

Investigation of the Isolator Flow of Scramjet Engines

Von der Fakultät für Maschinenwesen der
Rheinisch-Westfälischen Technischen Hochschule Aachen
zur Erlangung des akademischen Grades eines Doktors der
Ingenieurwissenschaften genehmigte Dissertation

vorgelegt von

Christian Max Fischer

Berichter: Universitätsprofessor Dr.-Ing. H. Olivier
 Universitätsprofessor Dr.-Ing. W. Schröder

Tag der mündlichen Prüfung: 10.06.2014

Diese Dissertation ist auf den Internetseiten der Hochschulbibliothek
online verfügbar.

Bibliografische Information der Deutschen Nationalbibliothek

Die Deutsche Nationalbibliothek verzeichnet diese Publikation in der Deutschen Nationalbibliografie; detaillierte bibliografische Daten sind im Internet über <http://dnb.d-nb.de> abrufbar.

ISBN 978-3-8439-1688-2

D 82 (Diss. RWTH Aachen University, 2014)

© Verlag Dr. Hut, München 2014
Sternstr. 18, 80538 München
Tel.: 089/66060798
www.dr.hut-verlag.de

Die Informationen in diesem Buch wurden mit großer Sorgfalt erarbeitet. Dennoch können Fehler nicht vollständig ausgeschlossen werden. Verlag, Autoren und ggf. Übersetzer übernehmen keine juristische Verantwortung oder irgendeine Haftung für eventuell verbliebene fehlerhafte Angaben und deren Folgen.

Alle Rechte, auch die des auszugsweisen Nachdrucks, der Vervielfältigung und Verbreitung in besonderen Verfahren wie fotomechanischer Nachdruck, Fotokopie, Mikrokopie, elektronische Datenaufzeichnung einschließlich Speicherung und Übertragung auf weitere Datenträger sowie Übersetzung in andere Sprachen, behält sich der Autor vor.

1. Auflage 2014

Danksagung

Diese Arbeit entstand während meiner Zeit als wissenschaftlicher Mitarbeiter und Stipendiat am Stosswellenlabor der RWTH Aachen. Sie wurde hauptsächlich finanziert durch die Deutsche Forschungsgemeinschaft (DFG) im Rahmen des Graduiertenkolleg "Aero-thermodynamische Auslegung eines Scramjet Antriebssystems für zukünftige Raumtransportsysteme".

Mein herzlicher Dank gilt Herrn Prof. Dr.-Ing. Herbert Olivier, dem Leiter des Stosswellenlabors für die Möglichkeit zu dieser interessanten Forschungstätigkeit und die Betreuung der Arbeit. Herrn Prof. Dr.-Ing. Wolfgang Schröder danke ich für das gezeigte Interesse an der Arbeit und die Übernahme des Korreferates.

Danken möchte ich weiterhin allen Kollegen und Mitstreitern am Stosswellenlabor für eine ganz besondere Arbeitsatmosphäre. Dabei ging es nicht nur um fachlichen Rat und Mithilfe oder die konstruktive Mithilfe bei unzähligen Probevorträgen. Auch die sportliche Ausgestaltung sommerlicher Mittagspausen oder Mithilfe bei einem legendären Grillabend im Rahmen einer Summerschool dürfen nicht vergessen werden.

Zusätzlicher Dank gilt auch der Institutswerkstatt unter der Leitung von Herrn Markus Eichler. Egal ob es darum ging Teile präzise zu fertigen, einen abgerissenen Stecker neu zu verdrahten oder auch über Nacht neues Öl für eine heiß gelaufene Pumpe zu beschaffen. Die unzähligen Versuche mit verschiedenen Modellen im Rahmen dieser Arbeit wären ohne ihre Unterstützung nicht möglich gewesen.

Ein ganz besonderer Dank gilt meiner Freundin und zukünftigen Frau Nadine, die die Entstehung dieser Arbeit mit all ihren Höhen und Tiefen mit durchlebt hat.

Christian Fischer

Juli 2014

Kurzfassung

Im Mittelpunkt dieser Arbeit steht der Isolator eines Scramjet Triebwerkes. Mit Hilfe verschiedener Windkanalmodelle wird der Einfluss der Wand- und Totaltemperatur auf das Strömungsfeld untersucht. Dabei wird die Wandtemperatur zwischen 300 K und 1000 K sowie die Totaltemperatur der Strömung zwischen 1150 K und 2200 K variiert. Die Anströmmachzahl des Scramjet Triebwerkes beträgt $Ma_\infty = 7.5$. Ergänzt werden die Versuchsdaten durch numerische Strömungssimulationen.

In einem typischen Scramjet Isolator mit Einlauf wird der Shock Train sowie eine Stoß/Grenzschicht-Wechselwirkung untersucht. Dabei zeigt sich, dass für diesen Strömungsfall das Verhältnis von Wand- zu Totaltemperatur ein guter Ähnlichkeitsparameter ist. Mit steigendem Verhältnis von Wand- zu Totaltemperatur wächst der Shock Train bei konstantem Gegendruck stromauf weiter in den Isolator hinein.

Allerdings beeinflusst der Einlauf das in den Isolator eintretende Strömungsfeld, z.B. durch Wandtemperatureffekte. Auch erzeugt der hier untersuchte Einlauf ein asymmetrisches Strömungsprofil. Um ausschließlich den Wandtemperatureffekt auf den Shock Train zu untersuchen wird ein Isolatormodell mit rechteckigem Querschnitt und homogener sowie wandtemperaturunabhängiger Einstromung untersucht. Die Machzahl dieser Einstromung wird zwischen $Ma = 2$ und $Ma = 3.5$ variiert. Es zeigt sich, dass bei konstanter Totaltemperatur die Länge des Shock Trains mit steigender Wandtemperatur abnimmt. Der Effekt ist besonders stark wenn sich die Wandtemperatur der Recoverytemperatur annähert. Dadurch ist das Verhältnis von Wand- zu Totaltemperatur kein Ähnlichkeitsparameter. Die existierende Korrelation von Waltrup und Billig kann den beobachteten Effekt nicht quantitativ beschreiben. Sie wird basierend auf den Messergebnissen entsprechend erweitert und beinhaltet nun den Einfluss des Wandwärmestroms.

Der Einfluss der Breite auf das Isolatorströmungsfeld mit rechteckigem Querschnitt wurde mit einem zusätzlichen Modell untersucht. Hier wurde die Breite des Isolators variiert. Die zusätzliche Kompression durch die Seitenwände führt zu kürzeren Shock Trains bei gleichem Gesamtdruckanstieg im Vergleich zu einem annähernd zweidimensionalen Strömungsfeld. Mit steigendem Verhältnis von Wand- zu Querschnittsfläche (Annäherung an einen quadratischen Querschnitt) steigen die Verluste durch die Grenzschicht und der Shock Train wird wieder länger.

Contents

Contents.....	I
List of Figures	III
Tables	IX
Nomenclature	XI
1 Introduction.....	1
1.1 Motivation	1
1.2 Research training group 1095	3
1.2.1 Overview	3
1.2.2 Context for presented research	3
2 Fundamentals.....	5
2.1 Navier Stokes equations	5
2.2 Boundary layer flow	8
2.2.1 Laminar boundary layers	8
2.2.2 Turbulent boundary layers	16
2.3 Shock boundary layer interaction	20
2.4 Scramjets	24
2.4.1 Scramjet development in Australia	29
2.4.2 Scramjet development in the USA	30
2.5 Shock trains	31
3 Experimental facility, measurement technique and numerical method	43
3.1 Hypersonic shock tunnel TH2	43
3.1.1 Principle of the shock tunnel	44
3.1.2 Free stream conditions	47
3.1.3 Measurement time	49
3.2 Measurement technique.....	56
3.2.1 Pressure measurements	56
3.2.2 Uncertainty of measured values	57
3.2.3 Schlieren and shadowgraph visualisation.....	57
3.3 Numerical method	60
3.3.1 Mesh splitting	60
3.3.2 Mesh sensitivity	61
4 Model Technique	65
4.1 Model heating technique	65

4.2	The Scramjet intake model	68
4.2.1	Flow field	68
4.2.2	Model modification for isolator investigation	69
4.2.3	Generation of pressure rise	71
4.2.4	Instrumentation	72
4.3	Heatable isolator model	73
4.3.1	Flow field	74
4.4	Cold isolator model	76
5	Results and discussion	79
5.1	Scramjet model	79
5.1.1	Basic flow field	79
5.1.2	Lip shock induced separation bubble	80
5.1.3	Shock train	88
5.1.4	Boundary layer suction	101
5.1.5	Comparison with DLR results	104
5.2	Heatable isolator model	107
5.2.1	Isolator entrance flow conditions	107
5.2.2	Basic flow field	107
5.2.3	Variation of back pressure	108
5.2.4	Wall temperature effects	116
5.2.5	Quantification of wall temperature effect	117
5.3	Model with variable isolator flow channel width	129
6	Conclusion	135
7	References	139
8	Appendix	149

List of Figures

Fig. 1-1: Specific impulse of air breathing propulsion systems compared to rocket engines [18]	1
Fig. 1-2: Basic setup of a Scramjet engine	2
Fig. 2-1: Boundary layer profiles (velocity left and temperature right) calculated with the Van Driest Technique with varying edge Mach number; $T_w = T_e = 300\text{ K}$ [35]	12
Fig. 2-2: Skin friction coefficient for a laminar and a turbulent boundary layer obtained with different methods	18
Fig. 2-3: Stanton number for a laminar and a turbulent boundary layer obtained with different methods	18
Fig. 2-4: Displacement thickness for a laminar and a turbulent boundary layer obtained with different methods	19
Fig. 2-5: Momentum thickness for a laminar and a turbulent boundary layer obtained with different methods	19
Fig. 2-6: Sketch of shock wave boundary layer interaction at a compression ramp (left) and caused by an impinging shock (right)	20
Fig. 2-7: Pressure distribution at a shock boundary layer interaction [66] (left) and sketch of the complete SWBL (right)	21
Fig. 2-8: Incipient separation pressure as a function of wall temperature	22
Fig. 2-9: Wall temperature effect on the separation length for three different correlations	24
Fig. 2-10: Cycle of thermal engines [34]	25
Fig. 2-11: Wall pressure distributions in combustion chamber without fuel injection [97]	27
Fig. 2-12: Wall pressure distributions in combustor with fuel injection (experimental data) [97]	27
Fig. 2-13: Combustion pressure rise as a function of total temperature ratio for different combustor area ratios at $Ma_3 = 2.5$ [5]	28
Fig. 2-14: Plane definition for hypersonic engines	29
Fig. 2-15: Pressure distribution in the HyShort II combustor with and without fuel injection [84]	30
Fig. 2-16: Drawing of a normal shock train flow field	31
Fig. 2-17: Drawing of an oblique shock train flow field with supersonic exit flow	32
Fig. 2-18: Pressure distribution at the wall and on the centerline of a normal shock train for two different back pressures and pressure contours for the two cases [88] ...	33
Fig. 2-19: Drawing of normal shock train flow field [12]	34

<i>Fig. 2-20: Shock Trains at $Ma_2 = 1.95$ with (bottom) and without forced transition of the boundary layer from [26]</i>	<i>41</i>
<i>Fig. 2-21: Shock Trains at $Ma_2 = 2.46$ with (top) and without forced transition of the boundary layer from [26]</i>	<i>42</i>
<i>Fig. 3-1: Drawing of the hypersonic shock tunnel TH2 in helium driven mode [66]</i>	<i>43</i>
<i>Fig. 3-2: Drawing of the shock tube section after bursting of the diaphragms ($t > 0$) [35]</i>	<i>45</i>
<i>Fig. 3-3: x-t diagram of the shock tube process in the TH2 facility in helium driven mode [35]</i>	<i>46</i>
<i>Fig. 3-4: Free stream conditions in the test section during an experiment at Condition I..</i>	<i>49</i>
<i>Fig. 3-5: Free stream conditions in the test section during an experiment at Condition I with a helium air mixture</i>	<i>50</i>
<i>Fig. 3-6: Pressure distribution in isolator model for Condition I, $Ma_2 = 3.5$, $T_W = 1000K$ with and without back pressure</i>	<i>51</i>
<i>Fig. 3-7: Wall pressure coefficients for different probe positions for the experiment shown in Fig. 3-6</i>	<i>52</i>
<i>Fig. 3-8: Wall pressure coefficients for different probe positions for the experiment shown in Fig. 3-6 (continued)</i>	<i>53</i>
<i>Fig. 3-9: Wall pressure coefficients for different probe positions of one experiment</i>	<i>54</i>
<i>Fig. 3-10: Wall pressure coefficients for different probe positions of one experiment (continued)</i>	<i>55</i>
<i>Fig. 3-11: Shadowgraph (top,) x-wise schlieren (middle) and y-wise schlieren (bottom) image for the same flow field.</i>	<i>58</i>
<i>Fig. 3-12: Operating principle for color schlieren (left) and one-directional schlieren (right) imaging</i>	<i>59</i>
<i>Fig. 3-13: Domains for mesh splitting</i>	<i>60</i>
<i>Fig. 3-14: Pressure distribution in the front part of the isolator for different meshes</i>	<i>62</i>
<i>Fig. 3-15: Pressure distribution in the rear part of the isolator with back pressure for different meshes</i>	<i>62</i>
<i>Fig. 3-16: Isolator pressure distribution without back pressure with different 2d and 3d meshes</i>	<i>63</i>
<i>Fig. 4-1: Cut through the cooling section [66]</i>	<i>65</i>
<i>Fig. 4-2: Temperature distribution on the wetted surfaces of the isolator at a nominal wall temperature of $T_W = 1000 K$ on the centerline in streamwise direction</i>	<i>67</i>
<i>Fig. 4-3: Temperature distribution on the wetted surfaces of the isolator at a nominal wall temperature of $T_W = 1000 K$ normal to flow direction at $x = 25 mm$</i>	<i>67</i>
<i>Fig. 4-4: Drawing of Scramjet intake model</i>	<i>68</i>
<i>Fig. 4-5: Scramjet intake model geometry with numerical schlieren image of intake</i>	<i>68</i>

Fig. 4-6: Mounting principle for isolator lip	69
Fig. 4-7: Drawing of bar underneath support parts (left) and drawing of lip fixation and additional bar to prevent sidewise opening of flow channel	70
Fig. 4-8: Old (top) and new (bottom) sidewall and new window frame (upper right corner).....	71
Fig. 4-9: Drawing of double wedge for back pressure generation	72
Fig. 4-10: Pressure probe positions in the isolator of the Scramjet model with shadowgraph image (Cond. I, $T_w = 300$ K).....	73
Fig. 4-11: Picture of the heatable isolator model in the TH2 test section	73
Fig. 4-12: Relation of isolator entrance Mach number Ma_2 and model angle of attack.....	75
Fig. 4-13: Basic flow field of isolator model	75
Fig. 4-14: Expected shock train length in isolator model as function for the overall pressure rise at different Mach numbers.....	76
Fig. 4-15: Layout of cold isolator model with varying isolator width	77
Fig. 4-16: Picture of cold isolator model	78
Fig. 4-17: Cut through cold isolator model.....	78
Fig. 5-1: Shadowgraph picture from experiment, schlieren picture with streamlines from numerical simulation and pressure distribution on ramp (bottom) and lip (top) for Condition I, $T_w = 300$ K.....	80
Fig. 5-2: Pressure distribution in the isolator at different wall temperatures for Condition I.....	81
Fig. 5-3: Color schlieren images ($T_w = 300$ K and $T_w = 1000$ K) and shadowgraph images ($T_w = 600$ K and $T_w = 800$ K) of lip shock induced separation bubble at Condition I.....	82
Fig. 5-4: Pressure distribution in the isolator at different wall temperatures for Condition II.....	83
Fig. 5-5: Shadowgraph images of lip shock induced separation bubble at Condition II at different wall temperatures	84
Fig. 5-6: Wall pressure distribution in the front part of the isolator for different wall temperatures at Condition I from numerical simulations with transition modeling	86
Fig. 5-7: Wall pressure distribution in the front part of the isolator for different wall temperatures at Condition I from numerical simulations with forced transition.....	86
Fig. 5-8: Schlieren images with stream lines created from numerical simulations for different wall temperatures at Condition I.....	87

<i>Fig. 5-9: Pressure distributions on ramp (bottom) and lip (top) for different pressure ratios, Condition I, $T_w = 300$ K, shadowgraph images in the middle</i>	90
<i>Fig. 5-10: Mach number profiles at isolator exit (pitot rake position) for different back pressures at Condition I, $T_w = 300$ K</i>	91
<i>Fig. 5-11: Detail of rear isolator section for pressure ratio 2.6 (frame in Fig. 5-9)</i>	91
<i>Fig. 5-12: Comparison between numerical and experimental results: pressure distribution on the lip (top), color schlieren image from experiment and schlieren image from numerical result (middle) and pressure distribution on ramp the (bottom), Condition I, $T_w = 600$ K</i>	93
<i>Fig. 5-13: Schlieren image from numerical result with back pressure, detail of rear isolator with stream lines, Condition I, $T_w = 600$ K</i>	94
<i>Fig. 5-14: Comparison between numerical and experimental results: Mach number profiles at isolator exit, Condition I, $T_w = 600$ K</i>	95
<i>Fig. 5-15 : Wall temperature influence: pressure distribution on the ramp (bottom), schlieren and shadowgraph images from these experiments (middle) and pressure distribution on the lip (top), Condition I</i>	96
<i>Fig. 5-16: Wall temperature influence: pressure distribution on the lip (top) with schlieren and shadowgraph images, Condition I</i>	97
<i>Fig. 5-17: Wall and total temperature influence: pressure distribution on the lip (top) with color schlieren images for the cases at $T_w = 1000$ K (bottom)</i>	99
<i>Fig. 5-18: Wall and total temperature influence: pressure distribution on the ramp (top) with shadowgraph images (bottom)</i>	100
<i>Fig. 5-19: Schlieren images created from numerical results for front part of the isolator without (top) and with different boundary layer suction geometries</i>	101
<i>Fig. 5-20: Schlieren images created from numerical results for front part of the isolator with different boundary layer suction geometries (continued)</i>	102
<i>Fig. 5-21: Comparison of pressure distributions on the ramp of SWL and DLR experiments with varying back pressure and different wall temperatures (SWL model only)</i>	104
<i>Fig. 5-22: Comparison of pressure distributions on the lip of SWL and DLR experiments with varying back pressure and different wall temperatures (SWL model only)</i>	105
<i>Fig. 5-23: Schlieren image (bottom) and wall pressure distribution (top) at Condition I, $Ma_2 = 3$, $T_w = 800$ K</i>	108
<i>Fig. 5-24: Wall pressure distributions at Condition I, $Ma_2 = 3.0$, $T_w = 800$ K with different back pressures</i>	109
<i>Fig. 5-25: Schlieren and shadowgraph images for the flow cases shown in Fig. 5-24</i>	109

<i>Fig. 5-26: Wall pressure distributions at Condition I, $Ma_2 = 3.5$, $T_W = 1000$ K with different back pressures</i>	110
<i>Fig. 5-27: Shadowgraph images of the rear part of the isolator for the flow cases shown in Fig. 5-26</i>	111
<i>Fig. 5-28: Wall pressure distributions at Condition I, $Ma_2 = 2.0$, $T_W = 300$ K with different back pressures</i>	112
<i>Fig. 5-29: Schlieren images for the flow cases shown in Fig. 5-28</i>	112
<i>Fig. 5-30: Wall pressure distributions at Condition II, $Ma_2 = 3.0$, $T_W = 300$ K with different back pressures</i>	113
<i>Fig. 5-31: Shadowgraph images for two flow cases of the wall pressure distribution in Fig. 5-30</i>	114
<i>Fig. 5-32: Wall pressure distribution in the isolator without back pressure at Conditions I and II from experiments compared to laminar and turbulent CFD calculations</i>	115
<i>Fig. 5-33: Shadowgraph images with constant pressure rise and varying wall temperature at Condition I; $Ma_2 = 3$; shock train starting points for $T_W = 300$ K and $T_W = 1000$ K marked by vertical lines</i>	116
<i>Fig. 5-34: Wall pressure distribution with constant pressure rise and varying wall temperature at Condition I, $Ma_2 = 3$</i>	117
<i>Fig. 5-35: Shock train pressure distribution for different wall temperatures at Condition I, $Ma_2 = 3.0$</i>	118
<i>Fig. 5-36: Shock train pressure distribution for different wall temperatures at Condition II, $Ma_2 = 3.0$</i>	119
<i>Fig. 5-37: Shock train pressure distribution for different wall temperatures at Condition I, $Ma_2 = 3.5$</i>	120
<i>Fig. 5-38: Shock train pressure distribution for different wall temperatures at Condition I, $Ma_2 = 2.0$</i>	121
<i>Fig. 5-39: Correlation parameter A for different wall temperatures normalized with the values for $T_W = 300$ K</i>	124
<i>Fig. 5-40: Boundary layer momentum thickness at position $x = 0.18$ m for different wall temperatures normalized with the values for $T_W = 300$ K</i>	124
<i>Fig. 5-41: Correlation parameter A normalized with $C = [2 \cdot \delta_2 / h]^{0.25}$ for different wall temperatures normalized with the values for $T_W = 300$ K</i>	125
<i>Fig. 5-42: Stanton number for undisturbed flow at isolator exit ($x = 0.18$ m) and for different wall temperatures normalized with the values for $T_W = 300$ K</i>	126
<i>Fig. 5-43: Correlation parameter A normalized with $C = [(\dot{q}_w A_w) / (\dot{m} c_p T_0)]^{0.25}$ for different wall temperatures, normalized with the values for $T_W = 300$ K</i>	127

<i>Fig. 5-44: Correlation parameter A normalized with C from equation 5.7 for different wall temperatures normalized with the values for $T_W = 300$ K.....</i>	<i>129</i>
<i>Fig. 5-45: Pressure distributions for 200 mm flow channel width.....</i>	<i>130</i>
<i>Fig. 5-46: Pressure distributions for 150 mm flow channel width.....</i>	<i>130</i>
<i>Fig. 5-47: Pressure distributions for 70 mm flow channel width.....</i>	<i>131</i>
<i>Fig. 5-48: Wall pressure rise along the shock train from model with variable width at Condition I, $Ma_2 = 3.0$.....</i>	<i>132</i>
<i>Fig. 5-49: Wall pressure distribution without back pressure at Condition. I, $Ma_2 = 3.0$...</i>	<i>133</i>
<i>Fig. 8-1: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.5$, $T_W = 300$ K.....</i>	<i>149</i>
<i>Fig. 8-2: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.5$, $T_W = 600$ K.....</i>	<i>149</i>
<i>Fig. 8-3: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.5$, $T_W = 800$ K.....</i>	<i>150</i>
<i>Fig. 8-4: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.0$, $T_W = 300$ K.....</i>	<i>150</i>
<i>Fig. 8-5: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.0$, $T_W = 600$ K.....</i>	<i>151</i>
<i>Fig. 8-6: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.0$, $T_W = 1000$ K.....</i>	<i>151</i>
<i>Fig. 8-7: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 2.5$, $T_W = 300$ K.....</i>	<i>152</i>
<i>Fig. 8-8: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 2.0$, $T_W = 600$ K.....</i>	<i>152</i>
<i>Fig. 8-9: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 2.0$, $T_W = 800$ K.....</i>	<i>153</i>
<i>Fig. 8-10: Wall pressure distribution for different back pressure at Condition II, $Ma_2 = 3.0$, $T_W = 600$ K.....</i>	<i>153</i>
<i>Fig. 8-11: Wall pressure distribution for different back pressure at Condition II, $Ma_2 = 3.0$, $T_W = 800$ K.....</i>	<i>154</i>
<i>Fig. 8-12: Wall pressure distribution for different back pressure at Condition II, $Ma_2 = 3.0$, $T_W = 1000$ K.....</i>	<i>154</i>

Tables

<i>Tab. 2-1: Flow condition for boundary layer calculation.....</i>	<i>17</i>
<i>Tab. 3-1: Design conditions of the research training group Scramjet.....</i>	<i>44</i>
<i>Tab. 3-2: Free stream condition at the models leading edge and gradients of free stream conditions.....</i>	<i>48</i>
<i>Tab. 3-3: Modified free stream conditions for numerical simulations.....</i>	<i>61</i>
<i>Tab. 4-1: Design parameters for the heatable isolator model.....</i>	<i>76</i>
<i>Tab. 5-1: Ratio between wall and total temperature for different wall temperatures at conditions I and II.....</i>	<i>85</i>
<i>Tab. 5-2: Relative bleed mass flow and relative total pressure level at exit for the different bleed configurations.....</i>	<i>103</i>
<i>Tab. 5-3: Isolator entrance flow conditions.....</i>	<i>107</i>
<i>Tab. 5-4: Isolator entrance flow condition for the experiments.....</i>	<i>108</i>
<i>Tab. 5-5: Boundary layer momentum thickness [$10^{-3}m$] for undisturbed flow ($x = 0.18 m$).....</i>	<i>122</i>
<i>Tab. 5-6: Parameter A for the investigated flow cases calculated from the pressure distributions.....</i>	<i>123</i>
<i>Tab. 5-7: Stanton number [-] for undisturbed flow ($x = 0.18 m$).....</i>	<i>128</i>

Nomenclature

A	amplification factor [-], correlation parameter [-]
A*	critical area [m ²]
C	Sutherland factor [kg/(m s K ^{0.5})], measured free stream quantity, wall temperature influence factor [-]
C _H	Stanton number [-]
D	flow channel diameter [m]
H	altitude [m], flow channel height [m]
L	characteristic length [m]
L _{Mess}	pressure pipe length [m]
M	molecular weight [g/Mol]
Ma	Mach number [-]
Pr	Prandl number [-]
R	specific gas constant [J/(kg K)]
Re	unit Reynolds number [1/m]
Re _x	Reynolds number [-]
S	Sutherland temperature [K]
T	temperature [K]
T _R	recovery temperature [K]
T ₀	total temperature [K]
V	volume [m ³]
X	probe signal [V]
a ₁₋₄	thermocouple calibration factors [K/V]
b	weighting factor [-]
c	specific thermal capacity [J/(kg*K)]
c _f	skin friction coefficient [-]
c _p	pressure coefficient [-]
c _μ	power law factor [kg/(m s K ^(1-ω))]
d _i	pressure pipe inner diameter [m]
e	internal energy [J/kg]
f _e	eigenfrequency [Hz]
h	flow channel height [m]
k	thermal conductivity [W/m*K]

l	length [m]
p	pressure [Pa]
p_0	total pressure [Pa]
p_p	pitot pressure [Pa]
q	dynamic pressure [Pa], $q = 0.5\rho u^2$
$\dot{q}$	heat transfer [W/m^2]
r	recovery factor [-]
s	coordinate in x-wise direction starting at the shock train [m]
t	time [s]
t_{95}	time for throttle device to achieve 95% of the mass flow [s]
t_r	response time [s]
u	velocity [m/s], x-wise velocity [m/s]
v	y-wise velocity [m/s]
w	flow channel width [m], z-wise velocity [m/s]
x	coordinate in main flow direction starting at isolator entrance [m]
y	coordinate normal to ramp plane starting at ramp [m]
β	ramp angle [$^\circ$]
γ	ratio of the specific heats
δ	boundary layer thickness [m]
δ_1	boundary layer displacement thickness [m]
δ_2	boundary layer momentum thickness [m]
ρ	density [kg/m^3]
τ	shear stress [N/m^2]
τ_s	time constant for t_{95} [s]
μ	dynamic viscosity [kg/(m s)]
ω	power law exponent [-]

Suffix

∞	free stream condition
0	stagnation condition
2	isolator entrance condition
3	isolator exit condition
1s	low pressure section of shock tube before experiment
2s	downstream of shock in low pressure section of shock tube

3s	downstream of contact surface in low pressure section of shock tube
4s	high pressure section of shock tube before experiment
e	edge condition
C	cavity in front of pressure probe
L	shock impingement location for inviscid flow
Mea	during measurement time
p	plenum
ref	reference condition
td	throttle device
W	wall condition
x	in x-wise direction
y	in y-wise direction

Superfix

*	critical condition ($Ma = 1$)
---	---------------------------------

1 Introduction

1.1 Motivation

Today rocket propulsion systems such as the Ariane V and the Sojus rocket are used as space transportation systems. A rocket engine carries its own fuel and oxidizer and works independent from the flight Mach number and atmospheric pressure (except the supersonic nozzle) but the maximum specific impulse is very limited. The payload ratio of these systems is in the range of low single digit percentage numbers. None of the in use systems is reusable and all require staging. For a drastic reduction of the costs of transport to space reusable systems ideally without staging are needed. To compensate for the higher structural weight of these propulsion systems caused by the structural mass of the reentry system and the additional mass due to the lack of staging systems with a higher specific impulse are needed. This can be achieved by using air breathing propulsion systems.

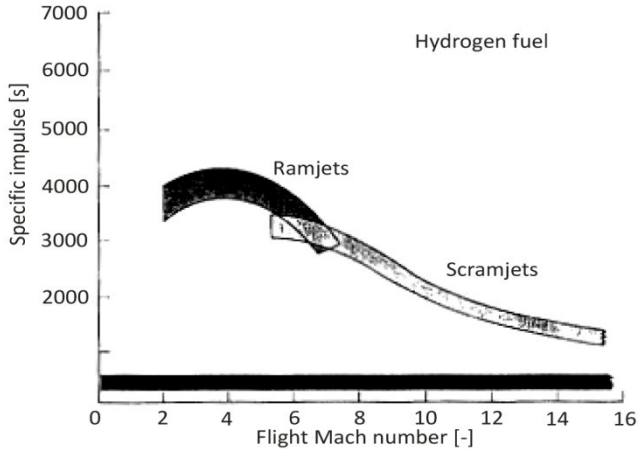


Fig. 1-1: Specific impulse of air breathing propulsion systems compared to rocket engines [18]

The potential of these air breathing systems is shown in Fig. 1-1. For supersonic flight between $Ma_\infty \sim 2$ and $Ma_\infty \sim 7$ Ramjets are used. In a Ramjet the incoming airflow is decelerated to subsonic speed. The achieved pressure rise from the deceleration is then used for the

combustion cycle. With increasing Mach number the total pressure loss created by the shocks in the compression process is increasing. From about $Ma_\infty = 6$ Scramjets offer a higher possible specific impulse. In a Scramjet (Supersonic combustion Ramjet) the air is only decelerated to lower supersonic velocities (about $1/3$ of Ma_∞ [33]). This creates lower total pressure losses but higher drag on the wetted walls due to the higher Mach number in the combustion chamber as well as the longer flow path needed to achieve mixing and combustion.

Also both Ramjet and Scramjet are not operational below a certain Mach number as they require the flight velocity to create compression. In an operational system they are also very likely to be combined (dual-mode operation) in a single flow path to enable operation over a wide Mach number range. The overall thrust created by Scramjet systems is the difference between thrust created by the system and the drag, which are both large numbers. Small changes in one of the two have a large impact on the overall system performance. Therefore a good understanding of the exact performance of every component in the system is needed. The basic setup of a Scramjet is shown in Fig. 1-2. The compression is fully achieved in the inlet. Fuel is injected from the walls or a central injector into the flow path and burned in the combustor. Fuel injection in the inlet is also possible as mixing takes a long time due to the high velocity. Thrust is then generated in the nozzle. The first section of the combustor is used as isolator. The isolator is necessary to separate the combustor from the inlet. During Ramjet operation the final normal shock and during Scramjet operation the combustion pressure rise lead to the formation of a shock train. This shock train has to be captured within the isolator. The isolator does not commit to the compression of the inlet. It has a constant cross sectional area or even a slight increase in area to counteract the buildup of the boundary layer. The goal of the design of a Scramjet engine is to make the isolator as short as possible as its wetted surfaces create drag. Therefore it is necessary to predict the flow field in the isolator which is dominated by a shock train. A detailed description of the relevant flow field will be given in chapter 2.5.

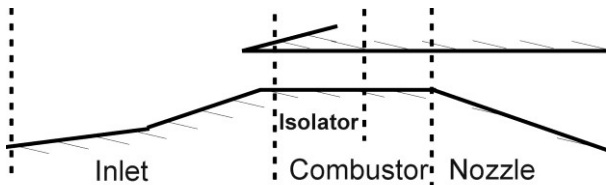


Fig. 1-2: Basic setup of a Scramjet engine

Goal of this thesis is the characterization of the isolator flow field. The focus will be on the effect of (a) elevated wall temperatures as well as the variation of (b) the total temperature.

Experiments and numerical simulations are carried out using an isolator in a typical Scramjet configuration. For a better understanding of underlying mechanisms of the shock train flow a simplified configuration with symmetric inflow will also be investigated. As both configurations feature a rectangular cross section a third model with variable width will also be investigated to study 3D effects. The results will be compared to existing correlations for shock train flows and these correlations will be expanded to cover the effects investigated.

For the Scramjet configuration the flow entering the isolator is not uniform. The final compression shock that originates from the leading edge of the lip induces a separation bubble on the ramp. The wall and total temperature effect on this separation bubble will also be investigated. It will also be discussed whether a boundary bleed is feasible here in order to reduce losses.

1.2 Research training group 1095

1.2.1 Overview

The presented research is part of Germany's research training group (GRK) 1095 "Aero-Thermodynamic Design of a Scramjet Propulsion System for Future Space Transportation Systems" which studies a Scramjet propulsion system with experimental, numerical and analytical means as well as from the conceptional point of view. It is a collaborative project of institutes at the University of Stuttgart, the Technical University of Munich, RWTH Aachen University and the DLR in Cologne. In an additional subproject combustion tests with a scramjet model are conducted at ITAM, Novosibirsk in Russia. The Research Training Group started in 2005 and is divided into 3 periods. The presented research was done in the second phase. The goal of the research training group is the design of a scramjet engine on paper. An optimized Scramjet configuration for flight tests is developed as a goal of the research training group but a flight test is not part of the project.

1.2.2 Context for presented research

The model and therewith the inlet and isolator flow path that is the core of the investigations of this thesis was designed during the first phase [31], [67]. Two almost identical inlet models were built to be tested at the DLR in Cologne as well as at the Shock Wave Laboratory (SWL) in Aachen. The tests at the DLR focused on the operational behavior of the inlet at different yaw angles [38] and additional sidewall compression [37]. A shock train was also generated with varying back pressure [29].

In the second phase of the research training group an additional three dimensional inlet was designed for a potential flight test configuration [28]. The research presented here uses

the model design from the first phase to investigate more fundamental flow phenomena in the isolator. Results from the experiments are also used as validation data for numerical subprojects [69]. An overview of the structure as well as the full research profile of the research training group can be found in [28] and [103].

2 Fundamentals

2.1 Navier Stokes equations

To reach an analytical description for any kind of flow the three basic conservation laws for mass, momentum and energy are used. The mass of a defined number of material elements is constant (eq. 2.1) while the volume can be a function of time.

$$m = \int_{V(t)} \rho dV = \text{const.} \rightarrow \frac{dm}{dt} = 0 = \frac{d}{dt} \int_{V(t)} \rho dV \quad (2.1)$$

This leads to the mass conservation equation:

$$\frac{\partial \rho}{\partial t} + \frac{\partial(\rho u)}{\partial x} + \frac{\partial(\rho v)}{\partial y} + \frac{\partial(\rho w)}{\partial z} = 0. \quad (2.2)$$

For the momentum conservation the product of mass and acceleration is the sum for all forces per volume element. They consist of mass forces per volume element $\vec{f}$ (e.g. force of gravity on the material elements; often neglected) and the surface forces per volume element $\vec{S}$ (stresses and pressure) (eq. 2.3).

$$\rho \left(\frac{D\vec{u}}{Dt} \right) = \vec{f} + \vec{S} \quad (2.3)$$

The complete derivation of the surface forces can be found in [80]. This leads to the momentum conservation of the Navier Stokes equations in x (eq. 2.4), y (eq. 2.5), and z (eq. 2.6) direction. It is assumed that the volume viscosity is of smaller magnitude than the dynamic viscosity (Stokes Hypothesis) and can therefore be neglected.

$$\begin{aligned} \rho \frac{du}{dt} = f_x - \frac{\partial p}{\partial x} + \frac{\partial}{\partial x} \left[\mu \left(2 \frac{\partial u}{\partial x} - \frac{2}{3} \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} + \frac{\partial w}{\partial z} \right) \right) \right] \\ + \frac{\partial}{\partial y} \left[\mu \left(\frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} \right) \right] + \frac{\partial}{\partial z} \left[\mu \left(\frac{\partial w}{\partial x} + \frac{\partial u}{\partial z} \right) \right] \end{aligned} \quad (2.4)$$

$$\begin{aligned} \rho \frac{dv}{dt} = f_y - \frac{\partial p}{\partial y} + \frac{\partial}{\partial y} \left[\mu \left(2 \frac{\partial v}{\partial y} - \frac{2}{3} \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} + \frac{\partial w}{\partial z} \right) \right) \right] \\ + \frac{\partial}{\partial z} \left[\mu \left(\frac{\partial v}{\partial z} + \frac{\partial w}{\partial y} \right) \right] + \frac{\partial}{\partial x} \left[\mu \left(\frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} \right) \right] \end{aligned} \quad (2.5)$$

$$\begin{aligned} \rho \frac{dw}{dt} = f_z - \frac{\partial p}{\partial z} + \frac{\partial}{\partial z} \left[\mu \left(2 \frac{\partial w}{\partial z} - \frac{2}{3} \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} + \frac{\partial w}{\partial z} \right) \right) \right] \\ + \frac{\partial}{\partial x} \left[\mu \left(\frac{\partial w}{\partial x} + \frac{\partial u}{\partial z} \right) \right] + \frac{\partial}{\partial y} \left[\mu \left(\frac{\partial v}{\partial z} + \frac{\partial w}{\partial y} \right) \right] \end{aligned} \quad (2.6)$$

For the conservation of energy the change of internal energy of a material element is given by the heat transfer and the work performed on the element (eq. 2.7).

$$\frac{de}{dt} = \dot{Q} + \dot{W} \quad (2.7)$$

The derivation of the complete energy conservation (eq. 2.8) of the Navier Stokes equations can again be found in [80].

$$\begin{aligned}
\rho \frac{de}{dt} + p \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} + \frac{\partial w}{\partial z} \right) &= \frac{\partial}{\partial x} \left(k \frac{\partial T}{\partial x} \right) + \frac{\partial}{\partial y} \left(k \frac{\partial T}{\partial y} \right) + \frac{\partial}{\partial z} \left(k \frac{\partial T}{\partial z} \right) \\
+ \mu \left\{ 2 \left[\left(\frac{\partial u}{\partial x} \right)^2 + \left(\frac{\partial v}{\partial y} \right)^2 + \left(\frac{\partial w}{\partial z} \right)^2 \right] - \frac{2}{3} \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} + \frac{\partial w}{\partial z} \right)^2 \right\} \\
+ \mu \left\{ \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right)^2 + \left(\frac{\partial w}{\partial y} + \frac{\partial v}{\partial z} \right)^2 + \left(\frac{\partial u}{\partial z} + \frac{\partial w}{\partial x} \right)^2 \right\}
\end{aligned} \tag{2.8}$$

These are the complete Navier Stokes equations for three dimensional flow. If steady two dimensional flow is assumed and the mass forces are neglected they can be simplified to the following set of equations.

Mass conservation:

$$\frac{\partial(\rho u)}{\partial x} + \frac{\partial(\rho v)}{\partial y} = 0 \tag{2.9}$$

Momentum conservation:

$$\begin{aligned}
\rho u \frac{\partial u}{\partial x} + \rho v \frac{\partial u}{\partial y} &= -\frac{\partial p}{\partial x} + \frac{\partial}{\partial x} \left[\mu \left(2 \frac{\partial u}{\partial x} - \frac{2}{3} \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} \right) \right) \right] \\
+ \frac{\partial}{\partial y} \left[\mu \left(\frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} \right) \right]
\end{aligned} \tag{2.10}$$

$$\begin{aligned}
\rho u \frac{\partial v}{\partial x} + \rho v \frac{\partial v}{\partial y} &= -\frac{\partial p}{\partial y} + \frac{\partial}{\partial y} \left[\mu \left(2 \frac{\partial v}{\partial y} - \frac{2}{3} \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} \right) \right) \right] \\
+ \frac{\partial}{\partial x} \left[\mu \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right) \right]
\end{aligned} \tag{2.11}$$

Energy conservation:

$$\begin{aligned} \rho \frac{de}{dt} + p \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} \right) &= \frac{\partial}{\partial x} \left(k \frac{\partial T}{\partial x} \right) + \frac{\partial}{\partial y} \left(k \frac{\partial T}{\partial y} \right) \\ + \mu \left\{ 2 \left[\left(\frac{\partial u}{\partial x} \right)^2 + \left(\frac{\partial v}{\partial y} \right)^2 \right] - \frac{2}{3} \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} \right)^2 + \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right)^2 \right\} \end{aligned} \quad (2.12)$$

2.2 Boundary layer flow

2.2.1 Laminar boundary layers

The Navier Stokes equations for steady flow can be further simplified for flow fields over wetted surfaces to obtain the laminar boundary layer equations by the estimation of the magnitude of the single terms. Therefore the flow field is divided into two parts. In the outer, inviscid part heat conduction and friction can be neglected. For the flow over a flat plate the conditions are constant (e) along the boundary layer edge. In a thin layer close to the wall the flow is dominated by heat conduction and friction. This is the boundary layer [77] and here flow properties are changing from wall to edge conditions. The following assumptions are taken.

- The boundary layer is thin compared to the characteristic length (L) of the flow case ($\delta/L \ll 1$). On the flat plate this is the running length of the boundary layer on the plate.
- The Reynolds number is large.
- The boundary layer thickness and the velocity in y-direction are inversely proportional to the square root of the Reynolds number.

The Reynolds number is defined in equation 2.13.

$$\text{Re}_x = \frac{\rho u L}{\mu} \quad (2.13)$$

The first assumption leads to the boundary layer theory being not valid close to leading edges. Then all quantities in the Navier Stokes equations are normalized as stated in (eq. 2.14). This leads to the boundary layer equations. The detailed derivation can be found in [80].

$$\begin{aligned}
x' &= \frac{x}{L}, \\
y' &= \frac{y\sqrt{\text{Re}_{e,x}}}{L}, \\
u' &= \frac{u}{u_e}, \\
v' &= \frac{v\sqrt{\text{Re}_{e,x}}}{u_e}, \\
p' &= \frac{p}{\rho_e u_e^2}, \\
T' &= \frac{T}{T_e}, \\
\rho' &= \frac{\rho}{\rho_e}, \\
k' &= \frac{k}{k_e}, \\
e' &= \frac{e}{e_e}, \\
\mu' &= \frac{\mu}{\mu_e}
\end{aligned} \tag{2.14}$$

The normalisations bring the values of x' , y' , ... to the same magnitude close to one. This is the reason for the use of the Reynolds number for y' and v' using the assumption taken above. After the normalizations are inserted into the equations some of the terms have the order of equation 2.15 and can therefore be neglected as Re is large and the terms have the same magnitude.

$$O\left(\frac{1}{\text{Re}_{e,x}}\right),$$

$$O\left(\frac{1}{\text{Re}_{e,x}^2}\right)$$
(2.15)

This leads to the boundary layer equations where the normalized quantities are replaced by the absolute ones again.

$$\frac{\partial(\rho u)}{\partial x} + \frac{\partial(\rho v)}{\partial y} = 0$$
(2.16)

$$\rho u \frac{\partial u}{\partial x} + \rho v \frac{\partial v}{\partial y} = -\frac{\partial p}{\partial x} + \frac{\partial}{\partial y} \left[\mu \left(\frac{\partial u}{\partial y} \right) \right]$$
(2.17)

$$\frac{\partial p}{\partial y} = 0$$
(2.18)

$$\rho u \frac{\partial h}{\partial x} + \rho v \frac{\partial h}{\partial y} = \frac{\partial}{\partial y} \left(k \frac{\partial T}{\partial y} \right) + u \frac{dp}{dx} + \mu \left(\frac{\partial u}{\partial y} \right)^2$$
(2.19)

The flow quantities (e.g. the velocity u) in the boundary layer depend on the distance from the wall (y). These boundary layer profiles vary with the running length (x), (eq. 2.20).

$$u = f(x, y)$$
(2.20)

To solve the equations a new coordinate η is needed to create similar profiles (eq. 2.21).

$$\eta = f(x, y),$$

$$u = f(\eta)$$
(2.21)

A solution for laminar incompressible flow over a flat plate was found by Blasius [7] in 1908 and for the same compressible flow case 1952 by Van Driest [95]. The huge time span between the two is a good indicator of the complexity of the compressible solution. For the

Van Driest solution the fluid has to be ideal and calorically perfect (eq. 2.22 and eq. 2.23). Also the Prandtl number Pr is assumed to be constant (eq. 2.24).

$$h = c_p T \quad (2.22)$$

$$p = \rho RT \quad (2.23)$$

$$Pr = \frac{\mu c}{k} = const. \quad (2.24)$$

Also the viscosity relation μ/μ_∞ needs to be described as a function of temperature. The viscosity can be modeled either with a simple power law (eq. 2.25) or with the Sutherland law (eq. 2.26). The appropriate exponent ω changes with temperature and the factor c_μ with ω . For low temperatures ($T \leq 200$ K) $\omega = 1$ is adequate. For higher temperatures ($T \geq 400$ K) $\omega = 0.65$ is used [36]. With the power law the viscosity relation can be obtained as function of T/T_∞ only, if ω is identical for both temperatures (eq. 2.27). This is not possible with the Sutherland law (eq. 2.28). Still Van Driest used the Sutherland law to obtain the viscosity relation μ/μ_∞ as a function of T/T_∞ and T_∞ as he found the power law with a constant exponent ω to be less accurate for the viscosity than the Sutherland law. Therefore the boundary profiles calculated by him are all set to a specific free stream temperature.

With the application of the power law the calculation procedure is possible without using a specific temperature T_∞ but will create less accurate data. Significantly different wall or free stream temperatures at a constant relation will result in identical profiles. This is not correct, as the exponent ω is a function of temperature.

$$\mu = c_\mu T^\omega \quad (2.25)$$

$$\mu = C \frac{T^{1.5}}{T + S} \quad (2.26)$$

$$\frac{\mu}{\mu_e} = \left(\frac{T}{T_e} \right)^\omega \quad (2.27)$$

$$\frac{\mu}{\mu_e} = \left(\frac{T}{T_e} \right)^{1.5} \frac{T_e + S}{T + S} \quad (2.28)$$

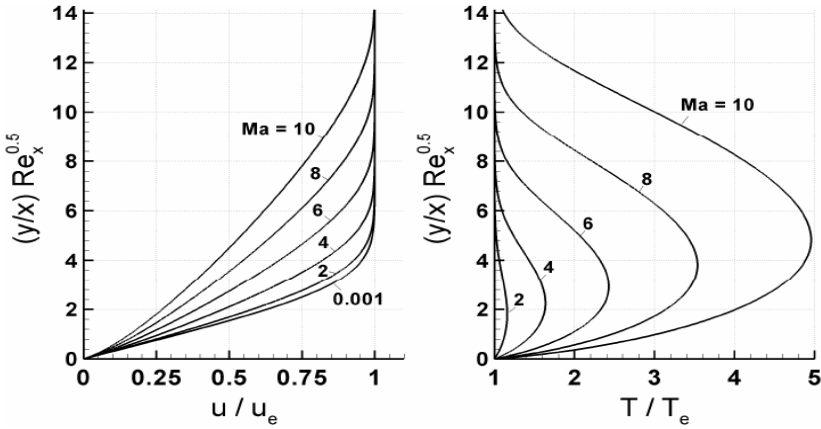


Fig. 2-1: Boundary layer profiles (velocity left and temperature right) calculated with the Van Driest Technique with varying edge Mach number; $T_w = T_e = 300 \text{ K}$ [35]

To create Van-Driest boundary layer profiles the shooting technique from [1] is used in the current work. Typical boundary layer profiles are shown in Fig. 2-1. The similarity coordinate η is $(y/x)\text{Re}^{0.5}$. The boundary layer is getting thicker with increasing edge Mach number and higher temperatures are reached in the boundary layer. The maximum temperature in the boundary layer is reached for adiabatic wall conditions and is defined as the recovery temperature (eq. 2.29). The recovery factor is $r < 1$ and usually set to $r = \text{Pr}^{1/2}$ for laminar boundary layers and $r = \text{Pr}^{1/3}$ for turbulent boundary layers [80].

$$T_r = T_e \left(1 + \frac{\gamma - 1}{2} r Ma_e^2 \right) \quad (2.29)$$

In Fig. 2-1 it is shown that it is not possible to separate the boundary layer sharply from the outer flow field. Therefore no natural definition of the boundary layer thickness is available. Usually the definition in equation 2.30 is used with $\varepsilon = 0.01$ [80].

$$u_e - u(y) \leq \varepsilon u_e \quad (2.30)$$

For the incompressible boundary layer this results in [7]:

$$\delta = 4.9 \frac{x}{\sqrt{\text{Re}_{e,x}}} . \quad (2.31)$$

For correlations often two other definitions of the boundary layer thickness are used. The boundary layer displacement thickness δ_1 (eq. 2.32) describes the offset of the outer flow field away from the wetted wall caused by the boundary layer in case of an equivalent inviscid flow. The boundary layer momentum thickness δ_2 (eq. 2.33) describes the momentum loss caused by the boundary layer in the term of the thickness of flow at edge conditions representing this loss.

$$\delta_1 = \int_{y=0}^{y=\delta} \left(1 - \frac{\rho u}{\rho_e u_e} \right) dy \quad (2.32)$$

$$\delta_2 = \int_{y=0}^{y=\delta} \frac{\rho u}{\rho_e u_e} \left(1 - \frac{u}{u_e} \right) dy \quad (2.33)$$

The wall shear stress (eq. 2.34) and the skin-friction coefficient (eq. 2.35) can also be calculated from the boundary layer profile.

$$\tau_w = \mu_w \left. \frac{\partial u}{\partial y} \right|_w \quad (2.34)$$

$$c_f = \frac{\tau_w}{q_e} \quad (2.35)$$

The wall heat transfer can be deduced for flat plate flow at constant wall temperature from the skin friction with the Reynolds analogy [80] using the recovery temperature defined in equation 2.29:

$$\dot{q}_w = \text{Pr}^{-2/3} \frac{\tau_w}{\rho_e u_e^2} \rho_e u_e c_p (T_r - T_w). \quad (2.36)$$

Then the Stanton number C_H can be calculated. For practical reasons here the Stanton number is calculated with the total temperature instead of the reference temperature (eq. 2.29) as the reference temperature depends on local edge conditions and therewith on the local model geometry and the state of the boundary layer.

$$C_H = \frac{\dot{q}_w}{\rho_e u_e c_p (T_0 - T_w)} \quad (2.37)$$

As an alternative to the elaborate calculation of the boundary layer in the Van Driest method the reference temperature method [23] can be used. For incompressible flow the temperature in the boundary layer is identical to the edge temperature. The quantities ρ and μ which depend on the temperature are identical to the edge conditions. Based on empirical observations the wall shear stress and wall heat transfer can be obtained for compressible flow based on the calculations for incompressible flow. Therefore the density ρ and the viscosity μ are calculated at a reference temperature (eq. 2.38). It combines the values of the fluid at the wall (W), at the boundary layer edge (e) and the recovery values (r, eq. 2.29).

$$T^* = 0.28T_e + 0.5T_w + 0.22T_r \quad (2.38)$$

If the pressure is constant through the boundary layer normal to the wetted surface equation 2.23 can be simplified to:

$$\rho^* T^* = \text{const.} = \rho_e T_e. \quad (2.39)$$

This is used to calculate density as a function of temperature. The viscosity relation μ/μ_e is calculated as a function of temperature relation T/T_e with equation 2.27 using the power law with a constant exponent ω . The Sutherland law is not used here due to its more complex structure. As an example the Reynolds number is transformed to reference conditions:

$$\text{Re}_x^* = \frac{\rho^* u_e x}{\mu^*} = \frac{\rho_e u_e x}{\mu_e} \frac{\rho^*}{\rho_e} \frac{\mu_e}{\mu^*} = \text{Re}_{e,x} \left(\frac{T_e}{T^*} \right)^{1+\omega}. \quad (2.40)$$

The same principle is used to obtain the skin friction (eq. 2.41) and wall heat transfer (eq. 2.42) for compressible flow [36].

$$\tau_w = 0.332 \frac{\rho_e u_e^2}{\sqrt{\text{Re}_{e,x}}} \left(\frac{T^*}{T_e} \right)^{0.5(\omega-1)} \quad (2.41)$$

$$\dot{q}_w = 0.332 k_e \text{Pr}^{1/3} (T_r - T_w) \left(\frac{T^*}{T_e} \right)^{0.5(\omega-1)} \sqrt{\frac{\text{Re}_\infty}{x}} \quad (2.42)$$

Here k is the thermal conductivity. It can also be calculated with a power law [36]:

$$k = 34.957 \cdot 10^{-5} T^{0.75} \quad (2.43)$$

According to Simeonides [84], Hirschel [36] and others based on the reference temperature the boundary layer displacement and momentum thickness can be expressed as follows:

$$\delta_1 = 1.72 \frac{x}{\sqrt{\text{Re}_{e,x}}} \left(-0.122 + 1.122 \frac{T_w}{T_e} + 0.333 \frac{\gamma-1}{2} Ma_e^2 \right) \left(\frac{T^*}{T_e} \right)^{0.5(\omega-1)} \quad (2.44)$$

$$\delta_2 = 0.66 \frac{x}{\sqrt{\text{Re}_{e,x}}} \left(\frac{T^*}{T_e} \right)^{0.5(\omega-1)} \quad (2.45)$$

2.2.2 Turbulent boundary layers

In chapter 2.2.1 the boundary layer equations were derived assuming steady flow. From a certain Reynolds number upward (and therewith a certain boundary layer running length) the boundary layer flow can become instable resulting in a turbulent boundary layer [78]. For defined flow conditions the boundary layer transitions to turbulent after a certain running length. Still the Reynolds number is not the only influencing parameter. In [1] 19 influencing parameters are listed including the surface structure.

Based on empirical observation a power formula can be used to describe the mean flow field of an incompressible turbulent boundary layer (eq. 2.46) [80].

$$\frac{u}{u_e} = \left(\frac{y}{\delta} \right)^{\frac{1}{7}} \quad (2.46)$$

Then the reference temperature can be used to obtain the skin friction (eq. 2.47), the wall heat transfer (eq. 2.48), the boundary layer displacement thickness (eq. 2.49) and the momentum thickness (eq. 2.50) [36].

$$c_f = 0.0592 \frac{u_e}{(\text{Re}_{e,x})^{0.2}} \left(\frac{T^*}{T_e} \right)^{0.2(\omega-4)} \quad (2.47)$$

$$\dot{q}_w = 0.0296 \cdot k_e \text{Pr}^{1/3} (T_r - T_w) \left(\frac{T^*}{T_e} \right)^{0.2(\omega-4)} \left(\frac{1}{x} \right)^{0.2} (\text{Re}_\infty)^{0.8} \quad (2.48)$$

$$\delta_1 = 0.0504 \frac{x}{(\text{Re}_{e,x})^{0.2}} \left(0.129 + 0.871 \frac{T_w}{T_e} + 0.648 \frac{\gamma-1}{2} Ma_e^2 \right) \left(\frac{T^*}{T_e} \right)^{0.2(\omega-4)} \quad (2.49)$$

$$\delta_2 = 0.036 \frac{x}{(\text{Re}_{e,x})^{0.2}} \left(\frac{T^*}{T_e} \right)^{0.2(\omega-4)} \quad (2.50)$$

Using the reference temperature the boundary layer quantities, except the wall heat transfer, become a function of the relation between wall and edge temperature only if the power law exponent ω is constant and therefore identical for edge, wall and reference conditions. The data becomes less accurate with increasing differences for these values. Still it is the only way to quickly obtain all the desired boundary layer parameters for turbulent flow. The wall heat transfer and the wall skin friction but not the boundary layer displacement and momentum thickness can be calculated from the boundary layer equations as shown in [94].

For a flat plate flow the skin friction coefficient, the Stanton number, the boundary layer displacement thickness, and the boundary layer momentum thickness obtained with the Van Driest method, the reference temperature method and with CFD simulations performed by the commercial software package CFX (chapter 3.3) are shown in Fig. 2-2 through Fig. 2-5 for different wall temperatures. The flow conditions for the calculations are listed in Tab. 2-1.

Tab. 2-1: Flow condition for boundary layer calculation

L [mm]	Ma _e [-]	T _e [K]	p _e [Pa]
110	3	574	16320

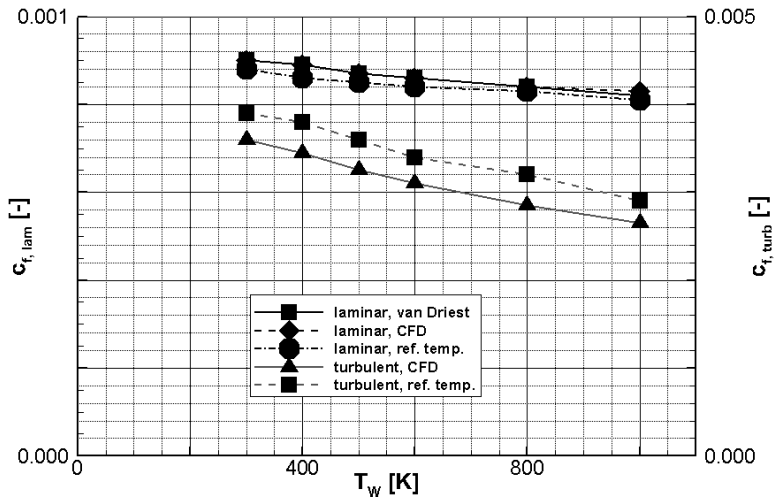


Fig. 2-2: Skin friction coefficient for a laminar and a turbulent boundary layer obtained with different methods

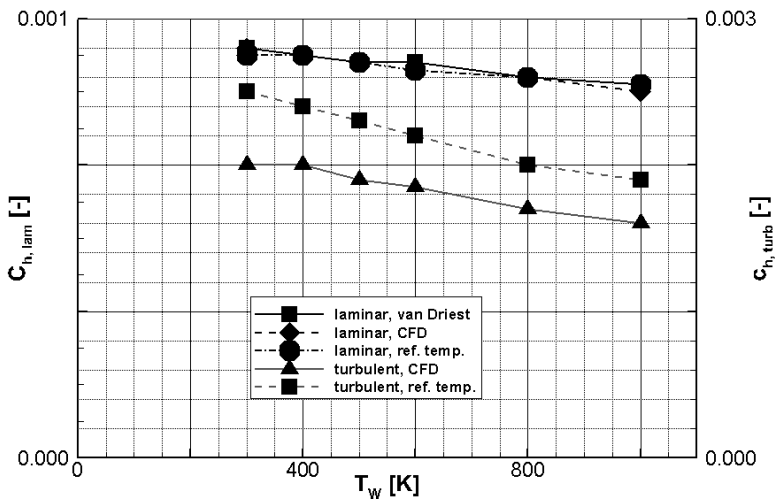


Fig. 2-3: Stanton number for a laminar and a turbulent boundary layer obtained with different methods

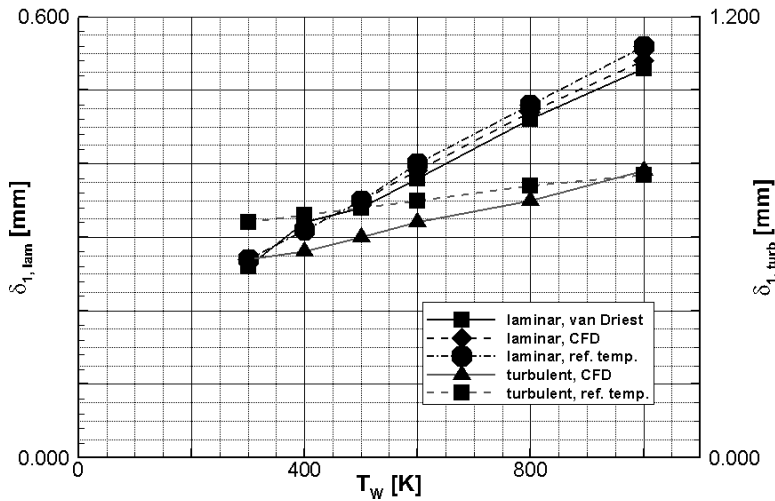


Fig. 2-4: Displacement thickness for a laminar and a turbulent boundary layer obtained with different methods

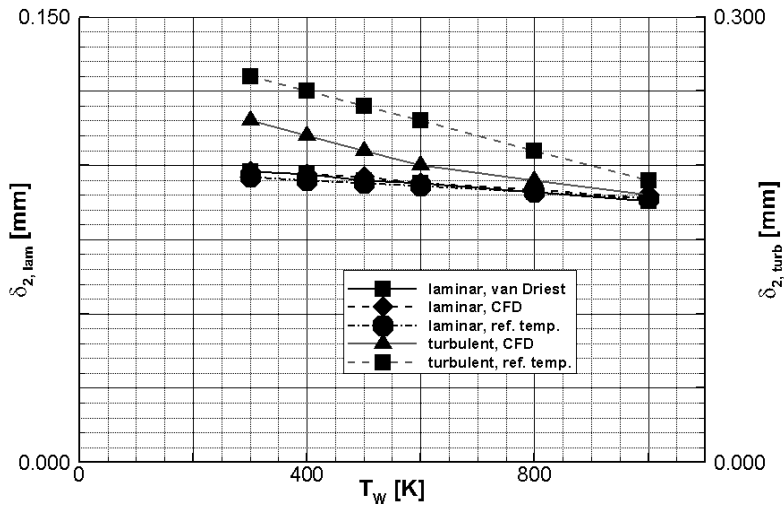


Fig. 2-5: Momentum thickness for a laminar and a turbulent boundary layer obtained with different methods

For the laminar boundary layer all three calculation methods result in almost identical values for the boundary layer parameters. For the turbulent boundary layer the results are of the same magnitude and the wall temperature effect is captured qualitatively identical with the two methods available here. Still an offset of up to about 10 % exists between results of the CFD simulations and the reference temperature method. Later to predict the pressure distribution inside a shock train the Stanton number and the boundary layer momentum thickness will be used with an exponent of 0.25. The offset of 10 % between the values is not problematic here and the easily available values from the reference temperature method will be used for correlation purposes.

2.3 Shock boundary layer interaction

If a supersonic boundary layer is hit by a compression shock this can result in local flow separation. The dynamic pressure in the boundary layer is reduced by friction. When the boundary layer is subject to the pressure rise by a compression shock it might not be able to follow this pressure rise. This results in the separation of the boundary layer. Two typical scenarios of shock boundary layer interaction are shown in Fig. 2-6.

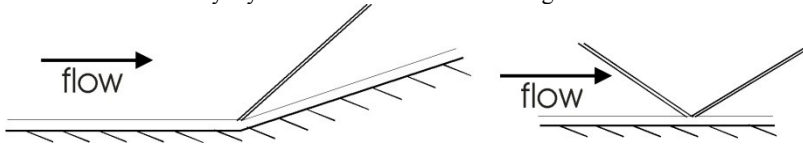


Fig. 2-6: Sketch of shock wave boundary layer interaction at a compression ramp (left) and caused by an impinging shock (right)

Shock boundary layer interactions at compression corners do occur at the hinge line of inlet ramps in supersonic air breathing engine inlets or at control surfaces for supersonic vehicles. An example for a shock boundary layer interaction caused by an impinging shock is the lip in the isolator of the research training group Scramjet (Fig. 4-5) where the deflection of the flow from the second inlet ramp into the isolator creates a shock originating at the leading edge of the lip impinging on the ramp.

The basic flow field is identical for both phenomena. The pressure rise from the single ramp compression or impinging shock is spread over two shocks at separation and reattachment (Fig. 2-7). The total pressure loss over the two shocks is smaller than over one with the same compression due to the smaller entropy production. Also the separation point lays upstream of the compression corner/impinging point where the boundary layer is thinner. This enables the separated shear layer to follow the pressure rise.

Geometrically the size of the separation bubble is variable as long as the separation and the reattachment shock together create the pressure rise defined by the compression corner. The actual size depends on the incoming boundary layer profile and the flow parameters stated in equation 2.51. The separation bubble will react to a thicker incoming boundary layer profile with upstream growth. The separation will then happen at a point with a thinner boundary layer profile that can then follow the pressure rise over the separation shock into the shear layer.

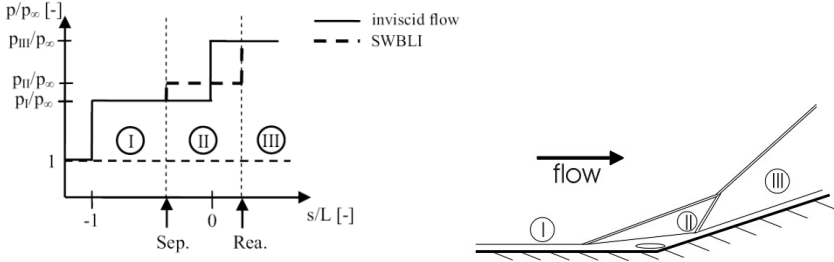


Fig. 2-7: Pressure distribution at a shock boundary layer interaction [67] (left) and sketch of the complete SWBL (right)

A correlation for the size of the separation bubble was found by Katzer [44] for the case of a shock impinging on a flat plate with a laminar boundary layer:

$$\frac{L_B}{\delta_1} = 4.4 \sqrt{\frac{\text{Re}_L}{\mu_W T_e}} \frac{\mu_e T_W}{Ma_I^3} \frac{p_{III} - p_{inc}}{p_I}. \quad (2.51)$$

Here p_{inc}/p_I (inc: incipient) is the minimum pressure jump at the compression corner or impingement position with an separation occurring. The following correlation is used to determine the incipient separation pressure:

$$\frac{p_{inc}}{p_I} = 1 + 0.5 \cdot Ma_I^2 \cdot \gamma \cdot f \cdot \sqrt{2} \cdot (c_{f,L})^{0.5} \cdot (Ma_I^2 - 1)^{-0.25}. \quad (2.52)$$

Katzer suggests a value of 1.85 for f in the correlation. Other researchers correlated its values between 1.57 [79] and 2 [39] from their experiments. The incipient separation pressure is mainly a function of Mach number. The wall temperature influence is very small as the small

wall temperature effect on $c_{f,L}$ (Fig. 2-2) is further weakened by the exponent of 0.5. This is shown in Fig. 2-8.

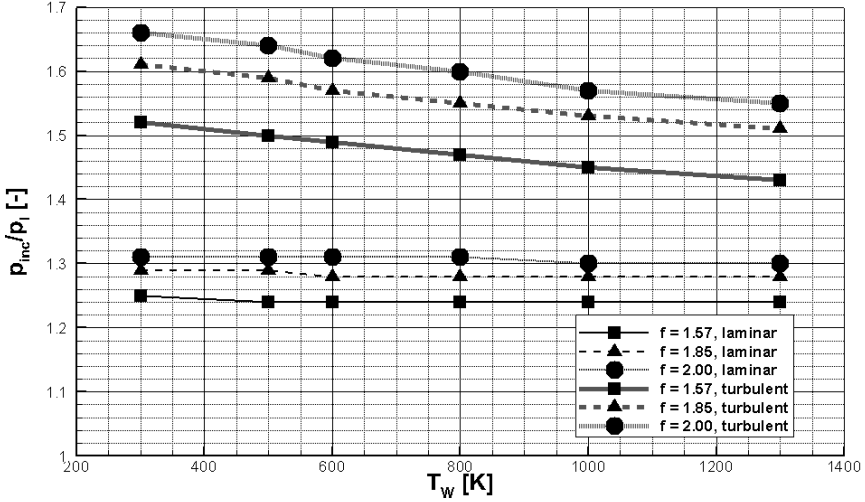


Fig. 2-8: Incipient separation pressure as a function of wall temperature

The correlation of Katzer has been modified by Bleilebens and Olivier [9] to fit the wall temperature effects they observed:

$$\frac{L_B}{\delta_{1,sep}} = \left(4.4 - 0.2 \frac{T_W}{T_e} \right) \sqrt{\frac{\text{Re}_L}{\frac{\mu_W T_e}{\mu_e T_W}}} \frac{p_{III} - p_{inc}}{p_I}. \quad (2.53)$$

They investigated the flow field at a compression corner with a 15° angle and a two dimensional flow field. The results show the separation bubble growing with increasing wall temperature and decreasing edge temperature. Also the separation bubble scales linear with the boundary layer displacement thickness.

For both correlations the wall temperature effects can be estimated using boundary layer values calculated with the reference temperature method and the power law to model the viscosity. Equation 2.27 is used for the viscosity, equation 2.44 or 2.49 for the boundary layer displacement thickness and equation 2.41 or 2.47 for the wall skin friction.

The shock wave boundary layer interaction at the hinge line of the two compression ramps of the inlet model used in this thesis was investigated prior to this work by Neuenhahn and Olivier [66]. They also showed a growing separation bubble with increasing wall and decreasing edge temperature. In addition they showed that the separation length is not only a factor of the relation between wall and edge temperature (T_w/T_e) but also the edge (or wall) temperature itself. The separation bubble reacts extremely sensitive to the incoming boundary layer profile therefore the use of the power law with a constant exponent ω to calculate the viscosity μ is not applicable here. Neuenhahn [67] also developed a correlation for the separation bubble length based on the momentum equation for the free shear layer over the separation bubble. In that correlation (eq. 2.54) also the empirically correlated incipient separation pressure is used and the difference of the shear stress on the upper and lower boundary of the free shear layer is estimated with the half value of the wall skin friction at the separation point for the undisturbed boundary layer based on empirical observation. The values for the skin friction coefficient c_f and the sonic height factor C_h can be obtained from boundary layer calculation e.g. using the Van Driest method described above. For the incipient separation pressure p_{inc} data has to be gained from empirical correlations or be predicted otherwise (e.g. by numerical simulations).

$$L_B = \frac{4}{\gamma} \frac{C_h}{c_f \sqrt{Re}} \frac{\sqrt{x}}{Ma_I^2} \left(\frac{p_{III} - p_{inc}}{p_I} \right) \quad (2.54)$$

The wall temperature effect on the incipient separation pressure is very small (Fig. 2-8). It can therefore be neglected to calculate the wall temperature effect on the separation bubble length. With this assumption it is possible to quantify the wall temperature effect on the separation length independently from the imposed pressure rise. This is shown in Fig. 2-9 assuming a laminar boundary layer. The sonic height required is calculated by CFD calculations. The flow conditions are the same as listed in Tab. 2-1. All three describe the wall temperature effect very similar as almost linear with a maximum deviation of about 10 %. The correlations by Katzer and its modification by Bleilebens are dominated by the wall temperature effect on the boundary layer displacement thickness as the effect of wall temperature on all other factors is very limited. The correlation by Neuenhahn is dominated by the sonic height of the boundary layer. This behaves very similar to the boundary layer displacement thickness. A strong effect of the wall temperature on the separation bubble is shown. For the typical maximum wall temperature of the shock tunnel experiments at the SWL of $T_w = 1000$ K the separation bubble is expected to grow to about 2.5 the size it has at $T_w = 300$ K.

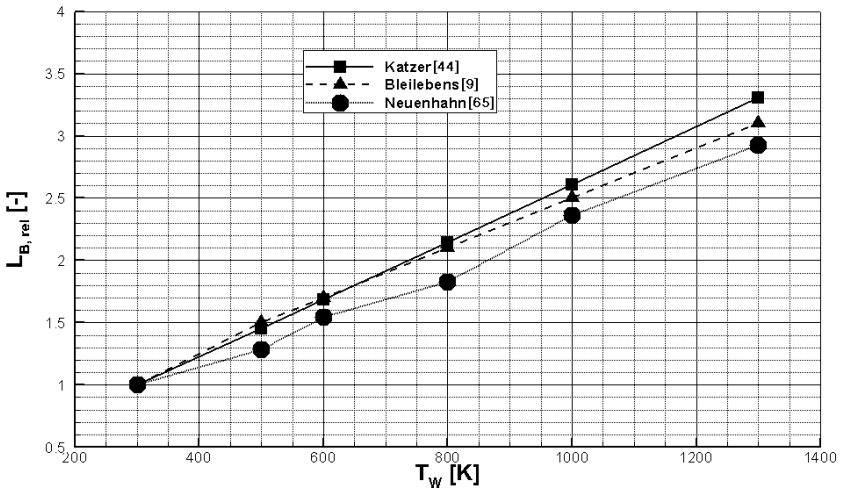


Fig. 2-9: Wall temperature effect on the separation length for three different correlations

2.4 Scramjets

Scramjets are hypersonic air breathing engines with combustion at supersonic velocity while ramjets burn the fuel at subsonic speeds. The ideal cycle for thermal engines is shown in Fig. 2-10. Both engine types use the compression by the deceleration of the incoming air-flow to generate the pressure required for the engine cycle (state 1-3). An isentropic process would reach a higher pressure level (state 2). In real systems the compression cannot be achieved isentropic as shocks are used to generate the pressure rise (Fig. 4-5). Turbofans and Turbojets use a compressor here. The combustion takes place between state 3 and 4.

In an ideal case the combustion takes place isobaric while losses are taken into account in Fig. 2-10 shown by the drop of the pressure from the isobaric line starting at state 3. The thrust is generated by the expansion between state 4 and 6. The losses here are shown by the pressure loss between state 4 and 5. Turbo engines have a turbine between state 4 and 6 to run the compressor between state 1 and 3. Thrust can be generated due to the divergence of the isobaric lines with increasing entropy. Thrust is lowered by the losses in the single components generated e.g. by skin friction on the walls. If these losses get too high the engine thrust can decrease to levels below the overall drag of the system.

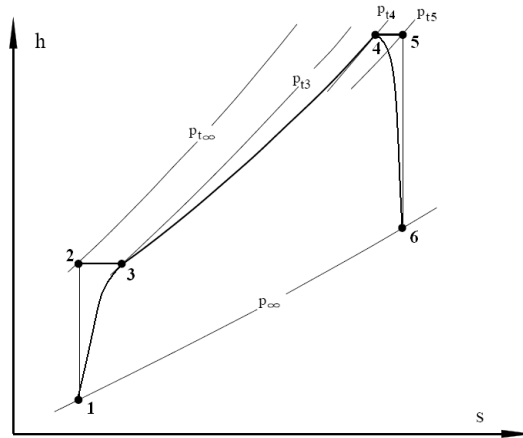


Fig. 2-10: Cycle of thermal engines [34]

To achieve this cycle three main components are needed for a Ramjet as well as a Scramjet.

- The inlet to achieve compression by deceleration of the incoming airflow
- The combustor to add heat to the compressed air. This is achieved by mixing a fuel (usually hydrogen or kerosene) into the airflow.
- A nozzle to expand the burned gas to atmospheric conditions in order to generate thrust.

The losses generated in the cycle limit the maximum operating Mach numbers for both Ramjets and Scramjets as shown in Fig. 1-1. Ramjets are limited to smaller Mach numbers as they need to decelerate the incoming airflow to subsonic speeds. With higher flight Mach numbers the total pressure losses created by the shocks during the decelerating process become too high. Scramjets create fewer losses due to the limited deceleration in the compression process. Therefore they are applicable for higher Mach number flight compared to Ramjets. The main issues for the successful operation of a Scramjet engine are:

- Scramjets as well as Ramjets cannot operate at no or low airspeed as they need the deceleration from the flight airspeed for the necessary compression for the engine cycle. They have to be combined together (Dual-Mode-Scramjet) and with an engine system for low flight Mach numbers
- High Mach number flow on all wetted surfaces in the engine is present. Therefore high drag is produced on all these surfaces. In operational Scramjet systems

overall drag and net thrust are close values of the same magnitude. The overall system performance reacts sensitive to small changes in the single system components.

For an effective Scramjet engine the area of the wetted surfaces and therewith the length of the engine flow path should be as short as possible. To achieve an effective compression in the inlet low compression angles and multiple shocks are desirable. An efficient compression in the inlet also has the advantage that the air is reaching lower temperatures at the end of the process. This leaves more margins for the addition of heat in the combustion process. Still both measures create larger areas of wetted surfaces. For the combustion process a rather long internal flow path is needed to mix the injected fuel before it can be burned. Here again this results in more drag due to skin friction. The simple basic design of a Scramjet has to be optimized to build a working engine.

As described above ideally a Scramjet would burn the fuel isobaric. In reality the heat addition is always accompanied by a pressure rise in a constant cross section combustor. An example is given by combustion experiments made within the research training group [98]. For the study two staged injection was investigated. In Fig. 2-11 the measured wall pressure distribution is shown for different combustion chamber entrance Mach numbers without fuel being injected. The first injection stage is a central injector. The leading and the trailing edge of this injector are shown by the first two vertical lines in the figure. The second stage is realized by wall injection further downstream marked in the figure by the third vertical line. Fuel is injected at a equivalence ratio of $\Phi_1 = 0.27$ at the first stage and $\Phi_2 = 0.24$ at the second stage. The central injector is generating a local pressure increase for the zero fuel case. For the fuel on cases the pressure is rising in downstream of the first stage caused by the combustion. This pressure rise does also affect the flow field upstream of the leading edge of the central injector.

In the supersonic flow field the pressure rise is communicated upstream through the subsonic part of the boundary layer on the wetted surfaces. If the inlet would end directly upstream of the central injector the combustion pressure rise would influence the inlet flow field causing thermal blockage of the engine. An isolator is necessary here to protect the inlet from this pressure rise. Due to the drag on the walls the length needed for the isolator is of great interest in order to achieve a good overall system performance. One conclusion on the length needed for the isolator can be drawn qualitatively from Fig. 2-12. The upstream influence of the combustion pressure rise is growing with decreasing Mach number.

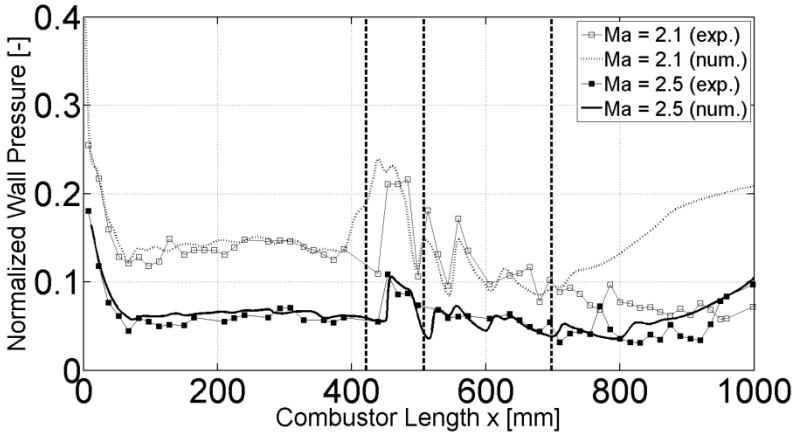


Fig. 2-11: Wall pressure distributions in combustion chamber without fuel injection [98]

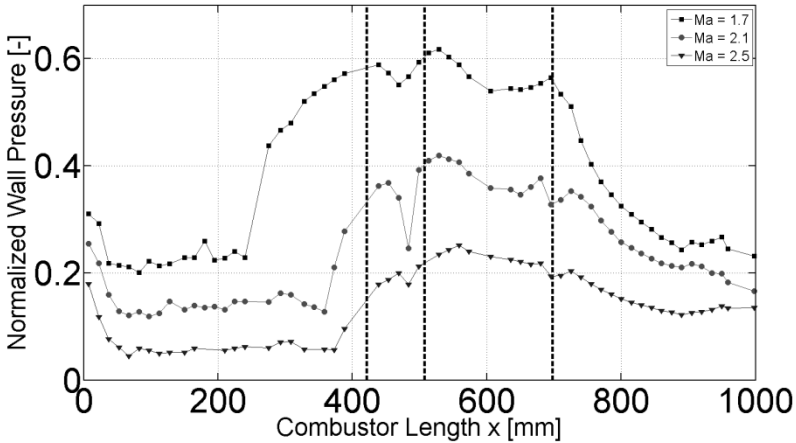


Fig. 2-12: Wall pressure distributions in combustor with fuel injection (experimental data) [98]

The combustion pressure rise can be quantified as a function of the combustion total temperature rise [5]. The pressure rise as a function of combustion heat release (T_{t4}/T_{t3}) is plotted for different combustor area ratios in Fig. 2-13 for a flow field entering the combustor with a Mach number of $Ma_3 = 2.5$. This shows that not only geometric blockage, but supersonic combustion itself leads to a significant pressure rise.

As mentioned above a real Scramjet will most likely work as a dual-mode system operating as a Ramjet as well as a Scramjet. For the Ramjet mode the flow has to be decelerated to

subsonic conditions. This is achieved by a final normal shock. If this shock would take place in the external compression system pressure fluctuations caused by the combustion could easily reach the inlet due to the subsonic flow field downstream of this final shock. Therefore this shock is placed inside the closed duct flow path of the engine. To capture the pressure rise of this shock inside the closed flow channel of an isolator is needed here as well.

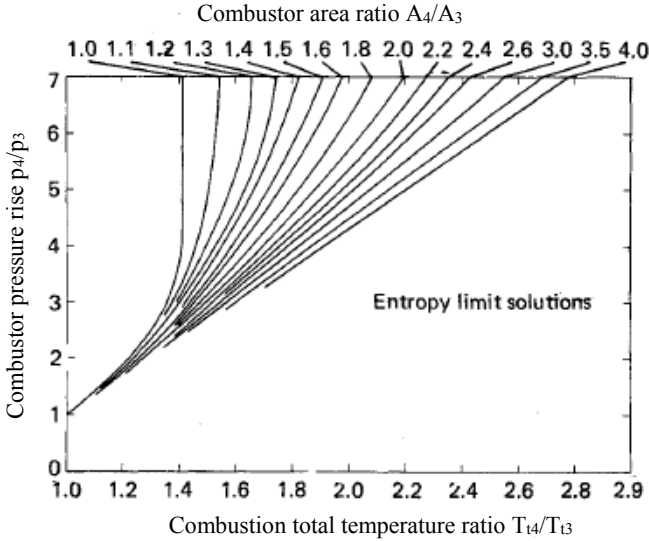


Fig. 2-13: Combustion pressure rise as a function of total temperature ratio for different combustor area ratios at $Ma_3 = 2.5$ [5]

In summary the isolator is needed in a Scramjet system for two reasons:

- To protect the inlet from the combustion pressure rise in supersonic combustion.
- To protect the inlet from the pressure rise by the final normal shock of the engine in Ramjet mode.

The compression necessary between the inlet and the combustor is achieved by the shock train. A shock train (chapter 2.5) occurs when the flow has to follow a pressure rise (e.g. by combustion or a throttle device).

Throughout the flow path of a hypersonic engine several conditions are defined to describe typical flow states. The typical numbering of these is shown in Fig. 2-14 following the most widely used scheme introduced by [33].

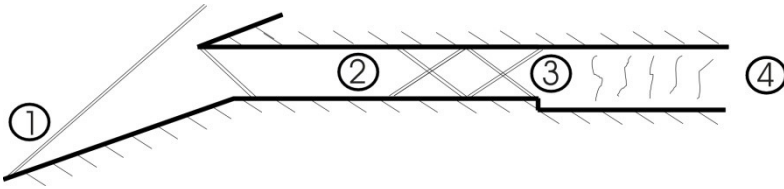


Fig. 2-14: Condition definition for hypersonic engines

Upstream of the inlet compression the flow conditions are noted with the subscript 1. If there is no forebody, like in the case of the scramjet model in the wind tunnel, this is also the free stream condition. The inlet compresses the flow to condition 2. This is the shock train entrance flow condition. The pressure at the combustion chamber inflow is noted with the subscript 3. This is the isolator exit flow condition. The pressure after the combustion process is noted with the subscript 4.

Hypersonic air breathing engines have been considered for fast air transport as well as space transportation purposes since the late 1930s. Still the biggest steps towards operational systems were taken within the last 15 years. Some of the more recent projects besides the research training group are briefly introduced in the following sections to provide some context.

2.4.1 Scramjet development in Australia

For the first time ever supersonic combustion was achieved during the HyShot II flight in Australia in 2001. The HyShot and the following HyCause and HiFire flight use sounding rockets to accelerate the test vehicle to the desired flight speed. A ballistic flight profile is used and the Scramjet experiments are performed during the reentry of the vehicle into the atmosphere. The flight Mach number for the combustion experiment was between 7.8 and 7.9 in an altitude of 35 to 23 km [85]. The inflow conditions for the engine vary throughout the free flight experiment due to the flight profile of the vehicle as well as wobbling movements induced by the sounding rocket. To generate comparable results for the fuel on and fuel off conditions two identical flow paths were built into the vehicle. The results show the pressure rise created by the combustion (Fig. 2-15).

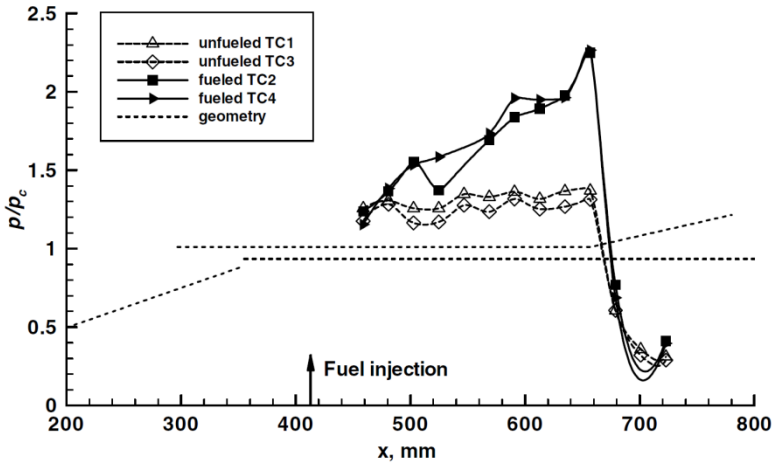


Fig. 2-15: Pressure distribution in the HyShort II combustor with and without fuel injection [85]

The design of the engine flow path is very simple. Aim of the experiment was not the generation of thrust but simply to achieve supersonic combustion. The results are then reference data for future flight experiments as well as ground tests and numerical simulations. The basic geometry of the engine is shown in Fig. 2-15. The inlet consists of a single compression ramp. The separation bubble induced by the shock forming at the leading edge of the lip is sucked away by a passive boundary layer bleed (not shown in Fig. 2-15). The combustor has a constant cross section and the thrust nozzle is as simple as sketched. The first part of the constant area combustor serves as isolator.

Founding on the successful HyShot flight the goal of the HiFire program is the development of a small free flying Scramjet glider to be tested in the final launch of the program [85]. Also in Australia within the Scramspace program a Scramjet demonstrator is developed for the potential use as engine for an intermediate stage of a transport to space system [10].

2.4.2 Scramjet development in the USA

Extensive Scramjet research was and is also carried out in the USA. Starting in 1985 a manned single stage to orbit (SSTO) air breathing system was developed for a first flight in 1999 within the National Aerospace Plane (NASP) program [104]. After the program failed due to the complexity of the ambitious task the hyperX program was initiated as a technology demonstrator for a Scramjet engine [62]. During the second test flight of the program positive thrust was generated for the first time ever with a Scramjet engine in free flight in 2004 and velocities up to $Ma = 9.7$ were reached with the X-43 with up to 11 s engine operation time

[59]. This was then followed by the X-51 with a first flight in 2010. It was flying at a lower Mach number of $Ma = 5$ with Kerosene as fuel operating for 200 s, to develop an engine for continuous operation [93].

2.5 Shock trains

Shock trains were first observed in 1949 in the diffusers of supersonic wind tunnels [68]. The expected discontinuous pressure jump of a single shock occurred over a longer distance. Shock trains occur in closed duct flows if a sufficiently strong pressure rise is forced on the flow. The overall pressure rise is limited by the pressure jump over a single normal shock. The pressure rise is communicated upstream through the subsonic part of the boundary layers on the wetted surfaces. A shock train can also develop if the pressure rise over a single shock in a closed duct is too high for the boundary layer to follow, leading to the separation and therewith upstream movement of the first shock. If boundary layers are present a single normal shock in a closed duct is usually only possible at very low supersonic Mach numbers.

Two different types of shock trains are known:

- The normal shock train with a strong normal shock with lambda feet at the beginning followed by weaker normal shocks. Downstream of the first shock the flow is accelerated back to supersonic speed and a new weaker shock occurs. It takes numerous shocks till the flow at the end of the shock train is subsonic. The flow field is sketched in Fig. 2-16.
- The oblique shock train initialized by oblique shocks. Here the pressure rise is lower at the beginning of the shock train. The flow can be either subsonic or supersonic after passing the shock train. Therefore an oblique shock train can also contain weak normal shocks towards the end.

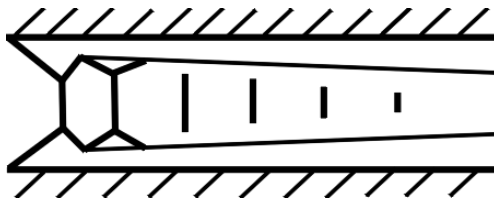


Fig. 2-16: Drawing of a normal shock train flow field

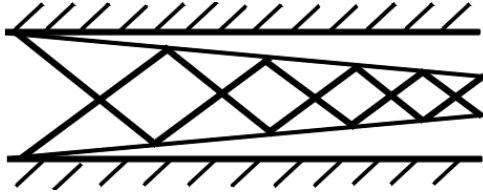


Fig. 2-17: Drawing of an oblique shock train flow field with supersonic exit flow

The beginning of the shock train is the first pressure jump caused by the first shock. If the shock train is caused by a pressure rise (e.g. a combustion pressure rise) the location of this pressure rise marks the end of the shock train. If the shock train is caused by a single normal shock interacting with the boundary layer the exit plane is defined by a uniform pressure and total pressure profile being reached. The normal shock train can be divided into two parts. In the first part which is the upstream region the supersonic flow is gradually decelerated and in the second downstream region the high speed flow in the central core is mixed with the low speed flow near the walls [90]. The main influencing factor on the type of shock train occurring is the isolator entrance Mach number Ma_2 . Usually the transition between normal and oblique shock train happens between $Ma_2 = 2$ and $Ma_2 = 3$.

A typical example for a normal shock train is given in Fig. 2-18 from [89]. Shown are the pressure distributions at the wall and at the centerline of the flow channel. The flow field is additionally illustrated with pressure contours for the two generated back pressures. The isolator entrance Mach number was $Ma_2 = 2$. For both shown cases the shock train starts with a normal shock with large lambda feet. The flow is accelerated back to supersonic speed after the shock by a nozzle like flow created by the lambda feet.

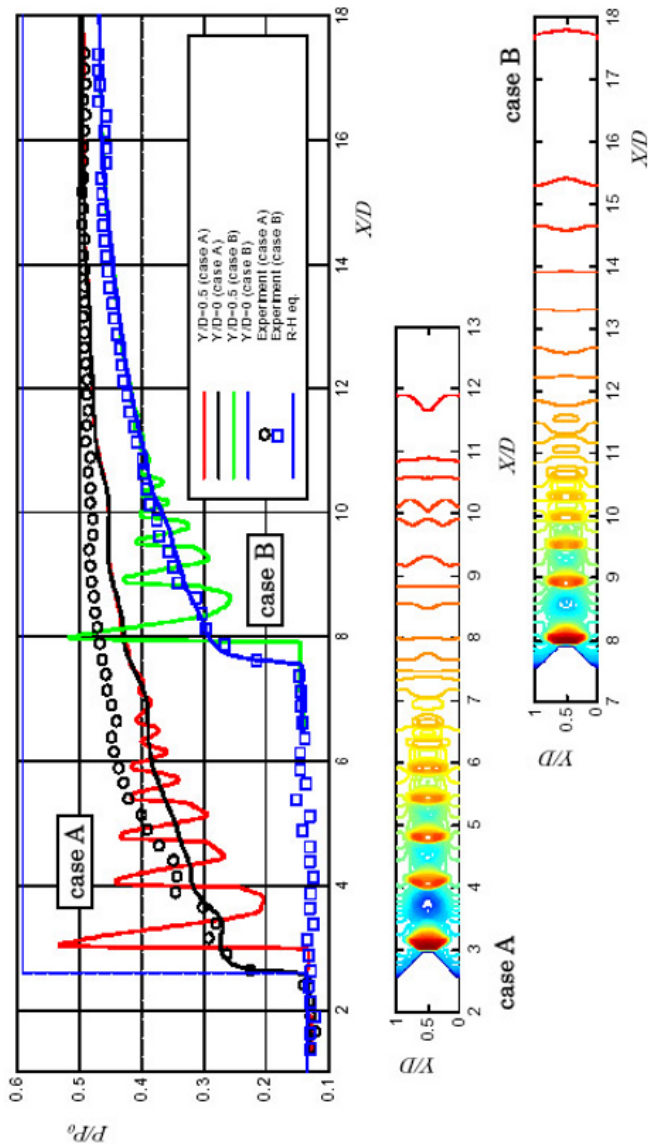


Fig. 2-18: Pressure distribution at the wall and on the centerline of a normal shock train for two different back pressures and pressure contours for the two cases [89]

The expansion caused by this is shown by the pressure distribution on the centerline but not on wall pressure distribution. The first shock of the shock train is the strongest one and the shocks are getting weaker further downstream. The wall pressure distribution with a large gradient at the beginning that is decreasing downstream is typical for normal shock trains.

Before the mechanism of the acceleration back to supersonic speed was unveiled normal shock trains were referred to as pseudo shocks. The visualization of the flow field showed a sequence of shocks. This was believed to be wrong as the flow was already subsonic after the first normal shock [15]. The wall pressure distribution does not show the existence of multiple shocks. The existence of the shocks was shown by static pressure measurements on the centerline of the duct in [72]. Also other names were used throughout the literature to describe shock trains. A list of these can be found in [58].

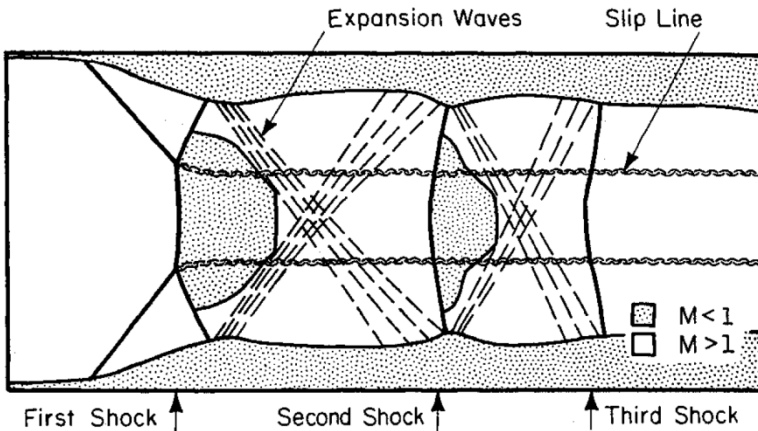


Fig. 2-19: Drawing of normal shock train flow field [12]

The nozzle effect was shown in [72], [3] and [88]. The boundary layer displacement thickness of the boundary layer was determined with a Laser Doppler Velocimeter (LDV) showing a decrease in the displacement thickness in between two successive lambda shocks. The same effect was also measured by Carroll and Dutton [12] and a drawing of the flow field is shown in Fig. 2-19.

Much effort was taken to predict the length and pressure distribution of the shock train. A first approach to an analytical description of the flow field was done by Crocco [15]. He described the whole process isentropic as he assumed the total pressure loss over the pseudo-shock small compared to the loss over a real normal shock. The flow enters the pseudo shock

at high speed and is continuously transferred to the lower speed state at the exit of the shock train by the mixing of inner high velocity core flow with the outer low velocity boundary layer. The pressure is constant at every plane throughout the mixing process and both mass flows add to the constant overall mass flow. The pressure is described by:

$$\frac{p}{p_2} = \left(\frac{1 - w'^2}{1 - w_2'^2} \right)^{\frac{\gamma}{\gamma-1}}. \quad (2.55)$$

with:

$$w = \frac{u}{\sqrt{2c_p T_0}} \quad (2.56)$$

The condition w' describes the core flow. The pressure relation above describes an isentropic compression of the core flow. This shock less model was improved by Ikui et al. [41] to allow a non isentropic deceleration of the core flow. The velocity variation in the main flow direction is described with the following two equations:

$$\frac{dw'}{dx} = -cw' \quad (2.57)$$

$$\frac{dw'}{dx} + b \frac{dw''}{dx} = 0. \quad (2.58)$$

The condition w'' describes the boundary layer flow. The constant c is determined empirically. The factor b can be determined from boundary condition at the beginning and end of the shock train. The two equations above are combined with the Navier Stokes equations (chapter 2.1) to obtain the wall pressure distribution:

$$\frac{p - p_2}{p_3 - p_2} = \frac{\{w_2'^2(w_2'^2 - 2w_2'^{*2}) + w_2'^2 w_2'^{*2} e^{-c(x/D)}\} (1 - e^{-c(x/D)})}{(w_2'^2 - w_2'^{*2})^2 - w_2'^2 (w_2'^2 - w_2'^{*2}) e^{-c(x/D)} (1 - e^{-c(x/D)})}. \quad (2.59)$$

This relation does not include any effects of the boundary layer profile on the shock train pressure distribution and is therefore not capable to describe e.g. wall temperature effects on

the shock train. It was later modified with empirical data to include these effects [42]. A more detailed description of these analytical models can be found in [58].

An analytical model of the flow field was combined with empirical knowledge to create a shock train model in [91]. The model is based on the determination of the flow field by the angles of the shocks and expansions. The angle of the first lambda foot of the first shock of the shock train is defined by the estimated flow turning angle of 10° . The second lambda foot turns the flow back into the main flow direction. The turning angle of the expansion wave right downstream of the second lambda foot has to be determined empirically. To resolve the whole flow field a lot of empirically gained data is necessary making it difficult for practical use.

The most widely used model to describe the pressure distribution throughout the shock train is the empirical correlation by Waltrup and Billig. In [100] they showed that the pressure distributions for the same flow conditions with different back pressures can be shifted over each other to create a single pressure distribution for the flow case. This means that the change of the boundary layer profile with the different shock train starting locations has no effect on the pressure distribution. Consequently the first correlation for the shock train length did not explicitly include boundary layer effects:

$$\frac{s(Ma_2^2 - 1) \text{Re}_D^{0.20}}{2(D - 1.25)} = 17.5 \left(\frac{p_3}{p_2} - 1 \right) + \left(13.5 \frac{p_3}{p_2} - 1 \right)^2. \quad (2.60)$$

The flow channel diameter D is inserted in inches here. The relation between the wall pressure and the shock train running length is described by a quadratic relation. The shock train length scales linear with flow channel diameter and inversely quadratic with the isolator entrance Mach number and weak inversely with the Reynolds number calculated with the flow channel diameter. The correlation was derived from experimental data for normal as well as oblique shock trains. It was modified to its final and most widely used form in [101] and [102]:

$$\frac{s(Ma_2^2 - 1) \text{Re}^{0.25}}{D^{0.5} \delta_2^{0.25}} = 50 \left(\frac{p_3}{p_2} - 1 \right) + 170 \left(\frac{p_3}{p_2} - 1 \right)^2. \quad (2.61)$$

The boundary layer profile is now included represented by the boundary layer momentum thickness for the undisturbed flow case. The independence of the pressure distribution from the exact location where the boundary layer profile is obtained was confirmed by Carroll and

Dutton in [11]. They investigated normal ($Ma_2 = 1.6$) as well as oblique shock trains ($Ma_2 = 2.45$) in a rectangular duct. They also found for both flow cases that the shock trains with different back pressure have an identical wall pressure distribution if they are shifted over each other. Different back pressures create different shock train lengths with different boundary layer profiles entering the shock train. For each defined flow case there is a certain pressure distribution for the shock train. For the considered variations of flow conditions the influence of the boundary layer profile entering the shock train that varies with the shock train length and therefore the back pressure can be neglected. They also observed that the oblique shock train has a tendency to become asymmetric and unsteady compared to the normal shock train.

Mateer et al [56] experimentally investigated shock train at a low supersonic isolator entrance Mach number of $Ma_2 = 1.4$. A variation of the Reynolds number calculated with the boundary layer just upstream of the first separation between values of $1.9 \cdot 10^5$ and $1 \cdot 10^6$ had no visible effect in opposition to the large influence of a small Mach number variation ($Ma_{2min} = 1.32$; $Ma_{2max} = 1.48$) on the same setup shown in [57].

For low supersonic Mach number ($Ma_2 = 1.3$) the stabilization of a single normal shock has been observed for relatively thin boundary layers in [71]. Naturally no single pressure distribution for the flow case can be obtained here. A single normal shock instead of a shock train for a thin boundary layer in front of a shock train was also observed at $Ma_2 = 2.4$ in [70]. It should be noted that the pressure profile for the shock train here was not measured at different locations throughout the rectangular duct but with a single probe and a constant back pressure variation. This unsteady flow field might have influenced the shock train as it easily becomes instationary for high isolator entrance Mach numbers as shown in [11].

By Lin et al. [53] an oblique shock train with an isolator entrance Mach number of $Ma_2 = 3$ was investigated numerically. A wide range of back pressures was investigated with the flow supersonic as well as subsonic at the isolator exit. For the cases with a supersonic flow at the exit clearly no complete mixture of the outer separated boundary layer flow with the inner high velocity flow was achieved. Still the flow field is identical compared to the cases with a subsonic exit flow. They compared their results to the correlation by Waltrup and Billig obtained from experiments in a cylindrical duct using the flow channel height instead of the flow channel diameter. The same trends were also found by the same authors for the lower isolator entrance Mach number of $Ma_2 = 1.8$ in [54]. They found a tendency to longer shock train for the rectangular case with a qualitatively identical wall pressure distribution. Corner vortices causing additional losses were found in [19]. Also a rectangular flow channel has a larger wetted surface compared to circular duct with the same cross section. Losses on the

surfaces e.g. by the boundary layer momentum thickness have a higher impact here. A good agreement between experimental results obtained in a rectangular duct and the correlation was found in [4] also with a slight increase in shock train length compared to the correlation. The same trend was observed for shock trains in annular ducts by [86].

One author of [100] stated the need for an additional scaling especially for rectangular isolators as they are often found in scramjet isolators [6]. Still this is difficult to realize as a rectangular duct can have a quadratic cross section as well as an almost two-dimensional one with a large impact on the ratio between wetted surface and cross section. A modified version of equation 2.61 was suggested by Sullins et al. in [87]:

$$\frac{s(Ma_2^2 - 1) \text{Re}^{0.2}}{D^{0.5} \delta_2^{0.3}} = 50 \left(\frac{p_3}{p_2} - 1 \right) + 170 \left(\frac{p_3}{p_2} - 1 \right)^2. \quad (2.62)$$

Here the exponent of the boundary layer momentum thickness is slightly increased. This gives the boundary layer losses a higher influence on the shock train length. This is logical considering the higher ratio of wetted surface to cross section but cannot quantify the geometry of cross section as it would be elementary here. Also the exponent of the unit Reynolds number is decreased to keep the dimensions in balance. There is no physical effect explaining this change. An additional scaling factor with the exponent 0.25 varying with the cross section geometry would be a more logical choice here. Most validation experiments for the correlation applied back pressure generated by a throttling device. Experiments by Le et al. [49] showed that it also works well when the back pressure is generated by supersonic combustion. Different back pressure levels were generated with different combustion equivalence ratios here.

Isolators with the same cross section area with rectangular and round cross section were compared experimentally in [51] for isolator entrance Mach numbers of $Ma_2 = 1.8$ and $Ma_2 = 2.2$. As expected the cylindrical isolator showed shorter shock trains with the same back pressure compared to the rectangular one. The same trend was found in the numerical simulations by Lin [55] for an isolator entrance Mach number of $Ma_2 = 2.6$.

All studies described above were done with cold air as flow gas ($T_0 \sim 300$ K). The influence of heat transfer on a shock train at $Ma_2 = 2.9$ was investigated by Cuffel and Back in [17]. They used precombustion to achieve a total temperature of the gas of $T_0 = 835$ K and applied a cooling system to the walls to vary the wall temperature. Two cases were compared, one with wall cooling at a wall temperature of $T_w = 384$ K and one where heat was added to the flow ($T_r = 782$ K) with a wall temperature of $T_w = 885$ K. The shock train showed an increasing length for the case with the hot wall where heat was added to the flow field. Also

they found a correlation to describe the wall heat flux in the shock train as a function of the pressure relative to the values upstream of the first separation:

$$\frac{c_h}{c_{h2}} \propto \left(\frac{p}{p_2} \right)^{0.8}. \quad (2.63)$$

This means the heat flux in the shock train scales very similar to the heat flux in a shock wave boundary layer interaction without the confinement of a duct where the relation is identical with an exponent of 0.85 [16].

The boundary layer momentum thickness in the shock train was measured in [12] and [72]. In both investigations the momentum thickness scales well with the pressure distribution very similar to the heat flux relation described above. An exponent of 1.07 was found for the correlation of pressure and momentum layer thickness with experimental data inside the shock train with data from [72]:

$$\frac{\delta_2}{\delta_{2,2}} \propto \left(\frac{p}{p_2} \right)^{1.07}. \quad (2.64)$$

In [51] two isolators with rectangular cross section were compared. One was made of a heat sink material and the other one had the same geometry but was coated with an isolating ceramic material resulting in lower heat transfer from the hot flow ($T_0 = 805$ K). Shorter shock trains were observed for the isolator with the ceramic coating. The authors speculated on a possible influence of the surface roughness here. The ceramic coating provides a smoother surface which is not quantified further. Still an influence of the smaller heat transfer due to the coating should be considered here as well. This would lead to the conclusion that the shock train length is decreasing with increasing wall temperature. This is in opposition to the results from [17]. Still here the effect of different wall temperatures which were all cooling the flow field was investigated in opposition to [17] where the effect of heat addition was shown.

The influence of the total temperature on the flow field in a rectangular isolator was investigated at $Ma_2 = 1.8$ and $Ma_2 = 2.2$ in [52]. The total temperature was varied by precombustion of the test gas then flowing through an isolator with heat sink walls. The heat addition was shown to potentially cause choking of the flow channel. For the unchoked flow cases the increase in total temperature led to an increase in shock train length. A colder wall in relation to the flow temperature led to an increase in shock train length.

The influence of the total temperature on the flow field in a rectangular isolator was investigated at $Ma_2 = 1.8$ and $Ma_2 = 2.2$ in [27]. There the wall temperature influenced the combustion process leading to higher isolator exit pressures p_3 with increasing wall temperatures. Still the pressure gradient at the beginning of the shock train seems to be lower for lower wall temperatures also indicating a longer shock train for decreasing values of T_w .

The effect of forced transition on the shock train in a rectangular duct was studied at isolator entrance Mach number between $Ma_2 = 1.7$ and $Ma_2 = 2.5$ in [26]. For $Ma_2 = 2.5$ and a laminar boundary layer the pressure distributions fits the correlation by Waltrup and Billig (equation 2.61) qualitatively but a linear pressure rise throughout the shock train is shown for the turbulent case. This is shown by the pressure distributions in Fig. 2-20 and Fig. 2-21. At $Ma_2 = 1.95$ the transition has no qualitative effect on the pressure distribution in the shock trains. The gradient is high at the beginning of the shock train and decreasing further downstream. At the higher inlet Mach number of $Ma_2 = 2.46$ the pressure distribution shows the same pattern for the experiments without boundary layer trip. With the turbulent boundary layer the pressure in the shock train rises linear.

From this the conclusion is drawn that for oblique shock trains (high Ma_2) the turbulent boundary layer upstream of the shock train results in a shock train with a linear pressure rise throughout the shock train.

All studies mentioned above describe idealized isolators with symmetrical inflow. An isolator in a Scramjet engine (e.g. in configuration of the Scramjet model used for the research describes in this thesis) will likely have an asymmetrical inflow. Naturally the results of a study for such a configuration are very specific to the configuration applied. An extensive study with a focus on the influence of the lip position and the boundary layer thickness entering the isolator was performed by Emami et al. [24] where 250 different inlet/isolator combinations were investigated.

Extensive overviews over the research on shock train are given in [6] and [58].

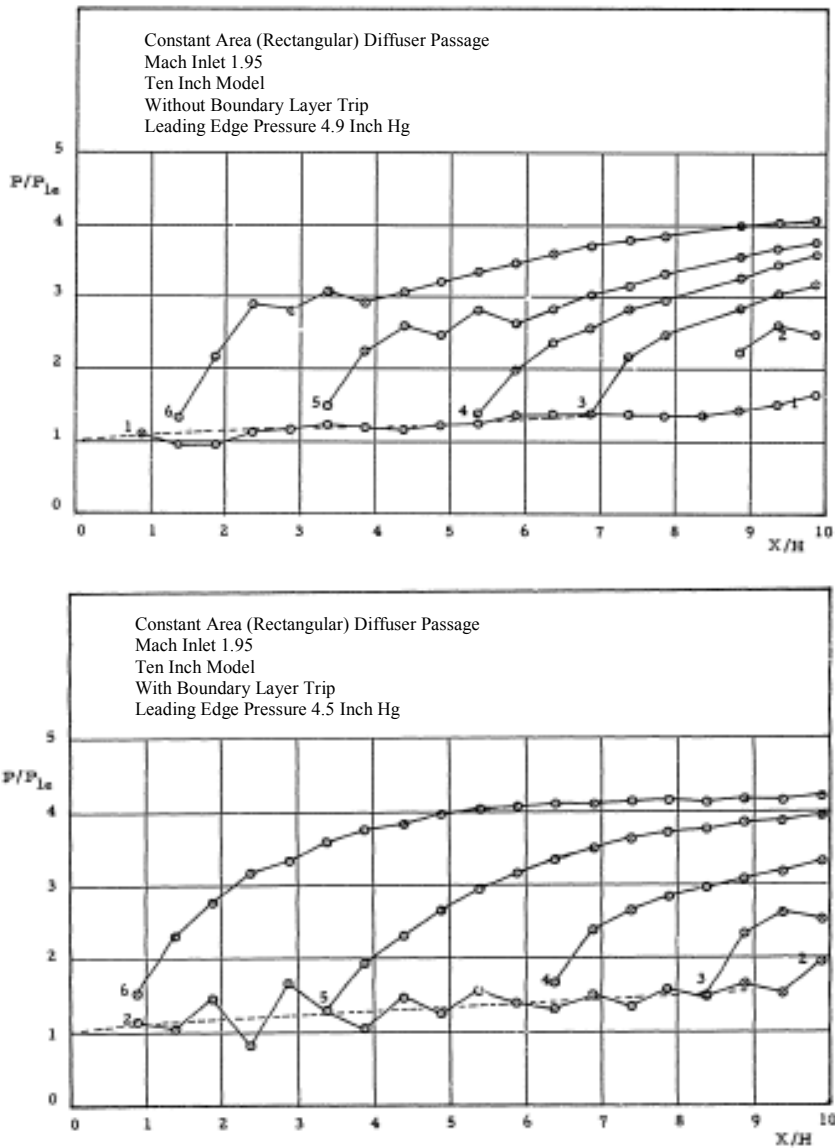


Fig. 2-20: Shock Trains at $Ma_2 = 1.95$ with (bottom) and without forced transition of the boundary layer from [26]

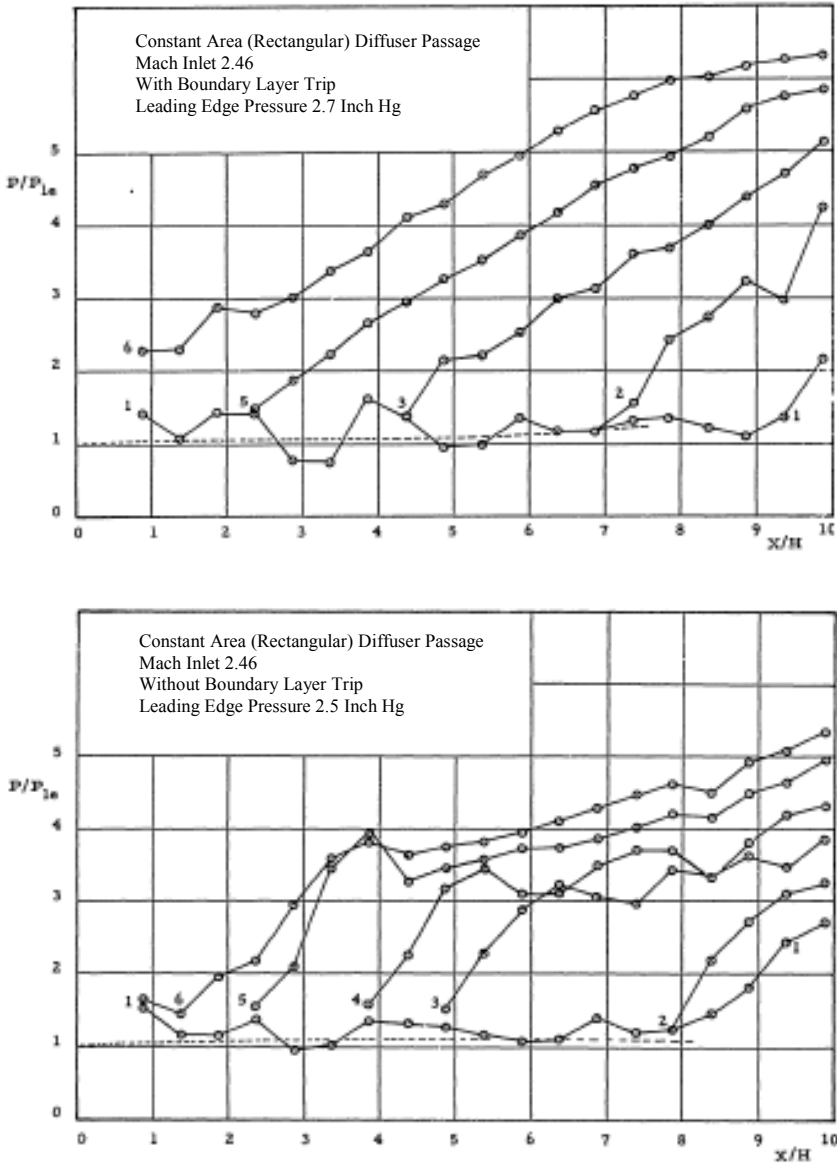


Fig. 2-21: Shock Trains at $Ma_2 = 2.46$ with (top) and without forced transition of the boundary layer from [26]

3 Experimental facility, measurement technique and numerical method

3.1 Hypersonic shock tunnel TH2

To simulate flight conditions for hypersonic vehicles at low enthalpies, i.e. for perfect gas behavior, in ground facilities, besides the high Mach and adequate Reynolds number, high stagnation temperatures are necessary to match the static temperature at the flight Mach number. Matching the flight Reynolds number is usually possible for smaller configurations, like used in the HyShot program (chapter 2.4.1). The correct reproduction of the static temperature is necessary for studies of the boundary layer phenomena in a Scramjet setup. As discussed in chapter 2 the relation between wall and static temperature can be used as a rough similarity parameter, but for the reproduction of shock wave boundary layer interactions realistic values of static temperatures are needed. For the experimental simulation of supersonic combustion naturally the realistic (high) values of flight static and total temperature are needed. The design point free stream conditions used for the research training group engine are listed in Tab. 3-1 showing this demand.

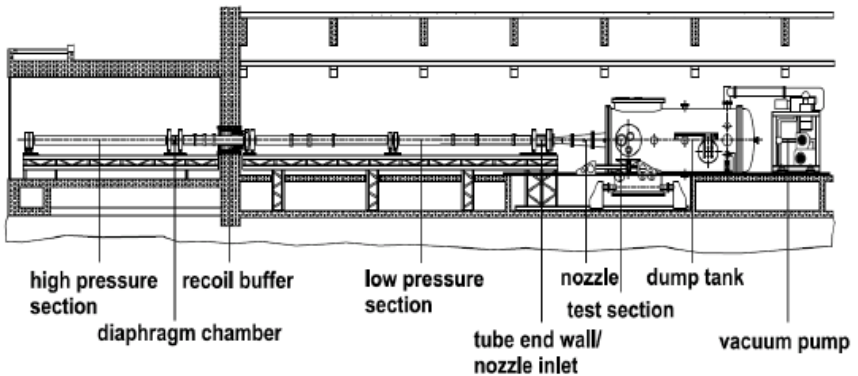


Fig. 3-1: Drawing of the hypersonic shock tunnel TH2 in helium driven mode [66]

The experiments described in this thesis were done in the reflected shock tunnel TH2 at the Shock Wave Laboratory using the helium driven mode [29] where stagnation temperatures up to 4600 K can be reached. The tunnel in this configuration is shown in Fig. 3-1. For higher

stagnation temperatures up to 7400 K a detonation driven mode [32], [75] is also available. The downside of the shock tunnel is the short measurement time in the range of a few milliseconds. The overall length of the tunnel is about 30 m.

Tab. 3-1: Design conditions of the research training group Scramjet

Ma_∞ [-]	H [m]	T_0 [K]
7	30000	2450

3.1.1 Principle of the shock tunnel

A shock tunnel with diaphragm, the TH2 in this case, is divided into 3 principle sections:

- the high pressure section
- the low pressure section
- the nozzle and the test section

The high and the low pressure section are working as a shock tube providing the test gas. This test gas is then accelerated to the desired Mach number in the nozzle to create the flow for the test section.

Before the experiment the whole tunnel is evacuated. The nozzle and the test section will remain at 0 bar. The high pressure section is filled with the driver gas ($p \geq 100$ bar) and the low pressure section with the driven gas ($p \leq 1$ bar). The driven gas will then be the test gas for the experiment. The two sections are divided by the diaphragm chamber. The chamber has a small volume filled with a gas at about half the pressure difference of the two sections. It is divided from the two sections by one diaphragm on each side. A third thin diaphragm is dividing the low pressure section from the nozzle. The experiment is started by opening the diaphragm chamber to a larger volume at lower pressure. The pressure drop results in a higher pressure difference between the high pressure section and the diaphragm chamber. The first membrane and shortly afterwards the second as well bursts and starts the experiment ($t=0$).

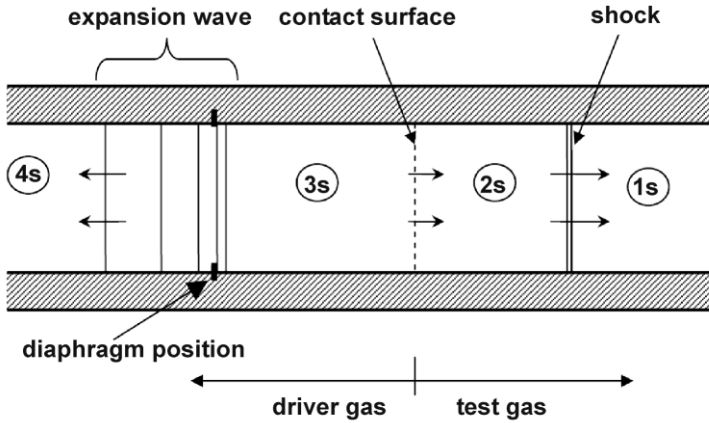


Fig. 3-2: Drawing of the shock tube section after bursting of the diaphragms ($t > 0$) [35]

The flow in the shock tube after the bursting of the diaphragms is drawn in Fig. 3-2. The discontinuity between high and the low pressure section leads to a shock moving into the low pressure section. This shock is driven by the contact surface between the two gases that is also moving into the high pressure section. An expansion wave is propagating into the high pressure section to lower the pressure from state 4s to 3s. All three phenomena start at the diaphragm position at $t = 0$. State 1s and 4s are the conditions in the low and the high pressure section at the beginning of the experiment. State 2s describes the condition behind the shock that is running to the right and the condition behind the contact surface is described by State 3s. The shock is moving with supersonic speed relative to gas with state 1s and creates an almost discontinuous jump in pressure, temperature, density and velocity. Over the discontinuity between state 2s and 3s only temperature and density are changing while pressure and velocity remain constant. The expansion wave is moving into the gas in the pressure section high reducing the pressure, the density and the temperature to the values of state 3s. The first characteristic at the border to state 4s is moving to the left with higher speed than the last one at the border to state 3s. This is caused by the higher speed of sound at state 4s and the movement of the gas at state 3s to the right. Therefore the wave is expanding with time.

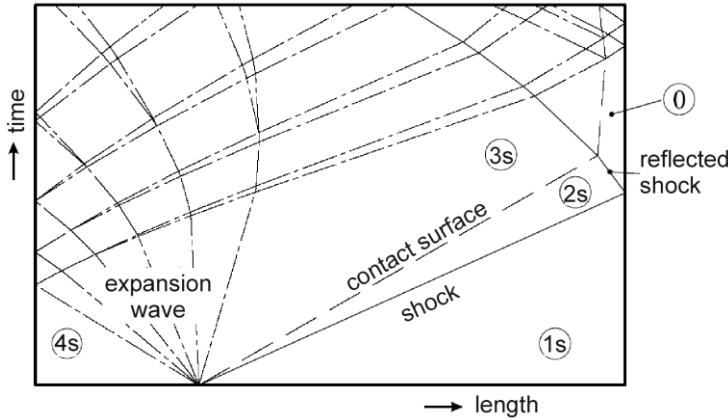


Fig. 3-3: *x-t diagram of the shock tube process in the TH2 facility in helium driven mode [35]*

The rightward moving shock is reflected at the right endwall of the shock tube. This creates a gas with high pressure and temperature. If the shock tube is closed at this end the velocity of the gas is zero. The conditions after the reflection at the endwall are the stagnation conditions for the free stream in the nozzle and the test section (0). The suddenly increased pressure at the endwall bursts the membrane between the low pressure section and the nozzle and the nozzle flow starts to build up. The whole process is shown in the *x-t* diagram in Fig. 3-3.

The theoretical limit for the measurement time is given by the time with constant nozzle reservoir conditions. For tailored interface conditions the interaction of the contact surface does not create a compression or expansion. Then the measurement time is the time between the impingement of the shock at the right endwall and the arrival of the first characteristic of the expansion wave at the right endwall. The reached nozzle stagnation condition (0) depends on the pressure relation p_{4s}/p_{1s} , the temperature relation T_{4s}/T_{1s} , the molecular weight relation M_{4s}/M_{1s} and the initial pressure p_{1s} and temperature T_{1s} . To reach high stagnation temperatures a relative hot and light driver gas in relation to the test gas is needed. This explains the use of helium in the conventional mode. In the conventional mode the high pressure section can be heated up electrically in order to achieve higher stagnation temperatures T_0 . A more detailed description of the shock tube process including the calculation of all the conditions can be found in [2].

3.1.2 Free stream conditions

The test gas at stagnation conditions is accelerated through the nozzle to reach the desired test free stream conditions. In an idealized case the free stream conditions could be calculated directly from the initial shock tube conditions (1s and 4s) assuming adiabatic nozzle flow. Deviation from these idealized conditions is created by viscous effects in the shock tube and nozzle and application effects like the opening of the membranes. Therefore the real free stream conditions are determined by the measurement of the pitot pressure and the heat flux in the stagnation point of a sphere in the test section [73], [99]. The static pressure in the test section is also required for the procedure. It can be either measured directly [45] or calculated from the stagnation pressure p_0 and the measured pitot pressure. Assuming a normal shock in front of the pitot probe the free stream Mach number is determined and from this the static pressure.

The free stream conditions for the experiments with the Scramjet and the isolator model are listed in Tab. 3-1. They are averaged over the test time which is in the range of about 2-5 ms. The nozzle of the tunnel is a slender conical nozzle with a half apex angle of 5.8° . This creates small axis gradients in the test section. They have been calibrated [74] and are listed in Tab. 3-2 for conditions I and II. For Condition I it has to be noted that the calibration was done with a slightly different driver gas mixture (see chapter 3.1.3). Still no significant offset for the gradients is expected and they have been proven in use for the flow parameters listed in Tab. 3-2 [35], [67]. They are also taken into account for the determination of the pressure coefficient (eq. (3.1)) and the Stanton numbers (eq. (3.2)) at the pressure probe and thermocouple positions by the evaluation of the local reference parameters [8]. To place the isolator parts of the model in the axis of the tunnels optical access the model is placed in a position with the leading edge about 265 mm upstream of the nozzle exit plane. The free stream conditions in Tab. 3-2 are listed for this position.

$$c_p = c_{p,x} = \frac{p - p_{\infty,x}}{0.5 \rho_{\infty,x} u_{\infty,x}^2} \quad (3.1)$$

$$C_H = C_{H,x} = \frac{\dot{q}}{\rho_{\infty,x} u_{\infty,x} c_p (T_0 - T_w)} \quad (3.2)$$

Tab. 3-2: Free stream condition at the models leading edge and gradients of free stream conditions

	Condition I	Condition II
Ma_∞ [-]	7.5	7.3
Re_∞ [1/m]	$5.8 \cdot 10^6$	$4.8 \cdot 10^6$
T_0 [K]	1150	2200
T_∞ [K]	100	220
p_∞ [Pa]	460	2000
$\Delta Ma_\infty / (Ma_\infty \cdot \Delta x)$ [1/m]	0.209	0.112
$\Delta p_\infty / (p_\infty \cdot \Delta x)$ [1/m]	-1.061	-0.525
$\Delta u_\infty / (u_\infty \cdot \Delta x)$ [1/m]	0.033	0.055
$\Delta \rho_\infty / (\rho_\infty \cdot \Delta x)$ [1/m]	-0.817	-0.443

Both conditions listed in Tab. 3-2 feature similar Mach and Reynolds numbers and differ in free stream and total temperature. Using the model heating system the ratio between wall and free stream temperature can be kept constant while changing the parameters themselves. Whereas Condition II is close to flight conditions with respect to total enthalpy, Condition I is more for comparison purposes ensuring the same wall to free stream temperature ratio as for Condition II. At Condition I also principal effects like the variation of the wall temperature are realized with less experimental effort. Also at Condition I wall temperature can reach values very close to the recovery temperature (eq. 2.29). This creates flow cases with very small wall heat transfer (eq. 2.36).

3.1.3 Measurement time

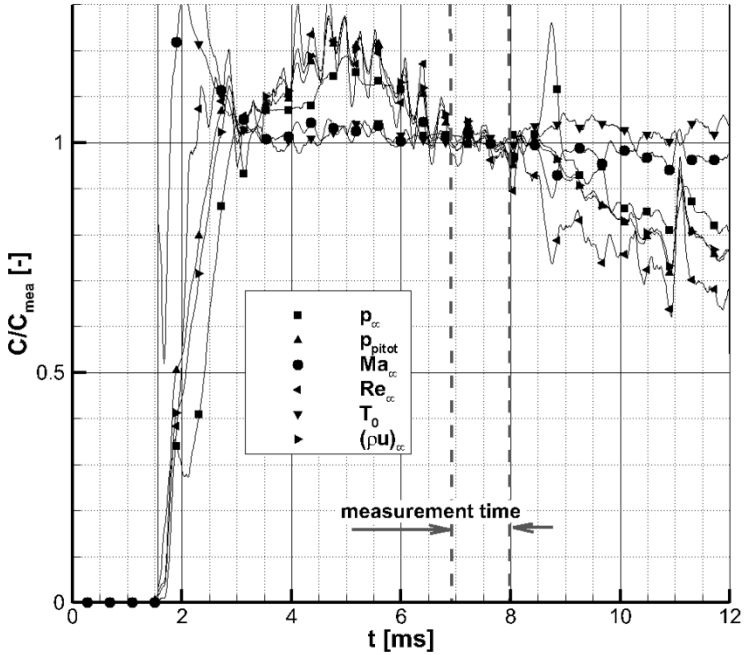


Fig. 3-4: Free stream conditions in the test section during an experiment at Condition I

The available measurement time is defined by the time frame with constant free stream flow conditions in the test section. Some typical free stream conditions C are plotted against time in Fig. 3-4. They are normalized with the mean values for the measurement time marked by the two dashed vertical lines. The experiment was done at Condition I using pure helium as driver gas in the high pressure section.

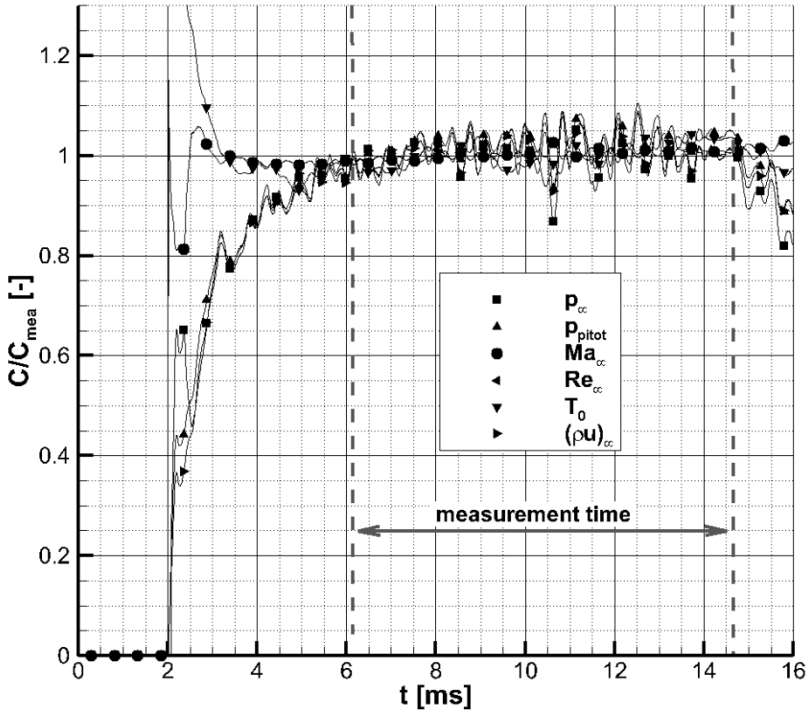


Fig. 3-5: Free stream conditions in the test section during an experiment at Condition I with a helium air mixture

Constant conditions are reached between about $t = 7$ ms and $t = 8$ ms leading to a measurement time of about 1 ms. The same quantities for an experiment also at Condition I just with a small amount of air added to the high pressure section are shown in Fig. 3-5. The air is added to the helium during the recycling of the gas. After the experiment the test section contains a mixture of helium from the high pressure section and air from the low pressure section. This mixture is combined with new clean helium to provide a gas with a constant composition. It has a composition of 97.5% helium and 2.5% air (amount of substance). This is the composition after a shot with pure helium in the high pressure section. The addition of air leads to a driver gas with a higher molar mass and therefore a reduction in total temperature in the free stream of the test section from about 1400 K to 1150 K at Condition I and from 2500 K to 2200 K at Condition II.

Besides the recycling of the helium the big advantage of the application of the helium air mixture is a massive increase in measurement time from 1 ms with pure helium to about 8 ms with the mixture. All results shown in this thesis were obtained with the mixture to take advantage of the longer measurement time.

For the experiments concerning the shock train a long measurement time is a huge advantage as the shock train contains large subsonic areas and a steady flow field might not develop during the test time. For the experiments with the helium air mixture not only the measurement time is longer but also the free stream quantities raise more shallow without the local maximum prior to the measurement time. This also increases the chance for the development of a steady flow field with a shock train during the measurement time. An example for a useable experiment is given in Fig. 3-6. Plotted are wall pressure coefficient distributions in the heatable isolator model for two flow cases with and without back pressure induced. The origin of the x coordinate is the isolator entrance. The shock train starts at about $x = 165$ mm and leads to a more or less linear increase in pressure. The single values shown in this graph are obtained by averaging the measured pressure signals over the measurement time.

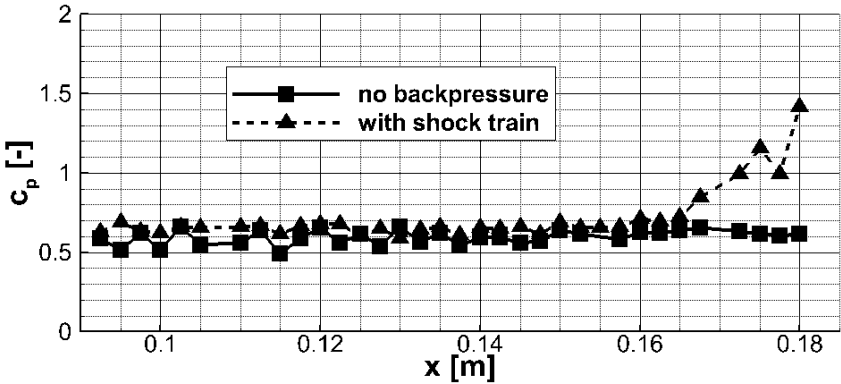


Fig. 3-6: Pressure distribution in isolator model for Condition I, $Ma_2 = 3.5$, $T_w = 1000K$ with and without back pressure

Some of those pressure coefficient plots over time are shown in Fig. 3-7 and Fig. 3-8. The first four plots for $x = 155$ mm to $x = 162.5$ mm are taken upstream of the shock train and show an even distribution over a large amount of time.

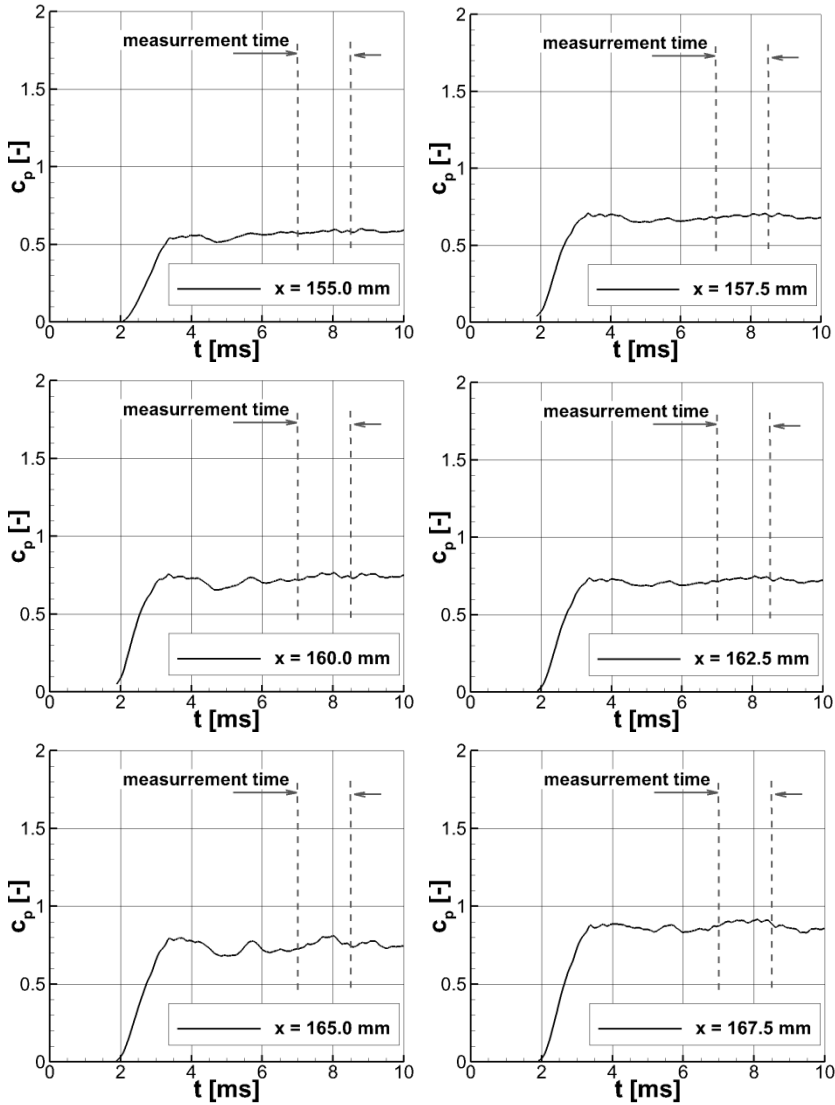


Fig. 3-7: Wall pressure coefficients for different probe positions for the experiment shown in Fig. 3-6

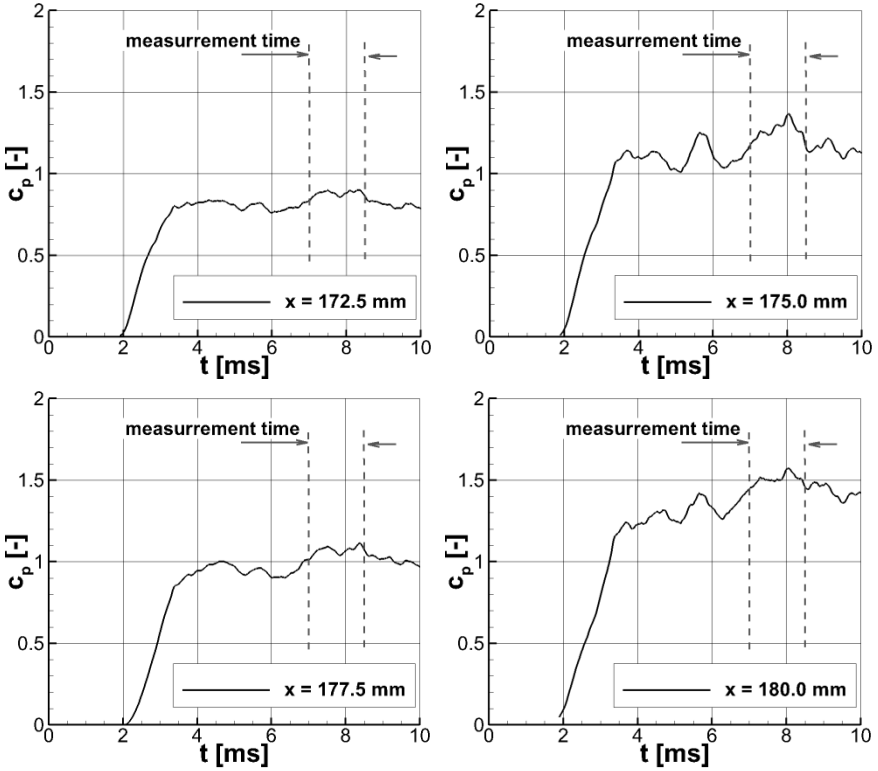


Fig. 3-8: Wall pressure coefficients for different probe positions for the experiment shown in Fig. 3-6 (continued)

In the four probes within the shock train the pressure level is rising and fluctuations over the time are visible with a first local maximum at about $t = 5.5$ ms and a second larger maximum between $t = 7$ ms and $t = 8.5$ ms where the measurement time is set. The fluctuations are visible in all four plots showing that the shock train is adapting to the downstream pressure fluctuation without a visible delay and therefore the measurement time is sufficient to capture the shock train.

An example for a non applicable experiment is shown in Fig. 3-9 and Fig. 3-10. No steady flow field for all pressure probes is reached at any point in time during the available measurement time between about $t = 6$ ms and $t = 10$ ms. The last pressure probe on the ramp ($x = 177.5$ mm) shows a more or less linear pressure increase during the measurement time. An almost constant pressure level is reached at about $t = 6$ ms until $t = 7$ ms.

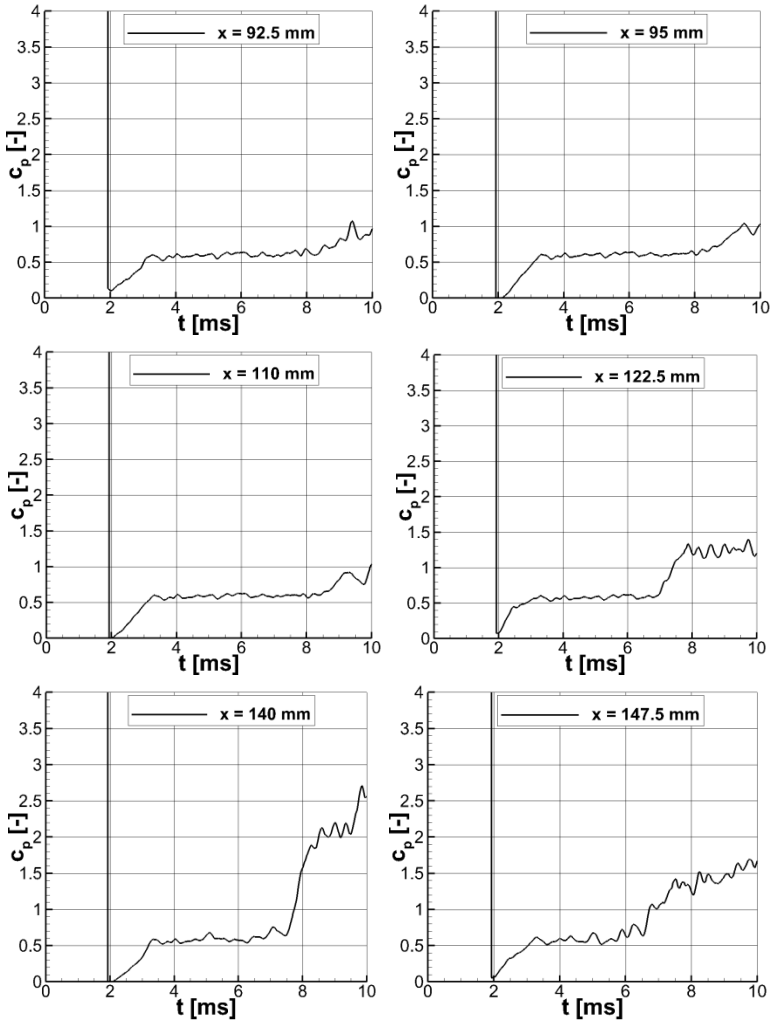


Fig. 3-9: Wall pressure coefficients for different probe positions of one experiment

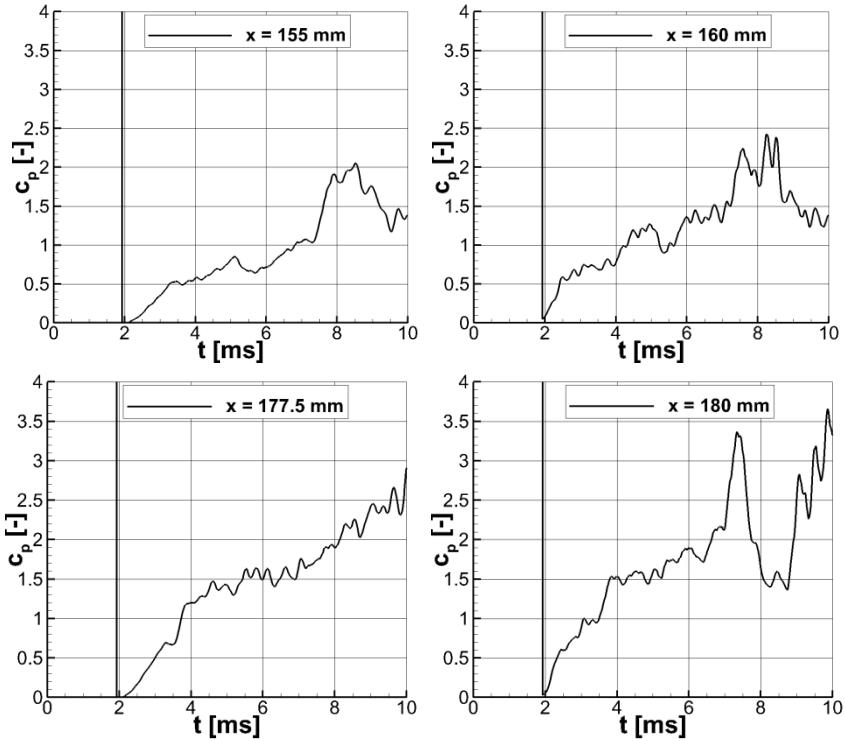


Fig. 3-10: Wall pressure coefficients for different probe positions of one experiment (continued)

In this time frame a pressure rise for the probes at $x = 147.5$ mm and 155 mm can be seen showing that the flow field is not stationary. Another pressure plateau is reached at the very end of the measurement time after $t = 9$ ms.

The last probe on the lip at $x = 180$ mm shows a high increasing pressure here ruling out a steady flow field in this time frame. The flow phenomenon is growing upstream into the isolator, the pressure probes downstream of $x = 147.5$ mm are inside the shock train and show increased pressure levels throughout the measurement time. The shock train reaches the probe at $x = 147.5$ mm at about $t = 7$ ms. All available pressure probes are affected at $t = 10$ ms. The growth of the shock train is asymmetric. The probe at $x = 122.5$ mm on the ramp is reached earlier by the shock train than the probe at $x = 140$ mm. The shock train is asymmetric and unsteady and therefore this experiment is not applicable. Naturally only experiments that

reached a steady symmetric flow field during the measurement time have been used for the results later presented in this thesis.

3.2 Measurement technique

Wall and pitot pressure measurements as well as schlieren and shadowgraph visualisation are used for the characterisation of the flow field. In the first measurement campaign with the Scramjet model the wall heat flux is also measured with type K thermoelements [82]. The more robust signals of the pressure probes proved to be much more applicable for the determination of the flow structure in the isolator. The heat flux measurements are used as validation data for numerical simulations [69]. As this thesis itself does not contain heat flux measurements they are not described in detail here.

The campaign showed the need for a higher resolution of wall pressure measurements to resolve the flow field fine enough to quantify the wall temperature effects. Due to the limited number of measurement channels of the tunnel the heat flux measurements were dropped for following measurement campaigns and more pressure probes were installed to achieve a higher spatial resolution.

3.2.1 Pressure measurements

For the wall and the pitot pressure measurements Kulite XCQ-080 pressure transducers are used. Due to their small membrane diameter of 0.7 mm their natural frequency is very high (300-500 kHz). This makes them applicable in short duration facilities. They have a cylindrical shape with an outer diameter of 2 mm enabling a high spatial resolution. Their maximum operation temperature is 390 K while the maximum surface temperature of the model is 1000 K. Therefore cooling is required as they are placed just 8 mm away from the hot wetted surfaces.

The geometry of the whole system pressure probe, cooler and connection pipe to the surface has a huge impact on the ability of the pressure probe to record the high temporal pressure gradients in the shock tunnel. The dynamic response of the system is limited by the lowest eigenfrequency of the system. In [8] the lowest value for this frequency was calculated using an approach from [21]. Applying the formula (eq. 3.3) to the current geometry of the system leads to an upper boundary of the dynamic response of $f = 3.0$ kHz. The static pressure probe reacts fastest to the pressure rise in the test section [45]. It needs 1.5 ms to reach the stationary static free stream pressure. This is slower than the limit set by the eigenfrequency of $\tau_r = 0.17$ ms (eq. 3.4). Therefore pressure probes in the coolers are fast enough for the use in the shock tunnel.

$$f_e = \frac{\sqrt{\gamma RT}}{8\pi} \sqrt{\frac{3\pi d_i^2}{L_{Mess} V_C}} \quad (3.3)$$

$$t_r = \frac{1}{2f_e} \quad (3.4)$$

Probes with different operating ranges are applied throughout the models. Probes with operating ranges of 1.7, 7 and 17 bar are available at the Shock Wave Lab. They survive 2 times their operating pressure without a change in their calibration value and 3 times without bursting. For models only tested at Condition I (cold isolator model and mass flow measurement test model) probes with all operating ranges are applicable. Pressures get higher for tests with Condition II. Probes with a measuring range of 17 bar are used in the pitot rake as pitot pressures over 7 bar are possible due to the effective pressure recovery of the many shocks the flow passes throughout the inlet compression and the shock train. Seven bar probes are used for the wall pressure measurements in the shock train area and 1.7 bar probes for upstream area of the isolator.

3.2.2 Uncertainty of measured values

A detailed analysis of the measurement uncertainties for heat flux and pressure determination is found in [8]. They are estimated with $\pm 7.1\%$ maximum for the pressure coefficient c_p . This value represents a worst case scenario based on worst case manufacturer data. For practical applications it can be assumed that the uncertainty is lower [76]. The calculation of the uncertainties does not take into account the effect of background noise. The background is an absolute fluctuation of the signal not a relative one. This applies for the static pressure measurement at Condition I. Here a very low static pressure $p_\infty = 460$ Pa is measured with a probe that has an operating range of 170000 Pa in order to survive the higher pressures in the test section after higher stagnation pressure experiments. The static pressure is needed to determine the pressure coefficients for the measured wall pressure. Due to the compression of the flow due to the ramps the wall pressure is considerably higher than the free stream pressure (e.g. ~ 9500 Pa in the isolator model for $Ma_2 = 3$) leading to an extremely small influence of this increased uncertainty on the values for the pressure coefficients.

3.2.3 Schlieren and shadowgraph visualisation

To determine the flow topology in the isolator schlieren and shadowgraph visualisation is applied. An optical system in Z-configuration is used. With shadowgraph imaging the second

derivative of the density is visualized. Parallel light is send through the test section. A change in density within the test section diffracts the light due to the change in light speed. The test section is imaged through an optical system and density changes are visualized as shown in Fig. 3-11 (top).

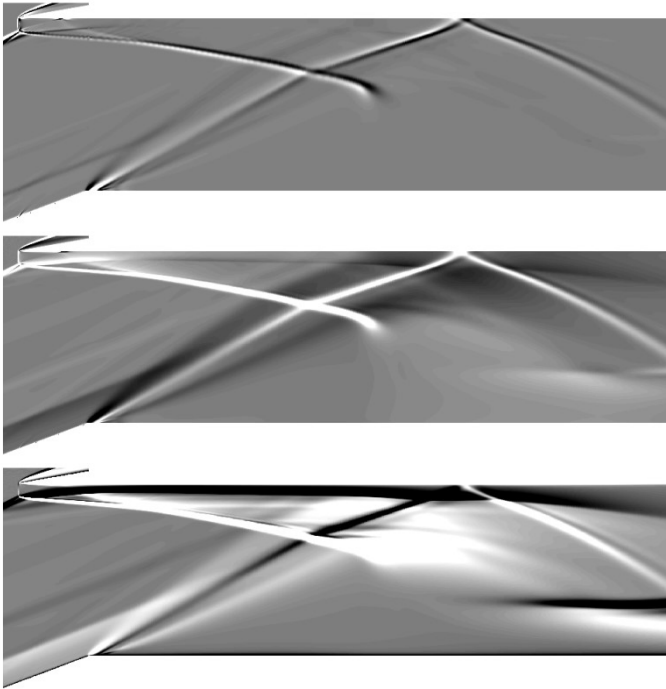


Fig. 3-11: Shadowgraph (top,) x-wise schlieren (middle) and y-wise schlieren (bottom) image for the same flow field.

The illumination intensity of the picture is proportional to the second derivative of the density. Therefore shocks are marked by the typical white and black stripe shown in the picture. Shadowgraph images are quite robust in case the optical path of the overall system passes through components (e.g. model windows) of poor optical quality.

Schlieren images show more details of the flow field (e.g. the y-wise schlieren image in Fig. 3-11 shows the boundary layers on ramp and lip). A small light source with a defined shape (e.g. a rectangle or a ring) is used here. Parallel light for the illumination of the test section is created by a parabolic mirror or a lens. The light source is projected onto a knife or a blend by a second lens or mirror. An additional lens is used to combine the second mirror

and this lens to project the test section onto the camera chip. The knife or blend is used to fade out a part of the light source. A density gradient will diffract the projection of the light source on the knife/blend. The two applied schlieren light source / blend or knife combinations are shown in Fig. 3-12. On the right side a one directional black and white schlieren system is shown. The light source (white) is projected onto the edge of a knife (yellow). A deviation normal to the knife will lead to a change in illumination in the final projection. The shape of the light source is used to influence the sensitivity of the setup. The left white light source is very sensitive as a small deviation has a large effect on the illumination. The opposite is the case for the right light source. The schlieren system shows only density gradient in the direction normal to the knife. Two typical resulting images are shown in the two lower pictures in Fig. 3-11. One shows the density gradient in x-wise direction and the other in y-wise direction. The x-wise schlieren image shows shocks consequently in white in the main flow direction and expansions black. The picture with the y-wise density gradient is not that consequent there but in principle is capable to show the boundary layers where only y-wise gradient exist.

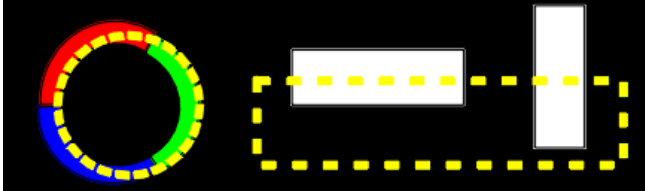


Fig. 3-12: Operating principle for color schlieren (left) and one-directional schlieren (right) imaging

It is also possible to apply a colored light source to show the direction of the gradient with different colors [46]. A ring shaped light source with three colored sections (Fig. 3-12) is used here. This ring is projected on a blend (yellow). The deviations caused by density gradients will then, depending on the direction, include or exclude the colors. Here the default image has a mainly green color. For this thesis shadowgraph, color schlieren as well as x-wise black and white schlieren imaging is used. More details on the visualisation techniques described here can be found in [81].

3.3 Numerical method

In addition to the experiments numerical simulations are carried out. They were used to enhance the understanding of the experiments as they provide the full coverage of the flow field in opposition to the limited data from the experiments. Simulations are also used to gain knowledge of certain flow parameters like the boundary layer momentum thickness and to study possible boundary layer bleed options. The commercial software package CFX by Ansys is used for all simulations. It solves the Navier Stokes equations and employs the finite volume method with a second order accurate scheme. Turbulence is modelled with the SST turbulence model [60]. The transition process is modelled with the Menter/Langtry model [48], [61]. In [67] it is described how it is used in CFX. The grids are generated with the package ICEM also from Ansys. Most performed calculations are two dimensional. There the two dimensional meshes are extracted by one cell to the third dimension to make them compatible with CFX.

The CFD code CFX was validated at the Shock Wave Laboratory for several hypersonic flow experiments in the TH2 shock tunnel [65], [67]. Also the parameters for turbulence and transition modelling were calibrated with data for two dimensional flow from [64].

3.3.1 Mesh splitting

To simulate the whole system of inlet and isolator mesh splitting was applied in order to keep the size of the meshes reasonable. The three domains are shown in Fig. 3-13. The whole inlet is simulated with one mesh. For the isolator the calculations are carried out in two steps with two meshes, one for the front part with the lip shock induced separation bubble and one for the rear part with the shock train.

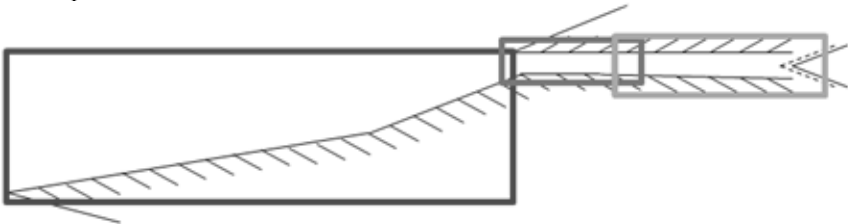


Fig. 3-13: Domains for mesh splitting

In opposition to the experiments for the simulations free stream conditions with identical Mach and Reynolds number are used here for the two test conditions. Also the free stream conditions differ slightly from the experimental conditions. They are listed in Tab. 3-3. They

are based on the pure helium conditions I and II of the TH2 shock tunnel [35] with small modifications to achieve identical Mach and Reynolds number.

Tab. 3-3: Modified free stream conditions for numerical simulations

	Condition I	Condition II
Ma_∞ [-]	8	8
Re_∞ [1/m]	$4.6 \cdot 10^6$	$4.6 \cdot 10^6$
T_∞ [K]	120	240
p_∞ [Pa]	723	1910

3.3.2 Mesh sensitivity

The mesh sensitivity for the outer compression part (inlet) of the model was investigated in [65]. The results of the mesh sensitivity study for the front part of the isolator are shown in Fig. 3-14. The study was done at Condition I, $T_w = 600$ K using turbulence and transition modeling as described above. An almost identical pressure distribution is shown for the three high resolution cases (730 x 400; 1000 x 400; 730 x 550). Some deviation is visible for the other two cases with lower resolution. Therefore the mesh with 730 x 400 cells is used for the CFD calculations.

The results of the mesh sensitivity study for the rear part of the isolator are shown in Fig. 3-15. Again turbulence and transition modeling is used. An overall pressure rise with $p_3/p_1 = 300$ was generated with an “opening” exit condition with the same flow and wall parameters as for the front part of the isolator. There is no mesh sensitivity present for any of the investigated flow cases. As the flow field here will vary intensely with different back pressures the mesh 700 x 300 cells was chosen for the calculations to ensure mesh independence for all possible calculated flow cases.

The calculations for the isolator of the Scramjet model are two dimensional since a very fine mesh is needed for the complex flow topology. Still two dimensional calculations have been proven useful and reliable to model flow phenomena for the given setup [67],

For the heatable isolator model three dimensional calculations were done to obtain pressure distributions for the flow case without back pressure assuming laminar and turbulent boundary layer flow in the isolator. These calculations are used to determine whether the flow field in the isolator in the experiments is laminar or turbulent.

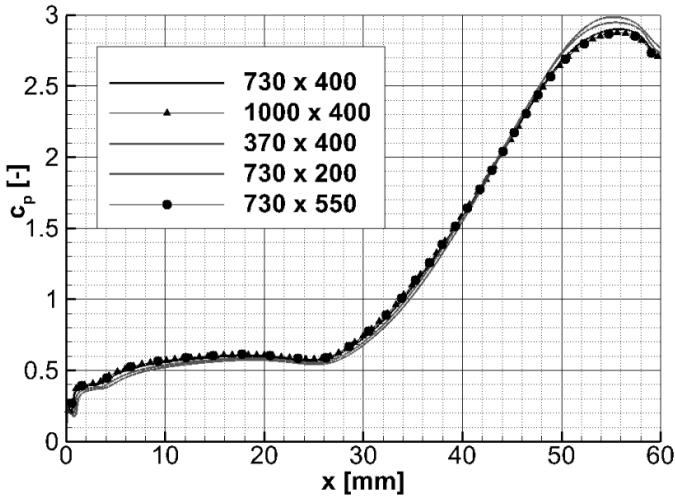


Fig. 3-14: Pressure distribution in the front part of the isolator for different meshes at Condition I, $T_w = 600\text{ K}$

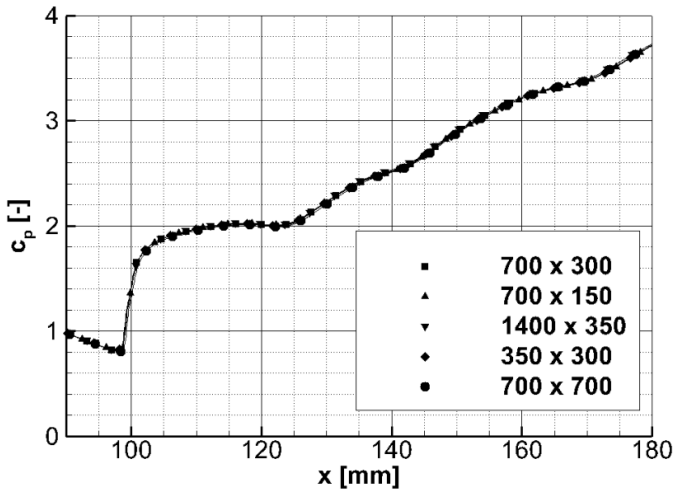


Fig. 3-15: Pressure distribution in the rear part of the isolator with back pressure ($p_3/p_1 = 300$) for different meshes at Condition I, $T_w = 300\text{ K}$

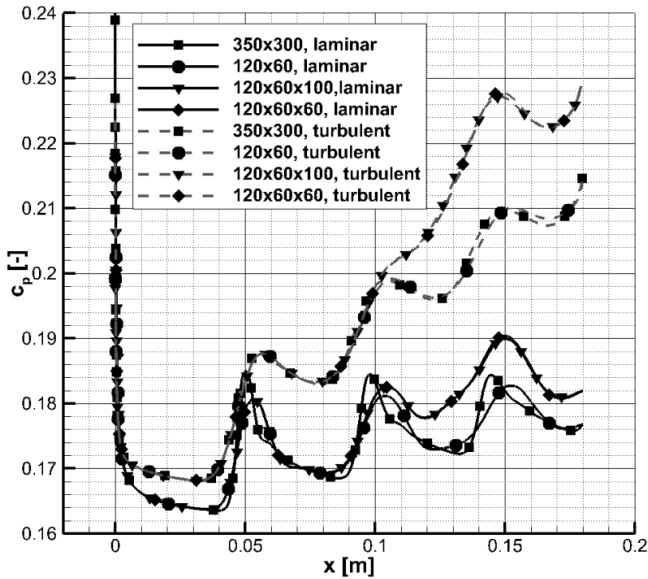


Fig. 3-16: Isolator pressure distribution without back pressure with different 2d and 3d meshes

As only the flow without back pressure is simulated and also only the pressure distribution at the walls is investigated a very coarse mesh is used with $120 \times 60 \times 60$ grid points. The mesh covers one quarter of the flow channel applying symmetric boundary conditions to cover the volume of the flow channel.

Based on a rather fine 2D mesh (350×300 cells) a coarse mesh (120×60) is first tested in a 2D calculation and then expanded to 3D (60 cells). To check for mesh sensitivity a finer mesh with 100 cells in the third dimension is also used. The results are shown in Fig. 3-16.

For the two 3D meshes the pressure values are taken on the centerline for the flow channel. The pressure in the flow channel grows steadily throughout the flow channel due to the confinement of the growing boundary layer. At the leading edges the buildup of the boundary layer creates shocks that are continuously reflected throughout the flow channel. The impingement of these shocks causes the pressure peaks at about $x = 50$ mm, $x = 100$ mm and $x = 150$ mm. The fine 2d mesh (350×300) shows these impingements for laminar flow very sharp due to the high resolution. They are more diffuse for the coarse 2d mesh (120×60) and

the 3d meshes. Still the overall development of the pressure level is still captured by the coarse 2d mesh for both laminar and turbulent flow.

The 3d calculations show increased pressure levels downstream of $x = 100$ mm, when the leading edge shocks from the sidewall reach the centerline of the model. The increased number of cells in the third direction does not influence the pressure distribution. Therefore the 3d meshes are considered fine enough to determine the pressure distribution in the isolator without back pressure.

4 Model Technique

Three models have been used for the research presented in this thesis:

- the Scramjet intake model,
- the heatable isolator model,
- the cold isolator model.

4.1 Model heating technique

The first two models have heatable wetted surfaces. The surfaces are heated electrically with heating elements. The heating technique was first developed for the application in the TH2 shock tunnel by Bleilebens [8], [9]. Surface temperatures of up to 800 K were reached. For the Scramjet model wall temperatures up to 1000 K as well a more even temperature distribution are reached due to a denser layout of the heating elements in the plates [67]. The elements are placed inside grooves in the plates and the grooves are filled with a ceramic compound. Thermocouples are also glued into the plates close to the elements to monitor and control the heating process. The heating elements are controlled analogue with transformers and a control system developed for this thesis. An automatic mode for this system is available but manual control is advised due to possible fluctuations in the temperature signals.

The pressure probes work linear up to 80°C and survive temperatures up to 120°C. Due to the use of a short duration facility a short distance from the probe to the surface is required. Active and passive cooling is used to protect them from the hot wetted surfaces.

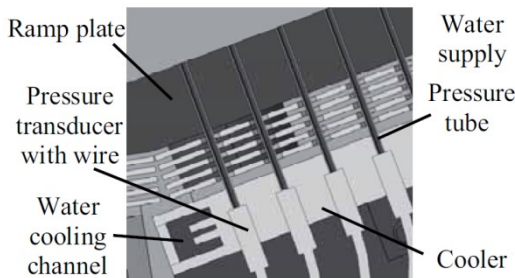


Fig. 4-1: Cut through the cooling section [67]

The cooling system for the pressure probes is shown in Fig. 4-1. The pressure probes are mounted into a cooler made out of brass. It is cooled by cold water (about 4-7 K below room temperature) flowing through. The coolers for the different areas of the models (ramp, lip, pitot rake) are connected parallel to the water supply in order to ensure the water entering all coolers at the same low temperature. The probes are connected to the wetted surfaces by stainless steel pipes with 0.8 mm outer and 0.68 mm inner diameter. The distance between the probes and the wetted surface is about 15 mm.

Between the cooler and the heated plate there is an assembly of five polished stainless steel plates for radiation isolation. The polished plates are divided by 0.5 mm washers. Originally these washers were made out of ceramics to ensure high thermal insulation. They are replaced with stainless steel ones as the ceramics washers broke due to the fitful forces in the shock tunnel. The stainless steel washers create higher heat conduction but the overall isolation performance was better as they fixed the sandwich plates and therefore circumvented contact between them. The overall assembly now also holds the coolers in the fixed position (Fig. 4-6).

The reached temperature distribution on the isolator centerline is shown in Fig. 4-2. The thermocouples for heat flux measurement on the centerline on the ramp and the lip of the isolator were used here. The test section was evacuated during the heating experiment to create a realistic environment for shock tunnel experiments. There is the option of a passive boundary layer bleed in the front part of the isolator. Therefore the front part of the ramp ($0 \leq x \leq 58.8$ mm; l_3 in Fig. 4-5) is not heated. This explains the low temperature in that area, also making it the coldest part of the intake and isolator ramps. The front part of the lip is also expected to be cold as there are no heating elements in that area due to design restrictions. The temperature distribution on the inlet ramps can be found in [66]. On the isolator ramp and lip as well as on the inlet ramps spatial temperature deviations were less than 10% of the nominal value of wall temperature in the heated area.

To determine the temperature distribution normal to the flow direction in the unheated area of the isolator ramp the windows in the side walls of the isolator model were removed. Through these openings thermocouples were put on the surface of the ramp at $x \approx 25$ mm. This is expected to be the coldest area of the model. The measured distribution is shown in Fig. 4-3. From the centerline outward until about 15 mm from the side wall the temperature stays within 5% of the center value then dropping rapidly. This is caused by the radiation of the red-orange glowing ramp plate side that was not covered by the window frame during the test.

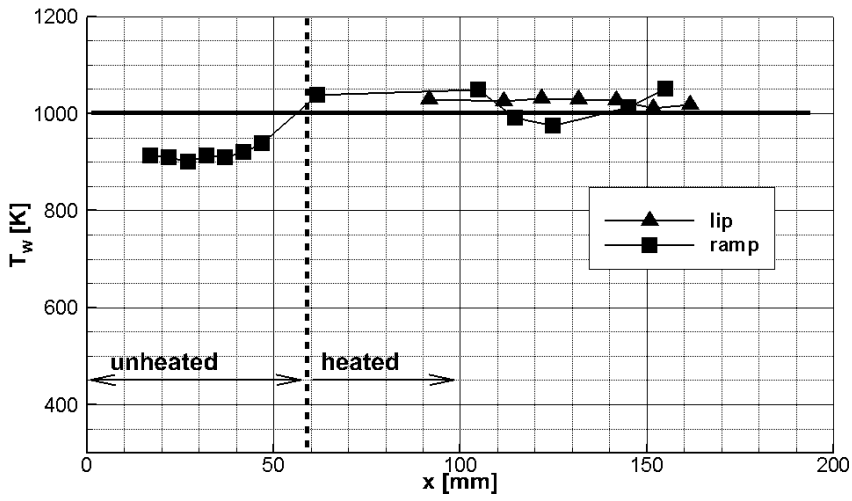


Fig. 4-2: Temperature distribution on the wetted surfaces of the isolator at a nominal wall temperature of $T_w = 1000$ K on the centerline in streamwise direction

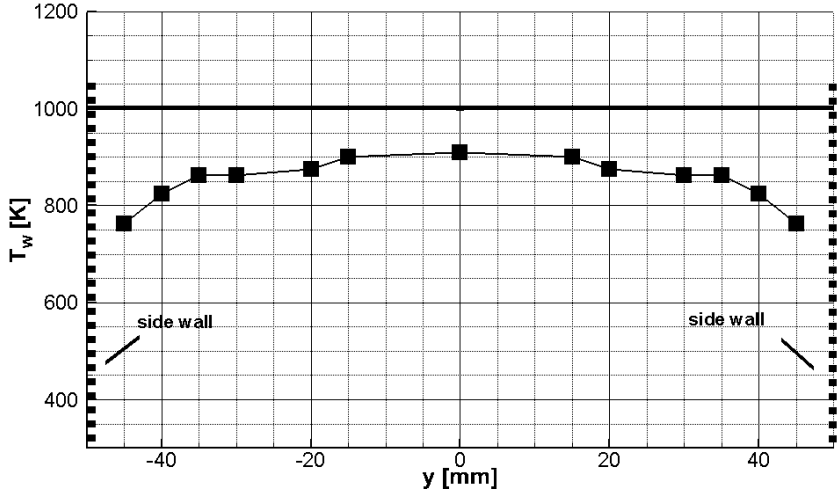


Fig. 4-3: Temperature distribution on the wetted surfaces of the isolator at a nominal wall temperature of $T_w = 1000$ K normal to flow direction at $x = 25$ mm

4.2 The Scramjet intake model

The Scramjet intake model was designed by Neuenhahn [67] following the inlet layout of the first phase of the research training group. It is a two dimensional inlet with two compression ramps followed by an isolator. A drawing of the model is shown in Fig. 4-4.

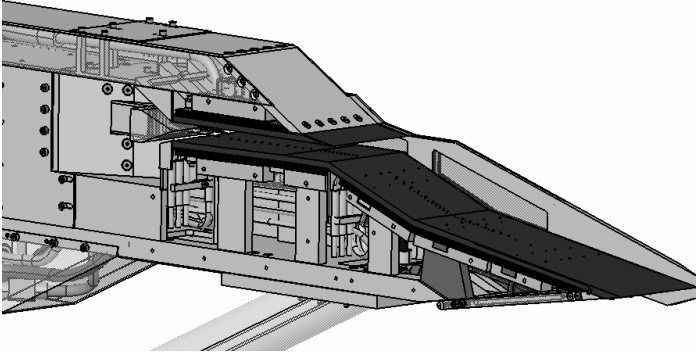


Fig. 4-4: Drawing of Scramjet intake model

4.2.1 Flow field

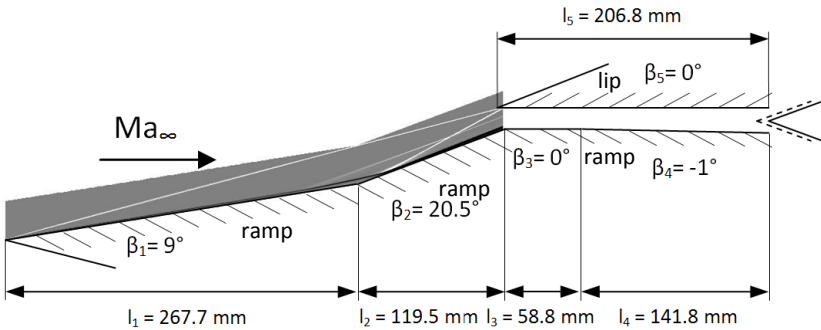


Fig. 4-5: Scramjet intake model geometry with numerical schlieren image of intake

The geometry of the model is shown in Fig. 4-5. The capture area of the model is 0.1 m x 0.1 m. After the first part of the isolator with constant cross section (l_3) an opening angle of -1° is applied to counteract the buildup of the boundary layer. The isolator is 15.5 mm high at the entrance and 18 mm at the exit. The leading edge of the lip is placed at the meeting point of the two compression shocks from the external compression. The shocks as well as separation bubble at the hinge line of the two ramps are visible. The separation

bubble was investigated in detail by Neuenhahn [66], [67]. At the exit of the isolator a movable double wedge is placed. It is used to generate back pressure for the investigation of the shock train.

4.2.2 Model modification for isolator investigation

During the tests for the shock wave/boundary layer interaction on the external compression ramps the lip was not attached to the model and therefore no isolator was present. Compared to the external compression the static pressure in the isolator reaches much higher values, especially when a shock train is created. With the pitot probes at the isolator exit pressures up to 10 bar at subsonic Mach numbers have been measured. Maintaining structural integrity becomes a major issue for the enclosed flow in the isolator. For the successful testing with the full model several modifications are taken from the original design.

In the original design of the model thermal expansion was considered the biggest issue. The mounting principle for the lip is shown in Fig. 4-6. The sidewalls are not connected directly to lip. The sandwich construction for radiation cooling is fixed by 4 screws. On each side one support part connects the sidewalls with lip trough the sandwich construction. The screws connecting the support parts with the sidewalls are loaded normal to their direction with the pressure force from the lip plate. This is not an issue as long as the absolute static pressure is low. This is the case for the external compression ramps. Assuming an average pressure of $p = 2$ bar (Condition II; $Ma_2 = 3$ $p_2/p_3 = 4$) the resulting pressure force on the lip would be 4000 N. This is obviously a critical shear load for the four M5 screws.

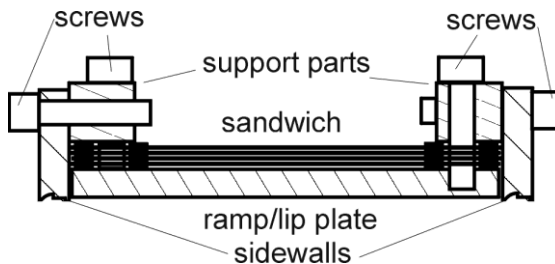


Fig. 4-6: Mounting principle for isolator lip

In addition the screws connecting the support parts with the lip are loaded in normal direction by the pressure force from the isolator sidewalls. This load is about five times smaller compared to the force of the lip ($h = 15.5\text{--}18$ mm; $w = 100$ mm). Problematic here is the

contact area to transport the forces. The support part has no direct contact to the lip but has to fix the sandwich construction with the washers as well.

Also the first screw fixing the lip is located about 100 mm downstream of the leading edge. This creates the risk of upward bending of the lips leading edge in cases due to high pressure inside the isolator.

Additional measures are necessary to unload the screws in normal direction and to prevent upward bending of the leading edge while maintaining thermal isolation. Also the mechanical stress on the windows in the sidewall has to be reduced to prevent cracking of the glass. On the external ramps this was achieved by a very loose installation of the windows. This was again possible due to the low absolute pressure there and is not advisable for the isolator.

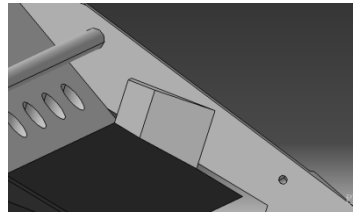
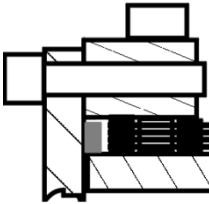


Fig. 4-7: Drawing of bar underneath support parts (left) and drawing of lip fixation and additional bar to prevent sideways opening of flow channel

The measures are shown in Fig. 4-7. Underneath the support parts 2 mm thin bars are attached to the sidewalls (Fig. 4-7 left). They prevent an upward movement of the lip and a downward movement of the ramp. These bars have to be placed exactly as they will define the flow channel height of the completed model. Therefore they are manufactured oversize for a later exact setting of the distance by the removal of some material. The front part of the lip is secured with clamps as shown in Fig. 4-7. To prevent the sideways opening of the flow channel an additional round bar is installed between the sidewalls (Fig. 4-7 right). An identical bar is used in the rear part of the isolator.

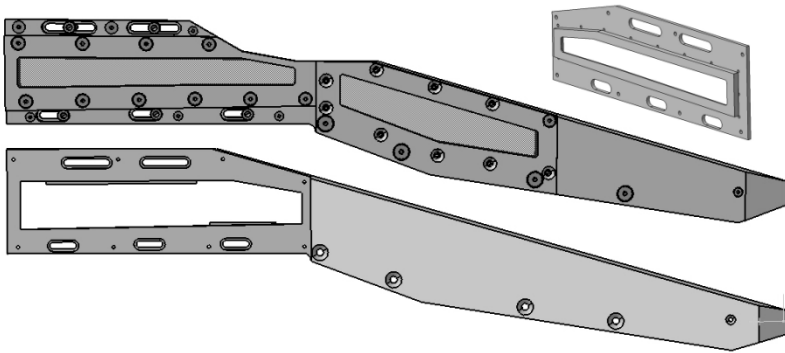


Fig. 4-8: Old (top) and new (bottom) sidewall and new window frame (upper right corner)

The original sidewalls were shaped with the lip leading edge and the following cover. This left no space for the needed reinforcements in the leading edge area of the lip. Therefore new sidewalls have been constructed (Fig. 4-8). They contain space for the lip fixation as well as the fixing bars described above. Also the windows are now installed in frames (Fig. 4-8 upper right corner). These frames are then installed in the sidewalls. For the sealing of the flow channel a sealing string is glued to the lip and creates clogging to the frame. This allows easy assembly and disassembly of the windows for cleaning purposes and decouples the windows from the sidewalls. The windows are fixed firmly in the frames using sealing strings. There is no direct contact between the metal and the glass anywhere in order to prevent cracking. The frames are then placed in the sidewall with a rather loose fit and are fixed with screws to prevent mechanical stress on the frames and therefore on the windows. The frames work as a cushion between the model and the windows. By default the new sidewalls do not contain the windows for the external compression ramps. The flow field there is currently not under investigation and placed outside of the optical axis of the tunnel. A later installment of windows there is possible at any time.

4.2.3 Generation of pressure rise

A movable double wedge is used to generate the back pressure for the investigation of the shock train. The principle assembly is shown in Fig. 4-9. The wedge has an angle of 20° on each side. The isolator with constant cross section is followed by adapter parts for the heating elements. They open the flow channel with 15° each. The smallest cross section is reached at the end of these parts. The back pressure is varied by a horizontal shift of the wedge.

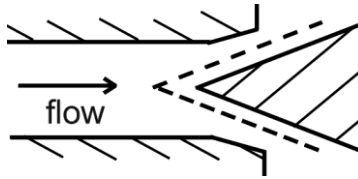


Fig. 4-9: Drawing of double wedge for back pressure generation

This allows the fast generation of back pressure. There are also issues with this setup that have to be mentioned here. The generated back pressure is not known before the experiment. The relation between generated pressure rise and wedge setting is determined empirically. The generated back pressure does also depend on the wall temperature in the isolator. The wall temperature influences the boundary layer on the wetted surfaces. For higher wall temperatures an increase in boundary layer displacement thickness δ_1 (Fig. 2-4) leads to higher blockage at the same horizontal position of the wedge. Therefore the relation between back pressure and wedge setting has to be experimentally determined for every tested wall temperature. This leads to a high number of experiments required for the whole study. Also the back pressure reacts very sensible to small changes of the wedge setting. Measures have to be taken to ensure an exact positioning ($\Delta x = 0.05$ mm) of the wedge. Still it might not be possible to generate identical back pressures at different wall temperatures. This issue has to be treated in the postprocessing of the experimental data.

A double wedge with an angle identical to the 15° opening angle of the adapter parts was also tested but no stationary flow conditions in the isolator were reached.

4.2.4 Instrumentation

The isolator of the Scramjet model has 48 positions for wall pressure on ramp and lip with a maximum spatial resolution of 5 mm. The double wedge is equipped with 8 water cooled Kulite pressure sensors as pitot probes creating a spatial resolution of 2 mm in the y-wise direction. All surface probes are placed on the middle axis of the ramps or the lip. The distribution of the wall pressure probes is shown in Fig. 4-10. Probes are clustered in the front part of the ramp to resolve the lip shock induced separation bubble and in the rear part to resolve the isolator flow field. There are no probes in the front part of the isolator ($x \leq 95$ mm) due to design restrictions as vertical space is required for the probe and its cable, the cooler and the sandwich construction. This space has to be created at a flat angle on top of the lip in order not to influence the isolator flow field.

There are 29 positions available for coaxial thermocouples on ramp and lip in a similar configuration compared with the pressure probe positions but there is a lower resolution

($\Delta x = 10$ mm) in the rear part. For flow visualization windows in the sidewall give optical access to almost the entire flow channel as it is shown in Fig. 4-10.

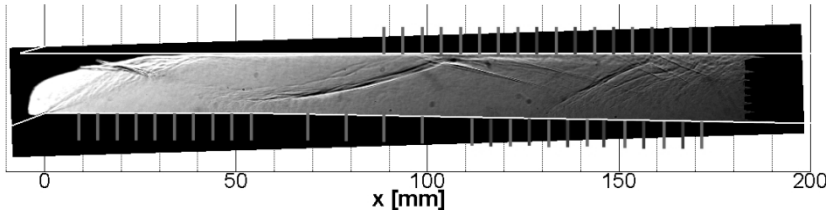


Fig. 4-10: Pressure probe positions in the isolator of the Scramjet model with shadowgraph image (Condition I, $T_w = 300$ K)

4.3 Heatable isolator model

In order to study the wall temperature effects on the shock train in a more universal flow case the heatable isolator model is used.

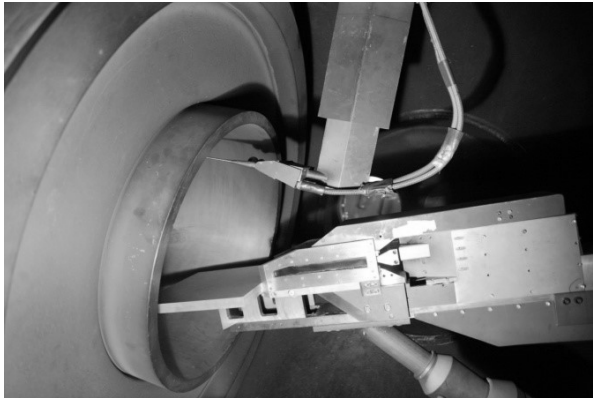


Fig. 4-11: Picture of the heatable isolator model in the TH2 test section

It is a derivation of the Scramjet intake model in order to use existing parts as well as to manufacture parts identical to those of the Scramjet model to save time and afford for manufacturing and construction. A picture of the model in the TH2 test section is shown in Fig. 4-11. The lip of the model is taken directly from the Scramjet model and an identical part with an x-wise offset of 2.5 mm for the probe positions is used as ramp. This creates a spatial resolution of 2.5 mm for the wall pressure measurements taking advantage of the symmetric flow field.

The back pressure is generated using the double wedge from the Scramjet model. It is equipped with 4 pitot probes in the center with a resolution of 2 mm to determine the isolator core exit Mach number Ma_3 . The cooler of the lip is reused and an identical one is used for the ramp. The pitot rake cooler is also recycled as well as the model windows.

4.3.1 Flow field

The basic flow field is drawn in Fig. 4-13. The incoming flow is decelerated with a single shock on a plate. The angle of attack of this plate and with it the parallel isolator is varied to create the desired isolator entrance Mach number Ma_2 . The relation between the angle of attack of the model and the isolator entrance Mach number is shown in Fig. 1-1. The boundary layer from the plate is disposed through a gap between the plate and the ramp of the isolator. The length of the plate is chosen to ensure the shock from the lip passing over the leading edge of the lip of the isolator. The ratio between length and width of the plate is set to 1 to fulfill the criteria by Lewis and Kubota for a two dimensional flow field on the plate [50]. The design parameters are shown in Tab. 4-1. Isolator length and width are identical to the Scramjet intake model. The height was set to 18 mm, the exit height of the Scramjet model. The isolator inlet Mach number reaches from 2 to 3.5.

The length of the plate is determined for the maximum isolator entrance Mach number flow case. This case has the lowest angle of attack and the lowest shock angle on the plate. A higher Mach number here would have resulted in a longer and therefore also wider plate. This would create higher mechanical loads on the plate. In addition the larger plate would catch up the boundary layer of the conical wind tunnel nozzle (chapter 3.1) as the leading edge would be placed further upstream (Fig. 4-11). It is not possible to move the whole model downstream as the isolator has to be placed within the optical axis of the tunnel to enable schlieren or shadowgraph visualization of the flow field. Also $Ma_2 = 3.5$ is a reasonable upper limit for the observation of shock trains as the shock train length is expected to decrease with $(Ma_2^2 - 1)$ (equation 2.61).

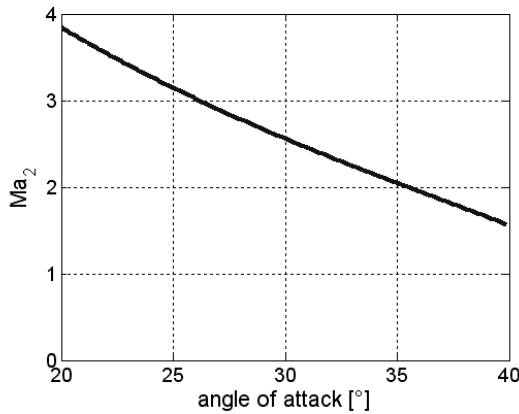


Fig. 4-12: Relation of isolator entrance Mach number Ma_2 and model angle of attack

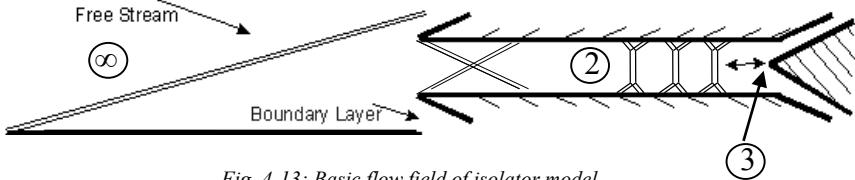


Fig. 4-13: Basic flow field of isolator model

The lower limit for the isolator entrance Mach number is set by the high angle of attack needed here. With the increasing angle the leading edge of the plate is approaching the lower nozzle wall and could also catch up the boundary layer here. Also low isolator entrance Mach numbers are not typical for Scramjet engines.

To get a first impression for the dimensions of the shock trains created with this model the correlation from equation 2.61 was used. The values for the boundary layer momentum thickness required here were gained with the reference temperature method (chapter 2.2). The results are shown in Fig. 4-14. The shock trains are shorter for higher isolator entrance Mach numbers. The flow channel is expected to block between a pressure rise $p_3/p_2 = 3.3$ at $Ma_2 = 2$ and $p_3/p_2 = 5.5$ at $Ma_2 = 3.5$. The model is instrumented downstream of $x = 0.0925$ m (x originating at the leading edge). This should be sufficient to capture shock trains over a wide range of pressure rises (2.5 at $Ma_2 = 2$ to 4.2 at $Ma_2 = 3.5$ in Fig. 4-14).

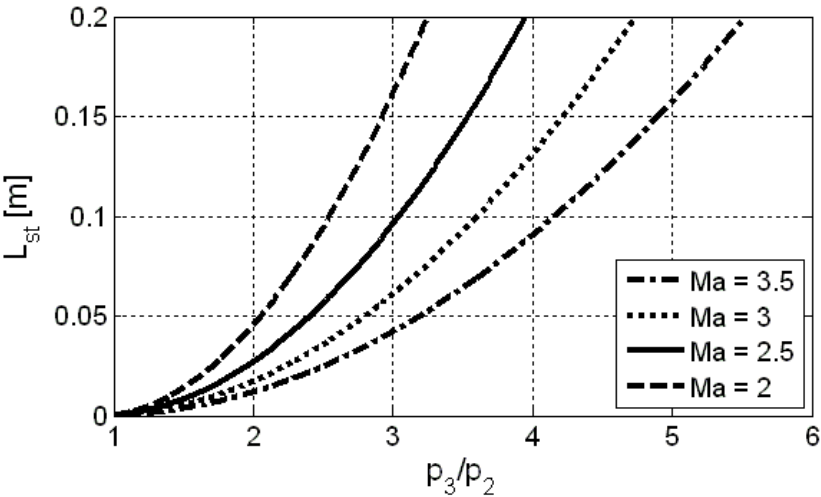


Fig. 4-14: Expected shock train length in isolator model as function for the overall pressure rise at different Mach numbers

Tab. 4-1: Design parameters for the heatable isolator model

chosen parameter		resulting parameter	
Ma ₂ [-]	2-3.5	angle of attack [°]	22-35
isolator width [m]	0.1	plate length [m]	0.3
isolator length [m]	0.2067	plate width [m]	0.3
isolator height [m]	0.018		

4.4 Cold isolator model

Both the Scramjet intake model as well as the heatable isolator model have a rectangular cross section with the same isolator width to height ratio (taken at isolator exit). To study possible width effects a model with variable width is used. A picture of the model is shown in Fig. 4-16. The basic flow field is identical to the heatable model as well as the isolator dimensions except the width. To ensure a two dimensional flow field the plate to reduce the Mach number Ma₂ was widened to 400 mm.

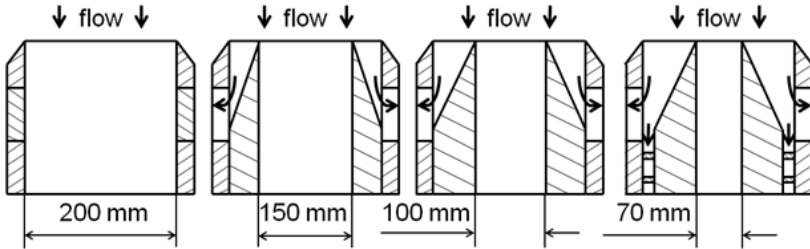


Fig. 4-15: Layout of cold isolator model with varying isolator width

The variation of the width is achieved with different inserts into the flow channel and closable orifices in the sidewalls, resulting in 70 mm to 200 mm channel width. The layout of the model with different inserts is shown in Fig. 4-15. At 200 mm width the whole flow channel is being used and the orifices in the side walls are closed. For the experiments at 150 mm width the orifices are opened and two inserts each 25 mm wide are installed in the flow channel. These are replaced by 50 mm wide inserts for 100 mm width. This is also the width of the heatable model. Results from the heatable model at 300 K wall temperature can therefore be used here as well. For 70 mm channel width 15 mm spacers are used to further separate the 50 mm inserts from the outer walls into the flow channel.

The construction of the model is very simple in order to save construction and manufacturing time. A cut through the model is shown in Fig. 4-17. The unwanted boundary layer from the inlet plate is guided past the isolator between the ramp and the base plate (lowest part in Fig. 4-17). The flow channel is only instrumented through the lip with a resolution of 5 mm on the center axis. The back pressure is generated using the same principle as the two heatable models.

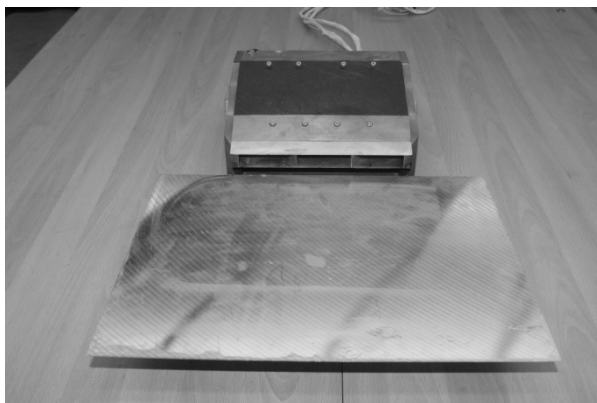


Fig. 4-16: Picture of cold isolator model

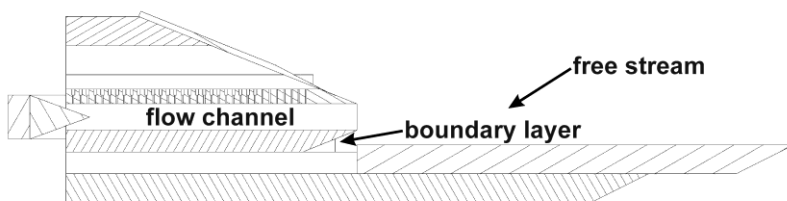


Fig. 4-17: Cut through cold isolator model

5 Results and discussion

5.1 Scramjet model

With this model the isolator flow field in a typical Scramjet engine is investigated. The flow passes the inlet with two compression ramps before entering the isolator with an asymmetric flow field. The flow field on the external compression ramps is shown by the numerical schlieren image in Fig. 4-5. An asymmetric flow field is entering the isolator.

5.1.1 Basic flow field

A shadowgraph picture, a numerically created schlieren picture as well as the pressure distribution on the isolator ramp and on the lip are shown in Fig. 5-1.

The expansion corner on the ramp is located at $x = 0$ mm. The numerical results are included for better interpretation of the flow field. The impact of the lip shock on the isolator ramp induces a separation bubble that fills about 50% of the flow channel. It is shown by the streamlines in the numerically created schlieren image. This phenomenon was also observed in [30]. The separation bubble fills the space upstream to the expansion corner. Both the experimental as well as the numerical picture show an expansion at the corner. At $x = 25$ mm, the lip shock hits the separation bubble. Directly downstream the wall pressure on the ramp starts to grow. This is not caused by the lip shock but by the beginning curvature of the shear layer upstream of the reattachment of the shear layer [20]. Reattachment takes place at $x = 55$ mm again inducing a small pressure jump. No back pressure was induced in this experiment. The numerical calculation (for details on the technique see chapter 3.3) fits the experimental results for the separation region, but gets less accurate for the reflection region of the shock on the ramp at $x \approx 140$ mm. The experimental shadowgraph image shows compression waves originating from the reattachment area which are also reflected at the lip. The pressure jump at $x = 140$ mm in the numerical solution on the ramp indicates this reflection to be a single shock, while the shadowgraph image as well as the pressure measurements show the reflection of the compression waves that are distributed over a larger area ($x = 150$ mm to $x = 170$ mm).

In the numerical schlieren image the x-wise density gradient is plotted with compression being white and expansion being black. It shows a shock (white) passing through the lip shock at $x = 20$ mm. This is the separation shock induced by the separation bubble downstream of the expansion corner. The influence of this shock on the flow field is minimal as the deflection of the stream lines caused by it is also minimal. In the shadowgraph picture from the experiments, the shock (black/white) also passes through the lip shock. Also visible are turbulent

fluctuations downstream the lip shock, caused by the interaction of the two shocks creating a shear layer.

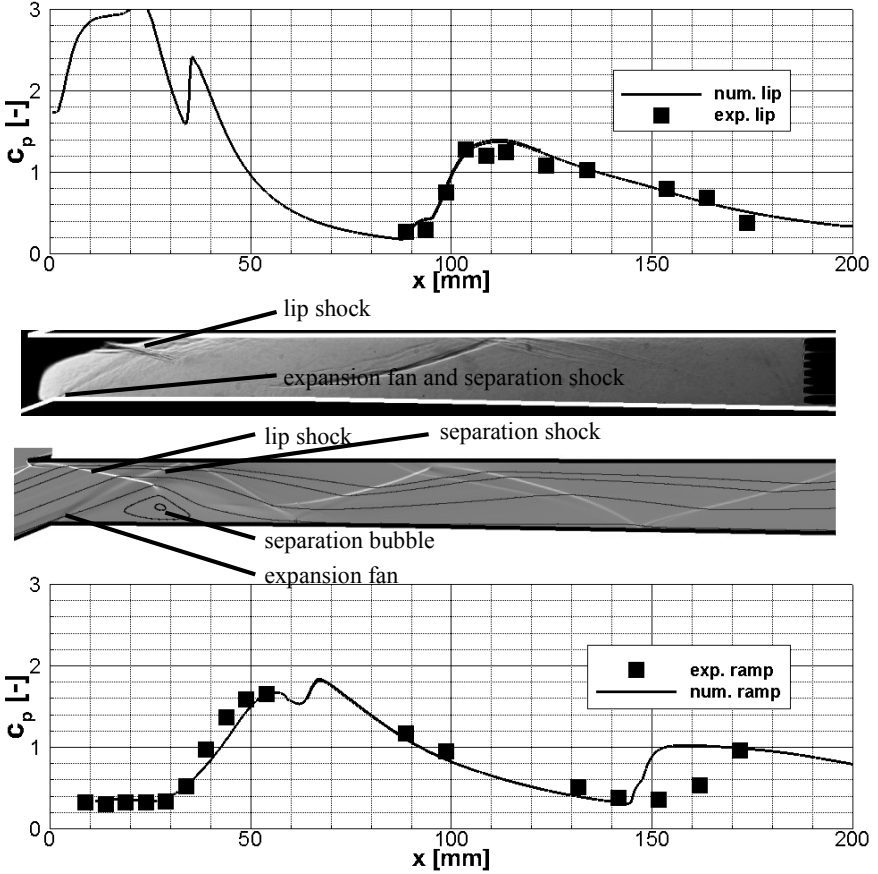


Fig. 5-1: Shadowgraph picture from experiment, schlieren picture with streamlines from numerical simulation and pressure distribution on ramp (bottom) and lip (top) for Condition I, $T_W = 300$ K

5.1.2 Lip shock induced separation bubble

The question to be answered by the experimental results is how the increase in wall temperature will influence the flow field. If the separation bubble grows upstream on the inlet ramps it can influence the flow field entering the isolator. If it fills the whole isolator cross

section the flow chokes. Experiments are performed at the shock tunnel conditions I and II and therefore it is also possible to investigate whether the wall to free stream temperature ratio is applicable as a similarity parameter here.

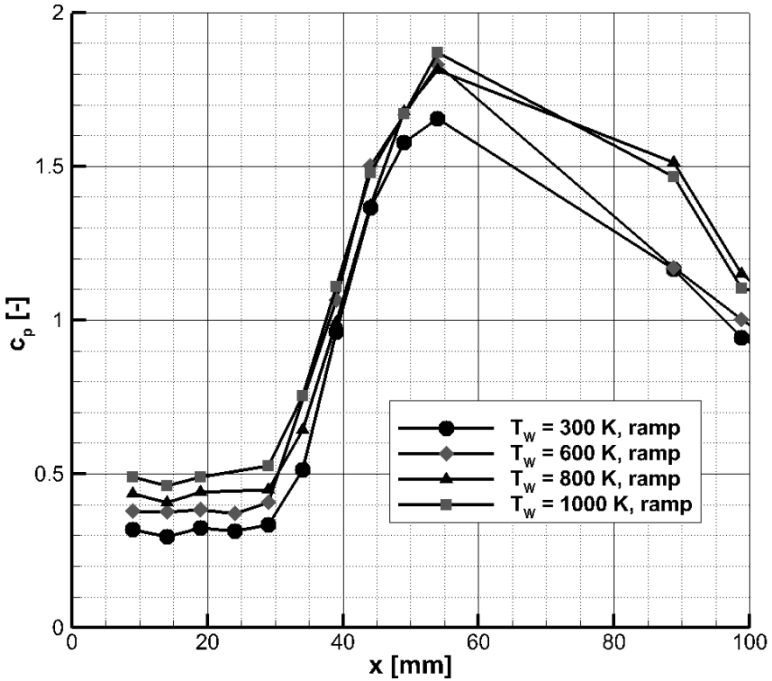


Fig. 5-2: Pressure distribution in the isolator at different wall temperatures for Condition I

The pressure distributions in the front part of the ramp (see Fig. 4-5) for different wall temperatures for experiments at Condition I are shown in Fig. 5-2. The pressure levels inside the first part of the lip shock induced separation bubble are rising steadily with increasing wall temperature from about $c_p = 0.3$ up to about $c_p = 0.5$. The pressure increase initiating reattachment shows no significant shift with increasing wall temperature. The higher pressure values between $x = 30$ mm and $x = 50$ mm are caused by the higher pressure upstream in the separation bubble. The origin of the increased pressure levels with increasing wall temperatures upstream of $x = 30$ mm will be discussed later.

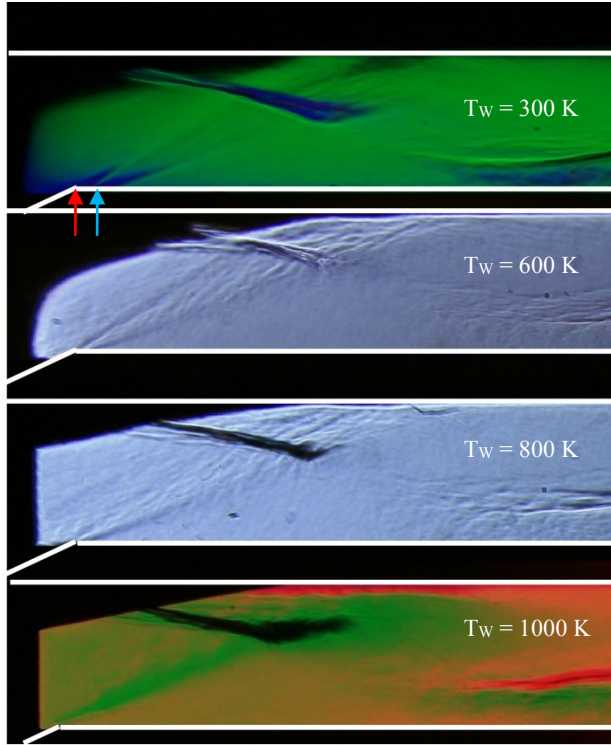


Fig. 5-3: Color schlieren images ($T_w = 300\text{ K}$ and $T_w = 1000\text{ K}$) and shadowgraph images ($T_w = 600\text{ K}$ and $T_w = 800\text{ K}$) of the area with the lip shock induced separation bubble (see Fig. 5-1) at Condition I

In Fig. 5-3 color schlieren and shadowgraph images of the lip shock induced separation bubble are shown for the four investigated wall temperatures. At $T_w = 300\text{ K}$ two flow phenomena are visible in the area of the expansion corner. They are first the expansion caused by the corner (red arrow) and then the separation shock of the separation bubble (blue arrow). It is also visible that the separation takes place slightly downstream of the expansion. The separation shock is moving upstream for $T_w = 600\text{ K}$ closer to the corner. Only a single flow phenomenon is shown for $T_w = 800\text{ K}$. The separation bubble has reached the corner. At $T_w = 1000\text{ K}$ only a single weak expansion occurs at the corner. This is also shown by the high pressure level in the separation bubble (Fig. 5-2).

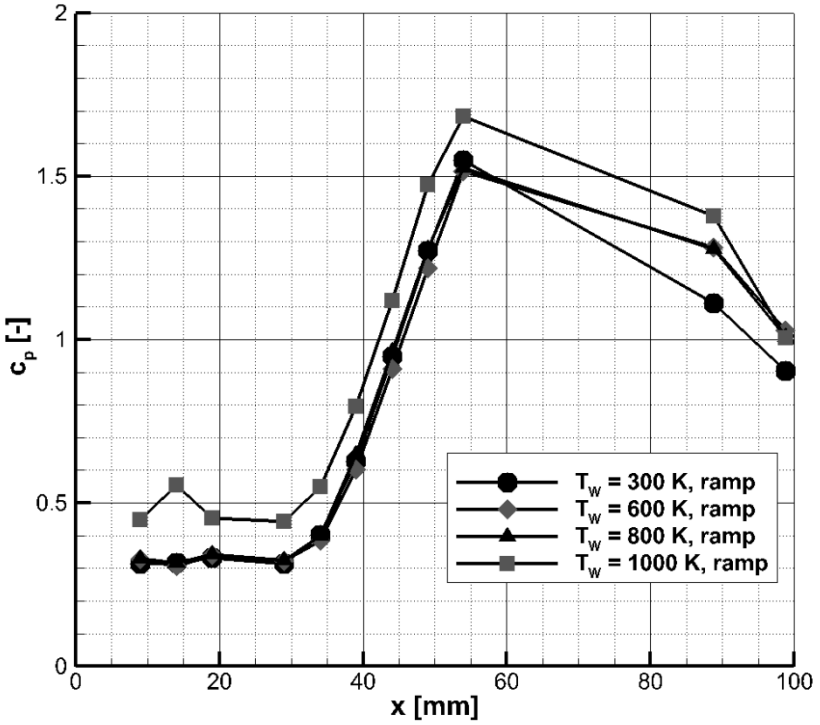


Fig. 5-4: Pressure distribution in the isolator at different wall temperatures for Condition II

For Condition II with the higher total temperature the pressure coefficient in the front part of the ramp as well as on the lip is plotted in Fig. 5-4. The distributions show no increase in the pressure levels in the lip shock induced separation bubble for the wall temperature from $T_w = 300$ K up to $T_w = 800$ K. A significant increase is reached then at $T_w = 1000$ K from about $c_p = 0.3$ for the colder wall temperature to about $c_p = 0.45$.

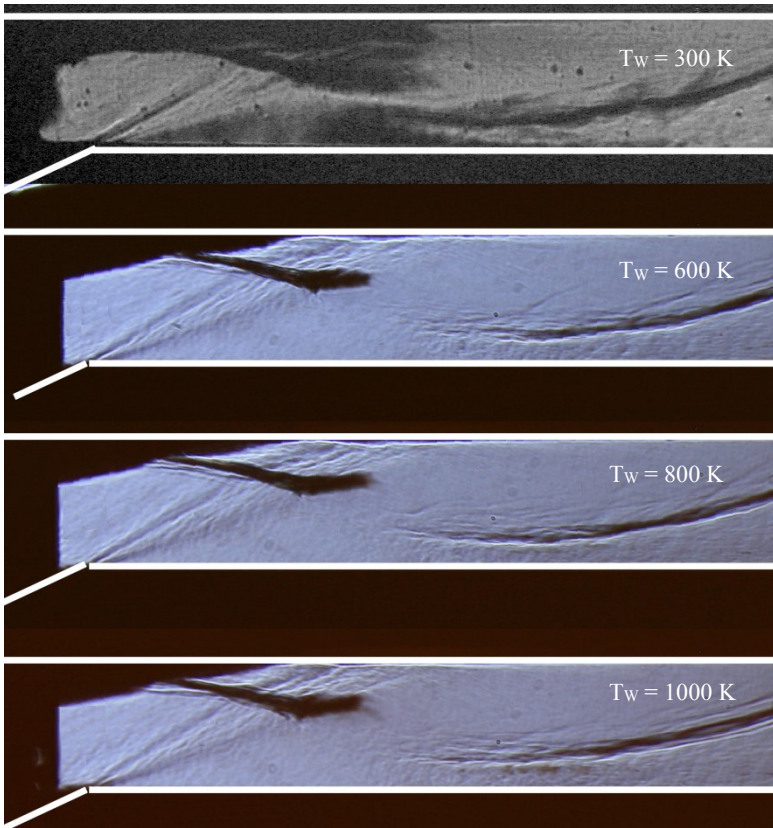


Fig. 5-5: Shadowgraph images of the area with the lip shock induced separation bubble (see Fig. 5-1) at Condition II at different wall temperatures

Shadowgraph images of the front section of the isolator for the different wall temperatures are shown in Fig. 5-5. All show the same basic flow field. At the corner the expansion as well as the separation shock slightly downstream are visible.

In Tab. 5-1 the ratios between wall and total temperature are listed for all investigated flow cases. This ratio is often referred to as similarity parameter for boundary layers and therefore a constant T_w/T_0 should result in identical flow fields if the incoming boundary layer is either laminar or turbulent, but the additional influence of the absolute values of T_w and T_0 was shown in [66]. In chapter 2.2 it was shown that the assumption of identical boundary layer

profiles for a constant T_w/T_0 is only true for constant exponents ω in the power law and therefore in a limited temperature range. This raises the question if T_w/T_0 is a useful similarity parameter here. Similar values of T_w/T_0 are reached for Condition I at $T_w = 300$ K and Condition II at $T_w = 600$ K as well as for $T_w = 600$ K at Condition I and $T_w = 1000$ K at Condition II. In the separation bubble at Condition I for $T_w = 300$ K and the corresponding case at Condition II the same pressure level of about $c_p = 0.3$ is reached. For the other case comparing $T_w = 600$ K at Condition I and $T_w = 1000$ K at Condition II the flow fields are not identical with a slightly higher c_p in the separation bubble for Condition II (0.45 compared to 0.4). It should be noted here that for the experiments at Condition I and II the Mach and Reynolds number are only similar and not identical (Tab. 3-2). Considering the sensitive reaction of the flow field to the wall temperature, especially at Condition I the wall temperature ratio is a good similarity parameter here.

Tab. 5-1: Ratio between wall and total temperature for different wall temperatures at conditions I and II

T_w [K]	T_w/T_0 at Condition I [-]	T_w/T_0 at Condition II [-]
300	0.26	0.14
600	0.52	0.27
800	0.70	0.36
1000	0.87	0.45

To further increase the understanding of the effects described above numerical simulations of the results described above are carried out. The pressure distributions on the ramp for different wall temperatures at Condition I are plotted in Fig. 5-6 and Fig. 5-7. For the simulations in Fig. 5-6 transition was modeled as described in chapter 3.3. The simulations in Fig. 5-7 were done fully turbulent preventing relaminarisation downstream of the expansion corner. For the lower wall temperatures ($T_w \leq 500$ K in Fig. 5-6 and $T_w \leq 800$ K in Fig. 5-7) the separation takes place downstream of the expansion corner with increasing pressure levels inside the separation bubble. For the higher wall temperatures the separation bubble slightly has grown upstream beyond the corner. This is also shown by the schlieren images in Fig. 5-8 which were created for the transitional cases.

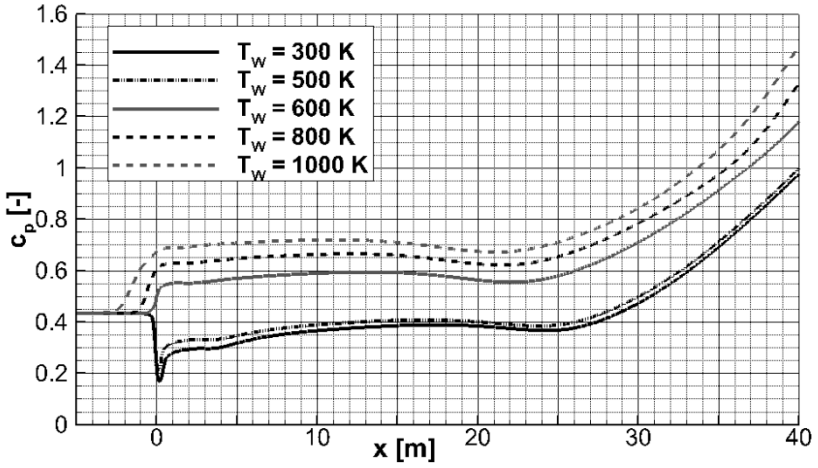


Fig. 5-6: Wall pressure distribution in the front part of the isolator for different wall temperatures at Condition I from numerical simulations with transition modeling

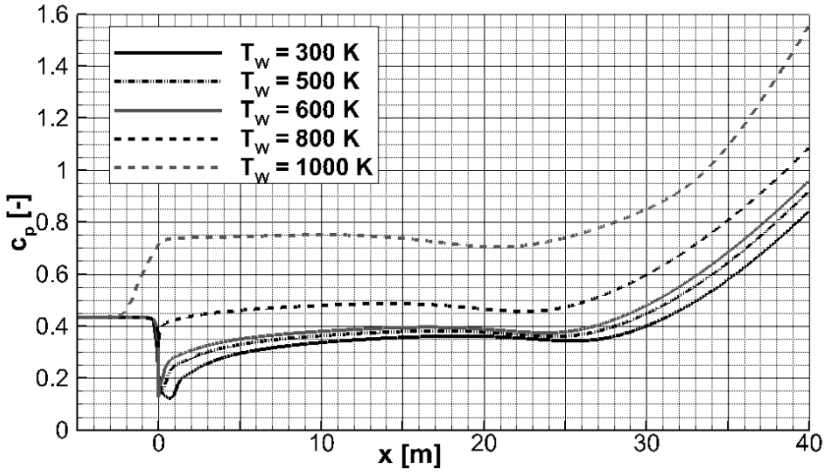


Fig. 5-7: Wall pressure distribution in the front part of the isolator for different wall temperatures at Condition I from numerical simulations with forced transition

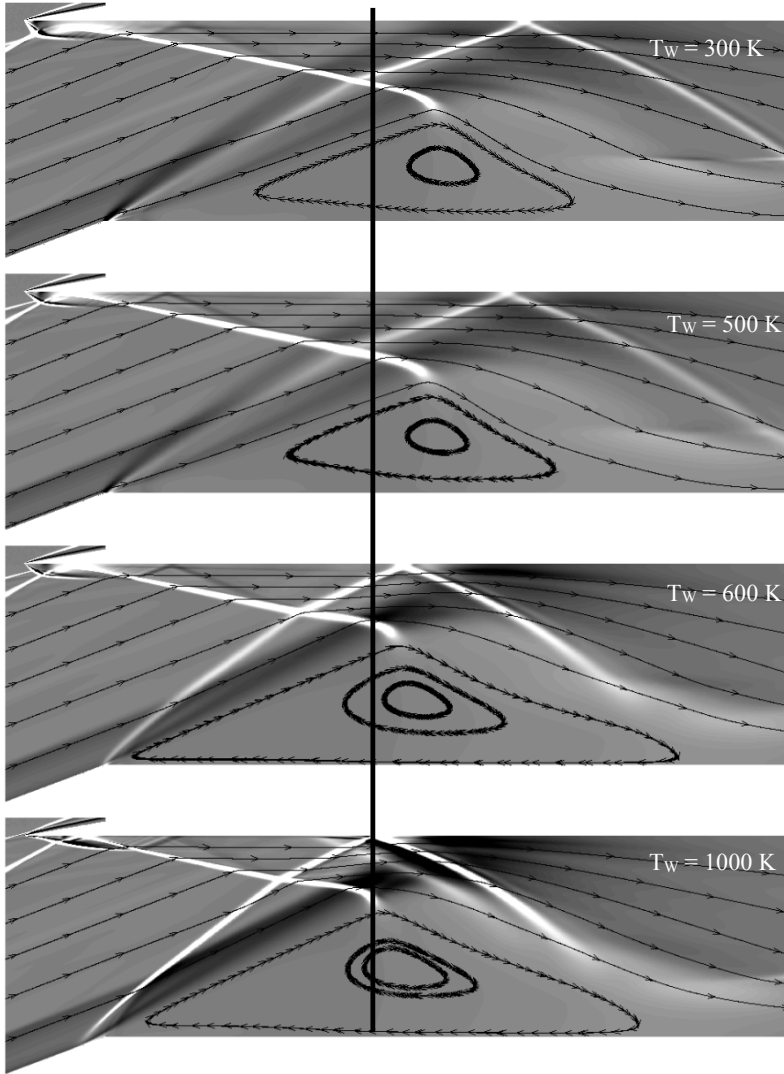


Fig. 5-8: Schlieren images with stream lines created from numerical simulations for different wall temperatures at Condition I

Schlieren images for four different wall temperatures created for the transitional cases for the relevant area of the isolator are shown in Fig. 5-8. They show the density gradient in

x-wise direction which is the main flow direction. Shocks are plotted white and expansions are plotted black.

For the two lower wall temperatures one can also see the separation bubble starting downstream of the expansion corner. The expansion at the corner (black) is followed by the separation shock (white). A slight upstream movement of the separation point is indicated by the weaker expansion originating at the corner caused by the upstream movement of the separation shock into the expansion.

For the two higher wall temperatures the separation bubble has grown upstream beyond the expansion corner and continues to grow with further increasing wall temperature. The new position of the separation shock upstream of the expansion corner leads to a significant change in the separation bubble geometry. The top of the separation bubble is the impingement point of the lip shock. The separation shock upstream of the expansion corner leads to an increased flow angle relative to the isolator main flow direction and therefore to an upstream movement of this impingement shock (black vertical line in Fig. 5-8). This leads to a vertical growth of the separation bubble and with it a higher blockage of the isolator flow channel. An upstream growth of the separation bubble could potentially block the isolator.

The effects are qualitatively identical for the fully turbulent calculations. For the shock tunnel Condition I the fully turbulent calculations (Fig. 5-7) match the experiments better than the transitional results showing the importance of a potential relaminarisation of the boundary layer downstream of the expansion corner for the flow field. Still there is a slight offset of the wall temperature effect as the separation bubble pressure level of the experiment at $T_w = 1000$ K match the CFD simulation at $T_w = 800$ K. This can be explained by the fact that the simulation is two-dimensional in opposition to the real 3D experiment. Altogether the simulations show the separation bubble growing into the expansion wave at the corner with increasing wall temperature. The raise in pressure level inside the bubble is caused by the partly cancelation of the expansion as the separation point closes in on the corner. The upstream movement of the separation point with increasing wall temperature is expected due to the increase in boundary layer displacement thickness (chapter 2.3). At Condition II the separation takes place a little more downstream therefore the separation point does not grow upstream into the expansion at the corner for the lower wall temperatures up to $T_w = 800$ K here (Fig. 5-4).

5.1.3 Shock train

In Fig. 5-9 the wall pressure distributions on ramp and lip for different back pressures created by different wedge settings are shown together with the corresponding shadowgraph images. All shown experiments have been done at Condition I with a wall temperature of

$T_w = 300$ K. The back pressure ratio was calculated by dividing the mean value of the last pressure probes on ramp and lip by the mean value of the same quantities from the experiment without back pressure. In Fig. 5-10 the Mach number profiles measured with the pitot probes installed on the movable wedge are shown. The Mach number profiles were calculated from the measured pitot pressures and the last static pressure measurement along the symmetry line on ramp and lip. The lowest and upmost pitot tubes are not located on the centerline and do therefore not interfere with the static pressure measurements in that area. Black symbols illustrate the flow field without back pressure for comparison. The Mach number profile for the case without back pressure shows the flow leaving the isolator supersonic with a Mach number of about 3 in the upper part of the isolator and about 2.5 in the lower part of the isolator. In the shadowgraph image for this case (Fig. 5-9) shocks are visible in front of all pitot probes also indicating supersonic flow. At a pressure ratio of 2.6 a shock train starts to build up at the most rear part of lip. This is shown in Fig. 5-11 by the pressure waves originating from the surface. Nothing comparable is visible on the ramp. The pressure distributions show a slight pressure increase on the lip as well as on the ramp. Therefore, the pressure increase does not reach most of the pitot probes, because the shock train starts just in front of the pitot rake. For this reason the static pressure at the probes is not identical with the values from the last static pressure probes on ramp and lip. Therefore a Mach number profile could not be determined.

At a back pressure level of 3.5 a shock train is visible on the wall pressure distribution as well as in the shadowgraph picture (Fig. 5-9). According to the shadowgraph image in Fig. 5-9 the boundary layer separates for the second time on the ramp at about $x = 125$ mm. The separation shock impinges on the lip creating a separation bubble with a separation shock at $x = 145$ mm visible in the shadowgraph picture as well as in the wall pressure distribution on the lip (Fig. 5-9). The shadowgraph shows a huge subsonic region on the ramp filling about 50 % of the flow channel. This is also visible in the Mach profile which shows the core flow at $y = 14$ mm with a Mach number of about 2.5 and extremely low Mach numbers in the lower part of the flow channel ($p_{\text{static}} \approx p_{\text{pitot}}$). The sharp cut off between the supersonic core and the extreme low energy flow of the separated region is typical for shock trains with supersonic exit flow (oblique shock trains).

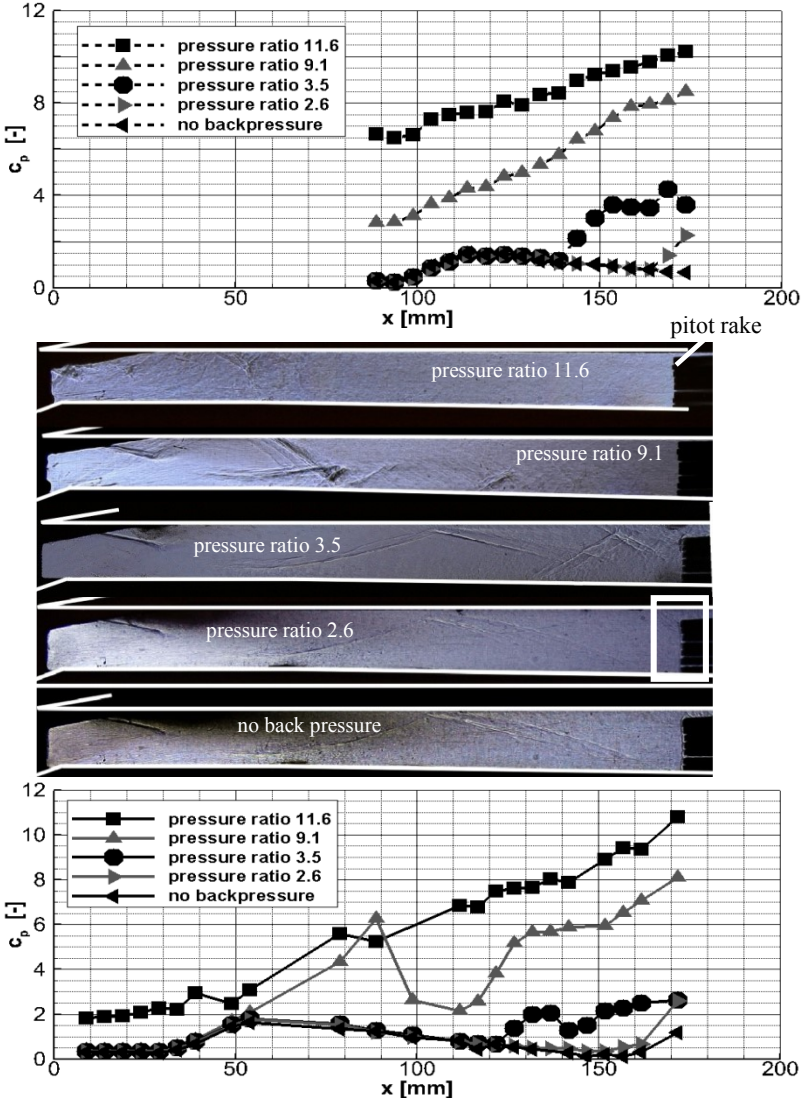


Fig. 5-9: Pressure distributions on ramp (bottom) and lip (top) for different pressure ratios, Condition I, $T_w = 300\text{ K}$, shadowgraph images in the middle

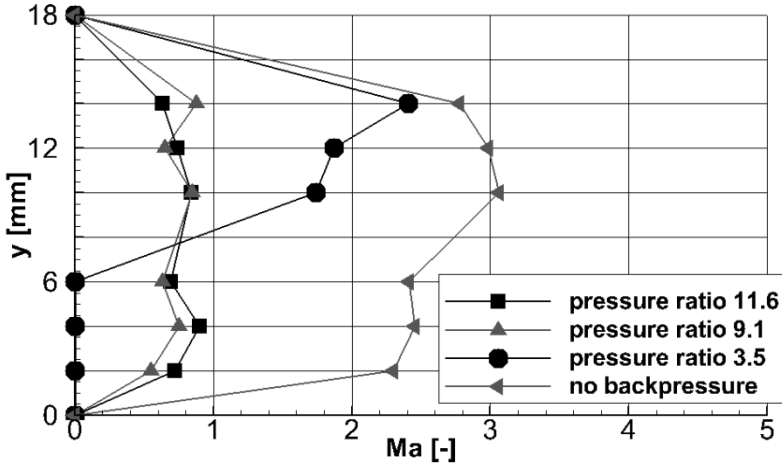


Fig. 5-10: Mach number profiles at isolator exit (pitot rake position) for different back pressures at Condition I, $T_w = 300$ K

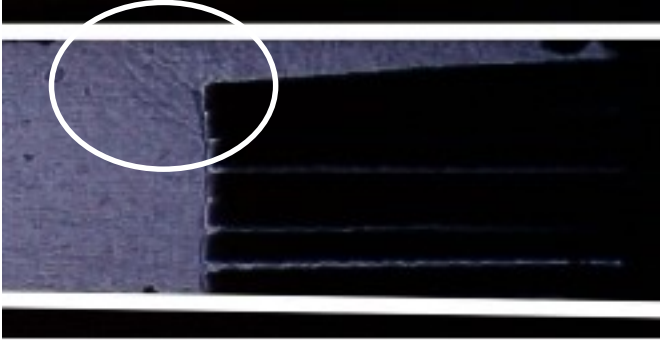


Fig. 5-11: Detail of rear isolator section for pressure ratio 2.6 (frame in Fig. 5-9)

At the back pressure level of 9.1 the shock train is moving further upstream into the isolator. The first separation occurs on the lip at about $x = 60$ mm (visible in shadowgraph picture only). The separation shock then hits the ramp leading to a small separation bubble there. This is shown in the shadowgraph image by the vertical offset of the impinging shock and its reflection from the ramp and a reattachment shock that is visible about 10 mm downstream. Then a separation shock originates at about $x = 120$ mm on the ramp (visible in shadowgraph image). The expansion after the reattachment at $x = 90$ mm as well as the pressure rise from the final separation is also visible in the pressure distribution on the ramp. The Mach profiles

show subsonic flow with uniform Mach number of about 0.7 at the exit plane. The even distribution is typical for shock trains with subsonic exit flow (normal shock trains). There are also no shocks visible in front of the pitot probes. This indicates subsonic flow as well. The asymmetric core flow is supersonic in the upstream part and subsonic in the downstream part of the isolator.

At a pressure ratio of 11.6 the flow structure breaks down completely in the isolator. There are some shocks visible in the shadowgraph image in the front part of the lip indicating supersonic flow there. The lip shock is pushed out of the isolator. Pressure levels are rising also in the very front part of the ramp. The isolator is blocked. The Mach numbers at the exit plane are subsonic. Over the shown range of back pressures the shock train grows from supersonic to subsonic exit flow before it blocks the inlet. For the considered cases, the shock train with supersonic exit flow (typical for scramjet operation) is caught easily within the 200 mm long isolator. For the flow case with partially supersonic isolator exit flow (pressure ratio = 3.5) the flow field is very asymmetric. This is most likely caused by the asymmetry of the isolator flow field.

To rule out an asymmetric back pressure generation a numerical simulation of the setup is conducted. In Fig. 5-12 a comparison between experimental and numerical results is shown. The results are for Condition I and a wall temperature of $T_w = 600$ K. In the numerical simulations a constant back pressure across the whole exit plane is used yielding to an overall pressure rise of $p_3/p_1 = 200$ for the whole inlet. The experimental results were chosen to fit the numerical results at the last pressure probes in the isolator. Numerical and experimental pressure distributions without any pressure rise are also plotted for comparison.

The experiment with back pressure shows a shock train with supersonic exit flow (Fig. 5-14). The shock train induced separation originates on the ramp at $x = 110$ mm shown by the pressure deviating from the flow case without backpressure (Fig. 5-12) and the separation shock in the schlieren image (Fig. 5-12). The separation shock impacts on the lip at $x = 130$ mm. It induces a separation there reaching upstream to about $x \sim 125$ mm. The lower rear half of the isolator is filled by a huge throughout separation bubble (Fig. 5-13). A supersonic core flow with a Mach number of about 1.8 is shown by the Mach profiles in Fig. 5-14. The continuously growing pressure in the experimental data on the ramp also indicates a throughout separation region on the rear part of the ramp.

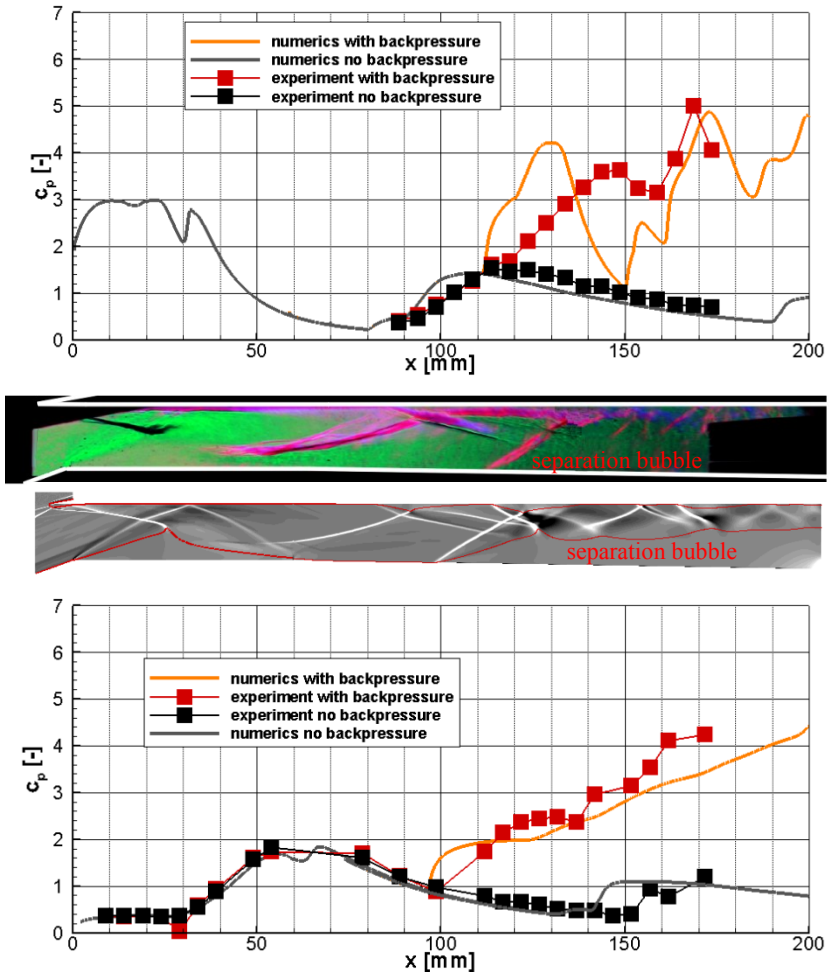


Fig. 5-12: Comparison between numerical and experimental results: pressure distribution on the lip (top), color schlieren image from experiment and schlieren image from numerical result (middle) and pressure distribution on ramp the (bottom), Condition I, $T_w = 600$ K

The schlieren image of the flow on the rear part of the lip as well as the corresponding pressure distribution with its more or less continuous pressure rise may indicate a throughout separation on the lip downstream to $x = 170$ mm which is limited to the region very close to the wall. The numerical schlieren image was created for the x -wise density gradient. It shows

shocks white and expansions black. The red line is an iso-line for Mach 1. The rear part of the isolator with numerically generated schlieren image is magnified in Fig. 5-13. The numerical results for the case with back pressure show the first separation of the shock train at $x = 95$ mm about 15 mm upstream of the experimental results on the ramp as well as for the impinging point on the lip. Despite the offset of the separation point the flow field is very similar on the ramp, visible by the continuous pressure rise. On the lip there is no long separation region, just two small separation bubbles at $x = 115$ mm and $x = 165$ mm.

The separation shocks cause the pressure peaks at $x = 125$ mm and $x = 168$ mm on the lip. Unlike the experiment, the numerical solution for the rear part of the lip shows a subsequent separating and reattaching flow (Fig. 5-13). The discrepancy between numerical and experimental results is not surprising as small differences between experiment and calculation sum up over the inlet with external compression and the isolator with the lip shock induced separation bubble. Other possible reasons for the discrepancy between numerical solution and the experiment are the influence of the turbulence model, the 3D features of the experimental flow field and differences in the free stream conditions that have been slightly modified for the numerical simulations.

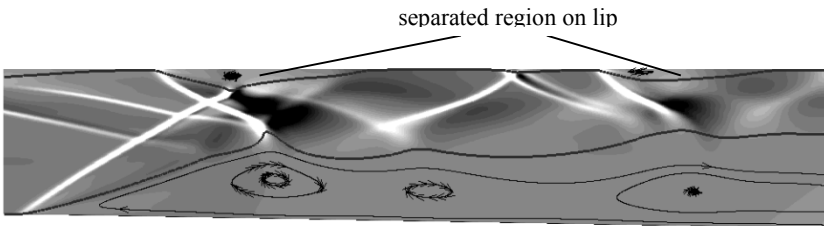


Fig. 5-13: Schlieren image from numerical result with back pressure, detail of rear isolator with stream lines, Condition I, $T_w = 600$ K

The Mach number profiles (Fig. 5-14) have an offset in the case without back pressure of about 0.5. The Mach numbers for the back pressure case match and show the basic flow features with a supersonic core and a huge separation region on the ramp.

The numerical results show the same flow case as the experiments, with an asymmetric shock train with supersonic core flow originating from the ramp. The shock train length has an offset of about 15 % visible in Fig. 5-12. Still the flow case is identical showing that the asymmetry of the isolator flow field with shock train is caused by the asymmetry of the isolator inflow and not an inhomogeneous back pressure generation.

In the numerical simulation the back pressure is imposed by a subsonic opening at the isolator exit.

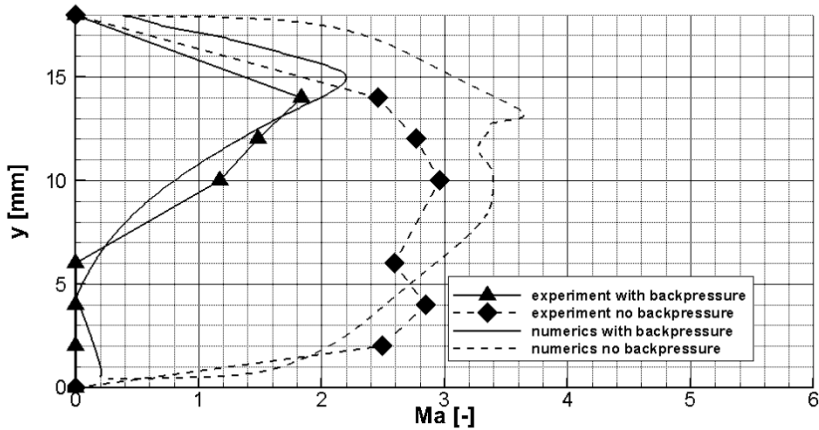


Fig. 5-14: Comparison between numerical and experimental results: Mach number profiles at isolator exit, Condition I, $T_w = 600$ K

Experiments with different wall temperatures at Condition I are compared in Fig. 5-15. They all reach about the same pressure at the last pressure probe on ramp and lip respectively and therefore the same back pressure. They reach a pressure ratio of about 9.1 creating a shock train with subsonic exit flow. The separation on the lip is visible as a shock originating at about $x = 50$ mm. At 300 K wall temperature this shock hits the ramp at $x = 75$ mm starting the shock train there. The flow reattaches at the ramp as indicated by the reattachment shock ($x = 80$ mm) as well as the final visible separation shock ($x = 115$ mm) originating directly from the ramp. For $T_w = 300$ K the flow field upstream of about $x = 50$ mm is not influenced by the shock train as the pressure values on the ramp for this case show. In the shadowgraph image there are still two flow phenomena visible at the expansion corner at $x = 0$ mm. For $T_w = 300$ K there is first an expansion and then slightly downstream the separation shock.

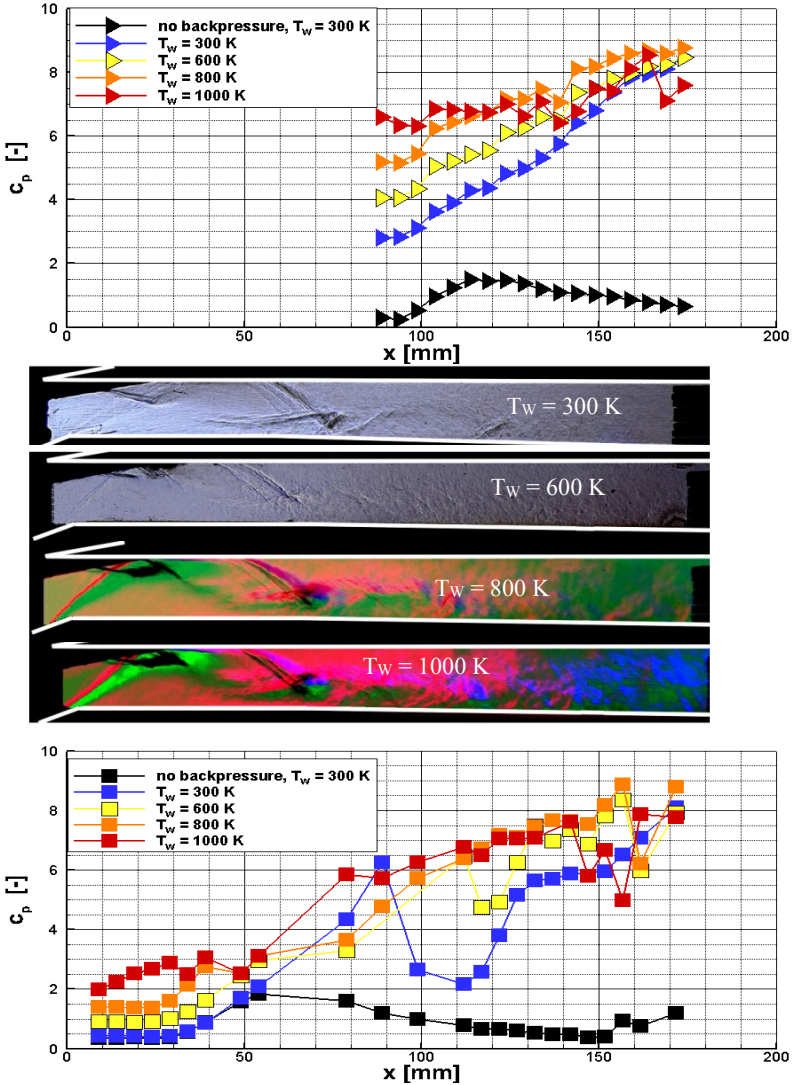


Fig. 5-15 : Wall temperature influence: pressure distribution on the ramp (bottom), schlieren and shadowgraph images from these experiments (middle) and pressure distribution on the lip (top), Condition I

With increasing wall temperature the separation shock from the lip moves slightly upstream. The impingement point on the ramp is therefore also moving upstream. For the three higher wall temperatures in the shadowgraph/schlieren images there is no expansion visible but a separation shock upstream of the corner. The separation point has moved upstream of the corner. For higher wall temperatures the shock train is interacting with the lip shock induced separation bubble. This is shown by comparing the schlieren images in Fig. 5-15 with those in Fig. 5-3. There without the shock train the separation bubble does not grow beyond the corner for the higher wall temperatures in opposition to Fig. 5-15. Also the pressure level in the lip shock induced separation bubble (ramp, $x < 50$ mm) grows with higher wall temperatures. This behavior is caused by the shock train as it has not been observed in this strength for the flow cases without back pressure (see Fig. 5-2).

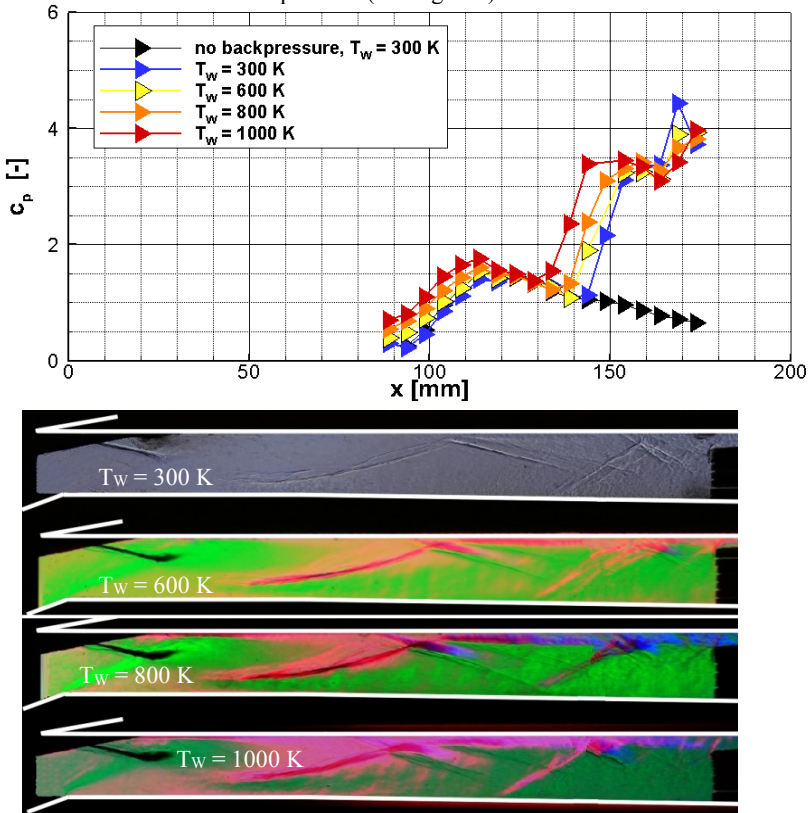


Fig. 5-16: Wall temperature influence: pressure distribution on the lip (top) with schlieren and shadowgraph images, Condition I

The pressure distributions on the lip at around $x = 100$ mm show a more or less linear pressure increase for all wall temperatures. The higher pressure levels are caused by the fact that the shock is now standing upstream of the expansion corner.

The wall temperature effect on a shock train with supersonic exit flow is shown by the pressure distributions on the lip in Fig. 5-16. The shock train starts on the ramp as shown by the shadowgraph and schlieren images. The pressure distributions on the lip show the impact of the first separation shock on the lip. The shock train is growing upstream constantly with increasing wall temperature from about $x = 145$ mm at $T_w = 300$ K to about $x = 135$ mm at $T_w = 1000$ K. The pressure values reached after the initial separation shock are identical in the four experiments. The separation point is clearly moving upstream with increasing wall temperature. Hence the shock train length is increasing with growing wall temperature with an identical flow topology.

For the shock trains with supersonic as well as with subsonic exit flow the shock train length is increasing with increasing wall temperatures. The influence of the total temperature in the shock train is investigated with the shock tunnel Condition II. The pressure distribution on the lip for different cases with constant back pressure is shown in Fig. 5-17. Plotted are three wall pressure distributions on the lip for Condition I with different wall temperatures between $T_w = 600$ K and $T_w = 800$ K. For these three cases the shock train is again growing with increasing wall temperature. For this condition, the experiment at $T_w = 1000$ K has developed a flow structure very similar to the $T_w = 1000$ K case in Fig. 5-16 which has a higher pressure ratio (4 vs. 3.4). Still the difference is that here the shock train on the lip is not started by the impact of the separation shock from the ramp. This happens at about $x = 165$ mm but the shock train is already starting at about $x = 140$ mm on the ramp. So here the shock train is started on both the ramp and the lip caused by the different boundary layer profile for the high wall temperature. This explains the high jump in shock train length on the lip. Also plotted is a pressure distribution obtained at Condition II with a wall temperature of $T_w = 1000$ K. With the increase in total temperature the shock train length is decreasing significantly, as shown by the green symbols. For the complex flow case in the Scramjet isolator here the change in total temperature between Condition I and Condition II with the same wall temperature $T_w = 1000$ K leads to different flow topologies. While the shock train is longer on the lip for Condition I as shown by the pressure distribution and the schlieren images the images show a longer shock train on the ramp for Condition II.

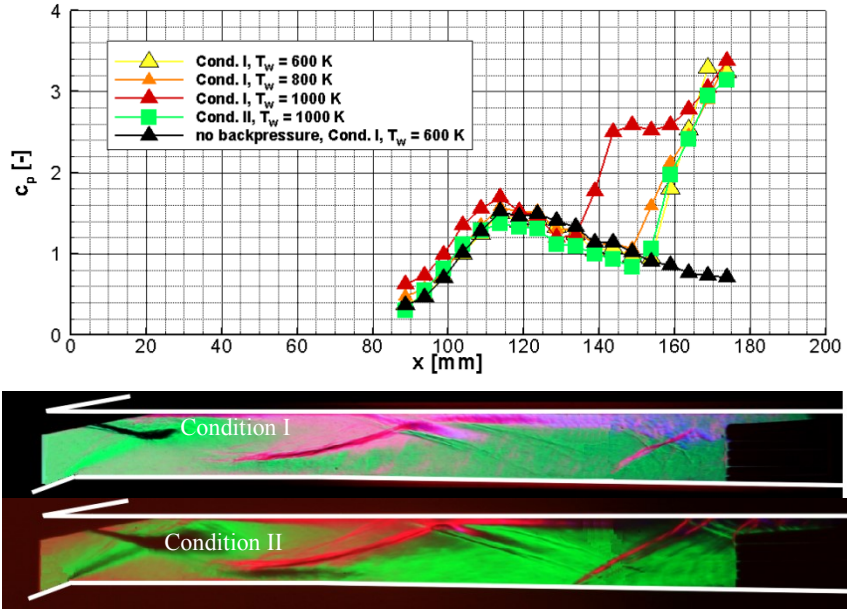


Fig. 5-17: Wall and total temperature influence: pressure distribution on the lip (top) with color schlieren images for the cases at $T_w = 1000$ K (bottom)

The experiments at Condition I, $T_w = 600$ K and Condition II, $T_w = 1000$ K have a similar ratio between total and wall temperature (Tab. 5-1). The measured shock train length for both cases is identical which may be indicating a similarity for the two flow cases.

With T_0/T_w as similarity parameter a longer shock train is expected for $T_w = 800$ K at Condition I compared to the two flow cases with similar T_0/T_w . This is confirmed by the pressure distributions in Fig. 5-17.

In Fig. 5-18 different flow cases also varying in wall and total temperature are compared with the pressure distributions on the ramp and shadowgraph images. The highest pressure rise with the shortest shock train is achieved at Condition I with $T_w = 300$ K. The whole pressure rise is achieved within the range of the last pressure probe and no shock train structure is visible in the shadowgraph image. The two other cases at Condition I with $T_w = 600$ K wall temperature and at Condition II with $T_w = 1000$ K wall temperature show a lower pressure rise but a significant shock train has developed with a huge subsonic region on the ramp also shown in the shadowgraph images. Both flow cases have a similar wall to total temperature ratio (Tab. 5-1). The pressure rise for the Condition I experiment is slightly higher as is

the shock train length. Still the pressure distribution for the two cases seems more or less identical and one can be shifted over the other. This also indicates that (T_w/T_0) is a good similarity parameter here.

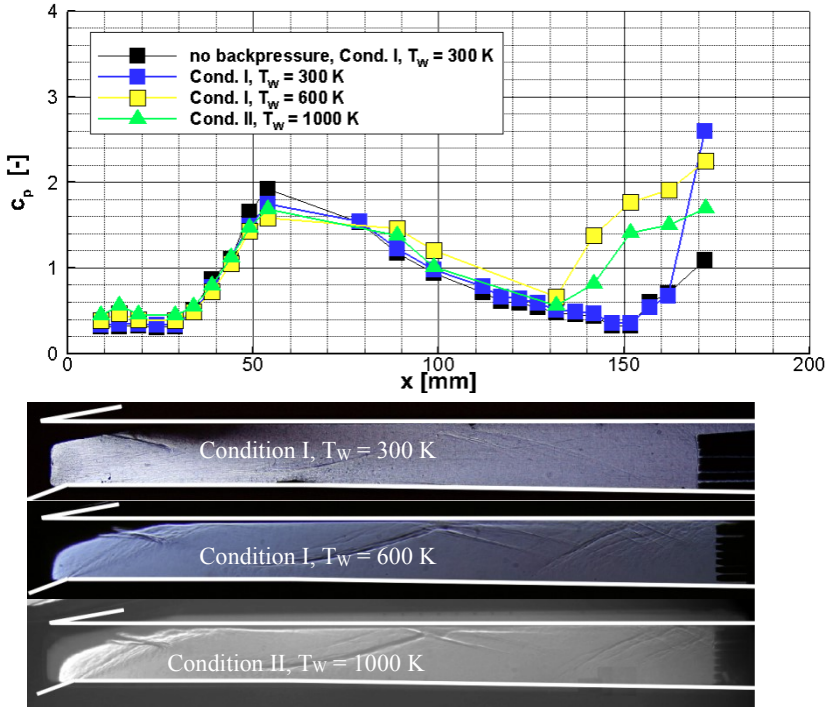


Fig. 5-18: Wall and total temperature influence: pressure distribution on the ramp (top) with shadowgraph images (bottom)

At all investigated back pressure levels there is a clear trend of increasing shock train length with increasing wall temperature. The flow topology of the shock train is complex due to the asymmetric inflow into the isolator. Therefore for a clear quantification of the wall temperature effect on the shock train is not possible here. To achieve this, a model with a symmetrical inflow is investigated in chapter 5.2.

5.1.4 Boundary layer suction

The potential of a boundary layer suction installed in the ramp of the isolator is investigated numerically. The goal of the suction is to eliminate the lip shock induced separation bubble on the ramp in order to prevent a possible blockage of the isolator. Six different potential geometries are tested. The geometries are shown in Fig. 5-19 and Fig. 5-20. For the exit of the slit a subsonic opening with p_∞ was used.

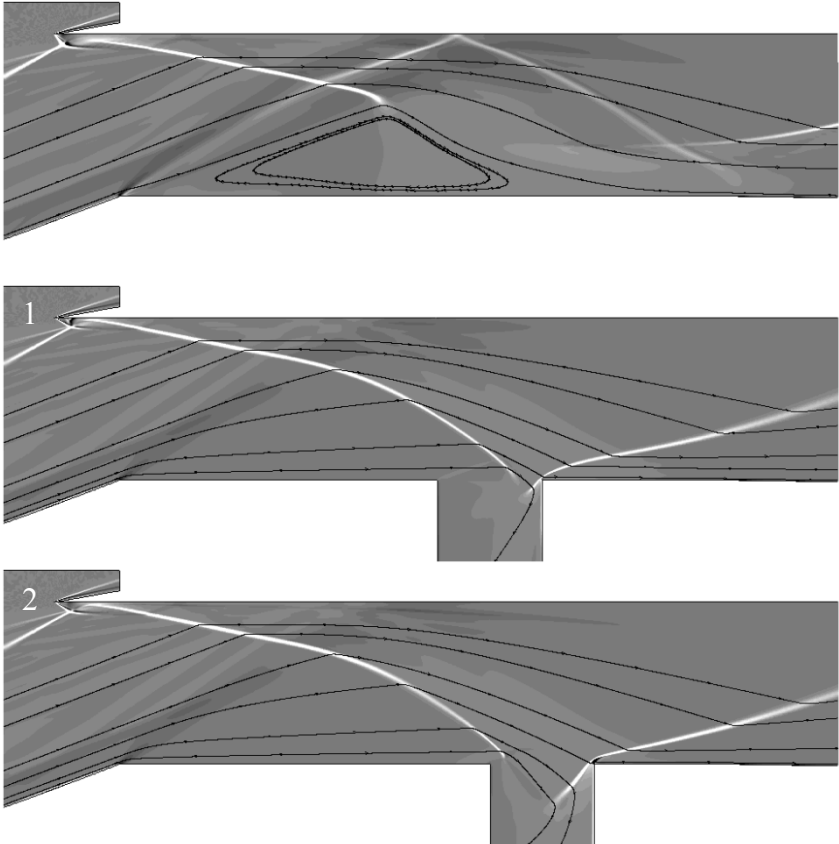


Fig. 5-19: Schlieren images created from numerical results for front part of the isolator without (top) and with different boundary layer suction geometries

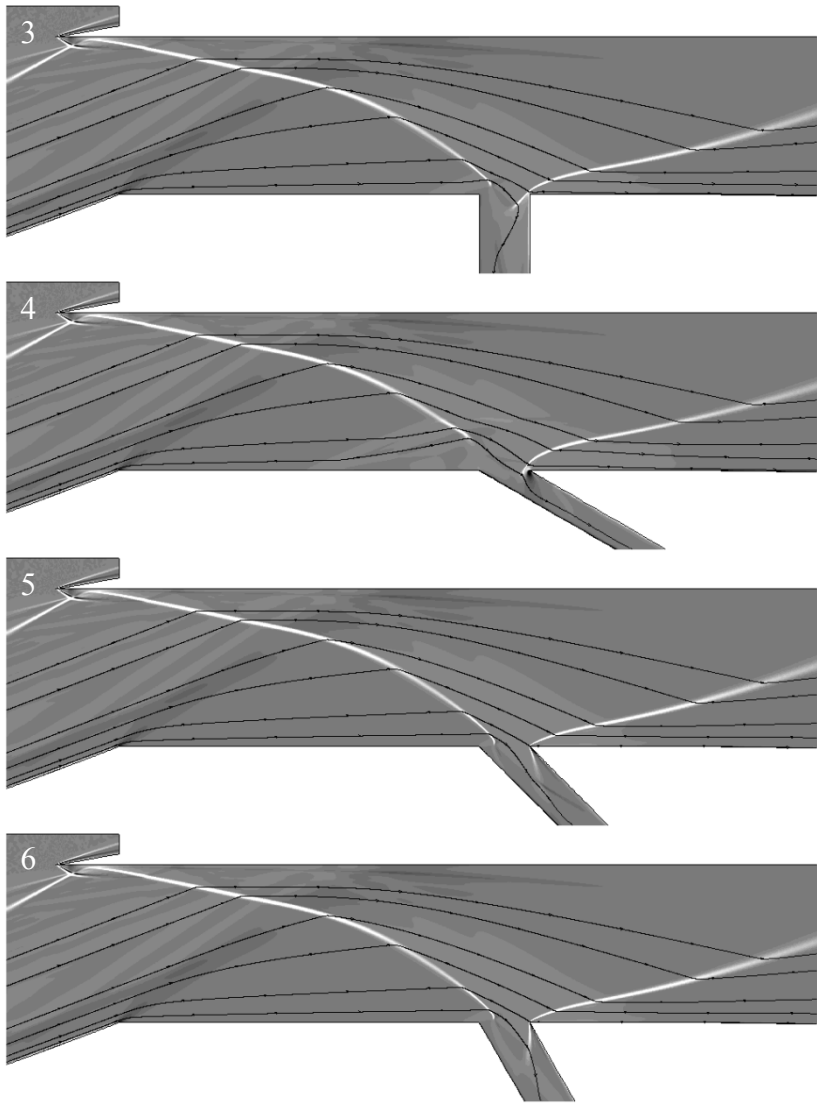


Fig. 5-20: Schlieren images created from numerical results for front part of the isolator with different boundary layer suction geometries (continued)

Shown are six different geometries together with the flow field without suction for comparison. Between the different geometries the location, the width and the angle of the slot was varied. The flow fields are illustrated by schlieren images showing the x-wise density gradient with compression being white and expansion black. Also shown are stream lines. All calculations are done at Condition I with a wall temperature of $T_w = 300$ K. The effect of the suction on the flow field is shown by comparing the slit geometry 1 with the flow field without suction. The elimination of the separation bubble is shown by the lip shock now reaching down to the slot as well as the stream lines passing through the area occupied by the separation bubble for the case without the slot. The shock originating at the end of the slot is caused by the alignment of the flow directly after the lip shock to the main flow direction. The separation bubble is eliminated also for all other geometries expect geometry 4. Still the size of the separation bubble is reduced drastically. The flow field shown is identical for the cases that eliminate the separation bubble. All investigated variations are capable to reduce or even eliminate the size of the separation bubble and therefore reduce the risk of a blockage of the isolator caused by the separation bubble.

The relative mass loss caused by the suction and the total pressure level at the isolator exit relative to the case without suction are listed in Tab. 5-2. The relative total pressure level is calculated by integrating the total pressure over the coordinate y .

Tab. 5-2: Relative suction mass flow and relative total pressure level at exit for the different suction configurations

Suction configuration	Relative suction mass flow [-]	Relative total pressure level at exit [-]
without	0 %	100 %
1	1.5 %	53 %
2	6.2 %	59 %
3	1.8 %	64 %
4	1.6 %	91 %
5	2 %	60 %
6	2 %	60 %

The mass loss trough the suction slot is more or less constant for all configurations expect configuration 2 with values between 1.5 % and 2 %. Considering the total pressure levels at the exit plane, the loss compared to the flow field without suction configuration 4 has losses significantly lower compared to the other cases. This is the case where the separation bubble is not completely eliminated. The boundary layer suction is capable of eliminating or reducing

the lip shock induced separation bubble with a small relative mass loss, still resulting in a remarkable total pressure loss at the isolator exit compared to the flow case without suction.

5.1.5 Comparison with DLR results

Within another subproject of the research training group an almost identical inlet model was investigated at the H2K hypersonic wind tunnel at the DLR in Cologne (chapter 1.2.2) [31]. The geometry of the model is identical besides the DLR model having a continuous transition on the ramp between inlet and isolator.

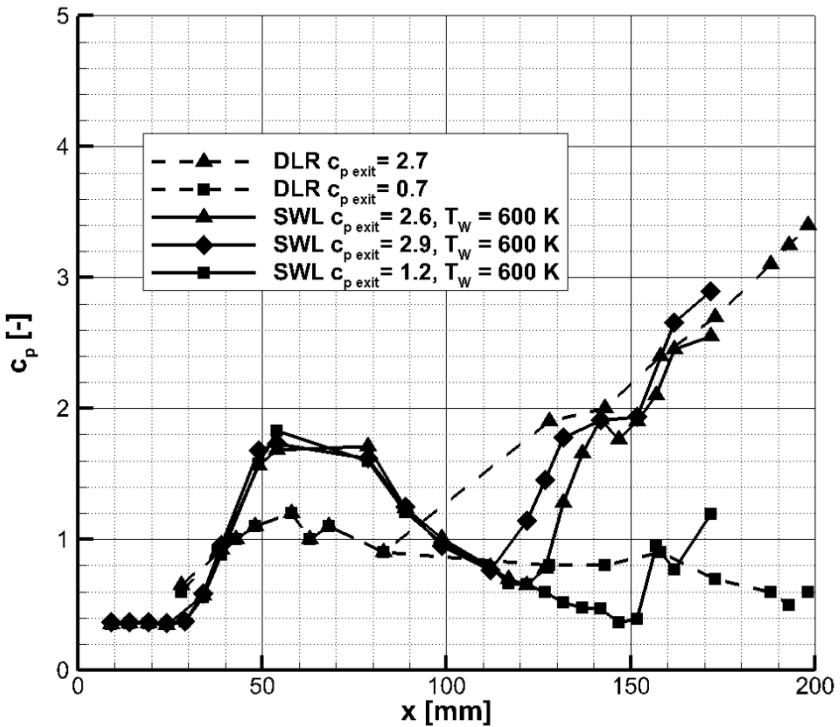


Fig. 5-21: Comparison of pressure distributions on the ramp of SWL and DLR experiments with varying back pressure and different wall temperatures (SWL model only)

The SWL model has a sharp expansion corner at this point due to constructive restrictions ($x = 0$). This leads to the difference in plateau pressure at impact location of the lip shock on

the ramp ($x \sim 55$ mm in Fig. 5-21). The DLR experiments were done in the H2K facility in Cologne at a free stream Mach number of 7 with a total temperature of 500 K. In Fig. 5-21 (ramp) and Fig. 5-22 (lip) pressure distributions on ramp and lip are compared. Experiments for comparison were chosen in order to match the pressure levels reached at the last static pressure probes in the SWL model. The SWL experiments shown here were made at Condition I and $T_w = 600$ K, despite one case at $T_w = 1000$ K where pressure values are only available on the lip. The ratio between total and wall temperature is not constant during the experiments at the H2K. At the beginning of the experiments it is about 1.5 and 1.2 at the end of the experimental time [31].

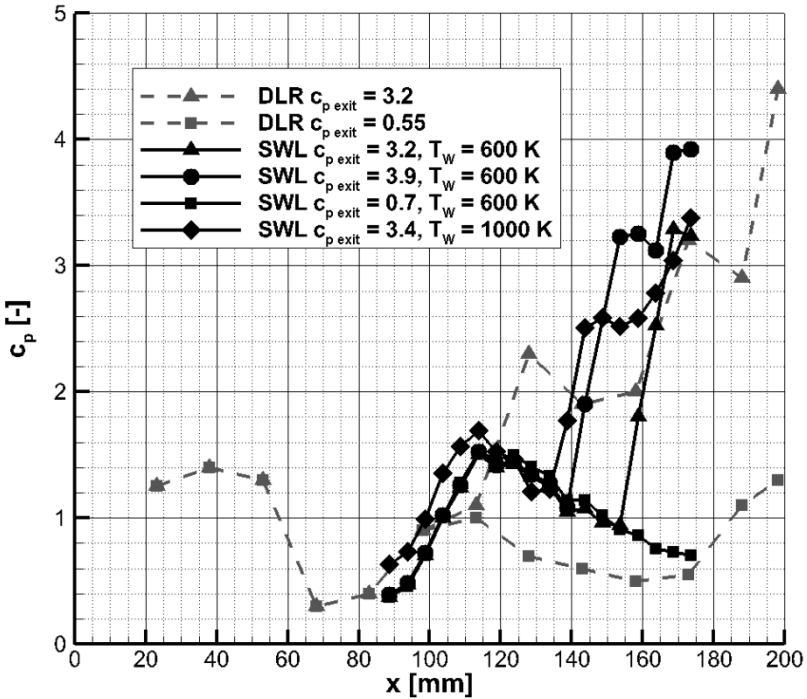


Fig. 5-22: Comparison of pressure distributions on the lip of SWL and DLR experiments with varying back pressure and different wall temperatures (SWL model only)

At Condition I and $T_w = 600$ K the ratio is 1.9 and 1.2 at $T_w = 1000$ K in the TH2 tunnel. The DLR experiments show longer shock trains compared to the $T_w = 600$ K cases at the SWL where the total to wall temperature ratio is higher. Pressure values are available up to

$x = 180$ mm at the SWL model due to constructive restrictions and up to $x = 200$ mm in the DLR model. For the comparison the exit plane is defined at the location of the last pressure probe in the SWL model. This approach is feasible assuming that the shock train is described by a single pressure function for constant flow conditions. This assumption is underlined by comparing the two SWL plots at $T_w = 600$ K that show a more or less constant offset. The SWL experiment with $c_{p, \text{exit}} = 3.2$ has pressure values that are identical at the exit plane on the lip compared to the DLR results with slightly higher pressure on the ramp.

For the other SWL experiment at $T_w = 600$ K the pressure is higher on the ramp as well as on the lip. For both cases the shock train is significantly shorter in the SWL model. This is coherent to the trends shown for the shock train length with the wall temperature. The shock train is longer for smaller values of total to wall temperature.

The results are closer together for the $T_w = 1000$ K case where the values for the ratio are closer together still with longer shock train for the DLR model. Here the total to wall temperature ratio is either identical or higher at the DLR. Therefore an identical or shorter shock train would be expected here in the DLR model.

It should be noted here that both models are not identical with a contoured transition between the last inlet ramp and the isolator ramp at the DLR model. Also the free stream Mach number is slightly lower in the DLR facility with $Ma_\infty = 7$. Later (chapter 5.2) it will be shown that the shock train length is also increasing with an increasing difference between the recovery temperature T_r and the wall temperature. This difference is very small for the flow cases at $T_w = 1000$ K at the SWL and results in smaller heat losses, which gives a possible explanation for the shorter shock train observed in the experiments at the SWL.

5.2 Heatable isolator model

5.2.1 Isolator entrance flow conditions

In Tab. 5-3 the isolator entrance flow conditions for the experiments described throughout the following chapter are listed.

Tab. 5-3: Isolator entrance flow conditions

Condition I				Condition II
Ma_2	3.5	3	2	3
T_2 [K]	333	411	639	786
T_{r2} [K]	1025	1038	1071	1984
Re_2 [$10^6/m$]	7.9	6.9	4.7	8.2
p_2 [Pa]	12440	16320	27790	39650

5.2.2 Basic flow field

A typical experimental result is shown in Fig. 5-23. Two shocks are originating from the leading edges of ramp and lip. In the part of the isolator that is equipped with pressure probes no pressure variations caused by these shocks are visible which proofs the weakness of these shocks. The pressure distribution is composed with values taken on the ramp as well as on the lip. The deviation of the values indicates a very uniform flow field in the isolator. As for all following figures the flow is coming from the left. At the right end of the schlieren image the tubes of the pitot probes are visible.

The shock train starts at $x = 145$ mm with multiple weak oblique shocks. The wall pressure rises linear. For a Mach number $Ma_2 = 2.46$ turbulent oblique shock trains this was also observed by Fejer et al. [26] (see chapter 2.5) At the pitot probes a core Mach number of 1.7 is determined from the measured pitot and wall pressure assuming that the static pressure is constant over the whole cross section.

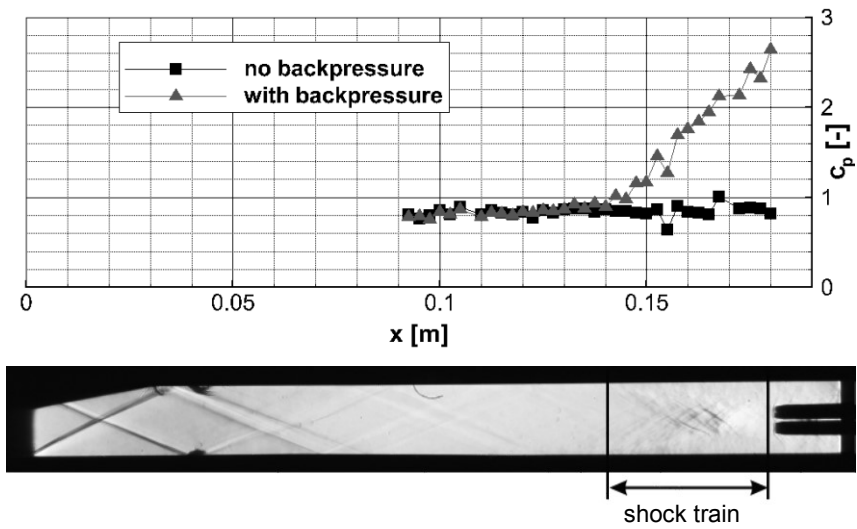


Fig. 5-23: Schlieren image (bottom) and wall pressure distribution (top) at Condition I, $Ma_2 = 3$, $T_w = 800\text{ K}$

5.2.3 Variation of back pressure

The flow field was investigated for the following isolator entrance Mach numbers and shock tunnel conditions as listed in Tab. 5-4. Some pressure distributions will be shown here in the following pictures. The flow cases not shown here can be found in the appendix.

Tab. 5-4: Isolator entrance flow condition for the experiments

	Condition I			Condition II
$T_w[\text{K}]$	$Ma_2 = 3.5$	$Ma_2 = 3$	$Ma_2 = 2$	$Ma_2 = 3$
300	$T_w/T_0 = 0.26$	$T_w/T_0 = 0.26$	$T_w/T_0 = 0.26$	$T_w/T_0 = 0.14$
600	$T_w/T_0 = 0.52$	$T_w/T_0 = 0.52$	$T_w/T_0 = 0.52$	$T_w/T_0 = 0.27$
800	$T_w/T_0 = 0.70$	$T_w/T_0 = 0.70$	$T_w/T_0 = 0.70$	$T_w/T_0 = 0.36$
1000	-	$T_w/T_0 = 0.87$	$T_w/T_0 = 0.70$	$T_w/T_0 = 0.45$

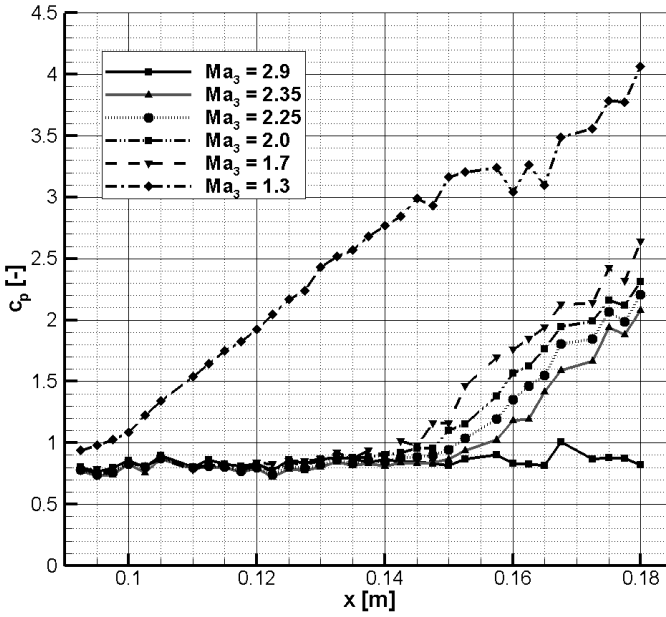


Fig. 5-24: Wall pressure distributions at Condition I, $Ma_2 = 3.0$, $T_w = 800$ K with different back pressures

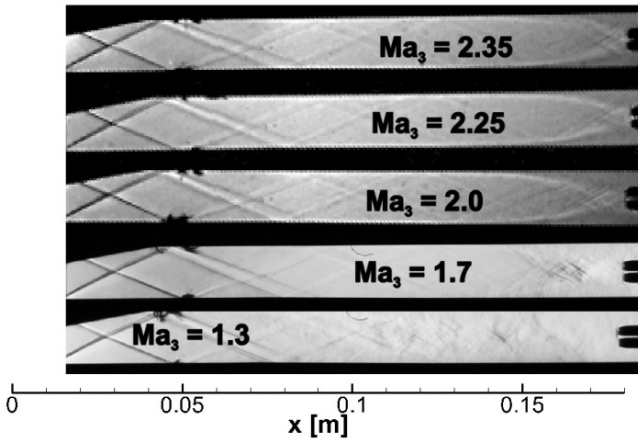


Fig. 5-25: Schlieren and shadowgraph images for the flow cases shown in Fig. 5-24

In Fig. 5-24 wall pressure distributions for different wall pressures at Condition I with $Ma_2 = 3.0$ and $T_w = 800$ K are shown. The corresponding shadowgraph and schlieren images can be found in Fig. 5-25. A wide range of back pressures is shown and the isolator exit Mach number was reduced to values from $Ma_3 = 2.35$ down to $Ma_3 = 1.3$. All pressure slopes are more or less parallel. This confirms the assumption of Waltrup and Billig [101] that the wall pressure distribution is identical for varying overall pressure rise at the same isolator entrance condition for a wide range of shock train lengths. Here a shock train length of up to about half of the isolator length was reached.

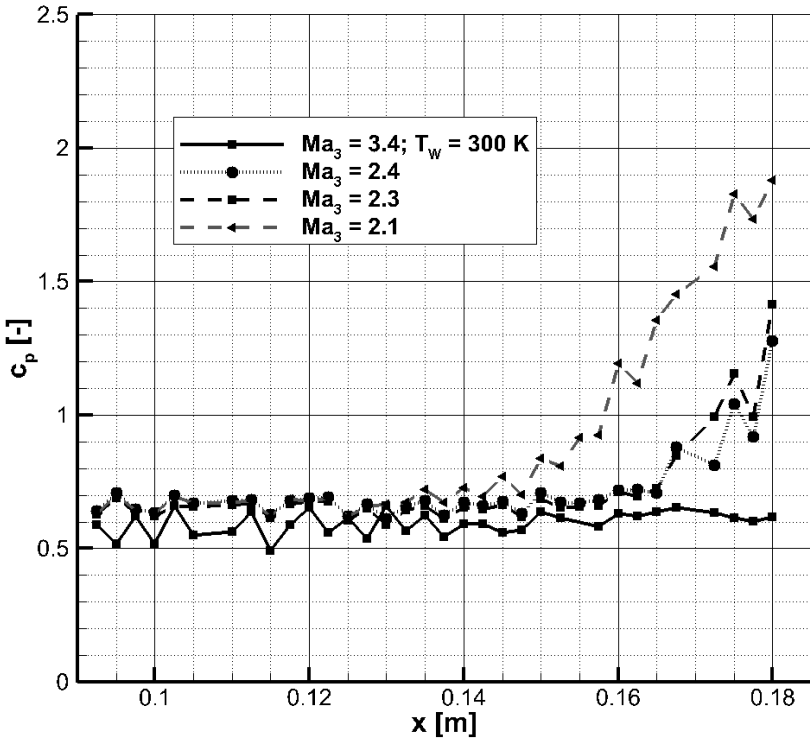


Fig. 5-26: Wall pressure distributions at Condition I, $Ma_2 = 3.5$, $T_w = 1000$ K with different back pressures

The pressure rise is a more or less a linear function of the shock train coordinate s starting with the shock train. The associated schlieren ($Ma_3 = 1.3$ and $Ma_3 = 1.7$) and shadowgraph

images ($Ma_3 = 2.0 - 2.35$) all show the shock trains without strong well defined shocks. This indicates boundary layer transition upstream of the shock train.

The wall pressure distributions for Condition I, $Ma_2 = 3.5$, $T_w = 1000$ K are shown in Fig. 5-26 with the corresponding shadowgraph images in Fig. 5-27. The observed flow fields are basically identical to the case at Condition I with $Ma_2 = 3.0$. The pressure rises more or less linear inside the shock train. The pressure distributions for the different back pressure levels are more or less parallel showing the existence of a single pressure distribution for the shock train for this defined flow case. Again in the shadowgraph images no well-defined shock structure is visible and the pressure rises linearly throughout the shock train. The oblique shocks visible at the entrance of the isolator are caused by the buildup of the boundary layer. They are relatively weak as they have no significant influence on the pressure distribution further downstream. Still they are shown stronger in the schlieren/shadowgraph images compared to the shock train (see e.g. Fig. 5-25). This shows that the imaging system is sensitive and that the low density of the flow at the higher Mach numbers at Condition I is not the reason for the absence of strong well defined shocks in the shock trains.

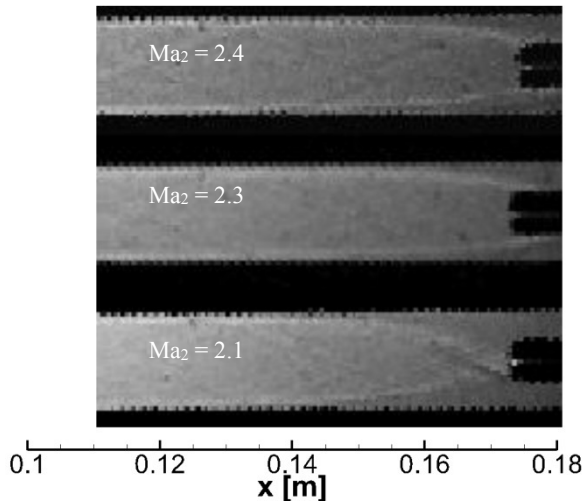


Fig. 5-27: Shadowgraph images of the rear part of the isolator for the flow cases shown in Fig. 5-26

For the isolator entrance Mach number $Ma_2 = 2.0$ at Condition I, $T_w = 300$ K the pressure distribution for varying back pressures are shown in Fig. 5-28 with the corresponding schlieren images in Fig. 5-29.

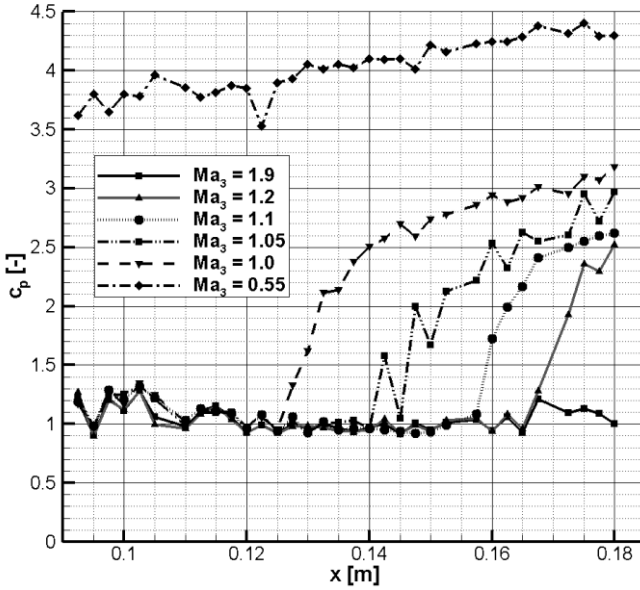


Fig. 5-28: Wall pressure distributions at Condition I, $Ma_2 = 2.0$, $T_w = 300$ K with different back pressures

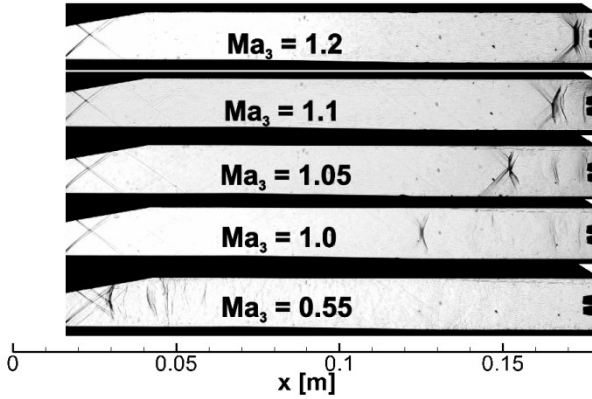


Fig. 5-29: Schlieren images for the flow cases shown in Fig. 5-28

It should be noted that for the $Ma_3 = 1.2$ case the calculation of this Mach number is not very reliable. The Mach number is determined assuming isobaric conditions between the last wall pressure probe and the location of the pitot tubes. Still as Ma_3 is not used of any correlations it will still be used to name the flow case. Here typical normal shock trains are created for all pressure rise levels investigated. The shock trains are initialized by a normal shock with λ -feet.

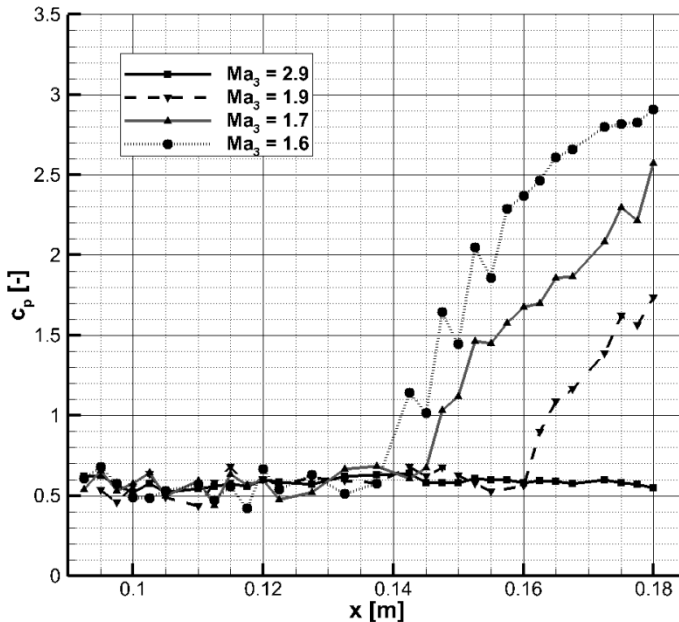


Fig. 5-30: Wall pressure distributions at Condition II, $Ma_2 = 3.0$, $T_w = 300$ K with different back pressures

The shock train length is increasing with increasing isolator pressure rise and has almost filled the complete isolator for the flow case with $Ma_3 = 0.55$ (Fig. 5-29). Downstream of the initial shock weaker normal shocks are visible. The schlieren images show strong well defined shocks at the beginning of the shock trains for all cases in opposition to the results at Condition I, $Ma_2 = 3.0$ (Fig. 5-24 and Fig. 5-25). The pressure distributions are still more or less self similar, but contrary to the cases discussed before the pressure is not a linear function of shock train running length as for $Ma_2 = 3$ and $Ma_2 = 3.5$ from above. A quadratic relation as used by Waltrup and Billig equation 2.61 appears to be a much better fit here.

Wall pressure distributions for $Ma_2 = 3$ at Condition II are shown in Fig. 5-30. The corresponding shadowgraph images for two of the flow cases in Fig. 5-31 show strong well defined oblique shocks at the starting points of the shock trains. Also the pressure distributions have a larger gradient at the beginning of the shock trains. The gradient is then decreasing downstream throughout the shock train. As for the $Ma_2 = 2$ case at Condition I here also a quadratic relation between pressure and shock train running length is adequate. As mentioned above in this case the pressure distributions also show a more or less self-similar behavior allowing a scaled pressure function of the form $p/p_{\text{ref}} = f(s/b)$.

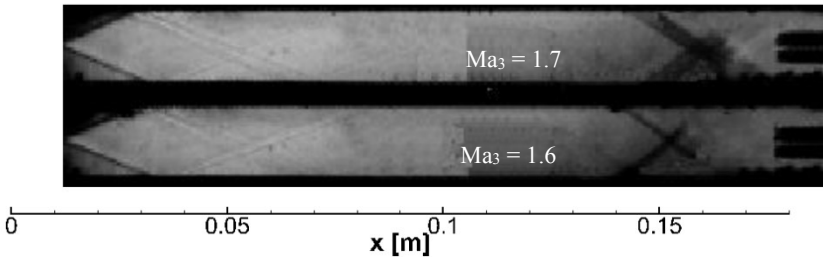


Fig. 5-31: Shadowgraph images for two flow cases of the wall pressure distribution in Fig. 5-30

Comparing the four flow cases described above at Condition I the shock trains with higher isolator entrance Mach number ($Ma_2 = 3$ and 3.5) feature a linear wall pressure distribution throughout the shock train. The shock train begins without clearly defined shocks (best shown in Fig. 5-23). At the lower Mach number at Condition I as well as at the higher total temperature at Condition II well defined strong shocks are visible at the beginning of the shock trains. The pressure gradient is high at the beginning of the shock train and then decreasing further downstream. This is not a wall temperature effect as the same characteristic distributions are shown independently from the wall temperature in all pressure distribution for these flow cases (chapter 8). The flow fields with elevated wall temperatures shown for Condition I $Ma_2 = 3.5$ and $Ma_2 = 3.0$ were chosen here due to the availability of shadowgraph and schlieren images for these cases as well as different successfully created back pressures. Also it cannot be an effect of the confinement in the flow channel as e.g. Condition II, $T_w = 1000$ K ($T_w/T_0 = 0.45$) features a thicker boundary layer compared to Condition I, $T_w = 300$ K ($T_w/T_0 = 0.26$).

As speculated above boundary layer transition upstream of the shock train can explain this different flow topologies shown by the experiments. In Fig. 2-21 [26] the influence of forced transition upstream of the shock train was identical to the results here at Condition I $Ma_2 = 3$ and $Ma_2 = 3.5$. The forced transition leads to a linear pressure rise in the shock train in oppo-

sition to the more typical pressure distributions for the other flow cases. The difference observed between shock trains at Condition I and Condition II at the constant $Ma_2 = 3.0$ can be a result of different free stream turbulence parameters for those. The weak visible shocks are also not originating from the surface as they do for the low Mach number and high total temperature cases. This is again an indication for a thicker and therefore turbulent boundary layer upstream of the shock train.

At the lower isolator entrance Mach number, the results from [26] show no qualitative difference. The flow field observed at $Ma_2 = 2$ can therefore include a laminar as well as a turbulent boundary layer upstream of the shock train. The trend seems to be a laminar boundary layer upstream of the shock train at Condition II and a turbulent boundary layer upstream of the shock train at Condition I with $Ma_2 = 3$ and $Ma_2 = 3.5$.

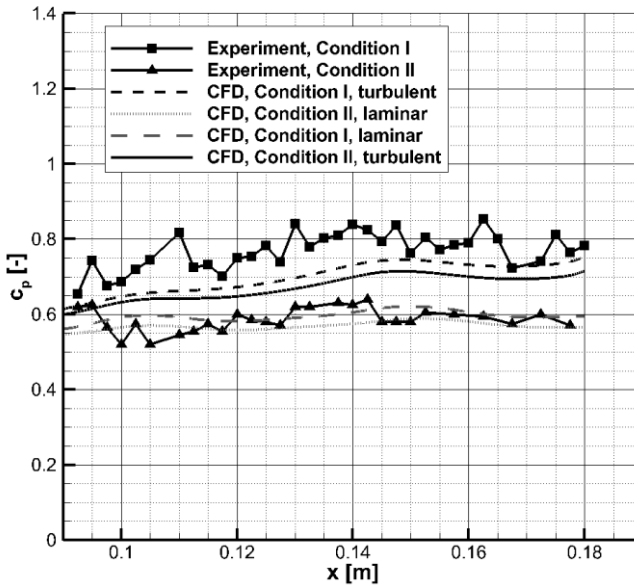


Fig. 5-32: Wall pressure distribution in the isolator without back pressure at Conditions I and II from experiments compared to laminar and turbulent CFD calculations.

To finally prove this theory of different transitional behavior for the two flow conditions 3d CFD calculations of the isolator flow field without back pressure are conducted at

$Ma_2 = 3$. At this Mach number both types of pressure distributions were observed in the experiments. The calculations shown in Fig. 5-32 are done with laminar and turbulent flow. The figure shows the pressure distributions on the centerline of the model from these calculations and the corresponding experiments all done at $T_w = 300$ K. The results show a significant offset in the pressure levels between the experimental results at conditions I and II. The comparison with the numerical results shows that this offset is caused by the different state of the boundary layer. The turbulent boundary layer at Condition I leads to the higher pressure values due to the higher displacement thickness. This is consistent with the boundary layer transitioning from laminar to turbulent at Condition I ($Re_\infty = 5.8 \cdot 10^6$) while it is laminar at Condition II ($Re_\infty = 4.8 \cdot 10^6$). This is consistent with the boundary layer transitioning to turbulent with increasing Reynolds number.

5.2.4 Wall temperature effects

Wall pressure distributions reaching about the same overall pressure rise are shown in Fig. 5-34 for Condition I, $Ma_2 = 3.0$. The increase in wall temperature leads to a constant decrease of the shock train length. This is also visible in the corresponding shadowgraph images in Fig. 5-33, where the shock train starting points for $T_w = 300$ K and $T_w = 1000$ K are marked to allow comparison.

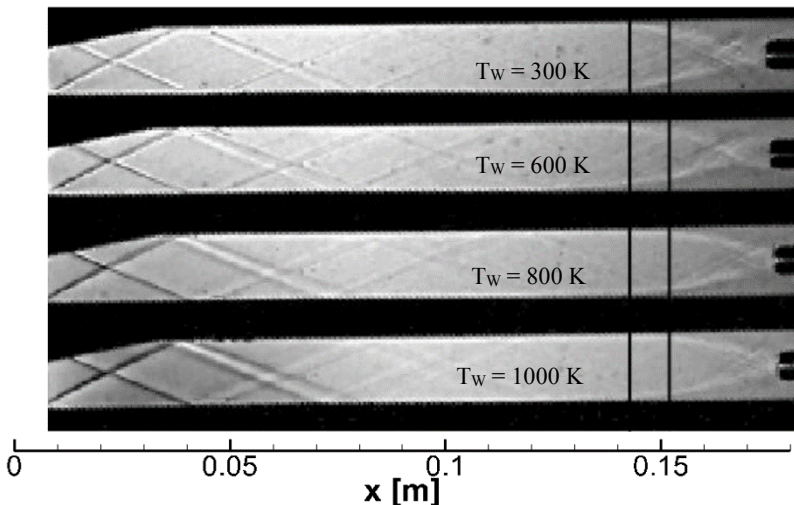


Fig. 5-33: Shadowgraph images with constant pressure rise and varying wall temperature at Condition I; $Ma_2 = 3$; shock train starting points for $T_w = 300$ K and $T_w = 1000$ K marked by vertical lines

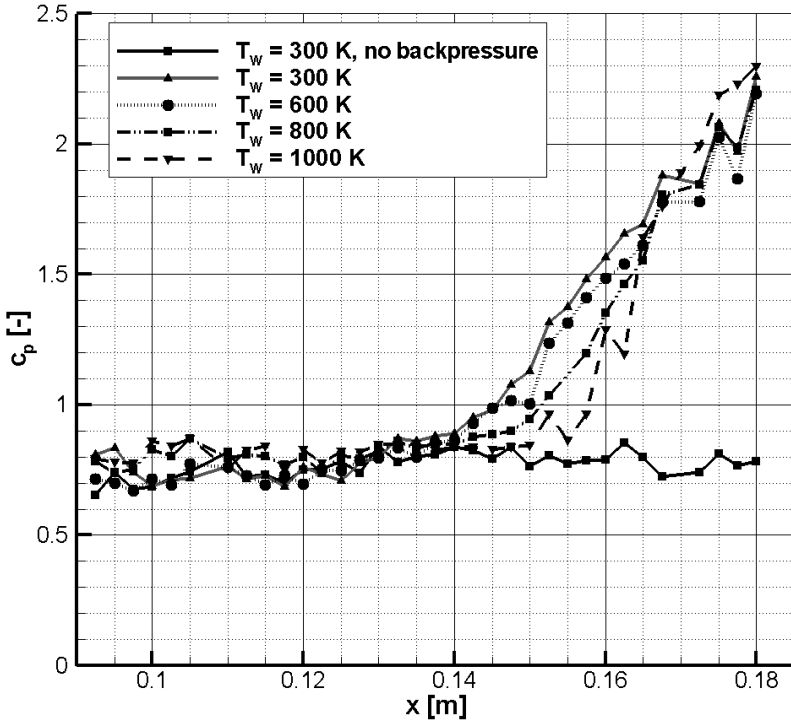


Fig. 5-34: Wall pressure distribution with constant pressure rise and varying wall temperature at Condition I, $Ma_2 = 3$

5.2.5 Quantification of wall temperature effect

Wall pressure distributions were aligned to create a single pressure distribution for every investigated flow case. The pressure in the shock train is now a function of the shock train coordinate s starting at the beginning of the shock train. After the alignment the available pressure values for every available coordinate s in the shock train were averaged. With these it is possible to compare the results of experiments with different back pressure as first done by Waltrup and Billig [101]. Also possible errors caused by inexact measurements, e.g. due to changing sensitivity of single pressure probes are also minimized by this procedure as values measured by different probes are combined for the pressure values of a fixed position s . The different lengths of the pressure distributions throughout the shock train shown for different flow conditions are caused by the different back pressures created with the double

wedge. At higher back pressures often asymmetric flow fields in the isolator were observed. Naturally only symmetric flow fields were taken into account for the present research.

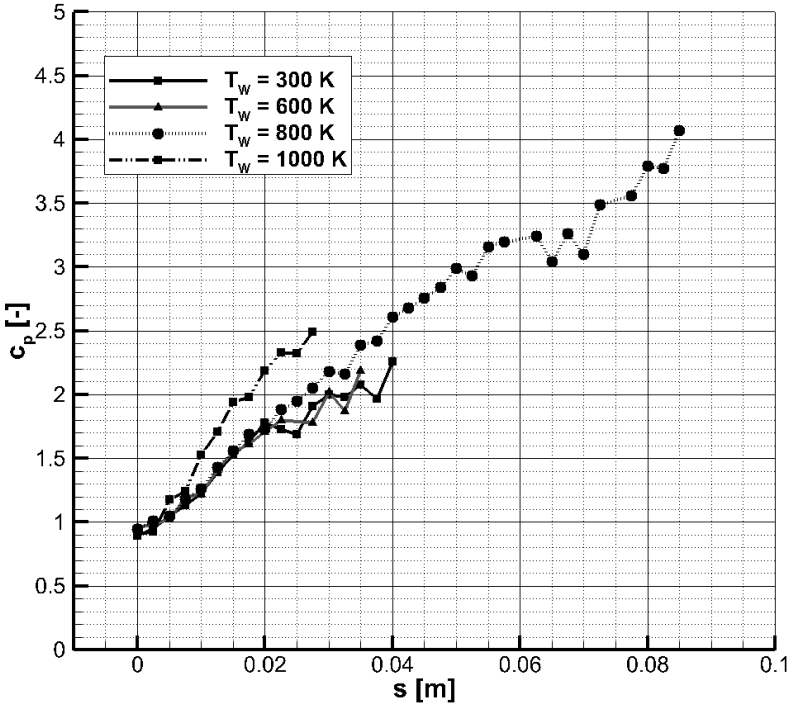


Fig. 5-35: Shock train pressure distribution for different wall temperatures at Condition I, $Ma_2 = 3.0$

For Condition I, $Ma_2 = 3$ these are shown in Fig. 5-35. For all investigated wall temperatures the pressure rise is more or less linear with no clear pressure jump at the beginning of the shock train. There is no difference visible between the pressure rise at 300 K and 600 K wall temperature. The pressure gradient is then increasing for 800 K and 1000 K wall temperature. The pressure gradient in the shock train is increasing with increasing wall temperature and therefore the shock train length is decreasing. The effect is strong for the higher wall temperatures and very weak in the lower temperature regime.

The shock train pressure distributions for the same isolator entrance Mach number ($Ma_2 = 3.0$) at Condition II are plotted in Fig. 5-36. Here the pressure distribution shows a high pressure gradient at the beginning of the shock train for all investigated cases that is decreasing further downstream. There is a general trend of increasing pressure values for

higher wall temperatures, but some scatter is present. The trend is best shown by comparing the pressure distributions for $T_w = 300$ K and $T_w = 1000$ K. For $T_w = 1000$ K the measured pressure is higher compared to the values for $T_w = 300$ K for most available probe positions. Still the overall trend is not as clear as for the same isolator entrance Mach number at Condition I.

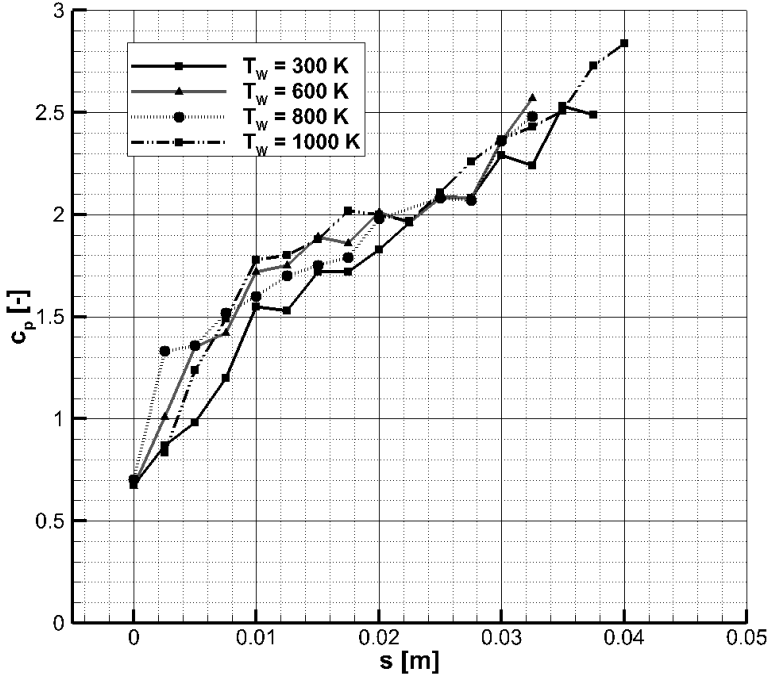


Fig. 5-36: Shock train pressure distribution for different wall temperatures at Condition II, $Ma_2 = 3.0$

For Condition I, $Ma_2 = 3.5$ the pressure distributions are plotted in Fig. 5-37. Here all pressure distributions show a more or less linear pressure increase throughout the shock train. Generally for identical positions s along the shock train the pressure values are increasing with the wall temperature with some scatter being present. For this high isolator entrance Mach number only rather short shock trains have been observed explaining the proximity of the pressure profiles for the different wall temperatures here.

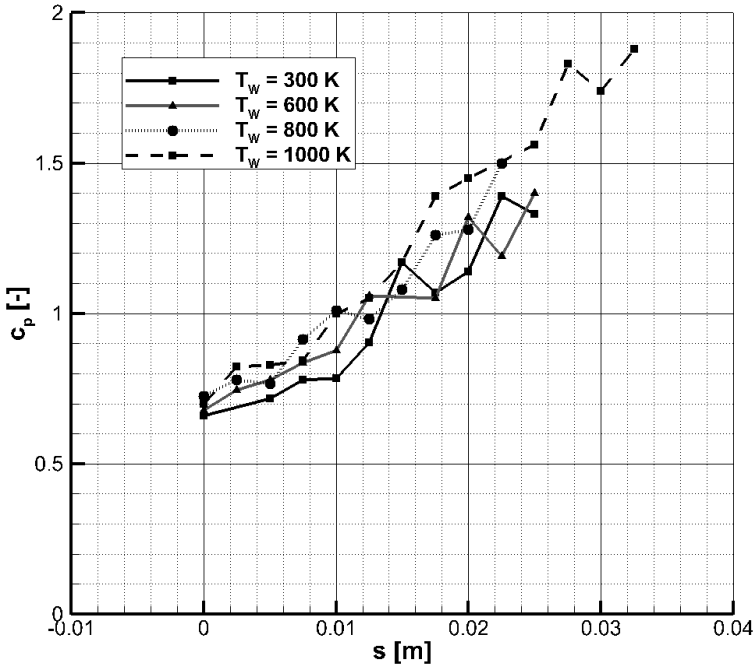


Fig. 5-37: Shock train pressure distribution for different wall temperatures at Condition I, $Ma_2 = 3.5$

The three available pressure profiles for $Ma_2 = 2.0$ are shown in Fig. 5-38. They all show a high pressure gradient at the beginning of the shock train that is decreasing further downstream. In this case at $T_w = 300$ K the pressure rise appears to be slightly higher than in the cases with elevated wall temperatures especially between $s = 0.01$ and $s = 0.02$. Further downstream the pressure gradients for $T_w = 600$ K and $T_w = 800$ K are higher and the pressure profiles are closing in again. No clear wall temperature effect is shown by the pressure distributions.

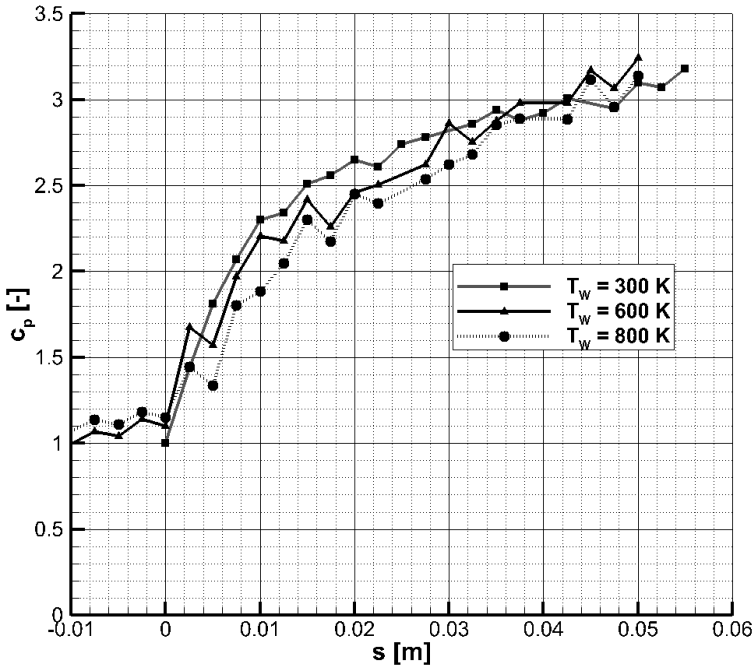


Fig. 5-38: Shock train pressure distribution for different wall temperatures at Condition I, $Ma_2 = 2.0$

Pressure distributions are created for different wall temperatures for four different flow cases. The cases with isolator entrance Mach numbers $Ma_2 = 3$ and $Ma_2 = 3.5$ show a decreasing shock train length with increasing wall temperature for both the shock trains with linear pressure rise as well the classical cases with a decreasing pressure gradient. For the normal shock train at $Ma_2 = 2.0$ no clear wall temperature effect is shown.

For the flow cases where a laminar boundary layer is suspected upstream of the shock train (Condition I, $Ma_2 = 2.0$ and Condition II, $Ma_2 = 3.0$) the empirical correlation of Waltrup and Billig (equation 2.61) can be applied due to the qualitatively identical development of the pressure with increasing shock train running length s .

The correlation will be used simplified to the following form to quantify only the wall temperature effects:

$$\frac{s}{L} = A \cdot \left[50 \left(\frac{p_3}{p_2} - 1 \right) + 170 \left(\frac{p_3}{p_2} - 1 \right)^2 \right] \quad (5.1)$$

Here all influencing flow parameters are combined into the factor A. This factor is derived from the pressure distributions for each wall temperature at the different flow conditions. The shock train length is nondimensionalised with the isolator length L. For the cases with a suspected transition upstream of the shock train (Condition I, $Ma_2 = 3.0$ and $Ma_2 = 3.5$) a linear correlation between the shock train running length and the pressure is more applicable. Therefore the factor A is calculated from the experiments based on the following correlation:

$$\frac{s}{L} = A \cdot \left(\frac{p_3}{p_2} - 1 \right) \quad (5.2)$$

The only factor influencing the shock train in the empirical correlation by Waltrup and Billig by wall temperature effects is the boundary layer momentum thickness δ_2 (equation 2.33) that will be nondimensionalised with the half flow channel height $h/2$. Then according to the correlation the factor A/L should be a function of $2\delta_2/h$:

$$A \sim \left(\frac{2\delta_2}{h} \right)^{0.25} \quad (5.3)$$

Tab. 5-5: Boundary layer momentum thickness [$10^{-3}m$] for undisturbed flow ($x = 0.18 m$)

T_w [K]	300	600	800	1000
Cond. I, $Ma_2 = 3.5$, turbulent	0.30	0.25	0.23	0.21
Cond. I, $Ma_2 = 3.0$, turbulent	0.34	0.285	0.26	0.24
Cond. I, $Ma_2 = 2.0$, laminar	0.15	0.14	0.14	
Cond. II, $Ma_2 = 3.0$, laminar	0.097	0.094	0.092	0.091

For a fixed flow channel geometry, Mach and Reynolds number the variation of the wall temperature leads to the following criteria:

$$\frac{A}{\left(2 \frac{\delta_2}{h}\right)^{0.25}} = \frac{A}{C} = \text{const.} \quad (5.4)$$

Values of δ_2 for the undisturbed flow field can be calculated with the reference temperature method as described in chapter 2.2. For the investigated flow cases they are listed in Tab. 5-5.

Values for the factor A were obtained from the measured pressure distributions with the curve fitting tool of Matlab using the custom function mode with equations 5.1 and 5.2.

Tab. 5-6: Parameter A for the investigated flow cases calculated from the pressure distributions

T_w [K]	300	600	800	1000
Cond. I, $Ma_2 = 3.5$	0.133	0.127	0.111	0.0889
Cond. I, $Ma_2 = 3.0$	0.157	0.146	0.131	0.0972
Cond. I, $Ma_2 = 2.0$	$2.94 \cdot 10^{-4}$	$2.77 \cdot 10^{-4}$	$2.83 \cdot 10^{-4}$	
Cond. II, $Ma_2 = 3.0$	$1.31 \cdot 10^{-4}$	$1.29 \cdot 10^{-4}$	$1.27 \cdot 10^{-4}$	$1.21 \cdot 10^{-4}$

Results for the factor A are plotted in Fig. 5-39. The parameter A is nonlinearly decreasing for the presumably turbulent cases (Condition I, $Ma_2 = 3$ and $Ma_2 = 3.5$) and the case at Condition II. At Condition I, $Ma_2 = 2.0$ the factor A is slightly decreasing for less than $T_w = 600$ K and then increasing. This little variation of the factor A for this case might indicate a small or even no wall temperature effect here.

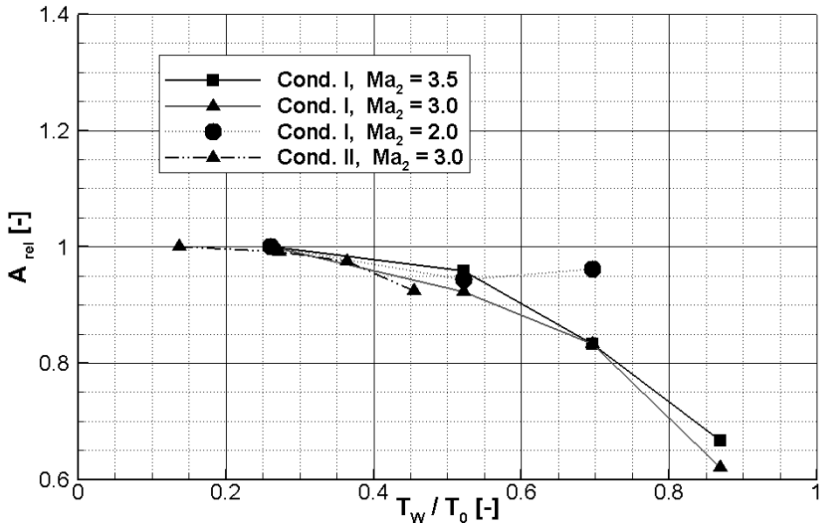


Fig. 5-39: Correlation parameter A for different wall temperatures normalized with the values for $T_w = 300$ K

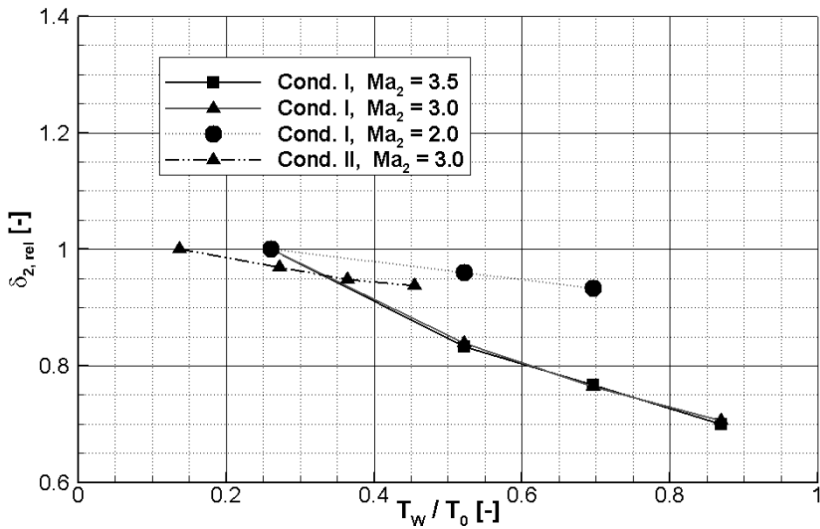


Fig. 5-40: Boundary layer momentum thickness for undisturbed flow at position $x = 0.18$ m for different wall temperatures normalized with the values for $T_w = 300$ K

The relative values for the momentum layer thickness are plotted in Fig. 5-40. They show δ_2 decreasing more or less linearly with the wall temperature for all flow cases. The relative wall temperature effect is stronger for the presumably turbulent cases as are the absolute values (Tab. 5-5). To match the effect of decreasing shock train length with increasing wall temperature ratio shown in the experiments qualitatively in the correlation of Waltrup and Billig (at this point leaving out Cond. I; $T_w = 800$ K; $Ma_2 = 2.0$) a much higher decrease in the momentum layer thickness would have been required. Therefore within the correlation of Waltrup and Billig the wall temperature effect on the boundary layer momentum thickness cannot explain the observed decreasing of the shock train length. This is shown in Fig. 5-41 where the influence of the boundary layer momentum thickness factor $(2*\delta_2/h)^{0.25}$ is too small to create a constant distribution with increasing wall temperature ratio. The correlation of Waltrup and Billig allocates an exponent of 0.25 to the momentum layer thickness.

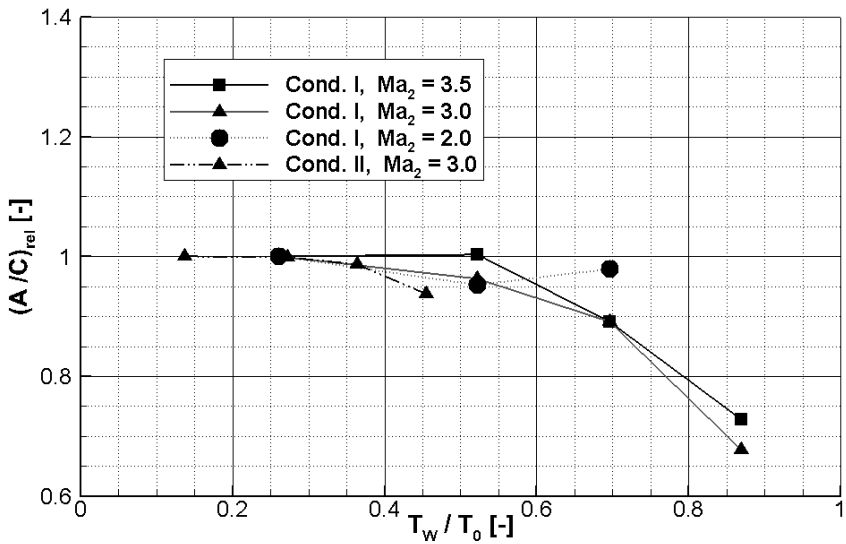


Fig. 5-41: Correlation parameter A normalized with $C = [2*\delta_2/h]^{0.25}$ for different wall temperatures normalized with the values for $T_w = 300$ K

This combined with the limited effect of the wall temperature on δ_2 leads to the effect that the influence of δ_2 on the factor A/C is too weak to compensate its dependence on the wall temperature ratio. The empirical correlation of Waltrup and Billig cannot explain the observed decrease in the shock train length with increasing wall temperature.

Still it has been validated by many experiments. The experiments of Waltrup and Billig and also most other investigations of shock trains were carried out using air at $T_0 = 300$ K with the same wall temperature and therefore it can be regarded as almost adiabatic. With the hot flow conditions of the shock tunnel this is not the case. Just the experiments with $T_W = 1000$ K at Condition I ($T_r = 1063$ K – 1071 K) get close to being adiabatic.

The wall temperature has a large impact on the wall heat transfer. The sensitivity towards wall temperature grows when it reaches values close to the recovery temperature:

$$\dot{q}_W = C_h \rho_e u_e c_p (T_r - T_W) \quad (5.5)$$

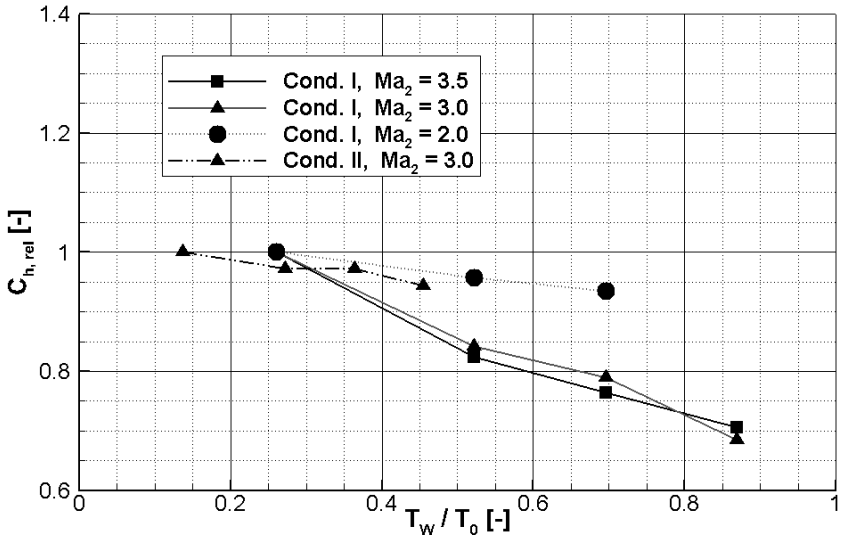


Fig. 5-42: Stanton number for undisturbed flow at isolator exit ($x = 0.18$ m) and for different wall temperatures normalized with the values for $T_W = 300$ K

For an undisturbed, flat plate boundary layer flow the values for the Stanton number C_h are calculated also with the reference temperature method. The calculated values for undisturbed flow at $x = 0.18$ m are listed in Tab. 5-7. For all four flow cases the values for the Stanton number are decreasing linearly with increasing wall temperature as it is shown in Fig. 5-42. The decrease is stronger for the presumably turbulent cases.

Both the heat transfer to the walls as well as the momentum layer thickness describe losses. Therefore it is assumed that the influence on the pressure rise in the shock train of the

heat transfer also has an exponent of 0.25. In Fig. 5-43 the correlation parameter A normalized with the wall heat transfer as stated in equation 5.6 is plotted.

$$\frac{A}{\left(\frac{\dot{q}_w A_w}{\dot{m} c_p T_0}\right)^{0.25}} = \frac{A}{C} = \text{const.} \quad (5.6)$$

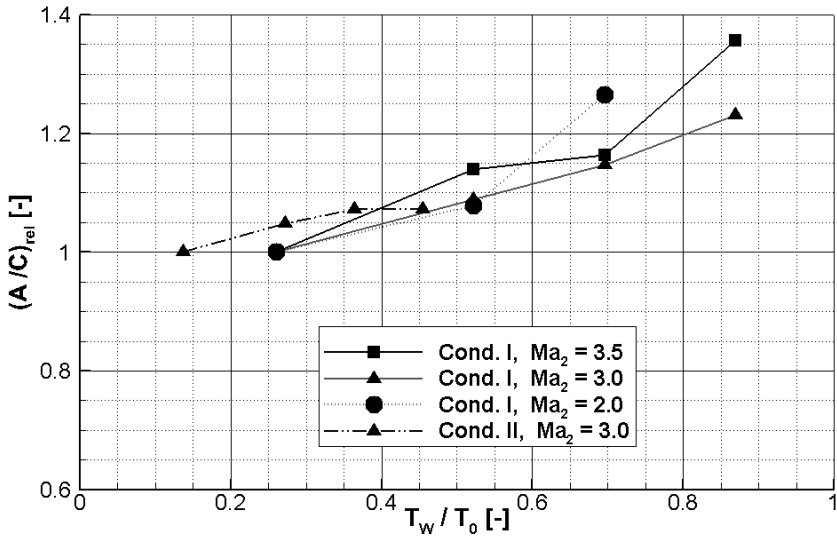


Fig. 5-43: Correlation parameter A normalized with $C = [(\dot{q}_w A_w)/(\dot{m} c_p T_0)]^{0.25}$ for different wall temperatures, normalized with the values for $T_w = 300$ K

To nondimensionalize the wall heat flux the calculated value of q_w is multiplied with the wetted surface area and divided by the total energy entering the isolator. Here the effect of this normalization is too strong leading to an increase in the correlation parameter A with increasing wall temperature observed for all flow cases. This is also observed for the experiments where the wall temperature is significantly lower than the recovery temperature for all considered wall temperatures (Condition II, $Ma_2 = 3.0$ and Condition I, $Ma_2 = 2.0$). Sensitivity caused by $(T_r - T_w)$ at $T_w = 1000$ K cannot be the source of the deviation of the shown value.

Tab. 5-7: Stanton number [-] for undisturbed flow ($x = 0.18$ m)

T_w [K]	300	600	800	1000
Cond. I, $Ma_2 = 3.5$, turb.	0.0017	0.0014	0.0013	0.0012
Cond. I, $Ma_2 = 3.0$, turb.	0.0019	0.0016	0.0015	0.0013
Cond. I, $Ma_2 = 2.0$, lam.	0.00046	0.00044	0.00043	
Cond. II, $Ma_2 = 3.0$, lam.	0.00035	0.00034	0.00034	0.00033

A modified version of the correlation is given in equation 5.7. Here b is a factor that weights the influence of the momentum layer against the influence of the heat transfer. For the different flow cases varying values for b were used with constant values of $b = 0.08$ for the x-type shock trains and $b = 2$ for the normal shock train without an effect caused by the transition of the boundary layer upstream of the shock train. The influences are added as they both describe losses. Also the influence of wall heat flux vanishes when the wall temperature reaches the recovery temperature leaving behind the existing form of the correlation. The exponents of 0.8 for the heat flux and 1.07 for the momentum layer thickness were taken from literature as described in the introduction. They describe the distribution of the two properties in the shock train as a function of the pressure rise. In case of adiabatic wall conditions ($\dot{q}_w = 0$) the exponent of 1.07 combined with 0.23 results in the value of 0.25 as used in equation 3 for the influence of the momentum layer thickness.

$$\frac{A}{\left[\left(\frac{\dot{q}_w A_w}{\dot{m} c_p T_0} \right)^{0.8} + b \cdot \left(\frac{2\delta_2}{h} \right)^{1.07} \right]^{0.23}} = \frac{A}{C} = \text{const.} \quad (5.7)$$

The relation A/C from the modified correlation using equation 5.7 is plotted against the wall temperature in Fig. 5-44. A much more even distribution of the correlation factor A/C with the wall temperature is reached now. With the modified factor C it is possible to describe the observed wall temperature effects. The old version of the correlation is included in the new version for adiabatic wall conditions.

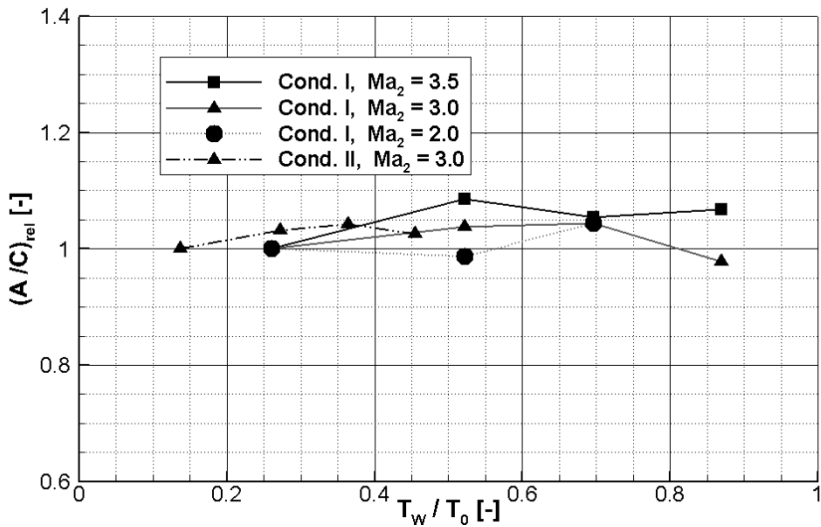


Fig. 5-44: Correlation parameter A normalized with C from equation 5.7 for different wall temperatures normalized with the values for $T_w = 300$ K

5.3 Model with variable isolator flow channel width

The model with variable flow channel width was investigated at Condition I with an isolator entrance Mach number of $Ma_2 = 3.0$. Wall pressure distributions for different flow channel widths are shown in Fig. 5-45 to Fig. 5-47. In Fig. 5-45 and Fig. 5-46 two experiments are shown for almost identical back pressures, while in Fig. 5-47 a variety of back pressures is reached. It is again obvious that the measured pressure profiles can be shifted over each other to create a single pressure function $[f(s)]$ for constant free stream conditions and the same model width. The width was varied from $w = 200$ mm over $w = 150$ mm down to $w = 70$ mm. For all three cases and all shown generated back pressures the pressure rises more or less linear throughout the shock train. The widest range of back pressure was for a width of $w = 70$ mm. The principal flow case is therefore identical to the results with the heatable isolator model. The measurements with different back pressures were combined to create pressure rise plots for the different flow cases as it is described in more detail in chapter 5.2.5. The results from the model with variable width are summarized in Fig. 5-48. The gradient is increasing with decreasing model width from $w = 200$ mm to $w = 150$ mm. The gradient is then decreasing again for $w = 70$ mm.

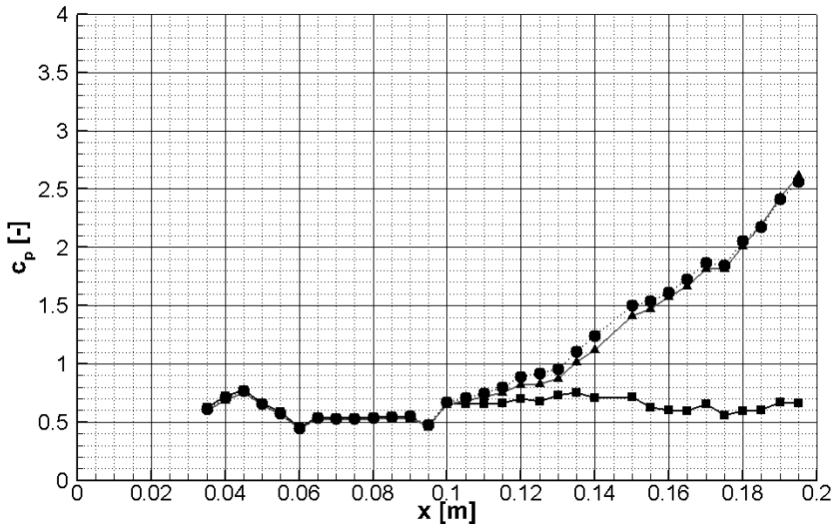


Fig. 5-45: Pressure distributions for 200 mm flow channel width

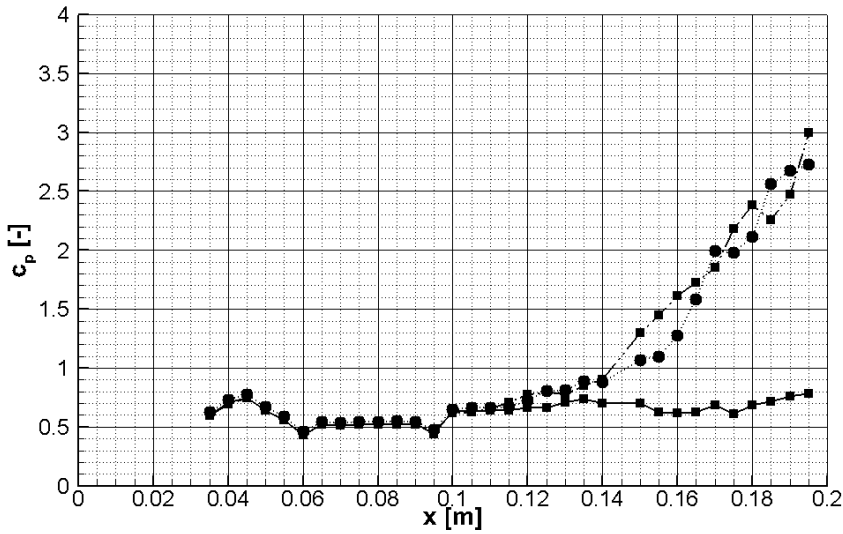


Fig. 5-46: Pressure distributions for 150 mm flow channel width

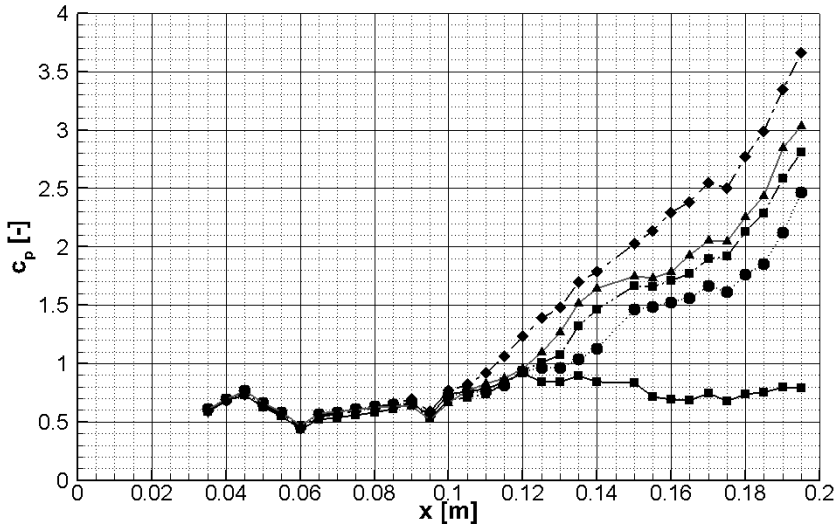


Fig. 5-47: Pressure distributions for 70 mm flow channel width

As described above the gradient of the pressure does scale with the energy losses of the flow which for the here considered constant flow condition can be estimated by equation 5.7. Both losses do occur on wetted surfaces and therefore scale with the wetted area. For a fixed flow condition the ratio between the incoming enthalpy and these losses therefore scales with:

$$\frac{A_E}{A_W} = \frac{wh}{2x(w+h)} \quad (5.8)$$

with:

$$w = h \rightarrow \frac{A_E}{A_W} = \frac{h}{4x} \quad (5.9)$$

$$w = \infty \rightarrow \frac{A_E}{A_W} = \frac{h}{2x} \quad (5.10)$$

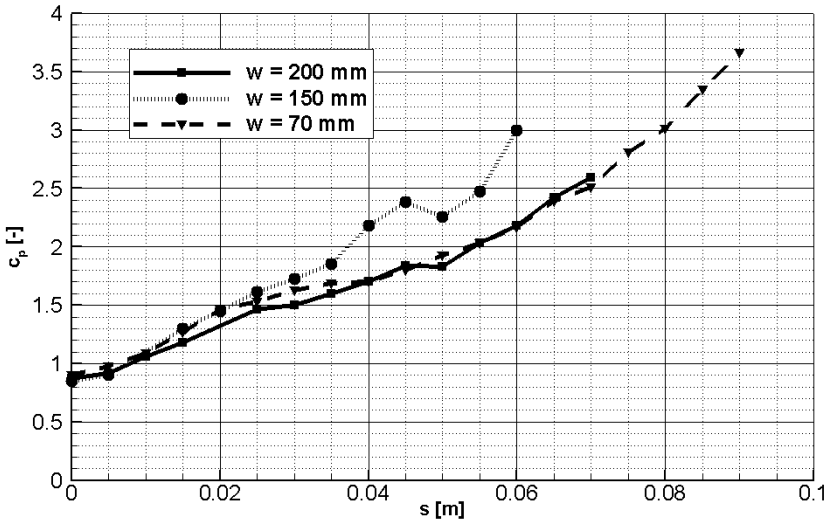


Fig. 5-48: Wall pressure rise along the shock train from model with variable width at Condition I, $Ma_2 = 3.0$

So the highest ratio and therewith the lowest losses are reached for the ideal 2D flow case and more confinement of the flow field leads to higher losses. This explains the decrease of the pressure gradient between $w = 150$ mm and $w = 70$ mm. The increase observed between $w = 200$ mm and $w = 150$ mm is reached due to the positive effect of the additional compression by the shocks from the sidewalls. This effect is shown for the undisturbed flow case in Fig. 5-49. At the end of the isolator a higher pressure level is reached for $w = 150$ mm compared to $w = 200$ mm. Therefore the same overall pressure rise in the isolator is achieved with a smaller relative pressure rise at the isolator exit comparing the shock train to the undisturbed flow case. For $w = 70$ mm and $w = 150$ mm the same pressure level at the exit plane is reached without back pressure. The additional losses over the wetted surfaces lead to the longer shock train at $w = 70$ mm compared to $w = 150$ mm (Fig. 5-48).

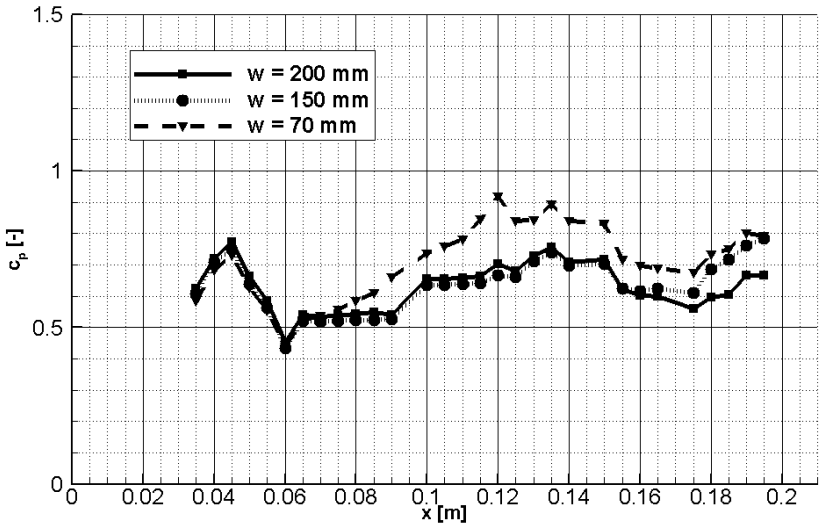


Fig. 5-49: Wall pressure distribution without back pressure at Condition. I, $Ma_2 = 3.0$

6 Conclusion

The aim of this thesis is to investigate wall and total temperature effects on the isolator flow field of a Scramjet engine, specifically the isolator of the research training group (chapter 1.2) engine. Two additional models are also investigated to gain principal knowledge about shock trains.

The isolator flow field of the Scramjet model features two main flow phenomena, the lip shock induced separation bubble in the front part of the isolator and the shock train in the rear part caused by the combustion pressure rise.

The model is investigated at wall temperatures between $T_w = 300$ K and $T_w = 1000$ K and at the total temperatures of $T_0 = 1150$ K and $T_0 = 2200$ K. In compliance with the previous investigations of wall and total temperature effects on separation bubbles [8], [67] the separation point moved upstream with growing wall and decreasing total temperature. The engine does not block due to the separation bubble at any of the investigated flow conditions. The comparison of the results with numerical simulations showed the importance of a possible relaminarisation downstream of the expansion corner. The data from the experiments is consistent with the numerical simulations with no relaminarisation taking place at the expansion corner. The separation bubble is located further downstream compared to simulations that allowed relaminarisation. The possibility of relaminarisation has to be considered for the transformation of the gained data set to another flight environment. Still if the separation bubble grows upstream beyond the expansion corner the isolator does not choke. The whole setup proved to be very sensitive to wall and free stream temperature. Still the ratio between wall and free stream temperature is a useful similarity parameter to quantify the temperature effects for this setup.

A boundary layer suction is investigated numerically to remove the separation bubble. Numerous configurations were investigated. In addition to the reduced mass flow all configurations lead to an additional loss of total pressure compared to the isolator without suction. As the isolator did not block at any investigated condition due to the separation bubble and additionally losses are found for such a system it is not considered further for the design of the Scramjet engine.

To investigate the shock train in the Scramjet isolator back pressure is generated. An asymmetric shock train with a huge separation on the ramp and only small separation bubbles on the lip is shown by numerical simulation. This is caused by the asymmetry of the flow field in the isolator and not by an asymmetric back pressure generation. The shock train is maintained well within the isolator for all flow cases with supersonic exit flow. In agreement with

the results in [31] a shorter isolator is suggested for a possible future isolator design. The shock train is growing further upstream into the isolator with increasing wall and decreasing total temperature. The ratio between wall and free stream temperature is a good similarity parameter here as well. Due to the asymmetric and design specific topology of the flow field no comparison with existing correlations is conducted.

Principal knowledge about the wall and total temperature effects on shock trains is gained by the investigation of two additional models, a heatable isolator model and a cold isolator model. Both have a uniform inflow. With these models also the isolator entrance Mach number is varied. For the different flow cases normal as well as oblique shock trains were created. In some cases the pressure function within the shock train shows a quadratic behavior as expected by the existing correlation by Waltrup and Billig. The shocks of the shock train are shown well defined in the schlieren and shadowgraph pictures. At higher Mach numbers at Condition I also shock trains with a linear pressure distribution were observed. The schlieren and shadowgraph pictures of those cases show numerous weak shocks throughout the shock train indicating turbulent flow probably caused by transition upstream of the shock train. For all investigated flow cases the pressure distribution could be described by a single pressure function. A turbulent boundary layer shows a higher compression in the duct without generated back pressure due to the higher displacement thickness. Numerical simulations show that the pressure distribution without generated back pressure is consistent with turbulent flow for the shock trains with linear pressure rise. For Condition II $Ma_2 = 3$ where a typical nonlinear pressure distribution is shown in the shock train it is consistent with a laminar boundary layer.

In all investigated flow cases the shock train length is decreasing with increasing wall temperature. This is consistent with the existing results of isolator flow with varying wall temperature below the recovery temperature ([51] and [52]). It is also qualitatively consistent with the existing correlation by Waltrup and Billig but the effect is largely underpredicted by the correlation. In opposition to most existing investigations of shock trains (including the results of [101]) the results presented in this thesis contain experiments with wall temperatures far below adiabatic conditions. Therefore it is proposed to include the losses produced by the heat transfer from the flow into the walls into the correlation. A modified version of the correlation is given in equations 5.7. In the given forms the heat transfer can be neglected for wall temperatures close to the recovery temperature due to the factor $(T_R - T_W)$. Naturally a much larger amount of data from many different wind tunnels will be needed to verify and finalize the modification.

The results for the symmetric shock train contradict the results for the shock train in the Scramjet model. The flow path upstream of the shock train in the Scramjet is subject to a wall

temperature influence. The displacement thickness is increasing with increasing wall temperature. This causes larger separation bubbles. The separation bubbles have a significant influence on the flow field (mass flux, total pressure profile) entering the isolator. In the Scramjet model the wall temperature does not only influence the shock train itself but also the isolator entrance flow field. This can explain the observed increase in shock train length with increasing wall temperature in the shock train model.

It has to be mentioned that validity of the given correlation is at the moment only demonstrated for the considered range of flow conditions and geometry. The modified correlation describes the wall and total temperature effects within the investigated variation of the Mach number (2 - 3.5) and the Reynolds number ($4.7 \cdot 10^6 - 8.2 \cdot 10^6$). The physical phenomena that lead to the observed wall temperature effect are also present at other flow conditions and geometries, so similar behavior of the flow field is expected there as well.

The width effect on the shock train was investigated with an additional model. Two influences on the shock train pressure distribution were identified. The additional compression by the sidewalls leads to shorter shock trains. This effect is dominant in the range between an ideal two dimensional case and the width where the weak shocks from the sidewalls reach the centerline of the isolator at the isolator exit plane. The ratio between incoming enthalpy and energy losses is decreasing with decreasing isolator width. This effect is dominant for smaller width to height ratios leading to longer shock trains.

7 References

- [1] Anderson, J., Hypersonic and High Temperature Gas Dynamics, 2nd edition, AIAA ISBN-1-56347-459-X, 2006
- [2] Anderson, J., Modern Compressible Flow – With historical perspective, 3rd edition, McGraw-Hill, New York, 2003
- [3] Arai, T., Sugiyama, H., Abe, F., Takahashi, T., Internal Structure of Pseudo-Shock Waves in a Square Duct, 17th International Symposium on Shock Waves and Shock Tubes, Bethlehem, USA, 1989
- [4] Bemet, D.A., Stevens, J.R., Thompson, M.W., Measured Operating Characteristics of a Rectangular Combustor/Inlet Isolator, AIAA Paper 90-2221, AIAA/SAE/ASME/ASEE 26th Joint Propulsion Conference, Orlando, USA, 1990
- [5] Billig, F.S., Combustion Processes in Supersonic Flow, Journal of Propulsion and Power, Vol.4, No. 3, 1988, pp. 209-216
- [6] Billig, F.S., Research on Supersonic Combustion, Dryden Lectureship on Research, Journal of Propulsion and Power, Vol. 9, No. 4, 1993, pp. 499-514
- [7] Blasius, H., Grenzschichten in Flüssigkeiten mit kleiner Reibung, Zeitschrift für Mathematik und Physik, Nr. 56, 1908, pp. 1-31
- [8] Bleilebens, M., Einfluss der erhöhten Wandtemperatur auf die Stoß/Grenzschicht-Wechselwirkung an einer Rampe im Hyperschall, Doctoral Thesis, RWTH Aachen University, Shaker Verlag, ISBN 3-8322-3822-3811-5, Aachen, 2005
- [9] Bleilebens, M., Olivier, H., On the Influence of Elevated Surface Temperatures on Hypersonic Shock Wave/Boundary Layer Interactions at a Heated Ramp Model, Shock Waves, Vol. 15, No. 5, 2006, pp. 301-312
- [10] Boyce, R., Tirtley, S., Brown, L., Creagh, M., Ogawa, H., SCRAMSPACE: Scramjet-based Access-to-Space Systems, 17th International Space Planes and Hypersonic Systems and Technologies Conference, San Francisco, USA, 2011
- [11] Carroll, B., Dutton, J.C., Characteristics of Multiple Shock Wave/Turbulent Boundary-Layer Interactions in Rectangular Ducts, Journal of Propulsion and Power, Vol. 6, No. 2, 1990, pp. 186-193
- [12] Carroll, B., Dutton, J.C., Multiple Normal Shock Wave/Turbulent Boundary-Layer Interactions, Journal of Propulsion and Power, Vol. 8, No. 2, 1992, pp. 441-448

- [13] Chapman, D. R., Kuehn, D. M., Larson, H. K., Investigation of Separated Flows in Supersonic and Subsonic Streams with Emphasis on the Effect of Transition, NASA Report Nr. 1356, 1956
- [14] Corpening, G.P., Van Wie, D., Mattes, L., Further Assessment of a Scramjet Inlet Mass Flow Measurement Technique for Use in Hypersonic Pulse Facilities, AIAA Paper 91-0551, 29th Aerospace Sciences Meeting, Reno, USA, 1991
- [15] Crocco, L., One-Dimensional Treatment of Steady Gas Dynamics, in: Fundamentals of Gas Dynamics, Princeton University Press, Princeton, 1958, pp.110-130
- [16] Cuffel, R.F., Back, L.H., Changes in Heat Transfer from Turbulent Boundary Layers Interacting with Shock Waves and Expansion Waves, AIAA Journal, Vol. 8, No. 10, 1970, pp. 1871-1873
- [17] Cuffel, R.F., Back, L.H., Flow and Heat Transfer Measurement in a Pseudo-Shock Region with Surface Cooling, AIAA Journal, Vol. 14, No. 12, 1976, pp. 1716-1722
- [18] Curran, E. T., Murthy, S. N. B., Eds.: Scramjet Propulsion - Progress in Astronautics and Aeronautics, Vol. 189, AIAA, Washington, DC, 2000
- [19] Davis, D.O., Gessner, F.B., Experimental and Numerical Investigation of Supersonic Turbulent Flow Through a Square Duct, AIAA Journal, Vol. 24, No. 9, 1986, pp. 1508-1515
- [20] Delery, J., Marvin, J.G., Shock-Wave Boundary Layer Interactions, AGARDograph No. 280, NATO Advisory Group for Aerospace Research and Development, ISBN 92-835-1519-6, 1986,
- [21] Doebelin, E.O., Measurement Systems – Application and Design, McGraw-Hill Publishing Company, New York , 1990
- [22] Dolvin, D.J., Hypersonic International Flight Research and Experimentation (HIFIRE) Fundamental Sciences and Technology Development Strategy, AIAA Paper 2008-2581, 15th AIAA International Space Planes and Hypersonic Systems and Technologies Conference, Dayton, USA, 2008
- [23] Eckert, E.R.G., Engineering Relations of Friction and Heat Transfer to Surfaces in High Velocity Flows, Journal of Aeronautical Sciences, Vol. 22, Nr. 8, 1955, pp. 585-587
- [24] Emami, S., Trexler, C.A., Auslender, A.H., Weidner, J.P., Experimental Investigation of Inlet-Combustor Isolators for a Dual-Mode Scramjet at a Mach Number of 4, NASA Technical Paper 3502, 1995
- [25] Esser, B., Die Zustandsgrößen im Stoßwellenkanal als Ergebnisse eines exakten Riemannlösers, Doctoral thesis, RWTH Aachen Universität, 1992

- [26] Fejer, A., Hearth, G., Driftmyer, R., An Investigation of Constant Area Supersonic Flow Diffusion, Aerospace Research Laboratories Report, ARL 64-81, 1964
- [27] Fotia, M.L., Driscoll, J.F., Ram-Scram Transition and Flame/Shock-train Interactions in a Model Scramjet Experiment, Journal of Propulsion and Power, Vol. 29, No. 1, 2013, pp. 261-273
- [28] Gaisbauer, U., Weigand, B., Research Training Group GRK 1095/2: "Aero-Thermodynamic Design of a Scramjet Propulsion System", ISABE 2011-1512, 20th ISABE Conference, Gothenburg, Sweden, 2011
- [29] Grönig H., Olivier H.: Experimental Hypersonic Flow Research in Europe, JSME Int. J., Series B, Vol. 41, No. 2, 1998, pp. 397-407
- [30] Häberle, J., Gülhan, A., Investigation of Two-Dimensional Scramjet Inlet Flowfield at Mach 7, Journal of Propulsion and Power, Vol. 24, No. 3, 2008, pp. 446-459
- [31] Häberle, J., Untersuchungen zum externen und internen Strömungsfeld eines Scramjet Triebwerkseinlaufs bei unterschiedlichen Betriebspunkten, Doctoral thesis, Universität Stuttgart, DLR Forschungsbericht 2009-14, ISSN 1434-8454, Cologne, 2009
- [32] Habermann, M., Erzeugung von Hochenthalpieströmungen im Hyperschall mit einem detonationsgetriebenen Stoßwellenkanal, Doctoral thesis, RWTH Aachen University, Shaker Verlag, ISBN 3-8265-8788-X, Aachen, 2000
- [33] Heiser, W.H., Pratt, D.T., Hypersonic Airbreathing Propulsion, AIAA Education Series, Fifth Printing, Washington D.C., 1993
- [34] Hennecke, D. K., Wörrlein, K., Vorlesungsskript Flugantriebe und Gasturbinen, Technische Universität Darmstadt, 2. Auflage WS 2000 / 2001
- [35] Heufer, K. A., Untersuchung zur Filmkühlung in supersonischen Strömungen, Doctoral thesis, RWTH Aachen University, Shaker Verlag, ISBN 978-3-8024-6, Aachen, 2008
- [36] Hirschel, E., H., Basics of Aerothermodynamics, Volume 204 Progress in Astronautics and Aeronautics, Springer Verlag, Heidelberg, 2005
- [37] Hohn, O., Gülhan, A., Experimental Investigation on the Influence of Sidewall Compression on the Flowfield of a Scramjet Inlet at Mach 7, AIAA Paper 2011-2350, 48th AIAA Aerospace Sciences Meeting, Orlando, USA, 2011
- [38] Hohn, O., Gülhan, A., Experimental Investigation on the Influence of Yaw Angle on the Inlet Performance at Mach 7, AIAA Paper 2010-938, 17th AIAA Space Planes Conference and Hypersonic Systems Conference, San Francisco, USA, 2011

- [39] Holden, M.S., Boundary-Layer Displacement and Leading-Edge Bluntness Effects on Attached and Separated Laminar Boundary Layers in a Compression Corner. Part II: Experimental study, AIAA Journal Vol. 9, No. 1, 1971, pp. 84-93
- [40] Honke, M., Messung des Massenstroms im Stosswellenkanal, Diploma thesis, Shock Wave Laboratory, RWTH Aachen University, 2010
- [41] Ikui, T., Matsuo, K., Nagai, M., The Mechanism of Pseudo-Shock Waves. Bulletin of JSME, Vol. 17, Nr. 108, 1974, pp. 731-739
- [42] Ikui, T., Matsuo, K., Sasaguchi, K., Modified Diffusion Model of Pseudo-Shock Waves Considering Upstream Boundary Layers, Bulletin of JSME, Vol. 24, No. 197, 1981, pp. 1920-1927
- [43] Jessen, C., Messung von Druck, Temperatur und Kraft an Modellen im Stoßwellenkanal, Doctoral thesis, RWTH Aachen Universität, Verlag Mainz, ISBN 3-925714-99-5, Aachen, 1993
- [44] Katzer, E., On the Length Scales of Laminar Shock/Boundary-Layer Interaction, Journal of Fluid Mechanics, Vol. 206, 1989, pp. 477-496
- [45] Kindl, H., Einfluss des Treibergases auf die Strömung in Stoßwellenkanälen, Doctoral thesis, RWTH Aachen University, Fahner, ISBN 3-00-007695-6, Lauf a.d. Pegnitz, 2001
- [46] Kleine, H., Verbesserung optischer Methoden für die Gasdynamik, Doctoral Thesis, RWTH Aachen University, 1993
- [47] Kulite Semiconductor Products, Inc, Internet: <http://www.kulite.com/docs/products/XCQ-080.pdf>, 2011
- [48] Langtry, R.B., Menter, F.R., Likki, S.R., Suzen, Y.B., Huang, P.G., Völker, S., A Correlation Based Transition Model Using Local Variables Part 2 – Test Cases and Industrial Applications ASME-GT2004-53454, ASME TURBO EXPO 2004, Vienna, Austria, 2004
- [49] Le, D.B., Goyne, C.P., Kraus, R.H., McDaniel, J.C., Experimental Study of a Dual-Model Scramjet Isolator, AIAA Paper 2005-23, 43rd AIAA Aerospace Sciences Meeting and Exhibit, Reno, USA, 2005
- [50] Lewis, J.E., Kubota, T., Lees, T., Experimental Investigation of Supersonic Laminar, Two-Dimensional Boundary Layer Separation in a Compression Corner With and Without Cooling, AIAA Journal, Vol. 6, No. 1, 1968, pp. 7-14

- [51] Lin, K.C., Tam, C.J., Jackson, K.R., Eklund, D.R., Jackson, T.A., Characterization of Shock Train Structures inside Constant-Area Isolators of Model Scramjet Combustors, AIAA Paper Nr. 2006-816, 44th AIAA Aerospace Sciences Meeting and Exhibit, Reno, USA, 2006
- [52] Lin, K.C., Tam, C.J., Eklund, D.R., Jackson, K.R., Jackson, T.A., Effects of Temperature and Heat Transfer on Shock Train Structures inside Constant-Area Isolators, AIAA Paper Nr. 2006-817, 44th AIAA Aerospace Sciences Meeting and Exhibit, Reno, USA, 2006
- [53] Lin, P., Rao, G.V.R., O'Connor, G.M., Numerical Investigation on Shock Wave/Boundary layer Interaction in a Constant Area Diffuser at Mach 3, AIAA Paper 1991-1766, AIAA 22nd Fluid Dynamics, Plasma Dynamics and Lasers Conference, Honolulu, USA, 1991
- [54] Lin, P., Rao, G.V.R., O'Connor, G.M., Numerical Analysis of Normal Shock Train in a Constant Area Isolator, AIAA Paper 1991-2162, AIAA/SAE/ASME/ASME 27th Joint Propulsion Conference, Sacramento, USA, 1991
- [55] Lin, P., Geometric Effects on Precombustion Shock Train in Constant Area Isolators, 29th AIAA/SAE/ASME/ASME Joint Propulsion Conference and Exhibit, Monterey, USA, 1993
- [56] Mateer, G.G., Brosh, A., Viegas, J.R., A Normal Shock-Wave Turbulent Boundary-Layer Interaction at Transonic Speeds, AIAA Paper 1976-161, AIAA 14th Aerospace Sciences Meeting, Washington D.C., USA, 1976
- [57] Mateer, G.G., Viegas, J.R., Mach and Reynolds Number Effects on a Shock-Wave/Boundary-Layer Interaction, AIAA Journal, Vol. 18, No. 8, 1980, pp. 1016-1018
- [58] Matsuo, K., Miyazato, Y., Kim, H.D., Shock Train and Pseudo-Shock Phenomena in Internal Gas Flows, Progress in Aerospace Sciences 35, 1999, pp. 33-100
- [59] McClinton, C.R., X-43-Scramjet Power Breaks the Hypersonic Barrier Dryden Lectureship in Research for 2006, AIAA 2006-1, 44th AIAA Aerospace Sciences Meeting, Reno, USA, 2006
- [60] Menter, F.R., Two-Equation Eddy Viscosity Turbulence Models for Engineering Applications, AIAA Journal, Vol. 32, No. 8, 1994, pp. 1598-1605
- [61] Menter, F.R., Langtry, R.B., Likki, S.R., Suzen, Y.B., Huang, P.G., Völker, S., A Correlation Based Transition Model Using Local Variables Part 1- Model Formulation ASME-GT2004-53452, ASME TURBO EXPO 2004, Vienna, Austria, 2004

- [62] Naidu., D.S., Buffington, J.L., Banda, S.S., Resurrection in Hypersonics: Why, What and When, AIAA-1999-4053, AIAA Guidance, Navigation and Control Conference, Portland, USA, 1999
- [63] Nau, M., Elektrische Temperaturmessung mit Thermoelementen und Widerstandthermometern, FAS 146, ISBN 978-3-935742-06-1, JUMO GmbH, Fulda, 2007
- [64] Neuenhahn, T., Olivier, H., Influence of the Wall Temperature and the Entropy Layer Effects on Double Wedge Shock Boundary Layer Interactions, AIAA Paper 2006-8136, 14th AIAA/AHI Space Planes and Hypersonic Systems and Technologies Conference, Canberra, Australia, 2006
- [65] Neuenhahn T, Olivier H, Numerical study of wall temperature and entropy layer effects on transitional double wedge shock wave/boundary layer interactions, Proc. of the 26th Int. Symposium on Shock Waves, Göttingen, Germany, July 2007, Vol. 1, pp. 683-688
- [66] Neuenhahn, T., Olivier, H., Influence of Temperature Effects on the Shock Wave/Boundary Layer Interaction of a Scramjet-Inlet, DGLR - Jahrestagung, Darmstadt, Germany, 2008
- [67] Neuenhahn, T., Investigation of the Shock Wave/Boundary Layer Interaction of Scramjet Intake Flows, Doctoral thesis, RWTH Aachen University, Shaker Verlag, ISBN 978-3-8322-9187-7, Aachen, 2010
- [68] Neumann E.P., Lustwerk, F., Supersonic Diffusers for Wind Tunnels, Journal of Applied Mechanics, Vol. 16, Nr. 2, 1949, pp. 195-202
- [69] Nguyen, T., Schieffer, G., Fischer, C., Olivier, H., Behr, M., Reinartz, B., Details of Turbulence Modeling in Numerical Simulations of Scramjet Intake, 27th Congress of the Int. Council of the Aeronautical Sciences (ICAS 2010), Nice, France, September 2010, Red Hook, NY: Curran, Vol. 1-5, 603-606
- [70] Nill, L.D., Mattick, A.T., An Experimental Study of Shock Structure in a Normal Shock Train, AIAA Paper 1999-799, AIAA 34th Aerospace Sciences Meeting and Exhibit, Reno, USA, 1996
- [71] Om, D., Childs, M.E., An Experimental Investigation and a Numerical Prediction of a Transsonic Normal Shock Wave/Turbulent Boundary Layer Interaction, AIAA Paper 1982-990, AIAA/ASME 3rd Joint Thermophysics, Fluids, Plasma and Heat Transfer Conference, St. Louis, USA, 1982
- [72] Om, D., Childs, M.E., Multiple Transsonic Shock-Wave/Turbulent Boundary-Layer Interaction in a Circular Duct, AIAA Journal, Vol. 23, No. 10, 1985, pp. 1506-1511

- [73] Olivier, H., An Improved Method to Determine Free Stream Conditions in Hypersonic Facilities, *Shock Waves*, Vol. 3, No. 2, 1993, pp. 129-139
- [74] Olivier, H., Zechner, M., Slender Conical Nozzle, MSTP Progress Report, HT-SF-E34-731-RWTH, Shock Wave Laboratory, RWTH Aachen University, 1996
- [75] Olivier, H., Jiang, Z., Yu, H.R., Lu, F., Detonation-Driven Shock Tubes and Tunnels, in: F. Lu, D. Marren (eds.) *Advanced Hypersonic Test Facilities*, Progress in Astronautics and Aeronautics, AIAA Inc., Vol. 198, 2002, pp. 135-203
- [76] Olivier, H., Personal Communication, 2012
- [77] Prandtl, L., Über die Flüssigkeitsbewegung bei sehr kleiner Reibung, *Proceedings of the Third International Mathematical Congress.*, Heidelberg; Germany, 1904, pp. 484-494
- [78] Reynolds, O., An Experimental Investigation of the Circumstances which Determine Whether the Motion of Water Shall be Direct or Sinuous, and the Law of Resistance in Parallel Channels, *The Philosophical Transactions of the Royal Society*, 1883
- [79] Rizetta, D. P., Burggraf, O. R., Jenson, R., Triple-Deck Solutions for Viscous Supersonic and Hypersonic Flow Past Corner, *Journal of Fluid Mechanics*, Vol. 89, Part 3, 1978, pp. 535-55
- [80] Schlichting, H., *Boundary-Layer Theory*, 7th edition, McGraw-Hill, New York, 1979
- [81] Settles, G.S., *Schlieren and Shadowgraph Techniques – Visualizing Phenomena in Transparent Media*, Springer-Verlag, Berlin, 2001
- [82] Shock Wave Laboratory, Coaxial Thermocouples for Heat Flux Measurement, Internet: <http://www.swl.rwth-aachen.de/en/industry-solutions/thermocouples/>, 2011
- [83] Simeonides, G., Hypersonic Shock Wave Boundary Layer Interactions Over Compression Corners, Doctoral thesis, University of Bristol, Bristol, 1992
- [84] Simeonides, G., Generalized Reference Enthalpy Formulations and Simulation of Viscous Effects in Hypersonic Flow. *Shock Waves*, Vol. 8, 1998, pp. 161-172
- [85] Smart, M., Hass, N., Paul, A., Flight Data Analysis of the HYShot 2 Scramjet Flight Experiment, *AIAA Journal*, Vol. 44, No. 10, 2006, pp. 2366-2375
- [86] Stockbridge, R.D., Experimental Investigation of Shock Wave/Boundary-Layer Interactions in an Annular Duct, *Journal of Propulsion and Power*, Vol. 5, No. 3, 1989, pp. 346-352
- [87] Sullins, G., McLafferty, G., Experimental Results of Shock Trains in Rectangular Ducts, AIAA Paper 92-5103, AIAA Fourth International Aerospace Planes Conference, Orlando, USA, 1992

- [88] Sugiyama, H., Arai, T., Uno, N., Takahashi, T., Three-Dimensional Structure of Pseudo-Shock Waves in a Rectangular Duct, 18th International Symposium on Shock Waves, Sendai, Japan, 1991
- [89] Sun, L.Q., Sugiyama, H., Mizobata, K., Fukuda, K., Numerical and Experimental Investigations on the Mach 2 Pseudo-Shock Wave in a Square Duct, *Journal of Visualization*, Vol. 6, No. 4, 2003, pp. 363-370
- [90] Tamaki, T., Tomita, Y., Yamane, R., A Study of Pseudo-Shock (1st Report, λ -Type Pseudo-Shock), *Bulletin of the JSME*, Vol. 13, No. 55, 1970, pp. 51-58
- [91] Tamaki, T., Tomita, Y., Yamane, R., A Study of Pseudo-Shock (2nd Report, x-Type Pseudo-Shock), *Bulletin of the JSME*, Vol. 14, No. 74, 1971, pp. 807-817
- [92] Triesch, K., Krohn, E., Verwendung von Kegeldüsen zur Drosselung und Durchsatzmessung bei Überschalleinläufen, DFVLR Bericht IB-39113-83-A-04, 1984
- [93] United States Air Force, X-51A Waverider Printabe Fact Sheet, Internet: www.af.mil/information/factsheets/factsheet.asp?fsID=17986
- [94] Van Driest, E.R., Turbulent Boundary Layers in Compressible Flow, *Journal of the Aeronautical Sciences*, Vol. 18, No. 3, 1951, pp. 145-160
- [95] Van Driest, E.R., Investigation of Laminar Boundary Layer in Compressible Fluids Using the Crocco Method, NACA TN2597, 1952
- [96] Van Wie, D., Corpening, G., Mattes, L., Carpenter, D., Mölder, S., McGreor, R., An Experimental Technique for the Measurement of Mass Flow of Scramjet Inlets Tested in Hypersonic Pulse Facilities, AIAA Paper 89-2331, 25th Joint Propulsion Conference, Monterey, USA, 1989
- [97] Van Wie, D., Techniques for the Measurement of Scramjet Inlet Performance at Hypersonic Speeds, AIAA Fourth International Aerospace Planes Conference, Orlando, USA, 1992
- [98] Vellaramkalayil, J.J., Schermann, T., von Wolfersdorf, J., Analysis of a Two-Stage Supersonic Combustion Chamber using Experiments and Simulations, AIAA Paper 2011-2247, 17th AIAA International Space Planes and Hypersonic Systems and Technologies Conference, San Francisco, USA, 2011
- [99] Vetter, M., Hyperschallumströmungen von Modellen im Stoßwellenkanal, Doctoral thesis, RWTH Aachen University, Verlag Mainz, ISBN 3-925714-81-2, Aachen, 1996
- [100] Waltrup, P.J., Billig, F.S., Precombustion Shock Structures in Scramjet Engines, AIAA Paper 1972-1181, AIAA/SAE 8th Joint Propulsion Specialist Conference, New Orleans, USA, 1972

- [101] Waltrup, P.J., Billig, F.S., Structure of Shock Waves in Cylindrical Ducts, AIAA Journal, Vol. 11, Nr. 10, 1973, pp. 1404-1408
- [102] Waltrup, P.J., Billig, F.S., Prediction of Precombustion Wall Pressure Distributions in Scramjet Engines, Journal of Spacecraft and Rockets, Vol. 10, No. 9, 1973, pp. 620-622
- [103] Weigand, B., Gaisbauer, U., An Overview on the Structure and Work of the DFG Research Training Group GRK 1095: "Aero-Thermodynamic Design of a Scramjet Propulsion System", AIAA Paper 2009-7276, 16th AIAA/DLR/DGLR Space Planes Conference and Hypersonic Systems Conference, Bremen, Germany, 2009
- [104] Wierzbanski, T., Kasten, T., Manned Versus Unmanned: The Implications To NASP, AIAA-90-5265, AIAA Second International Aerospace Planes Conference, Orlando, USA, 1990
- [105] Zhao, H., Development of a Mass Flux Probe Used in Short Duration Hypersonic Flow, Doctoral thesis, RWTH Aachen University, Shaker Verlag, ISBN – 3-8265-3720-2, Aachen, 1998

8 Appendix

Here the wall pressure distributions measured with the heatable isolator model but not included in chapter 5.2.3 are shown.

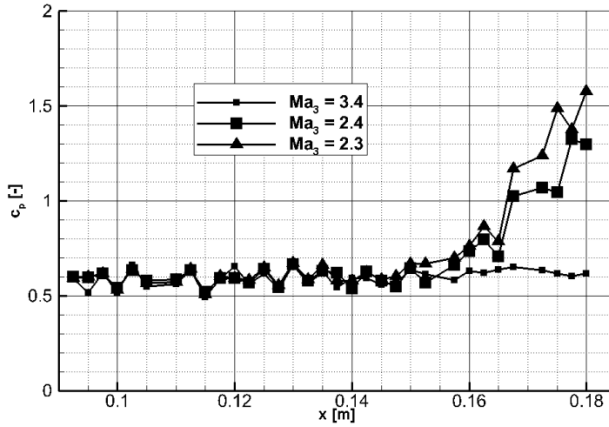


Fig. 8-1: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.5$, $T_w = 300\text{ K}$

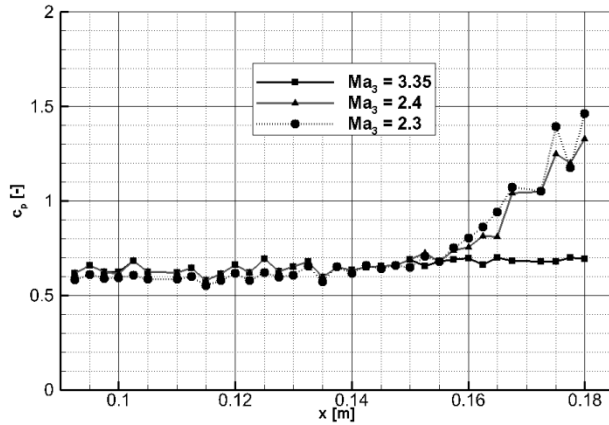


Fig. 8-2: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.5$, $T_w = 600\text{ K}$

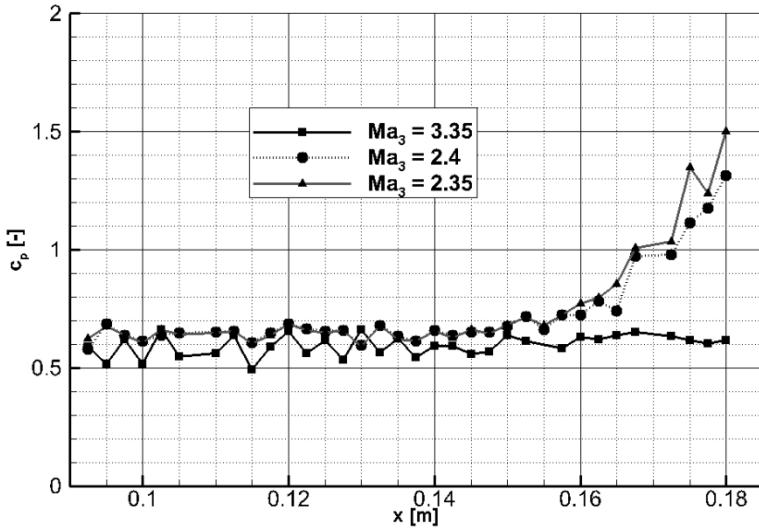


Fig. 8-3: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.5$, $T_w = 800$ K

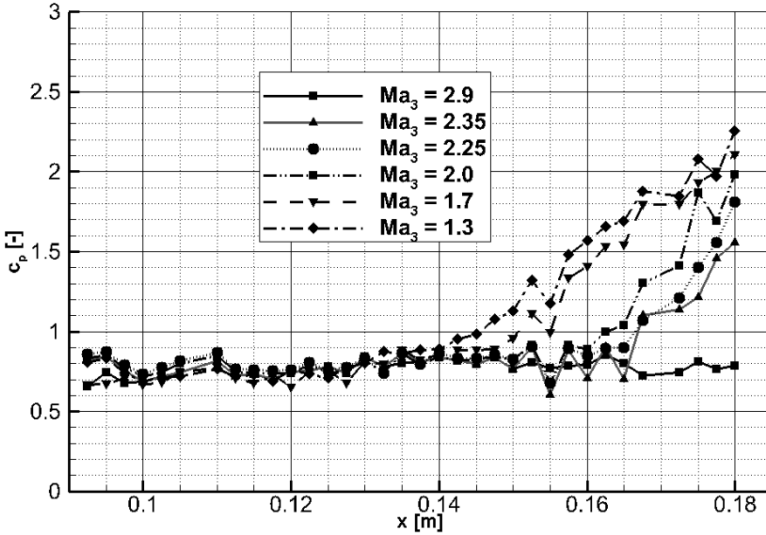


Fig. 8-4: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.0$, $T_w = 300$ K

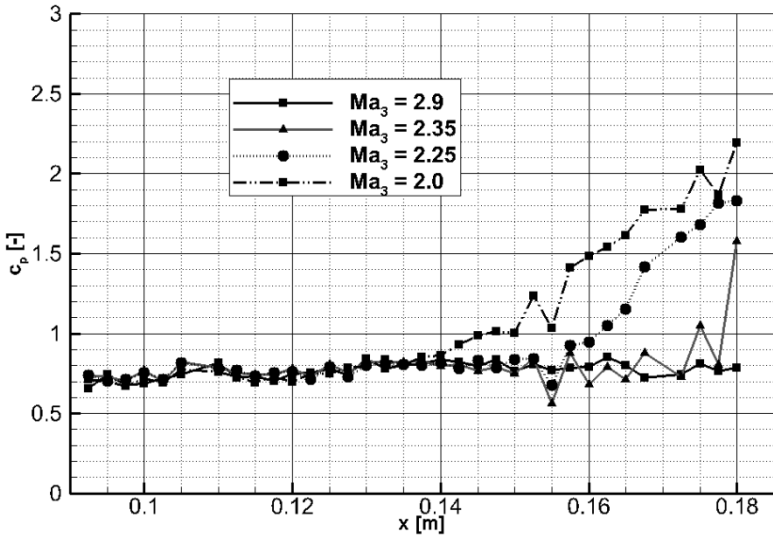


Fig. 8-5: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.0$, $T_w = 600\text{ K}$

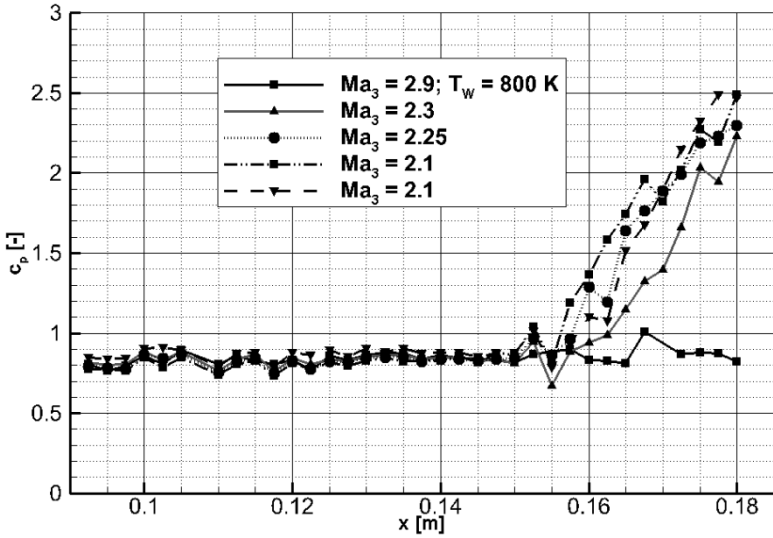


Fig. 8-6: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 3.0$, $T_w = 1000\text{ K}$

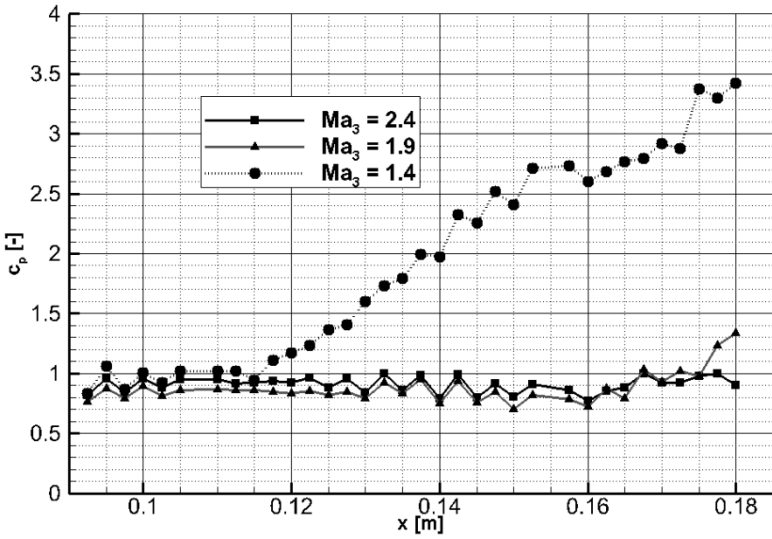


Fig. 8-7: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 2.5$, $T_w = 300$ K

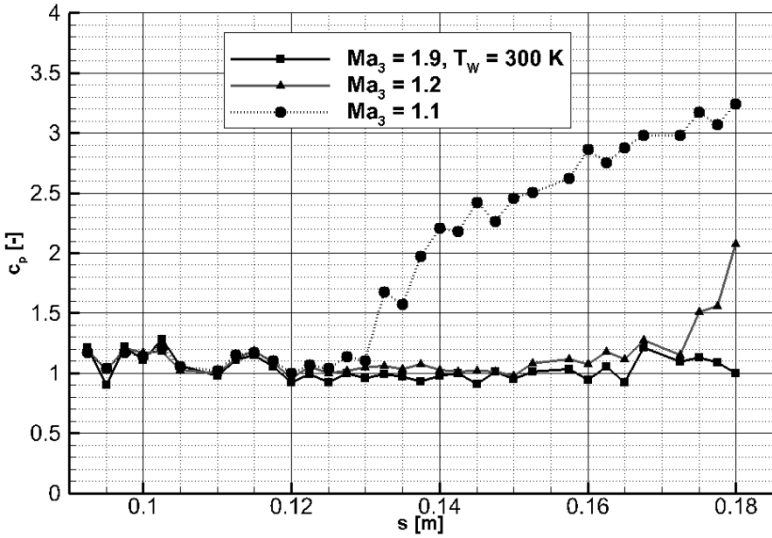


Fig. 8-8: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 2.0$, $T_w = 600$ K

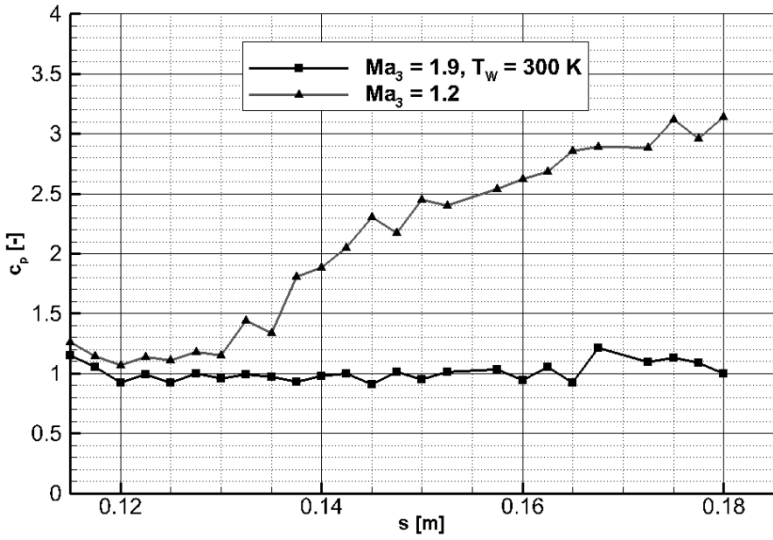


Fig. 8-9: Wall pressure distribution for different back pressure at Condition I, $Ma_2 = 2.0$, $T_w = 800$ K

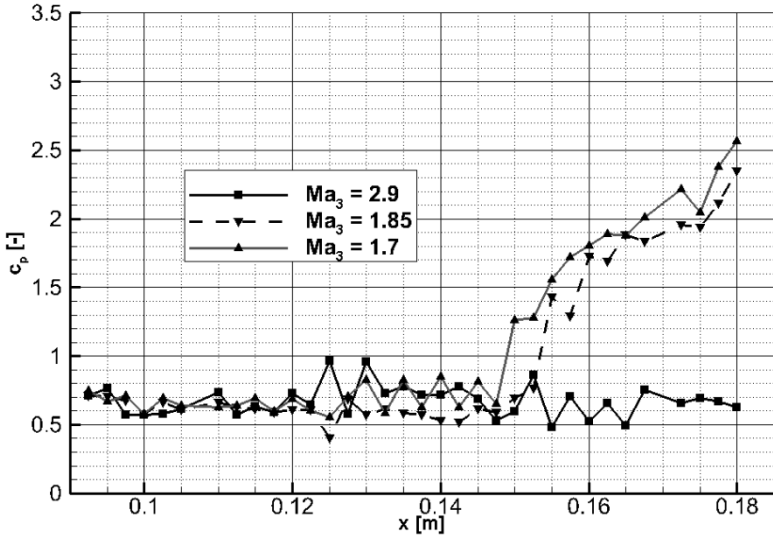


Fig. 8-10: Wall pressure distribution for different back pressure at Condition II, $Ma_2 = 3.0$, $T_w = 600$ K

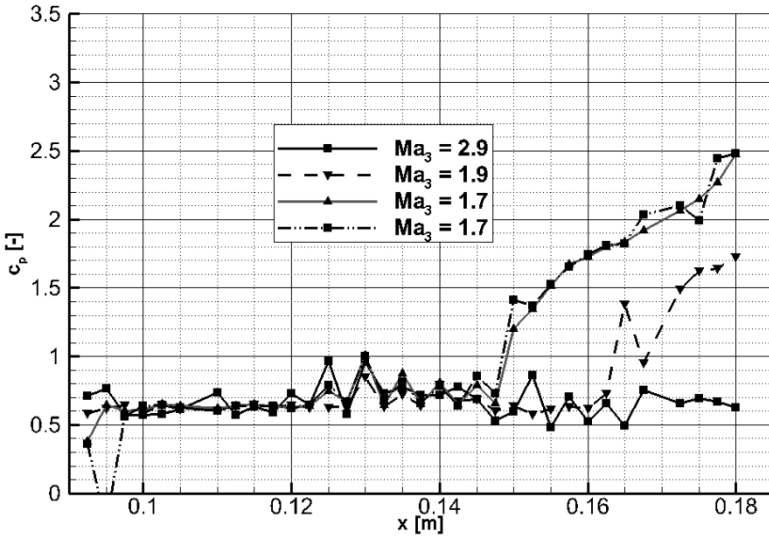


Fig. 8-11: Wall pressure distribution for different back pressure at Condition II, $Ma_2 = 3.0$, $T_w = 800$ K

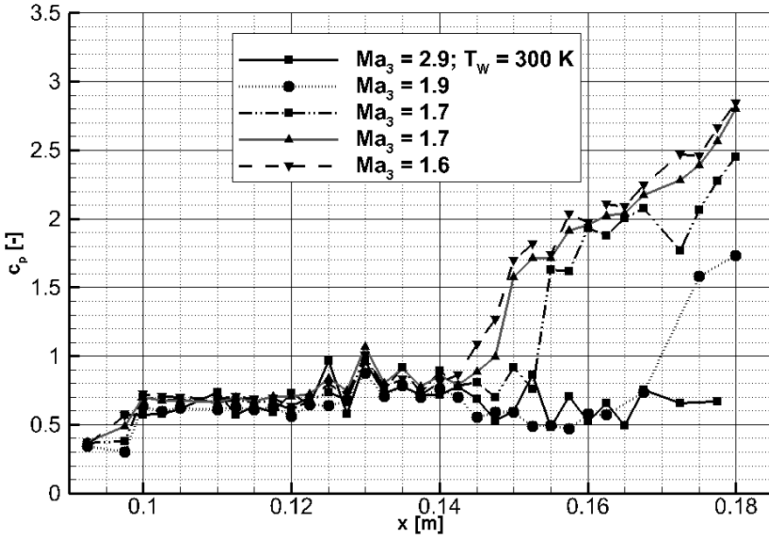


Fig. 8-12: Wall pressure distribution for different back pressure at Condition II, $Ma_2 = 3.0$, $T_w = 1000$ K