to obtain the information on axial surface static pressure distribution of an inlet, especially at throat area, since it allows for quick back-of-envelope calculation of throat Mach number\textsuperscript{33}. It also serves as a good indication of distortion level, given the fact that an ideal undistorted isolator would have a constant static pressure\textsuperscript{31,33}. In addition, the peak pressure locations identified by the map are also important for structural integrity consideration.

The pressure profiles along the centreline of inlet-isolator model calculated using various methods are shown in Figure 4-10. There were four discreet pressure tappings inside the isolator section, which covered roughly only half of the isolator length. The readings from PSP methods filled the gap in estimating the pressure profile in the other half of the isolator region. The in-situ calibrations were done, assuming linear and quadratic curve of Stern-
Volmer plot. When the linear calibration was used, the fit to discreet pressure data was quite poor, especially on the first and second ramps. Even though linear Stern-Volmer plot has been derived from PSP luminescent kinetics (see Appendix B), a curve of higher order could also be used to plot intensity versus pressure ratio\textsuperscript{132}. A quadratic calibration curve allows for better fitting of PSP and Kulite readings. The reason for this observation could be explained by the fact that the surface temperature changed significantly across every shock wave in the whole flow field, with the most dramatic surface temperature difference was between the external and internal part of the scramjet inlet-isolator. The CFD boundary layer temperature predictions gave an indication that the boundary layer reached a peak temperature of approximately 316 K at compression corner re-attachment point. On the other hand, at the re-attachment point downstream of shoulder, the boundary layer temperature could reach up to 356 K. It has been shown\textsuperscript{109} that PtTFPP-based PSP system has quite large temperature sensitivity of about 1.07% /\degree C. Thus, it makes more sense to differentiate between the external and internal part of the inlet-isolator and have two separate Stern-Volmer calibration curves. However, manufacturing constraint limits the number of pressure tappings at external and internal sections. This would produce two Stern-Volmer plots; each plot having a small interpolation range. Hence, a single quadratic Stern-Volmer plot was used to correlate intensity and pressure across both the external and internal surfaces. Mathematically, a quadratic equation is equivalent to the product of two linear equations. Quadratic equation fit has larger coefficient of determination, $R^2$ value compared to that using linear calibration as shown in Figure 4-11. Therefore, it was decided that quadratic curve fit will be used for inlet-isolator surface pressure map for all cases to follow.
The PSP and CFD showed similar pattern of pressure rise and drop throughout the inlet. The patterns of pressure rise before reaching a plateau at isolator entrance corresponds to the shoulder separation shock. The sudden pressure jump downstream of the plateau corresponds to severe re-attachment shock impinging on the surface. There are two smaller pressure peaks further downstream, which match the impingement locations of separation and re-attachment shock from the third separation bubble (see Figure 4-8 (a), page 137). The pressure profile predicted numerically has a better fit with Kulite pressure in comparison with PSP (quadratic), especially at Kulite position of x/L = 0.684 (see Figure 4-10, page 141). The CFD code also predicted that the pressure rise due to shoulder separation shock was higher and at a small distance upstream of that recorded from PSP. Additionally, PSP and CFD also differs in predicting the two smaller pressure peaks, where the separation and re-
attachment shock from the third separation bubble impinged on the isolator surface. The discrepancies between PSP and CFD could be attributed to error in calibration or in the weakness of the simulation itself to resolve the correct viscous interactions. It must also be noted that the simulation was strictly two-dimensional, while in real experimental conditions, the inlet-isolator usually have some three-dimensional effects.

The flow inside the isolator is considered as two-dimensional if the pressure distribution on the centreline is similar to the reading taken at intersection of sidewall and isolator surface. In Figure 4-10 (page 141), PSP measurement of pressure at sidewall intersection has good similarity with PSP readings on the centreline of the model. There were some minor differences between the two profiles, and the first difference was the higher pressure rise before plateau around the isolator entrance indicating that shoulder separation shock near sidewall was stronger in comparison to centreline. This finding is consistent with the postulated flow structures for glancing shock boundary layer interaction problem reported by Stollery\textsuperscript{189} and Kubota and Stollery\textsuperscript{190}. In their studies, it was stated that the oblique shock would have larger angle as it glances and interacts with boundary layer on the sidewall (see sketch in Figure 4-12). The oblique shock with larger angle at sidewall would result in higher pressure increase compared to that located at the middle plane away from sidewall effects.

In addition, there was another noticeable difference in pressure profile between the sidewall-intersection and middle plane. There was only one pressure peak in the range $x/L = 0.85$ and $x/L = 0.95$, where separation and re-attachment shocks from the third separation should impinge on the surface. It is suspected that the third separation bubble on the cowl surface was diminished near the sidewall. This can be seen clearly from isolator pressure map provided in Figure 4-13.
4.2.2.1 Error Estimation

The error in pressure prediction using quadratic calibration curve given in Figure 4-11 was estimated by using Equation 3-3. The error was found to be $\varepsilon_{\text{in-situ}} = 0.023$, while the average error from all Kulite transducer for baseline case was found to be $\varepsilon_{\text{transducer}} = 0.004$. Therefore, the true static pressure (in unit bar) was considered to be within $\pm 0.027$ from those given by the quadratic PSP calibration curve.
4.2.3 Isolator Surface Pressure Map

The overall scramjet inlet-isolator surface static pressure map for baseline case is shown in Figure 4-13. The pressure map is produced using quadratic calibration curve obtained in Section 4.2.2. It gives a good insight into the three-dimensionality of the flow inside the inlet. From the figure above, shocks can be seen glancing on the surface originating from sidewall leading edge upstream of the expansion corner. These sidewall shocks were not very significant as their footprints did not extend into the isolator region, similar to the observation in Figure 4-6 (page 135). Intense pressure spot with magnitude almost 50 times of freestream pressure was caused by the boundary layer re-attachment downstream of shoulder. The observed streak lines in the pressure map were artificially caused by the finishing on the quartz cowl component. Further downstream, shock footprints could be
observed, showing the impingement point of separation and re-attachment shock from the third separation bubble. The spanwise line-marks made by both shocks were perfectly straight, except at small distance close to sidewalls, where they coalesced. The true reason for this observation is still unknown.

Figure 4-14 Spanwise pressure profile at entrance (x/L = 0.59), middle (x/L = 0.78) and exit of isolator (x/L = 0.99)

The three-dimensionality of the inlet flow structures was further examined using spanwise pressure plot such as in Figure 4-14. The profiles are taken at three position of x/L = 0.59 (isolator entrance), x/L = 0.78 (middle of isolator) and x/L = 0.99 (just upstream of isolator exit). Most of the spanwise pressure fluctuations noises were due to manufacturing finish of quartz cowl component, which has wavy refraction properties in spanwise direction.

At the isolator entrance, the average normalized pressure was 12.18, with standard deviation of 0.66, rendering it to be the largest among the three locations. The main source of deviation to mean pressure value was the dramatic rise of pressure close to sidewall. The
rate of pressure rise in spanwise direction at the isolator entrance ($x/L = 0.59$) was hypothesized to be influenced by the rate of shoulder separation shock angle increase.

The high average pressure at $x/L = 0.78$ was influenced by the shoulder re-attachment shock just upstream of it. The lowest average normalized pressure was 10.37 measured spanwise at $x/L = 0.99$. This location also experienced the lowest spanwise pressure standard deviation of 0.33, which was half the value at the entrance. This finding demonstrates that the three-dimensional effects at isolator exit were minimal and the isolator had sufficient length to reduce large flow non-uniformities at throat. The small non-uniformity level near the isolator exit suggested that the pressure predicted using global measurement techniques on the sidewall surface would stay within 3% of the value obtained from middle plane. Such level was satisfactorily small and justified the assumption of two-dimensional overall flow.

### 4.2.4 Sidewall Plane Pressure Map

![Figure 4-15 Sidewall plane pressure contour layered on top of schlieren image for baseline case](image)

The sidewall plane pressure map contour for the scramjet inlet-isolator in baseline test condition is provided in Figure 4-15. The pressure contour was produced by first calibrating the measured intensity with static pressure. As there was no pressure tapping on sidewall surface, the intensity along the bottom line was calibrated against the static pressure at sidewall intersection taken from Figure 4-13 (page 146). The calibration curve had quite a
good fit to pressure data with $R^2$ value of 0.929. By using the linear calibration equation, Figure 4-15 was produced and its pressure distribution was normalized to freestream static pressure.

The pressure contours were imposed on grayscale schlieren image for comparison and flow phenomena identification. It is known that a schlieren image shows a spanwise integrated contour of density gradient, thus flow features observed with schlieren would not correspond exactly to phenomena identified with pressure map on sidewall plane. The isolator sidewall pressure map is an important tool in scramjet inlet analysis and has been used as an estimator for internal drag from which other flow properties can be derived.

This thesis introduces a new and simpler method to calculate flow Mach number utilising only static pressure increase ratio and shock wave angle.

From Figure 4-15, shock waves can be identified as steep pressure gradient where contour lines coalesced very closely together and the pressure downstream of it increased. At least five shocks were located and labelled. The first shock (i) was the cowl tip shock, followed by second shock (ii) located a small distance downstream. The second shock (ii) was thought to be related to the re-attachment at the cowl tip separation. The pressure imposed on the cowl inner surface by this re-attachment shock was very severe, with magnitude of about 100 times the freestream, which is twice the peak pressure caused by shoulder re-attachment shock. As the separation at cowl was induced by the shoulder separation shock, eliminating shoulder separation would consequently maintain the structural integrity of both cowl and isolator.

Shock (iii) was recognized as the re-attachment shock of shoulder separation bubble. It is interesting to note that separation shock was not detected in Figure 4-15. Shock (iv) was
located at the position of third separation bubble inside the isolator. As has been discussed previously, shoulder re-attachment shock impinges on the top isolator surface and induces a third separation bubble (see Figure 4-8, page 137). However, from Figure 4-15, it could not be concluded whether shock (iv) was a separation, re-attachment or just an oblique reflection shock. This observation added weight to the suspicion that the third separation bubble diminished near sidewall surface.

Shock (v) was an oblique reflection of shock (iv) and has lower pressure gradient compared to other shocks, hence making it more difficult to be detected in both schlieren (see Figure 4-8, page 137) and pressure contour map (see Figure 4-15, page 148). This very weak shock was approaching the Mach wave limit in terms of its behaviour, and exerted only subtle changes on the flow properties across it. The control volume around this final shock was considered as the interrogation window for Mach number estimation discussed in Section 4.2.6.

4.2.5 Inlet-isolator Exit Static Pressure and Distortion Index

There are several different inlet distortion computational methods that utilise different flow properties such as pressure, total pressure, and Mach number. However, Shukla et al.\textsuperscript{191} has shown that distortion index calculated by using static pressure is more meaningful and could be used for performance optimization objective. In the study, they inspected static pressure profile along inlet height at throat station and calculated the distortion. However, as has been explained by Emami et al.\textsuperscript{33}, the geometrical throat (inlet shoulder) and aerodynamic throat (axial station of the highest wall static pressure peak) locations do not always coincide. This is also supported by our observation, where the pressure peak location was
found to be quite far downstream of shoulder (see Figure 4-10, page 141). Hence, a static
pressure profile at isolator exit was employed to calculate the overall inlet-isolator distortion
index by using the formula given by Shukla et al.\textsuperscript{191}:

\begin{equation}
\sigma_p = \frac{1}{\bar{p}_e} \left[ \frac{1}{H} \int_0^H (p_e(y) - \bar{p}_e)^2 dy \right]^{1/2}
\end{equation}

Here, H and $\bar{p}_e$ denote the height of the isolator and mean isolator exit pressure,
respectively. Employing the pressure profile in Figure 4-15 (see page 148), the mean
pressure (normalized to freestream) and distortion index were found to be $\bar{p}_e = 17.70$ and
$\sigma_p = 0.126$. Since the mean of isolator exit pressure was already normalized to freestream
static pressure, it was also taken to represent the one-dimensional inlet-isolator
compression ratio $\tilde{\mathcal{C}}$. From Smart\textsuperscript{192}, it was found that scramjet inlet usually performs best if
$\tilde{\mathcal{C}}$ is between 50 and 100 regardless of Mach number. With compression ratio of less than 18,
this current baseline case could definitely achieve higher compression if contraction ratio
was increased beyond Kantrowitz limit\textsuperscript{80}.

4.2.6 Inlet-isolator Exit Mach Number and Performance Estimation

![Figure 4-16 Schematic of typical internal shock structure resulted from interactions between cowl shock and expansion wave](image)
The shock structure recorded by the pressure map in Figure 4-15 was simplified and shown in Figure 4-16. The control volume around the final shock was considered and the flow properties upstream and downstream of the shock were determined by oblique shock relations. The final shock was assumed to be weak and approaching Mach wave limit; thus the flow direction upstream, $\phi_u$, and downstream, $\phi_u$, of the shock can be taken as parallel to the horizontal. This assumption allowed for the shock angle to be measured directly from the horizontal axis. This assumption would be proven to be correct if the calculated flow deflection, $\theta$, is very small and negligible. Another important assumption to be made is that the flow inside the control volume must be two-dimensional, where the flow properties on the middle plane are similar to flow properties at sidewall. This assumption has been proved valid by the discussion in Section 4.2.3 (page 146).

From the interrogation window in Figure 4-15 (page 148), the final shock angle was estimated to be about $\beta = 28.09^\circ$. The average pressure increase ratio taken along the shock line was found to be $\frac{p_d}{p_u} = 1.025$, where $p_d$ and $p_u$ are static pressure downstream and upstream of the shock wave, respectively. This is the first proof that the weak shock wave assumption is valid since the pressure increase across the shock is almost unity.

From any standard compressible aerodynamics textbook, oblique shock relations are found:

**Equation 4-2**

$$\tan\theta = 2\cot\beta \frac{M_u^2 \sin^2 \beta - 1}{M_u^2(\gamma + \cos 2\beta) + 2}$$

**Equation 4-3**

$$\frac{p_d}{p_u} = 1 + \frac{2\gamma}{\gamma + 1} [M_u^2 \sin^2 \beta - 1]$$
Inserting Equation 4-3 into Equation 4-2, the flow deflection angle was found to be $\theta = 0.42^\circ$. This is consistent with our assumption of weak shock and negligible flow turning angle.

Inserting Equation 4-3 into Equation 4-4, Mach number downstream of the final shock was found to be $M_d = 2.13$. This value was taken as representative of one-dimensional Mach number at isolator exit $M_e$, which was comparable to the averaged one-dimensional value predicted by CFD, $M_e^{(CFD)} = 2.01$. Thus, the difference of PSP calculated Mach number to that of CFD is about 6%. Typically, there would be quite significant difference between CFD and other scramjet flow diagnostic methods. For example, in the paper by Haberle and Gulhan $^{73}$, the difference between CFD and pitot rake analysis in predicting isolator exit Mach number is as high as 12%. Similarly, in her thesis about computational scramjet analysis $^{151}$, Bosco found that pitot rake predicted Mach number would differ as much as 16.9% from CFD results. Stream thrust analysis also suffers from large uncertainty as shown by Wie $^{31}$, where the isolator Mach number differed 9.7% from CFD. These differences were not so much related to the imperfection of CFD codes used in those cited papers $^{73,151,31}$, since their CFD scheme has been validated with solid experimental data of streamwise pressure profile. Thus, it must be concluded that the difference in calculated isolator exit Mach number between CFD and pitot rake analysis, or between CFD and stream thrust analysis, must be due to the associated error inherent in those two techniques. Similarly, in this current study, the CFD scheme has been validated against streamwise pressure profile. Hence, the CFD
scheme has been considered correct and was taken as the basis to judge the accuracy of the Mach number calculated using PSP pressure map.

As the oblique shock relations break down at the Mach wave, a validation must be made to ensure that the measured shock angle $\beta$ is not lower than Mach angle. To check, first $M_u$ was calculated using Equation 4-3, and then Mach angle $\mu$ was calculated using:

Equation 4-5

$$\mu = \arcsin \left(\frac{1}{M_u}\right)$$

The Mach angle was found to be $\mu = 27.78^\circ$, which was lower than $\beta$, hence confirming the validity of using oblique shock relations for estimation of $M_d$. Closeness of magnitude of $\beta$ and $\mu$ shows that the final shock was very weak and had almost similar behaviour to a Mach wave, but it still can be described by oblique shock relations.

Utilising the calculated $M_e$ and the average one-dimensional static pressure $\bar{p}_e$ from Section 4.2.5, isolator exit temperature and total pressure can be estimated by using isentropic flow equations below:

Equation 4-6

$$\frac{p_e}{\bar{p}_e} = \left(1 + \frac{\gamma - 1}{2} M_e^2\right)^{-\gamma / \gamma - 1}$$

Equation 4-7

$$\frac{p_e}{\bar{p}_e} = \left(\frac{T_e}{\bar{T}_e}\right)^{\gamma / \gamma - 1}$$
The flow was assumed adiabatic with no loss in total temperature from freestream to isolator exit. All calculated flow properties and performance indicators are summarised in Table 4-5 below.

<table>
<thead>
<tr>
<th>$M_e$</th>
<th>$M_e^{(CFD)}$</th>
<th>$p_e$ (bar)</th>
<th>$p_{te}$ (bar)</th>
<th>$T_e$ (K)</th>
<th>$\bar{C}$</th>
<th>$\sigma_p$</th>
<th>$\pi_c$</th>
<th>$\eta_{KE(ad)}$</th>
<th>$\Delta s/c_p$</th>
<th>$\eta_{c(ad)}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.131 ± 9.8%</td>
<td>2.01</td>
<td>0.216</td>
<td>2.072</td>
<td>194.92</td>
<td>17.7</td>
<td>0.126</td>
<td>0.321</td>
<td>0.923</td>
<td>0.324</td>
<td>0.821</td>
</tr>
</tbody>
</table>

The low value of total pressure ratio demonstrates the extent of total pressure loss due to unwanted viscous and shock-shock interactions. The flow also has quite significant level of flow distortion with static pressure along vertical axis at isolator exit has about 12.6% deviation from mean value. The scramjet inlet-isolator also achieved poor overall adiabatic compression efficiency of $\eta_{c(ad)} = 0.82$, where theoretically, an inlet with three oblique compression shocks such as this current model could achieve $\eta_{c(ad)}$ close to 0.9. It must be noted that in real world conditions, the compression process would not be adiabatic, thus the cited $\eta_{c(ad)}$ must be taken as the upper limit of what the real performance is.
It is well known that there is a correlation between throat Mach number and adiabatic kinetic energy efficiency\(^{18,21}\), and four most common relations are provided in Figure 4-17. The empirical relations are taken from Waltrup et al.\(^{193}\), Billig et al.\(^{194}\), Tani et al.\(^{195}\) and Smart\(^{192}\). The current model was compared with the three empirical plots and was found to lie very close to the curve postulated by Tani et al.\(^{195}\). More experimental point plots from parametric studies will be added later to determine a better empirical model.

4.3 Conclusions

The subject of the study is a scramjet inlet-isolator which has been designed to achieve shock-on-lip condition in Mach 5 inviscid flow. However, in real life viscous flow conditions, the inlet is not generally expected to achieve zero mass flow spillage. The inlet was tested in Mach 5 flow, and viscous effects were found to only minimally displace the compression shocks. Thus, the inlet operates closely to the ideal condition of minimum mass flow spillage.
Separation at compression corner has been studied by using Kulite pressure transducers, pressure sensitive paint and numerical simulation. PSP paint mixture utilising PtTFPP as luminophore was found to be more accurate since its pressure readings had lower spatial noise compared to using Ru(II) mixture. Nevertheless, the Ru(II) paint has the ability to visualize streamwise vortices. This is quite impressive, considering that published literatures\textsuperscript{75–78,117,147,184,185} on this field usually employ heat flux measurement methods such as TSP and infrared thermography for vortices detection.

Good visualization of internal shocks structure has been achieved by colour schlieren method, where identification of different flow phenomena was made easier with different colour scheme. The shocks structure existed solely because of the interaction between cowl tip shock and shoulder expansion wave in viscous flow. Large shoulder separation bubble induced two more separations on top cowl surface.

The PSP has been applied successfully to analyse the isolator surface pressure distribution. This method fills a lot of the gap left by the very limited pressure transducer. It shows exactly the location of peak pressure due to boundary layer re-attachment. Furthermore, this method also acts as a basis for more convincing CFD validation. Generally, the CFD code currently employed is able to closely match the schlieren and PSP pressure distribution. This is a valuable advantage since the CFD code is two-dimensional and requires only minimal setup time. Rapid parametric numerical study of the inlet could be run prior to or as a complement to experimental investigations as it incurs only low computational cost.

For the in-situ calibration of PSP using overall inlet-isolator surface, quadratic curve seems to be better than linear curve in fitting discreet pressure data. The inlet-isolator could be thought as two separate components possessing two different average surface
temperatures, thus necessitating the use of two linear calibration curves together. A quadratic curve is a product of two linear equations.

PSP of isolator surface enables an inlet designer to assess the three-dimensionality inherent in the isolator section. Typically, the flow inside the isolator could be safely assumed to be two-dimensional, while three-dimensionality effects are usually contained in small area close to sidewall. Spanwise flow distortion was significantly lower at isolator exit in comparison to the entrance.

A simple method of calculating isolator exit Mach number was successfully demonstrated. The method employed oblique shock relations to analyse the final shock just upstream of the isolator exit. The shock was assumed to be very weak with its behaviour approaching the Mach wave. The assumption has been proven correct since the final shock deflects the flow with very small and negligible angle. With the validation of the assumptions, the Mach number can be estimated with high confidence. This method proves to be valuable since it does not require intrusive components or complex setup employed previously in published literatures. From Mach number estimation, complete set of performance indicators were calculated.

In conclusions, the global pressure measurement methods have been demonstrated to be able to fully characterize a scramjet inlet-isolator. The richness of information contained in the isolator pressure map has surpassed the previous state-of-the-art. The PSP methods are envisaged to be the method of choice for future scramjet inlet-isolator investigator.
CHAPTER 5 SCRAMJET INLET-ISOLATOR CHARACTERISTICS AT OFF-DESIGN CONDITIONS

This chapter discusses the behaviour of the scramjet inlet-isolator model tested under a variety of conditions in which it was not designed for. Angle-of-attack (AoA) was increased up to 4° to test flow behaviour on the external and internal part of the inlet-isolator. Cowl length, relative to shoulder horizontal axis location, was increased to simulate test conditions where the inlet over sped and external compression shock impinges inside the cowl section. The cowl was also made shorter to understand the change in performance, if any, of the inlet in case if the compression shocks impinge upstream miss the cowl tip.

5.1 Effects of Changes in AoA

Behaviour and performance of scramjet inlet-isolator at windward side of AoA = 2° and 4° were observed and compared to that of the baseline. All other freestream flow parameters such as unit Reynolds number, Mach number, stagnation temperature and pressure were similar to baseline test conditions. The test parameters are summarized in Table 5-1 below:

<table>
<thead>
<tr>
<th>Case</th>
<th>Mach Number</th>
<th>Reynolds Number</th>
<th>Stagnation Pressure</th>
<th>Stagnation Temperature</th>
<th>Cowl Geometry</th>
<th>AoA</th>
</tr>
</thead>
<tbody>
<tr>
<td>AoA-2</td>
<td>5 ± 0.4%</td>
<td>$13.2 \times 10^6 \text{ m}^{-1}$</td>
<td>6.5 ± 0.05 (bar)</td>
<td>375 ± 5 (K)</td>
<td>Base-cowl</td>
<td>2°</td>
</tr>
<tr>
<td>AoA-4</td>
<td>5 ± 0.4%</td>
<td>$13.2 \times 10^6 \text{ m}^{-1}$</td>
<td>6.5 ± 0.05 (bar)</td>
<td>375 ± 5 (K)</td>
<td>Base-cowl</td>
<td>4°</td>
</tr>
</tbody>
</table>
5.1.1 AoA Effects on Compression Corner Separation Size

From Figure 5-1, the size of the separation at compression corner decreased with every increase of AoA. The separation shock appeared “smeared” as the separation became more subtle in AoA-4 case (see Figure 5-1 (c) and Figure 5-1 (f)). Matsumura et al.\textsuperscript{77} reported that size of separation at compression ramp is proportional to unit Reynolds number. Thus, with an increase of inlet model AoA, the flow velocity downstream of leading edge ramp had diminished. This lower velocity consequently reduced the unit Reynolds number of the flow, thus reducing the separation size. Similar observation has been reported by Yang et al.\textsuperscript{117} recently.
5.1.2 AoA Effects on Compression Ramp Streamwise Vortices

From Figure 5-2, it can be noted that the heat transfer to ramp surface by the Goertler vortices became more intense with an increase in AoA, which is consistent with the observation made by Yang et al.\textsuperscript{117}. The root of the streamwise vortices also moved
upstream closer to compression corner with every addition in AoA, thus confirming the observation made by Ishiguro et al.\textsuperscript{186}. 
5.1.3 AoA Effects on Isolator Shock-structures

![Images of schlieren for baseline (a & b), AoA-2 (c & d), and AoA-4 (e & f)]

Figure 5-3 Comparison of experimental and numerical schlieren for baseline [(a) & (b)], AoA-2 [(c) & (d)] and AoA-4 [(e) & (f)] (angle-of-attack at windward).
From Figure 5-3, a few changes in isolator shock structures were observed as the inlet increased its AoA. For example, the cowl tip separation decreased in size with AoA. This is probably related to the fact that compression shock from ramp surface moved upstream away from cowl with an increase in AoA. Thus, cowl boundary layer in AoA-4 case (Figure 5-3 (e) and Figure 5-3 (f)) has not been weaken as much as baseline case (Figure 5-3 (a) and Figure 5-3 (b)), making it more resilient to separation.

Flow Mach number at isolator entrance decreased with an increase in AoA due to amplification on total compression. However, the flow turning angle of 22° was still needed for the flow to enter, regardless of AoA. Thus, it explains the formation of Mach stem that wa discovered experimentally in AoA-4 case (Figure 5-3 (e)). It must be noted that the numerical code could not simulate the Mach stem formation for the AoA-4 case (Figure 5-3 (f)). The Mach stem is part of Edney’s Type II shock-shock interaction. This shock-shock interaction is similar to Edney’s Type I interaction found in baseline case, where two shocks of different family (i.e. the shoulder separation shock and cowl tip shock) intersected each other but with such strength that a pocket of the flow must decelerate to subsonic to satisfy necessary flow turning angle. The natural mechanism of subsonic flow deceleration was shown by the formation of a Mach stem between the two interacting shocks. The subsonic pocket did not continue all the way downstream as reflections of oblique shocks could still be detected. Overall, AoA4-case suffered large flow distortion at aerodynamic throat area. Numerical code was not able to simulate the formation of Mach stem in AoA-4 case (see Figure 5-3 (f)).

The separation and re-attachment shock from the third separation bubble was observed to be much stronger at higher AoA when compared to that at baseline case. Their reflections in
AoA-2 and AoA-4 could be viewed more easily compared to that without AoA case. This finding suggests that the interrogation window for Mach number determination such as in Figure 4-15 (page 148) needs to move a little bit downstream.

5.1.4 Comparison of Scramjet Inlet-isolator Pressure Profile at Different AoA

The changes in shock structures discussed previously could be better apprehended by examining surface pressure plot in Figure 5-4. Pressure profiles of the whole inlet-isolator at different AoAs were measured experimentally and computed numerically, and plotted together for comparisons. All pressure sensitive measurements were calibrated in-situ with accompanying Kulite pressures. Calibration for both AoA-2 and AoA-4 cases used quadratic equation and had coefficient of determination $R^2$ value of 0.979 and 0.954, respectively. The
pressure predicted by PSP was correct to within ± 0.04 and ± 0.057 bar for AoA-2 and AoA-4 case, respectively.

The AoA-4 case has very large deviations in every Kulite readings on the compression ramp. Time-history of the transducers revealed significant static pressure oscillations. Some peculiarity has been remarked during the test campaign that if the model vibrates for any reason, wild pressure oscillation will be recorded. The main cause of model vibrations comes from loose connections in the sting holder. Certain measures have been taken to reduce the static pressure oscillation on the second ramp for AoA-4 case, but the errors were repeatable. Even when the model has been stopped from vibrating, the pressure recorded was still unsteady. There was a possibility that the pressure oscillation originated from the movement of separation and re-attachment shock around the first three transducers locations. Fast Fourier Transform analysis has been applied to every transducer on the compression ramps, but no significant frequency spike could be observed. Nevertheless, time-averaged pressures on second ramp were considered acceptable because of their closeness to CFD. Thus, only for this case, the first two Kulite readings on first ramp were not considered for PSP calibration process as their readings differ significantly with CFD.

For all three cases, similar patterns of pressure spike and drop could be observed. Sudden pressure jump in PSP-derived profile was always detected at the same location just upstream of isolator entrance, suggesting that shoulder separation points for all AoA were positioned at the same x-coordinate. However, CFD showed that with the increase of AoA, separation point moves upstream in small length increment. The shoulder separation length for all cases deduced from PSP and CFD showed roughly similar value. It is expected that the case with larger AoA would have better performance than baseline case since it could
achieved higher compression with smaller loss at shoulder separation. The Kulite’s pressure trends inside the isolator showed that the shoulder re-attachment pressure peak should be higher for larger AoA, but PSP patterns failed to concur. Overall, it could be concluded that Kulite, PSP and CFD have quite good pressure profile match between each other.

5.1.5 Comparison of Scramjet Inlet-isolator Pressure Map at Different AoA

The surface pressure map shown in Figure 5-5 provides a qualitative overview of the flow three-dimensionality at off-design conditions. All parts of the inlet experienced higher surface static pressure with an increase in AoA. The wave angle made by sidewall leading edge shocks remained roughly the same, unaffected by the increase in AoA, even though flow Mach number on the second ramp decreased. The line markings of separation and re-attachment shocks from the third separation bubble coalesced near the sidewall. This suggests that the separation bubble diminished close to sidewall for all AoA case. The size of the third separation bubble also decreased with an increase in AoA.
Figure 5-6 Spanwise pressure profile at entrance, middle and exit of isolator for AoA-2 case (angle-of-attack at windward)

Figure 5-7 Spanwise pressure profile at entrance, middle and exit of isolator for AoA-4 case (angle-of-attack at windward)
Table 5-2 Comparison of spanwise average and standard deviation of normalized pressure at entrance, middle and exit of isolator section

<table>
<thead>
<tr>
<th>Case</th>
<th>Avg pressure (S.Dev) at x/L=0.59</th>
<th>Avg pressure (S.Dev) at x/L=0.78</th>
<th>Avg pressure (S.Dev) at x/L=0.99</th>
</tr>
</thead>
<tbody>
<tr>
<td>Baseline</td>
<td>12.18 (0.66)</td>
<td>13.63 (0.60)</td>
<td>10.37 (0.33)</td>
</tr>
<tr>
<td>AoA-2</td>
<td>15.85 (0.81)</td>
<td>17.11 (0.89)</td>
<td>11.58 (0.54)</td>
</tr>
<tr>
<td>AoA-4</td>
<td>18.20 (0.87)</td>
<td>17.11 (0.99)</td>
<td>12.08 (0.98)</td>
</tr>
</tbody>
</table>

The spanwise normalized static pressure distributions inside the isolator for case AoA-2 and AoA-4 are shown in Figure 5-6 and Figure 5-7, respectively. Spanwise average and standard deviation of the static pressure for all cases are provided in Table 5-2. It was seen that case AoA-2 had similar three-dimensionality behaviour to baseline case, where the lowest spanwise standard deviation was located at isolator exit (x/L = 0.99), and the highest was at the entrance (x/L = 0.59). The inlet-isolator in case AoA-2 and AoA-4 experienced higher three-dimensionality effect shown by the increase in spanwise pressure standard deviation in comparison with that in the baseline case. It was also demonstrated that the AoA-4 case suffered the highest spanwise non-uniformity at isolator exit as its average static pressure has about 8% standard deviation.
5.1.6 Comparison of Scramjet Inlet-isolator Sidewall Pressure Map and Performance at Different AoA

The pressure maps given in Figure 5-8 depicted the traces of oblique shock reflections etched onto the sidewall as the inlet-isolator experienced changes in AoA. The pressure map contour for AoA-2 and AoA-4 case has been produced by first calibrating the intensity distribution on sidewall with measured sidewall intersection pressure from Figure 5-5 (page 167). Linear in-situ calibration equations were used for the three cases. The $R^2$ values of the calibration curves are 0.871 and 0.986 for AoA-2 and AoA-4 case, respectively. Low $R^2$ value for AoA-2 case increased the uncertainties of the magnitude of pressure contour plotted in
Figure 5-8 (b). Nevertheless, the shocks reflection pattern should remain the same regardless of the calibration error.

A shock was identified by close formation of pressure contour lines accompanied by pressure increment. It is interesting to note that with an increase in AoA, the number of contour lines associated with each shock decreased. This phenomenon occurred due to lower gap of high and low pressure region throughout the flowfield, suggesting higher level of uniformity for case with AoA.

In Figure 5-8, shock (i) was the mark made by cowl tip shock. The Mach stem connected to cowl tip shock found in the schlieren for case AoA-4 (see Figure 5-3 (e), page 163) was not detected by the pressure contours. Shock (ii) was the trace of re-attachment shock at cowl tip. Shock (iv), (v) and (vi) were the markings made by oblique reflections of shoulder re-attachment shock (iii). For all cases, the pressure gradient line traces of oblique shocks were always observed to be at larger angle (relative to horizontal) when compared to those measured from accompanying schlieren image. This could be seen clearly with shocks labelled (i) and (iv) for the three cases due to the appearance of “shock-envelope” associated with glancing oblique shock problem$^{190}$.

It was evident that the shock structures became more “compact”, with each shock having larger wave angle. Thus, the shock labelled (v) was not the final weak shock wave for AoA-2 and AoA-4 case. Interrogation window for Mach number determination must be placed more downstream to employ the final shock (vi) instead. The pressure increase ratio across shock (vi) for AoA-2 and AoA-4 were 1.075 and 1.016, respectively. The measured shock wave angles with respect to the horizontal axis were 28.97$^\circ$ and 34.35$^\circ$ for AoA-2 and AoA-4 case, respectively. Using calculation procedures demonstrated in Section 4.2.6 (page 151),...
Mach number and various performance indicators can be computed. Compression ratio $\tilde{\chi}$ and flow distortion $\sigma_p$ were also calculated from isolator exit pressure profile following the example demonstrated in Section 4.2.5 (page 150). Summary of all computed output are compiled in Table 5-3 below.

<table>
<thead>
<tr>
<th>Case</th>
<th>$M_e$</th>
<th>$M_{e(CFD)}$</th>
<th>$p_e$ (bar)</th>
<th>$p_{te}$ (bar)</th>
<th>$T_e$ (K)</th>
<th>$\tilde{\chi}$</th>
<th>$\sigma_p$</th>
<th>$\eta_{KE(ad)}$</th>
<th>$\Delta s/c_p$</th>
<th>$\eta_{c(ad)}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Baseline</td>
<td>2.13 ± 9.8%</td>
<td>2.01</td>
<td>0.216</td>
<td>2.072</td>
<td>194.92</td>
<td>17.7</td>
<td>0.126</td>
<td>0.321</td>
<td>0.923</td>
<td>0.324</td>
</tr>
<tr>
<td>AoA-2</td>
<td>2.08 ± 12.3%</td>
<td>2.05</td>
<td>0.332</td>
<td>2.980</td>
<td>198.65</td>
<td>27.2</td>
<td>0.153</td>
<td>0.46</td>
<td>0.951</td>
<td>0.221</td>
</tr>
<tr>
<td>AoA-4</td>
<td>1.77 ± 11.2%</td>
<td>1.84</td>
<td>0.472</td>
<td>2.615</td>
<td>228.10</td>
<td>38.7</td>
<td>0.046</td>
<td>0.405</td>
<td>0.941</td>
<td>0.258</td>
</tr>
</tbody>
</table>

From Table 5-3, the calculated isolator exit Mach numbers for AoA-2 and AoA-4 case matched closely with 2.05 and 1.84, respectively, predicted using CFD. Thus, the differences between CFD and PSP predicted Mach number were 1.9% and 3.6% for AoA-2 and AoA-4, respectively. For both baseline and AoA-2 case, Mach number was slightly under-predicted by CFD. On the other hand, the numerical scheme produced minor overprediction of Mach number for AoA-4 case because it cannot properly resolve for the Mach stem observed in schlieren image of Figure 5-3 (e) (page 163).

Improvements in vital aerodynamic performance indicators such as $\pi_c$, $\eta_{KE(ad)}$, and $\Delta s/c_p$ could be detected as the AoA was increased from 0° to 2°. This could be related to the decrease of cowl tip separation size as the compression ramp shock impinging away from the
cowl tip. The improvements produced dramatic jump on compression system efficiency $\eta_{c(ad)}$ from 0.821 to 0.888, which is close to the hypothesized compression performance for three oblique shock inlet postulated in Heiser and Pratt$^{21}$. The achieved compression efficiency along with almost 54 % increase in compression ratio $\tilde{C}$ has earned AoA-2 case as the best condition among the three cases.

The inlet operating with $\text{AoA} = 4^\circ$ had slightly lower aerodynamic performance compared to that in AoA-2 case, but still significantly better than that in baseline. The severity of Mach stem appearance was shown by the dramatic increase of isolator exit static temperature of about 30 K from AoA-2 to AoA-4. The increment was considered very significant since temperature difference between AoA-2 and baseline was only about 4 K. The AoA-4 case was expected to achieve higher aerodynamic performance than AoA-2 case if the Mach stem at isolator entrance could be eliminated. This is discussed in Section 6.3 (page 210), where variable geometry methods employed to optimize AoA-4 case were investigated. Nevertheless, AoA-4 case managed to have very slight overall compression system efficiency increase to $\eta_{c(ad)} = 0.89$.

### 5.2 Effects of Changes in Cowl Length

To study the effects of changes in cowl length, the cowl tip horizontal position was changed relative to expansion corner location. In one case, the tip position was moved upstream by 3 mm from base-cowl original design to allow for ingestion of compression ramp shock wave inside the inlet. In this case, the total cowl length upstream of geometric throat (shoulder) is 5 mm to simulate the inlet wave interactions that occur when an inlet operate at higher than the design Mach number. This case is termed long-cowl in the discussions to follow.
In another case, the cowl was made shorter by 3 mm from base-cowl to investigate the characteristics of the inlet if compression ramp shocks coalesced upstream and missed the cowl tip. This shock formation is similar to when an inlet operates at lower than the design Mach number. Analogous shock system could also be found when a scramjet inlet is deliberately shaped for a small spillage to ensure its startability. This case is labelled as short-cowl case in the following discussions. The summary of test case definitions is provided in Table 5-4 below.

Table 5-4 Parameters for long-cowl and short-cowl test conditions

<table>
<thead>
<tr>
<th>Test Name</th>
<th>Mach Number</th>
<th>Reynolds Number</th>
<th>Stagnation Pressure</th>
<th>Stagnation Temperature</th>
<th>Cowl Geometry</th>
<th>AoA</th>
</tr>
</thead>
<tbody>
<tr>
<td>Long-cowl</td>
<td>5 ± 0.4%</td>
<td>1.2 × 10^6 m^1</td>
<td>6.5 ± 0.05 (Bar)</td>
<td>375 ± 5 (K)</td>
<td>Long-cowl</td>
<td>0°</td>
</tr>
<tr>
<td>Short-cowl</td>
<td>5 ± 0.4%</td>
<td>1.2 × 10^6 m^1</td>
<td>6.5 ± 0.05 (Bar)</td>
<td>375 ± 5 (K)</td>
<td>Short-cowl</td>
<td>0°</td>
</tr>
</tbody>
</table>
5.2.1 Cowl-length Effects on Isolator Shock-structures

Figure 5-9 Comparison of isolator experimental and numerical schlieren for baseline [(a) & (b)], long-cowl [(c) & (d)] and short-cowl [(e) & (f)] cases.
Using images from Figure 5-9, internal shock structures for baseline, long-cowl and short-cowl cases were examined and compared. Since the same compression ramp was used for all cases, the change in isolator flow field was only related to cowl lateral position. For the long-cowl case, the compression shock impinged inside the cowl surface, thereby inducing extra cowl tip separations (see Figure 5-9 (d)). The two small separations combined into one bigger separation as observed experimentally in Figure 5-9 (c). Effective isolator entrance size in this case was further reduced in comparison with that in the baseline. The re-attachment shock from shoulder separation induced third separation bubble similar to shock wave pattern in baseline case (see Figure 5-9 (a) and Figure 5-9 (b)).

The short-cowl case did not have any separation at cowl tip whether by experimental or numerical observation (see Figure 5-9 (e) & (f)). The reason for this is that there was no interaction between cowl boundary layer and ramp compression shocks that could weaken the boundary layer and made it susceptible to separate.

Separation onset points for baseline and long-cowl case were similar at just upstream of shoulder expansion corner. The re-attachment points between the two cases were also approximately comparable. Separation onset point for the short-cowl case was located exactly at shoulder.

Overall, there was no striking difference in shoulder separation length regardless of different cowl shock interactions. Also, it can be concluded that results from CFD matched the experimental observation with high similarity.
5.2.2 Comparison of Scramjet Inlet-isolator Pressure Profile with Different Cowl Length

Figure 5-10 Isolator surface centreline pressure profile for baseline, long-cowl and short-cowl cases

The Long-cowl and Short-cowl isolator pressure profile were compared with baseline in Figure 5-10. Only isolator section was considered since the same compression ramp was used for all three cases. The accuracy of PSP and CFD was compared to Kulite readings for all cases. The PSP camera view in long-cowl case had been blocked by aluminium cowl leading edge at coordinate $x/L = 0.6$ until $x/L = 0.71$. Similarly, the short-cowl case had obstructed view at coordinate $x/L = 0.64$ until $x/L = 0.76$. As the PSP camera was blocked, the calibration process could not consider the pressure readings at coordinate $x/L = 0.63$ and $x/L = 0.68$ in the long-cowl case and readings at $x/L = 0.68$ and $x/L = 0.73$ in the short-cowl case. The reduce number of points in Stern-Volmer calibration curve has permitted for the use of
linear calibration curve. The resultant $R^2$ values for the long-cowl and short-cowl calibrations were 0.998 and 0.999, respectively. The pressure predicted by PSP is correct to within ± 0.018 and ± 0.006 bar for long-cowl and short-cowl case, respectively.

The long-cowl and short-cowl case pressure readings suffer from streamwise noise. The baseline pressure profile seemed smoother compared to the other two cases. This is due to different paint smoothness level between all cases.

The CFD filled the gap in predicting shoulder re-attachment peak pressure which PSP could not provide due to obstruction of observation window for the long-cowl and short-cowl cases. According to CFD, the peak pressure for the long-cowl case was almost similar to baseline case. This finding is related to comparable separation size between the two cases. On the other hand, short-cowl case has lower peak pressure with 20% decrement from baseline.
5.2.3 Comparison of Scramjet Inlet-isolator Pressure Map with Different Cowl Length

Figure 5-11 Comparison of isolator surface pressure map for (a) baseline, (b) short-cowl and (c) long-cowl case

All changes in three-dimensional effects on the isolator flow are shown in Figure 5-11. The analysis was limited by the obstruction of view by cowl aluminium frame for long-cowl and short-cowl case. It was observed that only 57% of isolator section width was visible for both
cases impeding the observation of three-dimensional effects near sidewall. The dash white lines in the figures denote the shock footprints made by separation and re-attachment shock from third separation. For the short-cowl case (Figure 5-11 (b)), the lines were almost straight similar to baseline case (Figure 5-11 (a)). For the Long-cowl case (Figure 5-11 (c)), the two lines curved significantly. This suggests that the large separation at cowl tip is highly three-dimensional and affects the separation downstream.
5.2.4 Comparison of Scramjet Inlet-isolator Sidewall Pressure Map and Performance at Different Cowl Length

Calibrating sidewall pressure contour proves to be challenging since isolator surface pressure maps for both long-cowl and short-cowl case in Figure 5-11 (b) and Figure 5-11 (c) did not allow for pressure profile measurement at sidewall intersection due to their small top window size. As such, the pressure profile at most extreme spanwise location permitted by the top window was used instead. Satisfactory $R^2$ values of 0.968 and 0.913 were obtained when linear calibration curves were used for both cases.
Interesting observations can be made from the comparison of sidewall static pressure contour shown in Figure 5-12. Region of high pressure, which is usually associated with re-attachment of cowl tip separation, was very large for long-cowl case (Figure 5-12 (b)). The re-attachment shock itself could not be detected, unlike the observation seen in Figure 5-12 (a) for baseline case. For all cases, pressure gradient line traces of oblique shocks were always at larger angle (relative to horizontal) if compared to accompanying schlieren image. This could be seen clearly with shocks labelled (i) and (iv) in Figure 5-12 (a), labelled (i) and (iii) in Figure 5-12 (b), and (i) and (iv) in Figure 5-12 (c). The appearance of “shock-envelope” due to glancing shock wave-boundary layer interaction was thought as the reason for this observation. Overall, shape of oblique shocks reflections and locations of shocks impingement did not vary much between the three cases, thus allowing for similar position of interrogation window to calculate Mach number. Calculated Mach number and all other properties are summarized in Table 5-5 below.

Table 5-5 Comparison of calculated flow parameters and performance indicators for baseline, long-cowl and short-cowl cases

<table>
<thead>
<tr>
<th>Case</th>
<th>$M_e$</th>
<th>$M_{e(CFD)}$</th>
<th>$p_e$</th>
<th>$p_{te}$</th>
<th>$T_e$ (K)</th>
<th>$\dot{c}$</th>
<th>$\sigma_p$</th>
<th>$\pi_c$</th>
<th>$\eta_{KE(ad)}$</th>
<th>$\Delta s/c_s$</th>
<th>$\eta_{c(ad)}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Baseline</td>
<td>2.13 ± 9.8%</td>
<td>2.01</td>
<td>0.216</td>
<td>2.072</td>
<td>194.92</td>
<td>17.7</td>
<td>0.126</td>
<td>0.321</td>
<td>0.923</td>
<td>0.324</td>
<td>0.821</td>
</tr>
<tr>
<td>Long-cowl</td>
<td>1.54 ± 10.5%</td>
<td>1.97</td>
<td>0.233</td>
<td>0.904</td>
<td>252.46</td>
<td>19.1</td>
<td>0.123</td>
<td>0.140</td>
<td>0.849</td>
<td>0.562</td>
<td>0.755</td>
</tr>
<tr>
<td>Short-cowl</td>
<td>2.07 ± 9.3%</td>
<td>2.17</td>
<td>0.249</td>
<td>2.174</td>
<td>200.15</td>
<td>20.4</td>
<td>0.046</td>
<td>0.337</td>
<td>0.927</td>
<td>0.311</td>
<td>0.836</td>
</tr>
</tbody>
</table>
From Table 5-5, the estimated one-dimensional isolator exit Mach number for long-cowl case was very low, at just 1.54. This severe decrement from $M_e = 2.13$ in baseline case is most probably due to the extra shock interactions at cowl tip separation. Impingements of compression shock inside the cowl surface prevented spillage and caused more unwanted shocks to appear inside the isolator region. The extra shock promoted modest amount of compression ratio increase in comparison with baseline case. Nonetheless, the extra shock also simultaneously caused larger total pressure loss, which explains the reason for poor total ratio efficiency of $\pi_c = 0.14$. The extra shock interactions also resulted in severe temperature increase of almost 60 K from baseline. The combined effects of low total pressure recovery and high temperature increase led to low adiabatic kinetic energy efficiency and high entropy increase. Ultimately, the long-cowl inlet suffered low compression system efficiency. It must be noted that CFD predicted a higher isolator exit Mach number for long-cowl, with $M_{e(CFD)} = 1.97$. Such large discrepancy (22% from CFD) was due to the weakness in CFD to properly model the large separation region at cowl tip. The large difference could also be attributed to the fact that the simulation was done in two-dimensional, while the sidewall pressure pattern used in calculating Mach number suffers from strong three-dimensional effect as shown in Figure 5-11 (c).

The short-cowl case, on the other hand, has comparable flow parameters and performance with baseline. It could be seen that the value was either slightly higher or lower on every category. For example, the total pressure ratio for short-cowl was 0.337, which was not very significant increase from baseline. The difference between the two cases was that the former lacked cowl tip separation which occurs in the latter, whilst all other shock interactions remained almost the same. The slight gain in $\pi_c$ means that the flow was more
efficient in converting total pressure into static pressure. It was observed that an inlet could maintain a good performance, even when a small spillage was allowed. The only drawback for allowing small inlet spillage is obviously the lower mass capture ratio, which could translate into lower thrust. For short-cowl case, the difference between CFD and PSP predicted Mach number was less than 5%.

5.3 Conclusions

The scramjet inlet-isolator was subjected to different AoA to test its robustness in producing optimum flow for combustion. The inlet was designed to achieve shock-on-lip condition in inviscid Mach 5 flow without any AoA. The introduction of AoA to inlet operation was found to improve the overall flowfield and associated performance. For example, the compression corner separation reduced in size with every increase in AoA. Shoulder separation size also diminished with AoA increment. This results in higher improvement of adiabatic compression efficiency, kinetic energy efficiency and entropy increase. However, at AoA-4 case, a Mach stem appeared and reduced the performance that could be obtained with an increase in AoA. Overall, AoA-2 case had the best performance per compression ratio.

For Baseline and AoA-2 cases, the lowest spanwise distortion was located at isolator exit, demonstrating the effectiveness of the isolator in regulating the flow uniformity suitable for combustion. The AoA-4 case maintained high spanwise distortion throughout the isolator due to its complex shock-shock and shock-boundary layer interactions at throat area. The interactions were highly three-dimensional. The overall spanwise distortion also increased with AoA.
The performance of the inlet-isolator also decreased dramatically when the compression shocks impinged inside the cowl. This kind of shock interactions was similar to those experienced by an inlet operating at larger-than-designed Mach number. Large total pressure loss was exerted onto the flow from the complex shock-shock and shock-boundary layer interactions near cowl tip. The short-cowl case was designed for its cowl to miss the compression shock to simulate an inlet operating at lower-than-designed Mach number. It achieved better aerodynamic performance than baseline case, even though it suffered with mass flow spillage. Its high compression ratio could compensate the low thrust resulted from low mass flow rate.

CFD generally matched the experimental observation qualitatively and quantitatively, except for the AoA-4 case, where it could not simulate the Mach stem at isolator entrance. Nevertheless, the CFD is still important in the characterization of scramjet inlet-isolator as it could fill the gap left by obstruction in PSP readings. The Kulite transducer, PSP and CFD complemented each other in helping an inlet designer to understand its characteristics.
CHAPTER 6 SCRAMJET INLET-ISOLATOR OPTIMIZATION

The investigation methodology employed during this test campaign has given comprehensive understanding of the generic scramjet inlet-isolator behaviour at off-design conditions. The PSP technique when applied on compression ramp surface has allowed for global pressure measurement, which can fill the gap between discreetly positioned pressure transducer. The three-dimensionality of the scramjet inlet, especially inside the isolator where sidewall effect is very prevalent, has also been studied thoroughly. On the other hand, the same technique when applied onto the sidewall, has opened door for flow properties calculation without the need for intrusive measurement device. Thus, from all the analysis done in previous sections, it could be recognized that the inlet is operating sub-optimally even at designed condition. The large shoulder separation is too strong and shocks emanating from the separation and re-attachment point induce more separations at their impingement locations.

Strategies to mitigate the shoulder separation problem are discussed in this section. The first strategy is carried out by deflecting the cowl inward to decrease the cowl shock strength and reducing the extent of separation. Das and Prasad\textsuperscript{59,68,69} have compared the effect of inward cowl deflection with the benefits of using boundary layer bleed for supersonic inlet, so Section 6.1 (page 187) will extend the investigation of cowl deflection to hypersonic inlet. The second mechanism is done by utilising micro-vortex generator (MVG) on the second ramp which is similar in principle to the strategy used by Reinartz \textit{et al.}\textsuperscript{60}.

It has also been found from Section 5.1.6 (page 170) that performance of the inlet improved considerably with increasing AoA, but inlet operating at too large AoA experienced spillage
and severe aerodynamic loss from formation of Mach stem at isolator entrance. In Section 6.3 (page 210), variable geometry inlet was employed to optimize the inlet flow at \( \text{AoA} = 4^\circ \).

The inlet-isolator flows were characterized using similar analysis done in previous sections.

### 6.1 Effectiveness of Cowl Deflection to Control Shoulder Separation

According to the author’s knowledge, there is serious lack of study that focuses on the applicability of cowl deflection for boundary layer control in scramjet inlet flow environment. In this study, cowls with deflection of \( 3^\circ \) and \( 5^\circ \) to the horizontal were studied and compared with baseline case. All flow parameters for each cowl deflection case are summarized in Table 6-1 below.

#### Table 6-1 Case definition

<table>
<thead>
<tr>
<th>Test Name</th>
<th>Mach Number</th>
<th>Reynolds Number</th>
<th>Stagnation Pressure</th>
<th>Stagnation Temperature</th>
<th>Cowl Geometry</th>
<th>AoA</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cowl-3</td>
<td>5 ( \pm 0.4% )</td>
<td>( 13.2 \times 10^6 ) m (^2) s (^{-1})</td>
<td>6.5 ( \pm 0.05 ) (bar)</td>
<td>375 ( \pm 5 ) (K)</td>
<td>3deg-cowl</td>
<td>0°</td>
</tr>
<tr>
<td>Cowl-5</td>
<td>5 ( \pm 0.4% )</td>
<td>( 13.2 \times 10^6 ) m (^2) s (^{-1})</td>
<td>6.5 ( \pm 0.05 ) (bar)</td>
<td>375 ( \pm 5 ) (K)</td>
<td>5deg-cowl</td>
<td>0°</td>
</tr>
</tbody>
</table>
6.1.1 Cowl Deflection Effects on Isolator Shock-structures

Figure 6-1 Comparison of experimental and numerical schlieren for baseline [(a) and (b)], Cowl-3 [(c) and (d)] and Cowl-5 [(e) and (f)] cases
From Figure 6-1, it was observed that the height of shoulder separation decreased with an increase in cowl deflection. This will ease up the passage of flow into isolator and reduce the probability of unstart. Inward cowl bend decreased the total turning angle the flow need to go through to enter the isolator section, thus the cowl shock would have lower strength. For both deflected cowl cases, no cowl tip separation could be observed.

For Cowl-5 case (Figure 6-1 (e) and (f)), the shock from shoulder separation was replaced by an expansion wave. The height of the shoulder separation bubble in this case reduced substantially, that the flow was turning away from itself in order to navigate around the bubble. Total pressure loss that was caused by shock-shock interaction at shoulder region was eliminated in this case, contributing to better performance.

Figure 6-2 Distance of separation onset point upstream of shoulder according to experimental and numerical schlieren

Movement of separation onset point with cowl deflection was tracked by measuring its distance relative to shoulder position using experimental and numerical schlieren images. The distance was normalised with length of isolator and plotted in Figure 6-2. For both numerical and experimental observations, the separation onset point moves downstream
closer to the expansion corner. With zero deflection, the separation onset point predicted numerically was very close to that observed experimentally. Nonetheless, for Cowl-3 case, the separation distance upstream of expansion corner was under-predicted by the simulation. Separation onset points for both experimental and numerical schlieren seem to lie exactly at shoulder expansion corner with 5° cowl deflection. The observed downstream movement of separation onset point was similar to the findings reported by Das and Prasad\textsuperscript{68}.

6.1.2 Comparison of Scramjet Inlet-isolator Pressure Profile with Different Cowl Deflection

![Isolator surface pressure profile for baseline, Cowl-3 and Cowl-5 cases](image)

Figure 6-3 Isolator surface pressure profile for baseline, Cowl-3 and Cowl-5 cases
Discussions of isolator shock structures in Section 6.1.1 are complemented by studying the accompanying surface pressure profiles given in Figure 6-3. Due to the cowl leading edge obstructing PSP measurement at $x/L = 0.615$ to $x/L = 0.727$ for both Cowl-3 and Cowl-5 cases, CFD provided the missing information on the pressure trends. Numerical data fitted closely the discreet Kulite readings inside that coordinate range. Good fit between PSP and Kulite was also observed. The $R^2$ values for both Cowl-3 and Cowl-5 cases intensity-pressure calibration were 0.998 and 0.997, respectively. The pressure predicted by PSP is correct to within $\pm 0.016$ and $\pm 0.017$ bar for Cowl-3 and Cowl-5 case, respectively.

Most inconsistencies between CFD and PSP pressure readings were located downstream of $x/L = 0.85$, where separation and re-attachment shock from top cowl surface impinges on isolator surface. Nevertheless, the peak pressure due to shoulder re-attachment predicted by CFD agreed very well with PSP.

Pressure profile in Figure 6-3 confirmed the observed increment in separation length with cowl deflection seen in schlieren image of Figure 6-1 (page 188). The magnitude of re-attachment peak pressure for Cowl-5 case was observed to decrease by half from that of baseline case. Even though the strength of shoulder re-attachment shock has been reduced, it was still strong enough to induce separation on the opposite surface.
6.1.3 Comparison of Scramjet Inlet-isolator Pressure Map with Different Cowl Deflection Angle

The spanwise flow features inside the isolator with different cowl deflection angle were visualized and compared together in Figure 6-4. The observation was obscured by the cowl frame that holds the Perspex window, allowing view of only 57% of middle isolator span.

Figure 6-4 Comparison of isolator surface pressure map for (a) baseline, (b) Cowl-3 and (c) Cowl-5 cases
From $0^\circ$ to $3^\circ$ deflections, no significant change in shock trace lines could be detected. The Cowl-5 case in Figure 6-4 (c) had large three-dimensionality effect shown by the curvy and almost double arch-like shock line trace. Large three-dimensionality effects present in Cowl-5 case was thought to cause the significant difference in pressure profile by PSP and CFD as discussed in Section 6.1.2.

6.1.4 Comparison of Scramjet Inlet-isolator Sidewall Pressure Map and Performance at Different Cowl Angle

Figure 6-5 Comparison of sidewall pressure contour for (a) baseline, (b) Cowl-3 and (c) Cowl-5 cases
The pressure contours of isolator sidewall for three different cases of cowl deflection angle are compared in Figure 6-5. As the frame of top windows in Figure 6-4 (b) and (c) blocked the isolator surface pressure measurement at sidewall intersection, the pressure profile at the edge of visible pressure map window was used to calibrate sidewall PSP. The calibration curve had surprisingly high value of $R^2$ at 0.996 (Cowl-3 case) and 0.924 (Cowl-5 case).

Unlike the baseline case, there was only one oblique shock emanating from cowl, which is labelled as (i) in both Cowl-3 (Figure 6-5 (b)) and Cowl-5 (Figure 6-5 (c)). Shock (i) impinged on isolator surface and reflected as shock (ii), (iii) and (iv) downstream. With less number of shock, improvements in performance were expected for Cowl-3 and Cowl-5 cases. Cowl leading edge experienced lower peak pressure when deflected. This was due to the elimination of cowl separation, thus saving it from very high peak pressure inflicted by boundary layer re-attachment.

By measuring final shock angle and pressure ratio, Mach and other flow properties were calculated and summarized in Table 6-2 below. Static pressure at isolator exit found by averaging the contour plot above was also included in the table below.
Table 6-2 Comparison of calculated flow properties for baseline, Cowl-3 and Cowl-5 cases

<table>
<thead>
<tr>
<th>Case</th>
<th>$M_e$</th>
<th>$M_e$ (CFD)</th>
<th>$\rho_e$ (bar)</th>
<th>$p_{te}$ (bar)</th>
<th>$T_e$ (K)</th>
<th>$c_e$</th>
<th>$\sigma_p$</th>
<th>$\pi_e$</th>
<th>$\eta_{KE(ad)}$</th>
<th>$\Delta s/c_p$</th>
<th>$\eta_{e(ad)}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Baseline</td>
<td>2.13 ±9.8%</td>
<td>2.01</td>
<td>0.216</td>
<td>2.072</td>
<td>194.92</td>
<td>17.7</td>
<td>0.126</td>
<td>0.321</td>
<td>0.923</td>
<td>0.324</td>
<td>0.821</td>
</tr>
<tr>
<td>Cowl-3</td>
<td>2.23 ±9.7%</td>
<td>2.25</td>
<td>0.242</td>
<td>2.722</td>
<td>186.18</td>
<td>19.8</td>
<td>0.036</td>
<td>0.422</td>
<td>0.944</td>
<td>0.246</td>
<td>0.860</td>
</tr>
<tr>
<td>Cowl-5</td>
<td>2.36 ±9.3%</td>
<td>2.39</td>
<td>0.233</td>
<td>3.206</td>
<td>175.78</td>
<td>19.1</td>
<td>0.196</td>
<td>0.497</td>
<td>0.956</td>
<td>0.200</td>
<td>0.880</td>
</tr>
</tbody>
</table>

In general, aerodynamic performance improves with cowl deflection angle. Even though contraction ratio decreases slightly with cowl deflection, the compression ratio actually increases rather than decreases. Single cowl shock has been made into two lower strength shocks by deflecting the cowl. Combination of two oblique shocks is more efficient in terms of total pressure recovery than a single shock in producing overall compression. This finding is similar to that demonstrated by Das and Prasad et al.\(^68,69\). The elimination of cowl tip separation further increased the compression efficiency, hence rendering improvements for all categories of performance indicators. However, with the introduction of expansion wave in Cowl-5 case, the isolator exit static pressure was found to be lower than that in Cowl-3 case. The expansion wave also decreased the level of flow uniformity in Cowl-5 case in comparison with the other two cases. For both Cowl-3 and Cowl-5, the PSP calculated Mach number is in the order of 1% deviation from CFD.
6.2 Effectiveness of MVG for Scramjet Inlet-isolator Flow Control

Based on discussions from previous section, the boundary layer on the inlet at baseline tests condition most probably starts as laminar upstream from compression corner, where a separation is easily induced. This is probably not a good representative of real flight condition, where the long forebody of a hypersonic aircraft would provide enough length for transition. Furthermore, the separation at the compression corner would induce lateral spillage, which increases drag and reduces engine mass capture ratio. Thus, some trip configurations suitable for current model were explored and discussed in this section. Comparisons of traditional sand paper strip and micro-vortex generator (MVG) array similar to previous studies\textsuperscript{185,196} were also included in this section.

Boundary layer re-attachment along with decay of streamwise vortices/Goertler vortices on second ramp could help in ensuring turbulent boundary layer flowing into isolator section. Nonetheless, it has been reported by Nguyen \textit{et al.}\textsuperscript{150} and Reinartz \textit{et al.}\textsuperscript{60} that flow “relaminarization” could occur due to pressure drop and acceleration of flow at expansion corner. Following the example from Reinartz \textit{et al.}\textsuperscript{60}, vortex generator was placed on the second ramp to control shoulder separation. Even though they applied wire turbulator to maintain turbulence boundary layer and prevent relaminarization, MVG array was utilised in this study to achieve the same effect. The MVG array was shaped from strip of sand paper, and the effectiveness three different grade of sand paper were discussed in Section 6.2.1.

6.2.1 Boundary Layer Trip Development for Compression Ramp

Exploratory study of boundary layer trip using sand paper strip has been done on the compression ramp at Mach 5 with no AoA. Three sand paper strips of grit size P60, P100 and
P150 have been selected for this study. They were cut into strips of size 5 mm × 36 mm. Their location on the first ramp was selected by considering the requirements of having low boundary layer edge Mach number and thick enough boundary layer to accommodate the strip\(^{196}\). The location selected was \(x/L = 0.12\), where the local boundary layer thickness was found to be 0.7 mm (see Figure 6-6). The strip was pasted onto the aluminium surface using double-sided tape of thickness 0.08 mm, where the leading edge of the roughness strip was aligned at the selected x-coordinate. Parameters for different cases are presented in Table 6-3 below.

### Table 6-3 Test definitions for different sand paper thickness

<table>
<thead>
<tr>
<th>Case</th>
<th>Total Strip Height (mm)</th>
<th>Mach Number</th>
<th>Freestream Reynolds Number</th>
</tr>
</thead>
<tbody>
<tr>
<td>P150-strip</td>
<td>0.28</td>
<td>5 ± 0.4%</td>
<td>(13.2 \times 10^6 \text{ m}^{-1})</td>
</tr>
<tr>
<td>P100-strip</td>
<td>0.43</td>
<td>5 ± 0.4%</td>
<td>(13.2 \times 10^6 \text{ m}^{-1})</td>
</tr>
<tr>
<td>P60-strip</td>
<td>0.65</td>
<td>5 ± 0.4%</td>
<td>(13.2 \times 10^6 \text{ m}^{-1})</td>
</tr>
<tr>
<td>MVG-array</td>
<td>0.65</td>
<td>5 ± 0.4%</td>
<td>(13.2 \times 10^6 \text{ m}^{-1})</td>
</tr>
</tbody>
</table>

**Figure 6-6** Position of sand paper strip from leading edge
(a) First ramp leading edge shock
Shock from roughness strip element
Separation shock
Re-attachment shock
Boundary layer edge visible up to separation shock

(b) First ramp leading edge shock
Shock from roughness strip element
Separation shock
Re-attachment shock
Boundary layer edge goes blurry indicating transition
Figure 6-7 Comparison of sand paper thickness effectiveness for eliminating compression corner separation: (a) P150-strip, (b) P100-strip, (c) P60-strip, and (d) MVG-array cases for boundary layer trip

The schlieren image in Figure 6-7 (a) shows that P150-strip case was not able to affect the compression corner separation. The boundary layer edge was clearly visible up to the separation point. With P100-strip case (Figure 6-7 (b)), the separation shock was barely visible, and the re-attachment shock moved closer to compression corner. Far upstream of compression separation, the solid boundary layer edge blurred before rapid boundary layer
growth, indicating transition to turbulence. Further increasing the roughness size in P60-strip case (Figure 6-7 (c)), the corner separation was eliminated, and the transition onset point moved further upstream.
Figure 6-8 Temperature increase after 3 s of flow for (a) P150-strip, (b) P100-strip, (c) P60-strip, and (d) MVG-array.
The movement of transition line with increase in roughness size was best demonstrated by temperature increase map of the surface as given in Figure 6-8. Sub-figure (a), (b) and (c) correspond to P150-strip, P100-strip and P60-strip case, respectively. The introduction of boundary layer strip has caused sudden temperature increase ahead of the compression corner, which was not visible in Figure 4-6 (see page 135) for case without roughness. This line of temperature increase moved upstream with an increase of roughness height.

For thin roughness strip (Figure 6-8 (a)), the re-attachment belt was located downstream of compression corner. However, with thicker roughness size, the re-attachment belt moved closer to compression corner in Figure 6-8 (b), and finally touched the corner in P60-strip case (Figure 6-8 (c)), hence indicating no separation.

The effectiveness of P60-strip case in eliminating compression separation has made it as a suitable candidate for exploratory study on MVG in scramjet inlet-application. Roughness element of P60-strip case has been cut into a vortex generator array of three connected triangular vortex generator. Each vortex generator has spanwise length of 7 mm and streamwise length of 5 mm as shown in Figure 6-9. This corresponds to hypotenuse length of \( c = 6.1 \text{ mm} \) for each single MVG. The half apex angle was set to \( A = 35^\circ \). The height of the vortex generator was similar to P60-strip element thickness of \( h = 0.65 \text{ mm} \). The dimension was chosen for purely convenient reason, which does not place too much importance in optimum shape of the MVG. This was due to this experiment being exploratory in nature, thus any possibility in optimization was reserve for future studies. Three single MVGs were positioned together into an array with distance between every apex was set to \( s = 7 \text{ mm} \) (see Figure 6-10). Its shape is similar to prism-type used in the paper by Schulein and Trofimov\textsuperscript{197} albeit with roughness effect included on its surface. Its spanwise location was set by aligning
the centreline of MVG array to coincide with the model centreline. The x-coordinate of MVG array was located at 15 mm from inlet leading edge to increase the percentage of vortex generator height to local boundary layer height, and also to ensure maximum effectiveness of MVG (see Figure 6-11).

Figure 6-9 Dimension of a single MVG

Figure 6-10 Distance between MVGs in an array
Figure 6-11 The MVG array was positioned with distance, $x/L = 0.11$ from leading edge and its middle apex coincides with inlet centreline

Schlieren image in (Figure 6-7 (d)) shows the effectiveness of the MVG in eliminating the compression corner separation. The transition region for MVG case was more upstream or at least comparable to that for P60-strip case. Separation was also eliminated successfully in this case. The infrared image for this case displayed the appearance of horse-shoe vortex line and shown as vortex pair from each MVG elements in Figure 6-8 (d). The presence of other vortex structures around the MVG such as postulated in the paper by Schulein and Trofimov$^{197}$ could not be detected because of the limited spatial resolution. The extent and length of present horseshoe vortices could not be determined exactly. It is currently unknown whether the vortices would decay or continue downstream until the isolator section.

On another note, the streamwise vortices that appeared downstream of re-attachment as observed in Figure 5-2 (page 161) were not present when turbulator strips were introduced (Figure 6-7 (a), Figure 6-7 (b), Figure 6-7 (c) and Figure 6-7 (d)). This means that the streamwise vortices in Figure 5-2 (page 161) originated from the leading edge before being
amplified by interactions with Goertler vortices. Introduction of turbulator strips prevented the streamwise vortices from leading edge to interact with the flow at compression corner. This does not mean that Goertler vortices were eliminated by the presence of turbulator strips, it just shows that streamwise vortices from the first ramp leading edge were responsible for the amplifications of the Goertler vortices in terms of temperature, just enough to be detected in Figure 5-2 (page 161).

6.2.2 MVG in Scramjet Inlet-isolator Flow Control Application

Even though MVG array has been able to control compression corner separation by inducing turbulent early on the first ramp, expansion corner relaminarization means that boundary layer separation could still occur. With that in mind, MVG array has been repositioned on the middle of second ramp to test its effect in suppressing shoulder separation. Reinartz et al.\textsuperscript{60} have demonstrated the method to minimize the extent of shoulder separation by placing boundary layer trip device on the compression surface closest to expansion corner without detail quantitative analysis. In this current study, PSP technique has been applied to quantitatively characterize the isolator flow in the present of MVG array. The MVG array was positioned at the middle of second ramp to avoid its interaction with re-attachment shock just upstream of that location (see Figure 6-12).
Figure 6-12 The MVG array was positioned with distance, $x/L = 0.48$ from leading edge and its middle apex coincide with inlet centreline.

Figure 6-13 Experimental schlieren image of isolator flow for (a) baseline (no MVG) and (b) with MVG-array.

Figure 6-13 shows the comparison of isolator shock structures of case without and with MVG. Shoulder separation still occurs in Figure 6-13 (b) but with smaller size. Smaller shoulder boundary layer produced separation shock that was not strong enough to induce separation around cowl tip region like in Figure 6-13 (a). The separation induced by shoulder re-attachment shock also appeared smaller in the MVG case. It was observed that the shock...
structures inside the isolator was faintly visible in MVG-array case, indicating that the shocks were mostly weak and there was a high level of flow uniformity.

The pressure profile taken on the centreline of the model for baseline and MVG case was compared in Figure 6-14. The intensity measured by the PSP camera was calibrated against Kulite readings distributed on both internal and external parts of the scramjet inlet. Significant oscillations in pressure were detected for all transducer locations downstream of MVG location. PSP calibration curve for MVG-array case was satisfactory, with coefficient of determination value, $R^2 = 0.976$. Pressure predicted by PSP in unit bar was correct to within $\pm 0.048$. 

Figure 6-14 Normalized static pressure profile without and with-MVG case
The PSP predicted pressure for MVG-array case was consistently lower than baseline case in the region downstream of MVG locations. The figure also shows that even though separation still occurs inside the isolator, severity of its re-attachment shock has dropped by more than 50%. The re-attachment peak pressure for MVG-array case also moved upstream, closer to shoulder. Subsequently, all pressure peaks inside the isolator have been reduced as well.

![Figure 6-15 Inlet-isolator surface pressure comparison of without and with-MVG](image)

The three-dimensional effect of MVG on the flow is shown in Figure 6-15. On the second ramp surface, the pressure drop was confined only at a small streak downstream of each individual vortex generator. Similarly, the reduction in re-attachment pressure peak was also limited in the middle region, which is under the influence of MVG, therefore resulting in the distortion of subsequent shock footprints further downstream.
Spanwise pressure has been taken at the start (coordinate $x/L = 0.59$), middle (coordinate $x/L = 0.78$) and end of isolator segment (coordinate $x/L = 0.99$) and compared in Figure 6-16. The spanwise pressure at position immediately downstream of MVG array ($x/L = 0.54$) has been included as well. The pressure drop associated with vortex streak emanating from each individual MVG has been detected (marked with red-dashed circle in Figure 6-16). This drop in pressure was followed by gradual pressure increase similar to observation made by Li and Liu$^{198}$ shown in Figure 6-17.
The position of the highest spanwise distortion from the plot was at \( x/L = 0.59 \) (isolator entrance), with standard deviation of 0.965. Further downstream, the spanwise pressure gained more uniformity, and the standard deviations reduced to 0.387 and 0.442 at position \( x/L = 0.78 \) (middle of isolator) and \( x/L = 0.99 \) (end of isolator), respectively.

### 6.3 Effectiveness of Variable-geometry Cowl to Optimize Flow at Off-design Conditions

Section 5.1.3 (page 163) has shown the appearance of Edney’s Type II\(^{st}\) shock-shock interaction at isolator entrance of inlet with \( \text{AoA} = 4^\circ \). The performances of the inlet at such condition are discussed in Section 5.1.6 (page 170), and the low aerodynamics performances are related to the flow structures. Large flow spillage also occurs as the compression shock missed the cowl tip. The objective of this section is to explore the possibility of optimizing inlet performance at off-design conditions with the help of variable-geometry cowl mechanism. The first mechanism was by increasing the vertical position of the cowl by 2.2 mm to obtain shock-on-lip condition. It is also to re-position the cowl shock impingement location downstream and altering the shoulder separation height, thus avoiding the formation of Mach stem. This case is termed \( \text{AoA4-Height} \) in all discussions to follow.
second mechanism was increasing the length of the cowl ahead of shoulder position by 3 mm with the objective of simply to obtain shock-on-lip condition. This case is named AoA4-Length case, and will be compared to AoA-4 and AoA4-Height case. The definitions of different optimized cowl case are given in Table 6-4 below:

Table 6-4 Variable geometry case definitions

<table>
<thead>
<tr>
<th>Test Name</th>
<th>Mach Number</th>
<th>Reynolds Number</th>
<th>Stagnation Pressure</th>
<th>Stagnation Temperature</th>
<th>Cowl Geometry</th>
<th>AoA</th>
</tr>
</thead>
<tbody>
<tr>
<td>AoA4-Height</td>
<td>5 ± 0.4%</td>
<td>$13.2 \times 10^6$ m$^{-1}$</td>
<td>6.5 ± 0.05 (Bar)</td>
<td>375 ± 5 (K)</td>
<td>Base-cowl with Height + 2.2mm</td>
<td>4°</td>
</tr>
<tr>
<td>AoA4-Length</td>
<td>5 ± 0.4%</td>
<td>$13.2 \times 10^6$ m$^{-1}$</td>
<td>6.5 ± 0.05 (Bar)</td>
<td>375 ± 5 (K)</td>
<td>Long-cowl</td>
<td>4°</td>
</tr>
</tbody>
</table>
6.3.1 Variable-geometry Effects on Scramjet Inlet-isolator Flow Structures at Off-design Conditions

Figure 6-18 Experimental and numerical schlieren for AoA-4 [(a) & (b)], AoA4-Height [(c) & (d)] and AoA4-Length [(e) & (f)] cases (angle-of-attack at windward)
Figure 6-18 shows the shock structures of the inlet with optimized cowl position at off-design conditions in comparison to the original case. Initially, AoA-4 suffers from large spillage area and Mach stem at isolator entrance. These problems were alleviated by increasing the cowl height. The two shocks from compression ramp coalesced and impinged the inner surface of cowl. As the cowl leading edge was at an angle to freestream, its expansion wave interacted with the focused compression shock. The interaction is classified as Type 3 off-design interactions in Bachchan and Hillier\textsuperscript{48}, and it triggered a Mach reflection. The unstable Mach reflection then moved upstream and detached at cowl leading edge. The current low frame rate of schlieren system could not capture the movement of the Mach reflection and shock detachment at cowl lip. However, similar observation of cowl shock detachment by using a high frequency camera can be found in Mahapatra and Jagadeesh\textsuperscript{63}.

The reflected shock emanating from triple point impinged on isolator surface and induced a separation bubble on shoulder. Similar to Cowl-5 case (see Figure 6-1 (c), page 188), the separation had expansion wave at its onset point. The subsonic flow pocket downstream of Mach reflection at cowl interacted with the expansion wave from shoulder and accelerated into supersonic again further downstream. The performance of the inlet in this case was expected to be either equally bad or worse than that of AoA-4 case. The pattern predicted by CFD was comparable to experimental observation.

For AoA4-Length case (Figure 6-18 (e) and (f)), the cowl length extension has allowed for shock-on-lip condition to be fulfilled with an added bonus of Mach stem elimination that was present in AoA-4 case (Figure 6-18 (a)). The exact mechanism in Mach stem elimination is unknown. Nonetheless, it is suspected to be associated with the changes in separation size at shoulder and cowl tip between the two cases. The overall shock waves pattern in this case
is akin to baseline case seen in Figure 4-8 (a) (page 137). Just from qualitative examination of the three cases in Figure 6-18, it could not be determined which will have better performance.

6.3.2 Variable-geometry Effects on Scramjet Inlet-isolator Pressure Profile at Off-design Conditions

Figure 6-19 Isolator surface static pressure for AoA-4, AoA4-Height and AoA4-Length cases (angle-of-attack at windward)

It was relatively difficult to investigate the correct pressure profile trends inside the isolator section for the optimized cowl cases due to inevitable pressure oscillations of Kulite pressure readings. In Figure 6-19, the readings from all four transducer locations in isolator for both AoA4-Height and AoA4-Length cases had significant oscillations, hence resulting in higher uncertainty in PSP calibration.
The PSP camera obstruction by cowl leading edge added more difficulties in calibrating intensity to pressure as it reduces the number of available discreet points in calibration plot. Goodness of fit of calibration curve to discreet data points was surprisingly good, with $R^2 = 0.981$ and 0.990 for AoA4-Height and AoA4-Length case, respectively. However, since the intensity was calibrated against the time average value of static pressure, averaging error was inherently high in PSP pressure profile, even though the calibration error was minimal. The pressure predicted by PSP is correct to within ± 0.072 and ± 0.085 bar for AoA4-Length and AoA4-Height case, respectively.

With low confidence and measurement gap in PSP readings, CFD fill the hole in understanding the pressure rise and drop inside the isolator. The isolator wall static pressure oscillations in AoA4-Height case are most probably associated with Mach reflection, which is known to be unstable. The nature and characteristics of pressure oscillations for both cases have been investigated by performing Fourier transform on the each transducer time history, but no significant pattern could be detected.

If the CFD predictions are to be considered as the best representative of true pressure profiles, it could be noted that AoA4-Length case has similar pressure patterns with AoA-4 case, but lower re-attachment peak at shoulder, thus indicating lower separation bubble size at shoulder. On the other hand, AoA4-Height case has similar pressure patterns with Cowl-5 (see Figure 6-3, page 190).
6.3.3 Variable-geometry Effects on Scramjet Inlet-isolator Pressure Map at Off-design Conditions

From Figure 6-20 (a), it could be observed that the inlet with cowl optimization cases have managed to reduce three-dimensionality effects symbolized by the curved shock traces.
made by impinging shock waves. Having said that, three-dimensionality effects are usually intensified near sidewall, where is not available for viewing in Figure 6-20 (b) and (c).

### 6.3.4 Variable-geometry Effects on Scramjet Inlet-isolator Pressure Map at Off-design Conditions

[Figure 6-21 Comparison of sidewall pressure contour for (a) AoA-4 and (b) AoA4-Length cases]

Comparison of sidewall pressure contour of the scramjet inlet at AoA = 4° without and with optimized cowl is shown in Figure 6-21. Case AoA4-Height could not be included in the comparison as the strong glancing shock inside the isolator has sliced and cut through the thin PSP layer, thus rendering it unusable.

Figure 6-20 (c) was used as a basis for calibrating sidewall surface pressure map produced in Figure 6-21 (b). The pressure profile at the edge of isolator top window has been assumed to
be similar to the profile at sidewall intersection. Sidewall intensity-pressure calibration managed to obtain acceptable value of $R^2 = 0.943$.

Similar shock patterns can be observed on the sidewall of AoA-4 and AoA4-Length case. However, the magnitude of peak pressure downstream of shock (ii) was lower, providing an advantage for the optimized cowl case.

The window of interrogation for Mach number calculation for AoA4-Length case has been chosen around shock (v) in Figure 6-21 (b). All calculated flow parameters and performance are included in Table 6-5 below:

Table 6-5 Comparison of flow parameters and performance of inlet at AoA = 4 without and with cowl optimization

<table>
<thead>
<tr>
<th>Case</th>
<th>$M_e$</th>
<th>$M_{e(CFD)}$</th>
<th>$p_e$ (bar)</th>
<th>$p_{te}$ (bar)</th>
<th>$T_e$ (K)</th>
<th>$\tilde{\zeta}$</th>
<th>$\sigma_p$</th>
<th>$\pi_c$</th>
<th>$\eta_{KE(ad)}$</th>
<th>$\Delta s/c_p$</th>
<th>$\eta_{c(ad)}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>AoA-4</td>
<td>1.77 ± 11.2%</td>
<td>1.84</td>
<td>0.472</td>
<td>2.615</td>
<td>228.10</td>
<td>38.7</td>
<td>0.046</td>
<td>0.405</td>
<td>0.941</td>
<td>0.258</td>
<td>0.890</td>
</tr>
<tr>
<td>AoA4-Length</td>
<td>1.93 ± 13.8%</td>
<td>1.89</td>
<td>0.379</td>
<td>2.656</td>
<td>213.25</td>
<td>31.1</td>
<td>0.113</td>
<td>0.412</td>
<td>0.942</td>
<td>0.254</td>
<td>0.882</td>
</tr>
</tbody>
</table>

From Table 6-5, it was noted that the inlet with cowl optimization managed to increase the isolator exit Mach number due to the elimination of Mach stem, which subsequently reduced the isolator exit static pressure and temperature. Improvements in aerodynamics performance indicators of $\pi_c$, $\eta_{KE(ad)}$, and $\Delta s/c_p$ were obtained. However, the overall compression efficiency, $\eta_{c(ad)}$, dropped by a small percentage.
6.4 Strategy for Scramjet Inlet-isolator Performance Optimization

Calculated inlet performance indicator from all cases was compiled and compared with value published by several literatures\(^ {37,31,199,200,201}\) and depicted in Figure 6-22 above. Accepted postulated correlations\(^ {193,194,195,192}\) that relate kinetic energy efficiency to throat Mach number ratio are presented as line plots. Only equation from Smart\(^ {192}\) has variable curvature depending on freestream Mach number. In this current plot, Smart’s equation only considers freestream Mach number of 5.

Discreet data points have been taken from Hohn and Gulhan\(^ {37}\) (circle), Wie and Ault\(^ {31}\) (square), Smart and Tetlow\(^ {199}\) (diamond), Molder\(^ {200}\) (star) and Turner and Smart\(^ {201}\) (downwards triangle). Current experimental data are marked with triangles inside the plot. It was discovered that data points from current experiment and literatures lie very close
within the bandwidth set by the equations from Billig et al.\textsuperscript{194} and Tani et al.\textsuperscript{195}. All current experimental data points remained within close proximity of curves from Waltrup et al.\textsuperscript{193} and Tani et al.\textsuperscript{195}.

The plot shown above in Figure 6-22 proved that none of the correlation functions could claim superiority over the others. Instead, they acted as statistical bandwidth limit of most reported data. The root cause of this problem is that there are so many different types, geometry and freestream Mach number of reported scramjet inlet investigations; hence a single equation to establish trend would probably be inadequate. Smart\textsuperscript{192} proposed that a different correlation equation should be used for different freestream Mach number. However, his proposed curve with freestream Mach number of 5 in the figure above was unable to fit current experimental data.

From the plot, it could be presumed that the best strategy for optimizing performance of a scramjet inlet is by increasing the ratio of throat Mach number to freestream. Maintaining high ratio of throat to freestream Mach number is counterbalanced by the needs to keep optimum compression ratio necessary for combustion. The current inlet achieved the best performance ($\eta_{KE(ad)} = 0.956$) when the cowl was deflected by 5\textdegree inwards. This is because it managed to reduce the strength of cowl tip shock, thereby increasing the throat Mach number while concurrently maintaining high compression ratio in contrast to baseline case. The worst performance of the scramjet inlet-isolator ($\eta_{KE(ad)}=0.849$) was experienced when the compression shock impinged the inner cowl surface. The extra shock-shock and shock-boundary layer interactions reduced the isolator exit Mach number without contributing positively towards compression process.
Having large Mach number ratio is also beneficial in the sense of having smaller uncertainty range of performance. For example, if the inlet has throat-to-freestream Mach number of 0.4, its kinetic energy efficiency could be between 0.917 and 0.984. On the other hand, if the inlet has Mach number ratio of 0.6, the performance probability range is smaller, from 0.981 to 0.998. For this reason, it is easier for an inlet engineer to quickly predict inlet performance at high throat to freestream Mach number.

6.5 Conclusions

The scramjet inlet-isolator suffers low performance in its baseline configurations due to the large boundary layer separation occurring at the isolator entrance. Cowl tip has been deflected inwards to mitigate the separation problem. The results were considered successful, where with step increase in cowl deflection angle, the performance of the inlet improved dramatically without sacrificing the overall compression. The separation onset point moved downstream with every increase in deflection angle, thus reducing the probability of inlet unstart. The peak pressure due to shoulder re-attachment has also diminished significantly with every cowl angle increment.

Exploratory study of suitability of using MVG to mitigate shoulder separation has also been done. The thickness of sand paper strip suitable for separation elimination was investigated. This was followed by cutting the best sand paper strip into triangular prism array of MVG. The qualitative performance of the MVG, and the original sand paper strip was found to be comparable in terms of compression corner separation elimination. The MVG-array, when positioned on the second ramp, was also observed to reduce the size of shoulder separations. The schlieren image hinted that the inlet obtained better uniformity compared
to no-MVG case. Even though boundary layer still separates at shoulder for with-MVG case, its re-attachment peak pressure decreased dramatically. The findings in this subchapter open up new possibilities and potentials for future studies.

The low inlet performance in case with AoA = 4° has been optimized with variable geometry cowl. The AoA-4 case suffers with large mass flow spillage as the compression missed the cowl tip. High total pressure loss was inflicted onto the internal flow in AoA-4 case, with the formation of unwanted Mach stem at isolator entrance. Increasing the cowl height or length both reduced mass flow spillage and eliminated the formation of Mach stem at throat. Better aerodynamic performance was achieved by the optimized cowl length case in terms of total pressure recovery, kinetic energy efficiency and entropy increase.

Adiabatic kinetic energy efficiency from all parametric cases has been compared with available data from published literatures. The pattern of the data points could be described by correlations functions proposed by various inlet investigators. The correlations assumed that scramjet inlet kinetic energy efficiency was dominantly influenced by throat Mach number. Thus, it can be concluded that the best strategy to improve inlet performance is by preserving high throat Mach number during the compression process.
CHAPTER 7 CONCLUSIONS AND FUTURE RECOMMENDATIONS

7.1 General Conclusions

A generic scramjet inlet-isolator model has been designed and subjected to various flow diagnostic techniques that have never been previously tried in a similar flow environment. A simple inlet-isolator design process has been applied as the focus of this study is weighted towards development of measurement techniques and also optimization of the inlet. The study was conducted in blow-down high supersonic wind tunnel under relatively long duration to simulate the flight real flight condition better. Mach number 5 was chosen as the test condition since it is generally regarded as the start of hypersonic speed regime. At such speed, it would be inefficient to decelerate the incoming flow to subsonic for combustion. Thus, supersonic combustion is needed.

One of the main focuses of this study is to develop a robust global surface measurement system in scramjet applications, and the first step was to determine which luminophore work best in such conditions. The study was limited to two of the most common luminophores; PtTFPP and Ru(II). PtTFPP-based PSP was shown to have lower spatial pressure noise in comparison to Ru(II)-based PSP. Low spatial pressure noise is important in any measurement where significant flow unsteadiness is expected, such as the SWBLI at compression corner. On the other hand, Ru(II)-based PSP was found to be effective as streamwise vortices detector. This serves as an alternative to more expensive IR thermography or TSP, which requires more post-processing in order to visualize similar vortices. The PtTFPP-based PSP, specifically the PSP mixture optimized for this study, has been shown to be very robust and was able to function normally from the flow speed of
Mach 4 (on the first ramp) to subsonic (at separation regions). This would add to the current limited list of high supersonic PSP mixture available in open literature.

From isolator surface pressure map, it could be mentioned that the flow inside the isolator is minimally three dimensional. The spanwise change in pressure was mostly small in magnitude and was confined only within 5% of spanwise length from the sidewall. Three-dimensionality of the flow in isolator was also found to be the lowest at isolator exit. This has validated the assumptions of two-dimensional flow, which then allowed high confidence in Mach number estimation using the final shock near the isolator exit. Minimal level of three-dimensional flow also allowed for satisfactory estimation of flow using two-dimensional CFD, which would significantly reduce computational time.

The visualization of flow on the internal sidewall surface by using PSP allows quantitative analysis of the overall inlet. With appropriate assumptions, the glancing oblique shock on the sidewall surface, which was detected by PSP, was utilised to predict the properties of flow exiting the isolator. Calculated Mach numbers by using this method were compared with CFD simulation for each unique case, and the results were very encouraging. Typically, the Mach number calculated using PSP differs within 6% or less from validated CFD data, except in Long-Cowl case, where the CFD could not properly simulate the large separation region observed experimentally. This level of accuracy is better than that provided by pitot rake or stream-thrust analysis. This new technique of analyzing scramjet inlet presented a breakthrough that would benefit overall scramjet inlet research efforts.

The separation bubble at shoulder has negatively impacted the overall flow quality inside the isolator. The separation has contributed to boundary layer separations on the cowl surface. At baseline test configuration, the re-attachment of shoulder separation bubble
typically exerted pressure of about 50 times freestream pressure. In addition, the re-attachment of cowl tip separation exerted pressure of about 100 times the freestream. The implication for this design is that a scramjet inlet designer must find a preventive measure to mitigate the appearance of shoulder separation. The structural integrity of the inlet must be given special consideration around the shoulder and cowl tip.

This thesis also found the relationship between separation on cowl surface and the overall performance. If the cowl tip is extended upstream, the boundary layer separation on cowl surface will grow in size and reduce the performance of the inlet. The detrimental effect in performance was severe even though the increase of cowl tip separation was minimal. It seems the best configuration is by deliberately designing the cowl to not achieve the shock-on-lip condition. This is shown by Short-cowl case, where the compression ramp shocks totally missed the cowl tip. Even though it suffers minimal mass flow rate, improvements in overall compression ratio and adiabatic compressive efficiency outweigh the disadvantages.

Two preventive measures to mitigate shoulder separation have been explored in this thesis. The first is by deflecting the cowl tip inwardly, which would reduce the strength of cowl shock impinging on surface near shoulder. The effect was dramatic, where the impact of shoulder re-attachment shock was reduced by half with 5 degree deflection. Cowl separation has been eliminated with cowl deflection, and thus reducing the structural load on cowl surface due to unwanted shock. The extra benefits of shoulder separation mitigation include the improvement of overall scramjet performance. However, an inlet designer must also consider the slight loss in compression ratio of the inlet with cowl deflection beyond the optimum degree.
Another boundary layer separation preventive measure tested in this thesis is the use of MVG located on the compression ramp upstream of the shoulder region. A non-optimized MVG array has been shown to be able to decrease the impact due to shoulder re-attachment by more than half. The resultant pressure exerted on the isolator surface was only 20 times the freestream pressure. The separation bubble was barely visible in the with-MVG case. This observation must be investigated further using optimized MVG geometry. The impact of adding the MVG on the scramjet inlet surface in terms of performance and drag must be considered in future studies.

Published scramjet inlet performance data has been compiled and compared with the current experiments. Four correlation equations\textsuperscript{193,194,195,192} that attempt to relate kinetic energy efficiency with throat-to-freestream Mach number ratio, \( \frac{M_{th}}{M_0} \), has been included in the plot as well. It seems that the performances of the scramjet inlet stay roughly within the bandwidth made by the equations (see Figure 6-22, page 219). From the plot, it could be concluded that scramjet inlet performance generally increase with \( \frac{M_{th}}{M_0} \). Hence, the combinations of variable-geometry inlet employed at off-design conditions must maintain high throat Mach number in order to produce high performance and thrust. This can be done by aiming only minimal compression ratio of about 50 times the freestream such as suggested by Smart\textsuperscript{192}.

7.2 Future Recommendations

The newly developed scramjet inlet-isolator exit Mach number measurement technique by using PSP discussed in this thesis has large potential for future studies. For example, similar inlet configurations could be tested using the traditional stream-thrust analysis and compared with the data from this thesis. This has not been done as part of this thesis, as the
problem of tunnel unstart prevented the attachment of mass flow meter component at isolator exit.

Concurrently, pitot pressure rake that is small enough to be fitted into the small isolator height could not be designed in time. This could provide another comparable estimate of isolator exit Mach number.

The concept of utilizing oblique shock relations in calculating isolator exit Mach number could be applied to global measurement techniques of Background-oriented Schlieren (BOS), quantitative colour schlieren, temperature sensitive paint (TSP), infrared thermography, and many others. The inlet-isolator, which is the subject of this thesis, has been designed with sidewall optical window that is suitable for the applications of BOS and TSP. Initial testing of BOS for this inlet configurations has been done and included in the appendix. This will definitely be the future direction and continuation of this thesis.

The window on the sidewalls of the isolator would make the inlet suitable for testing using Particle Image Velocimetry (PIV) as well. The velocity data provided using this technique could be used as comparison. Some minor damage on the Perspex windows are expected due to high speed impingement of the PIV seeder particles. However, the low cost of the Perspex window components makes this technique possible.

The PSP methods exploited in this study assume a steady flow on the compression ramps and inside the isolator. It would be very interesting to visualize the physical movements of shocks on the ramps and isolator. Anodized aluminium PSP technique, which has fast response to pressure, could provide the required unsteady flow investigative tool. There have been attempts to anodize the current model but were never successful, and the
reasons must be investigated. The author of this thesis has also tried to develop another type of fast response PSP using polymer-ceramic as binder, but currently it is very brittle and the paint layer could be adversely affected just by touching.

The inlet could also be tested in other off-design conditions such as change in Reynolds number and change in flow velocity. Reynolds number could be varied by modifying combinations of stagnation pressure and temperature. When the inlet was tested in higher stagnation temperature environment (lower unit Reynolds number), the Perspex window was damaged and needed replacement. A new window with higher temperature resistance would allow testing of boundary layer effect on the inlet operation. The initial plan for this thesis was to vary the Mach number to 4 and 6, but the amount of calibration works needed when the nozzle is changed is too time consuming. This is another potential study in the immediate future.

In this thesis, a simple numerical simulation assuming only two-dimensional flow was used. With the abundance in experimental data, validation of three-dimensional numerical scheme is exciting.

The Aero-physics Laboratory research group of which this author belongs to has initiated energy injection studies in hypersonic flow\textsuperscript{71,202–204}, and these will provide another aspect of investigative direction. The behaviour of the current inlet with gas injection on compression ramps is also interesting for future analysis.
REFERENCES


32. Mitani, T., Hiraïwa, T., Tarukawa, Y. & Masuya, G. Drag and total pressure distributions in scramjet engines at Mach 8 flight. *Journal of Propulsion and Power* 18,


77. Matsumura, S., Schneider, S.P. & Berry, S.A. Streamwise vortex instability and transition on the Hyper-2000 scramjet forebody. *Journal of spacecraft and rockets* 42,
78–89 (2005).


156. Slater, J.W. & Saunders, J.D. *Computational Fluid Dynamics (CFD) Simulation of Hypersonic Turbine-Based Combined-Cycle (TBCC) Inlet Mode Transition.* (National Aeronautics and Space Administration, Glenn Research Center: 2010).


222. Mirsepassi, A. & Dunn-Rankin, D. Particle image velocimetry in viscoelastic fluids and particle interaction effects. *16th Int Symp on Applications of Laser Techniques to Fluid


APPENDIX

A. High Supersonic Tunnel Calibration

![Graph showing time histories of stagnation pressure, pitot pressure, and stagnation temperature.](image)

Figure 0-1 Time histories of stagnation pressure, pitot pressure and stagnation temperature. The temperature scale is shown on the right (figure taken from Erdem). Calibration of the tunnel has been done by Erdem, and the process can be found extensively in his thesis. The data for calibration is shown in Figure 0-1. The figure shows time histories of pitot and stagnation values for a typical tunnel run time, with supplied total pressure and temperature of 6,450 mbar and 375 K, respectively. The jump in stagnation and pitot pressure was due to starting shock from the nozzle moving downstream before being swallowed by the diffuser. At around \( t = 8 \) seconds, oscillation was observed on pitot pressure reading, indicating end-of-tunnel stable running time. Between the initial pressure jump and oscillation, a stable plateau of pressure useful for test conditions can be obtained. The total steady flow duration was found to be roughly 7.5 seconds. The amplitude of oscillations during this stable plateau was less than 1%. Further analysis of frequency content of the oscillation can be seen clearly in Figure 0-2. Erdem explained that the
highest peak on the stagnation pressure signal (with and without filter) detected at 200 Hz was due to harmonic component of the electronic noise. Smaller peaks at 900, 1,650 and 2,000 Hz were related to the cavity frequency of the pressure chamber of the pitot rake. The slow stagnation temperature rise seen in Figure 0-1 was due to slow response of the thermocouple and did not reflect the true temperature history. Schlieren images in Figure 0-3 verified the assumed stable running time of 7.5 seconds.

Figure 0-2 Fast-Fourier transformed plot of stagnation and pitot pressure histories (figure taken from Erdem71)
Results of wind tunnel calibration using pitot rake is summarized in Table 0-1. Uncertainties in the values were a mixture of uncertainties from supply pressure set at dome valve, heater supply temperature set and measured pressure.

<table>
<thead>
<tr>
<th>( M )</th>
<th>( p_t ) (mbar)</th>
<th>( T_t ) (K)</th>
<th>( Re \times 10^6 )/m</th>
<th>( p_{pitot} ) (mbar)</th>
<th>( p_{\infty} ) (mbar)</th>
<th>( U_{\infty} ) (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>5 ± 0.5%</td>
<td>6450 ± 0.7%</td>
<td>375 ± 1.7%</td>
<td>13.1 ± 3.7%</td>
<td>396 ± 1%</td>
<td>12.18 ± 2.4%</td>
<td>790 1.0%</td>
</tr>
</tbody>
</table>

**B. Pressure Sensitive Paint (PSP) Theory**

**B.1 Photoluminescent Kinetics Theory**

Commonly, pressure transducers has been the method of choice to obtain pressure of a model in wind tunnel experiments. This method requires large manufacturing time and cost and provides only discrete point-based pressure readings. To supplement the discrete data, aerodynamicists have to resort to computational fluid dynamics to obtain the pressure map.
distribution. With advent of pressure sensitive paint (PSP), there is now an easier method to measure pressure distribution of model in wind tunnel experiment. There are many publications about PSP since it was first developed in the 1980s, but generally, the three good sources are [1] the paper by Sullivan et al.\textsuperscript{205}, which reports on the early works done in Purdue University; [2] a concise and very informative review of the method written by Bell et al.\textsuperscript{130}; and of course, [3] the most recent text book on this subject matter by Liu and Sullivan\textsuperscript{132}.

PSP method capitalizes on the sensitivity of certain luminescent molecules to oxygen to give out “visual” effect of pressure distribution. Typically consists of oxygen-sensitive luminescent molecules and oxygen permeable binder, photons are emitted to the molecules to excite them to an upper singlet energy state before the molecules are recovered by emission of another photon with longer wavelength (see Figure 0-4). However, if the excited molecules interact with oxygen, no photon is radiated, as excited energy is released to oxygen molecules instead. This radiationless transition from excited energy state to ground state is called oxygen quenching. Henry’s law states that oxygen concentration in air is linearly proportional to its partial pressure\textsuperscript{132}. With increase in pressure, more molecules go through oxygen-quenching process instead of radiating energy into photons. Thus, the intensity of PSP is inversely proportional to static pressure.
The luminescent kinetics of the luminophore working principle can be better understood with the Jablonski energy level diagram shown in Figure 0-5. The following explanation is an abridged version of detailed discussions found from Liu and Sullivan\textsuperscript{132} and Bell \textit{et al.}\textsuperscript{130}. The diagram shows that when photon from light source is absorbed by the luminophore, the molecules will become excited and their energy level are transformed from ground state, $S_0 \rightarrow S_1$, or, $S_0 \rightarrow S_2$. Absorption happens instantaneously in $10^{-15}$ s. Higher energy level of excited state, $S_2$, could transform into, $S_1$ by radiationless internal conversion, which typically
involves heat release, $\Delta$ (typically expressed as $S_2 \rightarrow S_1 + \Delta$). This process is also instantaneous, with $10^{12}$ s.

$S_1$ could return to ground state, $S_0$ by both radiationless ($S_1 \rightarrow S_0 + \Delta$) and radiative ($S_1 \rightarrow S_0 + \hbar \nu_f$) processes. This radiation is of fluorescence type, and the nature is characterized by Plank constant, $\hbar$, and excitation light frequency, $\nu_f$. $S_1$ could also be transformed by intersystem crossing to become excited triplet state, $T_1$ ($S_1 \rightarrow T_1 + \Delta$), even though this process is theoretically forbidden by quantum mechanics as it would involve changes in electron spin. Nevertheless, spin-orbit coupling effect of electrons turn molecules into a mix of the two pure states, where intersystem conversion is possible. Bell et al. explained that for luminophore of heavy metal types such as ruthenium and platinum, intersystem conversion has as much possibility to occur as intrasystem conversion.

$T_1$ would transform into ground state $S_0$ with radiative phosphorescence process ($T_1 \rightarrow S_0 + \hbar \nu_f$). As phosphorescence is an intersystem change, where it is theoretically forbidden, it would typically have longer lifetime compared to fluorescence. $T_1$ could also be deactivated without radiation as shown by ($T_1 \rightarrow S_0 + \Delta$). It is obvious from the figure that emission energy is always lower than excitation energy; thus there will be a shift to longer wavelength in comparison to excitation. This is called Stokes shift as he was the first to note this phenomenon.

If the excited molecules interact with environment, deactivation process would be affected. For example, if the excited molecules interact with oxygen molecules, then less energy will be spent by radiative conversion. This process is called oxygen quenching. Oxygen molecules are very effective energy quencher as it has triplet system ground-state energy level. Their excited singlet state requires only 1.0 eV of excitation energy but just as explained
previously, intersystem change from triplet ground-state to singlet excited-state (or vice versa) is forbidden by quantum mechanics.

From engineering point of view, the exact photophysical kinetics discussed above is non-essential with regards to pressure analysis. It is a standard practice to relate intensity directly to pressure (oxygen concentration) with Stern-Volmer plot\textsuperscript{132}. Nevertheless, the discussion shed some lights on the basis of using photoluminescence for pressure measurements.

B.2 Derivation of Stern-Volmer Equation for PSP

First, the population of excited luminophore molecules where excitation, emission, radiationless deactivation and oxygen quenching processes occur at the same time can be denoted by \([N]\). Then, the rate of change in \([N]\) can be expressed by:

\[
\frac{d[N]}{dt} = E_a - (k_r + k_{nr} + k_q[O_2])[N]
\]

Where \(E_a\) is the rate of excitation, \(k_r[N]\) is the rate of radiation, \(k_{nr}[N]\) is the rate of radiationless deactivation and \(k_q[O_2][N]\) is the rate of quenching by oxygen molecules \(O_2\).

When there is no outside interference to the system (i.e. no oxygen quench) and no net change in excited state concentration \([N]\), Equation 0-1 will become:

\[
E_a = (k_r + k_{nr})[N]
\]

However, if there is oxygen quenching and the net change in excited state is zero, Equation 0-2 will become:
Equation 0-3

\[ E_a = \left( k_r + k_{nr} + k_q [O_2] \right)[N] \]

Quantum yield or efficiency of the radiative process is defined as the ratio of luminescence rate to excitation rate. It is also defined as the ratio of number of photon emitted to photon absorbed. Thus, in the case of no oxygen quenching, the emission efficiency is given by:

Equation 0-4

\[ \varphi_0 = \frac{k_r[N]}{E_a} = \frac{k_r[N]}{(k_r + k_{nr})[N]} = I_o / I_a \]

Here, \( I_o \) is intensity of luminescence in condition without oxygen quenching. In the case of oxygen quenching, the efficiency becomes:

Equation 0-5

\[ \varphi = \frac{k_r[N]}{E_a} = \frac{k_r[N]}{(k_r + k_{nr} + k_q [O_2])[N]} = I / I_a \]

Stern-Volmer equation is set as the ratio of emission efficiency without oxygen quenching to quantum yield with quenching.

Equation 0-6

\[ \frac{\varphi_0}{\varphi} = \frac{k_r[N]}{(k_r + k_{nr} + k_q [O_2])[N]} = \frac{I_o}{I} \]

Rearranging the equation yields:

Equation 0-7

\[ \frac{I_o}{I} = 1 + \frac{k_q}{k_r + k_{nr}}[O_2] \]
Henry’s law states that concentration of oxygen inside the polymer binder that is available for interaction with luminophore molecules is proportional to static pressure:

Equation 0-8

\[
[O_2] = \mathcal{M} p
\]

Thus, Equation 0-7 becomes:

Equation 0-9

\[
\frac{I_o}{I} = 1 + \frac{k_q}{k_r + k_{nr}} \mathcal{M} p
\]

The equation now gives a practical relation between pressure and intensity. However, the equation poses a problem as it requires knowledge of luminescence intensity value in condition without oxygen quenching, which is not possible in normal wind tunnel operation with air (~21% oxygen) as test gas. Thus, it is more appropriate to measure intensity at reference condition with known pressure and another at unknown pressure.

Equation 0-10

\[
\frac{I_o}{I_{ref}} = 1 + \frac{k_q}{k_r + k_{nr}} \mathcal{M} p_{ref}
\]

Thus, by dividing Equation 0-10 with Equation 0-9, the final form of Stern-Volmer equation becomes:

Equation 0-11

\[
\left(\frac{I_o}{I_{ref}}\right) = \frac{I_{ref}}{I} = \frac{1 + \frac{k_q}{k_r + k_{nr}} \mathcal{M} p}{1 + \frac{k_q}{k_r + k_{nr}} \mathcal{M} p_{ref}}
\]

This may be conveniently expressed as:
Equation 0-12

\[
\frac{I_{ref}}{I} = A + B \left( \frac{p}{P_{ref}} \right)
\]

This final form of Stern-Volmer equation gives a convenient and simple linear equation which predicts pressure from measured intensity. This equation forms the strategy in applying PSP technique for accurate pressure measurement. Intensities at two pressure conditions are measured; one where pressure is known and referred to, and another one is at the unknown pressure. Then, the unknown pressure can be predicted using the equation, provided constants \( A \) and \( B \) are known. There are two methods in estimating constant \( A \) and \( B \). The first is called “a priori” calibration, where \( A \) and \( B \) are inferred from plot of intensity ratio change with respect to change in pressure ratio in a pressure calibration chamber. Then, the equation can be used for predicting pressure from measured intensity done in the wind tunnel facility of interest. The second calibration method correlates measured intensity ratios with measured pressure ratio near tap location (i.e. both intensity and pressure measured at the same time). Obviously, the second method, called “in-situ” calibration method, requires enough discreet pressure taps on interested model surface, whereas in a-priori method, none is required. However, the second method solves the problem of temperature dependencies of constants \( A \) and \( B \). With in-situ calibration process, Stern-Volmer equation could be expanded to form a polynomial equation in the form of:

Equation 0-13

\[
\frac{I_{ref}}{I} = A + B \left( \frac{p}{P_{ref}} \right) + C \left( \frac{p}{P_{ref}} \right)^2 + D \left( \frac{p}{P_{ref}} \right)^3 + \ldots
\]
C. Infrared Thermography

C.1 Theory

To reduce IR data into heat flux, three heat flux sensor models can be used. Detailed explanations of all mathematical models are given in Astarista et al. The “thin-film” model, which is most relevant to the current study, will be discussed in this part. The mathematical model actually follows the classical methods of measuring surface heat flux by bonding a very thin-film resistance thermometer onto subject’s surface. The thinness of the film is such that the film has negligible thermal resistance and heat capacity so as to not disturb the temperature history of main subject surface.

Consider the sketch in Figure 0-6, which shows thin film bonded onto semi-infinite wall surface. $Q_c$ is the convection heat transfer rate from the surrounding flow, $Q_r$ is the radiative heat transfer from the film and $Q_k$ is the conductive heat flux within the model test subject body. The net heat transfer rate $Q_n$ as function of time is given by:

\[
Q_n = Q_c + Q_r + Q_k
\]
Equation 0-14

\[ Q_s(t) = \frac{\rho c k}{\pi} \int_0^t \frac{dT_s(\tau)}{d\tau} \sqrt{t - \tau} \, d\tau \]

Where \( \rho = \) density, \( c = \) specific heat, \( k = \) thermal conductivity coefficient, \( t = \) duration of measurement, \( \tau = \) time variable, and \( T_s = \) surface temperature as function of time.

For simplification, the equation can be reduced by assuming isothermal surface at the start of measurement:

Equation 0-15

\[ Q_s(t) = \frac{\rho c k}{\pi} \left[ T_s(t) - T_s,i \right] + \frac{1}{2} \left[ t \frac{T_s(t) - T_s(\tau)}{(t - \tau)^{3/2}} \right] \]

Where \( T_s(t) = \) the final surface temperature at the end of measurement duration, and \( T_s,i = \) the initial surface temperature.

If the heat transfer rate is constant, the equation will reduce to:

Equation 0-16

\[ T_s - T_s,i = \frac{2Q_s}{\sqrt{\pi} \sqrt{\rho c k}} \]

The equation above shows that heat transfer is a function of temperature difference over time. Equation 0-16 also demonstrates that temperature increase will be maximized if the material possesses low thermal properties of \( \rho, c \) and \( k \).

The equation above was developed for measuring heat flux from physical thin film bonded onto surface. For heat flux measurement using infrared, there is no need for the thin film to
actually exists; instead, the paint coating on the surface is assumed as the thin film, and no modifications on the equations are necessary.\textsuperscript{148}

\section*{D. Background-Oriented Schlieren (BOS) in Scramjet Inlet-isolator Investigation}

\subsection*{D.1 Introduction}

The concept of analysing the final shock inside the isolator section for Mach number calculation presented in this thesis could be easily extended to utilise global density-based measurement techniques instead of PSP. Following the discussion in Section 2.2.5.3, if density-field could be determined, then Equation 2-24, Equation 2-27 and Equation 2-28 could be easily solved to find $M_3$.

BOS technique, which has been gaining its popularity today, is particularly suitable for calculating the density increase ratio across an oblique shock. Compared to method discussed in Chapter 4, where average isolator flow properties are inferred from the pressure map on sidewall, two-dimensional BOS technique would result in spanwise-integrated density field\textsuperscript{100} for the whole isolator. Concurrently, a three-dimensional BOS\textsuperscript{208,209}, which requires simple modifications from two-dimensional method could provide truly tomographic reconstruction of the isolator density-field along the spanwise direction. Hence, the calculated isolator exit Mach number and subsequently the inlet-isolator performances by using BOS method will be more reliable and accurate.
The method was first presented by Dalziel et al.\textsuperscript{210-212} in late 1990s as “synthetic schlieren”. They demonstrated that the displacements of light-ray path from random dots set up on the background of a changing density volume can be processed by using cross-correlation software borrowed from Particle Image Velocimetry (PIV) methods. The results are in the form of light-ray path-integrated density-gradient field, which explains why they associated it with the classical schlieren technique. At about the same time, Meier independently introduced and patented the concept of BOS in Germany\textsuperscript{213}.

This relatively new method has very simple setup, which requires only high resolution camera (typically Digital Single-Lens Reflex (DSLR) camera) and suitable background image, thus making it adaptable for many applications. For example, Richard and Raffel visualised compressible vortices from spinning helicopter rotor blade\textsuperscript{214} using BOS. They took images of a hovering helicopter with the camera positioned at an angle such that the tip of the rotor blade was set against background grass. The vortices from the blade tips were shown to be interacting with its engine exhaust plume. Elsinga \textit{et al.}\textsuperscript{99,100} had compared this method with another quantitative schlieren method called Calibrated Colour Schlieren (CCS) in analysing a two-dimensional wedge-plate model in flow Mach number of 1.94. BOS and CCS were shown to be effective in visualising shock and expansion wave. The resultant flow field properties calculated using BOS were close to analytical value predicted from oblique shock relations and Pandtl-Meyer expansion theory for that particular geometry. Venkatakrishnan and Meier\textsuperscript{208} applied this method to predict the flow around axisymmetric cone-cylinder geometry in a freestream Mach number of 2. Filtered back projection algorithm introduced in the article allows the authors to obtain slices of two-dimensional density-field around the model. The calculated density matched published data of cone tables\textsuperscript{215}. There are many
other exciting and interesting examples of recent applications of BOS in flow-diagnostics, and this thesis could only cite but a few. However, there has been no attempt yet to apply this method for scramjet inlet-isolator investigations.

D.2 Basic BOS Theory

![Figure 0-7](image)

Figure 0-7 Deflection of light path from background image due to change in density gradient (figure taken from Vekatakrishnan and Meier208)

The theory for BOS is relatively simple and can be described using Figure 0-7 above. In the figure, x-axis is parallel to freestream and starts at model tip at half-symmetry plane, y-axis is parallel with vertical plane and z-axis is along the line of sight. Random patterns of suitable shapes and sizes are printed onto single image and fixed on the background plane. The background plane is positioned at a distance $Z_D$ from the middle plane of density gradient volume investigated. The lens of the image capture device is located at length $Z_B$ from the background plane. $Z_i$ in the figure above denotes the focal length of the lens, where the light rays from the background plane converged onto the camera chip. From the figure, consider that as the volume changes its density, the light ray from a particular point on the
background plane will shift by $\Delta y$ on the camera chip. This displacement can be calculated by using PIV cross-correlation software using inputs of two still images of “wind-off” and “wind-on”. Wind-off is the initial condition where the density field is known and is usually the wind-tunnel condition without flow. Wind-on, as the name implies, refers to the condition where the density field changes due to wind tunnel freestream flow. This displacement is due to the light path being deflected vertically by angle $\varepsilon_y$. The deflection angle is typically very small and can be obtained by:

**Equation 0-17**

$$\sin (\varepsilon_y) \approx \tan (\varepsilon_y) \approx \varepsilon_y = \frac{\Delta y'}{Z_D}$$

Here, $\Delta y'$ denotes the virtual displacement of the point on background image and can be related to $\Delta y$ displacement detected by the camera by:

**Equation 0-18**

$$\frac{\Delta y'}{Z_B} = \frac{\Delta y}{Z_i}$$

The deflection angle $\varepsilon_y$ can be related to change in refractive index $\frac{\delta n}{\delta y}$ of the volume by:

**Equation 0-19**

$$\varepsilon_y = \frac{1}{n_0} \int_{Z_D}^{Z_D + \Delta Z_D} \frac{\delta n}{\delta y} dz$$

Here, $n_0$ denotes the initial refractive index of the flow, “wind-off” conditions prior to change in density field. The change in refractive index $\frac{\delta n}{\delta y}$ can be related to density gradient $\frac{\delta \rho}{\delta y}$ by partially-deriving the Gladstone-Dale equation:
Equation 0-20
\[
\frac{\delta(n - 1)}{\delta y} = \frac{\delta(k \rho)}{\delta y}
\]

Equation 0-21
\[
\frac{\delta n}{\delta y} = k \frac{\delta \rho}{\delta y}
\]

Here, \(k\) is the Gladstone-Dale constant and its value depends on type of gas used.

Equation 0-17 to Equation 0-21 can be similarly applied to consider the light path deflection on x-axis. Thus, the final density gradient at any point in the flow field is a product of \(\frac{\delta \rho}{\delta x}\) and \(\frac{\delta \rho}{\delta y}\).

D.3 Cross-correlation Theory

The background images of wind-off and wind-on can be discretized into a finite number of interrogation windows for cross-correlation analysis. The windows must be of the same size, \(W\), and contains enough shape elements for analysis. Since a window could contain many elements that have different displacement magnitudes, it is best to have the smallest window size possible to have a good representative of average particles displacement (see Figure 0-8). The cross-correlation function can be calculated by\(^{216}\):

![Figure 0-8 Movement of elements in an interrogation window for cross-correlation analysis](image)
Equation 0-22

$$\phi(\Delta x, \Delta y) = \frac{\sum_{x,y} W (I_a(x, y) - \bar{I}_a)(I_b(x + \Delta x, y + \Delta y) - \bar{I}_b)}{\left(\sum_{x,y} W (I_a(x, y) - \bar{I}_a)^2 \sum_{x,y} W (I_b(x, y) - \bar{I}_b)^2\right)^{0.5}}$$

Here, \(I_a\) denotes the pixel intensities at location \((x, y)\) within the interrogation window in the wind-off image. Similarly, \(I_b\) denotes the pixel intensities at location \((x, y)\) within the interrogation window in the wind-on image. \(\bar{I}_a\) and \(\bar{I}_b\) are the average \(I_a\) and \(I_b\) for the whole window size. \(\phi(\Delta x, \Delta y)\) is the correlation function in terms of displacement in x-y axis, \(\Delta x\) and \(\Delta y\). The function reaches peak value at the coincident points of elements match between wind-off and wind-on images.

Depending on the interrogation window size, the computing requirements could be really expensive. Thus, Fast Fourier Transform (FFT) approximation is typically used to produce each displacement vector\(^{217}\). However, Pust\(^{217}\) had shown that Direct Cross Correlation (DCC), which directly solves the equation without approximation, is significantly better in producing realistic results in comparison to FFT.

### D.4 Experimental Setup

The BOS setup for current scramjet inlet-isolator model was not so simple as it was thought initially. This is due to the sensitivity of BOS system, which requires the model to be located halfway between the background and camera lens (i.e. \(Z_D = 0.5Z_b\))\(^{208,218}\). Hargather\(^{218}\) demonstrated that as the model is gradually moved towards the background, the sensitivity of system drops significantly. The problem with this requirement is that with such small isolator height of only 6.8 mm, a sufficient compromise of camera focusing could not be found. If the lens is focused on the background image, then isolator section would appear
heavily blurred, rendering the resultant image useless. On the other hand, if the focusing is made onto the middle plane of the isolator section, then the patterns on the background image would be too out of focus to be detected by cross-correlations software. Compromise was made by applying the background image onto one sidewall, while the other sidewall provided the optical access (see Figure 0-9). Only base-cowl model was used for this first trial experiment.

Figure 0-9 Schematic of scramjet inlet-isolator with image of suitable pattern glued on the inside of sidewall
The setup for BOS experiments is shown in Figure 0-10. The camera used was Canon DSLR EOS-600D with lens of focal length 135 mm. The camera was positioned as such the lens touched the test-section window. This was done in order to increase the ratio of $Z_D/Z_B$ and thus increasing the sensitivity in measurement. The camera was set to capture a wind-off image after the test section was vacuumed right before the tunnel starting. The camera was also set to capture a series of image at 4 fps during the tunnel run, and the best image was chosen for wind-on. The two images chosen were then processed together to obtain the desired density gradient field.

Two red-head lamps with power of 650 W each was directed towards a large white screen. The reflective lights from the screen gave uniform illumination to the background image.
printed on sidewall. The experiments were conducted in total darkness except for the two light sources to obtain the best contrast for easy detection of particles deflection.

The pattern for the background image was made using open source general PIV processing software, PIVmat© written219 in MATLAB© environment. The software was set to draw 700,000 black particles of diameter 0.15 mm each in random pattern on an A4 sized image. Effectively, there were 11.22 particles/mm² and filled surface ratio was 0.198. The image was then printed on projector transparency sheet (A4 size) to allow for background illumination of the particles. Small rectangular strip of size 55 × 6.8 mm² was cut and pasted onto the inner face of sidewall as shown in Figure 0-9.
D.5 Results and Discussions

The raw pre-processed images of wind-off and wind-on conditions are shown in Figure 0-11. Any changes in particles cluster shapes and locations cannot be detected by naked eye. The model also experienced slight clockwise rotation in Figure 0-11 (b) due to heavy downwards force on the compression ramps. Even though the rotation was very subtle and cannot be observed visually, it is still of significant magnitude in comparison to each particle deflection value, and it can influence their final value as calculated by cross-correlation software. Thus, it must first be corrected prior to processing. The model movement was corrected using ‘cp2tform’ function in MATLAB 7.10.0 (R2010a), where control points were selected on both
figures and the algorithm automatically applied rotation and translation on Figure 0-11 (b), so that the control points on both figures agree with each other.

Both figures were processed by using open source PIVlab \textsuperscript{©} cross-correlation software written by Thielicke and Stamhuis\textsuperscript{220}. The software’s capability to cross-correlate PIV data has been demonstrated in papers by Mahanti \textit{et al.}\textsuperscript{221}, Mirsepassi and Dunn-Rankin\textsuperscript{222} and Ryerson and Schwenk\textsuperscript{223}. The small isolator size prohibited the use of small interrogation window size. This is due to requirements of sufficient particle elements in each interrogation window. Using smaller particles size and higher particles density in the background image might circumvent this issue, but they were limited by the printing capabilities. Thus, to overcome this problem, interrogation window of the size 192 × 192 pixel\textsuperscript{2} was chosen, with overlap value of 96%. Spatial resolution was improved significantly with high overlap value, but it comes with large computing cost. If using DCC scheme, rendering all displacement vectors took about 7 hours on a Dell M6300 mobile workstation with Core2Duo Extreme X7900 (2.8 Ghz) processor and 4 GB of RAM. The images in Figure 0-11 were also processed using FFT cross-correlation scheme, and the results came at much faster of less than 5 minutes on the same laptop. Comparison of the density gradient using standard colour schlieren, DCC and FFT are given in the figure below:
Figure 0-12 Schlieren images of baseline scramjet inlet-isolator model; (a) colour-schlieren (b) rendered from density gradient calculated using Direct Cross Correlation (DCC) and (c) rendered from density gradient calculated using Fast-Fourier Transform Cross Correlation (FFT-CC)

From Figure 0-12, it was observed that the BOS setup used in this experiment could not provide the required sensitivity to change in density. Only shoulder separation shock could be identified on both DCC and FTT cross-correlation scheme. Other shocks and flow phenomena, which could be observed easily in a normal schlieren setup, could not be spatially resolved with BOS. PIVlab© software produced a lot of spurious vectors near the bottom and top wall of isolator section, thus they are neglected from Figure 0-12 (b) and (c). This could be due to low particles density around the region or low signal-to-noise ratio.

There were significant difference between DCC and FFT cross-correlation scheme. In some region in the flow field, the density-gradient predicted using DCC could reach about twice the value calculated using FTT.
The issue of BOS sensitivity has been addressed in previous section, and it is known that the model should ideally be located halfway between background image and camera lens. For this current case, this is not possible due to very small isolator height preventing satisfactory focusing of background image viewed through the isolator. One possible way to increase the sensitivity for the current setup is to modify the setup according to the guidance from Tokgoz et al.\textsuperscript{224}. They measured the density and temperature changes in a fluid layer by using reflective surface.

![BOS setup with reflective surface](image)

**Figure 0-13 BOS setup with reflective surface (figure taken from Tokgoz et al.\textsuperscript{224})**

Consider Figure 0-13, where optical access is provided only on one side. Tokgoz et al.\textsuperscript{224} attached a reflective surface on the measurement plane, which contains the reflection of background image situated at an angle to the plane. This angle must be similar to camera optical plane viewing angle $\gamma$. The authors demonstrated that the setup has satisfactory sensitivity and could measure temperature difference of 1 K. The reflective surface BOS concept could allow a better result scramjet inlet-isolator analysis.
D.6 Conclusions

An exploratory study of applying BOS method to characterize a scramjet inlet-isolator has been performed. The setup could not be simpler, requiring only a DSLR camera and suitable background image. The processing software needed for rendering the density-gradient field is open-source and easily available. Using BOS to characterize the scramjet inlet-isolator would save a lot of cost in comparison to stream thrust analysis and pitot rake measurement. This method is also cheaper than using PSP method discussed in this thesis. However, the sensitivity of current BOS system for scramjet inlet-isolator application must be perfected first. This method has large potential for future studies.