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Prediction and Validation of the Aerodynamic Effects of Simulated Battle Damage on Aircraft Wings

by

Thomas W. Pickhaver

DOCTORAL THESIS

Submitted in partial fulfilment of the requirements for the award of

Doctor of Philosophy

of Loughborough University

May 2014

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Abstract

Aerodynamic analysis is an important area of survivability studies. There is a desire to be able to predict the aerodynamic effects of a given damage scenario on an aircraft wing with minimal wind tunnel testing or computational simulations. Due to the limited nature of previous studies, this has not generally been possible. The original contribution of this thesis is a predictive technique developed to estimate the aerodynamic effects of a simulated battle damage hole on an aircraft wing, resulting from a range of attack directions. This technique was successfully validated against experimental data.

Testing under two-dimensional conditions was undertaken on a NASA LS(1)-0417MOD aerofoil at a Reynolds number of 500,000. This project simulates the effect of attack direction by varying the offset between upper and lower surface damage holes in both chordwise and spanwise directions. Damage was modelled using circular holes. Lift, drag and pitching moment coefficients were measured and supplemented with surface flow visualisation and surface pressure measurements. Coefficient increments, defined as the difference between the damage cases and a datum undamaged case were used to quantify the effects of the damage, with the performance qualified in terms of weak and strong jets. Weak jets were found to have little effect on the flow and aerodynamic properties, while strong jets caused significant disruption. The effects increased in magnitude with hole size, incidence and proximity of the upper surface hole to the pressure peak. Spanwise offset on the holes had little effect on the jet strength but introduced asymmetry into the surface flow. This effect was found to be due to the behaviour of the flow within the cavity.

Three-dimensional testing was undertaken at a Reynolds number of 1,000,000 on a half wing model in order to investigate any changes in the aerodynamic characteristics of the damage when applied to a more representative aircraft wing. The higher Reynolds number exploited the larger wind tunnel working section and provided a value more representative of typical unmanned aerial vehicles. As the damage was moved towards the tip its effects were lessened and the transition from weak jet to strong jet delayed. Spanwise pressure variation from the tip also introduced asymmetry into the jet’s surface flow features.

Plotting coefficient increments for all attack directions against the pressure coefficient difference between upper and lower surfaces from an undamaged wing, across the equivalent damage hole region highlighted significant trends, which were used as the basis of a predictive technique for a range of hole sizes and attack directions. The validity of the technique was assessed by predicting a previously untested damage case and comparing it against subsequent wind tunnel tests. The results from this validation proved encouraging.

Keywords: Aerodynamics; Battle damaged wings; Wind tunnel testing; Survivability; Predictive technique; Two dimensional testing; Three dimensional testing
Acknowledgements

I would like to extend my thanks and gratitude to my PhD supervisor, Dr. Peter Render for his continued support, help and advice throughout this project. Without his input and guidance this project would not have been possible. Thanks are also extended to Dr. Andrew Irwin and Mark Lucking from BAE Systems who provided very useful feedback, comments and suggestions on all aspects of the project. In particular, Andrew’s previous experience on battle damaged aerodynamic studies, and Mark’s feedback and critique of the predictive methodology proved invaluable.

In addition, thanks are also extended to Rob Hunter, Stacey Prentice and Peter Stinchcombe for their help and expertise in designing, manufacturing and maintaining the models. Their assistance in setting up, operating and maintaining the wind tunnels proved invaluable throughout the course of all experimental testing. The help and flexibility of David Britton cannot go without thanks, for the prompt turnaround of requests to laser cut the removable skin panels. Finally, thanks are passed on to Chris Harvey for his advice and help on using the \LaTeX typesetting package during creation of customer reports, conference and journal papers and this thesis.

Without funding from BAE Systems and the European BaToLUS (Battle damage Tolerance for Lightweight UAV Structures) project there would not have been the resources available to achieve the full scope of this project. This unique opportunity allowed for the work to be shared with aircraft survivability experts throughout Europe and to receive validation of key results using resources from other partners in the consortium, in particular the computational fluid dynamics studies from ONERA.

Finally, but by no means least, special thanks and extreme gratitude and appreciation must be extended to my parents and family for their continued support and motivation throughout the duration of this PhD. This helped keep me motivated throughout, and in particular during writing up this thesis.
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Nomenclature

$A$  
Cross sectional area of the aerofoil

$a_0$  
Two-dimensional lift curve slope

$a_{3D}$  
Three-dimensional lift curve slope

$AR$  
Aspect ratio

$b$  
Wing span

$b_t$  
Wind tunnel working section width

$c$  
Wing chord

$\bar{c}$  
Geometric mean chord

$\bar{\bar{c}}$  
Mean aerodynamic chord

$c'$  
Chord normal to the leading edge

$C$  
Wind tunnel working section cross sectional area

$C_d, C_D$  
Drag coefficient

$C_{d0}, C_{D0}$  
Uncorrected drag coefficient at the zero lift incidence

$L, C_L$  
Lift coefficient

$C_m, C_M$  
Pitching moment coefficient

$C_p, C_P$  
Pressure coefficient

$C_{p, \text{hole}}$  
Area weighted pressure coefficient across a damage hole on an equivalent undamaged wing

$d C_d, d C_D$  
Drag coefficient increment due to damage

$d C_{dp}$  
Two-dimensional drag coefficient increment prediction

$d C_{DA}$  
Drag area coefficient increment

$d C_l, d C_L$  
Lift coefficient increment due to damage
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<td>$dC_{LA}$</td>
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<td>$dC_m, dC_M$</td>
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<td>$dC_{mp}$</td>
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<tr>
<td>$dC_{MA}$</td>
<td>Pitching moment area coefficient increment</td>
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<td>$dC_p, dC_P$</td>
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<tr>
<td>$D_{pred}$</td>
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<tr>
<td>$M$</td>
<td>Mach number</td>
</tr>
<tr>
<td>$p_{static}$</td>
<td>Static pressure</td>
</tr>
<tr>
<td>$p_{tap}$</td>
<td>Tapping pressure</td>
</tr>
<tr>
<td>$R$</td>
<td>Hole radius</td>
</tr>
<tr>
<td>$Re$</td>
<td>Reynolds Number</td>
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<tr>
<td>$s$</td>
<td>Wing span</td>
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<tr>
<td>$S$</td>
<td>Wing area</td>
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<td>$t$</td>
<td>Aerofoil thickness</td>
</tr>
<tr>
<td>TE</td>
<td>Trailing edge</td>
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</table>
### Nomenclature

- **V**: Free stream velocity
- **V_w**: Effective volume of wing
- **x**: Chordwise station
- **y_s**: Ordinate of the camber line of the aerofoil at a given chordwise station
- **y_u**: Ordinate of the upper surface of the aerofoil at a given chordwise station

### Fractions

- \( \frac{dC_D}{dC_L^2} \): Gradient of the straight line region of a plot of \( C_D \) against \( C_L^2 \)
- \( \frac{c}{H} \): Model chord to wind tunnel working section height ratio
- \( \frac{t}{c} \): Thickness to chord ratio

### Greek

- \( \alpha \): Incidence
- \( \beta \): Prandtl-Glaurert compressibility parameter
- \( \delta_{av} \): Mean value for upwash interference parameter over the wing planform
- \( \delta_D \): Upwash interference parameter relating to \( \Delta \alpha_D \)
- \( \delta_L \): Upwash interference parameter relating to \( \Delta \alpha_L \)
- \( \Delta A_n \): Area of a small square, \( n \), on the damage hole
- \( \Delta C_{p, n} \): Pressure coefficient value at the centre of a square, \( n \)
- \( \Delta C_d \): Incremental correction to the drag coefficient
- \( \Delta C_l \): Incremental correction to the lift coefficient
- \( \Delta C_m \): Incremental correction to the pitching moment coefficient
- \( \Delta C_p \): Incremental correction to pressure coefficients
- \( \Delta \alpha \): Incidence increment
<table>
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<td>$\Delta \bar{\alpha}$</td>
<td>Incidence increment due to lift interference</td>
</tr>
<tr>
<td>$\Delta \alpha_D$</td>
<td>Incidence correction relating to three-dimensional drag</td>
</tr>
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<td>$\Delta \alpha_L$</td>
<td>Incidence correction relating to three-dimensional lift</td>
</tr>
<tr>
<td>$\varepsilon_b$</td>
<td>Total blockage correction factor</td>
</tr>
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<td>$\varepsilon_s$</td>
<td>Solid blockage correction factor</td>
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<tr>
<td>$\varepsilon_{s0^\circ}$</td>
<td>Blockage correction at $0^\circ$ incidence</td>
</tr>
<tr>
<td>$\varepsilon_w$</td>
<td>Wake blockage correction factor</td>
</tr>
<tr>
<td>$\theta$</td>
<td>Angular co-ordinate at a specific point on an aerofoil</td>
</tr>
<tr>
<td>$\Lambda$</td>
<td>Angle of sweep</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Air density</td>
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<tr>
<td>$\tau$</td>
<td>Factor depending on wind tunnel working section shape</td>
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**Subscripts**

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<td>$2D$</td>
<td>Parameter from a wing tested under two-dimensional conditions</td>
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<tr>
<td>$3D$</td>
<td>Parameter from a wing tested under three-dimensional conditions</td>
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<tr>
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<td>Converted coefficient increment from a two-dimensional wing to a three-dimensional wing</td>
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<td>Two-dimensional drag</td>
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<td>$D$</td>
<td>Three-dimensional drag</td>
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<td>$dam$</td>
<td>Unknown damage case</td>
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<td>Coefficient with damage present</td>
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<tr>
<td>$f$</td>
<td>Final corrected parameter</td>
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<tr>
<td>lower</td>
<td>Parameter from the lower aerofoil surface only</td>
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<td>Three-dimensional pitching moment</td>
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<td>Reference to an incremental square on a damage hole</td>
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<td>undamaged</td>
<td>Coefficient with no damage present</td>
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<td>upper</td>
<td>Parameter from the upper aerofoil surface only</td>
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1.1 Project Overview

This project furthered research undertaken by Irwin [1] in 1993 at Loughborough University for British Aerospace (now BAE Systems). This work has formed part of a European-wide study into the survivability of unmanned aerial vehicles, through the Battle Damage Tolerance for Lightweight UAV Structures (BaToLUS) project. Irwin’s initial work formed the foundation for these studies into the aerodynamic effects of simulated battle damage on aircraft wings. In this context, ‘simulated battle damage’ refers to a consistent and repeatable model of a typical damage hole, and not a hole created by live fire. Irwin studied basic configurations of damage and formed the key principles, which were validated and applied during this project. This project took a more in-depth look at more complex damage cases, and explored damage configurations on an aerofoil not studied by Irwin. The experimental data was then used to develop a methodology that allows the aerodynamic effects of battle damage to be predicted with only a minimal amount of prior experimental or computational testing.

Ultimately the ability to predict the aerodynamic effects and consequences of battle damage would allow for a more robust assessment of aircraft survivability. Previously, such assessments had purely focused on structural survivability. However, as these found that aircraft wings could remain structurally intact with relatively large holes, the question of aerodynamic effects resulting from such a hole became more prominent. Additionally, it was identified that for a given damage size, since tailplanes have a comparatively smaller chord, the trends from Irwin’s work [1] suggested that tailplane damage could prove more significant than a wing. This is due to the potentially larger effect resulting from a small shell (in terms of percentage of lifting area removed). With potential controllability issues resulting from tailplane damage, it was evident more detailed analysis into aerodynamic survivability from battle damage was required. Finally, with a move towards greater use of unmanned aerial vehicles in combat situations, a desire to develop a more damage tolerant aircraft is apparent, since the operator may not be aware of the extent of any potential damage. This would therefore require knowledge of the changes to aerodynamic perfor-
mance and properties of a range of damage cases for a given aerofoil profile.

Understanding the aerodynamic effects resulting from sustained battle damaged requires knowledge of a number of variables in order to model the damage sufficiently. Examples of possible variables that would influence the shape, size and location of damage are summarised below:

- Threat type (e.g. missile, anti-aircraft artillery gunfire).
- Shell calibre/diameter.
- Attack angle, e.g. from below and in front of the wing, causing the hole through the lower surface to be closer to the leading edge than the upper surface, and location of attack on the wing (dictating chordwise and spanwise positions of the damage holes).
- Fragmentation of the shell, causing multiple small damage holes in close proximity.
- Material of the wing: metallic skins typically experience a “petalling” effect where small amounts of the skin material are ‘torn’ or ‘ripped’ up from the surface. These are small in size and random in shape, and will protrude up from the surface into the freestream. Composite skins do not experience this in the same fashion, as the fibre layers tend to delaminate and generate small splinters instead, which can be blown away by the airflow.
- High explosive shells causing different sized entry and exit holes.

Irwin’s work [1] restricted analysis to damage holes perpendicular to the wing chord line (thus, the entry and exit holes were at the same chordwise locations), with varying chordwise centres of the holes and varying hole size (representing different sized shells). Simulated gunfire was used, which combined with the thin nature of the aerofoil used, allowed the assumption to be made that entry and exit holes were of the same diameter. In addition it was assumed no fragmentation occurred, resulting in a single entry and exit hole, and thus greatly simplifying the analysis. No petalling was modelled, as it was assumed this would have a negligible effect due to its small size and irregular nature. The gunfire was assumed to generate a circular hole, which was found to be typical for non-exploding armour-piercing rounds. Irwin identified that the majority of damage occurred from projectiles fired at angles between 70° and 80° of the chord line. However a more realistic damage model would consider change in attack angle across a wider range, both in a chordwise and spanwise direction.

For the present project, using the key potential variables identified above, a sufficiently large data set would be constructed to allow the creation and evaluation of a methodology to predict the aerodynamic effects of a battle damage configuration which had not
been previously tested. The goal of the predictive methodology was to be able to predict the effects of battle damage on an aircraft wing with minimal wind tunnel testing, greatly reducing the requirement for expensive or time consuming wind tunnel testing or Computational Fluid Dynamics (CFD) simulations undertaken on large numbers of damage configurations. Wind tunnel testing would require a large number of models, increasing costs and CFD simulations would be limited by the time taken to generate meshes. By reducing computational time with a simple prediction method this would allow for an initial survivability analysis to be undertaken, in order to identify critical conditions for further, more detailed analysis. Previous predictive methods [1] were limited to specific aerofoils and configurations, and were rendered invalid when applied to more complex damage scenarios. Therefore a new technique was developed based on pressure coefficients from an equivalent undamaged aerofoil, which would allow the prediction of damage effects on a wing of finite aspect ratio.

1.2 Aims and Objectives

The aim of this project was to develop a method to predict the aerodynamic effects of a given battle damage configuration, consisting of a single entry and exit hole on a three-dimensional (finite aspect ratio) wing, without the need for extensive experimental testing or numerical (CFD) analysis. To fulfil this aim, it was necessary to build up a sufficient data set in order to develop and then validate the predictive method, and as such meet the following objectives:

- To create a methodology to allow the prediction of the effects of an “unknown” (or previously untested) damage configuration on a two-dimensional wing.

- To create a methodology to allow the prediction of the aerodynamic effects of simulated gunfire battle damage on a three-dimensional wing, consisting of a single entry and exit hole, using either experimental or predicted two-dimensional data.

- To investigate the effects of attack angle through varying skew (spanwise displacement) and obliquity (chordwise displacement) angles of the damage holes. Therefore, the hole located on the upper surface would no longer be directly above the hole on the lower surface.

- To compare the effects of skew and obliquity on a three-dimensional wing with a two-dimensional wing.

- To evaluate and critique the effectiveness of the predictive model through comparison with wind tunnel testing of a previously untested damage configuration.
• To gain a greater understanding of flow within the wing cavity in order to validate and support future CFD studies.

1.3 Definition of Two- and Three-Dimensional Wing Testing

Two-dimensional wing models are of infinite aspect ratio and therefore have no aerodynamic effects resulting from the tips of the model. This is achieved by the model spanning the wind tunnel working section, thus ‘eliminating’ the wing tips. A three-dimensional wing model is more representative of a typical aircraft wing, which has a finite aspect ratio and at least one free wing tip. As a result of the tips, additional flow features are introduced. The two-dimensional test technique is generally considered less complex than three-dimensional testing, with structurally simpler models. Two-dimensional testing allows a clearer understanding of the flow characteristics to be developed. This is due to the uniform pressure distribution across the span of the model, thus simplifying the analysis of the flow through and around the damage case.

It should be noted that on a two-dimensional model, the presence of damage introduces local three-dimensional flow features (i.e. the flow is no longer uniform across the span). Therefore, for simplicity, throughout this work “two-dimensional” and “three-dimensional” will refer to the testing configurations, rather than the flow conditions across the wing. The relative simplicity of two-dimensional testing meant that a greater number of test cases could be undertaken than would have been possible with three-dimensional testing.

Since aircraft wings have tips and therefore have three-dimensional flow characteristics, any potential effects resulting from these must be considered. The tip has a significant influence on the wing aerodynamics, ultimately resulting in reduced lift and increased drag. The difference between the high pressure on the underside of the wing and the lower pressure on the upper surface causes a migration of the airflow from the lower surface to the upper surface, around the wing tip. This “transfer” of pressure from the lower to the upper surface results in an outboard migration of the airflow on the lower surface from the root to the tip, and an inboard migration of the airflow on the upper surface from the tip towards the root. The presence of the tip, and this migration of flow also results in the formation of vortices at the wing tips, which dissipate into the downstream. These vortices induce a component of downwash into the airflow, in the vicinity of the wing. [2]

Crucially, this downwash has a significant effect on the aerodynamics of the wing, compared to a two dimensional aerofoil, and causes the greatest difference between the two testing techniques. This is illustrated in Figure 1.1, which shows an aerofoil segment of a three-
1.3. Definition of Two- and Three-Dimensional Wing Testing

The migration of pressure around the wing tip from the lower to upper surface generates vortices, which in turn creates a downwash component, altering the angle of the freestream velocity. This reduces the effective incidence of the wing and in turn reduces the local lift and increases the drag, through the creation of the induced drag from the downwash. In addition, due to the migration of airflow across the surface of the wing, and the leakage of high pressure around the tips of the wing the pressure distribution over the surface of the wing is no longer uniform, as per the two-dimensional configuration. Instead, the pressure distribution reduces across the span, towards the tip, such that it is zero at the wing tip. [2]
In contrast to two-dimensional testing, three-dimensional testing is more difficult and time consuming. It therefore makes sense to use two-dimensional test data as the basis of the three-dimensional predictive technique. Three-dimensional testing can then be used to evaluate the effects on a model with a wing tip present and to validate predictive techniques.
Chapter 2

Review of Existing Literature

2.1 Introduction

This section will provide a review of previous two- and three-dimensional studies undertaken to analyse the aerodynamic effects of battle damaged wings. Initial work into understanding the aerodynamic effects of battle damage was undertaken by National Aeronautics and Space Administration (NASA) in the late 1960s and the early 1980s [3, 4] and included relatively basic studies, primarily involving removing large portions of an aircraft wing. These papers lacked a systematic approach into developing an overall battle damage assessment, and offered little explanation into the aerodynamic behaviour of the flow features. Later work by Irwin [1, 5, 6, 7] in the 1990s formed the basis for developing an understanding of the aerodynamic behaviour of battle damage. Rather than a full aircraft, Irwin used a model based on the NACA 641-412 aerofoil, operating at a Reynolds Number of 500,000, and was tested under two-dimensional conditions.

Since Irwin’s work, additional studies have been undertaken in order to further develop an understanding of battle damaged wings. These have included studies on the effects of different shapes of damage [8], including actual damage from live fire [9]. In addition, Irwin’s original two-dimensional studies were progressed into more realistic three-dimensional tests, considering wings of finite span [10, 11].

2.2 Damage Simulation

2.2.1 Real Damage

Battle damage studies have tended to use simulated damage (a uniform, regular shape such as a circle) rather than a hole created by firing live ammunition through the test specimen. Using ‘real’ battle damage, caused by live ammunition, is not an ideal method to investigate the changes in aerodynamic characteristics caused by a damage hole on a wing. This is due to the random nature of the damage potentially creating irregular shapes which may vary from one damage case to the next. This would have a considerable adverse
effect on the repeatability the experiments. Such variations would generally be due to small manufacturing differences and imperfections in both the ammunition and the wing, and due to limits in the repeatability of firing the damage through the panel.

2.2.2 Damage Shape, Size and Location

Irwin modelled [1] gunfire damage as a simple circular hole through the wing after his research showed this to be the most common approximate shape of damage. Irwin’s model had a chord of 200 mm, and spanned the wind tunnel working section of 450 mm. The wing model was hollow, except for a 2 mm thick skin and two full-span spars, located at 25% and 75% chord. This resulted in three full-span cavities. Irwin limited the maximum hole diameter to 40% of the chord, due to structural restrictions of the wind tunnel models and on advice received regarding levels of survivable damage. The chosen damage diameters were 10% of the wing chord, c, 20%c, 30%c and 40%c, each located with their centres at the leading edge, quarter chord, half chord and trailing edge. This resulted in the leading and trailing edge damage being semi-circles rather than circular holes.

2.2.3 Attack Angle

Irwin assumed the holes through the upper and lower surfaces to be at the same chordwise location, with the holes on top of each other. This was the result of research that identified attack angles were typically in a range of $70^\circ - 80^\circ$ from the chordline. Combined with the typical thickness of military aerofoils, this resulted in only a small chordwise offset between holes, and as such was neglected.

2.3 Flow Structures

The key flow structures through the damage hole, as defined by Irwin [1], are a weak jet and a strong jet. The flow may also be a transitional jet, not exhibiting any unique features from the weak or strong jet as it transitions from one to the other (transitional jets were identified later in work by Samaad-Suhaeb [10]). Three key variables determine the jet strength at a given test condition, assuming constant wind tunnel speed: damage size, damage location and the incidence of the wing. The majority of Irwin’s jet analysis studies were undertaken on the upper surface. Irwin did however undertake a brief study into the flow characteristics on the lower surface of a battle damaged wing. He found that at incidences below the zero lift incidence, the jet features seen on the upper surface at positive incidences were replicated on the lower surface. When a jet was present on the
upper surface, a small wake, equal in width to the damage hole was present downstream of the lower surface hole. No further analysis beyond this had been undertaken.

2.3.1 Weak Jet

The weak jet is typically associated with smaller damage holes and low incidences for larger holes. A weak jet generally has a small effect on the aerodynamic coefficients and causes minimal disruption to the local airflow. A diagram illustrating the surface flow features of a weak jet is shown in Figure 2.1, from Irwin [1]. A key feature of the weak jet that differentiates it from the strong jet is that upon exiting the hole, the flow was immediately bent over by the freestream, attaching to the aerofoil surface downstream of the hole. The resulting wake from this flow was relatively narrow and moved in the freestream direction. Irwin’s work [1] noted that the weak jet was generally associated with small surface pressure differences between upper and lower surfaces, at the centre of the damage holes. The weak jet is created by a comparatively lower pressure difference across the damage holes, resulting in the flow through the hole experiencing less acceleration. Irwin defined [1] the pressure difference as the difference between the upper and lower surface pressure coefficients taken at the centre of the hole on an equivalent undamaged aerofoil. The direction of the flow through the hole is towards the suction surface, therefore for positive lift the throughflow is from the lower surface to the upper surface.

Irwin identified the following features on a weak jet, as shown on Figure 2.1, primarily from surface flow visualisation, but these were found to be similar to features witnessed in studies of jets in cross flow, and flat-plate jet mechanics. Smoke visualisation helped Irwin to identify that the weak jet did not penetrate far into the freestream flow, typically less than the aerofoil thickness, with the wake remaining of similar thickness.

- **Forward separation line:** The outer bound of the horse shoe vortex. The point at which the attached surface flow on the aerofoil meets the adverse pressure gradient from the damage disruption and separates.

- **Horse shoe vortex:** A recirculation resulting from the deflection of the boundary layer and surface flow due to the expansion of the jet through the hole. This bounds the wake resulting from the damage. Generally this runs parallel with the chord, once aft of the hole.

- **Secondary separation line:** The inner bound of the horse shoe vortex.

- **Contra-rotating vortices:** Two vortex centres were located on the trailing edge of the damage hole and moved forward around the jet hole as the jet strengthened.
Later computational studies showed these vortices to be the result of interactions between the jet as it exited the damage hole and the freestream [12].

- **Varying velocity profile**: Downstream of the hole, the flow moved in a freestream direction in the form of a wake, attached to the aerofoil surface. Depending on the position within this wake, the velocities varied, being greatest at the edges.

- **Laminar separation bubble**: This is an aerofoil property present at low Reynolds numbers and is not linked to the damage. A laminar separation bubble forms at the point where the laminar boundary layer separates from the aerofoil surfaces, undergoes transition to a turbulent boundary layer and then reattaches to the aerofoil. In this case, the downstream wake from the damage cuts the separation bubble.

![Figure 2.1: Sketch of the flow structures present for a weak jet, from Irwin [1].](image)

Through use of surface pressure tappings on a damaged wing, Irwin identified [1] a number of trends in the surface pressure coefficients, resulting from the damage. These supported surface flow visualisation photographs. The flow on the upper surface was seen to slow upstream of the damage hole, resulting in a positive pressure increment ($C_p$ becoming less negative). This loosely correlated with $C_p$ becoming more negative on the lower surface, as the flow was accelerated into the hole. As the jet accelerated through the hole, $C_p$ values for the upper surface downstream of the hole became more negative, before returning to the undamaged state.

### 2.3.2 Strong Jet

The strong jet is typically present with larger holes at most incidences and at high incidences for smaller holes. The strong jet typically causes greater disruption to the airflow, which results in more significant changes to the aerodynamic coefficients. Irwin identified that this resulted from a much larger pressure difference between upper and lower holes. The key features of a strong jet are shown in Figure 2.2, from Irwin [1]. The jet no longer remains attached to the wing surface, and instead penetrates the freestream flow, before
being bent over. The strong jet typically penetrated more than twice the distance into the freestream than the weak jet. The wake formed by the strong jet is considerably larger and thicker than for the weak jet and now consists of a region of reverse flow near the trailing edge. The reverse flow region includes air from the lower surface, which becomes entrained into the upper surface and flows towards the damage hole against the freestream, before separating and becoming entrained in the exiting jet.

As with the weak jet, Irwin identified flow features, labelled on Figure 2.2 for the strong jet, which have similarities with jets in cross flow.

- **Forward separation line**: The forward separation line, as with the weak jet, bounded the horse shoe vortex. This now extended out considerably in a spanwise direction, compared to the weak jet.

- **Contra-rotating vortices**: The two small contra-rotating vortices, as seen in the weak jet, have typically moved further forward towards the leading edge of the hole.

- **Reverse flow region**: An attached region downstream of the hole where flow from the lower surface becomes entrained around the trailing edge and travels upstream to the separation region.

- **Separation region**: The region at which the reverse flow meets the jet expanding downstream of the hole and separates from the aerofoil surface.

- **Entrainment region**: This consists of two large vortices, which are typically witnessed at the spanwise extremes of the wake. This is a recirculation of the flow, partly caused by the horseshoe vortex, at the point where the wake separates from the trailing edge of the wing.

![Figure 2.2: Sketch of the flow structures present for a strong jet, from Irwin [1].](image)

Irwin identified that the aerofoil surface pressure coefficient trends for the strong jet forward of the damage hole were similar to the weak jet. However, on the upper surface immediately aft of the hole, the $C_p$ values were smaller in magnitude than the undamaged data. This suggested the jet was no longer accelerating around the rear of the hole, which
was consistent with the flow no longer being attached to the aerofoil. Additionally, further
downstream of the hole the flow had accelerated, compared to the undamaged condition,
giving more negative $C_p$ values. This was found to be consistent with Irwin’s surface flow
visualisation photographs, which indicated the presence of reverse flow. The lower surface
pressures showed evidence of less positive $C_p$ values towards the trailing edge, supporting
the flow being accelerated as it becomes entrained into the reverse flow on the upper surface.
Irwin also found that the spanwise extent of the damage, when a strong jet was present,
was more significant than the surface flow visualisation suggested. The surface pressures
indicated disturbances extending beyond five hole diameters from the centreline.

2.3.3 Similarities with Jets in Crossflows

Irwin undertook an extensive analysis [1] of similarities between the components of weak
and strong jets with jets in cross flows, exiting holes in flat plates. He found that key
features, such as the contra-rotating vortices and horseshoe vortices were also present on
a flat plate with a circular jet exiting into a cross flow. Figure 2.3 from [13] provides a
visual representation of the key features of an arbitrary jet in a cross flow, and similarity
with Irwin’s experimental findings from battle damaged aerofoil studies (see Figures 2.1
and 2.2) are evident. As can be seen, the forward separation line and horseshoe vortex
identified by Irwin were present in Figure 2.3, and taking up a similar shape and position
around the jet. In addition, the wake downstream of the jet exit is present, although
without the surface detail Irwin noted. The vortex pair in Figure 2.3 represent the small
contra-rotating vortices Irwin noted to exit from the sides of the damage hole. The rotation
of these form the ‘kidney’ shape of the jet, which is a feature not identified by Irwin, due
to experimental limitations in visualising the jet.

Irwin also noted that there was a correlation between the change in jet properties of a jet
in crossflow as its velocity ratio (the ratio of the jet velocity to the freestream velocity)
changed, and the strengthening of a battle damaged jet. Andreopoulos and Rodi [14]
identified that for jets of a low velocity ratio (typically less than 0.5, thus with a jet
velocity less than the freestream), the jet behaved as if a partial, inclined ‘cover’ was
obstructing the front portion of the exit hole. This caused the jet streamlines to be bent
over completely upon exiting the hole into the freestream. In addition, these low velocity
ratio jets were found to not protrude significantly above the plate into the freestream. This
is illustrated in Figure 2.4a from [14], which shows the typical streamlines for a jet with a
velocity ratio of 0.5. Figure 2.4b from [14] shows the streamlines for a jet with a velocity
ratio of 2 (typically classed as a ‘high’ velocity ratio). It can be seen that with the increased
velocity ratio, and therefore jet velocity, the jet penetrated further into the freestream, and
2.3. Flow Structures

![Figure 2.3: Sketch of the key features in a typical jet in cross flow. Figure from [13].](image)

(a) Low velocity ratio (ratio = 0.5)  
(b) High velocity ratio (ratio = 2)

**Figure 2.4:** Streamlines for jets in crossflow of varying velocity ratios. Figures from [14].

formed a larger disturbance downstream, taking on the kidney shape shown in Figure 2.3. Irwin identified that the ‘weak jet’ from battle damaged studies was very similar in size and shape to that of a low velocity ratio jet, and the ‘strong jet’ was similar to that of a high velocity ratio jet. Irwin concluded that the weak jet would have a velocity ratio of less than 0.5, and the strong jet would have a velocity ratio greater than 2. A flat plate jet with a velocity ratio of 0.5 typically showed a small region of reverse flow downstream of the jet, whereas Irwin did not identify this on the weak jets on a battle damaged aerofoil. This difference was attributed to the change in surface pressure across the chord of the model. Particularly for the strong jets on aerofoils, it was noted that the flow characteristics were very similar to that of a flat plate jet.

More recent work on jets in crossflow by Mahesh [15] have reinforced these original findings. Mahesh confirmed earlier findings by Andreopoulos and Rodi [14] that as the high velocity
ratio jet exits the hole and is bent over into the freestream, a pair of large contra-rotating vortices formed on the aft edges of the hole, which distorted the jet into a kidney shape. This drew similarities with the strong jet identified by Irwin [1]. This pair of vortices were identified as a "signature feature" of jets in crossflow. Milanovic and Zaman [16] identified the contra-rotating vortices as resulting from the shearing of the jet fluid around the perimeter of the jet, as a result of the freestream airflow. This resulted in a redistribution of the vorticity at the jet edges, which caused the pair of contra-rotating vortices to form.

The horseshoe vortex was also identified as forming upstream of the jet’s leading edge, and then travelling downstream and around the jet [15]. This existed for both circular and rectangular jets, and was caused by the boundary layer of the plate encountering an adverse pressure gradient upstream of the jet, which was caused by the interaction of the freestream with the jet. This causes the freestream boundary layer to separate and form vortices that move initially spanwise around the jet, and then dissipate downstream. At velocity ratios typically below one (i.e. a jet speed less than the freestream), much smaller vortices were seen on the jet, resulting in the jet remaining attached to the plate surface [15]. These “hairpin vortices” were formed oriented to the freestream and dissipated downstream. These flow characteristics were similar to Irwin’s weak jet.

It should be noted that most work on jets in cross flow has been undertaken using flat plates, and not aerofoils. Irwin identified this [5] and stated that the damage jet would not be uniform upon exiting the hole, unlike on a flat plate. It was identified that as the wing incidence increased, the surface pressure differential between upper and lower surfaces at the damage hole location increased. This in turn increased the velocity ratio of the jet. It should be noted that despite similarities between battle damaged jets on aerofoils and jets in crossflow on flat plate, no attempts have been made to calculate the velocity ratio of a jet on an aerofoil. In addition, changing the damage hole size on an aerofoil would alter the pressure differential, and therefore the jet velocity ratio. Based on these methods it is felt that the identified flow characteristics from surface flow visualisation corresponded well to those from flat plate studies.

Milanovic and Zaman [16] investigated the effects of pitch and yaw on jets in crossflow. The jets were inclined at angles to the chord and span and velocity profiles taken perpendicular to the freestream. This paper identified that the basic flow features was similar when the jet was pitched by 20° to the plate, with the typical kidney shape of the jet present. However, in angling the jet further away from the vertical (perpendicular with the freestream, thus turning the jet into the freestream), this resulted in the velocity of the jet reducing, due to the jet losing energy in turning. Flow structures appeared very different for the yawed cases but this was likely due to the presented velocity profile plane no longer being perpendicular
2.4 Resulting Effects of Damage on Aerodynamic Coefficients

Irwin presents the effects of the battle damage on the lift, drag and pitching moment coefficients in terms of coefficient increments, $dC_l$, $dC_d$ and $dC_m$ respectively [1]. These express the change from an undamaged datum due to the presence of damage and are defined in Equations 2.1 to 2.3 below.

$$dC_l = C_l \text{ damaged} - C_l \text{ undamaged} \quad (2.1)$$

$$dC_d = C_d \text{ damaged} - C_d \text{ undamaged} \quad (2.2)$$

$$dC_m = C_m \text{ damaged} - C_m \text{ undamaged} \quad (2.3)$$

2.4.1 Damage Size

Figures 2.5 to 2.7 show lift, drag and pitching moment coefficient increment data plotted against incidence, from Irwin [1] for damage holes varying between 10%c and 40%c, centred at half chord.

Lift coefficient increment, $dC_l$ (Figure 2.5)

- The lift coefficient increment became more negative (greater loss of lift) as incidence increased.
- The larger diameter holes had the greatest effect on the lift coefficient increments (most negative values) and resulted in the greatest loss of lift.
- The greatest changes in lift were seen when the strong jet is present (above approximately $+4^\circ$ for the 20%c hole, $0^\circ$ for the 30%c hole and $-2^\circ$ for the 40%c hole).
Drag coefficient increment, $dC_d$ (Figure 2.6)

- Increasing the damage size increased the drag coefficient increments.
- Larger drag coefficient increments corresponded with wider wakes and greater disruption to the surface airflow, seen on surface flow visualisation photographs.
- The onset of the strong jet corresponded to an increase in the gradient of the $dC_d$ curve, leading to larger drag coefficient increments.

Pitching moment coefficient increment, $dC_m$ (Figure 2.7)

- Data showed the pitching moment coefficient increments to be relatively invariant with hole size, partly due to the centre of the hole remaining unchanged relative to the aerodynamic centre.
- The pitching moment coefficient increments became more negative (more nose down pitching moment) with increased angle of attack and jet strength, up to stall.

2.4.2 Chordwise Damage Location

In order to illustrate the effects of the chordwise location of the damage, a fixed damage hole diameter of 20%c will be discussed. This hole size exhibited both weak and strong jet characteristics for the NACA 641-412 aerofoil. Table 2.1 shows data extracted from Irwin’s thesis [1]. For this damage case at quarter chord and half chord locations, the weak jet was present at 0$^\circ$ and the strong jet was present at 8$^\circ$. Irwin also positioned a hole at leading edge, but the data were not presented in coefficient increment format, and as such general trends will only be briefly discussed. Figures 2.8 to 2.10 show coefficient increments for quarter and half chord holes – leading and trailing edge holes have been omitted, since these removed half the area from the wing compared to the quarter and half chord holes. The leading edge damage caused a large separated region downstream of the hole, the mid and half chord cases caused a jet and wake to form, while the trailing edge damage had a negligible effect.

<table>
<thead>
<tr>
<th>Damage location</th>
<th>$dC_l$ 0$^\circ$</th>
<th>$dC_l$ 8$^\circ$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Quarter chord</td>
<td>-0.025</td>
<td>-0.185</td>
</tr>
<tr>
<td>Half chord</td>
<td>-0.035</td>
<td>-0.100</td>
</tr>
<tr>
<td>Trailing edge</td>
<td>-0.015</td>
<td>-0.020</td>
</tr>
</tbody>
</table>

Table 2.1: Variation of $dC_l$ with chordwise location of damage for two incidence angles with a 20% chord hole. Data from graphs in Irwin’s thesis [1].
2.4. Resulting Effects of Damage on Aerodynamic Coefficients

Figure 2.5: Lift coefficient increments for various hole sizes, centred at half chord. Data from Irwin [1].

Figure 2.6: Drag coefficient increments for various hole sizes, centred at half chord. Data from Irwin [1].

Figure 2.7: Pitching moment coefficient increments for various hole sizes, centred at half chord. Data from Irwin [1].
As can be seen from Table 2.1, at $0^\circ$ incidence, the change in lift coefficient increment was greatest when the damage was centred about half chord. At $8^\circ$ the quarter chord location saw the greatest change. This change was found to be due to the aerofoil’s upper surface suction peak moving towards the leading edge as incidence increased. This is also demonstrated in Figure 2.8, which shows $dC_l$ for a range of incidences, for holes centred at quarter and half chord. Both holes had similar performance below $5^\circ$ incidence (identified as a weak jet). Once the quarter chord hole had developed into a strong jet, the $dC_l$ curves diverged, with the quarter chord hole generating a greater loss of lift.

<table>
<thead>
<tr>
<th>Damage location</th>
<th>$0^\circ$</th>
<th>$+8^\circ$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Quarter chord</td>
<td>0.005</td>
<td>0.045</td>
</tr>
<tr>
<td>Half chord</td>
<td>0.006</td>
<td>0.018</td>
</tr>
<tr>
<td>Trailing edge</td>
<td>0.004</td>
<td>-0.001</td>
</tr>
</tbody>
</table>

Table 2.2: Variation of $dC_d$ with chordwise location of damage for two incidence angles with a 20% chord hole. Data from graphs in Irwin’s thesis [1].

The change in drag coefficient increment with hole location due to the damage is shown in Table 2.2. As incidence increased the $dC_d$ values increased for quarter and half chord locations. As with $dC_l$, the greater difference between weak and strong jets was noticed at the quarter chord location. This was supported by Figure 2.9, which shows the drag coefficient increments for quarter and half chord holes. As with $dC_l$ a significant difference between quarter and half chord holes was noticed as the incidence increased. Unlike the other locations, the trailing edge location showed little change in $dC_d$ with incidence. This was a result of the jet and wake from the damage immediately dissipating into the freestream, due to there being no aerofoil surface downstream of the damage for the flow to reattach to.

Irwin suggested that the difference between the pressure coefficient on the upper and lower surface, $dC_p$, taken from an equivalent undamaged wing at the chordwise centre of the hole, was found to be the main driver of the flow characteristics. Irwin suggested further that the greater the pressure differential between the upper and lower surfaces, the stronger the jet was. At the trailing edge, $dC_p$ was considerably smaller than at quarter and half chord, so consequently any aerodynamic coefficient increments were smaller. It can be seen that $dC_l$ and $dC_d$ values increased in magnitude closer to the pressure peak, causing a stronger jet and greater disruption to the airflow.
2.4. Resulting Effects of Damage on Aerodynamic Coefficients

Figure 2.8: Variation of $dC_l$ with chordwise hole centre, for a 20%c hole. Data from Irwin [1].

Figure 2.9: Variation of $dC_d$ with chordwise hole centre, for a 20%c hole. Data from Irwin [1].

Figure 2.10: Variation of $dC_m$ with chordwise hole centre, for a 20%c hole. Data from Irwin [1].
Table 2.3: Variation of $dC_m$ with chordwise location of damage for two incidence angles with a 20% chord hole. Data from graphs in Irwin’s thesis [1].

<table>
<thead>
<tr>
<th>Damage location</th>
<th>0°</th>
<th>+8°</th>
</tr>
</thead>
<tbody>
<tr>
<td>Quarter chord</td>
<td>-0.002</td>
<td>-0.042</td>
</tr>
<tr>
<td>Half chord</td>
<td>-0.004</td>
<td>-0.010</td>
</tr>
<tr>
<td>Trailing edge</td>
<td>0.004</td>
<td>-0.003</td>
</tr>
</tbody>
</table>

Table 2.3 shows the change in pitching moment coefficient increments for each location. As with lift and drag, 8° for the quarter chord location saw the greatest magnitude of coefficient increment. However, at 0°, all locations had very small coefficient increments, close to zero. The trends indicate that while also being a function of the undamaged pressure difference between upper and lower surfaces across the hole, $dC_m$ is also a function of the distance from the aerodynamic centre. Figure 2.10 shows the $dC_m$ coefficient increments for the quarter chord case. This indicates the large change in the data, and demonstrates the effect of the distance of the hole from the moment reference centre on the pitching moment.

Work by de Waal [17] was undertaken at a lower Reynolds Number of 180,000, compared to Irwin’s work. This work took a different approach to Irwin’s testing and instead made use of a rectangular leading edge cut out of varying size. While the aerofoil and damage configurations were significantly different to Irwin’s setup the trends identified by de Waal were broadly similar to Irwin’s work on leading edge damage. Although a clearly defined transition from weak to strong jet was not identifiable in de Waal’s work (no surface flow visualisation was provided, and graphs were not presented in coefficient increment form), the general trend of decreasing lift coefficient and increasing drag coefficient with increasing incidence and damage size was identified.

### 2.4.3 Effects of Reynolds Number

During Irwin’s studies [1], most data were collected at a Reynolds number of 500,000. Given that this is relatively low compared to typical unmanned aerial vehicles and low speed aircraft, it was necessary to assess for the effects of changing Reynolds number. Irwin therefore carried out additional tests at 250,000, since all other testing had been undertaken at the maximum Reynolds number achievable in the wind tunnel. However, it was identified by project sponsors that a typical operating Reynolds number for a low speed unmanned aerial vehicle would be in the region of 2,000,000 - 4,000,000.

Irwin found that for both the lift and drag coefficient increments, particularly at lower incidences, there was no significant effect on the coefficient increments due to the change in Reynolds number. The pitching moment coefficient increments followed a similar trend.
Due to the reduction in Reynolds number, the dynamic pressure and forces acting upon the model were reduced and so greater inaccuracies would have been incurred in the balance measurements. As increasing Reynolds number from 500,000 to 4,000,000 would likely cause a significant change in maximum $C_l$ (increases with Reynolds number), minimum $C_d$ (decreases with Reynolds number) and stalling incidence (increases with Reynolds number) [2] it would be expected that troughs and peaks in the coefficient increment graphs (for example, the peak prior to the stall) would move to higher incidences. However, as coefficient increments were used to present data, thus comparing damaged aerodynamic data to undamaged data from the same Reynolds number, this would remove most of the influence of comparing coefficient increments from differing Reynolds numbers. It should be noted that at higher Reynolds numbers there is a greater likelihood of the flow being turbulent, whereas Irwin’s studies were undertaken with laminar flow. This would likely cause an increase in drag compared to laminar flow, but this effect would be minimised when converting to coefficient increments.

### 2.4.4 Wing Internal Structure

Throughout Irwin’s studies [1] hollow wing models were used, with a full-span cavity between spars at 25%c and 75%c. These allowed for a more representative model of an aircraft wing to be tested, and to determine if the presence of a cavity had any effect on the damage effects. However, due to the model construction, studies into the effects of damage on the internal flow within the cavity were not undertaken.

Irwin found [6] that the hollow wing generally produced a lesser magnitude of increment than the solid wing, although the general trends were similar. The effects of the solid wing were most noticeable when a strong jet was present. Irwin suggested that this was due to the hollow model causing a slight reduction in pressure coefficient on the lower internal surface around the hole, and a slight increase in pressure coefficient on the upper internal surface, thus causing a positive lift increment. Since this was not present with the solid wing, the lift coefficient increments became more negative. Irwin also found the drag coefficient increment to have a generally constant offset for the solid wing, compared to the hollow wing, resulting in more positive values. Typically the solid wing increased the $dC_l$ values by up to 20%, with $dC_d$ values increasing by around 25%. The change to $dC_m$ was less significant and resulted in a small change to the gradient of the $dC_m$–incidence curve.

These results suggest that for the most accurate damage modelling it is important to consider the dimensions of the internal structure of a wing as accurately as possible.
2.4.5 Camber Variation

Render et al [18, 19] investigated the effects of camber variation through testing different variants of the NACA 641-012 thickness form with different camber lines. Table 2.4 provides a summary of the incidences at which the jet transitions from weak to strong jet, showing data from [18]. It was found that the coefficient increments from different camber profiles followed the same basic trends as for holes changing in chordwise position. Additionally, the strength of the jet flow through the damage hole was altered, and the incidence at which transition from a weak to strong jet occurred was delayed. It was concluded that for positive incidences, reducing camber decreased the drag coefficient and resulted in a smaller loss of lift, with later jet transition.

The changes due to camber support Irwin’s work [1] which indicated that the pressure differential between upper and lower surfaces was a key factor in determining the jet strength. Generally, the greater the pressure differential, the stronger the jet. A reduction in camber results in a reduction in the pressure differential at a given chordwise location on the wing. Therefore, this would suggest a weakened jet. The work by Render et al [18], as shown in Table 2.4 supports this, by showing delayed transition with reduced camber, and therefore reduced pressure differential.

<table>
<thead>
<tr>
<th>Damage size</th>
<th>NACA Aerofoil</th>
<th>10% chord</th>
<th>20% chord</th>
<th>30% chord</th>
<th>40% chord</th>
</tr>
</thead>
<tbody>
<tr>
<td>NACA 641-012</td>
<td>8 – 10°</td>
<td>6 – 8°</td>
<td>4 – 6°</td>
<td>0 – 2°</td>
<td></td>
</tr>
<tr>
<td>NACA 641-412</td>
<td>6 – 8°</td>
<td>0 – 2°</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
</tbody>
</table>

Table 2.4: Comparison of the incidence of jet transition for two NACA aerofoils with different camber profiles. Table from Render et al [18].

Further studies by Render and Walton [19] investigated the effects of varying camber on battle damage through use of a trailing edge flap. The flap occupied the aft 25%c of a NACA 0012 aerofoil and could be angled in 10° increments between -20° and +20°. As with Irwin’s work, circular damage holes of between 10%c and 40%c diameter were centred at quarter chord. It was found that positive flap deflection, thus increasing the camber of the aerofoil, strengthened the jet, increased the magnitude of the coefficient increments and caused significant flow separation across the upper surface of the flap. Negative flap angles (decreasing the camber) were found to reduce the jet strength and the magnitude of the coefficient increments, and in some cases weaken the strong jet to a weak jet. It was therefore suggested that a trailing edge control device could be utilised in order to reduce the effects of the battle damage. For example, if the aircraft was equipped with split flaps or ailerons (i.e. divided into multiple, independent spanwise surfaces), that could be operated independently, a single control surface over which the jet is flowing could be
deflected upwards in order to weaken the jet. If the ailerons or flaps consisted only of a single spanwise element per wing, this would likely not have the required effect as the change in aircraft speed or angle of attack required to regain the lost lift from the control surface deflection may result in the jet strengthening.

2.4.6 Petalling

Petalling is generally only experienced on metallic skins. The metal buckles and essentially “peels” away from the surface, causing small raised regions around the damage hole. Petalling is often neglected due to it being very difficult to accurately model the small serrations on a small-chord wing. With composite wings, petalling becomes less of an issue since composite material tends to splinter and delaminate with any fibres protruding above the surface typically removed by the airflow [20].

Robinson and Leishman [9] briefly looked into the effects of petalling around a circular hole on a metallic helicopter rotor blade. This was done by adding serrations approximately 2.4%c high (where ‘c’ is the chord length), attached around the circumference of the hole. Flow visualisation studies showed the petalling did not have a significant impact on the flow, but highlighted that the wake was more energetic immediately downstream of the serrations and the reverse flow was stronger on the wing surface. Aerodynamic results showed that there was little difference to the change in lift coefficient when petalling was added to the damage. The drag coefficient saw the most significant change, with an increase of up to 10%. Small differences were noted in pitching moment, which suggested the petalling was having an effect on the chordwise pressure distribution, albeit minor. The report concluded that the hole was the primary source of degradation of aerodynamic properties, with serrations or petalling providing second order effects.

2.4.7 Non-circular Holes

It is recognised that the use of circular holes is an approximation to real battle damage. Studies by Render et al [18] looked into the effects of non-circular holes on the aerodynamic coefficients in order to verify the acceptability of using circular holes to simulate battle damage. This was done by using a six-point star shaped hole, with an area equal to that of a 23%c circular hole. Render et al used the same NACA aerofoil as in Irwin’s studies.

Surface flow visualisation studies (Figure 2.11, from Render et al [18] with original labels) for star shaped holes showed that the same basic jet features were present as with the circular holes. Closer analysis of the weak jet case showed that the horseshoe vortex was still present, having formed off the leading point of the star. It was also seen that three
separate small jets exited from the rear of the damage. Other vortices were present on the aft side apexes, suggesting jets also existed there. Despite the presence of multiple jets a weak jet was still produced at a similar incidence used in Irwin’s work.

It was evident from the strong jet surface flow visualisation photograph that the three separate jets from the weak jet had merged into one. Additionally, the side apexes of the star caused the primary separation line to be deflected outwards, resulting in a wider jet (Point B). These results also showed that the star shape delays the transition from a weak to a strong jet, by up to $4^\circ$ for the tested aerofoil, when compared to the circular holes from Irwin’s work [1].

There was generally a very good agreement between the lift coefficient increments of the star shape compared to the circular hole (Figure 2.12). The drag coefficient increments (Figure 2.13) matched reasonably well across the weak jet region, before diverging with the star producing slightly greater values as the strong jet developed. A similar trend was also noticed for pitching moment coefficient increments (Figure 2.14), although the star produced smaller magnitude values than the circle when operating as a weak jet. From this study it was indicated that simulating battle damage with circular holes provides a reasonable approximation, provided the areas and maximum diameters of the damage holes are similar.
2.4. Resulting Effects of Damage on Aerodynamic Coefficients

![Graph](image_url)

**Figure 2.12:** Comparing variation of $dC_l$ of a star-shaped hole with a circular shaped hole. Data from Render et al [18].

![Graph](image_url)

**Figure 2.13:** Comparing variation of $dC_d$ of a star-shaped hole with a circular shaped hole. Data from Render et al [18].

![Graph](image_url)

**Figure 2.14:** Comparing variation of $dC_m$ of a star-shaped hole with a circular shaped hole. Data from Render et al [18].
2.4.8 Actual Ballistic Damage

As part of a wider assessment of the effects of battle damage on helicopter rotor blades, Robinson and Leishman [9] studied the effects of actual ballistic damage. A representative projectile was fired through a specimen helicopter blade, which generated a single large hole with irregular edges, approximately triangular in shape, with several smaller holes in close proximity to the large hole. The effects were measured through wind tunnel testing and compared to an undamaged blade. In addition, a triangular hole of similar area to that removed by the live fire hole, but with uniform edges was also tested. Results were not presented in the form of coefficient increments, unlike other battle damage work.

It was found that the actual damage produced similar reductions in the maximum lift and the lift curve slope as the representative triangular damage. However, the actual damage caused a small reduction in the zero lift angle, which was attributed to the skin panels closest to the remaining thin section of model at the trailing edge splitting and distorting the shape of the aerofoil. This was not represented on the triangular damage. It was identified that there were more complex flow structures on the actual damage, resulting from the jagged edges around the hole and delamination of the skin. These also contributed to an increase of drag compared to the triangular hole. It is important to note that most of the differences between the actual and representative damage cases will be due to the material choice for the aerofoil. As was discussed in section 2.4.6, metallic aerofoils would experience petalling of the skin, rather than delamination and de-bonding, which has been shown to produce only a small change in drag. In addition, no statement was made by Robinson and Leishman into the repeatability of the actual ballistic damage, and as such the overall accuracy of the actual damage testing cannot be verified. This highlights the biggest concern with testing actual ballistic damage, in that the shapes generated may have poor levels of experimental repeatability, particularly due to splintering of composite structures, as witnessed by Robinson and Leishman.

Irwin had undertaken a detailed study [1] to investigate the most common damage shapes from live gunfire. Based on the assumptions that the aircraft would be attacked by a single anti-aircraft artillery shell, studies showed from live fire testing that the most common damage shape on lightweight metallic structures, such as aircraft wings, could be approximated to a circular hole. It is important to highlight that prior to Irwin’s studies, little work has been undertaken on the aerodynamic effects of battle damage, and in particular the accuracy of modelling simulated battle damage compared to actual ballistic damage. With the stated comparisons of more complex shapes (such as the star damage, discussed in section 2.4.7) against circular holes, it is felt that the circular hole provides a sufficient degree of accuracy to allow for the current battle damaged effects to be expanded.
2.5 Influence of a Finite Aspect Ratio Wing on Battle Damage Aerodynamics

Limited research has been undertaken into battle damage studies on either finite aspect ratio wing models, or models of entire aircraft. Early studies by NASA involved removing entire sections of an aircraft wing [3, 4] but did not provide a detailed analysis into the aerodynamic effects resulting from the damage. A more recent study by Shah at NASA [21] analysed the effects of circular holes and control surface loss on a generic commercial transport aircraft’s wing, horizontal and vertical tailplanes. Since the primary aim of this paper was to develop improvements for flight control systems, the focus was on assessing the effects of damage from a control perspective, rather than aerodynamic analysis. In addition, Shah’s paper stated that a large number of variables affected single-hole damage cases and as such did not provide as detailed a discussion compared to the effects of losing an entire wing section or skin panel.

Work by Render et al [22] progressed the battle damage studies from Irwin’s two-dimensional work and evaluated the effects on a, finite aspect ratio wing. A half model was mounted vertically in the wind tunnel with one free tip. The wind tunnel floor was utilised as a reflection plane. The free tip was at a sufficient distance to not be affected by any boundary layer effects from the wind tunnel walls. By using the half model technique this doubled the effective aspect ratio, which allowed a larger model to be tested and gave higher Reynolds numbers [23]. Wings of different aspect ratios (fixed chord, varying span) were tested, using the same aerofoil profile as Irwin. Due to construction limitations, only 20% chord diameter holes centred at half chord were tested. In contrast to Irwin’s work, the models were solid. The key focus of Render et al’s work was to investigate the effects of a varying spanwise pressure distribution on the aerodynamic effects of battle damage.

2.5.1 Flow Structures of Finite Aspect Ratio Wings

Surface flow visualisations studies were undertaken by Render et al [22] on a model of aspect ratio 8, with the damage located at 55% and 80% span from the root. A test was undertaken at 20% span but it was found that at this location the presence of the wind tunnel floor was adversely affecting the results. Flow visualisation was undertaken at 4° and 8° incidence.

Flow structures were similar to those witnessed on the two-dimensional wings, discussed in section 2.3. Figure 2.15 from Render et al [22] shows the 20%c hole located at 55% span. As can be seen in Figure 2.15a, the typical flow characteristics associated with a weak jet are present: the pair of contra-rotating vortices (point “B”); the forward separation lines;
Figure 2.15: Surface flow visualisation for a 20%c hole located at 55% span (445mm from root) on an aspect ratio 8 wing. Tip on the left. Photographs and annotations from Render et al [22].

Figure 2.16: Surface flow visualisation for a 20%c hole located at 80% span (650mm from root) on an aspect ratio 8 wing. Tip on the left. Photographs and annotations from Render et al [22].

and the horseshoe vortex (point “D”). However, the horseshoe vortex highlights the key difference between two-dimensional and three-dimensional wings: flow asymmetry, which caused the horseshoe vortex to be twisted away from the tip. The horseshoe vortex is clearly defined on the left hand (outboard) side of the photograph, but is weaker than on the right hand side (nearer the root). This would correlate with the reducing pressure difference between upper and lower surfaces along the span of the wing, towards the tip. As the incidence increases to 8°, shown in Figure 2.15b, the flow has developed into a strong jet. It is clear from these figures that the asymmetry witnessed with the weak jet is still present, since the inboard (right) vortex is more prominent than the outboard vortex.

Figure 2.16 shows the same damage condition located at 80% span. When compared to Figure 2.15 the increased effect of the reduction in spanwise pressure difference can be seen. This is shown by the jets appearing weaker at both incidences. In particular the flow structure has changed notably at 8° incidence (Figure 2.16b), with a significant reduction
in wake and vortex size. In addition, the horseshoe vortices have become more skewed on the outboard side, suggesting the wing tip vortex may be turning the jets. It is apparent from the surface flow visualisation that the reduction in spanwise pressure difference due to the tip has had a weakening effect on the jet, and by moving the hole closer to the tip, transition from weak to strong jet was delayed.

### 2.5.2 Variation in Aerodynamic Coefficient Increments due to Spanwise Damage Location

Render et al. [22] also analysed the effects of spanwise location of battle damage on lift, drag and pitching moment coefficient increments. Figures 2.17 to 2.19 show lift, drag and pitching moment coefficient increments for 20% c holes at two spanwise locations 55% and 80% span from the root. Render et al also plotted data from Irwin’s two-dimensional testing [1], which had been converted to the three-dimensional wing geometry by multiplying the coefficient increments by the wing area ratios. The results from the three-dimensional wing followed the trends of the two-dimensional results reasonably well. It can be seen that the lift and drag coefficient increments reduce in magnitude as the damage is moved towards the tip. The surface flow visualisation supported the reduction in magnitude of coefficient increments by showing the jets weakening. However, pitching moment coefficient results struggled to match those of the two-dimensional model. This was attributed to different mounting configurations, with Irwin’s model mounted horizontally with pin jointed struts and Render’s model mounted vertically and directly onto the balance. It was also identified by Render et al. that the half model configuration used introduced errors in pitching moment with aspect ratios below 8.

Additionally, work by Djellal and Ouibrahim [11] investigated the effects of circular damage on a 1/20th scale model of a small aircraft with tapered wings of span 450 mm. Varying sizes of damage were tested, between 20% and 40% c, inserted as circular holes on one or both wings at a variety of spanwise locations. It should be noted that the hole sizes were based on local chord, and as such the holes were smaller in diameter at the tip than at the root location (the tip hole diameters were approximately 60% of the root holes), and were thus removing different amounts of wing area. However, each damage hole was still the same non-dimensional diameter when expressed as a percentage of the local chord.

The damage was centred at either quarter or half chord. The Reynolds number was 173,000 based on mean chord, considerably lower than that of Render or Irwin’s work, which used a Reynolds number of 500,000. While Djellal and Ouibrahim did not present any data in terms of coefficient increments, or any data for pitching moment, some similarities can be drawn with the work by Render et al. [22]. It was noted that the tip location produced
significantly smaller lift and drag values compared to the damage located at the root or at
the location labelled ‘mean aerodynamic chord’. In addition, to support Irwin’s original
work [1], damage located at quarter chord produced a greater change in aerodynamic
coefficients than damage located at half chord. This would correlate to larger magnitude
coefficient increments.

2.6 Review of Studies using Computational Fluid Dynam-}
ics

Studies into simulating the aerodynamic effects of battle damage on aerofoils using Compu-
tational Fluid Dynamics (CFD) have generally attempted to replicate existing wind tunnel
data and simulate internal cavity flows. It should be noted that to date no wind tunnel
testing has attempted to quantify the flow inside the wing cavity, and as such no compar-
isons to wind tunnel results or assessments of accuracy can be undertaken on these data.
The replication of wind tunnel results has generally revolved around generating streamline
diagrams to compare against surface flow visualisation photographs, and comparing the
lift, drag, pitching moment and surface pressure results. Studies to date include a PhD
thesis by Samaad-Suhaeb [10] and conference papers by Yang et al [24] (based upon work
from Samaad-Suhaeb’s thesis) and Saeedi et al [12].

It is important to note that the accuracy of any CFD simulations depend on the quality of
the modelling and resolution of the meshing. A coarse mesh results in a loss of accuracy, but
increasing the mesh density or resolution increases processing time. Additionally, the sets
of governing equations used will affect the accuracy, depending on the flow being modelled,
and will also affect computational time. Typically, for aeronautical applications, the k-$\varepsilon$
and Spalart-Allmaras (S-A) turbulence models are used, which offer the optimum modelling
of flows bounded by walls (e.g. within a wind tunnel) or flow within the boundary layer.
This latter factor is important for considering flow close to the surface of an aerofoil, and
when attempting to replicate surface flow visualisation photographs. Samad-Suhaeb [10]
undertook a detailed comparison of the k-$\varepsilon$ and S-A turbulence models on battle damaged
wings, and compared the simulations to experimental results. Samad-Suhaeb determined
that while the k-$\varepsilon$ model produced more accurate models aft of the damage, the S-A model
produced a better match to the experimental data when considering flow across the entire
aerofoil and across the incidence range. This was likely due to the S-A model being able to
calculate boundary layers experiencing adverse pressure gradients more accurately.
2.6. Review of Studies using Computational Fluid Dynamics

Figure 2.17: Lift coefficient increment for 20%c damage at varying spanwise locations from the root of an AR8 wing. Data from Render et al [22].

Figure 2.18: Drag coefficient increment for 20%c damage at varying spanwise locations from the root of an AR8 wing. Data from Render et al [22].

Figure 2.19: Pitching moment coefficient increment for 20%c damage at varying spanwise locations from the root of an AR8 wing. Data from Render et al [22].
2.6.1 Replicating Wind Tunnel Results

Work by Saeedi et al [12] focused on simulating a 30% chord diameter hole located at half chord on a two-dimensional wing, as tested by Irwin [1]. In addition, the star shaped damage investigated by Render et al [18] was also simulated, albeit equivalent to a 31%c circular hole and not the 23%c circular hole used by Render et al. Saeedi’s simulations modelled the presence of the wind tunnel walls and were undertaken using the k-ε model. A relatively high mesh density was used of 2,000,000 cells for the circular damage and 6,000,000 for the star shaped damage. It is worth noting that Saeedi et al modelled the boundary conditions of the wind tunnel working section, unlike other CFD work.

Flow field plots for the circular hole showed that many of the features identified on surface flow visualisation photographs by Irwin were replicated in the CFD simulations, including the presence of weak and strong jets. The increase in wake size as the jet strengthened was also indicated in the computational simulation, however at 10° incidence the experimental surface flow visualisation showed a wider wake than was predicted by the CFD results. This was believed to be limitations and errors within the numerical solutions used, and results from the simulation being unable to capture the forward separation lines, so therefore could not model or show the horseshoe vortex.

When predicting lift and drag coefficient increments for the 30%c circular hole (no predictions of pitching moment coefficient increments were made), it was found that across the positive incidence range the simulation results generally under predicted compared to experimental data. The comparisons for the 30%c circular hole are shown in Figures 2.20 and 2.21 for \( dC_l \) and \( dC_d \) respectively, with all data from [12]. Both \( dC_l \) and \( dC_d \) simulations appear to follow the general trends of the experimental data well.

Saeedi et al also attempted to predict the flow field and coefficient increments for Render et al’s [18] star shaped damage case. The flow field results showed a strong distinction between weak and strong jets, identifying many of the features from experimental flow visualisation photographs. Comparisons with experimental data for the star shaped damage hole were also presented, against work undertaken by Render and Mani in [8, 22]. The data from the latter paper was presented in coefficient form, rather than as coefficient increments, which does not allow for the trends to be compared with the circular holes discussed previously. The lift and drag coefficient results from Saeedi et al are shown in Figures 2.22 and 2.23 respectively. As can be seen from Figure 2.22, the lift coefficients match reasonably well until the onset of stall, when the CFD results diverge. A constant offset is witnessed with the drag coefficient results (Figure 2.23), which was attributed to restrictions in the modelling, which set the transition point for the boundary layer at the leading edge. This resulted in the entire boundary layer being turbulent, unlike the experimental data.
2.6. Review of Studies using Computational Fluid Dynamics

Figure 2.20: Comparison of $dC_l$ values from CFD simulations and experimental results for a 30\%c circular hole at half chord. Data from Saeedi [12].

Figure 2.21: Comparison of $dC_d$ values from CFD simulations and experimental results for a 30\%c circular hole at half chord. Data from Saeedi [12].

Figure 2.22: Comparison of $C_l$ values from CFD simulations and experimental results for a star shaped hole at half chord. Data from Saeedi [12].
Samaad-Suhaeb et al [10, 24] undertook analysis on a half-wing model of aspect ratio 8 to mirror the work of Render et al [22]. A 20% c hole located 55% span and 80% span, centred at half chord, was simulated using CFD and compared with the experimental results. A considerably more coarse model was used, compared to that of Saeedi, with only 690,000 cells. The CFD model was initially validated against an undamaged wing. Generally it was found that $C_L$ matched well, with divergence occurring at onset of stall. However, $C_D$ values were generally over-predicted by the simulation.

Predicted plots of streamlines showed a generally good match with the surface flow visualisation for the damage case presented by Render et al. Of worthy note is that the asymmetry noted by Render et al [22] was also present in the simulations, which gave reassurance that the skewing of the horseshoe vortex was a genuine effect. While the flow features were generally well identified on the simulation, it was noted that at 4° incidence at both the mid and tip locations, the simulated jet displayed flow features consistent with a transitional jet, while the experimental surface flow visualisation showed the jet to be fully strong. As the incidence was increased to 8° the simulation showed the jet strengthening and more features in common with the strong jet being present. This inconsistency in the jet strength was also present when analysing surface pressure coefficient values, with a significant difference noted between simulated and experimental results at 8° incidence. The simulated results were consistent with that of a weaker jet.

Figure 2.23: Comparison of $C_d$ values from CFD simulations and experimental results for a star shaped hole at half chord. Data from Saeedi [12].
2.6.2 Simulating Flow Inside the Hole

Work by Saeedi et al [12] and Samaad-Suhaeb et al [24] attempted to simulate the flow through the damage hole. For the purposes of all simulations, the wings were modelled as solid. Flow fields inside the hole presented by Saeedi et al for the 30%c circular hole showed the jet exiting from the rear half of the hole. This confirmed the experimental surface flow visualisation photographs. Additionally a low pressure region in the forward section of the hole was identified, caused by suction from the contra-rotating vortices on the upper and lower wing surfaces. As the incidence was increased, the recirculating flow within the forward section of the hole was lifted up above the upper surface of the wing to allow a greater airflow from the jet through the hole.

The chordwise stream traces from Samaad-Suhaeb et al [24] showed that at all simulated incidences, (4°, 8° and 10°) at both the tip and mid span locations, the flow generally exited through the rear half of the hole, with the forward section acting like a cavity, causing the flow to recirculate within this region. This is illustrated in Figure 2.24 from [24], which shows streamlines for a hole at the tip location. The flow features shown in these figures indicate that the hole was not running full, which was backed up by surface flow visualisation from Render et al [22] and supports Saeedi et al’s work. At the higher incidences (Figure 2.24a) the large separation bubble was witnessed downstream of the hole, identifying the reverse flow region associated with the strong jet.

**Figure 2.24:** Chordwise stream traces at the centre of a damage hole, with flow from left to right, for incidences at 4° (a), 8° (b) and 10° (c). Image from Samaad-Suhaeb et al [24].

The CFD studies did identify new flow features close to the damage hole, which had not been identified previously by surface flow visualisation photographs. Such features included side wall vortices, found either side of the hole at approximately the location where the forward portion of the jet exits the hole. The stream traces identified a low pressure region in the forward portion of the hole, which from the resulting suction of the flow, caused two groups of contra-rotating vortices to form within the hole: an upper pair and lower pair. It was suggested that the larger of these vortices, closest to the hole walls were feeding the pair of small contra-rotating vortices, seen on the upper surface flow visualisation close to
the jet exit. In addition, the flow from the jet was seen to be feeding and thus influencing the horseshoe vortex. This highlights the major benefit of CFD, in that it allows the flow inside the hole or cavity to be visualised with relative ease, although it should be noted that it can be very difficult for wind tunnel testing to validate any flow features within the cavity identified by computational studies. It is likely that the internal flow features will differ on a hollow wing, given the differences in coefficient increments (see section 2.4.4), however the papers discussed here have shown the extent that the internal hole geometry can drive the flow.

2.7 Methodologies for Predicting Battle Damage Effects

2.7.1 Review of Two-Dimensional Methodologies

Irwin developed empirical methodologies [1] for determining the aerodynamic effects of a previously untested battle damage configuration using an interpolation method. The methodology was proved to be valid for damage located at quarter or half chord on the NACA 641-412 aerofoil. Irwin related the effects of all of his tested damage cases to the difference between the pressure coefficients for the upper and lower surfaces on an undamaged wing at the chordwise centre of the holes. From this, Irwin developed interpolation methods for lift, drag and pitching moment coefficients. The methods were only validated against the NACA 641-412 aerofoil, for damage between quarter and half chord. A detailed summary of Irwin’s methodology is given in Appendix A.

Irwin’s methodology for predicting lift coefficient increments was based upon the area of the wing surface that was affected by the resulting wake from the damage. As this was not accurately known, it was assumed to be a function of the area of the damage, the throughflow and the reduction in pressure coefficient difference over the region affected. In order to calculate the lift coefficient increments, Irwin derived a number of factors, which were obtained by plotting graphs from experimental data and calculating gradients of linear best fit curves, omitting any points which were at or beyond stall. Different graphs were constructed for each hole size and chordwise position in order to obtain the appropriate gradients. Irwin then plotted these gradients against hole size to obtain the final scaling factors.

To predict the drag coefficient increments, the change in drag was related to the lift coefficient increments. Irwin identified that when $dC_d$ was plotted against $dC_l$ two distinct data sets were formed for different hole sizes: a cluster of weak jet data points with no clear trends, and distinct set of strong jet data points, which formed approximately linear relationships. Irwin then developed curves to allow the drag coefficient increments for inter-
mediate damage sizes to be predicted. Irwin’s experimental results identified that pitching moment coefficient increments at a given chordwise location were generally invariant with hole size, and as such did not require any additional predictive methodologies to be defined. Consequently for the pitching moment coefficient increments, all intermediate hole sizes were calculated using averaged values of Irwin’s experimental data at the incidence required.

Irwin validated the predictive techniques by predicting the resulting aerodynamic coefficient increments from a 24%c hole at quarter chord, and verifying the prediction with wind tunnel tests. Figures 2.25 to 2.27 show the predicted versus actual lift, drag and pitching moment coefficient increment data respectively. All data shown in the figures are from Irwin’s thesis [1]. The prediction for lift coefficient increments (Figure 2.25) was seen to be reasonably accurate compared to the experimental data, up to approximately 8° incidence. It was noted by Irwin that above this incidence the prediction methodology breaks down due to onset of stall, which was outside of the region of validity for the method.

A good match between predicted and actual drag coefficient increments (Figure 2.26) was witnessed between 0° and +10°. As with the \( dC_l \) prediction, the results break down as stall onsets. This was due to the drag coefficient increment predictions being heavily reliant on predicted lift coefficient increment values. Therefore, any errors in the \( dC_l \) prediction would compound errors in the \( dC_d \) prediction. The prediction is less accurate below approximately 2°, when a weak jet was present, and was due to Irwin’s method relying on strong jet data to approximate weak jet results. Irwin’s results for pitching moment coefficient increment predictions (Figure 2.27) showed a good match with experimental data across all incidences. This was due to the lack of variation of \( dC_m \) with hole size for a given chordwise location.

A key area of Irwin’s methodology that would have an impact on the accuracy was the use of a single chordwise point to determine the surface pressure coefficients from an equivalent undamaged aerofoil. By using a single point at the centre of a hole, this restricts the accuracy to holes which are small and have minimal variation in pressure distribution across their diameter. If the hole is large or close to the aerofoil pressure peak this method would potentially lead to considerable errors. A more accurate approach would be to either take the average of several points along the hole, or adopt an area weighted integration technique.

It is felt that the heavy reliance on experimental data and the empirical nature of the predictive methodology adversely affects its usability. In order to make use of the technique it would be necessary to build up a sufficiently large data set of coefficient increments from a range of damage cases for the aerofoil in question through either wind tunnel testing or
Figure 2.25: Comparison of experimental and predicted lift coefficient increments obtained by Irwin for a 24%c hole at quarter chord. Data from Irwin [1].

Figure 2.26: Comparison of experimental and predicted drag coefficient increments obtained by Irwin for a 24%c hole at quarter chord. Data from Irwin [1].

Figure 2.27: Comparison of experimental and predicted pitching moment coefficient increments obtained by Irwin for a 24%c hole at quarter chord. Data from Irwin [1].
computational simulations. This may be prohibitively expensive and would increase the
time taken to gain a prediction for damage on a new aerofoil.

In general it is felt that Irwin’s prediction methodology was of good accuracy when applied
to the NACA 64₁-412 aerofoil. However, it has a number of limitations, significantly,
that it was only validated against one aerofoil, and was only valid for holes located at
quarter or mid chord, between 10% chord and 40% chord in diameter. However, given
the methodology’s reliance on pre-existing experimental data for the different damage
configurations, it is felt that this methodology may be limited in validity, and may not be
applicable to alternative aerofoils. If the methodology is not applicable to other aerofoils,
then it would not provide a practical alternative to wind tunnel testing for determining
the aerodynamic effects of a previously untested damage configuration, unless the aerofoil
in question had already undergone significant testing.

2.7.2 Review of Three-Dimensional Methodologies

In order to obtain the coefficient increments for damage on a three-dimensional wing it
may be necessary to carry out testing or a prediction on a two-dimensional wing first. As it
is unlikely the two- and three-dimensional wings will be of the same area it is necessary to
convert the two-dimensional results to the area of the three-dimensional wing. Render et
al [22] published such a method, which converted Irwin’s two-dimensional battle damage
data [1] and validated this against three-dimensional models of three different aspect ratios,
using the same aerofoil and chord as Irwin’s work. It should be noted that this method
of using identical chords resulted in the hole areas being identical on the two and three-
dimensional models. In addition, it was stated in the paper that this method was not
a complete predictive technique but demonstrated a possible basis for predicting three-
dimensional coefficient increments from two-dimensional data.

Render et al used the link identified by Irwin [1] between coefficient increments and the
pressure coefficient difference in the region of the hole, \(dC_P\), which is defined as the difference
between the upper and lower surface pressure coefficients on an undamaged wing at
the chordwise centre of the damage. As with Irwin’s work, the values used were taken at
a single point located at the chordwise centres of the hole.

The methodology by Render et al began by demonstrating that spanwise variation in sur-
face pressure coefficient along a wing could be accounted for by plotting the coefficient
increments against the pressure coefficient difference, \(dC_P\). Irwin’s two-dimensional coef-
icient increments were converted to the three-dimensional wing geometry using Equation
2.4 for \(dC_L\) and similar equations for \(dC_D\) and \(dC_M\).
\[ dC_{L \, 3D} = dC_{l \, 2D} \left( \frac{S_{2D}}{S_{3D}} \right) \]  \hspace{1cm} (2.4)

Where:

- \( dC_{L \, 3D} \) = Three-dimensional lift coefficient increment
- \( dC_{l \, 2D} \) = Two-dimensional lift coefficient increment
- \( S \) = Wing area

The method was extended by plotting coefficient increment areas (see Equations 2.5 to 2.7) against the corresponding \( dC_p \) value, taken at the centre of the hole. This was found to collapse the two- and three-dimensional data well, but no further work was undertaken to extract the three-dimensional coefficients for specific incidences, or to predict intermediate hole sizes.

\[ dC_{L A} = dC_{L} \cdot S \]  \hspace{1cm} (2.5)
\[ dC_{D A} = dC_{D} \cdot S \]  \hspace{1cm} (2.6)
\[ dC_{M A} = dC_{M} \cdot S \cdot c \]  \hspace{1cm} (2.7)

Where:

- \( dC_{L A}, dC_{D A}, dC_{M A} \) = Three-dimensional lift, drag or pitching moment coefficient increment area
- \( dC_{L}, dC_{D}, dC_{M} \) = Three-dimensional lift, drag or pitching moment coefficient increment
- \( S \) = Wing area
- \( c \) = Wing chord

Figures 2.28 to 2.30 from Render et al [22] show the results for the conversion of two-dimensional data to three-dimensional coefficient increment areas, for lift, drag and pitching moment coefficient areas (\( dC_{L A}, dC_{D A} \) and \( dC_{M A} \)) respectively. Three-dimensional data were presented for wings of three aspect ratios, AR6, 8 and 10, with damage located 450 mm from the root on all wings (73%, 55% and 45% span respectively). Through utilising coefficient increment areas and plotting against \( dC_P \), the three figures show that Render et al had removed the spanwise dependency, with reasonable success. However the two-dimensional pitching moment area increments (Figure 2.30) do not match as well with the three-dimensional data, in particular for aspect ratio 6, which was significantly
offset from all other three-dimensional experimental data. It was identified that this was likely due to the differences in model mounting, as discussed in section 2.5, and the previously discussed errors introduced with low aspect ratio half wing models. Despite these errors, it is felt that the accuracy of the predictions supports using this methodology as a starting point to predict three-dimensional coefficient increments from two-dimensional experimental data.

As with Irwin’s two-dimensional prediction method, the use of a single point value for calculating the pressure coefficient difference across a hole raises concerns about the overall accuracy of the prediction. This is of particular concern on the three-dimensional wing, where there will not only be a chordwise variation in pressure but a spanwise variation across the hole that will become more prominent as the hole is moved towards the tip, or increased in diameter. Therefore, a more appropriate method would be to adopt an area weighted technique, which would consider both the chordwise and spanwise pressure variation across each hole. In the figures presented by Render et al, coefficient increments were only plotted against the pressure difference. No further working was undertaken to plot the coefficient increments against incidence. However, due to the use of $dC_P$ to normalise for span it is likely this would not have a significant effect once coefficient increments were plotted against incidence.
Figure 2.28: Lift area coefficient increment against pressure difference at the centre of a hole 450mm from the root on different aspect ratio three-dimensional wings. Data from Render et al [22].

Figure 2.29: Drag area coefficient increment against pressure difference at the centre of a hole 450mm from the root on different aspect ratio three-dimensional wings. Data from Render et al [22].

Figure 2.30: Pitching moment area coefficient increment against pressure difference at the centre of a hole 450mm from the root on different aspect ratio three-dimensional wings. Data from Render et al [22].
Chapter 3

Model Design and Configuration

3.1 Introduction

Testing was carried out on separate two- and three-dimensional models, using different wind tunnels. This section outlines the design criteria and rationale for each model, including any wind tunnel configuration limitations that impacted on the model design.

3.2 Aerofoil Selection

Two key factors drove the aerofoil selection:

- The project sponsors, BAE Systems desired an aerofoil that was close to that used on a current reconnaissance unmanned aerial vehicle (UAV).
- To allow for ease of publishing, an aerofoil which had base data (surface pressure coefficients and profile geometry) in the public domain was desired.

The chosen aerofoil was the NASA LS(1)-0417MOD (its design and construction is shown in section 3.3), which was developed from the LS(1)-0417 aerfoil formerly known as the NASA GA(W)-1. This aerofoil is particularly suited to low-speed general aviation applications, while having superior performance when compared to the older National Advisory Committee for Aeronautics (NACA) aerofoils, including an improved lift to drag ratio and an increased maximum $C_L$. The modification involved re-profiling the contours to reduce the pitching moment coefficient, increasing the forward loading and decrease the aft upper surface pressure gradient. [25].

A key driver in the selection of the aerofoil was the variation in chordwise pressure coefficients between the upper and lower surfaces. Irwin’s work [1] demonstrated that the pressure differential is an important parameter in battle damage. Figure 3.1, which shows data from [25], compares the theoretical pressure distributions from the original LS(1)-0417 to the modified aerofoil. The LS(1)-0417 featured a relatively constant pressure coefficient on both the upper and lower surfaces between approximately 5% and 55% chord, at 0° incidence. This would have resulted in little variation between the aerodynamic coefficients
for different chordwise damage locations. However, the LS(1)-0417MOD showed a more significant change in pressure differential across the same region. This would produce a greater difference in aerodynamic coefficient increments when testing different battle damage cases, thereby improving the range of the planned test investigations. For this reason the LS(1)-0417MOD was chosen.

### 3.3 Two-Dimensional Model Sizing and Design

Two-dimensional testing (see Chapter 4 for the testing techniques) was undertaken in Loughborough University’s Low Turbulence wind tunnel. This imposed the following restrictions and considerations on the model design:

- Working section dimensions of 0.45 m by 0.45 m.
- Maximum operating speed of 37 m/s, with a turbulence intensity of 0.1%.
- Horizontal mounting for incidence control.
- Mounting via two struts and an incidence arm to an under floor balance (the incidence arm typically consisting of two struts to prevent it from affecting any damage flows).
- Convention for this wind tunnel dictated a forward strut on the model be located at the balance reference centre, which is typically at 25% chord, as this is likely to be close to the aerodynamic centre of the aerofoil.
3.3. Two-Dimensional Model Sizing and Design

3.3.1 Sizing the Two-Dimensional Model

During wind tunnel testing it is important to maintain as high a Reynolds number as possible, maximising the accuracy of the data when scaling results to full-sized aircraft [23]. The higher Reynolds numbers produce greater accuracy due to the higher wind tunnel speeds exerting greater forces and moment on the balance. The higher Reynolds number also resulted in reduced viscous effects, due to the thinner boundary layer, thus providing more representative results. The customers’ specifications for the UAV on which this project was based gave a typical design operating Reynolds number range of between 2.3 million and 4.7 million, based upon the wing chord. While it was not possible to match this in the low turbulence wind tunnel, this aided the desire to test at the maximum possible Reynolds number in the wind tunnel.

By matching the Reynolds number, boundary layer conditions are representative and similar. As the Reynolds number reduces, the drag coefficient typically increases, the maximum lift coefficient generally reduces and the lift curve slope also decreases. Additionally, if the Reynolds number is low, boundary layer transition may be delayed and laminar flow separation becomes more likely [23]. From this, it is therefore ideal to operate at as high a Reynolds number as the wind tunnel will allow. For a given wind tunnel speed this therefore corresponds to a larger model size.

The Reynolds number requirements, combined with the maximum operating speed of the wind tunnel helped drive the chord sizing for the model. Irwin’s work [1] demonstrated a Reynolds number of 500,000 was achievable when testing wing models in the Low Turbulence wind tunnel. When run at its maximum speed this yielded a model chord of 200 mm. Barlow et al [23] recommend a maximum ratio of model chord to working section height (c_H) of 0.35 in order to limit blockage effects. The model size for this Reynolds number however gives a \( \frac{c}{H} \) value of 0.44. Barlow et al also recommend that the model frontal area to working section frontal area ratio is between 0.01 and 0.1, but ideally around 0.05. This model, with a 200 mm chord had a ratio of 0.076. While this was higher than the ideal value, it was still within recommended limits.

Ideally, in order to allow for comparison with Irwin’s results [1] without any concerns of Reynolds number effects, the 200 mm chord model size was preferred. However, in order to determine if this larger model was causing too great a blockage in the wind tunnel, it was tested alongside a smaller model of 141 mm chord and 24 mm thickness. This was of the same thickness as Irwin’s models, and was within the limit of \( \frac{c}{H} \leq 0.35 \). However, due to the smaller chord, the maximum achievable Reynolds Number for this model was 300,000. The validation, which is discussed in greater detail in section 4.7, concluded that the 200 mm model could be adequately corrected for wind tunnel blockage effects, and was
thus selected for all two-dimensional testing.

3.3.2 Two-Dimensional Model Construction and Design

A key decision with regards to the model design was whether to use a solid model or a hollow model. The former, traditional approach to wind tunnel testing, gives better resistance to the loads experienced, while the latter, more novel approach, provides a more realistic simulation of the internal structure of an aircraft wing. A compromise was reached between a realistic model of an aircraft wing and model integrity. The model had a hollow centre cavity to represent a typical aircraft wing structure, and solid leading and trailing edges, simulating wing spars, to improve the strength of the model.

Throughout Irwin’s work [1], a significant number of individual wing models were used, with a new one required for each damage case. With the large number of damage cases required for this current project, Irwin’s approach was prohibitively expensive. Therefore, the upper and lower surfaces around the cavity region were made from removable panels.

Figure 3.2 shows a chordwise cut through of the final model design. The solid leading and trailing edge and solid tips around the cavity were manufactured from ProLab65, a machinable synthetic modelling board. The removable panels were moulded fibreglass and finished in resin to ensure a smooth finish and that the outer profile of each panel matched well with the aerofoil profile. The panels were attached to the model by countersunk screws. Figure 3.3 shows a plan view of the model. The chordwise bounds of the cavity (24% chord and 75% chord) were intended to correspond with typical spar locations in an aircraft wing. Four mounting struts (two towards the leading edge, two towards the trailing edge) were connected to the solid section at the spanwise limits of the cavity, so as not to interfere with any flow effects within the cavity.

![Figure 3.2: Chordwise drawing of the wing illustrating the solid leading and trailing edges (hatching), the removable skin panels and the cavity.](image-url)
Figure 3.3: Top view of the two-dimensional model (200mm chord) illustrating the solid frame, damage hole locations and panel attachments. Hatching indicates the solid region, filled in regions indicate the screws.
3.4 Three-Dimensional Model Sizing and Design

The three-dimensional model testing was undertaken in Loughborough University’s 1.3 m x 1.9 m wind tunnel (see chapter 5). This was to accommodate a model of finite aspect ratio (this was not possible in the Low Turbulence wind tunnel) and to allow a higher testing Reynolds number. The wind tunnel could operate at a maximum speed of 45 m/s, with a turbulence intensity of 0.15% [26].

A half model configuration was chosen for a number of reasons. Primarily, there is a well-developed test technique for using half models in the large wind tunnel at Loughborough University. The wind tunnel had been optimised for half model configurations, and previous work [10, 22] had demonstrated the validity and accuracy of the half model technique in this wind tunnel. A half model would be mounted vertically, and use the wind tunnel floor as a reflection plane. This theoretically doubles the effective aspect ratio of the model. A further benefit of the half model configuration is that there are no struts penetrating the airflow. However, due to the boundary layer over the wind tunnel floor, a true reflection plane is not achieved, although the ratio of boundary layer thickness to wing span is likely to be small, thus minimising any possible effect.

3.4.1 Sizing the Three-Dimensional Model

Typically to avoid interactions between the working section walls and the wing tips, the model span is restricted to 75% of the working section width [23]. Based on a working section height of 1.3 m, the half model span is 0.975 m. Samad-Suhaeb [10] demonstrated a model with an effective aspect ratio of 6 would provide satisfactory battle damage data. Given the actual aspect ratio is half of this, the wing chord would therefore be 0.325 m. This gave a Reynolds number of 1,000,000 at the wing tunnel’s maximum speed, and a value closer to the typical design range of the baseline UAV. A relatively small aspect ratio was selected as it was desired to have as large a chord as the wind tunnel balance would permit in order to maximise loads and therefore accuracy of measurements.

3.4.2 Three-Dimensional Model Construction and Design

The three-dimensional model used the same basic design as the two-dimensional model, with a solid leading edge forward of 24% chord and a solid trailing edge aft of 75% chord, and a solid tip. The centre section remained a hollow cavity. A purpose built aluminium block was used to attach the model’s spars to the strut mounted on the under-floor balance. Aluminium spars were used to form the cavity bounds and to transfer the model loads.
3.5. Damage Definitions

through the block onto the balance. The construction of the model is shown in Figure 3.4. The leading edge spar was an I-beam, in order to minimise weight without sacrificing strength, however for ease of manufacturing and to maintain integrity of the solid tips, the beam transitioned to a rectangular cross section in the solid regions. Due to size constraints, the trailing edge spar was a simpler rectangular profile. The solid leading and trailing edges and the tip were manufactured from ProLab65.

The cavity, which for balance tests extended from 15% to 85% span, was bounded by fibreglass panels screwed into the leading and trailing edge spars. The model and cavity extended below the tunnel floor to allow for easy removal of pressure tapping tubes, without disrupting the airflow over the model. This was however sealed to 15% span for balance tests to prevent leakage. Additional screws were affixed into ribs at the chordwise edges of each panel. The ribs ensured that the edges of the panels matched to provide a smooth joint. To maximise the cavity extents, the ribs had cut-outs which were similar in size to the cavity’s internal geometry. Four fibreglass panels were used on each surface, with the panel at the wind tunnel floor permanently fixed in place. The other three panels on each surface were interchangeable between three spanwise locations: “Root” (panel centre at 25% span); “Centre” (panel centre at 50% span); and “Tip” (panel centre at 75% span). The panels were designed such that any panel could be tested at the three different spanwise locations.

3.5 Damage Definitions

As was discussed in chapter 1 a number of factors affect potential damage configurations. A primary factor is the type of damage to be considered. Gunfire damage was selected to maintain similarities with Irwin’s work [1] and to extend Irwin’s findings. Based on studies by Render et al [18] and Robinson and Leishman [9], and in line with studies by Irwin [1], circular holes were assumed to be an adequate approximation of gunfire damage.

Upon consultation with project sponsors, the following variables were selected for this project: hole size, chordwise attack angle (obliquity angle), spanwise attack angle (skew angle), and combined attack angles. These were chosen to explore more “realistic” damage scenarios, which had not been investigated under previous studies. Hole diameters of 5%c, 10%c, 20%c, 30%c and 40%c were selected. These matched diameters used in Irwin’s original tests [1], with 40% chord being the maximum achievable without compromising the structural integrity of the panels. A 5% damage case was also included to investigate potential minimum damage sizes. In varying the chordwise and spanwise attack angles, this moved the upper hole relative to the lower hole along either the chord line or span line.
Figure 3.4: Diagram showing the construction of the finite aspect ratio model.
The range of damage cases that could be considered was restricted by the model geometry. In order to maintain structural integrity it was decided that no spars would be cut by the damage, thus restricting all holes to within the bounds of the cavity and imposing limits on obliquity and skew angles. This is discussed in the following sections.

### 3.5.1 Sign Conventions and Hole Location

The sign convention used for obliquity angles is shown in Figure 3.5, with angles being taken from the vertical. Damage holes had a constant chordwise axis of 50%. This was a compromise between the pressure distribution variation between the upper and lower surfaces and the number of feasible damage cases. If the hole was centred about a location closer to the leading edge, the pressure differential would be greater, thus yielding greater magnitude increments, but would greatly restrict the obliquity angle obtained. It was desired to test as great a range of obliquity angles as possible, which dictated centring the holes about mid chord. Figure 3.6 shows the use of the chord line as a datum for applying the obliquity angle. The intersection of the obliquity angle line with the upper and lower surfaces marked the centre of the hole, as shown by the shaded regions in Figure 3.6.

![Figure 3.5](image)

**Figure 3.5:** Diagram illustrating the obliquity angle convention where “LE” and “TE” are the leading and trailing edges of the wing respectively.

![Figure 3.6](image)

**Figure 3.6:** Diagram illustrating the hole location (20% chord diameter, +60° obliquity, located at 50% chord) from a chordwise view. The solid region indicates the hole, the hatched region the solid leading and trailing edges. The dashed line indicates the chord line and the dotted line the obliquity angle, centred on the chord line.

No sign convention was required for skew angles on the two-dimensional model, due to the lack of spanwise variation in surface pressure. For consistency, however, all holes were skewed in the same direction, with the upper hole to the right hand side of the model.
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Figure 3.7: Diagram illustrating the sign convention for skew on the three-dimensional model, with negative skew shown.

<table>
<thead>
<tr>
<th>Hole diameter</th>
<th>Obliquity, degrees from vertical</th>
</tr>
</thead>
<tbody>
<tr>
<td>5% chord</td>
<td>0°</td>
</tr>
<tr>
<td>10% chord</td>
<td>Valid</td>
</tr>
<tr>
<td>20% chord</td>
<td>Valid</td>
</tr>
<tr>
<td>30% chord</td>
<td>Valid</td>
</tr>
<tr>
<td>40% chord</td>
<td>Valid</td>
</tr>
</tbody>
</table>

Table 3.1: Summary of valid and invalid test cases for various hole diameters (expressed as percentages of chord) for damage centred about the half-chord point.

For the three-dimensional model, due to the spanwise variation in pressure, positive and negative skew angles were required. Therefore, a damage hole with positive skew involved the upper surface hole moving towards the tip, and negative skew moved the upper surface hole towards the root of the wing (see Figure 3.7).

3.5.2 Permissible Damage Hole Locations

Table 3.1 shows the possible damage sizes and obliquity angles and indicates whether each test case is valid. A test case was deemed “invalid” if the hole cuts one or both spars. It should be noted that at larger hole sizes, portions of the upper hole overlapped the lower hole at certain skew and obliquity angles. Table 3.2 summarises all obliquity and skew angles tested for each hole size on the two-dimensional model. The axis of all holes was centred about half chord unless otherwise stated. Due to limited resources, and based on previous work by Irwin [1] it was decided to focus testing on the 20%c hole, as it was felt this was likely to produce both weak and strong jets (thus giving the greatest analysis into damage properties), and allow for the widest range of obliquity and skew angles to be tested. It was felt that additionally testing skew and obliquity angles (where possible) on the extreme cases of 5%c and 40%c would permit sufficient analysis of the trends for the intermediate hole sizes.

Due to wind tunnel availability a limited number of cases from two-dimensional testing
3.5. Damage Definitions

were tested on the three-dimensional model. Therefore, key “extreme” cases only were tested on the three-dimensional model. The three-dimensional model was however designed such that all permissible damage cases from the two-dimensional model could be tested if required. All damage cases were tested at three spanwise centres, 25%, 50% and 75% span. A summary of the three-dimensional test cases is given below:

- 20%c with no obliquity and no skew
- 20%c with ±60° obliquity no skew
- 20%c with ±45° skew and no obliquity
- 40%c with no obliquity and no skew

<table>
<thead>
<tr>
<th>Skew Angle</th>
<th>-60°</th>
<th>-30°</th>
<th>0°</th>
<th>+30°</th>
<th>+60°</th>
</tr>
</thead>
<tbody>
<tr>
<td>0°</td>
<td>5%c, 20%c</td>
<td>20%c</td>
<td>5%c, 10%c, 20%c</td>
<td>20%c, 40%c</td>
<td>20%c centred at 0.35c</td>
</tr>
<tr>
<td>30°</td>
<td>None</td>
<td>20%c</td>
<td>20%c</td>
<td>20%c</td>
<td>None</td>
</tr>
<tr>
<td>45°</td>
<td>5%c, 20%c</td>
<td>None</td>
<td>5%c, 20%c, 40%c</td>
<td>None</td>
<td>5%c, 20%c</td>
</tr>
<tr>
<td>60°</td>
<td>None</td>
<td>None</td>
<td>20%c</td>
<td>None</td>
<td>None</td>
</tr>
</tbody>
</table>

**Table 3.2:** Summary of all test cases undertaken for the two-dimensional testing.
4.1 Introduction

This chapter highlights the testing techniques used for all two-dimensional wind tunnel testing. Throughout this thesis “two-dimensional” refers to battle damage effects on an aerofoil, tested under two-dimensional conditions. When damage is introduced to a wing under two-dimensional conditions, this will generate local three-dimensional flow. Two-dimensional aerofoil testing was undertaken in Loughborough University’s Low Turbulence Wind Tunnel and utilised the under-floor balance to measure lift and drag forces, and pitching moment. These were supplemented with surface flow visualisation and surface pressure measurements. A brief summary of the wind tunnel corrections will be presented in this chapter. The accuracy and validity of the model design will also be discussed, providing comparisons against published data. As was discussed in section 3.3 due to the large model size, compared to the wind tunnel working section cross sectional area, it was necessary to validate the wind tunnel blockage corrections.

4.2 Wind Tunnel Setup

4.2.1 Low Turbulence Wind Tunnel Description

The Low Turbulence wind tunnel is an open return wind tunnel with a closed working section, measuring 0.45 m x 0.45 m. Typical turbulence intensity was approximately 0.1%. The model was mounted on a three-component balance, designed by Aerotech (now ATE Aerotech), situated below the wind tunnel floor. The model connected to the balance by four struts, with the rear pair providing incidence control. The typical maximum operating speed of the wind tunnel was 37 m/s, which yielded a Reynolds number of approximately 500,000 when using a model with a 200 mm chord.

Figure 4.1 shows the wind tunnel configuration with a wing model mounted in the working
section, connected to the under-floor balance. To achieve the required two-dimensional conditions, the model spanned the tunnel working section. The wind tunnel speed was set using a pitot-static probe positioned upstream of the model. The wind tunnel also has a shielded thermocouple mounted in the airstream to measure the air temperature. A GE Sensing RPT410V barometric pressure sensor mounted away from any exhaust flow from the wind tunnel measured ambient pressure.

The balance utilised load cells for each of the three components. Voltages from these were sent to a desktop computer configured for data acquisition, running a custom-designed application created in National Instruments’ “LabVIEW” to view and log all data. Outputs from all sensors were passed into the data acquisition software and logged. The software carried out conversions from voltages and applied calibration corrections where appropriate. The lift component load cell was calibrated to within 0.03% of full scale deflection when a 20 kg load was applied. Pitching moment was calibrated to within 0.04% of full scale deflection under a 400 kg.mm load, and the drag component was calibrated to within 0.05% of full scale deflection under a 2 kg load.
4.2. Wind Tunnel Setup

4.2.2 Model Mounting and Installation

Figure 4.2 shows the model mounted inside the working section. In order to maximise the internal cavity the rear struts were located at 75% chord. Incidence control was provided through a potentiometer and gearbox. The incidence of the model was verified after installation by means of a digital inclinometer, placed on a frame parallel with the model chord and machined to sit flush on the upper surface of the model. The model incidence was verified across the range of testing incidences. The digital inclinometer had a stated accuracy of ±0.1°, which was deemed to be the limiting accuracy of the gearbox and model installation. Backlash in the gearbox and installation of the model inside the tunnel was verified using the same inclinometer, with any variation less than the accuracy of the inclinometer.

The pin joints used to affix the struts to the model caused some degree of friction between the pin and the strut, which resulted in a slight bias of the results for pitching moment. As this was a consistent error in all experiments, and as testing would focus on comparing data with damage present to undamaged data, any error was effectively cancelled. To prevent balance earthing and hysteresis, a nominal gap between each tip of the model and the tunnel sidewalls of approximately 0.5 mm was maintained. This allowed a trade-off between a sufficient gap to prevent touching while still preserving the two-dimensional flow over the wing. Surface flow visualisation and surface pressure measurements indicated that despite the gaps, two-dimensional conditions were maintained up to the onset of stall.

![Figure 4.2: Chordwise view of the two-dimensional model mounted in the low turbulence wind tunnel.](image)
4.2.3 Balance Testing Techniques

Prior to each test run, a set of “wind off” data were obtained for each incidence, to allow for static loads on the balance to be eliminated. Each data set of “wind off” and “wind on” data were collected over a range of incidences, from $-8^\circ$ to $+12^\circ$ in $2^\circ$ increments, and from $+13^\circ$ to $+17^\circ$ in $1^\circ$ increments in order to give increased resolution of results at the anticipated onset of stall. The incidence was only moved in one direction (increased) during test runs to eliminate any hysteresis and backlash from the incidence mechanism gearbox. Verifications to ensure no hysteresis was present were performed at the end of each test run.

At least two incidence traverses with the wind tunnel running were taken. Anomalies were removed from the data set and the remaining data were averaged. The wind off data were subtracted and strut and wind tunnel corrections were applied to the data. Corrections are discussed in section 4.6.

4.3 Boundary Layer Transition Strips

A boundary layer transition strip was included in order to avoid any potential Reynolds number effects, particularly evident with testing at low Reynolds numbers. By forcing transition, this ensured the region downstream of the transition strip remained turbulent at all incidences. This however introduced a drag penalty, which would effectively be cancelled out when comparing damaged data to undamaged data. Based on recommendations in existing literature [23], and NASA data for the aerofoil [25], the transition strip used 80-grit carborundum grains. The 3 mm wide strip was located at 7.5% chord to correspond to NASA’s original testing. NASA tested over a wide range of Reynolds numbers, maintaining a transition location of 7.5% chord, although varying the transition height accordingly. The carborundum grains were affixed to the model using a spray adhesive. The strip was only applied to the upper surface of the model. It was found that a strip had minimal effect on the lower surface and was therefore omitted. Although no simulations using software such as X-Foil were undertaken to determine the location of transition, surface flow visualisation studies showed the transition strips worked successfully over the incidence range of interest by demonstrating transition occurred at the location of the strips. Without the strip, natural transition was seen to occur at approximately half chord at lower incidences, migrating forwards to approximately 10% chord as the incidence increased prior to stall.
4.4 Surface Flow Visualisation

In order to develop a better understanding of the flow physics present with different damage cases, and to help explain trends in the balance data, surface flow visualisation techniques were utilised. The flow visualisation mixture consisted of paraffin, white titanium dioxide powder and linseed oil. The mixture was applied normal to the flow direction to allow brush strokes to be more easily distinguished from flow features. Flow visualisation experiments were recorded on a high definition digital video camera. All photographs or videos were taken while the tunnel was running to prevent the mixture distorting under gravity.

4.5 Surface Pressure Tappings

4.5.1 Pressure Tapping Setup

Surface pressure measurements were taken on an undamaged condition and four damaged cases. Using a similar arrangement to Irwin’s work [1], ten chordwise pressure taps were placed on the panel at fixed intervals of 10 mm. These were positioned at five spanwise locations: 0R, 0.5R, 1.5R, 2.5R and 5R, where ‘R’ is the radius of the damage hole. The tappings were kept in the same positions irrespective of the damage case to minimise variables and to provide a constant reference. As shown in Figure 4.3, only one half of the model span was pressure tapped, due to the two-dimensional nature of the model. A summary of the chordwise locations of each tapping is given in Table 4.1. No tappings were placed in the area removed by the hole. Due to restrictions in the model design, it was necessary to locate the taps on the upper and lower panels at different chordwise positions. For simplicity, only the removable panels were pressure tapped. This was due to the flow visualisation showing the primary area of focus to be in close proximity to the hole. Figure 4.4 shows the experimental setup, with the pressure tapping tubes exiting the wind tunnel. An oversized slot through the model and wind tunnel door was required for installation of the tubes, which was then sealed with plasticine to prevent leakage.

The pressure tappings consisted of 1.6 mm diameter brass tubes bonded onto the removable panels such that they were flush with the outer surface, in order to minimise any wakes or disruption resulting from the tappings. Plastic tubing connected the tappings (maximum 20 at any one time: ten on the upper surface, ten on the lower surface) into array of four Pressure Systems 16TC/DTC pressure scanners. These were connected to a Chell CANdaq data acquisition unit, which allowed the data to be logged to a computer. The data acquisition software sampled each pressure tapping 8,192 times over a period of approximately 30 seconds. The pressure scanners had a nominal accuracy of ±0.0696
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Figure 4.3: Location of the pressure tappings on the two-dimensional model. An example damage hole is shown by the dashed line.

<table>
<thead>
<tr>
<th>Tapping Number</th>
<th>Upper panel chordwise location, x/c</th>
<th>Lower panel chordwise location, x/c</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.275</td>
<td>0.255</td>
</tr>
<tr>
<td>2</td>
<td>0.325</td>
<td>0.305</td>
</tr>
<tr>
<td>3</td>
<td>0.375</td>
<td>0.355</td>
</tr>
<tr>
<td>4</td>
<td>0.425</td>
<td>0.405</td>
</tr>
<tr>
<td>5</td>
<td>0.475</td>
<td>0.455</td>
</tr>
<tr>
<td>6</td>
<td>0.525</td>
<td>0.505</td>
</tr>
<tr>
<td>7</td>
<td>0.575</td>
<td>0.555</td>
</tr>
<tr>
<td>8</td>
<td>0.625</td>
<td>0.605</td>
</tr>
<tr>
<td>9</td>
<td>0.675</td>
<td>0.655</td>
</tr>
<tr>
<td>10</td>
<td>0.725</td>
<td>0.705</td>
</tr>
</tbody>
</table>

Table 4.1: Chordwise tapping locations for upper and lower panels on the two-dimensional model.

mmH₂O. Unused brass taps remained on the panels during pressure testing.

4.5.2 Data Collection and Reduction

Surface pressure data were recorded at each incidence used for balance testing. The data collected for each tapping were averaged to a single value with anomalous data points omitted. Data were post-processed to calculate pressure coefficients, $C_p$ (Equation 4.1), where $p_{tap}$ is the pressure recorded by the tapping and $p_{static}$ is the static pressure from the wind tunnel’s pitot-static system. No blockage corrections were applied to the pressure data, as will be discussed in the following section.

$$C_p = \frac{p_{tap} - p_{static}}{\frac{1}{2} \rho V^2}$$  \hspace{1cm} (4.1)
4.6 Two-Dimensional Wind Tunnel Corrections

It is necessary to apply corrections to any balance measurements, because the flow over a model inside a closed wind tunnel is different to that of a model in free air. The corrections adjust the data to effective “free air” values [23]. The corrections take the form of two components: lift interference corrections and blockage corrections. Due to the model size relative to the wind tunnel, simplified methods such as Engineering Sciences Data Unit (ESDU) [27] were not valid. ESDU limited the model chord to working section height ($c/H$) to less than 0.35 (this configuration had a $c/H$ value of 0.44). The ESDU methodology was a simplified approach taken from AGARD [28], which in turn simplified equations from Goldstein [29, 30]. The AGARD corrections were selected as they did not have the $c/H$ restriction that ESDU had, as involved a simpler approach than Goldstein’s original work, while remaining valid for the model.

To maintain accuracy for relatively thick aerofoils, the AGARD method retained the fourth power of $c/H$ as used by Goldstein [30]. It was stated that for the lift interference corrections, this power cannot be ignored when $c/H$ is greater than 0.3. The AGARD method assumed the model was small in comparison to the working section, but unlike the ESDU corrections, the model fell within the stated criteria. In order for the lift interference corrections to be valid, the following general assumptions were made:

- The model is mounted in the centre of the wind tunnel working section.
- The flow was assumed to be incompressible.

Blockage corrections (comprising of wake and solid blockage corrections) were applied to
drag coefficients and kinetic pressure corrections. AGARD [28] suggested that to account for the change in the blockage by the model due to changes in incidence, a scale factor proportional to $\alpha^2$ is applied to the solid blockage term, this assumes flow is incompressible. AGARD further suggest that for compressible flow, an additional factor in the form of the Prandtl-Glaurert compressibility parameter, $\beta$ can be included. Given $\beta$ was equal to 0.994 during testing, this was omitted and the flow assumed to be incompressible. Further assumptions were made for the blockage corrections:

- The model must be mounted in the centre of a closed wind tunnel.
- Wake and solid blockage corrections are assumed to be independent of each other and independent of the lift.
- The lift is not large. AGARD do not qualify this further, and it was assumed that the lift generated was not “large”.
- Blockage is assumed to only influence the longitudinal component of the flow around the model.

Table 4.2 summarises the changes to the aerodynamic coefficients from the wind tunnel corrections at a selection of incidences. The coefficient corrections are also expressed as a percentage of the uncorrected value. A summary of the AGARD equations used for correcting two-dimensional data are given in Appendix B.

<table>
<thead>
<tr>
<th>Incidence</th>
<th>$\Delta \alpha$</th>
<th>$\Delta C_l$</th>
<th>$\Delta C_d$</th>
<th>$\Delta C_m$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0°</td>
<td>+0.0022°</td>
<td>-0.0155 (5.8%)</td>
<td>-0.0009 (4.1%)</td>
<td>+0.00389 (5.8%)</td>
</tr>
<tr>
<td>+12°</td>
<td>+0.496°</td>
<td>-0.0980 (7.5%)</td>
<td>-0.0065 (7.3%)</td>
<td>+0.0132 (15.7%)</td>
</tr>
<tr>
<td>+17°</td>
<td>+0.696°</td>
<td>-0.137 (9.5%)</td>
<td>-0.0177 (10.5%)</td>
<td>+0.0167 (16.6%)</td>
</tr>
</tbody>
</table>

*Table 4.2:* Changes to the aerodynamic coefficients from the two-dimensional wind tunnel corrections.

Few methods of correcting pressure coefficients for the effects of wind tunnel blockage exist. Correction methods presented by AGARD [28] were analysed but were found not to suit the testing approach used. Given that the blockage, and therefore dynamic pressure, vary along the chord of the model it was not possible to generate a single correction value. Corrections were calculated, with several assumptions made, which yielded changes of around 1 – 2%. Given the small degree of correction and issues in recording reliable data to use in the corrections (only pressure data could be recorded, as the setup shown in Figure 4.4 effectively earthed the balance, preventing valid measurements from being taken), all pressure data remain uncorrected.

The struts used to mount the model to the balance, were also subjected to aerodynamic forces from the airflow, particularly drag. A simple method, also used by Irwin [1], was
used to correct for any strut forces. The model was removed from the struts, and the struts run in the wind tunnel at the testing speed to collect lift, drag and pitching moment force values for the struts at each incidence tested. These were then subtracted from the forces obtained during tests with the model. After strut corrections had been applied, the AGARD correction method was used. Corrected data from damaged and undamaged configurations were used to obtain the coefficient increments using Equations 2.1 to 2.3 for lift, drag and pitching moment coefficient increments respectively.

4.7 Model Size Validation

In order to validate the AGARD corrections discussed in section 4.6, and to confirm the acceptability of using a large model, a smaller model of the same aerofoil was tested. The smaller model was of such a size that it conformed to the previously-discussed blockage constraints [23], having a chord of 141 mm and a thickness of 24 mm (the same thickness as models used by Irwin [1] in the same wind tunnel). This therefore gave a \( \frac{c}{T} \) value of 0.313, which was within the generally-accepted limits. However, due to the reduced chord, the maximum Reynolds number achievable was 300,000. At the time of the tests, the hollow model had not been manufactured, so two solid models, with 200 mm and 141 mm chords were used instead.

The comparison tests were carried out with and without transition strips. When the transition strips were installed this gave a better correlation between the two sets of data, reinforcing the proposal to force boundary layer transition. Figure 4.5 shows the variation of lift coefficient for the two models, with transition forced. A reasonable match is shown between the two sets of data, within experimental repeatability bands. An analysis of the experimental repeatability using the hollow model is provided in section 4.8. Figure 4.6 shows the drag polars for both models. In the centre region (\( C_l = 0.2 - 1.2 \), corresponding to incidences between -2\(^\circ\) and +8\(^\circ\)), there is a good match between the two models. Figure 4.7 shows the variation in pitching moment coefficient with incidence. Unlike lift and drag coefficients, no convergence was witnessed with \( C_m \), instead the curves displayed different shapes, but following similar trends. This offset in pitching moment was noted to be due to a manufacturing error on the 200 mm model, which placed the struts further forward than the design location. The results shown were mathematically corrected by adjusting the moment arm, although this did not completely remove the error. Due to time constraints it was not possible to modify the model and re-test it.

The deviations witnessed in the three figures at higher incidences (above approximately 12\(^\circ\) incidence) occurred at stall onset. This will have an adverse effect on experimental
repeatability, reducing confidence in the results beyond this point. In addition, the wind tunnel corrections used were not valid for separated flow, and as such no comparison is required between the models post-stall (approximately 12° incidence). It was noted that the trailing edge of the smaller model was prone to flutter at higher incidences. This was not present on the larger model, due to the thicker trailing edge. This would account for the decrease in lift and increase in drag at the higher incidences with the flutter further destabilising the air flow.

4.8 Experimental Repeatability Analysis

4.8.1 Factors Affecting Experimental Repeatability

The following is a brief review of the key factors that may affect experimental repeatability and the estimated significance. This review is not exhaustive.

Model Removal and Re-installation

It was necessary at times to remove the entire model from the wind tunnel. This typically occurred after flow visualisation experiments when the model required cleaning or when the transition strip needed to be re-applied. To minimise repeatability errors a technique was developed to ensure accurate model placement within the tunnel. This consisted of temporarily affixing metal shims to the tunnel walls and ensuring the model was pushed firmly up against the shims (the shims were removed prior to testing). This gave a constant gap to the tunnel side wall. After each re-installation, a test run was carried out and compared with a known datum to ensure accurate model placement and consistent data. The model incidence was verified across the incidence range using a digital inclinometer, with a resolution of ±0.1°. A template, manufactured from the same computer model of the aerofoil was used to ensure the inclinometer was accurately seated on the model, and was parallel to the chord. Tabs on either end of the template ensured accurate placement onto the model.

Transition Strip

After flow visualisation experiments it was necessary to remove and reapply the transition strip, since the mixture occasionally collected in the grit and reduced its performance. After the strip had been reapplied, it was visually inspected and the performance of the model tested and verified against a datum case. It was considered that application of the transition strip had a high level of repeatability.

Use of Different Panels

The panels were manufactured by hand laying of fibreglass. Each set was manufactured
4.8. Experimental Repeatability Analysis

Figure 4.5: AGARD-corrected [28] lift coefficient variation for the 200mm and 141mm models at Re = 300,000, with transition strips installed.

Figure 4.6: AGARD-corrected [28] drag coefficient variation for the 200mm and 141mm models at Re = 300,000, with transition strips installed.

Figure 4.7: AGARD-corrected [28] pitching moment coefficient variation for the 200mm and 141mm models at Re = 300,000, with transition strips installed.
from the same moulds. While the aerodynamic surface of each panel was accurate, small variations in panel thickness did occur. All panels were inspected prior to testing and unacceptable panels were either modified or discarded. The checks consisted of both visual checks and aerodynamic checks. Results from each batch of undamaged panels were compared to the previous batches, and outliers were either modified to bring the results in line or rejected.

**Incidence Mechanism**

The incidence mechanism on the balance consisted of a gearbox and potentiometer, controlled by a thumb-screw. The mechanism was assessed to have a tolerance better than $\pm 0.1^\circ$ (the accuracy of the inclinometer used to verify the incidence). In order to remove any possible effects from backlash in the gearbox, the incidence was only adjusted in one direction during testing.

### 4.8.2 Analysis of Undamaged Panel Repeatability

Before any damage holes were added to a panel, it was tested in an undamaged state. Generally, it was found that across all panels there was a good level of experimental repeatability. Between some panels being tested, the model had also been removed from the tunnel. From the data, an “average” panel was determined and the upper and lower limits across the incidence range for all panels were calculated. This then quantified the tolerance and repeatability for the experimental setup.

Figures 4.8 to 4.10 show the repeatability bands for an undamaged wing configuration, plotted with averaged data from all panels. The repeatability bands represent the maximum and minimum variations at each incidence across the range of panels tested. It should be noted that for clarity, the incidence repeatability bands are not shown. As can be seen in Figure 4.8, the experimental repeatability for the lift coefficient data were deemed acceptable, until the onset of stall (beyond $12^\circ$). Due to the nature of balance testing, this made it difficult to get repeatable data from the stalled aerofoil. Figure 4.9 shows the repeatability of the drag coefficient. This repeatability band (which will be affected partly by the repeatability of $C_l$, due to both being used in plotting the graph) is minimal and deemed at an acceptable level up to values of $C_l = 1.0$ ($8^\circ$ incidence). Above this, the aerofoil begins to approach stall, where the drag levels vary greatly due to the separated flow. Figure 4.10 shows the repeatability of the pitching moment coefficient. As mentioned, this has a wider spread than other data due to the friction in the pin joints, but the repeatability is still considered reasonable and on a par with previous work [1], which used the same balance and wind tunnel. The repeatability bands resulting from variations across all the undamaged panels are given below. These bands are only applicable for
testing carried out below stall ($12^\circ$).

- Incidence ($\alpha$) repeatability: $\pm 0.1^\circ$
- $C_l$ repeatability: $\pm 0.018$
- $C_d$ repeatability: $\pm 0.0016$
- $C_m$ repeatability: $\pm 0.0021$

### 4.8.3 Impact of Repeatability on the Coefficient Increments

Since the data would be presented in the form of coefficient increments (see Equations 2.1 to 2.3 for definitions), the undamaged data from a specific panel (the same panel as the one that had damage applied), and not from an averaged data set would be used for calculating the coefficient increments. Therefore the repeatability band across the range of panels would not be applicable. Only the repeatability resulting from the removal and installation of the model and panels would be relevant. Therefore repeatability bands for the coefficient increments were obtained by assessing the typical ‘entry repeatability’ of a single panel, with these bands then doubled (one ‘entry repeatability’ band for the undamaged panel, and a second for testing the damaged panel). Figures 4.11 to 4.13 show the repeatability bands on two damage cases for the lift, drag and pitching moment coefficient increments. Data from a 5% and a 20% chord diameter hole, both with -60$^\circ$ obliquity, are shown.

Figure 4.11 shows the repeatability bands superimposed on the lift coefficient increment values. These are shown to have a potentially significant impact when considering 5% chord diameter holes, due to the very small increments produced. The increments were often less than the repeatability at a given incidence. This therefore reduced the confidence in data obtained from 5% chord diameter holes. The drag (Figure 4.12) and pitching moment coefficient increments (Figure 4.13) showed similar trends to the lift coefficient increment repeatability, with the repeatability bands increasing beyond stall (approximately $12^\circ$), and remaining generally constant below. For all three increments the onset of stall was likely to produce significant repeatability concerns, due to the difficulties in accurately measuring unstable flow on two different test runs (undamaged and damaged cases). The levels of repeatability were deemed acceptable and comparable to the values from the previous section, although care must be taken when comparing different damage combinations using holes of 5% chord diameter, since the increments at times were shown to lie within the experimental repeatability. This applied to all coefficient increments, but was most applicable to the lift coefficient increment.
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Figure 4.8: Error bars indicating the experimental repeatability for the two-dimensional lift coefficient against the average of all data for an undamaged wing at \( \text{Re} = 500,000 \).

Figure 4.9: Error bars indicating the experimental repeatability for the two-dimensional drag coefficient against the average of all data for an undamaged wing at \( \text{Re} = 500,000 \).

Figure 4.10: Error bars indicating the experimental repeatability for the two-dimensional pitching moment coefficient against the average of all data for an undamaged wing at \( \text{Re} = 500,000 \).
4.9 Validation of the Panel Technique

A summary of the coefficient increment repeatability bands are given below:

- Incidence ($\alpha$) repeatability: ±0.2°
- $dC_l$ repeatability: ±0.015
- $dC_d$ repeatability: ±0.0013
- $dC_m$ repeatability: ±0.0020

4.9 Validation of the Panel Technique

The suitability of the undamaged hollow wing was assessed using tests carried out, with the transition strip installed on the upper surface, at a Reynolds Number of 500,000. Results were compared with data from NASA [25], obtained at a Reynolds number of 2,000,000 in two-dimensional conditions, with transition strips installed on the upper and lower surfaces. It should be noted that the NASA results were obtained from pressure measurements, and not balance measurements. Experience suggests that there are often small differences between coefficient data obtained from the two methods. For example, drag coefficients measured from a pressure rake compared to those measured from a balance tend to be smaller in magnitude, partly due to the effects of skin friction, which cannot be detected by a pressure rake. All balance data collected for this project have been corrected for blockage effects using the method outlined in Section 4.6; the NASA data are uncorrected – given the size of the NASA model versus the size of the wind tunnel, it is unlikely to have a significant correction due to blockage effects.

4.9.1 Balance Data Validation

In terms of the lift coefficient, reducing the Reynolds number would be expected to reduce both the lift curve slope and the maximum lift coefficient. In Figure 4.14, this is seen to be the case when comparing the hollow wing with the NASA data. There is also a small offset in the zero lift angle (increased by approximately 0.5°) for the hollow model. This cannot be fully accounted for by experimental repeatability, and is likely to be due to the edges of the panels introducing small geometric changes in the model profile. Barlow et al [23] state that with an increase in Reynolds number, the lift curve slope should increase in gradient slightly and the stall become more abrupt. It would have been anticipated that the transition strip would remove such effects, however given that Figure 4.14 demonstrates these trends it is likely that the transition strip was not working fully as designed, with some residual Reynolds number effects remaining.
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Figure 4.11: Error bars indicating experimental repeatability for the lift coefficient increment for two damage cases.

Figure 4.12: Error bars indicating experimental repeatability for the drag coefficient increment for two damage cases.

Figure 4.13: Error bars indicating experimental repeatability for the pitching moment coefficient increment for two damage cases.
In terms of drag, reducing Reynolds number will increase the drag coefficient. It would be expected that the transition strip would remove any major Reynolds Number effects due to laminar flow. However, as stated by Barlow et al [23], the change in $C_d$ resulting from the Reynolds Number changing from 500,000 to 2,000,000 would be approximately 10%, when a turbulent boundary layer is present. Given that the experimental difference (see Figure 4.15) was closer to 30%, it is therefore likely other factors are affecting the drag coefficient. These are likely to include any small steps introduced by the panel, resulting in small but abrupt increases in aerofoil thickness at the leading edge of the panel. Additionally, the height of the transition strip and the different methods of obtaining $C_d$ data would have resulted in skin friction not being considered for the NASA data. Additionally, the skin friction would have increased with the reducing Reynolds number for the hollow model. While the blockage of the low turbulence wind tunnel was considerably greater than for the NASA data, this was unlikely to affect the results, as it had been demonstrated in section 4.6 that the effects of blockage could be removed using corrections.

Figure 4.16 shows the pitching moment coefficient variation with incidence for the NASA and hollow models. It is acknowledged that the pitching moment is the weakest of the three balance measurements due to the friction in the joints on the model struts. The NASA model would not have suffered such issues. The differences are also likely to be compounded by the small geometric changes in the profile at the edges of the panels.

It is important to recognise that all battle damage results will be presented in terms of coefficient increments. The effects discussed are likely to be present and reasonably consistent for all tests and will thus be eliminated when data are expressed in terms of coefficient increments. This consistency was confirmed in section 4.8, where the hollow wing model showed acceptable repeatability for a number of different panels.

### 4.9.2 Surface Pressure Data Validation

The data from the upper surface at $0^\circ$ and $8^\circ$ (Figures 4.17 and 4.18) incidence shows a reasonable match with the published NASA data [25]. However, the lower surface pressure data for both incidences showed a generally constant offset. This offset was most prominent at the extents of the experimental data, close to the leading and trailing edges of the panel. The model was inverted, such that the struts were effectively on the upper surface, and a transition strip installed on the lower surface. The discrepancies were still evident, which concluded that the offset was due to the presence of the lower panel, and not the struts. The good match across the centre chordwise region suggested the panel was accurately manufactured and conforming to the wing profile, and a step was present at the leading edge of the lower panel.
Figure 4.14: Lift curve slope comparison between the hollow model and NASA data, both with transition installed. NASA data from [25].

Figure 4.15: Lift/drag curve comparison between the hollow model and NASA data, both with transition installed. NASA data from [25].

Figure 4.16: Pitching moment coefficient comparison between the hollow model and NASA data, both with transition installed. NASA data from [25].
When considering the spanwise variation in pressure coefficients, it can be seen that the repeatability across each spanwise tapping was very high for both incidences, with the points for each spanwise location generally collapsing on each other well. The spanwise consistency of the data across both upper and lower panels was very encouraging, indicating across the region of the panel the model was conforming to two-dimensional flow conditions. It also suggests that there are no concerns about the panels deforming (particularly at $8^\circ$ incidence, which shows less spread across the spanwise locations than at $0^\circ$) during testing and distorting the aerofoil profile.

In addition, the surface profile of the model was accurately measured and compared with the original drawings. It was found that the solid regions of the wing matched well with the original drawings. No twist was identified across the span of the model. The only issue identified was a reduction in thickness towards the trailing edge, due to manufacturing limitations. This was most noticeable on the lower surface. This would have had a small effect on the airflow, similar to changing the camber of the wing, which would have altered the lower surface pressure coefficients. The upper surface panel was seen to follow the profile of the drawings reasonably well. A ‘step’ change in the surface profile was however identified at the leading edge of the lower panel, where there was a uniform change in thickness of approximately 0.5 mm above the ideal wing surface. This was due to the leading edge of the panel not sitting flush with the model and being raised relative to the wing, for the span of the panel. This would have further altered the aerodynamic profile of the lower surface, likely contributing to the change witnessed in the pressure coefficients. As no pressure taps were placed on the solid region of the wing the true effect of this step cannot however be proven.
Figure 4.17: Spanwise variation in pressure coefficients for an undamaged wing at $0^\circ$ incidence. NASA data from [25], at $Re = 2,000,000$. 

Chapter 4. Two-Dimensional Testing Techniques and Validation
Figure 4.18: Spanwise variation in pressure coefficients for an undamaged wing at $+8^\circ$ incidence. NASA data from [25], at $Re = 2,000,000$. 
5.1 Wind Tunnel Setup

5.1.1 Wind Tunnel Description

Loughborough University’s Large Wind Tunnel has a closed working section with a height of 1.3 m and a width of 1.9 m. As with the low turbulence wind tunnel, the large wind tunnel has an open circuit, although as shown in Figure 5.1 from Johl et al [26], this adopts a ‘U-shape’, due to space constraints. Airflow enters through the bellmouth inlet, and flows through the contraction into the working section. Air exits via the fan and diffuser. The maximum operating speed is 45 m/s, which gives a Reynolds number of 1,000,000 for a 325 mm chord model. Turbulence intensity is 0.15% at the centre of the working section.

A six-axis under floor balance provides measurements for lift, drag and side forces, and pitch, roll and yaw moments, with an accuracy better than 0.01% full scale deflection for all axes. However, for the purposes of this project, only the lift, drag and pitching moment channels would be considered. The wind tunnel floor incorporated a turntable, which

Figure 5.1: Loughborough University’s Large Wind Tunnel. Image from [26].
allowed the model incidence to be controlled to within $\pm 0.1^\circ$, with an accuracy greater than $\pm 0.001^\circ$.

### 5.1.2 Model Mounting and Installation

Figure 5.2 shows the half wing model mounted vertically inside the working section, looking downstream with the lower surface visible. In the photograph, the damage is located on the centre spanwise panel.

It was necessary to preserve a gap between the turntable and the model to prevent earthing of the balance. To avoid airflow near the junction of the wing root and the floor leaking into the working section, an approach similar to that of Malik and Render [31] was used where the gap was offset from the leading edge of the model by 100 mm. A dummy floor was installed on the model to provide this offset, which had a nominal gap of 5 mm to the tunnel floor. This is shown in Figure 5.3, which shows one half of the turntable installed around the dummy floor. Although the drag of the dummy floor was measured by the balance as well it was felt this would be negligible, and would fall within the experimental repeatability bands. In addition, the presence of the floor would be constant across all configurations and thus essentially removed when comparing damaged configurations to undamaged configurations.

It should be noted that mounting the model vertically presented challenges with the surface flow visualisation. This made the mixture prone to running under gravity, potentially distorting any results. Where possible photographs were taken with the wind tunnel running, although higher quality photographs necessitated entering the working section post-run.

### 5.2 Boundary Layer Transition Strips

As with the two-dimensional model boundary layer transition was forced. A similar size transition strip was used, of width 3 mm, located at 7.5%c, thus forcing transition at the same chordwise location as the two-dimensional testing. However, the height of the strip required resizing in order to facilitate the increase in Reynolds number. This was scaled according to Barlow et al [23] and resulted in a transition strip which consisted of 56 grit carborundum grains for testing undertaken at a Reynolds number of 1,000,000.
5.2. Boundary Layer Transition Strips

Figure 5.2: Three-dimensional model mounted inside the Large Wind Tunnel (lower surface shown).

Figure 5.3: Dummy floor installed on the three-dimensional model, with one half of the turntable insert installed on the right.
5.3 Surface Pressure Tappings

Surface pressure data were obtained using the same Chell CanDAQ pressure scanner as in the two-dimensional testing. Pressure measurements were taken on undamaged panels only. The tappings were located at five spanwise and ten chordwise locations, with the chordwise locations corresponding to those used in the two-dimensional pressure tests. In order to locate the forward tapping on the upper surface closer to the pressure peak, the upper surface tapping locations on the three-dimensional wing were at the same chordwise locations as the lower tappings on the two-dimensional wing, and vice versa. Tappings were not placed on the solid regions of the model, or at the rib locations, to preserve the model integrity. The tapped panels were placed at each of the three spanwise panel locations shown in Figure 3.4 and discussed in section 3.4.2.

Figure 5.4 illustrates the locations of the pressure tappings on the upper surface panel, with the dashed lines outlining the cavity bounds, and Table 5.1 summarises the chordwise locations of the tappings on each panel. The spanwise locations for the tappings at each of the three panel locations are given in Table 5.2 for the upper and lower panels. These consisted of a centreline set, two sets centred at each of the 45° skew holes, and two sets at the outboard edges of the panels. Chordwise tappings are indicated by “Cx”, where ‘x’ is the tapping number (1 is nearest the leading edge, 10 is nearest the trailing edge). Similarly, the spanwise tappings are labelled “SPx”, where ‘x’ is the spanwise tapping number (1 nearest the root, 5 nearest the tip).

Due to the increased size of the model longer tubes connecting the pressure tappings to the scanner would be required (approximately twice the length of those used in two-dimensional testing). Prior to pressure data being collected an investigation into settling time was undertaken, which found a minimum settling time of 45 seconds was required after changing wind tunnel speeds and model incidences.

5.4 Three-Dimensional Wind Tunnel Corrections

As with the two-dimensional model, corrections are required to return the values measured for the three-dimensional model to free air conditions. However, a different method was required, as the formulae used for the two-dimensional testing were only valid for wings under two-dimensional conditions. These consist of lift interference and blockage corrections. The lift interference results from the upwash on the wing, which becomes more significant towards the wing tip. AGARD [28, 32] and ESDU [33] both provide corrections for lift interference that are valid for half-wing configurations. The ESDU method provides
5.4. Three-Dimensional Wind Tunnel Corrections

Figure 5.4: Diagram of an upper surface panel on the three-dimensional model showing the spanwise locations of the pressure tappings.

<table>
<thead>
<tr>
<th>Tapping Number</th>
<th>Upper surface chordwise location, %c</th>
<th>Lower surface chordwise location, %c</th>
</tr>
</thead>
<tbody>
<tr>
<td>C1</td>
<td>0.255</td>
<td>0.275</td>
</tr>
<tr>
<td>C2</td>
<td>0.305</td>
<td>0.325</td>
</tr>
<tr>
<td>C3</td>
<td>0.355</td>
<td>0.375</td>
</tr>
<tr>
<td>C4</td>
<td>0.405</td>
<td>0.425</td>
</tr>
<tr>
<td>C5</td>
<td>0.455</td>
<td>0.475</td>
</tr>
<tr>
<td>C6</td>
<td>0.505</td>
<td>0.525</td>
</tr>
<tr>
<td>C7</td>
<td>0.555</td>
<td>0.575</td>
</tr>
<tr>
<td>C8</td>
<td>0.605</td>
<td>0.625</td>
</tr>
<tr>
<td>C9</td>
<td>0.655</td>
<td>0.675</td>
</tr>
<tr>
<td>C10</td>
<td>0.705</td>
<td>0.725</td>
</tr>
</tbody>
</table>

Table 5.1: Chordwise pressure tapping locations for upper and lower panels on the three-dimensional model.

<table>
<thead>
<tr>
<th>Tapping Number</th>
<th>Spanwise location on root panel, %b</th>
<th>Spanwise location on centre panel, %b</th>
<th>Spanwise location on tip panel, %b</th>
</tr>
</thead>
<tbody>
<tr>
<td>SP1</td>
<td>0.15</td>
<td>0.40</td>
<td>0.65</td>
</tr>
<tr>
<td>SP2</td>
<td>0.2184</td>
<td>0.4684</td>
<td>0.7184</td>
</tr>
<tr>
<td>SP3</td>
<td>0.25</td>
<td>0.50</td>
<td>0.75</td>
</tr>
<tr>
<td>SP4</td>
<td>0.2616</td>
<td>0.5316</td>
<td>0.7616</td>
</tr>
<tr>
<td>SP5</td>
<td>0.35</td>
<td>0.60</td>
<td>0.85</td>
</tr>
</tbody>
</table>

Table 5.2: Spanwise pressure tapping locations on the three-dimensional model.
a simpler approach than the AGARD method, using tables of data obtained from a large quantity of experiments to provide constants for equations, whereas the AGARD method uses more complex integrations and analysis of the vortex profile across the wing. Given the simplicity of the three-dimensional model (unswept and untapered), and its relatively small size compared to the wind tunnel working section, it was desired to use the simpler ESDU corrections. The ESDU lift interference corrections were only applied to incidence, lift and drag; pitching moment coefficients remained uncorrected.

The ESDU corrections stated geometric constraints for the model and wind tunnel, with key constraints summarised below. As these were all met, it was deemed that the ESDU corrections would be valid for use on the three-dimensional model.

- Model span (0.975m) to tunnel width (physical tunnel height, due to the model being mounted vertically; 1.3m) ratio between 0.4 and 0.8 (the model gave a ratio of 0.75).
- The ratio of wind tunnel working section height (equal to the physical width of 1.9m, due to the vertically-mounted model) to width (double the physical height of 1.3m due to the half model configuration [33]) must lie between 0.7 and 1.43 (the wind tunnel had a ratio of 0.731).

AGARD’s methods for wake blockage corrections [28] were used. The wake blockage correction is of similar form to the two-dimensional corrections but considered model span. However, it should be noted that the blockage is dependent on the drag produced by the model, which is in turn a function of the tip vortex. AGARD therefore noted that since the blockage is independent of lift (“assuming the lift is not large” [28]), the drag at zero lift should be used. This is not valid, however, for separated flow. As the wind tunnel is sufficiently large compared to the model, it is assumed the addition of a damage jet would have negligible additional blockage effects.

The AGARD corrections for solid blockage, however, were not valid for the model, as these were valid only for “small” wings, with a span to tunnel width ratio less than 0.5. Taken into account the half model configuration, the ratio for the model was 0.75. Earlier derivations by Herriot [34] represented the wing as line sources and sinks for wind tunnels whose width to height ratios are between 0.29 and 3.5, and model span to tunnel width ratios are between 0 and 1. This method was used instead of the AGARD method for calculating solid blockage. Herriot’s method adopted scale factors based upon tunnel shapes and aerofoil profiles. However, due to the age of the paper, scale factors were only available for certain low-drag NACA aerofoils. Scale factors for various thicknesses of the NACA 66-0XX series aerofoils were interpolated for the required 17% thickness. This aerofoil was selected as it provided the closest match to the LS(1)-0417MOD’s performance.
When considering both lift interference and blockage corrections it is important to note that any equation for lift interference or blockage corrections that utilises either the wind tunnel width (equal to the physical height of 1.3 m) or the model span (0.975 m) requires this value to be doubled, due to the half model configuration using the wind tunnel floor as a reflection plane. A summary of all equations used are presented in Appendix C.

The following general assumptions were made for the three-dimensional corrections, based on the associated texts for the correction formulae.

- The model is positioned close to the mid-plane of the wind tunnel, and in the case of the half model configuration, the root of the model is on the wind tunnel floor.
- The wind tunnel floor is assumed to act as a perfect reflection plane.
- Model upwash or lift interference is not directly affected by model thickness or the size of the wake.
- The flow field is generally uniform in the vicinity of the model.
- The flow is subsonic and assumed to be incompressible.
- The wing can be represented as a series of line sources and sinks.
- Wake blockages are independent of vortex drag and lift, assuming lift is not large (no further clarification of ‘large’ was provided by AGARD [28]).
- For the purposes of blockage corrections, drag values used at zero lift incidence (-2.5°) were uncorrected.

As per the AGARD method [28], the blockage correction formulae to pressure coefficients are identical to those for a two-dimensional wing. Therefore, for the same rationale as was discussed previously, blockage corrections will not be applied to the three-dimensional pressure data. The corrections were typically less than 1% for all measured values of $C_P$ and made little difference to the final values.

Table 5.3 below summarises the changes to the incidence and aerodynamic coefficients when lift interference and blockage corrections are applied to the three-dimensional LS(1)-0417MOD wing.

<table>
<thead>
<tr>
<th>Incidence, °</th>
<th>$\Delta \alpha$</th>
<th>$\Delta C_L$</th>
<th>$\Delta C_D$</th>
<th>$\Delta C_M$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0°</td>
<td>+0.170°</td>
<td>-0.0009 (0.5%)</td>
<td>+0.0004 (2.3%)</td>
<td>+0.0003 (0.5%)</td>
</tr>
<tr>
<td>+6°</td>
<td>+0.679°</td>
<td>-0.0038 (0.6%)</td>
<td>+0.0068 (17.5%)</td>
<td>+0.0003 (0.5%)</td>
</tr>
<tr>
<td>+12°</td>
<td>+1.093°</td>
<td>-0.0070 (0.7%)</td>
<td>+0.0182 (22.0%)</td>
<td>+0.0003 (0.5%)</td>
</tr>
</tbody>
</table>

Table 5.3: Changes to the aerodynamic coefficients on an undamaged wing from the three-dimensional wind tunnel corrections (data for $Re = 1,000,000$).
5.5 Data Processing

After the wind tunnel data have been corrected for lift interference and blockage effects, the data are converted into coefficient increments, as with the two-dimensional data. The same equations as defined by Irwin [1] are used, however a nomenclature change is introduced to identify the three-dimensional coefficients, adopting the convention of uppercase subscripts. Equations 2.1 to 2.3 are presented in Equations 5.1 to 5.3 below, for three-dimensional lift, drag and pitching moment coefficient increments respectively.

\[ dC_L = C_{L \text{ damaged}} - C_{L \text{ undamaged}} \]  
\[ dC_D = C_{D \text{ damaged}} - C_{D \text{ undamaged}} \]  
\[ dC_M = C_{M \text{ damaged}} - C_{M \text{ undamaged}} \]

5.6 Experimental Repeatability Analysis

As with the two-dimensional method a number of factors affect the experimental repeatability. The three-dimensional model configuration added further factors which required consideration, while some factors from two-dimensional testing were also applicable. Such factors included the boundary layer transition strip, repeatability of individual panel sets and verifying the entry repeatability against a datum case.

5.6.1 Factors Affecting Experimental Repeatability

Three factors were identified as potentially affecting the repeatability of the three-dimensional testing. A number of factors from two-dimensional testing (see section 4.8) also apply to the three-dimensional testing.

Entry Repeatability

Entry repeatability covered factors that only affected the model during installation into the wind tunnel. Due to the availability of the wind tunnel and length of testing slots, multiple entries were required, resulting in the model being re-installed each time. The following additional factors unique to the three-dimensional testing were deemed to affect entry repeatability.
5.6. Experimental Repeatability Analysis

- **Floor Inserts:** Floor inserts around the base of the model provided the necessary gap between the dummy floor and the wind tunnel turntable to allow balance measurements to be made. These inserts formed part of the turntable (see Figure 5.3). Care was taken to ensure the gap was constant, although priority was taken over ensuring the inserts were not touching the dummy floor and model. Issues with floor inserts typically affected the pitching moment results more than lift and drag.

- **Model Installation:** The model was designed such that its location on the strut, and the location of the strut on the balance were highly repeatable, using dowel pins and reamed bolts. All bolts were tightened using a torque wrench to a predefined torque value to minimise movement and possible effects on balance zeros.

- **Validation:** For validation purposes, the undamaged datum consisted of three pairs of panels (upper and lower surface panels) that remained in the same position for all testing, unless replaced by a damaged set of panels. This removed any additional variables of panel geometry from the undamaged spanwise locations.

**Panel Repeatability**

As with the two-dimensional testing, each set of panels were tested prior to damage being applied to ensure they all fitted within an acceptable repeatability band and matched well against the datum case. Any panel outside of the band was either discarded or modified.

**Location Repeatability**

In addition to the panel repeatability tests, each undamaged panel was tested at the three spanwise locations, with the specified undamaged datums occupying the other spanwise locations. It was found that the spanwise variation of a single panel was minimal, and less significant than other repeatability factors.

5.6.2 Quantifying the Repeatability

A number of repeatability runs were undertaken. For these, the model was removed from the tunnel, all panels were removed, and the model then re-installed in the tunnel and the panels reinstalled. The magnitude of the repeatability band on the lift coefficient at $\text{Re} = 1,000,000$ (Figure 5.5) was generally small below the onset of stall, with the size of the $\Delta C_D$ repeatability band (Figure 5.6) identified as being due to the variation in individual panels. The pitching moment coefficient (Figure 5.7) was found to be most susceptible to entry repeatability. This was identified to be due to the gap between the floor inserts on the turntable and the model. Ideally this was to be kept as small as possible but due to the bending loads exerted on the model at high incidences, this occasionally resulted in
the model floor touching the inserts, and thus corrupting the results in pitch. Such data sets were discarded, the model floor re-seated and the test re-run, and as such this does not affect the repeatability bands shown. It should be noted that at a Reynolds number of 1,000,000, testing was restricted to 14° incidence, due to the model approaching the balance load limits.

Figures 5.8 to 5.10 show the repeatability bands for lift, drag and pitching moment coefficients at a Reynolds number of 500,000. These are included to allow the effects of Reynolds number to be analysed. In all cases, the bands are shown to be larger than for the higher Reynolds number runs (see Figures 5.11 to 5.13). This was due to smaller magnitude forces being exerted on the balance, resulting from the lower operating speed. This therefore put any measurements closer to the minimum sensitivity of the balance load cells. The trends however in the repeatability bands are broadly similar between the two cases.

As with the two-dimensional repeatability, when considering coefficient increments, only the entry repeatability bands would be applicable. Figures 5.11 to 5.13 show the effects of the entry repeatability on lift, drag and pitching moment coefficient increments for a sample damage hole (20% chord diameter with no obliquity or skew, at the central location) at a Reynolds number of 1,000,000. The trends shown in the three figures are similar to those from the two-dimensional repeatability study and show that unlike the 5%c diameter hole from the two-dimensional testing, there should be few concerns with increments produced from the 20%c holes falling within repeatability bands.

The maximum repeatability bands for both Reynolds Numbers are given in Table 5.4 below, for undamaged panels. These were taken for pre-stall conditions, due to the bands becoming significantly larger once stall onset had begun. In addition, the repeatability bands for coefficient increments are also shown.

<table>
<thead>
<tr>
<th>Panel</th>
<th>Repeatability band</th>
<th>Re = 500,000</th>
<th>Re = 1,000,000</th>
</tr>
</thead>
<tbody>
<tr>
<td>Incidence (α)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>CL</td>
<td>±0.01°</td>
<td>±0.01°</td>
<td></td>
</tr>
<tr>
<td>CD</td>
<td>±0.018</td>
<td>±0.011</td>
<td></td>
</tr>
<tr>
<td>CM</td>
<td>±0.0021</td>
<td>±0.0015</td>
<td></td>
</tr>
<tr>
<td></td>
<td>±0.0024</td>
<td>±0.0023</td>
<td></td>
</tr>
<tr>
<td>Coefficient Increments</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Incidence (α)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>dCL</td>
<td>±0.02°</td>
<td>±0.02°</td>
<td></td>
</tr>
<tr>
<td>dCD</td>
<td>±0.012</td>
<td>±0.0072</td>
<td></td>
</tr>
<tr>
<td>dCM</td>
<td>±0.025</td>
<td>±0.0018</td>
<td></td>
</tr>
<tr>
<td></td>
<td>±0.013</td>
<td>±0.0012</td>
<td></td>
</tr>
</tbody>
</table>

Table 5.4: Three-dimensional testing repeatability bands.
Figure 5.5: Experimental repeatability bands for lift coefficient at a Reynolds Number of 1,000,000.

Figure 5.6: Experimental repeatability bands for drag coefficient at a Reynolds Number of 1,000,000.

Figure 5.7: Experimental repeatability bands for pitching moment coefficient at a Reynolds Number of 1,000,000.
Figure 5.8: Experimental repeatability bands for lift coefficient at a Reynolds Number of 500,000.

Figure 5.9: Experimental repeatability bands for drag coefficient at a Reynolds Number of 500,000.

Figure 5.10: Experimental repeatability bands for pitching moment coefficient at a Reynolds Number of 500,000.
5.6. Experimental Repeatability Analysis

![Graph](image)

**Figure 5.11:** Experimental repeatability bands for lift coefficient increment for a centrally-located 20%c hole with no obliquity or skew at a Reynolds Number of 1,000,000.

![Graph](image)

**Figure 5.12:** Experimental repeatability bands for drag coefficient increment for a centrally-located 20%c hole with no obliquity or skew at a Reynolds Number of 1,000,000.

![Graph](image)

**Figure 5.13:** Experimental repeatability bands for pitching moment coefficient increment for a centrally-located 20%c hole with no obliquity or skew at a Reynolds Number of 1,000,000.
5.7 Model Validation

The model was used in its undamaged configuration to carry out balance and surface pressure measurements for validation purposes. The three “datum” undamaged panels were used during this validation. For surface pressure measurements a fourth undamaged panel pair was used, which were tested at each of the three spanwise locations. Surface flow visualisation was also undertaken to assess the performance of the transition strip. The transition strip was found to be performing adequately.

5.7.1 Undamaged-Wing Balance Data Analysis

Initial undamaged runs were conducted at Reynolds numbers of 500,000 and 1,000,000 to compare the performance of the model to the two-dimensional data (Re = 500,000). Figures 5.14 to 5.16 show the lift, drag and pitching moment coefficients at both Reynolds numbers, compared to the two-dimensional data.

The lift coefficients (Figure 5.14) showed trends consistent with a change in aspect ratio. The zero-lift angle had shown little change, while there was a reduction in lift curve slope and maximum lift coefficient for the three-dimensional model. Both Reynolds numbers from the three-dimensional testing produced similar results. This trend was consistent with the published NASA data [25] (albeit for a two-dimensional aerofoil and for higher Reynolds numbers), which showed at incidences below approximately 13° there was little change in $C_L$ with Reynolds number.

The drag curves (Figure 5.15) showed trends consistent with changing Reynolds number for the three-dimensional model, with a reduced $C_D$ for increased Reynolds number across all values of $C_L$. The increased drag coefficient for the three-dimensional model, compared to the two-dimensional model was expected, given the presence of wing tips and the associated flow effects. However, at lower incidences the drag for the three-dimensional model at Re = 500,000 was lower than for the two-dimensional model. Although the two models had nominally identical aerofoil sections the physical models and panel geometries were slightly different. Additionally the two-dimensional model, due to its smaller chord, was more susceptible to manufacturing tolerances on the panels, as both sets were manufactured to the same tolerances.

The pitching moment coefficients (Figure 5.16) showed the greatest differences to the two-dimensional data. This was due to the two wind tunnel configurations using very different methods for mounting the model and obtaining pitch data. The two-dimensional model was mounted on pin-jointed struts connected to the balance, resulting in some friction between the pin and the strut. This would have skewed the data compared to the finite
aspect ratio testing, which did not use any pin joins and instead affixed the model direct to the balance.

Figure 5.17 shows $C_D$ against $C_L^2$ for both models. This shows that the three-dimensional wing demonstrates the expected behaviour below stall, with the $C_D/C_L^2$ curve adopting a straight line. From this figure, the drag coefficient at zero lift, $C_{D0}$ can be calculated for both models. Theoretically, at zero lift (and therefore no induced drag on either model) both models should produce the same $C_{D0}$ [2]. As can be seen an offset is present between the data sets. The two-dimensional $C_{d0}$ is 0.0216, while the three-dimensional $C_{D0}$ is 0.0190 at $Re = 500,000$ and 0.0156 at $Re = 1,000,000$. The offset between two-dimensional data and three-dimensional data at $Re = 500,000$ has likely arisen from small model geometry differences, primarily in the interaction between the panel and the solid leading and trailing edges.

The lift curve slope of the two-dimensional model was 5.307. At $Re = 500,000$, the three-dimensional lift curve slope was 4.008, and at $Re = 1,000,000$ it was 4.060. Equation 5.4 from Anderson [2] calculates the equivalent lift curve slope on a three-dimensional wing, assuming an elliptic pressure distribution.

$$a_{3D} = \frac{a_0}{1 + \frac{a_0}{\pi AR}}$$

Where:

- $a_{3D}$ = Equivalent three-dimensional lift curve slope
- $a_0$ = Two-dimensional lift curve slope
- $AR$ = Aspect ratio

Using the two-dimensional lift curve slope and a theoretical aspect ratio of 6 in Equation 5.4, the equivalent three-dimensional lift curve slope is 4.141. However, the three-dimensional wing will not have an elliptic pressure distribution due to its rectangular profile. The gradient of the straight line region of the $C_D/C_L^2$ curve (Figure 5.17) can be expressed as $\frac{K}{\pi AR}$, where $K$ is a factor dependent on the induced drag. Anderson states [2] that this factor is typically between 1.05 and 1.25. Estimations from ESDU [35, 36] state $K$ to be equal to 1.103 for the specified geometry. Therefore, rearranging Equation 5.5, taken from ESDU 66031 [35], the effective aspect ratio of the three-dimensional wing can be found.
Figure 5.14: Lift coefficients at different Reynolds numbers from the three-dimensional model, compared against the two-dimensional model data.

Figure 5.15: Lift/drag curves at different Reynolds numbers from the three-dimensional model, compared against the two-dimensional model data.

Figure 5.16: Pitching moment coefficients at different Reynolds numbers from the three-dimensional model, compared against the two-dimensional model data.
Figure 5.17: $C_D$ against $C_L^2$ at different Reynolds numbers for the three-dimensional model, compared to the two-dimensional data.

\[
\frac{dC_D}{dC_L^2} = \frac{K}{\pi AR} \tag{5.5}
\]

Where:

- $\frac{dC_D}{dC_L^2}$ = Gradient of the straight line region of the $C_D/C_L^2$ plot. Note these are not coefficient increments.

- $K$ = Induced drag factor

Using the gradient of the straight line region of the Re = 1,000,000 curve from Figure 5.17 of 0.0815 and a $K$ value of 1.103 from ESDU [35, 36], the effective aspect ratio is 4.30. Due to the nature of half model testing it would not be expected to achieve the theoretical aspect ratio, due to the presence of the boundary layer from the wind tunnel floor. These results are encouraging and indicate that the three-dimensional model is of suitable accuracy.

5.7.2 Analysis of Surface Pressure Data for an Undamaged Wing

Surface pressure data are presented in Figures 5.18 to 5.20 for the five spanwise taps at each of the three spanwise panel locations (root, centre and tip respectively) at a Reynolds number of 1,000,000 at 8° incidence. The data are compared to two-dimensional NASA data from [25], obtained at a Reynolds number of 2,000,000. All data are uncorrected for blockage effects. All figures show a general offset for both upper and lower surfaces from the NASA data, with a reduction in the pressure difference between upper and lower
surfaces. This is as a result of the finite aspect ratio effects. The trends along the chord matched well with the NASA data, indicating that the panels used on the wing were of reasonable accuracy and match to the published aerofoil profile.

A small degree of variation across the span on the upper surface was witnessed at the root location (Figure 5.18). This was surprising given that it would be anticipated that the region near the root would experience virtually no spanwise variation in pressure. The figure shows that towards the leading edge the pressure coefficients at the spanwise taps nearest the floor (SP1 and SP2) on the upper surface were reduced, until approximately 40%c, compared to the more outboard taps. Previous testing in the same wind tunnel [22] indicated that the boundary layer on the wind tunnel floor disrupted flow up to approximately 150 mm from the wind tunnel floor (equivalent to 15.4% span), which would be consistent with the pressure data shown. Towards the trailing edge, aft of approximately 55% chord, the outboard data set (SP5) for the upper surface appeared anomalous, going against the trends set by the rest of the data. Given such a discrepancy does not appear at the other panel locations, the cause is likely to be from a small installation error, causing a small wake to form off either the outboard leading edge of the root panel or the inboard leading edge of the centre panel.

Figure 5.19 shows the spanwise variation in pressure coefficients for the central panel location. The data for both upper and lower surfaces showed very little spanwise variation in the pressure coefficients, indicating that the presence of the wing tip was not having a significant effect at this location. However, at the tip panel location, between 65% and 85% span, the data (Figure 5.20) began to show the effects of the reducing pressure coefficients as a result of the wing tip. It can be seen that the upper surface tappings at SP5 (85% span) were significantly reduced compared to other spanwise locations. Additionally, the pressure difference between upper and lower surfaces was seen generally to decrease as the tappings move towards the tip. Changes were less significant on the lower surface due to the smaller magnitude coefficients, but the SP1 data set (inboard) generally has the least negative values, while the SP5 data set generally has more negative values.
5.7. Model Validation

Figure 5.18: Surface pressure coefficients, at $Re = 1,000,000$, taken at five spanwise locations from the root panel location, at $+8^\circ$ incidence. Two-dimensional NASA data from [25].
Figure 5.19: Surface pressure coefficients, at $Re = 1,000,000$, taken at five spanwise locations from the central panel location, at $+8^\circ$ incidence. Two-dimensional NASA data from [25].
Figure 5.20: Surface pressure coefficients, at Re = 1,000,000, taken at five spanwise locations from the tip panel location, at +8° incidence. Two-dimensional NASA data from [25].
5.8 Reynolds Number Effects

Given that there was a desire to undertake three-dimensional testing at a Reynolds number of 1,000,000 and that as previously stated, two-dimensional testing was carried out at Re = 500,000, it is necessary to determine if any changes in data arose from this difference. Since both sets of data will be used to develop predictive techniques, it is necessary to assess any potential effects of increasing the Reynolds number from 500,000 to 1,000,000. To accomplish this, tests were undertaken on the three-dimensional model at both Reynolds numbers, using a representative 20% chord diameter hole with no obliquity or skew located at the central spanwise location. The corresponding lift, drag and pitching moment coefficient increments are shown in Figures 5.21 to 5.23. Reference should be made to the repeatability bands stated in section 5.6, which indicated slightly poorer experimental repeatability for testing at Re = 500,000.

The three figures show that the variation between the coefficient increments at different Reynolds numbers is generally small and falls within the repeatability bands. The lift coefficient increments (Figure 5.21) show the greatest spread at low incidences (below 5°), although still within the experimental repeatability bands (section 5.6). A possible source for this error was from the much smaller balance forces resulting from running at lower speeds for the Re = 500,000 run, coupled with the low incidence. However, it would be anticipated this would apply to all three coefficient increments, but it appears that this offset only affected the lift coefficient increments. Drag (Figure 5.22) and pitching moment coefficient increments (Figure 5.23) showed negligible change across the incidence range. It should be noted that the maximum incidence discrepancy between the two Reynolds numbers is due to balance load limits experienced at Re = 1,000,000. Given these figures show the cases with the typical scatter and variation it can be said that with the use of the transition strip, any potential Reynolds number effects have been successfully removed. Therefore, data from lower Reynolds number runs (including the two-dimensional data) can be compared with the more reliable data from higher speed runs on the three-dimensional model.

It should be noted that the recovery and sign change of $dC_L$, $dC_D$ and $dC_M$ above approximately 13° was a consistent trend across most damage cases. This was likely due to a phenomenon similar to that experienced with blown flaps and will be discussed in due course. A full analysis of the battle damaged effects and trends in these graphs are undertaken in the next two chapters.
5.8. Reynolds Number Effects

Figure 5.21: Variation of lift coefficient increment with Reynolds number for a 20% chord diameter hole with no obliquity or skew located at the centre.

Figure 5.22: Variation of drag coefficient increment with Reynolds number for a 20% chord diameter hole with no obliquity or skew located at the centre.

Figure 5.23: Variation of pitching moment coefficient increment with Reynolds number for a 20% chord diameter hole with no obliquity or skew located at the centre.
Initially damage configurations with no obliquity or skew (referred to as “straight through” holes) were tested in order to allow baseline trends to be identified, and to compare the effects of varying hole size on the two-dimensional LS-series aerofoil with previously published work by Irwin [1]. The effects of obliquity (chordwise displacement) and skew (spanwise displacement), and a combination of both were investigated. It should be noted that from here on “two-dimensional” generally refers to the testing configuration.

6.1 Summary of Surface Flow Visualisation Features

A summary of the key surface flow visualisation features obtained from experimental results is presented, using a 5% and 20% chord diameter straight through holes. An example annotated flow visualisation photograph of a weak jet is shown in Figure 6.1.

- Point “A” shows the horseshoe vortex. This bounds the wake of the jet and is generally a similar width to the hole. In some cases, as shown here, the horseshoe vortex does not surround the entire hole. This indicates that the jet is not running full (filling the hole), and thus the forward portion of the hole is acting as a cavity.

- Point “B” shows a pair of small contra-rotating vortices. With a weak jet, these are typically small and close to the spanwise centreline of the hole, on the trailing edge of the hole. Flow visualisation mixture has collected at this point, resulting in the two vortices becoming indistinguishable from each other.

- Point “C” shows a small jet downstream of the hole, attached to the aerofoil surface, and the resulting disturbance to the surface flow.

- Point “D” shows a narrow wake from the fasteners used to attach the removable panels to the wing.

- Point “E” shows effects from the panel edges.
Figure 6.1: Example annotated weak jet surface flow visualisation photograph, showing a 5% hole with no obliquity or skew at 0° incidence. Flow direction is from bottom to top.
6.1. Summary of Surface Flow Visualisation Features

Figure 6.2 shows an annotated surface flow visualisation photograph for a strong jet. The most obvious difference between the weak jet (Figure 6.1) and the strong jet is the presence of a much wider wake with a region of reverse flow within it.

- Point “A” shows the beginning of the forward separation line which bounds the horseshoe vortex.
- Point “B” shows the two contra-rotating vortices, which have moved further round the hole towards the leading edge of the wing.
- Point “C” shows a more clearly defined horseshoe vortex bounding the wake. As the wake has now increased in size the horseshoe vortex has become wider.
- Point “D” shows a region of reverse flow, where flow was entrained from the lower surface and attached to the upper surface of the aerofoil, flowing against the freestream. It then separated and became entrained into the exiting jet at Point ’E’, having been turned towards the freestream again.
- Point “E” shows where the jet has exited the hole and begun to ‘spread out’ in a spanwise direction and grow larger in size as it travels downstream. This was not present in the weak jet, which generally remained a constant size.
- Point “F” shows a small wake from the fasteners used to attach the panels.
- Point “G” shows effects from the panel edges.
Figure 6.2: Example annotated strong jet surface flow visualisation photograph, showing a 20% hole with no obliquity or skew at 8° incidence. Flow direction is from bottom to top.
6.2 Damage Holes with 0° Obliquity and 0° Skew

In order to highlight the differences in flow structure between the straight through holes, surface flow visualisation photographs are presented for 0° and 8° incidence. Figures 6.3 to 6.5 show the 5%c, 20%c and 40%c cases respectively. Focus will be on these three cases since the 10%c hole exhibited very similar characteristics to the 5%c hole, and the 30%c hole followed the trends of the 20%c hole. In all surface flow visualisation photographs, the direction of airflow is from the bottom of the photograph to the top.

It can be seen that at 0° incidence, the 5%c hole (Figure 6.3a, also annotated in Figure 6.1) had little effect on the airflow. A small wake no wider than the hole was witnessed downstream of the hole, remaining attached to the aerofoil surface. Small contra-rotating vortices were visible on the trailing edge of the hole, although due to their proximity, the flow visualisation mixture had collected, giving the appearance that the vortices merged. The location of the horseshoe vortex indicated that the hole was not running full (i.e. the jet is not filling the hole, and is exiting through the aft portion of the hole only), which would suggest the front of hole was instead behaving like a cavity. The wake shown at 0° incidence can be termed a “weak jet”, as identified in Irwin’s work [1] (see also Figure 2.1).

As the incidence increased to 8° (Figure 6.3b) it can be seen that the wake had increased in size, but had not fully strengthened to the “strong jet” shown in Irwin’s work (see Figure 2.2 and Figure 6.2). This case showed a more developed weak jet, with the wake widening slightly, allowing the horseshoe vortex bounding the hole to be identified. As this was now located forward of the hole, it indicated the hole was now running full. In addition, the two contra-rotating vortices were more identifiable, having moved towards the leading edge of the hole, suggesting the jet was strengthening with more airflow through the hole. The absence of a significant separated wake or region of reverse flow downstream of the hole supports the claim that the flow remained a weak jet and had not transitioned to a strong jet.

Figure 6.4 shows the surface flow visualisation for the 20%c case. At 0° (Figure 6.4a) a similar flow pattern to the 5%c hole (Figure 6.3a) can be seen, with little disturbance to the airflow. A small separation region was noted where the wake meets the wing trailing edge, shown by the collection of flow visualisation mixture. Again, there was little flow through the damage hole, shown by the contra-rotating vortices being located at the most aft point of the hole and the wake not spreading beyond the width of the hole. However, once the incidence was increased to 8° (Figure 6.4b) the surface flow structure changed significantly. The wake was considerably wider than the damage hole, with the forward separation line showing the jets had diverted the freestream flow in a spanwise direction, around the damage hole. At 8°, the hole was running more full than at lower incidences,
with the small contra rotating vortices having moved round towards the leading edge of the hole, and the horseshoe vortex having moved forwards. This was consistent with Irwin’s findings, which indicated a strengthening jet as incidence increased. The additional feature of the strong jet is a region of reverse flow, between the trailing edge of the wing and the large vortices sitting towards the rear of the panel. As can be seen, the strong jet caused significant disruption to the airflow over the surface of the aerofoil, with the wake occupying a large area. This would have a significant effect on the aerodynamic coefficients.

Figure 6.5 shows the surface flow visualisation for the 40%c case. The two flow visualisation photographs are similar to the 20%c cases shown in Figure 6.4. The greatest difference between the two holes was witnessed at 0\degree incidence (Figure 6.5a). A narrow region of reverse flow, at the centre of the hole was witnessed, with a more clearly defined wake either side, occupying a similar spanwise extent to the hole. In addition, the two contra-rotating vortices at the aft edge of the hole were more clearly defined. These features suggest that the jet had strengthened from a weak jet, but with the absence of a large wake and extensive region of reverse flow, had not fully developed into a strong jet. Instead the jet has likely become a “transitional” jet. As the incidence increased (Figure 6.5b), the jet became more similar to the 20%c case, with the only noticeable difference being the wider region of reverse flow, due to the larger diameter hole.

Lift, drag and pitching moment coefficient increments for the five straight through damage cases are shown in Figures 6.6 to 6.8. Reference to absolute $C_l$, $C_d$ and $C_m$ values can be made to the repeatability figures, shown in Figures 4.8 to 4.10. Two distinct trends were noticed in the coefficient increment data: the 5%c and 10%c diameter holes produced very small $dC_l$ values which generally decreased with incidence; and the larger diameter holes produced curves with steeper gradients, which varied with incidence. When compared to the surface flow visualisation photographs in Figures 6.3 to 6.5 it can be seen that the smaller $dC_l$ values generally corresponded to weak jets, with the more negative values corresponding to the stronger jets. The same two distinct trends were also identified in the drag coefficient increments, with the 5%c and 10%c values producing very small $dC_d$ values, and the the larger diameter holes presenting a greater variation in $dC_d$ with incidence. Similarly, the largest wakes corresponded to the greatest $dC_d$ values (Figure 6.7), due to the larger area of disruption on the aerofoil surface. Irwin’s work [1] identified that weak jets had a minimal influence on the surface pressure distribution of the aerofoil, with any effects remaining close to the damage hole. Conversely the strong jet caused significant disruption to the pressure distribution over a wider spanwise region of the aerofoil. Given that the trends in coefficient increments correlate with Irwin’s work it can be said that the larger changes in $dC_l$ from the strong jet were due to the greater disruption to the pressure distribution across the upper surface.
6.2. Damage Holes with $0^\circ$ Obliquity and $0^\circ$ Skew

Figure 6.3: Surface flow visualisation for a 5% chord diameter hole with no obliquity or skew.

Figure 6.4: Surface flow visualisation for a 20% chord diameter hole with no obliquity or skew.

Figure 6.5: Surface flow visualisation for a 40% chord diameter hole with no obliquity or skew.
The general trend was for $dC_l$ and $dC_d$ to increase in magnitude with hole size. However, the pitching moment coefficient increments (Figure 6.8), with the exception of the 5%c hole, were reasonably invariant with hole size across the central incidence range (approximately $-5^\circ$ to $+10^\circ$). This was consistent with Irwin’s findings and was due to the centres of the holes not moving with respect to the moment reference centre. Irwin found that $dC_m$ was strongly dependent on the distance of the hole from the moment reference centre, essentially linked to the moment arm of the damage hole. The increase in scatter at higher incidences in all figures was a result of stall onset (at approximately $12^\circ$ incidence), while the lack of variation in the 5%c hole was likely due to the small hole diameter having a small effect on the flow characteristics, as illustrated by the surface flow visualisation. Above approximately $12^\circ$ incidence, after the onset of stall, some cases saw the damage effects reverse, resulting in either a decrease in drag or an increase in lift from the undamaged case. While the large experimental repeatability bands reduces the confidence in the results at this point, surface flow visualisation indicated that the jet was partly helping to reattach the flow to the upper surface, creating an effect similar to that of a blown flap. This helped to delay stall in the immediate vicinity of the jet, reducing the lift loss and drag increase.

The lack of variation in $dC_l$, $dC_d$ and $dC_m$ with incidence for the 5%c hole, and to a lesser extent the 10%c hole, suggest that the hole was too small to have any noticeable effect on the surface flow and coefficient increments. It should be noted that across much of the incidence range for the three coefficient increments, the values for the 5%c hole were within experimental repeatability bands. The surface flow visualisation indicated little throughflow at $0^\circ$, and a very small jet (compared to all other cases) through the hole at $8^\circ$ which suggested that the hole was instead acting as a cavity at the lower incidences. Without additional data from negative lift incidences it is difficult to confirm the trends of the small holes below $-2.5^\circ$ incidence (approximate zero lift angle). It is felt that the 5%c and 10%c holes are too small to allow definite conclusions to be drawn.

It can be seen from Figure 6.6 that the 20%c and 30%c curves had a shallow gradient below approximately $+2^\circ$ incidence, and after this point the gradient increased, to produce a similar gradient to that of the 40%c hole. Surface flow visualisation images indicated that these changes in gradient corresponded to a transition from a weak jet to a strong jet. Figure 6.9 illustrates this using the 20%c hole as an example, by presenting flow visualisation photographs for a series of incidences: $0^\circ$, $2^\circ$, $4^\circ$, $8^\circ$ and $12^\circ$.

As the incidence was increased from $0^\circ$ (Figure 6.9a) to $2^\circ$ (Figure 6.9b) the wake size increased, indicating a strengthening jet. At $2^\circ$, elements of the strong jet were more visible with a narrow region of reverse flow beginning to form aft of the hole. By $4^\circ$ (Figure 6.9c) the flow was exhibiting all the typical features of a strong jet, with a well
6.2. Damage Holes with 0° Obliquity and 0° Skew

Figure 6.6: Lift coefficient increments for holes of varying diameter, with no obliquity or skew.

Figure 6.7: Drag coefficient increments for holes of varying diameter, with no obliquity or skew.

Figure 6.8: Pitching moment coefficient increments for holes of varying diameter, with no obliquity or skew.
defined region of reverse flow, wide wake and large area of recirculation downstream of the hole at the point the reverse flow separated from the aerofoil surface.

Increasing the incidence to $8^\circ$ (Figure 6.9d) resulted in the jet strengthening and the wake increasing in width to approximately three times the hole diameter. This corresponded to the greater loss of lift and increase in drag in Figures 6.6 and 6.7. At $12^\circ$ (Figure 6.9e), a larger wake was witnessed. This incidence marked the onset of the stall. However, when compared to surface flow visualisation from an undamaged wing and for the solid wing it was apparent that the presence of the panel was promoting early separation of the flow. On the damage case, this manifested in the flow structure that appeared similar in shape to the horseshoe vortex but further forward of the hole. Behind the separated region, the strong jet feature was still present, although smaller than at $8^\circ$ incidence. The horseshoe vortex could also be seen inside this separated region, close to the hole. The surface flow visualisation confirmed the trends in Figures 6.6 and 6.7 of the lift and drag coefficient increments reducing in magnitude as the stall approached.

Figure 6.10 shows the surface pressure coefficients across the panels at five spanwise locations (expressed as multiples of the hole radius, $R$) for the 20%c hole, compared to the undamaged data at the central location of $R = 0$, at $8^\circ$ incidence. Figure 6.10 shows that for the straight through damage hole there was a reduction in the magnitude of the $C_p$ values for the upper surface across the data set, with one exception of a small region aft of approximately 65% chord on the upper surface. The lower surface $C_p$ values became more negative, resulting in the pressure differential between the upper and lower surfaces reducing compared to the undamaged case. Forward of the damage hole, there was a significant decrease in the magnitude of the $C_p$ values, compared to the undamaged data. This suggests there was a large reduction in the pressure peak at the leading edge of the model. Similar results were noticed by Irwin [1] for an extensively pressure tapped model. This reduction was responsible for the negative pitching moment coefficient increments (reducing $C_m$), seen in Figure 6.8. This effect of the reduced pressure peak was, however, partially offset by the changes in $C_p$ behind the pitching moment reference centre, at quarter chord. This in turn was offset by the increase in magnitude of $C_p$ values aft of the damage hole which, based on Irwin’s findings, would have extended beyond the sampled region towards the trailing edge of the model.

The reduction in magnitude of the $C_p$ values for the damaged upper panel forward of the hole, indicated that the flow slowed down as it approached the damage jet. This was evident as far out as $R = 5$, although the surface flow visualisation (Figure 6.9d) indicated this station was outside of any damage flow features. Although the change from the undamaged $C_p$ was small compared to other spanwise locations, it was evident that the effects from the damage were influencing the flow over a much wider region than the flow
6.2. Damage Holes with $0^\circ$ Obliquity and $0^\circ$ Skew

Figure 6.9: Surface flow visualisation sequence for a 20% chord diameter hole with no obliquity or skew, showing the upper surface.
visualisation would suggest, as was found by Irwin [1]. Essentially this meant the surface flow visualisation on its own cannot be relied on to give a complete indication of the extent of the damage effects. Close to the sides of the hole (at $R = 1.5$ and $R = 2.5$, between 0.4c and 0.6c), there was little variation with chord in the upper surface $C_p$ distributions, which indicated there was little acceleration around and immediately downstream of the hole. Surface flow visualisation indicated this region to be where the jet was spreading out in a spanwise direction, beyond the hole. It should be noted that, with the exception of the most outboard set of tappings ($R = 5$), the upper surface pressure coefficients failed to return to the corresponding undamaged values. Irwin [1] identified this as a feature of the strong jet, which is caused by accelerating flow from the jet and the resulting wake.

The presence of the damage caused the $C_p$ values on the lower surface to become more negative for all tapping locations, although the general shape of the curves was similar to the undamaged case. The exceptions to this were the central tappings, $R = 0$ and $R = 0.5$, where small deviations in $C_p$ occurred from the other data sets, both upstream and downstream of the hole. Forward of the hole $C_p$ became more negative, indicating the flow was accelerating into the damage hole. Downstream of the hole, for $R = 0$ and $R = 0.5$, there was little change compared to the undamaged pressure coefficients. This indicated

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**Figure 6.10:** Spanwise variation of surface pressure coefficients for a 20%c hole with no obliquity or skew.
that in the immediate downstream vicinity of the hole the velocity remained unchanged, relative to the undamaged case. Surface flow visualisation on the lower surface showed the rear of the hole was acting like a cavity, with a small attached wake, with a width approximately equal to the hole diameter (see Figure 6.4b). The pressure coefficients were consistent with this by showing little change compared to the undamaged coefficients.

### 6.3 Effects of Obliquity

While damage cases with obliquity were tested on 5%c and 20%c holes, only data from the 20%c cases will be discussed, due to the very small increments generated by the 5%c holes. Changing obliquity had a noticeable effect on the flow characteristics, as is shown in Figure 6.11, which shows obliquity angles from +60° to -60° in 30° steps. All figures show upper surface flow visualisation at 8° incidence. The photographs demonstrate that as the upper hole moves forward, towards the leading edge the wake increases in size, consistent with the jet strengthening. With the exception of +60° obliquity (Figure 6.11a), the other obliquity cases showed a wake structure very similar to the previously discussed straight through case (Figure 6.11c). The key differences were that as the obliquity became more negative, the small contra-rotating vortices that sat on the trailing edge of the hole were seen to move further round towards the leading edge, indicating that the jet was strengthening. This suggested the jet at -60° obliquity (Figure 6.11e) was the strongest of all the cases. It should be noted that the wake for the +60° obliquity case was significantly smaller than for the other cases, which indicated a considerably weaker jet. Therefore, based on the findings from the “straight through” holes, smaller magnitude $dC_l$ and $dC_d$ values would be expected for the +60° obliquity case, compared to the other obliquity values.

As the upper surface hole moves closer to the leading edge the pressure differential across the hole increases in magnitude. The pressure distribution from the undamaged aerofoil (see Figure 4.18 for 8° incidence) can be used to obtain a pressure coefficient difference across the regions removed by the damage holes. Taking the difference between upper and lower surface $C_p$ values at a single chordwise point at the centres of the -60° obliquity hole (approximately 35%c and 65%c for the upper and lower surfaces respectively) yields the pressure differential across the holes. The pressure coefficient difference changes from -1.08 at 8° for the -60° obliquity hole to -0.57 for the +60° obliquity hole. This increase in magnitude of the pressure coefficient difference correlates with the jets strengthening as the obliquity becomes more negative.
Figure 6.11: Surface flow visualisation on the upper surface for 20% diameter holes with varying obliquity, at 8° incidence.
Figures 6.12 to 6.14 show the coefficient increments for each of the oblique cases. The lift and drag coefficient increments supported the trends shown by the surface flow visualisation of positive obliquity weakening the jet and negative obliquity strengthening the jet. By moving the upper hole closer to the leading edge this increased the length of the wake, thus causing greater disruption to the surface pressures over a larger portion of the chord. The longer wakes therefore resulted in larger magnitude \( dC_d \) values. The effects of a straight through hole positioned in line with the upper hole for a -60° obliquity case are discussed later in this section.

Three key regions were identified in both coefficient increment plots: weak jet, strong jet and post stall. Weak jets occurred at the lower incidences and were characterised by small coefficient increments and a relatively shallow gradient. The incidence at which the jet transitioned from a weak jet to a strong jet varied depending on the obliquity angle, with transition occurring at 0° for -60° obliquity, and between 2° – 4° incidence for +60° obliquity. In all cases, the strong jet region was characterised by lift and drag coefficient increments that rapidly increased in magnitude up to the point of stall (the exception being +60° obliquity, which will be discussed in detail later). This was coupled with the surface flow visualisation showing larger wakes with distinct regions of reverse flow. Beyond approximately 12° the stall onset began, where there was a significant reduction in the magnitude of \( dC_l \) and \( dC_d \). This correlated with a large separation region migrating towards the leading edge in the surface flow visualisation photographs. These trends were consistent with all obliquity cases, with more negative obliquity angles experiencing stronger jets.

As with \( dC_l \) and \( dC_d \) negative obliquity saw the largest magnitude pitching coefficient increments (Figure 6.14). This was a result of the holes changing in chordwise position and the resulting impact on the pressure distribution in the vicinity and downstream of the hole being further from the moment reference centre. These findings were consistent with results from Irwin [1], which investigated the effects of varying the chordwise locations of straight through holes. As previously stated the forward holes cause a greater disruption across the aerofoil chord, resulting in a more significant drop in pressure from the forward pressure peak. As a result, this causes a more negative \( dC_m \). As with \( dC_l \) and \( dC_d \), beyond stall trends in \( dC_m \) broke down, with few identifiable trends between different obliquity angles. This was due to the resulting separated flow over the wing. Above stall (approximately 12°), \( dC_l \) and \( dC_d \) showed, for some obliquity cases, a reversal of sign. While this was likely to fall within the wide experimental repeatability bands (which may also explain why the effect is not consistent across other obliquity cases), it does highlight an interesting effect of the jet beyond stall: the jet may be helping to re-attach the flow to the aerofoil surface, in a similar method to a blown flap. This would therefore delay
stall in the vicinity of the jet, thus increasing lift and decreasing drag, compared to the undamaged case.

It is interesting to note that the effects of obliquity appeared to reverse for negative lift (below $-2.5^\circ$), with positive obliquity producing stronger jets and larger magnitude coefficient increments. This was particularly apparent when analysing the drag coefficient increments. Below the zero-lift incidence, the pressure peak was essentially reversed, occurring on the lower surface. This therefore resulted in the lower hole for $+60^\circ$ obliquity being the closest to the pressure peak. The magnitude of the coefficient increments are not mirrored about the zero lift line since the aerofoil is not symmetric.

Of note is the behaviour of the $+60^\circ$ obliquity case, which at approximately $6^\circ$ incidence diverged from the trends set by the other configurations, with the magnitudes of both $dC_l$ and $dC_d$ decreasing, which would indicate a weakening jet. Figure 6.15 shows flow visualisation photographs for the $+60^\circ$ obliquity case at $0^\circ$, $4^\circ$, $6^\circ$ and $8^\circ$ incidence. Between $0^\circ$ and $4^\circ$ (Figures 6.15a and 6.15b) the jet is seen to develop and strengthen in a similar fashion to the other cases. However at $6^\circ$ (Figure 6.15c) the jet was seen to weaken, with the wake narrowing. It remained in this state, with the wake maintaining a similar size at $8^\circ$ (Figure 6.15d). The internal geometry was identified as a likely cause of this effect, due to the upper surface hole being very close to the aft boundary of the cavity. This will be discussed in detail in section 6.6.

Surface pressure measurements were recorded for the $\pm30^\circ$ obliquity cases and compared to the straight through and undamaged cases at $8^\circ$ incidence in Figure 6.16. The leading edges of the damage holes on the upper surface are located at $34%c$ for $-30^\circ$ obliquity, $40%c$ for the straight through case and $45%c$ for the $+30^\circ$ obliquity case. The five figures show that the trends for the oblique cases were broadly similar to the trends previously identified for the straight through damage (Figure 6.10). On and near the centreline ($R = 0$ and $R = 0.5$; Figures 6.16a and 6.16b) both oblique cases show a deceleration of the flow on the upper surface as it approaches the hole (shown by the smaller magnitude $C_p$ values). This deceleration was more rapid for the negative obliquity case due to the hole being located further forward on the aerofoil. It is interesting to note that despite the differing rates of deceleration and different changes in pressure compared to the undamaged case, the two oblique cases produce a very similar value of $C_p$ immediately forward of the hole to the straight through case. This was also apparent on the lower surface. This indicates that the jets were occupying a similar area of the damage hole, and that the horseshoe vortex was positioned at a similar location forward of the hole. Little variation of the upper surface $C_p$ along the chord between the three cases downstream of the hole indicated that all were running as strong jets.
6.3. Effects of Obliquity

Figure 6.12: Lift coefficient increments for 20\%c diameter holes with varying obliquity.

Figure 6.13: Drag coefficient increments for 20\%c diameter holes with varying obliquity.

Figure 6.14: Pitching moment coefficient increments for 20\%c diameter holes with varying obliquity.
Figure 6.15: Surface flow visualisation on the upper surface for a 20% hole with +60° obliquity.
The variation in obliquity was also apparent away from the hole centreline when considering $R = 1.5$ and $R = 2.5$ (Figures 6.16c and 6.16d). The cause of the greater loss of lift and increase in pressure drag (skin friction is likely to vary due to the wake structure) on the $-30^\circ$ obliquity is evident by the greater reduction in upper surface $C_p$ forward of the hole compared to the straight through and $+30^\circ$ obliquity cases. The shapes of all curves were generally similar, with less negative $C_p$ values on the upper surface closest to the pressure peak for the negative obliquity case. This was a result of the greater deceleration of the flow, compared to the straight through and positive obliquity cases, which due to the reduced pressure coefficient caused a greater loss of lift. Given that this occurred over a greater spanwise extent than for the other cases, this supports the $dC_l$ data, which showed the greatest loss of lift for negative obliquity. Downstream of the damage, at both spanwise locations, the three damage configurations produced reasonably constant $C_p$ values on the upper surface, increasing slightly towards the trailing edge. This was due to relatively similar velocities inside the wake. The difference at $R = 2.5$ (Figure 6.16d) between $+30^\circ$ obliquity and the other cases was due to the narrower wake, with these tappings being outside of the wake identified on the flow visualisation photographs. At $R = 5$ (Figure 6.16e) any distinction between the different obliquity cases had been removed, with all showing a similar small loss of $C_p$ on both upper and lower surfaces, as was seen with the straight through case. As with the straight through case, it is apparent from this that the damage effects extend beyond the extent shown by the flow visualisation photographs. This trend was also identified in Irwin’s work [1].

The presence of obliquity produced little effect on the lower surface with all holes producing similar $C_p$ values across the spanwise range. Away from the hole the $-30^\circ$ obliquity case appeared to have an offset present compared to the other damage cases. It is believed this was due to the leading edge of the lower panel being slightly raised, which was not detected by balance testing.

To further understand the effects of obliquity, and to support the claim that the chordwise position of the upper surface hole determines the jet characteristics, a $20\%c$ hole centred at $35\%$ chord was tested. This was the chordwise centre of the upper surface hole for the $-60^\circ$ obliquity case. Irwin’s work [1] had shown the chordwise centre of “straight through” holes was a key factor in the jet properties, with holes closer to the pressure peak producing stronger jets than holes further away. By comparing this damage case to the $-60^\circ$ obliquity hole, this will determine if the upper hole location is the driving factor in determining jet characteristics.
Figure 6.16: Variation of surface pressure coefficients with obliquity, across the span at 8° incidence.
Figures 6.17 to 6.19 suggest that for positive lift, the chordwise position of the upper surface hole is important in driving the jet properties. This was shown by the good match between the -60° obliquity data and the straight through data at 35% chord. The pressure difference between the upper and lower holes on the -60° obliquity case, taken at the chordwise centres of the holes was –1.08, and for the straight through hole at 35% chord –1.14. While similar, this helps to explain why a greater loss of lift was witnessed for the straight through hole.

The drag coefficient increments (Figure 6.18), showed a close match across the positive lift region (above -2.5°), which suggested potential similarities in the wake or jet structure. Pitching moment coefficient increments (Figure 6.19) did not show as strong a correlation between the -60° obliquity case and the equivalent straight through hole. This was likely due to the differing positions of the lower surface holes affecting the pressure distribution. On all three figures the correlation between the straight through hole centred at 35% chord and the oblique hole broke down below the negative lift incidence. This was due to the effects of obliquity being reversed at negative lift, since the flow through the damage was also reversed.
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Figure 6.17: Lift coefficient increments for a 20%c hole with varying chordwise centres.

Figure 6.18: Drag coefficient increments for a 20%c hole with varying chordwise centres.

Figure 6.19: Pitching moment coefficient increments for a 20%c hole with varying chordwise centres.
6.4 Effects of Skew

Due to the constant spanwise pressure on the undamaged model it was not anticipated that skew would have a significant effect. The discussion will focus on the 20%c cases. 40%c results followed the same trends as the 20%c cases. Across the incidence range, up to stall, the addition of skew had little effect on the lift, drag and pitching moment coefficient increments. Any variations that were present were within experimental repeatability bands.

Surface flow visualisation photographs for 0° and 45° skew cases are shown in Figure 6.20, for 8° incidence. 30° and 60° skew were very similar. These figures indicate that the jets were of similar strength, and the addition of skew did not change the jet properties and characteristics significantly, and were consistent with the coefficient increment results. The main difference between the surface flow visualisation for the straight through case (Figure 6.20a) and the skew case (Figure 6.20b) was that asymmetry was introduced into the flow. Compared to Figure 6.20a it is apparent that with the addition of skew, the jet was twisting as it exited the hole. In all skew cases, the small contra-rotating vortices that sit on the hole edges were no longer at the same chordwise locations: the left hand vortex was slightly enlarged and had not moved as close towards the leading edge of the hole, compared to the right hand vortex. In addition, the large vortices at the trailing edge of the reverse flow region were seen to be no longer in the same chordwise locations. The right hand vortex can be seen to be sitting closer to the rear edge of the panel than the left vortex. Later investigations, discussed in section 6.6, found the asymmetry to be due to internal flows within the cavity.
Figure 6.20: Surface flow visualisation for the upper surface on 20% c holes with varying skew at 8° incidence.
6.5 Effects of Obliquity and Skew Combined

A more realistic attack scenario would likely involve a combination of both obliquity and skew. Figures 6.21 to 6.23 show the coefficient increments for a 20%c hole with +60° obliquity and 45° skew, compared to a hole with just +60° obliquity and a straight through hole. For $dC_l$ (Figure 6.21), the combined case matched well to the case with only obliquity, including the previously discussed weakening of the jet beyond approximately 6° incidence. With the exception of some deviation at stall the combined case generally produced $dC_l$ values comparable to those of the case with just obliquity. Any differences in $dC_l$ were within experimental repeatability bands.

Similarly, $dC_d$ (Figure 6.22) for the combined case showed minimal change from the oblique case, indicating the jets were operating at comparable strengths and size. Finally, a reasonable match with the oblique data at incidences below 8° was witnessed for $dC_m$ (Figure 6.23). Figures 6.21 to 6.23 showed that for extreme positive obliquity angles, the addition of skew did not greatly affect the aerodynamic coefficients, with the damage hole behaving very similarly to an equivalent hole with just obliquity. This was expected, since adding skew did not alter the pressure difference between upper and lower holes on a two-dimensional wing. The positive $dC_l$ and negative $dC_d$ values above 12° incidence were due to the onset of stall. As has been previously stated, recording accurate and repeatable data post-stall was more challenging and as such repeatability was reduced. The errors were likely introduced from different stall characteristics between the undamaged and damaged aerofoils.

Figures 6.24 to 6.26 show the coefficient increments for a 20%c hole with -60° obliquity and 45° skew, compared to straight through and -60° obliquity holes. Similar to the combined case with positive obliquity, the combined case with negative obliquity showed a reasonably close match to the -60° obliquity data. The lift coefficient increments (Figure 6.24) showed a greater loss of lift at higher incidences for the combined case, compared to the -60° obliquity data. This suggests a strengthening jet, however surface flow visualisation and the slight reduction in drag coefficient increments (Figure 6.25) indicated this was not the case. Surface flow visualisation showed little change in wake size and that asymmetry had been introduced by the addition of skew. This asymmetry appeared generally similar for both combined cases, and was not seen to vary with the addition of negative obliquity.

As previously stated, making the obliquity angle more negative increased the strength of the damage jet with the wake, covering a larger area of the aerofoil surface. It is possible that the asymmetry introduced by adding skew to the hole caused a small change to the coefficient increments, compared to an equivalent hole without skew, which became more apparent as the wake increased in size.
Figure 6.21: Lift coefficient increments for a 20%c hole with +60° obliquity and 45° skew.

Figure 6.22: Drag coefficient increments for a 20%c hole with +60° obliquity and 45° skew.

Figure 6.23: Pitching moment coefficient increments for a 20%c hole with +60° obliquity and 45° skew.
6.5. Effects of Obliquity and Skew Combined

Figure 6.24: Lift coefficient increments for a 20%c hole with -60° obliquity and 45° skew.

Figure 6.25: Drag coefficient increments for a 20%c hole with -60° obliquity and 45° skew.

Figure 6.26: Pitching moment coefficient increments for a 20%c hole with -60° obliquity and 45° skew.
6.6 Discussion of Flow Features Within the Cavity

To gain an understanding of the flow behaviour within the cavity, and in particular to explain the unusual behaviour of the $+60^\circ$ obliquity case, the inner surfaces of the panels were smoothed and flow visualisation mixture applied. The internal surface was not as smooth as the external resin surface but it was deemed any roughness present on the internal surface was not adversely affecting the flow features. It should be noted that unlike flow visualisation undertaken on the upper surfaces, the wind tunnel was switched off and the panels removed from the model before photographs were taken. This resulted in the mixture flowing under gravity, which would have distorted some of the images. All photographs are presented for the upper and lower panels, at $0^\circ$, $4^\circ$ and $8^\circ$ incidence.

Figure 6.27 shows a series of photographs for a 20%c straight through hole. A sketch illustrating the deduced flow paths at half span for the aerofoil at $8^\circ$ incidence is shown in Figure 6.28. At the lower incidences, a jet structure similar to that witnessed on the outer surface appeared to form on the lower surface inside the cavity (Figures 6.27a and 6.27c). A wake downstream of the hole (Point A) was visible, although by $4^\circ$ this had been disrupted by flow impinging upon the rear spar. In addition the small contra-rotating vortices on the hole trailing edge were present (Point B), suggesting the flow had similar characteristics to a weak jet, as seen on the outer surfaces. This is not surprising given that the flow through a hole on a panel in isolation would still cause the jet to spread upon exiting the hole, irrespective of whether there is any cross flow at that location or not. Due to the smaller pressure difference between the internal and external sides of the lower surface, this jet was generally exhibiting weak jet characteristics. As the flow entered the cavity it was seen to flow along the lower surface and impinge upon the rear spar. This then caused the flow to be deflected forwards (Point C) towards the exit hole. This deflection may have then resulted in the flow recirculating on either side of the jet, forming the vortices shown in Point D.

At $0^\circ$ the flow appeared to spread further into the upper cavity than at $4^\circ$ or $8^\circ$, with the lower surface showing a contrast. Combined with Point D migrating forwards as incidence increases, this suggests that the jet is spreading within the cavity and then at the higher incidences impacting on the upper surface. This is likely due to the greater momentum possessed by the flow at higher incidences. A recirculation effect was seen on the upper surface of the cavity at all incidences (Point D, and Point Y on Figure 6.28), and on the lower surface at $8^\circ$. This was caused by the flow through the lower surface hole impacting the rear of the cavity. Comments were made during analysis of the external surface flow visualisation regarding the hole becoming full as the jet strength increased. This can be seen by comparing the point at which the region of disturbed flow intersected the upper
surface hole at different incidences or jet strengths (Point E, Figures 6.27b, 6.27d and 6.27f). It can be seen that as the incidence increased this disturbance moved towards the leading edge of the hole and directly correlated to the point the jet exited on the outer surface (Point E). This does not however show any airflow which may be exiting straight through the holes without spreading into the cavity. This throughflow has been illustrated in the sketch, in Figure 6.28, and labelled as Point Z.

Figure 6.29 shows photographs for a 20%c hole with +60° obliquity. A sketch showing the deduced flow at half span for this configuration at 8° incidence is shown in Figure 6.30. At 0°, both lower and upper surfaces (Figures 6.29a and 6.29b) show similar features to the straight through hole: weak jet features were visible on the lower surface, including the small contra-rotating vortices (Point A), and the jet was seen to spread along the rear spar and onto the upper surface. The spreading and resulting flow field features were more apparent than for the straight through case, likely due to the proximity of the upper surface hole to the rear of the cavity. This left the flow nowhere to spread downstream of the hole and forcing it to instead spread out in a spanwise direction (Point B). This spreading into the cavity was seen to increase at 4° on the upper panel (Figure 6.29d), which was likely the result of flow from the lower hole hitting the rear spar and recirculating around the cavity, spreading in a spanwise direction, before exiting the upper hole. It is interesting to note that the jet does not appear to strengthen much on the lower surface between 0° and 4°, with the wake remaining a similar size and the contra rotating vortices in a similar place. This was attributed to a comparatively small change in the pressure across the hole on the lower surface as the incidence increased.

At 8° (Figures 6.29e and 6.29f), both upper and lower panels show very different flow structures. Little evidence of flow attaching to the lower surface (Figure 6.29e) was noticed. There was a collection of fluid downstream of the hole (Point C), which was acknowledged to be due to a large separation region in this area. This would suggest that the flow had become entrained into the jet at this point, as illustrated at Point W in Figure 6.30, and impinged on the rear spar, causing some disturbance at the aft of the cavity (Point X). The flow visualisation patterns on the upper surface appeared to support this theory. Forward of the upper hole, and approximately in line with the lower hole, it is apparent where the flow through the lower hole has struck the upper panel (Point D in Figure 6.29e, and Point Z in Figure 6.30) and proceeded to spread out to approximately one hole diameter either side of the upper surface hole, before exiting through the hole. Low speed smoke testing confirmed that, unlike other damage configurations, at higher incidences (above 6°), a significant portion of the airflow appeared to travel straight through the two damage holes on the +60° obliquity case (Point Y), with little spreading into the cavity. This would support the large separation region on the lower panel at 8°, caused by the flow separating
Figure 6.27: Internal flow visualisation for a 20%c hole with no obliquity or skew (leading edge at the bottom of the photograph).

Figure 6.28: Chordwise sketch of the flow through the cavity for a 20%c straight through hole at 8° incidence.
aft of the hole, and the apparent weakening of the jet on the outer surface as incidence increased beyond 6°.

When the internal flow visualisation for the +60° case was compared to other cases, the lack of spreading into the rear of the cavity at 8° was apparent. This therefore suggests that the cause of the “anomaly” in the +60° obliquity data was a result of the hole and cavity geometry. This allowed the airflow to pass through the cavity from the lower surface to the upper surface with little recirculation inside the cavity, and the only significant spreading of the flow occurring close to the hole on the upper surface of the cavity. As a result, the airflow produced a different wake structure on the upper external surface, resulting in a weakened jet, compared to other obliquity cases, due to the lost momentum.

Photographs for a 20%c hole with -60° obliquity are shown in Figure 6.31. Sketches of flow paths at mid span and at the chordwise centre of the lower hole for 8° incidence are shown in Figure 6.32. The three photographs from the lower surface (Figures 6.31a, 6.31c and 6.31e) show generally similar structures, with little change due to incidence. Unlike the straight through and positive obliquity cases, there was no visible spreading of the flow along the rear spar on the lower surface. This was due to the proximity of the lower hole to the spar, leaving little space downstream of the hole for the flow to spread. Instead the flow impacted the upper surface (Point B in Figures 6.31b, 6.31d and 6.31f, Point Z in Figure 6.32) and spread along the rear of the upper panel. This caused a large wake, which resulted in internal recirculation along the chord, at the edges of the wake (Point D in Figure 6.31e, and Point W in Figure 6.32), shown by the two long chordwise collections of flow visualisation mixture. This then caused weak separation lines to form on the lower surface in a chordwise direction as the flow became entrained, approximately matching the shape of the upper spreading region (Point C in Figure 6.31f, and Point Y in Figure 6.32b). The spread on the upper surface (Point X, Figure 6.32) caused the flow visualisation mixture to disperse generally in a chordwise direction forwards, towards the upper hole (streak lines are visible in line with the lower surface hole). Some spanwise spreading of the flow was also witnessed, although primarily the flow spread upstream towards the hole. It can be seen that the flow exited in a similar position around the upper hole, typically half to one third back from the leading edge of the hole (Point E). These are positioned forward of the corresponding contra-rotating vortices on the external surface, which may indicate the angle the flow is exiting the hole at.

The lack of disturbance to the lower surface indicated that the flow was occupying the upper portion of the cavity, and flowing along the upper surface. At all three incidences, collections of flow visualisation fluid were seen on the lower surface, either side of the damage hole, originating from a point in line with the centre of the upper surface hole. This appeared to be due to a suction effect from the flow near the upper surface, as it
Figure 6.29: Internal flow visualisation for a 20% c hole with $+60^\circ$ obliquity (leading edge at the bottom of the photograph).

Figure 6.30: Chordwise sketch of the flow through the cavity for a 20% c hole with $+60^\circ$ obliquity at $8^\circ$ incidence.
6.6. Discussion of Flow Features Within the Cavity

exited the hole. Such an effect was not seen in other photographs, which suggests the relative strength of the flow through the cavity had increased.

Figure 6.33 shows the internal flow visualisation for a 20%c hole with 60° skew. A sketch showing the spanwise flow profile, taken at the chordwise centre of the holes is shown in Figure 6.34. Similarities between the skew case and the straight through case (Figure 6.27) are apparent, particularly on the lower surface, with a weak jet forming and the small contra-rotating vortices visible at the trailing edge of the hole (Point A). As with the straight through hole, the flow spread into the spanwise cavity downstream of the hole was witnessed (Point B), with the degree of spreading increasing as the incidence increased. However, the effects of skew were apparent when analysing the upper surface photographs (Figures 6.33b, 6.33d and 6.33f). Due to the spanwise offset of the holes the jet from the lower hole was flowing as a jet towards the trailing edge (Point C, and Point Z on Figure 6.34), where it then impacted and spread out sideways. This is similar to the straight through case, but with the spanwise offset due to the hole locations. However, it can be seen that as the incidence increases, particularly at 8° (Figure 6.33f) the flow is exiting at a significant angle. This is due to the recirculation within the cavity, as illustrated by Point Y in Figure 6.34. This was identified as the primary cause of the asymmetry witnessed in the external flow visualisation. Due to the increased momentum of the flow, the jet was entraining flow from the lower surface (Point D, and Point X in Figure 6.34), which led to a greater wake on the upper surface.

Generally, the flow within the cavity was constrained to a width of approximately three to four times the hole diameter. It could therefore be suggested that cavity sizes beyond this would not have a significant effect on the jet properties, but reducing the cavity size may begin to adversely affect the hole properties. However, this was not validated during the project. A restricted access CFD report by ONERA [37] undertaken in parallel with this project provided internal streamline simulations for flow within the cavity, presented as chordwise streamlines. Generally the results from the CFD study (the skew case was not presented) were consistent with the internal and external flow visualisation results discussed here.
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Figure 6.31: Internal flow visualisation for a 20%c hole with -60° obliquity (leading edge at the bottom of the photograph).

(a) 0° incidence, lower surface  (b) 0° incidence, upper surface
(c) 4° incidence, lower surface  (d) 4° incidence, upper surface
(e) 8° incidence, lower surface  (f) 8° incidence, upper surface

Figure 6.32: Sketch of the flow through a -60° obliquity hole at 8° incidence.

(a) Chordwise cross section  (b) Spanwise cross section, at the lower hole
6.6. Discussion of Flow Features Within the Cavity

Figure 6.33: Internal flow visualisation for a 20%c hole with 60° skew (leading edge at the bottom of the photograph).

Figure 6.34: Spanwise sketch of the flow through the cavity for a 20%c hole with 60° skew at 8° incidence.
6.7 Summary and Conclusions

The following points can be concluded from the two-dimensional testing:

- Similar flow features and coefficient increment trends were identified on straight through holes, when compared to Irwin’s previous studies [1].

- Jet strength was dependent on hole size, obliquity angle and incidence.

- The effects of damage holes were found to influence the surface pressures over a considerably greater region than was identified by the flow visualisation.

- Positive obliquity weakened the jet, while negative obliquity strengthened the jet.

- The pressure difference between the upper and lower surface holes was found to be the key driver of jet characteristics, with a greater pressure difference (upper surface hole closer to the leading edge pressure peak) present for negative obliquity configurations.

- The geometry of the wing was partially responsible for the $+60^\circ$ obliquity hole not following trends, and the jet not strengthening beyond $6^\circ$ incidence. The flow was found to be separating on the lower internal surface of the cavity, and thus not spreading within the cavity as much as other cases.

- Adding skew to a damage hole caused little difference to the coefficient increments. Asymmetry was witnessed, however, in the surface flow visualisation.

- When a hole had a combination of skew and obliquity, the results tended towards the equivalent hole with only obliquity.

- Flow spread downstream inside the cavity upon entering, unless hole geometry prevented it from doing so. For the negative obliquity cases, the flow instead spread along the upper surface of the cavity. At high incidences with extreme positive obliquity, a large proportion of the flow went straight through the cavity with little spreading.
Three-dimensional testing was undertaken in order to provide validation for a predictive technique, and to determine if the trends identified on two-dimensional models occurred on a wing of finite aspect ratio, in the presence of a spanwise variation in pressure difference. As was discussed in section 5.8, all data presented in this section were obtained at a Reynolds number of 1,000,000.

All surface flow visualisation photographs presented have been rotated from their original orientation such that the flow is from the bottom of the photograph to the top, to remain consistent with the two-dimensional photographs. The root of the wing is shown on the left hand side and the tip is on the right. As previously stated, it should be noted that the flow visualisation mixture was prone to running under gravity. Work by Samaad-Suhaeb [10] to investigate these effects had shown that when the mixture collected as a result of flow separation, despite being prone to flowing under gravity, the location of the vortices and other flow features were not affected. This was proven by horizontal tests on a finite aspect ratio wing. Typically when mounted vertically, the liquid flowed from the upper large vortex, along the reverse flow separation line and into the lower vortex.

7.1 Effects of Spanwise Pressure Variation on Straight Through Damage Holes

As a baseline reference for all damaged cases, surface flow visualisation from the undamaged datum is presented at the three spanwise locations in Figures 7.1 to 7.3. The disturbance seen at mid-span of Figure 7.2 was the result of fluid collecting behind the panel and leaking out. This was not indicative of any flow features in this region. Flow visualisation for the 20%e straight through holes is presented at 8° incidence for the three spanwise locations in Figures 7.4 to 7.6. The surface flow visualisation showed that the root location (Figure 7.4) was being adversely affected by the boundary layer from the wind tunnel floor (reference can also be made to the corresponding undamaged photograph, shown in Figure 7.1),
which showed a similar flow pattern. Despite using experience gained from Render et al [22] to move the root damage further outboard, and incorporating the method from Malik et al [31] to remove leakage effects, it was apparent these were only partially effective. As a result, this constrained the development of the jet and wake. Experimental validation undertaken for a different project in the same wind tunnel, shortly after the completion of this testing indicated the boundary layer to be approximately 80 mm thick. This was consistent with Johl et al [26]. When comparing Figure 7.4 to the centre and tip locations (Figures 7.5 and 7.6) it was apparent that the root jet had been skewed towards the tip, while the centre and tip jets had been skewed towards the root. Because of this influence from the boundary layer, no further discussion will take place on the root cases.

The photograph for the centre location (Figure 7.5) showed that the flow structure of the straight through hole was broadly similar to that of the two-dimensional wing (see Figure 6.4b), but with a small degree of asymmetry introduced. The asymmetry became more prominent at the tip location (Figure 7.6). It should be noted that some of the larger collections of flow visualisation mixture bounding the reverse flow region were affected by gravity, despite photographs being taken while the wind tunnel was running. This led to more fluid collecting on the left hand (root) side of the vortices than on the opposite side. However, this was determined to not affect the flow structures significantly. Unpublished work undertaken prior to this project at Loughborough University had conducted surface flow visualisation on a horizontally mounted battle damaged three-dimensional wing, and found that the flow patterns were very similar, when compared to those from a vertically mounted model. This therefore confirmed that the effects of the fluid running under gravity were negligible.

The asymmetry in the jet at the centre and tip locations confirms the trends found by Render et al [22] and was caused by the spanwise variation in pressure across the wing, and resulting downwash due to the presence of the wing tip. This reduced the pressure difference between the upper and lower surfaces on the outboard edge of the hole, compared to the inboard edge, essentially weakening the jet. This therefore resulted in the jet appearing skewed as it exited the hole. Figure 7.5 shows the asymmetry to be present in the expected orientation (skewing the jet away from the tip) at the central location. The tip location (Figure 7.6), had a more significant degree of asymmetry compared to the central location. At the trailing edge of the panel, the outboard horseshoe vortex had moved inboard by approximately one screw pitch, compared to the central location. With the inboard edge remaining in a similar position, this had also resulted in the wake reducing in width. The disturbance from the tip vortex (Point A) can be seen on the outboard edge of the photograph, and was also visible in Figure 7.3 on the undamaged wing.
7.1. Effects of Spanwise Pressure Variation on Straight Through Holes

**Figure 7.1:** Surface flow visualisation for the undamaged three-dimensional wing, at the root location, at 8° incidence.

**Figure 7.2:** Surface flow visualisation for the undamaged three-dimensional wing, at the centre location, at 8° incidence.

**Figure 7.3:** Surface flow visualisation for the undamaged three-dimensional wing, at the tip location, at 8° incidence.
Figure 7.4: Surface flow visualisation on the three-dimensional wing for a 20\%c straight through hole, at the root location, at $8^\circ$ incidence.

Figure 7.5: Surface flow visualisation on the three-dimensional wing for a 20\%c straight through hole, at the centre location, at $8^\circ$ incidence.

Figure 7.6: Surface flow visualisation on the three-dimensional wing for a 20\%c straight through hole, at the tip location, at $8^\circ$ incidence.
Three-dimensional coefficient increments for the 20\%c straight hole are presented in Figures 7.7 to 7.9 respectively. As was anticipated a significant reduction in magnitude, particularly at the higher incidences, was noted for the tip case when compared to the central case. This was consistent with the reduction in pressure differences towards the tip, and confirmed findings by Render et al [22]. The drag coefficient increments (Figure 7.8) supported this by indicating reduced $dC_D$ values and therefore a weakened jet at the tip location. This effect was most noticeable at incidences where the strong jet was present, above approximately 6°.

Figure 7.9 shows an approximately constant offset present between $dC_M$ at the tip and centre locations across all incidences. It was expected that spanwise positioning would have little effect on pitching moment, given that two-dimensional testing showed this to be driven more by the chordwise location of the hole relative to the moment reference centre, which remained unchanged, since the wing was unswept and untapered. However, given that the pressure difference across the hole was also a key factor in determining $dC_M$ it is therefore likely that the small reduction in $dC_P$ towards the tip has caused the change in $dC_M$ between the centre and tip locations. The values for both locations were however shown to be within the repeatability bands.

Testing of a 40\%c straight through hole was undertaken to determine if the effects from the 20\%c hole were repeatable for different hole sizes. Figures 7.10 to 7.12 show the coefficient increments for this case. The lift and drag coefficient increments (Figures 7.10 and 7.11) both show larger coefficient increments compared to the 20\%c straight through hole. This was consistent with the two-dimensional coefficient increment trends (see section 6.2). In addition, $dC_M$ (Figure 7.12) showed little change from the 20\%c straight through hole, consistent with the holes having the same chordwise centres. As with the 20\%c hole, the $dC_L$ values (Figure 7.10) for the 40\%c hole showed a significant reduction as the damage moved from the central location towards the tip. A significant variation between centre and tip was noted with the drag coefficient increments (Figure 7.11). This was more significant than for the 20\%c hole and was due to the outboard edges of the hole, and therefore the wake, being closer to the tip, which resulted in a greater pressure difference between the inboard and outboard edges of the hole. $dC_M$ values (Figure 7.12) again showed a similar pattern to the 20\%c results, with a similar small offset between centre and tip to the 20\%c case (see Figure 7.9). This therefore supports the statement that it is likely due to the change in $dC_P$ across the hole, between the two spanwise locations. In addition, the trends identified in two-dimensional testing, of increasing magnitude of $dC_l$ and $dC_d$ with hole size were identified in the three-dimensional results.
Figure 7.7: Three-dimensional lift coefficient increments for a 20%c straight through hole, at two spanwise locations.

Figure 7.8: Three-dimensional drag coefficient increments for a 20%c straight through hole, at two spanwise locations.

Figure 7.9: Three-dimensional pitching moment coefficient increments for a 20%c straight through hole, at two spanwise locations.
7.1. Effects of Spanwise Pressure Variation on Straight Through Holes

Figure 7.10: Three-dimensional lift coefficient increments for a 40%c straight through hole, at two spanwise locations.

Figure 7.11: Three-dimensional drag coefficient increments for a 40%c straight through hole, at two spanwise locations.

Figure 7.12: Three-dimensional pitching moment coefficient increments for a 40%c straight through hole, at two spanwise locations.
7.2 Effects of Obliquity

Damage holes 20%c in diameter with ±60° obliquity were tested on the three-dimensional wing. Surface flow visualisation for the -60° obliquity holes are shown in Figures 7.13 and 7.14 for the centre and tip locations. Generally, both cases showed strong jet flow features, as was expected from the two-dimensional results (Figure 6.11e), although the flow was now asymmetric due to the spanwise pressure variation.

The spanwise asymmetry was evident in the large wake at the tip (Figure 7.14) and was consistent with the previously discussed straight through case. Unlike the centre location (Figure 7.13), the wake did not extend beyond the outboard (right) edge of the panel, but was seen to extend considerably beyond the inboard edge of the panel. This was in contrast to the central location, and was a strong indication of the extent that the flow around the wing tip was disrupting the jet development. The extent of the asymmetry made it difficult to fully assess if the jet had weakened, compared to the central location. Unlike the straight through case, which showed a definite weakening of the jet between the central and tip locations, the wakes appeared to be similar in size.

Lift, drag and pitching moment coefficient increments for the -60° obliquity hole are shown in Figures 7.15 to 7.17. Due to limits on balance load, the incidence range was limited for the -60° obliquity case. The basic trends and shapes of \( dC_L \), \( dC_D \) and \( dC_M \) were consistent with the two-dimensional results (Figures 6.12 to 6.14). Small variations in the \( dC_M \) trends, when compared to the two-dimensional data were present and were attributed to the different model mounting configurations. In addition, spanwise variation was consistent with the results from the three-dimensional straight through hole (Figure 7.7). These points indicate that when damage was applied to a more representative aircraft wing there were no significant changes in the trends. Whilst there was a reduction in magnitude of \( dC_L \) towards the tip, the trends were still consistent across the span and with two-dimensional results, showing a steeper gradient after the transition to a strong jet had occurred, compared to the weak jet region.
7.2. Effects of Obliquity

Figure 7.13: Surface flow visualisation on the three-dimensional wing for a 20%c hole with -60° obliquity and no skew, at the centre location, at 8° incidence.

Figure 7.14: Surface flow visualisation on the three-dimensional wing for a 20%c hole with -60° obliquity and no skew, at the tip location, at 8° incidence.
Figure 7.15: Three-dimensional lift coefficient increments for a 20\%c hole with -60° obliquity and no skew, at two spanwise locations.

Figure 7.16: Three-dimensional drag coefficient increments for a 20\%c hole with -60° obliquity and no skew, at two spanwise locations.

Figure 7.17: Three-dimensional pitching moment coefficient increments for a 20\%c hole with -60° obliquity and no skew, at two spanwise locations.
7.2. Effects of Obliquity

Surface flow visualisation photographs at 8° incidence for the +60° obliquity case at the centre and tip locations are shown in Figures 7.18 and 7.19. The figures show generally similar wake sizes and shapes between the central and tip locations, in contrast to the straight through and -60° obliquity cases. In addition, a small degree of asymmetry was noticed. This suggests that the effects of the asymmetry were most significant on fully developed strong jets. A similar flow pattern to the two-dimensional flow visualisation (Figure 6.11a) was noticed at both locations. It is apparent the jet had not developed to the same strength of the straight through or -60° obliquity cases, and the wake was relatively small as a result. This indicates that the effect of the jet failing to strengthen, which had been noted for the +60° obliquity case during two-dimensional testing were repeatable and could be attributed to the internal geometry.

Coefficient increments for the +60° obliquity case are shown in Figures 7.20 to 7.22. Both lift (Figure 7.20) and drag coefficient increments (Figure 7.21) showed a significant reduction in magnitude compared to the straight through and -60° obliquity holes. This was consistent with the two-dimensional data (Figures 6.12 and 6.13), which also showed lower magnitude coefficient increments for the +60° obliquity case. It is also important to note that the “unusual” behaviour identified during two-dimensional testing of the jet failing to strengthen beyond approximately 8° incidence was repeatable on the three-dimensional model. This was not surprising, given the identical internal geometry (spar placement and hole location) between the two models.

The drag coefficient increments (Figure 7.21) showed little variation across the span between the centre and tip locations. This suggested that the effects of spanwise variation only became prominent when jets above a certain strength were present. Alternatively the jet properties are a function of the pressure distribution in the vicinity of the hole, rather than just across the hole. It was also likely that any spanwise effects for weaker jets could not be distinguished outside of experimental repeatability bands. Surface flow visualisation showed similar wake sizes, which was supported by the similar $dC_D$ values between the centre and tip cases. When compared to other obliquity angles, the pitching moment coefficient increments (Figure 7.22) were again consistent with the expected general trends from two-dimensional testing (Figure 6.14), which showed little to no effect of the battle damage on the pitching moment coefficient increments at lower incidences (below approximately 7° incidence). It is worth noting that, as with the two-dimensional data, the +60° obliquity case showed a very small, and at times, negligible effect on the aerodynamics of the wing.
Figure 7.18: Surface flow visualisation on the three-dimensional wing for a 20%\(c\) hole with +60\(^\circ\) obliquity and no skew, at the centre location, at 8\(^\circ\) incidence.

Figure 7.19: Surface flow visualisation on the three-dimensional wing for a 20%\(c\) hole with +60\(^\circ\) obliquity and no skew, at the tip location, at 8\(^\circ\) incidence.
7.2. Effects of Obliquity

Figure 7.20: Three-dimensional lift coefficient increments for a 20%c hole with $+60^\circ$ obliquity and no skew, at two spanwise locations.

Figure 7.21: Three-dimensional drag coefficient increments for a 20%c hole with $+60^\circ$ obliquity and no skew, at two spanwise locations.

Figure 7.22: Three-dimensional pitching moment coefficient increments for a 20%c hole with $+60^\circ$ obliquity and no skew, at two spanwise locations.
7.3 Effects of Skew

Unlike the two-dimensional testing, the direction of skew was potentially important, due to the spanwise variation in pressure on the three-dimensional wing. By testing positive and negative angles of skew (where positive skew moved the upper hole towards the tip) this could be validated. Analysis of the surface flow visualisation for both positive and negative skew cases (Figures 7.23 and 7.24) allowed for identification of any changes in the flow features resulting purely from spanwise pressure variation due to the skew angles. For reference, the 20%c straight through flow visualisation at both locations is reproduced in Figure 7.25. Given the variation in surface pressure distribution across the holes, any such changes in pressure difference were likely to be minor, and most prominent at the tip. Little difference was present between the two cases at the centre location (Figure 7.23). The changes in asymmetry can be identified by the differing chordwise positions of the two large vortices downstream of the hole. Both wakes were of approximately equal size, and therefore strength. Depending on the skew angle, the asymmetry from the spanwise pressure variation masked the asymmetry induced by the skew, as was apparent with the negative skew, shown in Figure 7.23a.

The surface flow visualisation for the tip location (Figure 7.24) showed a smaller, weaker jet for positive skew, when the upper surface hole was closest to the tip (Figure 7.24b), compared to the negative skew (Figure 7.24a). The asymmetry introduced by the skew (in addition to the asymmetry present from the three dimensional flow characteristics) was least apparent at this location, due to the much greater skewing of the jet by the spanwise pressure gradient across the wing. At the tip the pressure gradient across the hole would have been greater than at the central location, hence the larger difference between positive and negative skew.

Lift, drag and pitching moment coefficient increments for $\pm 45^\circ$ skew on a 20%c hole are shown in Figures 7.26 to 7.28 respectively, with central and tip locations plotted separately. When comparing the lift coefficients for the two skew angles at the central location (Figure 7.26a), very little difference was noticed, until approximately 10$^\circ$ where the two curves diverged. This was unexpected, and was not noticed in the two tip cases (Figure 7.26b), which showed a close match across all incidences. The offset at the central location was attributed to a minor and previously undetected installation error on the $+45^\circ$ skew case causing a slightly premature stall. Aside from this error, the data indicated that the small spanwise change in pressure difference across the hole did not have a noticeable effect on the coefficient increments.

Similarly, for the drag coefficient increments (Figure 7.27) a close match was noticed between the positive and negative skew cases, at the central spanwise locations (Figure 7.27a).
7.3. Effects of Skew

Figure 7.23: Surface flow visualisation on the three-dimensional wing for a 20%c hole with ±45° skew, at the centre location, at 8° incidence.

Figure 7.24: Surface flow visualisation on the three-dimensional wing for a 20%c hole with ±45° skew, at the tip location, at 8° incidence.

Figure 7.25: Reference surface flow visualisation images for the 20%c straight through case, at 8° incidence.
This supported the similar wake sizes shown in the flow visualisation photographs (Figure 7.23). A more significant variation was witnessed at the tip location, with the negative skew case (Figure 7.27b) yielding larger $dC_D$ values above approximately 7° incidence. This correlated with the stronger jet witnessed in the surface flow visualisation, compared to the +45° skew case (Figure 7.24). With the upper surface hole closer to the wing tip on the +45° skew case, this would have resulted in a reduced pressure differential across the holes, compared to the negative case. While this difference was not the same order of magnitude as the variation between positive and negative obliquity, it was sufficient to reduce the drag coefficient increments and weaken the jet, as shown in the surface flow visualisation photographs. This did not appear to influence $dC_L$ to the same degree, which suggests the size of the wake is more critical in determining the change in drag than the change in lift. The pitching moment coefficient increments (Figure 7.28) followed the same trends as the lift and drag results, showing little variation across the span, apart from at extreme ends of the data set.
7.3. Effects of Skew

![Graphs showing lift coefficient increments with varying skew angles.](image)

(a) Central location  
(b) Tip location

**Figure 7.26:** Three-dimensional lift coefficient increments at both spanwise locations for a 20%c hole with varying skew angles.

![Graphs showing drag coefficient increments with varying skew angles.](image)

(a) Central location  
(b) Tip location

**Figure 7.27:** Three-dimensional drag coefficient increments at both spanwise locations for a 20%c hole with varying skew angles.

![Graphs showing pitching moment coefficient increments with varying skew angles.](image)

(a) Central location  
(b) Tip location

**Figure 7.28:** Three-dimensional pitching moment coefficient increments at both spanwise locations for a 20%c hole with varying skew angles.
7.4 Effects of Different Panel Combinations on Coefficient Increments

The design and structural integrity of the three-dimensional model allowed for testing of more extreme damage combinations, including removing entire panels, which was not possible on the two-dimensional model. Testing on different combinations of damage holes, as summarised in Table 7.1 allowed further analysis into the jet characteristics, and to provide an insight into more complex damage configurations, where the entry and exit hole differed in area. Notation used in figures is also shown in Table 7.1, consisting of the form “upper hole/lower hole”. 20%c and 40%c holes were used as extensive coefficient increment data existed, and time did not permit the manufacture of intermediate hole sizes. All testing was undertaken at the central location to minimise spanwise effects. Damage holes were centred at half chord and mid span of the panel. When a configuration was tested with a panel removed, no attempt was made to smooth the recesses into which the panel sat, as it was felt this provided a more realistic scenario, when an entire panel has been lost due to damage (possibly from an internal explosion, such as a fuel tank, a structural failure, or through hydrodynamic ram effects from fuel).

<table>
<thead>
<tr>
<th>Upper surface</th>
<th>Lower surface</th>
<th>Notation</th>
</tr>
</thead>
<tbody>
<tr>
<td>20%c</td>
<td>Undamaged panel</td>
<td>20/U</td>
</tr>
<tr>
<td>20%c</td>
<td>40%c</td>
<td>20/40</td>
</tr>
<tr>
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<td>No panel</td>
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</tr>
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<td>20%c</td>
<td>N/20</td>
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<tr>
<td>No panel</td>
<td>No panel</td>
<td>N/N</td>
</tr>
</tbody>
</table>

Table 7.1: Summary of panel combinations tested, showing notation used throughout.

Flow visualisation photographs of the lower surface have been digitally enhanced to improve clarity as a result of lower light levels due to the lighting configuration in the wind tunnel. Lower surface flow visualisation photographs are presented in the same orientation as upper surface photographs, with airflow from bottom to top, and the wing tip towards the right of the photograph. Figure 7.29 shows the surface flow visualisation photographs for the datum case of a 20%c hole on the upper and lower panels, at 8° incidence. Figure 7.29a repeats the upper surface photograph shown previously. Figure 7.29b shows the corresponding lower surface photograph; it is apparent the contrast in wake structure between the two surfaces. Very little disturbance from the hole on the lower surface was present, which was consistent with the small pressure changes witnessed on the lower surface during undamaged testing. A minimal wake was present downstream of the damage, no wider than the hole diameter.
Figure 7.30 shows the flow visualisation photograph for the 20%c hole through the upper surface only. The surface flow features identified were representative of those found around an “open” cavity, with a small wake and region of sluggish flow downstream of the hole. ESDU [38] define an “open cavity” as having a length to depth ratio of between 1 and 2: a 20%c hole centred at half chord generates a length to depth ratio of approximately 1.2 at maximum diameter. Render et al [19] undertook studies on battle damage repairs, which involved covering a hole on one surface, thus leaving a cavity present on the other. Flow features in Figure 7.30 supported those suggested by ESDU [38, 39] and Render et al [19], with a wake and recirculation region aft of the cavity. No presence of a jet was witnessed as there was no throughflow. However, contrary to suggestions from the ESDU literature, the surface flow visualisation showed the disturbance to extend aft to the trailing edge of the aerofoil. This resulted in a region of sluggish flow, which in places appeared to be flowing against the freestream. This may have been partly due to the effects of gravity on the mixture. ESDU data were presented for flat plates, and therefore it is likely that the varying chordwise surface pressure of the aerofoil was altering the wake formation. When the damage was located on the lower surface only (Figure 7.31) a narrow wake similar to that witnessed in Figure 7.30 was seen, but with less disturbance to the flow. The wake from the hole on the lower surface also had similarities with cavity flows, although it did not appear to affect the flow beyond the trailing edge of the panel, likely due to a small step present in the transition between the panel and solid model. In both cases, the wake was bounded by a dividing streamline, in a similar location to the horseshoe vortex on through damage cases. This ran parallel to the freestream, and exiting from the cavity at the widest point of the hole. For both configurations, the wake itself generally remained no wider than the hole.

Figures 7.32 to 7.34 compare $dC_L$, $dC_D$ and $dC_M$ values from a case with a 20%c hole on the upper surface and an undamaged panel on the lower surface, and an undamaged panel on the upper surface and a 20%c hole on the lower surface with the conventional 20%c straight through case. The lift coefficient increments (Figure 7.32) show very little change in lift from the undamaged case, compared to the through hole. The loss of lift was generally constant, with no weak or strong jet effect witnessed due to there not being any throughflow. A greater loss of lift (although still within experimental repeatability bands) was noted for the 20/U configuration. This result was consistent with findings by Render et al [19], when repairing a through damage hole on one surface. By restoring the geometry of the upper suction surface, a greater improvement would be expected, as the greater surface pressure is present on the upper surface and thus a hole on this surface would result in a greater loss of pressure, and therefore loss of lift.

Similar results were found for the drag coefficient increments (Figure 7.33), with the two
Figure 7.29: Surface flow visualisation at 8° incidence for a 20%c straight through hole (Configuration 20/20).

Figure 7.30: Surface flow visualisation at 8° incidence showing the upper surface for a 20%c hole, with an undamaged panel on the lower surface (Configuration 20/U).

Figure 7.31: Surface flow visualisation at 8° incidence showing the lower surface for a 20%c hole, with an undamaged panel on the upper surface (Configuration U/20).
7.4. Effects of Different Panel Combinations on Coefficient Increments

“cavity” cases again producing a negligible change in drag from the undamaged case compared to the straight through hole. This supported the surface flow visualisation photographs, which showed small wakes downstream of the hole, compared to the large wake with strong vortices and reverse flow of the through holes. The $dC_D$ data showed slightly larger coefficient increments at higher incidence for the hole through the upper surface, which supported the more visible wake, compared to the negligible increments of the hole through the lower surface.

Pitching moment coefficient increments (Figure 7.34) showed similar trends to the lift and drag coefficient increments, with both damage cases causing little change from the undamaged data. With the hole on the upper surface, the pitching moment generally became more negative, whereas the reverse was true for the hole on the lower surface. This was consistent with the undamaged surface pressure coefficients across the holes, which were generally larger in magnitude for the hole through the upper surface.

The effects of different sized holes, and therefore the effects of a small damage hole potentially “throttling”, or restricting a larger hole, are considered by placing a 20%c hole on one surface and a 40%c hole on the other. Flow visualisation for the case with a 20%c upper hole and a 40%c lower hole is shown in Figure 7.35a. When comparing the upper surface to the 20%c straight through case (Figure 7.35b) it can be seen that the wake was slightly wider and the jet was running more full (the small contra-rotating vortices had moved further round towards the leading edge of the hole). This is consistent with the upper surface hole being supplied with a greater flow rate, due to the larger inlet hole.

Figure 7.36 provides an interesting insight into the flow being constrained by the entry hole, rather than the exit hole. The 40%c straight through flow visualisation photograph is provided in Figure 7.36b for reference. The flow features on the upper surface for the 40/20 configuration (Figure 7.36a) suggested that the flow structure was now much more complex, with two pairs of contra-rotating vortices present: one pair the result of flow from the lower surface through the smaller lower hole (label A), generated by the 20%c hole; and another from the flow exiting the cavity through the larger upper hole (label B on both figures) generated by the 40%c hole. The latter were in a similar position to the 40%c straight through hole (see Figure 7.36b). The wake structure was also noticeably different to previous cases, with a much narrower region of reverse flow. The inner pair of vortices (label A) were caused by flow from the 20%c hole. The flow exiting through the 40%c hole was driving the wake size, but the flow from the 20%c hole was likely becoming entrained into the reverse flow aft of the hole, which resulted in the reverse flow region appearing narrower. This was likely due to the flow from the 20%c hole disrupting and occupying the area where the reverse flow region would normally have occurred. In addition, the hole is running less full, with the horseshoe vortex now intersecting the 40%c hole approximately
Figure 7.32: \(dC_L\) values for varying combinations of a 20\% lower hole and an undamaged panel.

Figure 7.33: \(dC_D\) values for varying combinations of a 20\% hole and an undamaged panel.

Figure 7.34: \(dC_M\) values for varying combinations of a 20\% hole and an undamaged panel.
7.4. Effects of Different Panel Combinations on Coefficient Increments

a quarter of a diameter aft of its leading edge, unlike the 40/40 configuration where the horseshoe vortex intersects the hole at the leading edge.

Figures 7.37 to 7.39 show the coefficient increments for these two cases. Additionally 20\%c and 40\%c straight through holes are shown for reference. When considering the lift coefficient increments (Figure 7.37), it can be seen that for the 20/40 configuration, the results tended towards that of the 20\%c straight through hole. Similarly, for the 40/20 configuration, above approximately 4° incidence the \(dC_L\) values were closer to the 20\%c straight through hole than the 40\%c straight through configuration. This therefore suggests that as well as the chordwise position of the holes, the mass flow through the hole is also critical. In both cases, the throughflow was restricted by the smaller hole, which caused the results to tend toward the equivalent smaller straight through hole (i.e. the 20/20 configuration). Above approximately 5° incidence the 20/40 configuration caused a greater loss of lift compared to the 40/20 due to the larger lower hole allowing more mass flow through. This led to the hole running in an “over-full” state (i.e. the jet flow spreading outwards significantly more upon exiting the hole than the other cases), as was seen by the forward position of the horseshoe vortex in Figure 7.35a. This had become more prominent, causing a greater loss of lift once the strong jet has developed, from approximately +5° incidence. Conversely, with the 40\%c hole on the upper surface, initial lift loss was consistent with the 40\%c straight through case, but as soon as the jet began to develop and strengthen the restriction from the smaller inlet hole became apparent. Little increase in jet strength was noted, based on the reasonably constant coefficient increments above approximately 2° incidence. Beyond 7° incidence the loss of lift was less than for the 20\%c straight through configuration, due to the jet not strengthening.

Similar results were witnessed for \(dC_D\) (Figure 7.38). The key difference was that both “throttled” cases had similar weak jet performance, initially matching that of the 40\%c straight through case, but then tended towards the 20\%c straight through case. This suggested that for the 20/40 configuration, the larger inlet hole was providing a greater flow rate through the hole than in the 20\%c straight through case, and thus causing a larger wake at the lower incidences. This increase in wake size correlates with the increase in drag and a loss of lift. However, as incidence increased the \(dC_D\) data began to match that of the 20\%c straight through case, indicating that the smaller hole on the upper surface was restricting the airflow and jet formation. This was supported by the surface flow visualisation photographs. It is likely that close to the zero lift incidence the flow was behaving more like a cavity, which would have been dominated by the 40\%c hole.
Figure 7.35: Surface flow visualisation on the upper surface at 8° incidence for a 20%c hole on the upper surface and a varying hole size on the lower surface.

Figure 7.36: Surface flow visualisation on the upper surface at 8° incidence for a 40%c hole on the upper surface and a varying hole size on the lower surface.
7.4. Effects of Different Panel Combinations on Coefficient Increments

Figure 7.37: $dC_L$ values for varying combinations of a 20%c hole and a 40%c hole.

Figure 7.38: $dC_D$ values for varying combinations of a 20%c hole and a 40%c hole.

Figure 7.39: $dC_M$ values for varying combinations of a 20%c hole and a 40%c hole.
The 40/20 configuration matched the 20/40 configuration well, until approximately 6° incidence. Above this incidence there was no increase in $dC_D$, which indicated the jet was not strengthening. This behaviour was similar to that of the +60° obliquity hole, where the flow was becoming entrained into the freestream more quickly than other cases, and thus not spreading inside the cavity. In addition, this would have resulted in little spreading onto the upper surface upon exiting the hole. Surface flow visualisation confirmed the lower $dC_D$ values, with the wake appearing narrower and less energetic than either of the cases with the 20%c hole on the upper surface. Pitching moment coefficient increment data (Figure 7.39) for all cases was consistent with previous testing of straight through holes centred at half chord, showing no significant variation with hole diameter, irrespective of the combination of hole sizes used.

It is apparent from this set of data that there is not a simple link between hole size alone and jet properties, as the combination of hole sizes is important. If the hole is not able to operate full (as with the 40/20 configuration), then the jet cannot develop fully. However, once the hole is able to operate in a full or “over-full” condition (as with the 20%c hole on the upper surface and the 40%c hole on the lower surface), a significant effect on the aerodynamic coefficients occurs.

The final set of tests involved removing an entire panel. Initially, one panel was removed from the upper or lower surface and a 20%c hole left on the opposite. Flow visualisation was limited when a panel was removed due to the large surface area removed, however some flow features could still be identified. Figure 7.40 shows flow visualisation from the upper surface with a 20%c hole and no panel on the lower surface, and Figure 7.41 shows the opposite, with no panel on the upper surface. It is apparent from Figure 7.40 the effects due to increased mass flow through the 20%c hole, having removed the lower panel. Compared to Figure 7.35 (40%c on the lower surface) and Figure 7.29a (20%c on the lower surface), the position of the forward horseshoe vortex is the most noticeable difference. On previous cases it was sitting on the leading edge of the hole, whereas in this case it had moved a significant distance (approximately a quarter diameter) forward. This has resulted from a large over expansion of the jet as it exits the hole, due to the increased mass flow from the larger inlet area. In turn this increased the size of the wake which would likely have a significant impact on $dC_D$. Another feature which suggests a greater flow rate of air through the hole was the position of the two small contra-rotating vortices on the hole perimeter. For the original straight through case, these were aft of the point of maximum diameter, whereas in this case they have both moved forward. From the internal flow visualisation on the two-dimensional cavity this typically indicated where the jet was exiting the hole, and thus in this configuration the jet was seen to be occupying this greatest area of the hole, running at maximum strength.
With the upper panel removed and a 20%c hole restricting flow through from the lower surface (Figure 7.41), a significant wake was present. However, upon closer analysis of the upper surface it is apparent, as with the 40%c hole being throttled by the 20%c hole, a small jet had formed, exiting from the 20%c hole. This was inside the main wake from the large cavity. This has generated a small region of reverse flow at the centre of the panel, around the larger vortices and wake coming from the cavity. The large wake was likely a result of separation caused by the cavity bounds, and from flow spreading out of the cavity, into the freestream. Based on ESDU [38], the cavity, with a length to depth ratio of approximately 3 is likely to be classed as either open or open-transitional (i.e. on the border between an open and transitional cavity). Typically open cavities have length to depth ratios of up to approximately 2 [38]. The flow features around the cavity appeared consistent with this, although additional disturbances would have originated from the steps present on the chordwise ribs.

Removing both panels produced the flow pattern shown in (Figure 7.42). Compared to Figure 7.41, which had the lower surface restricted by a 20%c hole, it is evident that the large disruption had developed into a strong jet, with a greater degree of recirculation, and a clear region of reverse flow in the centre of the trailing edge. This region was approximately three quarters the width of the cavity. The overall spanwise width of the wake was similar to Figure 7.41, which suggests the wake size was dictated by the wing area removed. Despite the considerably larger hole, which was also rectangular in shape, instead of circular, the basic jet features were still present, in particular the horseshoe vortex and the region of reverse flow. This is consistent with published work on rectangular jets in cross flow, from Krothapalli and Shih [40], who studied rectangular inclined jets in crossflow. Krothapalli and Shih showed that when the jet was perpendicular to the freestream, the same features from circular jets and battle damage studies were identified. This suggests that flow through a square hole would be expected to generate the same features as flow through a circular hole, as was witnessed in the flow visualisation in Figure 7.42.

The coefficient increments for the configurations with entire panels removed are shown in Figures 7.43 to 7.45; it should be noted that the axes on these figures are different scales to previous figures. The 20%c and 40%c straight through data are included as reference datums. As would be expected, all three figures show a substantial loss of aerodynamic performance due to the removal of a panel. The lift coefficient increments (Figure 7.43) show that when both panels were removed, there was a substantial loss of lift (approximately twice that of the 40%c straight through case and six times that of the 20%c straight through case at 10° incidence). This was partly due to the loss of a significantly larger proportion of the surface area on each side of the wing compared to the two circular holes, but a more prominent factor was the increased mass flow through the large resulting hole.
Figure 7.40: Surface flow visualisation at $8^\circ$ incidence showing the upper surface for a case with a 20\%c hole on the upper surface and no panel on the lower surface (Configuration 20/N).

Figure 7.41: Surface flow visualisation at $8^\circ$ incidence showing the upper surface for a case with no panel on the upper surface, and a 20\%c hole on the lower surface (Configuration N/20).

Figure 7.42: Surface flow visualisation at $8^\circ$ incidence showing the upper surface for a case with no panels on the upper and lower surfaces (Configuration N/N).
With significant flow through the cavity, this would have caused substantial disruption to the lifting surface. The effects of this increased mass flow through the hole and the resulting over expansion were apparent, with the $dC_L$ results from the $N/20$ configuration approximately 50% larger than the $20/N$ configuration at $10^\circ$ incidence. This demonstrates that the loss of lift from the case with both panels removed was driven primarily by volume of flow through the “hole”. With the flow constrained by a 20%c hole on the lower surface, despite the loss of lifting area on the upper surface, the coefficient increments were more in line with the 40%c straight through hole.

The drag coefficient increments (Figure 7.44) were consistent with the lift coefficient increments. When just one panel was removed, generally similar $dC_D$ values were produced, irrespective of which panel was removed, up until approximately 5°. The $dC_D$ values at 8° incidence for the two cases with a 20%c hole suggested both would have similar wakes. However, the flow visualisation (Figures 7.40 and 7.41) showed very different wake sizes and structures, with one being a very strong jet and the other a wider but weaker jet. The coefficient increment values suggest that the two were therefore equivalent.

Pitching moment coefficient increments (Figure 7.45) produced somewhat unexpected results. Previous results have shown $dC_M$ to generally be invariant with hole size, and to only be affected by the chordwise centre of the hole. With the removal of the panel, there was a minimal change in the chordwise centre of the larger rectangular hole (approximately 0.5%c forward). Therefore it was anticipated the $dC_M$ values would be similar to the 20%c straight through case. This was not the case however. No trends were identified, only that having just the upper panel removed caused a greater nose-down pitching moment than both panels removed. This was possibly a combination of the jet from the 20%c hole and the loss of part of the suction surface in the vicinity of the pressure peak on the upper face of the wing. With the upper surface removed this would have significantly altered the pressure distribution, and potentially the centre of pressure (see Figure 5.19 for the undamaged $C_P$ plot at centre span). However, with the lower surface remaining in place, as with the $N/20$ configuration the airflow would still be able to react a force onto the centre region of the aerofoil, and thus potentially cause the more negative $dC_M$ values when coupled with the removal of the upper surface. It is likely that the removal of a panel in these three configurations and the resulting significant disturbance to the pressure distribution was responsible for the large differences in $dC_M$ when compared to the $20/40$ and $40/20$ configurations (Figure 7.39), which still had a large proportion of the suction or pressure surfaces intact.
Figure 7.43: $dC_L$ values for varying combinations of a 20%c hole and a removed panel.

Figure 7.44: $dC_D$ values for varying combinations of a 20%c hole and a removed panel.

Figure 7.45: $dC_M$ values for varying combinations of a 20%c hole and a removed panel.
7.5 Summary and Conclusions

The following points can be concluded from the three-dimensional testing:

- The variation of spanwise pressure distribution associated with a three-dimensional wing has an effect on the coefficient increments, weakening the jet towards the tip, which caused a reduction in magnitude of the lift and drag coefficient increments. These effects increased as the hole was moved closer to the tip.

- The downwash associated from the wing tip causes a skewing of surface flow features away from the tip.

- Surface flow visualisation showed similar features for straight through and oblique holes, when compared with two-dimensional testing.

- The effects of skew were dependent on the skew angle used. Coefficient increments did not see a significant change, but the asymmetry introduced by the skew was affected by the asymmetry from the spanwise pressure variation.

- When a single panel was removed, the hole was found to behave in a similar fashion to a cavity, producing very small coefficient increments and a narrow wake.

- When holes of different sizes were used on the upper and lower surfaces, it was found that the size of the hole on the upper surface determined the flow characteristics, while the size of the hole on the lower surface determined the mass flow rate through the cavity, and therefore the jet strength.

- With a larger hole on the lower surface compared to the upper surface, under positive lift conditions, this led to the jet producing a very large wake and a stronger jet, and the hole being in an “over-full” state.
Chapter 8

Predicting the Aerodynamic Effects of an Untested Damage Hole on a Two-Dimensional Wing

8.1 Introduction

The collected experimental data were used to develop and validate a predictive technique. The prediction method begins with predicting the effects of battle damage on two-dimensional aerofoils, and is then developed into a three-dimensional predictive technique, as will be described in Chapter 9. The aim was to develop a predictive technique that determines the aerodynamic effects of battle damage on the wing of a UAV, but is likely to be suitable for any low speed aircraft. The methods were developed from wind tunnel data obtained between test Reynolds Numbers of 500,000 and 1,000,000, based on the wing chord.

Irwin’s predictive method [1], reviewed in section 2.7 and shown in Appendix A, was found to be invalid when applied to the LS(1)-0417MOD aerofoil. It was evident from Irwin’s methodology that it primarily focused on predicting the strong jet conditions. Due to the performance of the LS-series aerofoil used in this work compared to the NACA aerofoil used by Irwin, this meant that fewer data points were available for plotting the figures shown in Appendix A. As a result the accuracy was heavily degraded, and for a number of cases the methodology could not predict any results. The reduced number of strong jet points (partly due to the earlier stall of the LS aerofoil) particularly hampered the prediction of the drag coefficient increments. As a result, a new methodology was developed.

The goal of the two-dimensional predictive technique was to predict the aerodynamic effects of battle damage on any given aerofoil, using minimal experimental data and without the need for extensive testing of battle damaged cases. In this context, “experimental data” could be obtained from either wind tunnel testing or CFD simulations. It is ideal to minimise the amount of experimental data required for a predictive technique as this reduces the cost and time required for predicting the effects of a previously unknown damage case,
and allowing for a more rapid survivability assessment of the aircraft. However, as will be explained in later sections, a trade-off between accuracy and quantity of experimental data are required. A full worked example of the two-dimensional predictive methodology is presented in Appendix D.

8.1.1 Definition of the Problem

As has been outlined throughout this thesis, when considering the aerodynamic effects of battle damage, a number of variables must be considered. For a two-dimensional predictive technique these include, but are not limited to:

- Variation in hole size;
- Chordwise centres of the upper and lower holes;
- Varying attack directions (i.e. differing obliquity and skew angles).

Therefore a technique is required that will allow for an assessment of a range of damage scenarios as quickly and efficiently as possible. This highlights a number of issues with current testing and aerodynamic analysis techniques: CFD can be too time consuming, due to the time taken to generate the mesh, and the additional time taken to generate the results. Additionally wind tunnel testing would not not feasible for a large range of damage cases, due to costs and time of manufacturing the models and time to carry out tests. Render et al [22] had demonstrated that two-dimensional damage data could be used to predict the effects of damage on a three-dimensional wing. While this reduces testing costs, it is still impractical to test every possible damage case on a two-dimensional model. This illustrates the requirement for a predictive technique which can produce a summary of the aerodynamic effects of battle damage, with minimal experimental or computational testing required.

8.1.2 Key Definitions

The following terms are used throughout the prediction methodology:

- **Two-dimensional:** A wing, with or without damage, tested in a two-dimensional experimental setup.

- **Three-dimensional:** A wing tested with at least one free wing tip, giving a spanwise variation of the surface pressure on an undamaged wing.

- **Pressure Coefficient Difference** ($dC_p$): Difference in upper and lower surface pressure coefficients from an undamaged aerofoil across the location of the hole.
- **Reference data:** Experimentally-obtained two-dimensional data, used to form best fit curves (this can also be obtained through CFD simulations.

- **Experimental data:** Coefficient increment data obtained through experimental testing (although not used in this methodology, this data could also be obtained through CFD simulations or from tools such as XFOIL).

- **Scale factors:** Relationships based on existing experimental data and are used to apply corrections for different parameters.

### 8.1.3 Methodology Overview

The aim of the predictive methodology in this chapter is to predict and obtain two-dimensional lift, drag and pitching moment coefficient increments for an aerofoil with a previously untested single damage hole through the wing. The methodology uses lift, drag and pitching moment coefficient increment data from two reference data sets covering all required incidences. These two data sets are typically taken at the extremes of chordwise locations and for a hole size close to the required value (typically $\pm 10\% c$ of the required diameter). Scatter plots, scale factors and pressure coefficient differences from an undamaged aerofoil at the location of the damage hole are used to obtain the prediction. This data can then be converted to a three-dimensional wing, using the methodology to be discussed in chapter 9. By utilising the pressure coefficient differences, this essentially removed any dependency on the type of jet, and therefore allowed one method to be used for both weak and strong jets. This was due to weak jets occurring when the pressure coefficient difference was small, and the strong jets occurring when the difference was comparatively larger. While there was not a definitive ‘switching point’ for transition from a weak jet to a strong jet for a given aerofoil, the change in gradient of a $dC_p$ curve was scaleable from one case to another.

A brief outline of the methodology is given below:

1. Define the parameters of the damage case to be predicted, including hole size, obliquity and skew angles (thus giving the chordwise locations of the hole centreline).

2. Obtain undamaged pressure coefficient differences between upper and lower surfaces at the damage hole location for the undamaged two-dimensional wing.

3. Select at least two ‘extreme’ damage cases from existing two-dimensional experimental data sets, inside which the chordwise locations of the hole to be predicted fall. This will form the “Reference data set”. A greater number of experimental data sets will improve prediction accuracy. These data sets can be obtained from either wind
tunnel testing or CFD simulations. The reference data set should meet the following conditions:

- Covers extreme chordwise locations of the hole. The location of the hole to be predicted must lie within this range.
- Close to the hole size to be predicted, although an exact match is not required (typically ±10%c from the required diameter).
- Data set contains lift, drag and pitching moment coefficient increments.

4. Calculate pressure coefficient differences as in step 2 for all reference data cases, at all incidences of interest. Create a scatter plot of the coefficient increments against undamaged pressure coefficient difference.

5. Interpolate the scatter plots for the pressure coefficient differences of the damage hole to be predicted.

6. Apply corrections for hole size to give predicted two-dimensional results.

8.1.4 Limitations

A single limitation is applied to the two-dimensional predictive technique. It has only been developed for a single entry and exit hole on the aerofoil (one on each of the upper and lower surfaces), or multiple holes that have sufficient separation between them that there is no interaction and each hole is acting independently. Additional limitations are driven by the data sets used to generate the reference data. However, if a new set of data are being collected for the reference data then the limits can be tailored around the desired damage case. Such limits include:

- Surface pressure coefficient range of the aerofoil
- Obliquity and skew angles
- Incidence range
- Hole size and shape

8.2 Definition of Final Prediction Equations

Equations 8.1 to 8.3 show the three final equations used to predict the lift, drag and pitching moment coefficient increments respectively, for the damage case considered. Each equation makes use of the predicted coefficient increments and scale factors to correct for
8.3. Calculation of Pressure Coefficient Difference

The following sections will define each term within the equations. The absence of $F_{hs}$ should be noted in Equation 8.3, and will be discussed in due course.

\[
dC_{l \, dam} = F_{hs} \times dC_{lp} \quad (8.1)
\]
\[
dC_{d \, dam} = F_{hs} \times dC_{dp} \quad (8.2)
\]
\[
dC_{m \, dam} = dC_{mp} \quad (8.3)
\]

Where:

- $dC_{l \, dam} =$ Predicted two-dimensional lift coefficient increment
- $dC_{d \, dam} =$ Predicted two-dimensional drag coefficient increment
- $dC_{m \, dam} =$ Predicted two-dimensional pitching moment coefficient increment
- $F_{hs}$ = Scale factor for hole size (Equation 8.9)
- $dC_{lp}$ = Lift coefficient increment prediction from a set of experimental data (Equation 8.6)
- $dC_{dp}$ = Drag coefficient increment prediction from a set of experimental data (Equation 8.7)
- $dC_{mp}$ = Pitching moment coefficient increment prediction from a set of experimental data (Equation 8.8)

8.3 Calculation of Pressure Coefficient Difference

As discussed previously, it was determined from earlier studies [1] that the pressure coefficient difference across the damage hole was an important factor in damage jet properties and the effects of damage on the aerodynamic coefficients. Early predictive techniques [1, 22] used the pressure coefficient difference between upper and lower surfaces of the undamaged wing, taken at the chordwise centre of the hole. While this may prove adequate for small holes away from the pressure peak, it is unlikely to yield acceptable accuracy for larger holes, or holes closer to the leading edge. To improve the accuracy of this, the undamaged pressure coefficient was integrated across the surface area of the hole for each of the upper and lower wing surfaces. This method would also allow for spanwise variations in the pressure to be taken into account in the later three-dimensional predictions.

The damage hole on a given surface was divided up into a number of small squares, $n$, of equal size. A value for the pressure coefficient across a given square, $\Delta C_{p \, n}$, was taken...
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(assumed to be a point value at the centre of the square) and multiplied by the area of the square, $\Delta A$. The area weighted pressure coefficient over the hole, $C_{p \, \text{hole}}$, was calculated for the upper and lower surface holes individually and is given by Equation 8.4 below:

$$ C_{p \, \text{hole}} = \frac{\sum (\Delta C_{p \, n} \cdot \Delta A_n)}{\sum \Delta A_n} \quad (8.4) $$

Where:

- $C_{p \, \text{hole}}$ = Area weighted pressure coefficient across the damage hole
- $\Delta C_{p \, n}$ = Pressure coefficient value at the centre of a small square, $n$
- $\Delta A_n$ = Area of a small square, $n$

A trade-off between processing time and accuracy was required. It was found that having a square size of approximately 0.25%c provided the hole area within 3% accuracy and a $C_{p \, \text{hole}}$ value converged to within four decimal places without having a significant effect on processing time.

The pressure difference between upper and lower holes is then found from Equation 8.5, using the $C_{p \, \text{hole}}$ values for the upper and lower surfaces from Equation 8.4.

$$ dC_p = (C_{p \, \text{hole}})_{\text{upper}} - (C_{p \, \text{hole}})_{\text{lower}} \quad (8.5) $$

Where:

- $dC_p$ = Area weighted pressure coefficient difference across the upper and lower surface holes
- $C_{p \, \text{hole}}$ = Pressure coefficient across the hole
- $\text{upper}, \text{lower}$ = Surface on which the hole is being considered

8.4 Predictions for Chordwise Hole Location

The prediction of damage effects centred on the use of the pressure coefficient difference between the damage holes, $dC_p$. This removed a number of geometric factors, including model incidence and chordwise positions of the hole (including obliquity angle). Figures 8.1 to 8.3 show the lift, drag and pitching moment coefficient increments, plotted against $dC_p$, for a range of two-dimensional damage cases using 20%c holes. For each figure, coefficient increment values from the $-60^\circ$, $-30^\circ$, $+30^\circ$ and $+60^\circ$ obliquity, $45^\circ$ skew and straight through cases, including the straight through case centred at 35% chord, were
8.4. Predictions for Chordwise Hole Location

plotted against the corresponding area-weighted $dC_p$ value for the specific hole location and incidence. Values approaching stall onset, and the ‘anomalous’ values above $6^\circ$ incidence from the $+60^\circ$ obliquity case were omitted (see section 6.3). The extreme obliquity cases, $\pm 60^\circ$, have been highlighted. A second order line of best fit through all data sets have been included. The lines of best fit were forced through the origin in all three cases. Experimental data generally showed that when the pressure coefficient difference between upper and lower surfaces was zero, the coefficient increments were very small and could thus be approximated to zero. It should be noted that there is likely to be a very small drag coefficient due to the cavity nature of the hole at zero $dC_p$: this is evident in the experimental data, but due to scatter and inaccuracies associated with measuring the very small values, the approximation of $dC_d = 0$ when $dC_p = 0$ will be used.

Figures 8.1 to 8.3 all show that coefficient increment data from different obliquity angles can generally be collapsed onto a single curve when plotted against the pressure coefficient difference across the hole. Scatter was reasonable, although greater than experimental repeatability, for all three parameters, with the spread increasing as $dC_p$ became more negative (corresponding to stronger jets and higher incidences on the negative obliquity cases). By highlighting the extreme obliquity cases it identifies trends which may be relevant to the predictive methodology. For $dC_l$ (Figure 8.1) and $dC_d$ (Figure 8.2) it can be seen that the best fit line “under estimates” the results for the $+60^\circ$ obliquity case. This was more significant at lower incidences, or smaller $dC_p$ values. Conversely, the $-60^\circ$ obliquity case was generally “over estimated” by the best fit line, with the best fit line becoming a closer match as $dC_p$ became more negative. This offset is likely due to the $-60^\circ$ obliquity hole increasing the internal flow path compared to other holes, essentially increasing losses within the cavity and weakening the jet for a given $dC_p$. This was supported by analysing surface flow visualisation photographs for each obliquity angle for a similar $dC_p$, which demonstrated the $-60^\circ$ obliquity case had a weaker jet than the other cases. While the $-60^\circ$ obliquity case appeared out of line, data from the straight through hole centred at 35%c (discussed in section 6.3) matched better with the remaining data. This therefore indicates that the offset was not due to the location of the upper hole, but was instead due to the jet path within the cavity. This contributed to the overall scatter witnessed on the plots. However, no such trends applied to obliquity angles less than $-60^\circ$ in this instance.

The best fit curves shown in Figures 8.1 to 8.3 could be used to predict the coefficient increments for any intermediate obliquity case, assuming the hole size and aerofoil remains unchanged. As shown by the three figures, the $\pm 60^\circ$ obliquity cases followed the general trend of the best fit curves, although they tended to bound the upper and lower ranges of the data. Therefore, best fit lines could be constructed using just these two extreme data sets. This has the benefit of considerably reducing the amount of testing or computational
Figure 8.1: $dC_l$ against $dC_p$ scatter plot for all 20%e cases, with a line of best fit through all points shown.

Figure 8.2: $dC_d$ against $dC_p$ scatter plot for all 20%e cases, with a line of best fit through all points shown.

Figure 8.3: $dC_m$ against $dC_p$ scatter plot for all 20%e cases, with a line of best fit through all points shown.
8.5. Adjusting Prediction for Varying Hole Size

It would take a significant amount of wind tunnel testing or computational time to generate an adequate data set to develop Equations 8.6 to 8.8 for all hole sizes likely to be of interest. Therefore, it would be advantageous to apply a ‘scale factor’ or ‘correction’ to a set of data from a different sized hole. Figures 8.7 to 8.9 shows scatter plots of straight through $dC_l$, $dC_d$ and $dC_m$ data against area weighted $dC_p$, from 10%c, 20%c, 30%c and 40%c holes. All holes were centred about half chord and had no obliquity or skew. For the purpose of
Figure 8.4: $dC_l$ against $dC_p$ scatter plot for $\pm 60^\circ$ obliquity cases, with a line of best fit.

Figure 8.5: $dC_d$ against $dC_p$ scatter plot for $\pm 60^\circ$ obliquity cases, with a line of best fit.

Figure 8.6: $dC_m$ against $dC_p$ scatter plot for $\pm 60^\circ$ obliquity cases, with a line of best fit.
8.5. Adjusting Prediction for Varying Hole Size

Hole size variation it is not necessary to consider the effects of varying chordwise location (and therefore varying obliquity), as this would have already been considered in Equations 8.6 to 8.8. Given that experimental testing (section 6.2) had shown $dC_m$ to be generally invariant with hole size, no correction for hole size would be required to Equation 8.8.

Using linear best fit lines in Figures 8.7 and 8.8, combined with a minimal data set, introduced a significant degree of scatter. Further, the 10% c hole did not show a trend in $dC_l$ that could be approximated to a linear best fit, and as a result the best fit line is not shown. The gradients from these lines of best fit were normalised by the gradient for the 20% c hole, and the results of this are shown in Table 8.1. The $dC_l$ gradient for the 10% c hole has been omitted due to the best fit line not providing a good approximation. Equation 8.9 shows a formula for a ratio, $F_{hs}$, which relates the diameter of the hole whose effects are to be predicted ($D_{pred}$) against the diameter of a reference hole, $D_{ref}$.

$$F_{hs} = \frac{D_{pred}}{D_{ref}}$$  \hspace{1cm} (8.9)

It can be seen that the normalised $dC_l$ and $dC_d$ gradients in Table 8.1 for the 30% c and 40% c holes were similar to the $F_{hs}$ diameter ratios. This therefore indicates that the ratio of diameters, $F_{hs}$, could be used to approximate the normalised gradients and therefore predict damage for a different hole size. More complex methods of determining a correction for hole size were attempted but showed little to no improvement in the final predicted coefficient increments.

<table>
<thead>
<tr>
<th>Hole size</th>
<th>$dC_l$ Normalised Gradient</th>
<th>$dC_d$ Normalised Gradient</th>
<th>$F_{hs}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>10% c</td>
<td>N/A</td>
<td>0.36</td>
<td>0.5</td>
</tr>
<tr>
<td>30% c</td>
<td>1.49</td>
<td>1.36</td>
<td>1.5</td>
</tr>
<tr>
<td>40% c</td>
<td>2.13</td>
<td>1.82</td>
<td>2</td>
</tr>
</tbody>
</table>

Table 8.1: Gradients of linear best fit curves from straight through holes of varying sizes normalised against a 20% c straight through hole, with ideal $F_{hs}$ values shown.

By comparing the normalised gradients of the lines of best fits, shown in Table 8.1 with the ideal $F_{hs}$ values the potential accuracy of any prediction can be assessed. It is apparent from the table that, when using a 20% c hole as the reference case, the gradients of $dC_l$ and $dC_d$ are reasonably close to the value of $F_{hs}$, with the exception of the 10% c
Chapter 8. Predicting the Two-Dimensional Effects of an Untested Hole

Figure 8.7: \( dC_l \) against \( dC_p \) for a range of hole sizes centred at half chord with 0° obliquity, with lines of best fit shown.

Figure 8.8: \( dC_d \) against \( dC_p \) for a range of hole sizes centred at half chord with 0° obliquity, with lines of best fit shown.

Figure 8.9: \( dC_m \) against \( dC_p \) for a range of hole sizes centred at half chord with 0° obliquity.
hole. The gradient for the 30%c hole matched $F_{hs}$ very well, with the error increasing to approximately 7% for the 40%c hole.

Because the 10%c hole was generally dominated by a weak jet, whereas all other holes exhibited both weak jets and strong jets, this led to errors when comparing the normalised gradients against $F_{hs}$. The gradient for $dC_l$ was omitted due to the data set not following the same trends, and the $dC_d$ gradient had an error of 28%, compared to an error of up to 10% for the larger holes. Since the 10%c hole never experienced a strong jet its data cannot therefore be accurately scaled from a hole that exhibits a strong jet. Therefore, this is likely to impose a lower limit on the range of permissible hole sizes.

Given that the $dC_l$ and $dC_d$ normalised gradients for 30%c and 40%c holes were reasonably close to the $F_{hs}$ values, it is likely that the proposed $F_{hs}$ scaling factor could provide a reasonable prediction for use when the hole size to be predicted varies from the reference data set. However, it is unlikely to provide an accurate prediction when the new hole is significantly smaller, such that the jet properties are not similar to that of the reference hole (i.e. the hole experiences both a weak jet and a strong jet across the incidence range in question). This therefore means any predictions of a 10%c hole, using the 20%c hole as a datum, are likely to prove inaccurate. Similarly, a divergence in accuracy was seen as the hole size increased. This therefore means that accuracy for 40%c (and larger, if data existed) would be reduced. As a result of this a limitation must exist on this method in that the diameter of the predicted hole must be “close” to that of the hole used to obtain the base data. In this instance, “close” would likely place the 40%c hole on the upper limit of applicable hole sizes. An acceptable range for $F_{hs}$ would therefore be: $0.5 < F_{hs} \leq 2$. Therefore, to predict the effects of a damage hole which is of different size, the $dC_l$ and $dC_d$ values must be multiplied by the scale factor defined in Equation 8.9. No correction for hole size is applied to $dC_m$.

8.6 Review of Accuracy

In order to validate the predictive techniques, virtually all previously tested two-dimensional cases were predicted. The exception were cases using a 5%c hole, as it was felt the results were too close to experimental repeatability to give meaningful evaluations of the method. In addition, as will be discussed in due course, the 10%c hole produced errors due to it operating entirely as a weak jet. All predictions were carried out using best fit curves generated from 20%c diameter holes with ±60° obliquity only. A selection of results are presented here, to demonstrate different conditions, and to highlight any short-fallings in the prediction method. Reference should be made to Figures 4.11 to 4.13 in section
4.8, which show the two-dimensional lift, drag and pitching moment coefficient increment repeatability bands.

Figures 8.10 to 8.12 show $dC_l$, $dC_d$ and $dC_m$ predictions for a 20%c hole with no obliquity or skew. As can be seen in Figure 8.10 a good match between predicted and experimental data were witnessed, up to approximately 4° incidence, with the results under predicted at higher incidences. This was due to the relationship of the best fit line used to predict the data relative to the experimental data. Based on the comparison of the best fit lines with experimental data (Figures 8.4 to 8.6) it would be anticipated that a reasonable match for $dC_l$ would be seen across the weak jet range (low values of $dC_p$), with this degrading as the jet strengthened, leading to an under prediction of $dC_l$ at more negative $dC_p$ values. The results for $dC_d$ (Figure 8.11) showed a consistent offset up to 10° incidence, which resulted in an under prediction. This was due to the previously discussed issue with the -60° data producing a smaller $dC_d$ for a given $dC_p$, which as a result skewed the best fit curve. Due to the small scatter in the $dC_m$ plots (Figure 8.6), the prediction for $dC_m$ (Figure 8.12) produced a good match with experimental data. Lift and drag coefficient increment predictions fell outside of the experimental repeatability bands (section 4.8), with the lift showing the worst error when the strong jet was present. $dC_m$ predictions, however, were generally within the experimental repeatability bands of the original data.

The relationship of experimental data to the $dC_l$ best fit curve used (shown in Figure 8.4) was expected to generate an over prediction of the -60° obliquity case at lower incidences, with the best fit curve generating more negative values of $dC_l$ than the original -60° obliquity data points. The best fit curve suggested the match would improve as $dC_p$ became more negative. This was evident in the results of the $dC_l$ prediction, shown in Figure 8.13. A significant deviation at stall onset (12° incidence) was noticed in all three coefficient increment predictions (Figures 8.13 to 8.15) between actual and predicted results, due to the reduced experimental repeatability and inability to accurately and consistently predict the aerodynamic effects at stall. Drag and pitching moment coefficient increments (Figures 8.14 and 8.15) showed good matches to actual data, given the dominance the -60° obliquity data had over the +60° data in the scatter plots (the -60° data occupied a larger $dC_p$ range than the +60° data on the plots, and thus biased the best fit curve accordingly). This was due to the much larger $dC_p$ range of the -60° obliquity configuration.

Despite +60° obliquity data being used to generate the best fit curves, the match between predicted and actual data for the +60° obliquity case (Figures 8.16 to 8.18) were not as close as for the -60° obliquity case. This was due to the offset between the two reference cases, and the best fit line being skewed more towards the -60° obliquity case. $dC_l$ and $dC_d$ were both under predicted up to approximately 8° incidence, at which point the results were over predicted. This was caused by the jet failing to expand beyond this incidence
Figure 8.10: Prediction of two-dimensional lift coefficient increments for a 20%c straight through hole.

Figure 8.11: Prediction of two-dimensional drag coefficient increments for a 20%c straight through hole.

Figure 8.12: Prediction of two-dimensional pitching moment coefficient increments for a 20%c straight through hole.
and not following the trends of all other damage cases (as discussed in section 6.3).

Considering all predicted 20% cases (straight through, ±30° obliquity, ±60° obliquity and 45° skew), the maximum differences to the original, experimental data, and the RMS of these differences are shown for each increment in Table 8.2 below. It should be noted that the “Maximum difference” values will be skewed by values approaching stall and by the +60° obliquity case above 8° incidence. The RMS values were generally within two to three times of the experimental repeatability.

<table>
<thead>
<tr>
<th>Maximum Δ</th>
<th>$dC_l$</th>
<th>$dC_d$</th>
<th>$dC_m$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.0798</td>
<td>0.0174</td>
<td>0.0233</td>
<td></td>
</tr>
<tr>
<td>RMS</td>
<td>0.0262</td>
<td>0.00495</td>
<td>0.00585</td>
</tr>
<tr>
<td>Experimental Repeatability</td>
<td>±0.015</td>
<td>±0.0013</td>
<td>±0.0020</td>
</tr>
</tbody>
</table>

Table 8.2: Error analysis for 20% two-dimensional predictions. Experimental repeatability from section 4.8.

The accuracy of the 10% straight through hole predictions (Figure 8.19 to 8.21) were varied. $dC_l$ (Figure 8.19) showed a poor match, and did not follow the trends. However, this was due to the experimental data showing trends inconsistent with the larger holes, as shown in Figure 8.7. When the larger holes showed a more negative $dC_l$ as $dC_p$ became more negative, the 10% hole showed comparatively little variation in $dC_l$. $dC_d$ (Figure 8.20) however, produced a good match, despite concerns about the error in the scale factor for the 10% hole. The pitching moment coefficient increment prediction (Figure 8.21) was also of reasonable accuracy, supporting the claim that no correction for hole size was required to $dC_m$.

Figures 8.22 to 8.24 show the predictions for a 40% hole with 45° skew. As can be seen from Figure 8.22, the $dC_l$ prediction was significantly less than the experimental data. Table 8.1 suggested a larger scale factor of 2.13 than the value used of 2 would be required for $dC_l$, indicating a potential 7% under prediction. However, Figure 8.22 shows up to a 30% error. This suggests a larger, non-linear scale factor would be required to correct for hole size. As $dC_d$ (Figure 8.23) also shows an under prediction (albeit not as significant as for $dC_l$), when an over prediction would have been expected it is unlikely that this is the only reason for the errors. If the actual normalised gradient of 2.13 were used instead of 2, it would not have fully removed the errors in $dC_l$ and $dC_d$. The $dC_l$ prediction would have improved slightly, while the $dC_d$ prediction would have worsened. This suggests that $F_{hs}$ is only having a second order effect on the results and is not a primary factor in the predictions. It is therefore likely that the hole size scale factor may not be a linear relationship, and may need to include an additional power term relating to hole size, given the divergence as the hole size has increased. Attempts to identify this term were not successful, although it is important to state that the trends shown by the predictions were
correct. Given that the prediction for $dC_m$ (Figure 8.24) was a very good match, this supports the exclusion of $F_{hs}$ in pitching moment predictions.

An additional factor to correct successfully for large differences in hole size is likely to arise from the different flow properties with the 40% $c$ hole, which generally exhibited a strong jet across all incidences. Combined with the considerably larger hole, these factors are all likely to work against the accuracy of the prediction methodology. Therefore, a limitation would be imposed which restricts the prediction of holes close in size to the reference data. The results presented here suggest the limit could be a range of $\pm 10%c$ on hole diameter for optimum accuracy, although the results have demonstrated that general trends can be predicted for holes twice the size, thus $F_{hs}$ can range between 0.5 and 2.

To summarise, while relying on a reference set of experimental data, the prediction method is reasonably robust and able to cope with changes in chordwise centre, within the bounds of the original data set. Due to the simplified method used for accounting for different hole diameters, accuracy in the prediction was reduced, particularly when there was a large difference between hole size. A note of caution is therefore added to the predictive technique: the hole diameter of the reference data should be as close as possible to the diameter of the hole to be predicted, in order to minimise such errors.
Figure 8.13: Prediction of two-dimensional lift coefficient increments for a 20%c diameter hole with -60° obliquity.

Figure 8.14: Prediction of two-dimensional drag coefficient increments for a 20%c diameter hole with -60° obliquity.

Figure 8.15: Prediction of two-dimensional pitching moment coefficient increments for a 20%c diameter hole with -60° obliquity.
8.6. Review of Accuracy

Figure 8.16: Prediction of two-dimensional lift coefficient increments for a 20\%c diameter hole with +60° obliquity.

Figure 8.17: Prediction of two-dimensional drag coefficient increments for a 20\%c diameter hole with +60° obliquity.

Figure 8.18: Prediction of two-dimensional pitching moment coefficient increments for a 20\%c diameter hole with +60° obliquity.
Figure 8.19: Prediction of two-dimensional lift coefficient increments for a 10\%c diameter hole with no obliquity or skew.

Figure 8.20: Prediction of two-dimensional drag coefficient increments for a 10\%c diameter hole with no obliquity or skew.

Figure 8.21: Prediction of two-dimensional pitching moment coefficient increments for a 10\%c diameter hole with no obliquity or skew.
Figure 8.22: Prediction of two-dimensional lift coefficient increments for a 40\%c diameter hole with 45° skew.

Figure 8.23: Prediction of two-dimensional drag coefficient increments for a 40\%c diameter hole with 45° skew.

Figure 8.24: Prediction of two-dimensional pitching moment coefficient increments for a 40\%c diameter hole with 45° skew.
 Use of Data from an Alternative Aerofoil

From Irwin’s work [1], data were available for straight through holes between 10}%c and 40}%c centred at quarter and half chord. These data were collected from the NACA 641-412 aerofoil, in the same wind tunnel and at the same Reynolds number as the LS(1)-0417MOD aerofoil, however it was limited by pressure data generally only being available for between 0° and 8° incidence. Given that the NACA aerofoil operated over a similar, but slightly larger $C_p$ range to the LS aerofoil, data from the 20}%c holes centred at both quarter and half chord was plotted with the data from the LS aerofoil. Both sets of data had blockage corrections applied to lift, drag and pitching moment coefficient increments, and $dC_p$ values were obtained using the area weighted method.

This section will demonstrate that coefficient increment data from similar aerofoils could be collapsed into a single curve, which allowed for an alternative prediction method to be developed. If no experimental data were available for the aerofoil in question, but it had a similar operating $C_p$ range to the LS(1)-0417MOD and NACA 641-412 aerofoils, the lines of best fit from these graphs could be used to predict the damage case in question, with hole size corrections being applied as required. The accuracy would likely be reduced due to the greater scatter on these plots, but it would allow for a crude approximation.

Figures 8.25 to 8.27 show scatter plots of the NACA and LS coefficient increments, plotted against $dC_p$. A line of best fit going through all the data are also shown. The LS data were taken from 20}%c holes with 0°, ±30° and ±60° obliquity. As can be seen in all three cases, the NACA data produced similar trends to the LS data, although the NACA data were offset from the LS data for all increments below a $dC_p$ of approximately -0.7. There was deviation at the more negative $dC_p$ values (higher incidences), due to the different strong jet performance of the two aerofoils. The LS aerofoil typically formed a strong jet at smaller, less negative $dC_p$ values compared to the NACA aerofoil. The three figures showed that when a single trend line was placed through both the LS and NACA data, there was only significant scatter at the more negative $dC_p$ values. The equations from these best fit curves could therefore be used to predict the aerodynamic effects of a previously untested hole in place of the previously discussed predictive technique. In all cases, and as with the LS scatter plots shown previously, second order polynomial fits were used. These three graphs would also allow for additional experimental data from similar aerofoils to be included, assuming the Reynolds number and operating conditions matched, in order to increase the data set and improve the accuracy of the best fit curves. The resulting equations from Figures 8.25 to 8.27 are given in Equations 8.10 to 8.12.
8.7. Use of Data from an Alternative Aerofoil

\[ dC_{lp} = -6.69002 \times 10^{-3} (dC_p)^2 + 0.12598 (dC_p) \]  \hspace{1cm} (8.10)

\[ dC_{dp} = 0.0157409 (dC_p)^2 - 0.0129068 (dC_p) \]  \hspace{1cm} (8.11)

\[ dC_{lp} = -0.0252005 (dC_p)^2 - 3.27908 \times 10^{-3} (dC_p) \]  \hspace{1cm} (8.12)

The \(-60^\circ\) obliquity case with a 20\%c hole on the LS aerofoil was predicted using the trend lines shown in Figures 8.25 to 8.27. The results are shown in Figures 8.28 to 8.30. The previous prediction using best fit lines from the LS aerofoil with \(\pm 60^\circ\) obliquity are also included for reference. The \(-60^\circ\) obliquity case was chosen as this had the largest \(dC_p\) range, and would thus be utilising the regions of the best fit curves that were subjected to the greatest scatter. A consistent trend across all three graphs was for the magnitude of the coefficient increments to be over predicted at lower incidences, compared to the experimental data and the previous prediction. Above approximately 5\(^{\circ}\) incidence, in all cases, the magnitude of the coefficient increments was then under predicted. The point of this switching corresponded to a \(dC_p\) value of approximately \(-0.8\). As can be seen from Figures 8.25 to 8.27 this is approximately the point at which the NACA and LS data sets began to diverge, as they developed into strong jets. As the best fit line consistently lies below the LS data, this would cause the under prediction. Similarly, a hole on the NACA wing would likely be over predicted. The variation between predicted and experimental results above 10\(^{\circ}\) incidence was consistent with previous predictions and was due to the onset of stall.

The above prediction was also used to predict Irwin’s 20\%c straight through hole centred at half chord on the NACA 641-412 aerofoil [1], with the results shown in Figures 8.31 to 8.33. Surface pressure data were only available between 0\(^{\circ}\) and \(+8^\circ\) incidence, which has limited the scope of the predictions. The trends shown differ to those in Figures 8.25 to 8.27 for the LS aerofoil. All three predicted coefficient increments showed a tendency to over predict, as was suggested by the position of the NACA data relative to the best fit curves. However, with the general trends reasonably well represented, this gives confidence to the methodology. It is not known if the more detailed trends above 8\(^{\circ}\) incidence (i.e. those above the available band of pressure data) would be well predicted by the data, although based on the relationships identified from other predictions, it is likely that as stall approaches the accuracy of the prediction will deteriorate.

If the size of the NACA data set matched that of the LS data set, it is likely that the best fit lines would be better suited to produce more accurate matches. It should be noted that the concept of using coefficient increment data from multiple aerofoils should be reserved only for when it is not possible to obtain data from the aerofoil in question, for use in
Figure 8.25: Scatter graph of $dC_l$ against area weighted $dC_p$ for LS(1)-0417MOD and NACA 641-412 data. NACA data from Irwin [1].

Figure 8.26: Scatter graph of $dC_d$ against area weighted $dC_p$ for LS(1)-0417MOD and NACA 641-412 data. NACA data from Irwin [1].

Figure 8.27: Scatter graph of $dC_m$ against area weighted $dC_p$ for LS(1)-0417MOD and NACA 641-412 data. NACA data from Irwin [1].
the predictive methodology as detailed in earlier sections. This section has illustrated that using a combination of aerofoils with similar pressure distributions can be used as a substitute method for obtaining the reference set of coefficient increments. However, the closer the pressure distributions to that of the aerofoil to be predicted, the more accurate the data is likely to be.
Figure 8.28: Prediction of $dC_l$ for a 20%c hole with -60° obliquity on the LS aerofoil, using best fit curves derived from LS and NACA data.

Figure 8.29: Prediction of $dC_d$ for a 20%c hole with -60° obliquity on the LS aerofoil, using best fit curves derived from LS and NACA data.

Figure 8.30: Prediction of $dC_m$ for a 20%c hole with -60° obliquity on the LS aerofoil, using best fit curves derived from LS and NACA data.
8.7. Use of Data from an Alternative Aerofoil

**Figure 8.31:** Prediction of $dC_l$ for a 20% $c$ straight through hole centred at half chord on the NACA aerofoil, using best fit curves derived from LS and NACA data.

**Figure 8.32:** Prediction of $dC_d$ for a 20% $c$ straight through hole centred at half chord on the NACA aerofoil, using best fit curves derived from LS and NACA data.

**Figure 8.33:** Prediction of $dC_m$ for a 20% $c$ straight through hole centred at half chord on the NACA aerofoil, using best fit curves derived from LS and NACA data.
Chapter 9

Three-Dimensional Predictive Technique

The goal of the three-dimensional predictive technique is to use two-dimensional coefficient increment data and undamaged pressure data for the three-dimensional wing to predict the effects of battle damage on a wing of finite aspect ratio. The two-dimensional coefficient increment data can be obtained either experimentally, by CFD or from the previously-discussed predictive technique.

Work by Render et al, [22], reviewed in section 2.7, provided a method for converting two-dimensional data to three-dimensional wing geometry, which could then be used to predict the aerodynamic effects of battle damage (although this was not proven in the paper). This method was limited in that both two- and three-dimensional wings were of identical chord, and as such the hole areas were identical. When applied to the present study, it was found that Render’s method [22] had to be modified in order to account for the physically larger holes in the three-dimensional wing, resulting from the increased chord size. A refined predictive technique was therefore developed.

The initial data requirements for this prediction are:

- Two-dimensional lift, drag and pitching moment coefficient increments (obtained from wind tunnel testing, CFD calculations or from the predictive technique discussed in the previous chapter).

- Damage hole geometry including chordwise and spanwise locations of the hole on the three-dimensional wing.

- Pressure distributions from an undamaged three-dimensional wing at the spanwise locations of the damage.
9.1 Conversion of Two-Dimensional Data to Three-Dimensional Wing Geometry

When the damage hole area is expressed as a function of the chord, the two-dimensional coefficient increments can be expressed in terms of the three-dimensional model geometry (indicated with a subscript ‘\(3Dc\)’). These can be obtained from using the three formulae shown in Equations 9.1 to 9.3, for a straight, untapered wing. These formulae are valid for models of different spans and chords, provided the damage holes on both wings are of the same diameter, when expressed as a percentage of the chord. They are not the final three-dimensional coefficient increments, this is discussed in Section 9.2. The derivation of Equations 9.1 to 9.3 is shown in Appendix E. In addition, formulae are also derived in Appendix E for unswept tapered wings and swept wings with taper. They are presented in Equations 9.4 to 9.6 and Equations 9.10 to 9.12, respectively, for completeness but are not used or validated in the following chapters.

\[
dC_{L_{3Dc}} = dC_{l_{2D}} \left( \frac{b_{2D}}{c_{2D}} \right) \left( \frac{c_{3D}}{b_{3D}} \right) \tag{9.1}
\]
\[
dC_{D_{3Dc}} = dC_{d_{2D}} \left( \frac{b_{2D}}{c_{2D}} \right) \left( \frac{c_{3D}}{b_{3D}} \right) \tag{9.2}
\]
\[
dC_{M_{3Dc}} = dC_{m_{2D}} \left( \frac{b_{2D}}{c_{2D}} \right) \left( \frac{c_{3D}}{b_{3D}} \right) \tag{9.3}
\]

Where:

- \(dC_{L_{3Dc}}\) = Two-dimensional lift coefficient increment converted to three-dimensional wing geometry
- \(dC_{l_{2D}}\) = Lift coefficient increment from a two-dimensional wing
- \(dC_{D_{3Dc}}\) = Two-dimensional drag coefficient increment converted to three-dimensional wing geometry
- \(dC_{d_{2D}}\) = Drag coefficient increment from a two-dimensional wing
- \(dC_{M_{3Dc}}\) = Two-dimensional pitching moment coefficient increment converted to three-dimensional wing geometry
- \(dC_{m_{2D}}\) = Pitching moment coefficient increment from a two-dimensional wing
- \(b_{2D}\) = Span of two-dimensional wing
- \(c_{2D}\) = Chord of two-dimensional wing
- \(c_{3D}\) = Chord of three-dimensional wing
9.1. Conversion of Data to Three-Dimensional Wing Geometry

- $b_{3D} = \text{Span of three-dimensional wing}$

9.1.1 Conversion Equations for Unswept Wings with Taper

For a three-dimensional wing with taper, but no sweep, the general conversion equations are defined below in Equations 9.4 to 9.6.

\[
dC_{L_{3D}} = dC_{l_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \left( \frac{c_{3D}^2}{c_{2D}^2} \right) \quad (9.4)
\]

\[
dC_{D_{3D}} = dC_{d_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \left( \frac{c_{3D}^2}{c_{2D}^2} \right) \quad (9.5)
\]

\[
dC_{M_{3D}} = dC_{m_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \left( \frac{c_{3D}^2}{c_{2D}^2} \right) \quad (9.6)
\]

All terms are as previously defined, except that $S_{3D}$ is defined as:

\[
S_{3D} = b_{3D} \times \bar{c}_{3D} \quad (9.7)
\]

$\bar{c}_{3D}$ is defined as the geometric mean chord, such that:

\[
\bar{c}_{3D} = \int_0^{\frac{b_{3D}}{2}} c_{3D} \, dy. \quad (9.8)
\]

and $\bar{c}$ is the mean aerodynamic chord, and defined as:

\[
\bar{c} = \frac{2}{S_{3D}} \int_0^{\frac{b_{3D}}{2}} c_{3D}^2 \, dy. \quad (9.9)
\]

9.1.2 Conversion Equations for a General Wing with Sweep and Taper

For a three-dimensional wing with taper and an angle of sweep $\Lambda$, the general conversion equations are as defined in Equations 9.10 to 9.12 below:

\[
dC_{L_{3D}} = dC_{l_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \frac{c_{3D}^2}{c_{2D}^2} \cos^2 \Lambda \quad (9.10)
\]

\[
dC_{D_{3D}} = dC_{d_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \frac{c_{3D}^2}{c_{2D}^2} \cos^2 \Lambda \quad (9.11)
\]

\[
dC_{M_{3D}} = dC_{m_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \left( \frac{c_{3D}^2}{\bar{c}_{3D}} \right) \left( \frac{c_{3D}^2}{c_{2D}^2} \right) \cos^2 \Lambda \quad (9.12)
\]
The value $c'_{3D}$ is defined as the chord normal to the leading edge at the damage location. \( \tilde{c} \) is defined in Equation 9.9.

\section*{9.2 Obtaining Final Coefficient Increments}

In order to obtain the final predicted three-dimensional coefficient increments, it is necessary to plot the converted two-dimensional coefficient increments (obtained from Equations 9.1 to 9.3 for a straight, untapered wing) against the two-dimensional undamaged pressure coefficient differences. For lift, drag and pitching moment coefficient increments, inputting the model geometry used throughout this work, the conversion factor from Equations 9.1 to 9.3 equals 0.75. It should be noted that the two-dimensional experimental data were obtained at a Reynolds number of 500,000 and the three-dimensional experimental data were obtained at a Reynolds number of 1,000,000. The use of differing Reynolds numbers is justified in section 5.8.

It was found that when plotting the converted two-dimensional coefficient increments from Equations 9.1 to 9.3 and the experimental three-dimensional coefficient increments against the corresponding pressure coefficient differences (see section 8.3), the data matched well, removing any spanwise dependency. This is shown in Figures 9.1 to 9.3 for converted lift, pitching moment and drag coefficient increments respectively. The three-dimensional experimental data for the two spanwise locations was plotted against the undamaged pressure coefficient difference across the hole, $dC_P$, at the given spanwise location, and converted two-dimensional coefficient increments were plotted against two-dimensional $dC_P$ values. It should be noted that the area weighting method used for calculating the pressure differences for three-dimensional holes used the experimental pressure data from the finite aspect ratio wing and therefore considered spanwise pressure variation.

The figures show that, when converted to three-dimensional geometry, the two-dimensional case collapsed onto the three-dimensional experimental data at both spanwise locations, taking into account experimental repeatability (see section 5.6). The tip location produced a less accurate match for $dC_L$ than the centre location, but this was still within experimental repeatability bounds. Despite this, the lift and drag coefficient increments collapse well, giving confidence to the prediction method. The converted two-dimensional drag coefficient increments (Figure 9.2) produced values slightly larger in magnitude than the three-dimensional experimental values at low $dC_P$ values (below -0.2, corresponding with the lowest incidences). The pitching moment coefficient increments however, had a significant offset (Figure 9.3). Small errors in the pitching moment would likely have been introduced due to different model mounting configurations between the two- and
9.2. Obtaining Final Coefficient Increments

Figure 9.1: Comparing converted two-dimensional $dC_{L,3D}$ values with actual three-dimensional values at two spanwise locations for a 20%$c$ straight through hole.

Figure 9.2: Comparing converted two-dimensional $dC_{D,3D}$ values with actual three-dimensional values at two spanwise locations for a 20%$c$ straight through hole.

Figure 9.3: Comparing converted two-dimensional $dC_{M,3D}$ values with actual three-dimensional values at two spanwise locations for a 20%$c$ straight through hole.
three-dimensional testing. Comparing the results to those of Render et al [22] show that a similar trend is identified with the pitching moment coefficient increment plots, and was attributed to the low aspect ratio of the three-dimensional model, as discussed in previous chapters.

9.3 Predicting the Three-Dimensional Coefficient Increments

The two-dimensional to three-dimensional predictive technique, discussed in this chapter uses two-dimensional data from wind tunnel tests, as this will provide the most accurate assessment without having to consider errors generated by the two-dimensional predictive methodology. The full prediction method (incorporating both two and three-dimensional predictions) was also applied to complex cases, involving changes in hole size, obliquity and skew angle. This is discussed in detail in Chapter 10.

In order to obtain the final coefficient increments, the variation of $dC_P$ with incidence is required. It should be noted that $dC_P$ is taken at the required spanwise location on the three-dimensional wing. Plots of $dC_P$ against incidence can be interpolated for intermediate incidences as required. These $dC_P$ values are then used to interpolate the converted two-dimensional coefficients ($dC_{L_{3Dc}}$, $dC_{D_{3Dc}}$ and $dC_{M_{3Dc}}$) shown in Figures 9.1 to 9.3 in order to obtain the predicted $dC_L$, $dC_D$ and $dC_M$. The converted two-dimensional data set is extrapolated where necessary. A worked example for the three-dimensional predictive methodology is shown in Appendix F.

To summarise, the methodology for the three-dimensional predictive technique is as follows:

1. Obtain two-dimensional coefficient increment data for the damage case being considered, either from wind tunnel testing, CFD calculations or from the two-dimensional predictive technique (see chapter 8).

2. Convert the two-dimensional coefficient increments to the three-dimensional wing geometry, using equations 9.1 to 9.3.

3. Plot the converted two-dimensional coefficient increments against the area-weighted undamaged two-dimensional pressure differentials, as shown by the “2D Converted” data in Figures 9.1 to 9.3.

4. Calculate the undamaged three-dimensional pressure coefficient differences at the incidences and spanwise locations of the damage to be considered.
5. Interpolate the two-dimensional converted coefficient increments for the three-dimensional $dC_P$ values obtained in the previous step to give the three-dimensional coefficient increments at each incidence.

To demonstrate the accuracy of the prediction method a series of damage cases are presented. The experimental repeatability bands for the three-dimensional data should be noted when making comparisons. These are shown in Figures 5.11 to 5.13 from section 5.6. Reference should also be made to the two-dimensional repeatability bands from section 4.8. Converting the two-dimensional coefficient increment repeatability bands to the three-dimensional geometry, using Equations 9.1 to 9.3 gives:

- $dC_{L3Dc} = \pm 0.0113$
- $dC_{D3Dc} = \pm 0.000975$
- $dC_{M3Dc} = \pm 0.0015$

Figures 9.4 to 9.7 show the experimental and predicted three-dimensional lift coefficient increments for a 40%c straight through hole and 20%c holes with $+60^\circ$ and $-60^\circ$ obliquity and $+45^\circ$ skew respectively. The skew case is located at the tip to demonstrate the use with varying spanwise pressures, and all others are at the central location. The figures generally show a good match between the predictions and experimental results, with the general trend to under predict by a small amount. The 40%c straight through hole (Figure 9.4) was predicted well, which indicated the prediction method was adaptable to different hole sizes.

Results from the other cases were encouraging, with the lift prediction for the $+60^\circ$ obliquity case (Figure 9.5) showing a good match, with a generally consistent offset, across most of the incidence range. The prediction broke down at approximately $+8^\circ$ incidence. This was a trend with all 20%c holes (but was less evident at the tip) and was identified as likely being due to premature separation identified on the two-dimensional model. This resulted from the panels, but due to the larger model was not present on the three-dimensional model. The tip location for the skew case (Figure 9.7) showed a larger under prediction than the central results, likely due to the additional skew effects caused by the proximity of the tip. This was less significant for the central location with a skewed hole, suggesting the presence of the tip vortex was disrupting the jet structure in such a way that could not be accurately modelled by the predictive method, by considering the spanwise pressure variation across the hole only.

In order to understand the divergence at approximately $+6^\circ$ incidence on the $-60^\circ$ obliquity
case (Figure 9.6), surface flow visualisation for two and three-dimensional testing at the same $dC_P$ were compared. The divergence occurred at a $dC_P$ of approximately -0.8, which corresponded to 4° incidence on the two-dimensional model ($dC_p = -0.791$) and 8° incidence on the three-dimensional model ($dC_P = -0.768$). By measuring and scaling the sizes of the horseshoe vortices, it was found that the jet on the three-dimensional model was slightly stronger and wider, despite the smaller $dC_P$. This difference in jet strength may be due to the relatively small working section used for the two-dimensional testing. It is possible that the proximity of the tunnel walls constrained the development of the jet for very strong jets at extreme obliquity angles. This would also account for the divergence above approximately +4° incidence on the 40%c case. No other damage cases showed such an offset.

Figures 9.8 to 9.11 show the predicted and experimental drag coefficient increments. The predictions generally show a reasonable match to the experimental data, particularly below +8° incidence, although most points were within the experimental repeatability bands. There was a general trend for the increments to be over-predicted as the jet strengthened. This was most apparent with the 40%c case (Figure 9.8), although below approximately 5° incidence the match was very good. Surface flow visualisation photographs showed the wake for the 40%c case to be affected more by the spanwise pressure variation, even at the centre location, than the 20%c case, which would account for a reduction in $dC_D$, compared to the two-dimensional data. All other cases showed a reasonably consistent match.

Both obliquity cases (Figures 9.9 and 9.10) showed reasonable matches for $dC_D$ across the incidence range. The negative obliquity case (Figure 9.10) shows a good match across the incidence range, with some deviation in the upper incidence range. This was however consistent with other $dC_D$ predictions. A reasonable match was also seen with the skew case located at the tip (Figure 9.11).

Figures 9.12 to 9.15 show the experimental and predicting pitching moment coefficient increments. A general trend for the pitching moment coefficient increments was to under predict at most incidences, on all four cases. This effect was consistent with findings by Render et al [22], who suggested that this was a result of testing lower aspect ratio wings in half model configuration. Additionally, the differences in the model mounting between the two-dimensional and three-dimensional testing should be taken into consideration. This would help account for the offset.
9.3. Predicting the Three-Dimensional Coefficient Increments

Figure 9.4: Prediction of $dC_L$ for a 40%c straight through hole from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.5: Prediction of $dC_L$ for a 20%c hole with $+60^\circ$ obliquity from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.6: Prediction of $dC_L$ for a 20%c hole with $-60^\circ$ obliquity from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.7: Prediction of $dC_L$ for a 20%c hole with $+45^\circ$ skew from two-dimensional experimental data, compared with three-dimensional experimental data for the tip location.
Figure 9.8: Prediction of $dC_D$ for a 40%c straight through hole from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.9: Prediction of $dC_D$ for a 20%c hole with $+60^\circ$ obliquity from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.10: Prediction of $dC_D$ for a 20%c hole with $-60^\circ$ obliquity from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.11: Prediction of $dC_D$ for a 20%c hole with $+45^\circ$ skew from two-dimensional experimental data, compared with three-dimensional experimental data for the tip location.
9.3. Predicting the Three-Dimensional Coefficient Increments

Figure 9.12: Prediction of $dC_M$ for a 40% straight through hole from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.13: Prediction of $dC_M$ for a 20% hole with +60° obliquity from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.14: Prediction of $dC_M$ for a 20% hole with -60° obliquity from two-dimensional experimental data, compared with three-dimensional experimental data for the centre location.

Figure 9.15: Prediction of $dC_M$ for a 20% hole with +45° skew from two-dimensional experimental data, compared with three-dimensional experimental data for the tip location.
Chapter 10

Validation of the Predictive Technique Using Untested Unique Cases

Two “unique” cases comprising of damage conditions which had not been previously evaluated through two- or three-dimensional wind tunnel testing were tested at the end of the three-dimensional wind tunnel testing programme. These unique cases were used to validate the predictive techniques. The two cases were a 28% chord diameter hole with -50° obliquity (the greatest obliquity achievable with this hole size) and a 28% chord diameter hole with -35° obliquity, +55° skew. For the second case, the axis of the hole centres were moved such that the upper hole was as close to the leading edge as the panel design would permit. As a result the axis of the holes was centred at approximately 43% chord (from the leading edge). Various geometric values relating to the damage cases are summarised in Table 10.1 below.

<table>
<thead>
<tr>
<th>Chordwise Position</th>
<th>28%c, -50° Obliquity</th>
<th>28%c, -35° Obliquity, +55° Skew</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Upper Surface</td>
<td>Lower Surface</td>
</tr>
<tr>
<td>Hole leading edge</td>
<td>27%c</td>
<td>45%c</td>
</tr>
<tr>
<td>Hole centre</td>
<td>41%c</td>
<td>59%c</td>
</tr>
<tr>
<td>Hole trailing edge</td>
<td>55%c</td>
<td>73%c</td>
</tr>
</tbody>
</table>

Table 10.1: Hole geometries for two unique cases, both featuring 28% chord diameter holes.

No two-dimensional testing was carried out on the two unique damage configurations. Therefore it was necessary to use the two-dimensional predictive method (see chapter 8) to obtain the data input required for the three-dimensional predictive technique (Equations 8.6 to 8.8), correcting for hole size (Equation 8.9) and then applying the three-dimensional conversion (see chapter 9 and Equations 9.1 to 9.3). Results are only presented for the central spanwise location. Errors were generally consistent between the two spanwise locations.
10.1 28% Chord Diameter Hole with -50° Obliquity

Figures 10.1 to 10.3 show the predicted coefficient increments compared to the experimental data for the -50° obliquity hole. The lift coefficient increment (Figure 10.1) was predicted with a reasonable degree of accuracy. A trend similar to that of the two-dimensional -60° obliquity prediction was noticed (Figure 8.13), in that the $dC_L$ results were overpredicted up to approximately 6° incidence, beyond which they were under predicted. As discussed in section 9.3, the divergence above 6° incidence was potentially due to the jet from the two-dimensional -60° obliquity case being restricted by the wind tunnel. This would have affected the gradient of the best fit curve and therefore the prediction equations, particularly at the most negative $dC_p$ values, which would typically only be experienced with extreme negative obliquities.

The drag coefficient increment prediction (Figure 10.2) improved as incidence increased. The prediction had a similar and generally constant offset from the experimental data below 6° incidence. This exhibited similar trends to the three-dimensional prediction for a 20%c hole with -60° obliquity (Figure 9.10), which showed an offset in the centre incidence range (between 2° and 8° incidence), and the prediction improving at higher incidences. The offset at lower incidences between the prediction and experimental results had not been witnessed on other predictions previously, and is likely to be a combination of factors relating to the obliquity angle, hole size and conversion to three-dimensional geometry.

The prediction for the pitching moment coefficient increments (Figure 10.3) were the least encouraging. The prediction followed the same basic trend as the experimental data but had a significant offset, giving an under-prediction, which would result in a less nose-down moment than the damage would cause in a wind tunnel test. The offset was however consistent with other results for predicting three-dimensional results from two-dimensional data, as discussed in the previous chapter. The data also follows trends noticed by Render et al [22], in that testing lower aspect ratios in the half model configuration caused an offset in pitching moment. This error can therefore be largely attributed to experimental setup differences.
10.1. 28% Chord Diameter Hole with -50° Obliquity

Figure 10.1: Prediction of $dC_L$ for a 28% hole with -50° obliquity, compared to experimental three-dimensional data, for the centre location.

Figure 10.2: Prediction of $dC_D$ for a 28% hole with -50° obliquity, compared to experimental three-dimensional data, for the centre location.

Figure 10.3: Prediction of $dC_M$ for a 28% hole with -50° obliquity, compared to experimental three-dimensional data, for the centre location.
10.2 28% Chord Diameter Hole with -35° Obliquity, +55° Skew and a Forward Central Axis

The coefficient increments predictions for this case, with -35° obliquity, +55° skew and a forward central axis case are shown in Figures 10.4 to 10.6 respectively. Moving the chordwise centre of the hole resulted in $dC_P$ values that were very similar to the other 28%c case. As a result, the trends for lift, drag and pitching moment coefficients closely matched the simpler case shown in Figures 10.1 to 10.3, with the exception of a greater offset in $dC_D$. The predicted data followed the general trends of the experimental data, giving encouragement to the methodology, and suggesting that the offset in the pitching moment coefficient increment prediction had arisen from differing experimental test configurations, and from limits with lower aspect ratio wings.

10.3 Summary

Generally the predictive technique was robust enough to produce satisfactory results for larger holes, and for complex configurations. This gave confidence to the method for reducing hole locations to the pressure coefficient differences across the hole. In addition, changes in hole size appeared to be reasonably well accounted for, despite the basic method used. The method also highlights a potential issue in the original experimental data, in that the jet for the -60° obliquity case may have been restricted in the two-dimensional wind tunnel. This adversely affected the lift coefficient increment predictions. Despite this, the prediction followed the general trends of the lift and drag coefficient increments reasonably well. The pitching moment coefficient increment prediction had a significant offset, which was attributed to differing model configurations and a low aspect ratio wing being used for three-dimensional testing.

The two predicted cases also showed that, with some limitations, it was possible to predict the effects of a complex damage hole with only minimal wind tunnel testing of “limited” two-dimensional damage cases, in order to provide a suitable range of chordwise locations. With more than two damage configurations from the same aerofoil used to form the base data set, it is therefore likely that the accuracy of the results would improve with the inclusion of more data sets.
Figure 10.4: Prediction of $dC_L$ for a 28% c hole with $-35^\circ$ obliquity, $+55^\circ$ skew, compared to experimental three-dimensional data, for the centre location.

Figure 10.5: Prediction of $dC_D$ for a 28% c hole with $-35^\circ$ obliquity, $+55^\circ$ skew, compared to experimental three-dimensional data, for the centre location.

Figure 10.6: Prediction of $dC_M$ for a 28% c hole with $-35^\circ$ obliquity, $+55^\circ$ skew, compared to experimental three-dimensional data, for the centre location.
The following conclusions could be drawn from two-and three-dimensional experimental testing, and from the predictive technique:

1. General trends for battle damage which were identified in previous work on the NACA 641-412 aerofoil were repeatable on the LS(1)-0417MOD aerofoil. The presence of damage caused a loss of lift, increase in drag and a more nose down pitching moment, when the damage hole is located aft of the aerodynamic centre.

2. Surface flow visualisation identified similar features between the two aerofoils with damage, which corresponded with similarities in trends with the aerodynamic coefficients. The effects of damage fell into typically two categories: a weak jet or a strong jet. Weak jets were generally linked with lower incidences, smaller hole sizes and with holes through the upper surface located towards the trailing edge. Strong jets generally formed at higher incidences, with larger holes and for upper hole locations closer to the leading edge. Weak jets were found to produce small changes in aerodynamic coefficients with small wakes of similar size to the hole, affecting a minimal area of the wing surface. Strong jets typically produced much larger changes in coefficients with wakes several times wider than the hole, affecting a greater region of the aerofoil.

3. The addition of obliquity produced distinct effects. Negative obliquity (upper surface hole forward of the lower surface) resulted in a stronger jet and greater change in aerodynamic coefficients compared to holes with no obliquity, while positive obliquity reduced the effects of damage. The jet strength and damage effects of an oblique hole were driven by the difference in surface pressure coefficients across the area of the holes, between the upper and lower aerofoil surfaces. The effects of negative obliquity were similar to that of an equivalent straight through hole centred at the location of the upper hole. Additionally, changing the obliquity angles was found to significantly change the flow within the cavity, as will be summarised later in this section.
4. Adding skew to the damage on a two-dimensional wing produced a negligible change to the aerodynamic coefficients. This was attributed to the lack of spanwise variation in pressure. However, surface flow visualisation showed that asymmetry had been introduced into the flow. When obliquity and skew were combined, the coefficient increments tended towards those of the equivalent hole with obliquity only.

5. Lift, drag and pitching moment coefficient increments for different obliquity cases could be collapsed and reduced to a single trend line. This was achieved by plotting the coefficient increments against the pressure coefficient differences between upper and lower surfaces, taken across the damage hole. This formed the basis for a method to predict two-dimensional coefficient increments, where these curves were then interpolated for the pressure coefficient differences for the hole to be predicted. The ratios of predicted hole size to the hole size from the base data were used to provide a basic correction for variation in hole size from the base data.

6. It was demonstrated that a limited set of experimental data, consisting of two extreme obliquity cases, could be successfully used to predict the effects of a damage hole. This represents potentially significant cost and time savings by reducing the need for wind tunnel testing and CFD simulations.

7. By plotting coefficient increments against the pressure coefficient difference it was found that data from a different aerofoil followed similar trends to data from the LS(1)-0417MOD aerofoil. This gives rise to the possibility of predicting coefficient increments for similar aerofoils using these data, without the need for experimental testing of reference cases.

8. A technique was defined to convert two-dimensional coefficient increments to a three-dimensional wing of different geometry. Through use of the pressure coefficient differences across the hole at the spanwise location of the damage on the three-dimensional wing and use of the converted two-dimensional coefficient increments, the effects of battle damage could be predicted for a hole on a wing of finite aspect ratio. This allowed the prediction of a unique damage hole to be carried out using the simple two-dimensional prediction method. Results for lift and drag were good, with relatively consistent offsets. More significant errors were present in the predictions for pitching moment coefficient increments, due to the different wind tunnel configurations and model mounting methods used in the two and three-dimensional tests.
9. A complex and previously untested damage hole was used to validate the predictive technique. Lift coefficient increments produced a good match, tending to show a small over-prediction at lower incidences. Drag coefficient increments showing a consistent under prediction, correlating with validations against other configurations. Pitching moment coefficient increments matched the trends of the experimental data, although with an offset present. This offset was consistent with other cases and attributed to experimental testing techniques.

10. In order to better understand the flow characteristics further, surface flow visualisation was used inside the cavity. During positive lift, flow was seen to spread downstream of the lower surface hole into the cavity. Recirculation was noticed, which caused the flow to impact on the upper surface of the cavity. Negative obliquity saw considerable spreading of the flow along the upper surface, with the flow migrating in the opposite direction to the freestream in order to exit the hole. The external flow asymmetry present on skew cases was found to originate from within the cavity, and resulted from the flow exiting the upper surface hole asymmetrically.

11. The internal construction of the wing was seen to restrict the jet development on high positive obliquity angles. When the upper surface hole was placed close to the aft boundary of the cavity, this prevented the airflow from spreading into the cavity and weakened the jet exiting from the upper surface hole.

12. Consistent with previous studies, coefficient increments were found to be affected by the presence of a wing tip on a three-dimensional model. The resulting spanwise pressure variation reduced the magnitude of lift and drag coefficient increments as the hole was moved towards the tip. Little change in the pitching moment coefficient increment was noticed. Asymmetry was also introduced into the flow, generally skewing the jet away from the tip. This increased as the hole moved towards the tip.

13. The trends for positive and negative obliquity from two-dimensional testing were repeated on a three-dimensional wing. However asymmetry was introduced into the flow due to the spanwise variation in pressure. There was little difference between positive and negative skew angles on the three-dimensional wing. However, a variation in the level of asymmetry was noticed between positive and negative skew, with negative skew (upper hole located furthest from the tip) showing less asymmetry than positive skew.

14. When a single damage hole was made through one surface, and the other remained undamaged, this had a significant effect on the aerodynamic coefficients, compared to through damage. For the cases tested this resulted in a significant reduction in the effects of the damage, due to the removal of the through flow.
15. For configurations with different sized holes on the upper and lower surfaces, the results generally tended towards those from an equivalent pair of in-line damage holes of the same diameter as the smaller hole, due to this hole “throttling” the airflow. If the larger hole were on the lower surface, this produced a stronger jet than the opposite case due to the greater mass flow through the damage. When an entire panel was lost on one surface and a damage hole present on the other, the results were similar to those of varying hole diameters. When both panels were removed, there was a substantial loss of lift and increase in drag, which was considerably greater than all other cases tested.
Chapter 12

Future Work and Recommendations

It is recommended that the following areas be investigated further, in future projects, in order to gain a more detailed understanding of the behaviour of battle damaged aerodynamics, and to further refine and improve the predictive technique. Due to time and scope limitations it was unfortunately not possible to achieve these within this project.

1. Undertake two-dimensional testing in the Large Wind Tunnel, used for three-dimensional testing, in order to verify the strongest jets are behaving in the same way. During two-dimensional testing, and as a result of the predictive technique, concerns were raised regarding the largest jets (40%c and 20%c with -60° obliquity cases) being constricted by the proximity of the wind tunnel walls of the low turbulence wind tunnel, but this did not appear to have any significant effect on results. This would be possible in the large wind tunnel due to the larger working section, in relation to the model thickness, which would allow sufficient room for the jet to expand and remain well clear of the tunnel walls. By utilising a vertical model this could also remove any potential effects from friction in the strut pin joints, which may have been affecting the pitching moment results.

2. To verify or disprove any potential effects of gravity on the three-dimensional flow visualisation images. This would require either modifications to the existing three-dimensional model such that it can operate in a vertically-mounted two-dimensional configuration, therefore eliminating any effects due to the presence of wing tips, but still to permit any potential gravitational effects to appear, or to mount a three-dimensional model horizontally in a wind tunnel to determine effects purely related to the wing tip presence. Techniques such as pressure sensitive paint may provide a viable alternative to surface flow visualisation mixture in this scenario.

3. To validate the wind tunnel blockage corrections for a wing with a jet present, using computational fluid dynamics. Wind tunnel corrections to date were only valid for attached flow. With the strongest jets producing a significant separated region on the aerofoil and a large wake and jet downstream of the model it is likely this will increase the resultant blockage. For more accurate blockage corrections, the configurations could be simulated using CFD to determine the corrections required to return the
flow to a free air condition. However, in order for this to be accurately completed, sufficient confidence in the accuracy of CFD studies into battle damage effects would need to be achieved. This would likely require the CFD to be first validated against ‘jets in crossflow’ studies, with the similarities between jets in crossflow and battle damage identified and discussed in the previous chapters.

4. To include multiple holes in close proximity to each other, and missile damage (a large array of smaller holes arranged in a grid pattern) in the predictive technique, in order to widen the scope of the predictive methodology. Irwin [1] had undertaken initial studies on holes in close proximity to each other, but left plenty of scope for expansion in this area.

5. To consider the effects of different cavity sizes on the jet properties, in particular reducing the spanwise extent of the cavities. Previous work had shown that a solid wing produced significantly different results to a hollow wing, and internal flow visualisation undertaken during this project had identified a significant spanwise extent of flow spreading within the cavity. Therefore, restricting the cavity size could potentially have an effect on the coefficient increments.

6. Investigate the effects of battle damage on tapered and swept wings. Little work has been undertaken to date on the effects of battle damage on swept or tapered wings. As these are a more realistic representation of an aircraft wing, it is important to ensure the trends identified in this and previous studies are consistent.

7. Improvements to the two-dimensional predictive method could be made. In particular, an improvement to the prediction of different hole sizes and refinement of the $F_{hs}$ term. This was identified as one of the weaker elements of the two-dimensional predictive technique and with more available time and data (including greater testing of hole sizes other than 20%c, and testing of intermediate hole sizes with obliquity angles) could be improved to more accurately adjust the prediction for larger holes. With more extensive data sets, it may be possible to derive a predictive technique that requires little to no experimental testing or input from wind tunnel/computational data.
8. The predictive technique has only been validated against a single aerofoil. To improve accuracy and robustness, the following additional parameters could be added to testing programmes:

- Include different aerofoils, including those with different pressure distributions.
- Higher speed aerofoils and testing.
- Swept and/or tapered wings.
- The effects of control surface deflection or deployment of high lift devices.

9. Adjust the two and three-dimensional predictive techniques to take into consideration the changing size of the damage wake with different cases. This was identified as a potential cause for error in drag predictions when converting data from a two-dimensional aerofoil to a three-dimensional wing, in particular when damage was located close to the tip and the wake was being skewed significantly by the tip vortex.

10. To use particle image velocimetry (PIV) or similar techniques to develop a greater understanding of the behaviour of the airflow within the jet, and in particular, the flow within the cavity. This would allow velocity profiles to be constructed in order to gain a much clearer understanding of the flow directions and velocities inside the cavity, and to understand further the behaviour of the different oblique jets. This does present significant challenges due to lighting and access restrictions within the cavity, making it difficult to install the required cameras inside the cavity. An alternative may be to construct a transparent model, but this would present the challenge of ensuring there is no optical distortion.
Summary of Irwin’s Methodology for Predicting Damage Effects at Mid Chord

This appendix contains a summary of the method developed by Irwin [1] to predict the effects on lift, drag and pitching moment coefficients from a circular damage hole located at mid chord, on a NACA 641-412 aerofoil. All equations and images in this section are extracted from Irwin’s thesis [1]. The aim was to predict a previously untested hole size on the NACA 641-412 aerofoil.

A.1 Lift Loss Interpolation for Mid Chord Damage

Irwin assumed that the non-dimensional area of the wing surface affected by the damage ($A_{loss}$) is a function of the non-dimensional area of the damage, $A_d$, the average change in pressure differential over the area affected by the damage ($\Delta C_p_{loss}$) and a through flow parameter for the aerofoil section ($\mu$). $A_{loss}$ is given by Equation A.1.

$$A_{loss} = \mu.A_d.\Delta C_p_{loss} = \frac{\mu.\pi (\frac{d}{\tau})^2}{4}.\Delta C_p_{loss}$$

(A.1)

Where:

- $A_{loss}$ = Non-dimensional area of wing surface affected by the hole
- $\mu$ = Aerofoil section property, defined in Equation A.6. Assumed to remain constant for a given damage size, $d$
- $A_d$ = Non-dimensional area of damage
- $\Delta C_p_{loss}$ = Pressure differential reduction due to the hole
- $\frac{d}{\tau}$ = Hole diameter to wing chord ratio
The lift coefficient increment, $dC_l$, can be defined as a function of $A_{loss}$ and $\Delta C_p^{loss}$ when expressed in terms of the lift coefficient increment resolved normal to the chord, $dC_{lN}$. The value of $\Delta C_p^{loss}$ is a function of the average pressure coefficient differential over the area removed by the hole of an undamaged wing, $\Delta C_p^{UD}$, and changes in the pressure field surrounding the hole, $\delta C_p$. The pressure coefficient differential is defined as the difference between the surface pressure coefficients on the upper and lower surfaces, taken at a single point at the chordwise centre of the damage hole. This is shown in Equation A.2.

$$\Delta C_p^{loss} = \Delta C_p^{UD} + \delta C_p$$  \hspace{1cm} (A.2)

Where:

- $\Delta C_p^{loss} =$ Change in pressure coefficient differential due to the hole
- $\Delta C_p^{UD} =$ Undamaged pressure differential coefficient at the damage centre
- $\delta C_p =$ Change in pressure field coefficient surrounding the damage. Initially assumed to remain constant for a given damage size, but was found to vary with damage size and location

Equation A.3 combines Equations A.1 and A.2 to give the lift coefficient increment normal to the chord, $dC_{lN}$. It is assumed that $dC_{lN}$ is zero when the damage diameter is zero (i.e. no change in lift if there is no damage).

$$dC_{lN} = \frac{\mu \pi}{4} \left( \frac{d}{c} \right)^2 (\Delta C_p^{UD} + \delta C_p)^2$$  \hspace{1cm} (A.3)

Where:

- $dC_{lN} =$ Lift coefficient decrement normal to the chord
- $A_{loss} =$ Non-dimensional area of wing surface affected by the hole (Equation A.1)
- $\Delta C_p^{loss} =$ Pressure differential reduction due to the hole (Equation A.3)

Irwin initially assumed $\mu$ and $\delta C_p$ were invariant with hole diameter. Therefore, Equation A.3 can be simplified by extracting constant terms and writing these as a single value, $m_1$, as shown in Equation A.4. This then allows Equation A.3 to be rewritten in the form shown in Equation A.5.

$$m_1 = \frac{1}{2} \left( \frac{d}{c} \right) \sqrt{\mu \pi}$$  \hspace{1cm} (A.4)
A.1. Lift Loss Interpolation for Mid Chord Damage

Where:

- $m_1$ = Constant derived to simplify the lift coefficient increment normal to the chord
- $\mu$ = Constant defined in Equation A.6

\[
\sqrt{dC_{lN}} = m_1 \Delta C_{p UD} + m_1 \delta C_p
\]  \hspace{1cm} (A.5)

Where:

- $\sqrt{dC_{lN}}$ = Square root of the lift coefficient increment normal to the chord, from Equation A.3
- $m_1$ = Constant defined in Equation A.4

In order to obtain the various constants derived above it is necessary to plot a series of graphs from previously-obtained experimental data covering different hole sizes and locations. From previous experiments, a range of $dC_l$ values will be known. These are converted to $dC_{lN}$ values, the square roots taken and plotted against the known undamaged pressure coefficient differential, $\Delta C_{p UD}$, as shown in Figure A.1. It should be noted that different graphs will be required for the positive and negative $\Delta C_{p UD}$ ranges. From these graphs, $m_1$ values are obtained from the gradients of a straight line fit (see Figure A.1), and $\delta C_p$ values obtained from the interception of the line with the x-axis.

In order to obtain $\mu$, the relationship shown in Equation A.6 is used. The parameter, $m_2$, is obtained from the gradient of a graph of $m_1$ values against damage size, shown in Figure A.2. As shown, this graph collapses onto a straight line with minimal scatter.

\[
\mu = \frac{4m_2^2}{\pi}
\]  \hspace{1cm} (A.6)

Where:

- $\mu$ = Aerofoil section property
- $m_2$ = Straight line gradient from a plot of $m_1$ against $\frac{d}{c}$

Despite the original assumption, Irwin found that $\delta C_p$ was not constant for all diameters tested. Therefore, using previous experimental data, $\delta C_p$ can be interpolated for any required damage size. The graph created by Irwin for this purpose is shown in Figure A.3. The previous experimental data shown in Figure A.2 is then used to interpolate $m_1$ for the desired damage size, and $\Delta C_{p UD}$ will be known for the selected aerofoil.
Figure A.1: $\sqrt{\Delta C_l N}$ against pressure differential, for 20% chord damage at quarter chord for the negative $\Delta C_p UD$ range. Figure from Irwin [1].

Figure A.2: Graph of $m_1$ against damage size, to allow for interpolation of $m_1$ for any damage size. Figure from Irwin [1].
A.2 Drag Increase Interpolation for Mid Chord Damage

Irwin’s work showed that the drag coefficient increments from damage typically resulted from surface flow separation, causing an increase in pressure drag, and from pressure drag caused by the through-flow impinging upon the wing internal structure. It was found that neither of these contributions individually showed a direct correlation to the measured drag increments, but the combined effects cause the increments witnessed. From this, Irwin developed an interpolation method that correlated the drag increment, $dC_d$, to the

Once a hole size to be interpolated has been selected the interpolation method reduces to:

1. Interpolate $\delta C_p$ from Figure A.3 for the required damage size.
2. Interpolate $m_1$ from Figure A.2 for the required damage size.
3. $\Delta C_p UD$ is known from existing data.
4. Substitute all values into Equation A.5.
5. Manipulate $\sqrt{dC}_{1N}$ to obtain $dC_l$.

It should be noted that Irwin only validated this methodology against the NACA 641-412 aerofoil. When it was applied to the LS(1)-0417MOD aerofoil used in this project, the method was found to be invalid and could not produce any results.

Figure A.3: $\delta C_p$ change with damage size, for the negative $\Delta C_p UD$ range. Figure from Irwin [1].
lift loss. Using experimental data, the drag increments for 10%c and 30%c diameter holes at quarter chord were plotted against the lift increments, within the negative $dC_l$ range, as shown in Figure A.4.

As can be seen from Figure A.4 the weak jet data points for both damage cases formed a cluster close to the origin. This was a result of the small drag coefficient increments associated with a weak jet. No clear trends could be formed from this cluster as the scatter was too great. However when considering the strong jet points a clear trend forms and, prior to the onset of stall, there is an approximately linear relationship. Figure A.4 also shows that different damage sizes yield a different gradient, with the smaller holes generating a steeper gradient. Figure A.5 shows the straight-line gradients taken from Figure A.4 (primary y-axis) and the $C_d$ offsets (secondary y-axis) plotted against damage size. This graph allows the drag coefficient increment for intermediate damage sizes to be found by interpolating the appropriate gradient and intercept from the graph and using the lift coefficient increment calculated previously for the given damage size. This method requires the lift coefficient increment to be calculated first.
Irwin found that no relationship could be determined for the smallest damage test case, 10%c, but for 20%c to 40%c diameter holes, a linear relationship between pitching moment increment and incidence was found (only valid between -2° and +10°), with all damage sizes producing values of a similar magnitude. Therefore, to allow for interpolation of different damage sizes, Irwin used mean values based on the results from the three larger damage sizes, and assumed a linear variation of the data.
Two-Dimensional Wind Tunnel
Corrections: Equations

The equations used for the two-dimensional wind tunnel corrections, as discussed in section 4.6 are shown below. Unless otherwise stated, all equations are from AGARD’s method [28].

B.1 Lift Interference Corrections

The lift interference corrections comprised of three parts, a correction to the effective incidence of the wing ($\Delta \alpha$, Equation B.1), an increment to the lift coefficient ($\Delta C_l$, Equation B.4) and an increment to the pitching moment coefficient ($\Delta C_m$, Equation B.5). The lift and pitching moment corrections require the kinetic pressure corrections ($G$, Equation B.12) to be applied to yield the final, corrected values.

B.1.1 Incidence Correction

The incidence correction is given by Equation B.1 below:

$$\Delta \alpha = \frac{\pi^2}{96} \left( \frac{c}{H} \right)^2 (2\alpha + D_2) + \frac{\pi^4}{92160} \left( \frac{c}{H} \right)^4 (-2\alpha + 20D_2 - 21D_4)$$  \hspace{1cm} (B.1)

Where:

- $\Delta \alpha$ = Incidence correction, rads
- $c$ = Model chord, m
- $H$ = Wind tunnel working section height, m
- $\alpha$ = Uncorrected model incidence, rads
- $D_2$, $D_4$ = Equation B.2
Appendix B. Two-Dimensional Wind Tunnel Corrections: Equations

The equation for $D_n$, where $n$ is an integer between 1 and 4 is given in Equation B.2. From Equation B.3, the limits of 0 and $\pi$ are equivalent to the leading and trailing edges of the aerofoil. To solve this equation, numerical integration by the trapezium method was used. This divided the aerofoil up into a number of trapeziums of equal length, along the chord. Each value of $x$ has a corresponding value of $y_s$ and $\theta$, from Equation B.3.

However, at the limits of 0 and $\pi$, the denominator of Equation B.2 becomes zero, causing the solution to tend to infinity. Therefore, to allow the equations to be solved, the first and last ‘segments’ of the aerofoil are omitted (when $x = 0$ and $x = c$) when solving by the trapezium rule.

$$D_n = \frac{4}{\pi} \int_0^\pi \frac{y_s}{c} \frac{\sin n\theta}{\sin \theta} d\theta$$

(B.2)

Where:

- $y_s =$ Ordinate of the camber line of the aerofoil at a given chordwise location $x$ (see Equation B.3). $y_s$ should be such that when added to the distance from the chord line to the point of half thickness it yields the vertical location of the upper surface of the aerofoil.
- $c =$ Chord
- $n =$ 1, 2, 3 or 4, as required
- $\theta =$ Equation B.3, below

$$\theta = \cos^{-1}\left(1 - \frac{2x}{c}\right)$$

(B.3)

Where:

- $\theta =$ Angular co-ordinate at a specific point on the aerofoil, rads
- $x =$ A point along the chord, m
- $c =$ Chord, m
B.1. Lift Interference Corrections

B.1.2 Lift Interference Correction

The lift interference correction is given in Equation B.4 below:

$$
\Delta C_l = \left[ -\frac{\pi a_1 C_l}{96} \left( \frac{c}{H} \right)^2 \right] + \left[ \frac{7\pi^3 a_1}{30720} \left( \frac{c}{H} \right)^4 \{3C_l + 2\pi (2\alpha - D_1 + D_2 + D_3)\} \right] \quad \text{(B.4)}
$$

Where:

- $\Delta C_l$ = Lift coefficient increment due to lift interference
- $a_1 = $ Two-dimensional lift curve slope, $(rad)^{-1}$.
- $C_l = $ Uncorrected two-dimensional lift coefficient
- $c = $ Chord, m
- $H = $ Wind tunnel working section height, m
- $\alpha = $ Uncorrected incidence, rads
- $D_1, D_2, D_3 = $ Equation B.2

B.1.3 Pitching Moment Coefficient Correction

The pitching moment correction is given in Equation B.5 below. This is based on the lift interference correction.

$$
\Delta C_m = - \left[ \frac{1}{4} \Delta C_l \right] + \left[ \frac{7\pi^5}{61440} \left( \frac{c}{H} \right)^4 (2\alpha + D_2) \right] \quad \text{(B.5)}
$$

Where:

- $\Delta C_m = $ Pitching moment coefficient increment due to lift interference
- $\Delta C_l = $ Lift coefficient correction, Equation B.4
- $c = $ Chord, m
- $H = $ Wind tunnel working section height, m
- $\alpha = $ Uncorrected incidence, rads
- $D_2 = $ Equation B.2
B.2 Blockage Corrections

The blockage corrections, which are applied to the drag coefficient and kinetic pressure corrections are formed of two parts: the solid blockage corrections, $\varepsilon_s$, and the wake blockage corrections, $\varepsilon_w$. These are combined in Equation B.6 to give the total blockage correction, $\varepsilon_b$.

$$\varepsilon_b = \varepsilon_s + \varepsilon_w$$  \hspace{1cm} (B.6)

B.2.1 Solid Blockage Correction

The solid blockage correction is derived from Goldstein’s work [29], and as with the lift interference corrections take the form of a fourth-power in $\frac{c}{H}$ and an integral along the chord of the aerofoil to determine a camber profile. Equation B.7 gives the solid blockage correction for a model at 0° incidence. Equation B.8 is then applied to take account of the model at incidence. As with the lift interference correction, Equation B.7 is solved using the trapezium numerical integration method, with the first and last “segments” omitted to allow the integration to be solved.

$$\varepsilon_s^{0^\circ} = \frac{\pi A}{6H^2} + \left[ \frac{\pi^3}{960} \left( \frac{c}{H} \right)^4 \int_0^\pi \frac{y_u}{c} \cos \theta \sin \theta d\theta \right]$$  \hspace{1cm} (B.7)

$$\varepsilon_s = \varepsilon_s^{0^\circ} \left[ 1 + 1.1 \left( \frac{c}{t} \right) \alpha^2 \right]$$  \hspace{1cm} (B.8)

Where:

- $\varepsilon_s^{0^\circ}$ = Blockage correction at 0° incidence
- $A$ = Cross sectional area of the aerofoil, m$^2$
- $H$ = Wind tunnel working section height, m
- $c$ = Model chord, m
- $y_u$ = Ordinate of the upper surface of the aerofoil at point $x$ (see Equation B.3), m
- $\theta$ = Equation B.3
- $\varepsilon_s$ = Solid blockage correction taking into account model incidence
- $t$ = Model thickness, m
- $\alpha$ = Model incidence, rads
From the solid blockage, the drag coefficient correction is calculated using Equation B.9.

\[ \Delta C_d = -C_d \varepsilon_s \]  

(B.9)

Where:

- \( \Delta C_d \) = Drag coefficient correction due to blockage
- \( C_d \) = Uncorrected drag coefficient
- \( \varepsilon_s \) = Solid blockage correction

### B.2.2 Wake Blockage Correction

The wake blockage formula is given in Equation B.10 and takes into account the effects of the streamline Mach number on the compressibility of the flow. This is substituted into Equation B.6, along with the solid blockage correction to give the total blockage correction.

\[ \varepsilon_w = \frac{1}{4} \left( \frac{c}{H} \right) \frac{1 + 0.4M^2}{\beta^2} C_d \]

(B.10)

\[ \beta = \sqrt{1 - M^2} \]

(B.11)

Where:

- \( \varepsilon_w \) = Wake blockage correction
- \( c \) = Model chord, m
- \( H \) = Wind tunnel working section height, m
- \( M \) = Free stream Mach number
- \( \beta \) = Compressibility parameter, see Equation B.11
- \( C_d \) = Uncorrected drag coefficient
B.3 Kinetic Pressure Correction

The correction for kinetic pressure (Equation B.12) is applied to the final correction equations (section B.4). This method uses a formula from ESDU [27]. The ESDU equation is a rearrangement of the AGARD method, reducing the terms to a single function.

\[
G = \frac{1}{1 + (2 - M^2) \varepsilon_b} \tag{B.12}
\]

Where:
- \(G\) = Kinetic pressure correction
- \(M\) = Free stream Mach number
- \(\varepsilon_b\) = Total blockage correction

B.4 Final Correction Formulae

The following equations apply the kinetic pressure correction to the corrections determined previously to return the final corrected parameters. A subscript \(f\) indicates the final parameter corrected for freestream; parameters without this subscript refer to uncorrected values. The corrected incidence is given in Equation B.13. The corrected lift, drag and pitching moment coefficients are given in Equations B.14 to B.16 respectively.

\[
\alpha_f = \alpha + \Delta \alpha \tag{B.13}
\]

\[
C_{lf} = G (C_l + \Delta C_l) \tag{B.14}
\]

\[
C_{df} = G (C_d + \Delta C_d) \tag{B.15}
\]

\[
C_{mf} = G (C_m + \Delta C_m) \tag{B.16}
\]
C.1 Lift Interference Corrections

All equations in this section, unless otherwise stated, are from ESDU [33]. It should be noted that the wing span is doubled when using the half model configuration. In addition, values interpolated from the data tables given in the ESDU data sheet are included. In addition, because the model is mounted vertically, the physical tunnel width becomes the height, and the physical tunnel height becomes the width, which is doubled due to the half-model configuration.

C.1.1 Incidence Correction

The first stage calculates the change in incidence due to upwash:

\[ \Delta \bar{\alpha} = \delta_{uv} \frac{S C_L}{C} \]  \hspace{1cm} (C.1)

Where:

- \( \Delta \bar{\alpha} \) = Incidence increment due to lift interference
- \( \delta_{uv} \) = Mean value for the upwash interference parameter over the wing planform. Interpolated as 0.138 from Table 13.2 in ESDU [33].
- \( S \) = Effective wing planform area.
- \( C_L \) = Uncorrected lift coefficient
Appendix C. Three-Dimensional Wind Tunnel Corrections: Equations

- \( C \) = Wind tunnel working section cross sectional area. Note, this should be double the physical area for half model configurations

### C.1.2 Corrections to Lift, Drag and Pitching Moment Coefficients

The corrections to lift, drag and pitching moment coefficients assume \( \Delta \bar{\alpha} \) is small. As a result of this, no lift interference correction is applied to \( C_M \), the pitching moment coefficient. The correction to the lift coefficient is given in Equation C.2 below:

\[
\Delta C_L = -C_D \Delta \alpha_L
\]  

(C.2)

Where:
- \( \Delta C_L \) = Correction to lift coefficient due to lift interference
- \( C_D \) = Uncorrected drag coefficient
- \( \Delta \alpha_L \) = Incidence correction relating to lift, see Equation C.3

\( \Delta \alpha_L \) is defined as:

\[
\Delta \alpha_L = \delta_L \frac{SC_L}{C}
\]  

(C.3)

\( \delta_L \) is the upwash interference parameter related to \( \Delta \alpha_L \). This was interpolated as 0.145 from Table 13.3 in ESDU [33]. Similarly, the correction to the drag coefficient due to lift interference is given as:

\[
\Delta C_D = C_L \Delta \alpha_D
\]  

(C.4)

Where:
- \( \Delta C_D \) = Correction to drag coefficient due to lift interference
- \( C_L \) = Uncorrected lift coefficient
- \( \Delta \alpha_D \) = Incidence correction relating to drag, see Equation C.5

\( \Delta \alpha_D \) is defined as:

\[
\Delta \alpha_D = \delta_D \frac{SC_L}{C}
\]  

(C.5)
δ_D is the upwash interference parameter related to \( \Delta \alpha_D \). This was interpolated as 0.123 from Table 13.3 in ESDU [33].

### C.2 Blockage Corrections

Blockage corrections are applied to lift, drag and pitching moment coefficients. As with the two-dimensional corrections these are broken into two parts, the solid blockage, \( \varepsilon_s \), and the wake blockage, \( \varepsilon_w \). These are combined in Equation C.6 below, in the same form as Equation B.6 before:

\[
\varepsilon_b = \varepsilon_s + \varepsilon_w \tag{C.6}
\]

#### C.2.1 Wake Blockage Correction

The wake blockage correction is similar to the two-dimensional correction, given in Equation B.10, taking into account the ratio between the model span and the wind tunnel working section dimensions. The three-dimensional wake blockage correction is given in Equation C.7 below from AGARD [28], noting the inclusion of zero lift drag. The value chosen for this was taken to be the average drag at zero lift incidence (approximately equal to \(-2.5^\circ\)) for all undamaged panels. The zero lift drag did change for each case with damage present but the effect on the corrections was unnoticeable. Therefore a single value of \( C_{D_0} = 0.0156 \) was selected for all cases.

\[
\varepsilon_w = \frac{1}{4} \left( \frac{S}{b_t H} \right) \frac{1 + 0.4M^2}{\beta^2} C_{D_0} \tag{C.7}
\]

Where:

- \( \varepsilon_w \) = Wake blockage correction
- \( S \) = Model chord
- \( b_t \) = Wind tunnel working section width
- \( H \) = Wind tunnel working section height
- \( M \) = Free stream Mach number
- \( \beta \) = Compressibility parameter, see Equation B.11
- \( C_{D_0} \) = Uncorrected drag coefficient at zero lift incidence. Average value of 0.0156 used
C.2.2 Solid Blockage Correction

The solid blockage correction is obtained from Herriot’s work [34]. This requires obtaining a scale factor based upon the aerofoil profile. Due to the age of the paper, this only contained the older NACA-series aerofoils, and as such the 66-0XX series was chosen, as this provided the closest match from the data provided to the LS(1)-0417MOD aerofoil. Data were interpolated for the 17% thickness required. Similarly, the tunnel shape factor was interpolated. These interpolated values are stated where appropriate. As with the lift interference, all parameters relating to wing span or wind tunnel width should be doubled due to the half model configuration. The solid blockage correction is given as:

\[ \varepsilon_s \left( \frac{1}{\beta^3} \right) \left( \frac{K_1 \tau V_w}{C^2} \right) \]

(C.8)

Where:

- \varepsilon_s = Solid blockage correction
- \beta = Prandtl-Glaurert compressibility parameter, see Equation B.11.
- \(K_1\) = Factor depending on the shape of the aerofoil base profile. Interpolated for a 17% thick NACA 66-0XX aerofoil to 1.049.
- \(\tau\) = Factor dependent on wind tunnel working section shape and wing span to tunnel width ratio. Interpolated to 0.874.
- \(V_w\) = Effective volume of the wing
- \(C\) = Wind tunnel working section cross sectional area

The change in drag caused by the solid blockage, \(\Delta C_D\), is given by Equation C.9, using the same \(C_{D_0}\) value stated previously:

\[ \Delta C_D = - \left( 1 - 0.4M^2 \right) \varepsilon_s C_{D_0} \]

(C.9)
C.3 Final Correction Formulae

The following equations correct the incidence and lift, drag and pitching moment coefficients for lift interference and blockage effects to free air values, indicated with a subscript $f$. Note that as mentioned earlier, pitching moment does not require a correction for lift interference. $M$ refers to the free stream Mach number and $\varepsilon_b$ to the total blockage correction, given in Equation C.6. All formulae are from AGARD [28].

\[ \alpha_f = \alpha + \Delta \bar{\alpha} \quad (C.10) \]

\[ C_{Lf} = C_L \left[ 1 - \left( 2 - M^2 \right) \varepsilon_b \right] + \Delta C_L \quad (C.11) \]

\[ C_{Df} = (\Delta C_{D_b} + \Delta C_D) + C_D \quad (C.12) \]

\[ C_{Mf} = C_M \left[ 1 - \left( 2 - M^2 \right) \varepsilon_b \right] \quad (C.13) \]
A worked example for the two-dimensional predictive techniques, outlined in Chapter 8 is demonstrated for an LS(1)-0417MOD aerofoil with a 30% chord diameter hole with $-30^\circ$ obliquity and $0^\circ$ skew, using the LS(1)-0417MOD aerofoil.

**Data Required:**

- Chordwise locations of the hole to be predicted.
- A reference set of experimental coefficient increment data ($dC_l$, $dC_d$, $dC_m$), comprising of at least two cases covering the incidence range required. The cases should be distributed either side of the chordwise location to be predicted. The diameter of the hole to be predicted should be within $10\%c$ of the reference holes.
- Undamaged pressure distribution for both surfaces of the two-dimensional wing at all incidences of interest.

**D.1 Starting Data: Generating Best Fit Lines**

Two sets of experimental data will be used to form the best fit lines. These will be from 20%c holes with $-60^\circ$ and $+60^\circ$ obliquity. Data are omitted at stall onset, and where the $+60^\circ$ obliquity case begins to exhibit unusual behaviour (above $+6^\circ$ incidence). The data obtained from wind tunnel experiments are summarised in Tables D.1 and D.2 below (note in Table D.2 the higher incidences have been omitted due to the unusual behaviour). ‘Upper’ and ‘Lower’ $C_p$ values are taken across the hole, using the area weighted technique (see Equation 8.4).

Using Equation 8.5 (reproduced below), the area weighted pressure coefficient difference across each hole can be calculated using the upper and lower $C_p$ values provided in Tables D.1 and D.2. These are summarised in Table D.3 for both holes.
Table D.1: Experimental reference for the 20%c -60° obliquity hole.

<table>
<thead>
<tr>
<th>Incidence, °</th>
<th>$dC_l$</th>
<th>$dC_d$</th>
<th>$dC_m$</th>
<th>Upper $C_p$</th>
<th>Lower $C_p$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.34</td>
<td>-0.010</td>
<td>0.0021</td>
<td>-0.0007</td>
<td>-0.555</td>
<td>-0.282</td>
</tr>
<tr>
<td>-0.19</td>
<td>-0.030</td>
<td>0.0048</td>
<td>-0.0021</td>
<td>-0.685</td>
<td>-0.226</td>
</tr>
<tr>
<td>2.03</td>
<td>-0.057</td>
<td>0.0094</td>
<td>-0.0044</td>
<td>-0.818</td>
<td>-0.185</td>
</tr>
<tr>
<td>4.14</td>
<td>-0.092</td>
<td>0.0190</td>
<td>-0.0137</td>
<td>-0.940</td>
<td>-0.140</td>
</tr>
<tr>
<td>6.26</td>
<td>-0.139</td>
<td>0.0307</td>
<td>-0.0268</td>
<td>-1.052</td>
<td>-0.092</td>
</tr>
<tr>
<td>8.47</td>
<td>-0.180</td>
<td>0.0409</td>
<td>-0.0348</td>
<td>-1.145</td>
<td>-0.049</td>
</tr>
<tr>
<td>10.52</td>
<td>-0.190</td>
<td>0.0502</td>
<td>-0.0417</td>
<td>-1.209</td>
<td>-0.017</td>
</tr>
</tbody>
</table>

Table D.2: Experimental reference data for the 20%c +60° obliquity hole. Higher incidences omitted due to unique behaviour.

<table>
<thead>
<tr>
<th>Incidence, °</th>
<th>$dC_l$</th>
<th>$dC_d$</th>
<th>$dC_m$</th>
<th>Upper $C_p$</th>
<th>Lower $C_p$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.26</td>
<td>0.003</td>
<td>0.0017</td>
<td>0.0013</td>
<td>-0.412</td>
<td>-0.499</td>
</tr>
<tr>
<td>-0.16</td>
<td>0.011</td>
<td>0.0019</td>
<td>0.0013</td>
<td>-0.468</td>
<td>-0.406</td>
</tr>
<tr>
<td>2.02</td>
<td>-0.032</td>
<td>0.0037</td>
<td>0.0010</td>
<td>-0.519</td>
<td>-0.297</td>
</tr>
<tr>
<td>4.10</td>
<td>-0.055</td>
<td>0.0063</td>
<td>0.0010</td>
<td>-0.557</td>
<td>-0.198</td>
</tr>
<tr>
<td>6.17</td>
<td>-0.077</td>
<td>0.0087</td>
<td>0.0010</td>
<td>-0.580</td>
<td>-0.105</td>
</tr>
</tbody>
</table>

Scatter plots are generated from both sets of data of the coefficient increments (Tables D.1 and D.2) against the corresponding $dC_p$ values (Table D.3). Best fit lines are inserted into the data, using a second order polynomial. Figures 8.4 to 8.6 are reproduced here, along with the resulting best fit equations (Equations 8.6 to 8.8).

\[
dC_p = (C_p \text{ hole})_{upper} - (C_p \text{ hole})_{lower}
\]

\[
dC_{lp} = -0.0785237 (dC_p)^2 + 0.0679725 (dC_p)
\]

\[
dC_{dp} = 0.0376766 (dC_p)^2 + 3.87867 \times 10^{-3} (dC_p)
\]

\[
dC_{mp} = -0.0450634 (dC_p)^2 - 0.0180391 (dC_p)
\]

Table D.3: Area weighted $dC_p$ values across the damage holes for -60° and +60° obliquity holes.
D.1. Starting Data: Generating Best Fit Lines

Figure D.1: $dC_l$ vs $dC_p$ scatter plot for experimental data from $\pm 60^\circ$ obliquity holes, with a best fit line shown.

Figure D.2: $dC_d$ vs $dC_p$ scatter plot for experimental data from $\pm 60^\circ$ obliquity holes, with a best fit line shown.

Figure D.3: $dC_m$ vs $dC_p$ scatter plot for experimental data from $\pm 60^\circ$ obliquity holes, with a best fit line shown.
D.2 Predicting the Unique Case

D.2.1 Prediction for Chordwise Hole Location

Using the area weighted method, a summary of the $dC_p$ values for the 30%c hole with -30° obliquity, at various incidences, are given below.

<table>
<thead>
<tr>
<th>Incidence, °</th>
<th>Upper $C_p$</th>
<th>Lower $C_p$</th>
<th>$dC_p$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.34</td>
<td>-0.501</td>
<td>-0.404</td>
<td>-0.097</td>
</tr>
<tr>
<td>-0.19</td>
<td>-0.602</td>
<td>-0.333</td>
<td>-0.269</td>
</tr>
<tr>
<td>2.03</td>
<td>-0.705</td>
<td>-0.271</td>
<td>-0.433</td>
</tr>
<tr>
<td>4.14</td>
<td>-0.795</td>
<td>-0.202</td>
<td>-0.592</td>
</tr>
<tr>
<td>6.26</td>
<td>-0.874</td>
<td>-0.135</td>
<td>-0.738</td>
</tr>
<tr>
<td>8.47</td>
<td>-0.930</td>
<td>-0.075</td>
<td>-0.855</td>
</tr>
<tr>
<td>10.52</td>
<td>-0.960</td>
<td>-0.029</td>
<td>-0.931</td>
</tr>
<tr>
<td>12.56</td>
<td>-0.934</td>
<td>0.002</td>
<td>-0.936</td>
</tr>
</tbody>
</table>

Table D.4: Summary of pressure coefficients for a 30%c hole with -30° obliquity.

The values from Table D.4 are then used in Equations 8.6 to 8.8) to generate the predictions for a 20%c hole with the desired obliquity angle. These values have not been corrected for hole size. These values are summarised in Table D.5 below.

<table>
<thead>
<tr>
<th>Incidence, °</th>
<th>$dC_{lp}$</th>
<th>$dC_{dp}$</th>
<th>$dC_{mp}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.34</td>
<td>-0.007</td>
<td>0.0000</td>
<td>0.0013</td>
</tr>
<tr>
<td>-0.19</td>
<td>-0.024</td>
<td>0.0017</td>
<td>0.0016</td>
</tr>
<tr>
<td>2.03</td>
<td>-0.044</td>
<td>0.0054</td>
<td>-0.0006</td>
</tr>
<tr>
<td>4.14</td>
<td>-0.068</td>
<td>0.0109</td>
<td>-0.0051</td>
</tr>
<tr>
<td>6.26</td>
<td>-0.093</td>
<td>0.0176</td>
<td>-0.0112</td>
</tr>
<tr>
<td>8.47</td>
<td>-0.116</td>
<td>0.0242</td>
<td>-0.0175</td>
</tr>
<tr>
<td>10.52</td>
<td>-0.131</td>
<td>0.0290</td>
<td>-0.0222</td>
</tr>
<tr>
<td>12.56</td>
<td>-0.132</td>
<td>0.0293</td>
<td>-0.0225</td>
</tr>
</tbody>
</table>

Table D.5: Predicted lift, drag and pitching moment coefficients for a hole with -30° obliquity, prior to any hole size corrections.
D.2.2 Adjusting Prediction for Variations in Hole Size

As the size of the hole to be predicted is different to that of the reference data, a correction to the data in Table D.5 is required. This utilises Equation 8.9, reproduced below, and used to correct $dC_l$ and $dC_d$ for hole size.

$$F_{hs} = \frac{D_{pred}}{D_{ref}}$$

In this case, $D_{pred}$ is 30\%c and $D_{ref}$ is 20\%c, therefore $F_{hs}$ equals 1.5. Equations 8.1 to 8.3 (reproduced below) are now used with the values from Table D.5 to give the final predicted two-dimensional coefficient increments, shown in Table D.6.

$$dC_{l \text{ dam}} = F_{hs} \times dC_{l p}$$

$$dC_{d \text{ dam}} = F_{hs} \times dC_{d p}$$

$$dC_{m \text{ dam}} = dC_{m p}$$

<table>
<thead>
<tr>
<th>Incidence, $^\circ$</th>
<th>Predicted $dC_l$</th>
<th>Predicted $dC_d$</th>
<th>Predicted $dC_m$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.34</td>
<td>-0.011</td>
<td>0.0000</td>
<td>0.0013</td>
</tr>
<tr>
<td>-0.19</td>
<td>-0.036</td>
<td>0.0025</td>
<td>0.0016</td>
</tr>
<tr>
<td>2.03</td>
<td>-0.066</td>
<td>0.0081</td>
<td>-0.0006</td>
</tr>
<tr>
<td>4.14</td>
<td>-0.102</td>
<td>0.0163</td>
<td>-0.0051</td>
</tr>
<tr>
<td>6.26</td>
<td>-0.139</td>
<td>0.0264</td>
<td>-0.0112</td>
</tr>
<tr>
<td>8.47</td>
<td>-0.173</td>
<td>0.0362</td>
<td>-0.0175</td>
</tr>
<tr>
<td>10.52</td>
<td>-0.197</td>
<td>0.0434</td>
<td>-0.0222</td>
</tr>
<tr>
<td>12.56</td>
<td>-0.199</td>
<td>0.0439</td>
<td>-0.0225</td>
</tr>
</tbody>
</table>

Table D.6: Final predicted lift, drag and pitching moment coefficients for a 30\%c hole with -30$^\circ$ obliquity.
Derivation of Two-Dimensional to Three-Dimensional Wing Area Conversion

E.1 General Wing Planform

Using the drag coefficient increment, $dC_d$ as an example, below is the derivation for converting two-dimensional increments to three-dimensional geometry. The derivation is identical for $dC_l$, but changes for $dC_m$. The derivation for $dC_m$ is shown in Appendix E.4. Throughout the derivation, a subscript $h$ refers to parameters across the hole (using the hole as the reference area, as opposed to the wing).

With reference to a generic three-dimensional wing, with taper and hence chord variation, shown in Figure E.1, a hole is located at a distance $y$ from the root and has a diameter of $Xc'$, where $c'$ is the local chord, defined as the chord normal to the leading edge. The leading edge of the wing is swept back at an angle $\Lambda$. Therefore, the change in drag due

\[ \text{Hole} = Xc' \]

Figure E.1: Location of a hole at $Xc'$ on a general wing planform.
to the hole on a three-dimensional wing, $\Delta D$ is:

$$\Delta D = (C_{Dh}) \left( \frac{1}{2} \rho V^2 \cos^2 \Lambda \right) \frac{\pi}{4} (Xc'_{3D})^2$$  \hspace{1cm} (E.1)

Where:

- $\Delta D = \text{Change in drag due to the presence of the hole}$
- $C_{Dh} = \text{Drag coefficient across the hole, based on hole area}$
- $\rho = \text{Density}$
- $V \cos \Lambda = \text{Velocity normal to the leading edge of the wing}$
- $\Lambda = \text{Angle of sweep}$
- $c'_{3D} = \text{Local chord of the three-dimensional wing, normal to the wing leading edge}$
- $Xc'_{3D} = \text{Diameter of the hole on the three-dimensional wing, where } X \text{ is a constant and is equal on both two- and three-dimensional wings}$

The drag coefficient based on hole area can be determined from two-dimensional testing and is given by:

$$C_{dh} = \frac{dC_{d2D}}{2} \left( \frac{1}{2} \rho V^2 S_{2D} \right) \frac{\pi}{4} (Xc_{2D})^2$$  \hspace{1cm} (E.2)

Where:

- $C_{dh} = \text{Drag coefficient across the damage hole}$
- $dC_{d2D} = \text{Two-dimensional wing drag coefficient increment due to the hole}$
- $S_{2D} = \text{Two-dimensional wing area}$
- $c_{2D} = \text{Chord of the two-dimensional wing}$

Simplifying Equation E.2 gives:

$$C_{dh} = dC_{d2D} \frac{S_{2D}}{\frac{\pi}{4} X^2 c_{2D}^2}$$  \hspace{1cm} (E.3)

Therefore, substituting Equation E.3 into E.1 gives:
\[ \Delta D = dC_{d2D} \frac{S_{2D}}{\frac{\pi}{2} X^2 c_{2D}^2} \left( \frac{1}{2} \rho V^2 \cos^2 \Lambda \right)_{3D} \frac{\pi}{4} X^2 c_{3D} \]

Simplifying, and noting the hole sizes are expressed as the same fraction of a chord \((X)\) on both two- and three-dimensional wings gives:

\[ \Delta D = dC_{d2D} S_{2D} \frac{c_{3D}^2}{c_{2D}^2} \left( \frac{1}{2} \rho V^2 \cos^2 \Lambda \right)_{3D} \tag{E.4} \]

The drag coefficient increment from the hole on a three-dimensional wing, \(dC_{D3D}\) can be expressed as:

\[ dC_{D3D} = \frac{\Delta D}{\left( \frac{1}{2} \rho V^2 S \right)_{3D}} \tag{E.5} \]

Substituting \(\Delta D\) from Equation E.4 into Equation E.5 gives:

\[ dC_{D3D} = dC_{d2D} S_{2D} \frac{c_{3D}^2}{c_{2D}^2} \left( \frac{1}{2} \rho V^2 \cos^2 \Lambda \right)_{3D} \frac{S_{3D}}{S_{2D}} \]

Simplifying:

\[ dC_{D3D} = dC_{d2D} \frac{S_{2D}}{S_{3D}} \frac{c_{3D}^2}{c_{2D}^2} \cos^2 \Lambda \tag{E.6} \]

Where:

\[ S_{2D} = b_{2D} \times c_{2D} \tag{E.7} \]

\[ S_{3D} = \int_{0}^{\frac{b_{3D}}{2}} c_{3D} \, dy. \tag{E.8} \]

- \(dC_{D3D}\) = Drag coefficient increment on a three-dimensional wing
- \(dC_{d2D}\) = Drag coefficient increment on a two-dimensional wing
- \(S_{2D}\) = Two-dimensional wing area, defined in Equation E.7
- \(S_{3D}\) = Three-dimensional wing area, defined in Equation E.8
- \(c_{3D}\) = Three-dimensional wing chord, normal to the leading edge
- \(c_{2D}\) = Two-dimensional wing chord
- \(\Lambda\) = Angle of leading edge sweep
• $b_{3D}$ = Three-dimensional wing span
• $c_{3D}$ = Three-dimensional wing chord
• $b_{2D}$ = Two-dimensional wing span
• $c_{2D}$ = Two-dimensional wing chord

E.2 For a Wing of Constant Chord, Without Sweep

For a straight, untapered wing with constant chord, $c_{3D}$, and no sweep, such that:

• $\Lambda = 0$
• $c'_{3D} \equiv c_{3D}$
• $S_{3D} = b_{3D} \times c_{3D}$

Equation E.6 can therefore be rewritten as:

$$dC_{D_{3D}} = dC_{d_{2D}} \frac{b_{2D}}{b_{3D}} \frac{c_{2D}}{c_{3D}} \frac{c_{3D}^2}{c_{2D}^2}$$

$$\therefore \quad dC_{D_{3D}} = dC_{d_{2D}} \left( \frac{b_{2D}}{c_{2D}} \right) \left( \frac{c_{3D}}{b_{3D}} \right)$$

(E.9)

Equation E.9 simply involves the ratio of aspect ratios of the two- and three-dimensional wings.

E.3 For an Unswept, Tapered Wing

For a wing with taper, but no sweep, such that:

• $\Lambda = 0$
• $S_{3D} = b_{3D} \bar{c}_{3D}$
• $c'_{3D} \equiv c_{3D}$, where $c_{3D}$ is the local wing chord

$\bar{c}$ is defined as the geometric mean chord, such that:

$$\bar{c}_{3D} = \int_0^{b_{3D}} c_{3D} \, dy.$$  (E.10)
In addition, $c_{3D}$ is defined as the local chord at the damage. Therefore, Equation E.6 can be rewritten as:

$$dC_{D_{3D}} = dC_{D_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \left( \frac{c_{3D}^2}{c_{2D}^2} \right)$$ (E.11)

### E.4 Derivation of Conversions for Pitching Moment Coefficient Increments

When considering the change in pitching moment, $\Delta M$, due to damage is:

$$\Delta M = (C_{Mh}) \left( \frac{1}{2} \rho V^2 \cos^2 \Lambda \right) \frac{\pi}{4} (Xc'_{3D})^2 \left( c'_{3D} \right)$$ (E.12)

Where:

- $\Delta M$ = Change in pitching moment due to the presence of a hole
- $C_{Mh}$ = Pitching moment coefficient across the hole, based on hole area
- All other terms as defined in Equation E.1

The two-dimensional pitching moment coefficient based on hole area can be expressed as:

$$C_{mh} = \frac{dC_{m2D} \left( \frac{1}{2} \rho V^2 S_{2D} c \right)_{2D}}{\frac{1}{2} \rho V^2 \frac{\pi}{4} (Xc')_{2D} c}$$ (E.13)

Simplifying Equation E.13 gives:

$$C_{mh} = dC_{m2D} \frac{S_{2D}}{\frac{1}{4} X' c'_{2D}} \frac{c_{2D}}{c_{2D}}$$ (E.14)

Substituting Equation E.14 into E.12 and simplifying gives:

$$\Delta M = dC_{m2D} S_{2D} c_{3D} \left( \frac{1}{2} \rho V^2 \cos^2 \Lambda \right)_{3D}$$ (E.15)

The pitching moment coefficient increment from the hole on a three-dimensional wing, $dC_{M_{3D}}$, can be expressed as:

$$dC_{M_{3D}} = \frac{\Delta M}{\frac{1}{2} \rho V^2 S_{3D} \bar{c}}$$ (E.16)
Where $\bar{c}$ is the mean aerodynamic chord, and defined as:

$$\bar{c} = \frac{2}{S_{3D}} \int_{0}^{b_{3D}} c_{3D}^2 y \, dy. \quad (E.17)$$

Substituting $\Delta M$ from Equation E.15 into Equation E.16 and simplifying gives:

$$dC_{M_{3D}} = dC_{m_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \left( \frac{c_{3D}}{c} \right) \left( \frac{c_{3D}^2}{c_{2D}^2} \right) \cos^2 \Lambda \quad (E.18)$$

This is the general equation for converting two-dimensional pitching moment coefficient increments to three-dimensional values.

For an untapered wing with constant chord and no sweep, as defined in section E.2, it can be said that $\bar{c} = c_{3D}$, therefore Equation E.18 becomes:

$$dC_{M_{3D}} = dC_{m_{2D}} \left( \frac{b_{2D}}{b_{3D}} \right) \left( \frac{c_{3D}}{c_{2D}} \right) \quad (E.19)$$

For a wing with taper, and no sweep, as defined in Section E.3, Equation E.18 becomes:

$$dC_{M_{3D}} = dC_{m_{2D}} \left( \frac{S_{2D}}{S_{3D}} \right) \left( \frac{c_{3D}}{c} \right) \left( \frac{c_{3D}^2}{c_{2D}^2} \right) \quad (E.20)$$
Appendix F

Three-Dimensional Predictive Technique: Worked Example

The three-dimensional predictive technique, as outlined in Chapter 9, will be demonstrated for a 20% chord diameter damage hole with -60° obliquity and 0° skew. The two-dimensional data used for this worked example have been calculated using the prediction method from Chapter 8 and Appendix D (note that the example in Appendix D uses a 30%c hole with -30° obliquity).

Data Required:

- Two-dimensional and three-dimensional wing chord and span lengths.
- Two-dimensional coefficient increment data for the case to be considered (obtained from either experimental data, CFD or the two-dimensional predictive technique).
- Undamaged pressure data at all incidences for the spanwise location to be considered, for both two- and three-dimensional wings.

Step 1: Converting Two-Dimensional Data to Three-Dimensional Wing Geometry

Using Equations 9.1 to 9.3, each coefficient increment from the two-dimensional predictive technique (or from experimental data) is converted to a two-dimensional coefficient increment based on three-dimensional wing area. The equations are reproduced below for convenience.

\[
\begin{align*}
\frac{dC_{L3Dc}}{C_{3D}} &= \frac{dC_{L2D}}{C_{2D}} \left( \frac{b_{2D}}{b_{3D}} \right) \left( \frac{c_{3D}}{c_{2D}} \right) \\
\frac{dC_{D3Dc}}{C_{3D}} &= \frac{dC_{D2D}}{C_{2D}} \left( \frac{b_{2D}}{b_{3D}} \right) \left( \frac{c_{3D}}{c_{2D}} \right) \\
\frac{dC_{M3Dc}}{C_{3D}} &= \frac{dC_{M2D}}{C_{2D}} \left( \frac{b_{2D}}{b_{3D}} \right) \left( \frac{c_{3D}}{c_{2D}} \right)
\end{align*}
\]
The values used were as follows:

- $b_{2D} = 0.45$
- $b_{3D} = 0.975$
- $c_{2D} = 0.2$
- $c_{3D} = 0.325$

This gives a scaling factor of 0.75 for all two-dimensional coefficient increments. Using the predicted two-dimensional coefficient increments shown in Table F.1, the three-dimensional coefficient increments, based on the two-dimensional prediction are as shown in Table F.2.

<table>
<thead>
<tr>
<th>Incidence ($^\circ$)</th>
<th>$dC_l$</th>
<th>$dC_d$</th>
<th>$dC_m$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.34</td>
<td>-0.024</td>
<td>0.0016</td>
<td>0.0017</td>
</tr>
<tr>
<td>-0.19</td>
<td>-0.048</td>
<td>-0.0012</td>
<td>0.0061</td>
</tr>
<tr>
<td>2.03</td>
<td>-0.074</td>
<td>-0.0066</td>
<td>0.0126</td>
</tr>
<tr>
<td>4.14</td>
<td>-0.105</td>
<td>-0.0143</td>
<td>0.0209</td>
</tr>
<tr>
<td>6.26</td>
<td>-0.138</td>
<td>-0.0242</td>
<td>0.0309</td>
</tr>
<tr>
<td>8.47</td>
<td>-0.169</td>
<td>-0.0343</td>
<td>0.0409</td>
</tr>
<tr>
<td>10.52</td>
<td>-0.193</td>
<td>-0.0425</td>
<td>0.0488</td>
</tr>
<tr>
<td>12.56</td>
<td>-0.198</td>
<td>-0.0445</td>
<td>0.0508</td>
</tr>
</tbody>
</table>

**Table F.1:** Predicted two-dimensional coefficient increments for a 20%$c$ hole with -60$^\circ$ obliquity.

<table>
<thead>
<tr>
<th>Incidence ($^\circ$)</th>
<th>$dC_{L_{3Dc}}$</th>
<th>$dC_{D_{3Dc}}$</th>
<th>$dC_{M_{3Dc}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.34</td>
<td>-0.018</td>
<td>0.0013</td>
<td>0.0012</td>
</tr>
<tr>
<td>-0.19</td>
<td>-0.036</td>
<td>0.0046</td>
<td>-0.0009</td>
</tr>
<tr>
<td>2.03</td>
<td>-0.056</td>
<td>0.0094</td>
<td>-0.0050</td>
</tr>
<tr>
<td>4.14</td>
<td>-0.078</td>
<td>0.0157</td>
<td>-0.0108</td>
</tr>
<tr>
<td>6.26</td>
<td>-0.103</td>
<td>0.0232</td>
<td>-0.0181</td>
</tr>
<tr>
<td>8.47</td>
<td>-0.127</td>
<td>0.0307</td>
<td>-0.0257</td>
</tr>
<tr>
<td>10.52</td>
<td>-0.145</td>
<td>0.0366</td>
<td>-0.0319</td>
</tr>
<tr>
<td>12.56</td>
<td>-0.149</td>
<td>0.0381</td>
<td>-0.0334</td>
</tr>
</tbody>
</table>

**Table F.2:** Converted three-dimensional coefficient increments from the predicted two-dimensional results.
Step 2: Interpolating to obtain final coefficient increments

The converted coefficient increments in Table F.2 are then plotted against the undamaged two-dimensional $dC_p$ values for the damage case to be considered (-60° obliquity in this example). This is calculated using the area weighting method discussed earlier, and Equation 8.5. The undamaged pressure coefficient differences are then found, using the same method, for the hole location on the three-dimensional wing at the required spanwise station. Table F.3 shows the corresponding two- and three-dimensional pressure coefficient difference values, for a -60° obliquity hole at the central spanwise location.

<table>
<thead>
<tr>
<th>Incidence (°)</th>
<th>Two-Dimensional $dC_p$</th>
<th>Three-Dimensional $dC_p$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-2.00</td>
<td>-0.27309</td>
<td>-0.26632</td>
</tr>
<tr>
<td>0.16</td>
<td>-0.45835</td>
<td>-0.39837</td>
</tr>
<tr>
<td>2.30</td>
<td>-0.63281</td>
<td>-0.52507</td>
</tr>
<tr>
<td>4.45</td>
<td>-0.79934</td>
<td>-0.64715</td>
</tr>
<tr>
<td>6.62</td>
<td>-0.95995</td>
<td>-0.76843</td>
</tr>
<tr>
<td>8.73</td>
<td>-1.09647</td>
<td>-0.87285</td>
</tr>
<tr>
<td>10.82</td>
<td>-1.1924</td>
<td>-0.96246</td>
</tr>
<tr>
<td>12.97</td>
<td>-1.21495</td>
<td>-1.00432</td>
</tr>
</tbody>
</table>

Table F.3: Two-dimensional and three-dimensional pressure coefficient differences for a -60° obliquity hole at the central spanwise location. Incidence stated is the three-dimensional blockage corrected incidence.

The data from Table F.2 is plotted against the two-dimensional $dC_p$ values from Table F.3. This is shown for $dC_{L_{3Dc}}$, $dC_{D_{3Dc}}$ and $dC_{M_{3Dc}}$ respectively in Figures F.1 to F.3.

In order to allow the three-dimensional coefficient increment values to be interpolated from these figures, the variation of the three-dimensional $dC_P$ (Table F.3) with incidence is required, as shown in Figure F.4 for the -60° obliquity hole at the central spanwise location. The three-dimensional $dC_P$ for any incidence can be interpolated from this figure.

Interpolating Figures F.1 to F.3 for each value of three-dimensional $dC_P$ corresponding to the required incidences (from Figure F.4), and extrapolating where necessary yields the final three-dimensional coefficient increments. These are summarised in Table F.4, and shown as incidence-coefficient increment plots in Figures F.5 to F.7.
Appendix F. Three-Dimensional Predictive Technique: Worked Example

Figure F.1: Converted three-dimensional lift coefficient increments from the two-dimensional prediction for a -60° obliquity hole, against the corresponding two-dimensional $dC_p$ values.

Figure F.2: Converted three-dimensional drag coefficient increments from the two-dimensional prediction for a -60° obliquity hole, against the corresponding two-dimensional $dC_p$ values.

Figure F.3: Converted three-dimensional pitching moment coefficient increments from the two-dimensional prediction for a -60° obliquity hole, against the corresponding two-dimensional $dC_p$ values.
Figure F.4: Variation of three-dimensional $dC_P$ with incidence for a $-60^\circ$ obliquity hole at the central spanwise location.

Table F.4: Final predicted three-dimensional coefficient increments, from estimated two-dimensional data, for a 20\%c hole with $-60^\circ$ obliquity at half span.
Figure F.5: Predicted three-dimensional $dC_L$ from two-dimensional predicted results for a 20\%c hole with -60° obliquity.

Figure F.6: Predicted three-dimensional $dC_D$ from two-dimensional predicted results for a 20\%c hole with -60° obliquity.

Figure F.7: Predicted three-dimensional $dC_M$ from two-dimensional predicted results for a 20\%c hole with -60° obliquity.
List of Publications

The following publications have been released as part of this project.

Conference Papers


Journal Papers
T. W. Pickhaver and P. M. Render. A technique to predict the aerodynamic effects of battle damage on an aircraft’s wing. Submitted to The Aeronautical Journal September 2013.

Restricted access reports released for the BaToLUS programme
T. W. Pickhaver and P. M. Render. Review of the aerodynamic effects of battle damage on aircraft wings. BaToLUS/2/2.5/TR/BAES/001, BaToLUS Battle Damage Tolerance for Lightweight UAV Structures and Loughborough University, January 2010.


References


