Investigation into the aerodynamic effects of simulated battle damage to a wing

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INVESTIGATION INTO THE AERODYNAMIC EFFECTS OF SIMULATED BATTLE DAMAGE TO A WING

by

Andrew J. Irwin, B.Eng

A Doctoral Thesis
Submitted in partial fulfilment of the requirements for the Award of
Doctor of Philosophy of Loughborough University

May 1999

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Sincere thanks go to Mr. P. Stinchcombe and Mr. G. Cunningham for their roles in supporting the practical aspects of the research.

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Synopsis

A key stage in the design-cycle of a military aircraft is the assessment of its vulnerability to hostile threat mechanisms. Such mechanisms inflict battle-damage to the aircraft structure and systems. This experimental investigation considered the aerodynamic consequences of simulated battle-damage to a two-dimensional wing. Key assumptions and techniques were identified leading to the modelling of both gunfire and missile fragmentation damage. Wind tunnel balance measurements were undertaken, together with surface pressure measurements and flow-visualisation methods. Force and moment results indicated extensive changes in coefficient values, whilst both smoke and surface visualisation paint successfully indicated the flow mechanisms present. Using these techniques the influences of damage and experimental variables were investigated, including damage type, size, location and Reynolds Number. Studies were also made into cases of multiple gunfire holes and the influence of internal wing construction.

Results indicated that damage at quarter and half-chord locations gave greater coefficient changes than those seen for either leading or trailing edge damage. This was primarily due to reductions in the upper surface pressure peak due to through-flow. Such reductions were seen to extend in both a chordwise and spanwise direction. The flow mechanism identified indicated both similarities and differences to those of flat-plate jets in crossflows. Analysis of both gunfire and missile damage data lead to the development of a set of empirical relationships, which related damage location and size to coefficient changes.
NOMENCLATURE

a  Fragment damage grid spacing (mm)
a_l  Lift curve slope = \delta C_L / \delta \alpha
A  Cross sectional area of wing section
A_d  Non-dimensional damage area
c  Chord (mm)
C_d  Drag coefficient
C_{df}  Equivalent freestream drag coefficient
C_L  Lift coefficient
C_{lf}  Equivalent freestream lift coefficient
C_m  Pitching moment coefficient (about c/4 location, positive nose up)
C_{mf}  Equivalent freestream pitching moment coefficient
C_N  Normal force coefficient
C_p  Pressure coefficient
d, D  Damage diameter (mm)
dA  Ratio of wing area removed to undamaged area
D_n  Tunnel correction coefficient
dC_N  Change in normal force coefficient
d[C_d]  Change in C_d due to damage (at a given incidence)
d[C_d]'  Calculated change in C_d due to damage (at a given incidence)
d[C_L]  Change in C_L due to damage (at a given incidence)
d[C_L]'  Calculated change in C_L due to damage (at a given incidence)
d[C_L]_N  Change in C_L due to damage, resolved normal to chord
d[C_m]  Change in C_m due to damage (at a given incidence)
d[C_m]'  Calculated change in C_m due to damage (at a given incidence)
d(C_p_{upper})  Change in upper-surface pressure coefficient
d[\Delta C_p]  Change in pressure coefficient differential
G  Kinetic pressure correction factor
h  Tunnel height (mm)
LE  Leading edge
m_1, m_2  Lift loss factors
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>$p_0$</td>
<td>Static pressure of undisturbed tunnel flow (N/m²)</td>
</tr>
<tr>
<td>$p_t$</td>
<td>Local surface tapping static pressure (N/m²)</td>
</tr>
<tr>
<td>$r$</td>
<td>Damage hole radius (mm)</td>
</tr>
<tr>
<td>$S_{TE \text{ region}}$</td>
<td>Area of TE region considered (mm²)</td>
</tr>
<tr>
<td>$S_{UD}$</td>
<td>Undamaged wing area (mm²)</td>
</tr>
<tr>
<td>$t$</td>
<td>Aerofoil thickness (mm)</td>
</tr>
<tr>
<td>TE</td>
<td>Trailing edge</td>
</tr>
<tr>
<td>$V$</td>
<td>Tunnel velocity (m/s)</td>
</tr>
<tr>
<td>$V_{\text{freestream}}$</td>
<td>Crossflow freestream velocity (m/s)</td>
</tr>
<tr>
<td>$V_{\text{jet}}$</td>
<td>Crossflow jet velocity (m/s)</td>
</tr>
<tr>
<td>$x$</td>
<td>Distance across flat-plate surface (parallel to freestream)</td>
</tr>
<tr>
<td>$y$</td>
<td>Distance across flat-plate surface (perpendicular to freestream)</td>
</tr>
<tr>
<td>$z$</td>
<td>Distance above flat-plate surface</td>
</tr>
<tr>
<td>$Z_c$</td>
<td>Ordinate of camber line</td>
</tr>
<tr>
<td>$Z_t$</td>
<td>Aerofoil thickness</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Incidence (degrees)</td>
</tr>
<tr>
<td>$\alpha_f$</td>
<td>Equivalent freestream incidence (degrees)</td>
</tr>
<tr>
<td>$\Delta \alpha$</td>
<td>Incidence correction (degrees)</td>
</tr>
<tr>
<td>$\Delta C_L$</td>
<td>Lift coefficient correction</td>
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<tr>
<td>$\Delta C_m$</td>
<td>Pitching moment coefficient correction</td>
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<td>Pressure coefficient differential loss</td>
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<tr>
<td>$\epsilon_B$</td>
<td>Total blockage correction factor</td>
</tr>
<tr>
<td>$\epsilon_S$</td>
<td>Solid blockage correction factor</td>
</tr>
<tr>
<td>$\epsilon_{SP^0}$</td>
<td>Solid blockage correction factor (at 0° incidence)</td>
</tr>
<tr>
<td>$\epsilon_w$</td>
<td>Wake blockage correction factor</td>
</tr>
<tr>
<td>$\mu$</td>
<td>Lift loss factor</td>
</tr>
<tr>
<td>$\theta$</td>
<td>Grid rotational angle (degrees)</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Air density (kg/m³)</td>
</tr>
<tr>
<td>$\rho_{fr}$</td>
<td>Fragment density (holes per m²)</td>
</tr>
</tbody>
</table>
CHAPTER 1. INTRODUCTION

1.1 Background

This chapter provides a general introduction to the topic of Battle Damage and its assessment. More specifically, it outlines the need to consider the effects of anticipated levels of battle damage in the design process of military aircraft. In addition to a brief introduction to the history of battle damage assessment, the present increased requirement to develop accurate assessment methodologies is explained. The characteristics of relevant threat mechanisms are outlined together with their associated damage effects. The chapter closes by describing the aim of the work contained within this thesis; to contribute to the subject of aerodynamic assessment of battle damaged wings.

1.2 The role of Battle Damage Survivability Assessments

Given the function of military aircraft, their design and manufacture must consider not only their ability to perform a given task, but also take account of the environment in which they undertake it. It must be assumed that such aircraft will operate in hostile environments, i.e. with various types of anti-aircraft threats in operation. To assume the contrary would be to misunderstand the basic function of military aircraft.

As a result of being placed in a hostile environment, the aircraft must be expected to sustain some level of battle damage. As a consequence of this, one of the key design requirements of any military aircraft is Combat Survivability. This is defined as “the capability of an aircraft to avoid and/or withstand a man-made hostile environment” (Ref.1).

Survivability can be reduced to two key considerations. The first, susceptibility, is the ability to “avoid” the hostile threats (such as radar, gunfire, guided missiles, etc.) and can be considered to be a function of both aircraft design and operational tactics employed. Reduced radar-cross section and increased manoeuvrability are functions of design, whilst terrain masking to avoid detection or suppression of opposing air-defences are operational tactics.
The aircraft's susceptibility is measured as a probability of the aircraft being hit by a threat mechanism.

However, vulnerability is the ability of an aircraft when hit, to "withstand" the damage caused by the threat mechanism, i.e. the more vulnerable an aircraft is, the more likely it will be killed when hit. Vulnerability is a function of both threat mechanism and aircraft design. Thus for a given mechanism, aircraft vulnerability is determined by survivability features incorporated into the design which reduce the extent of the effects of damage. Vulnerability is measured by the conditional probability of an aircraft kill, once it has been hit.

During the design process, survivability enhancement techniques ensure that the final aircraft design is one that can successfully withstand specified hostile-environment levels without undue weight, cost or performance penalties. Survivability assessments are undertaken as part of this iterative process in order to successfully achieve a considered balance.

It is vital that survivability assessments use methodologies that accurately assess the implications of the anticipated threats. As part of this approach, vulnerability assessments consider the aircraft in terms of both its systems and structure. Critical components are those which if destroyed or damaged would lead to an aircraft loss, whilst non-critical components would lead to differing levels of aircraft incapacitation. Each individual component is assessed in terms of functionality against level of damage. When all the components (both system and structure) are combined together an overall appraisal of the total aircraft's vulnerability is obtained.

However, consideration of the aircraft structure in such vulnerability assessments has tended to concentrate mainly on structural integrity, equipment hardening, shielding vital components and the like. It appears, and will be shown later, that few detailed investigations have been undertaken into the aerodynamic effects of damage to the structure, in particular, damage to lifting surfaces. Given that aerodynamic integrity is of paramount importance to the continued functioning of any aircraft, quantifying such effects and understanding the mechanism involved must be a key requirement for a complete vulnerability assessment. Basic factors include the extent and characteristics of lift loss and pitching moment changes.
Whilst drag increments could lead to significant reductions in aircraft range, thereby preventing a successful return to base.

It must be remembered that although some, if not most, fixed-wing military aircraft may operate at high-speed (>Mach1), all must pass through the low-speed aerodynamic range, if only to land! Hence the low-speed investigations undertaken here.

1.3 Historical Considerations of Battle Damage Effects

The first steps towards improving aircraft survivability came with one of the earliest deployments of aircraft into a hostile environment, namely World War I. The pilots involved quickly identified the need for increased self-preservation and took to wearing steel infantry helmets, and placing steel-plates beneath their seats. There was an immediate benefit to the pilots in taking this action, and it quickly became apparent that the design of the aircraft had failed outright to consider how easily the pilot could be injured when flying the aircraft. During the course of the conflict, attempts were made to correct this critical weakness in the design. Methods of improving the survivability of both the crew and aircraft were identified and implemented, progressing from all-steel seats to nickel-chrome steel armour plating around not only the crew, but newly identified critical components such as the fuel tanks, radiators and engines.

During World War II the greatest threats to aircrews and aircraft came from anti-aircraft shells. It was not expected that the shell would hit the aircraft directly, rather the process relied on the penetrative effects of the shell fragments when it burst in close proximity to the aircraft. Given the lightweight method of aircraft construction at the time, protection against fragmentation was in the form of body-armour to protect the crew. Unfortunately, in the early stages of the conflict, the body-armour issued was very heavy and uncomfortable to wear, so rarely used. This resulted in high aircrew casualty rates. The redesign of body protection lead to the development of less awkward ‘flak-jackets’ which reduced the casualty rate by 65%. Also at this time, shielding of vital aircraft components was incorporated as part of the design process, and the development of self-sealing fuel tanks played a significant role in reducing aircraft losses. Attention was also paid to the suppression of fuel fires; voids surrounding fuel
tanks were filled with Balsa wood to reduce fuel leakage, and early attempts were made to fit internal fire extinguishers to fuel tanks.

Considerable emphasis was placed on researching specific problems of aircraft vulnerability and protection during World War II, with particular attention being directed towards structural effects associated with penetration of the aircraft by bullets and shell fragments.

The U.S. Army Ballistic Research Laboratory held the First Working Conference on Aircraft Vulnerability in 1948, involving representatives from the U.S. Air Force, ordnance companies and various research establishments. Meetings were held to discuss and define the problems associated with the vulnerability of military aircraft of the time, and identify improvements and methodologies that could be adopted into future aircraft designs. However, despite a very good start, the activities of the group were soon abandoned when it was decided that "...all future wars would be fought with nuclear weapons." (Ref. 2). This is a very important statement as it was the first clear indication of a perception that would be held true by amongst others, military strategists and equipment manufacturers, for many years. The consequence of this perception was the development of a philosophy, which in turn became one of the key driving forces of the ensuing arms race, of "Nuclear deterrence preventing aggression". Unfortunately, this philosophy resulted in the decline of research into the topic of combat survivability in the early 1950's.

In general, from the 1950's onwards, work on combat survivability has tended to concentrate on specific problems highlighted by specific conventional conflicts.

In the late 1950's, the U.S. Air Vehicle Environmental Research Team was formed by the U.S. Army, to consider the likely threats to aircraft operating in direct support of forward area units. The team developed the first concepts for protection against ballistic threats, which were then adopted in all U.S. Army combat aircraft. The USAF and Navy also adopted some of the concepts developed, which included new lightweight armour materials, damage tolerant components and further advances in fuel protection.
During the Korean conflict, interest in survivability was revived again. The resulting advancements were particularly in the application of steel armour-plating, self-sealing fuel tanks, and bullet-resistant glass. However, following this conflict non-nuclear considerations were again overshadowed by an emphasis on design directed towards nuclear war considerations.

It was not until the use of large numbers of U.S. aircraft in Vietnam, in the mid 1960's, that it became apparent once again that they were to be used in a non-nuclear hostile environment. Fighter aircraft, and more especially helicopters, were used for the first time in combat roles where exposure to enemy gunfire was commonplace. The ensuing large numbers of rotary-wing aircraft shot down or critically damaged by small-calibre weapons provided the motivation for the U.S. Army to conduct new survivability research. In addition to the U.S. Army, both the US Airforce and Navy also experienced unacceptable aircraft losses, and embarked on separate programmes to analyse the problems, and develop methods of improving survivability. This included the use of foam inside fuel cells to prevent fires, and light-weight ceramic composite armours to provide protection against armour-piercing projectiles. Active counter-measures and other electronic warfare devices were also developed and used in conjunction with aural and infrared emission suppression.

With each of the three U.S. services undertaking separate survivability programmes, an uncoordinated approach towards improving survivability developed. The need for an integrated effort to co-ordinate and standardise the work was recognised, and it was to this end that the U.S. Joint Technical Co-ordinating Group on Aircraft Survivability (JTCG/AS) was established in 1971. Over the past 25 years, JTCG/AS has published a number of documents relating primarily to the definition of survivability design requirements, guidelines for obtaining improved survivability, and developing assessment methodologies (including vulnerability assessments). However, even this major source of data concentrated little resources on research into the aerodynamic consequences of battle damage.
1.4 Threat Characteristics

The following section gives a clear introduction to the physical causes and effects of battle damage to aircraft. Consideration is given to the different types of threat mechanisms which may be encountered by aircraft in hostile environments, as well as the characteristics of the resulting damage processes.

1.4.1 Threat Types

There are two main conventional threat mechanisms considered to be of primary significance (Refs. 3, 4); projectiles and guided missiles. Both mechanisms are specifically designed to cause damage to aircraft structural components, including the wings.

Additionally high-intensity radiation is also considered as an threat mechanism to aircraft, but the type of internal electrical damage caused is relevant to aircraft systems rather than structure, and will not be considered further.

a) Projectiles

A projectile is an object propelled by an initial impulse and continuing in motion due to its own inertia. No guidance capabilities are present following the initial impulse trajectory, and the object will fall under gravity. Typically a bullet or artillery shell will be described as a projectile.

Small Arms are projectiles with diameters less than 20mm in diameter, and are usually one of the following types;

(i) Ball-type projectiles have a relatively soft-metal core designed to flatten on impact and give a larger damage area.

(ii) Armour-Piercing (AP) consists of a penetrating hardened-metal core encased by a softer metal sleeve (required to prevent rapid gun-bore attrition). Additionally the round may also contain incendiary (I) material designed to ignite combustible materials within the target. Figure 1.1 illustrates a typical AP-I projectile.
Anti-Aircraft Artillery (AAA) is used to describe projectiles of 20mm diameter and greater. Such AAA threats usually consists of;

(i) Armour Piercing shells which may include incendiary material, as in the case of small arms rounds.

(ii) 23mm High-Explosive (HE) rounds are considered to be the most common form of anti-aircraft artillery (Ref.3). HE rounds have a high-explosive charge within a metal casing, and typically have either tracer or incendiary components (Figure 1.2). A fuse is used to control the point of detonation, and may be of the ‘super-quick’ or ‘delayed’ contact type. Super-quick fuses cause detonation on immediate contact with the target, i.e. just external to the structure. Delayed fuses allows detonation as the shell passes through the aircraft structure, thus causing greater internal damage.

Projectiles may be fired by many different types and configurations of guns available. The threat may range from a single-round handgun, through aircraft mounted cannon, to highly lethal mobile gun platforms such as the Russian ‘ZSU-23’ gun-system, capable of firing up to 2000 rounds/min. using radar targeting.
b) Guided Missiles

A guided missile is defined as a self propelled aerospace vehicle with guidance capabilities, designed to inflict damage by warhead detonation. Two distinct types of guided missiles, namely, Air-to-Air (AAM) and Surface-to-Air (SAM) Missiles, threaten aircraft. AAMs are launched from an attacking aircraft and attempt to home in on the target. As the AAM is carried on an aircraft its mass is kept as low as possible, which results in a relatively small warhead. However, SAMs are launched from either land or sea based platforms and so are very much less constrained by weight. This allows more complex onboard sensor systems and larger warheads, which are more likely to kill the aircraft than produce survivable damage.

There are four main types of missile warhead;

(i) **Blast warheads** are designed to cause damage by a massive High-Explosive (H-E) over-pressure on the structure of the target. However the range for such a warhead is relatively small. This is due to the rapid reduction in pressure pulse with distance from warhead to target (a cubic dissipation).

(ii) **Fragmentation** is the most common type of warhead, and will be discussed later.

(iii) **Continuous Rod** warheads consist of a series of metal rods joined together, in turn, at each end. When detonated, the blast causes the rod bundle to expand outwards, creating an expanding jagged ring. The expanding ring is designed to ‘cut’ through the aircraft structure and so cause significant damage.

(iv) **Shaped Charge** warheads generate focused beams of fragments. By controlling the casing and explosive geometry it is possible to focus the direction of the fragment spray.

1.4.2 Damage Processes

As indicated, conventional threat mechanisms are penetrators, fragments, incendiary particles, blast and cutting mechanisms. This project is restricted to the two primary methods of inflicting aircraft damage; penetration and fragmentation effects. These were chosen because they are both are common forms of survivable damage. The consequences of
incendiary, blast and cutting mechanisms are usually serious fire and structural damage, which are likely to result in an immediate aircraft kill, and so are of no interest here.

a) Penetration
Penetration is defined as the damaging effect of a projectile, i.e. small arms or AAA round. Although missile fragments also enter the aircraft structure their damage mechanism is different and will be considered separately.

The purpose of the penetrator is to pass through the aircraft causing the maximum internal damage possible. Each projectile generates a single damage path through the aircraft, with both entry and exit holes. The size of the holes generated by the projectile are usually far greater than that of the projectile itself. An armour-piercing round will result in a relatively small hole, whilst a contact fused High-Explosive (H-E) round will produce a significantly larger hole.

In addition, if the projectile passes through a fluid filled cavity such as a wing fuel tank, then hydraulic ram occurs. Hydraulic ram is when kinetic energy from the projectile is transferred into the fluid and generates a large pressure wave. This pressure wave passes through the fluid and creates an impulsive load on the tank panels. Large-scale damage usually results and in some instances, whole panels may be ‘blown-out’.

b) Fragmentation Effects
Fragmentation effects result from a combination of H-E charge and weakened metal casing. This is designed to generate metal fragments of a specific size, which are then dispersed in a pre-defined spray pattern, optimised for maximum effect. The desired fragment dimensions can be obtained by scoring the metal casing or wrapping it with wire (Figure 1.3). Some designs use cases composed of pre-formed metallic cubes or balls encased in an epoxy matrix.
With the detonation of the High-Explosive charge, the warhead bursts. The fragments are then ejected in an approximately uniform distribution around the missile axis, resulting in a divergent cylindrical spray pattern (Figure 1.4). As the fragments are incident on the structure of the aircraft, the damage is spread over a wide area of the aircraft skin. The damage processes and terminal effects for each individual fragment are similar to those associated with a projectile, i.e. ballistic impact followed by penetration, with possible hydraulic ram.
Fatal missile damage usually results from a burst close to the aircraft. The high density of fragments penetrating the structure will invariably lead to significant structural and system damage, and thus an aircraft kill. However, if the fragment density is low due to a large burst distance, or a limited, non-critical region of the aircraft lies within the fragment beam, then the damage is more likely to be survivable. For example, the fragment beam may only intercept part of a wing. This would be unlikely to result in the loss of wing structural integrity. However, with no method to assess the aerodynamic effects, it would not be known if the wing would continue to provide adequate lift, with acceptable drag and pitching moment.

Unclassified illustrations of missile damage are rare. However, Figure 1.5 illustrates such damage to a civil aircraft’s wing control surface. The distribution of the relatively small holes produced can clearly be seen.

![Figure 1.5 Survivable Damage to a BAe 125 (Stbd flap)](image)

1.5 Literature Review

The aerodynamic research listed here is believed to be the most comprehensive possible. The publications identified were all based on US military funded research programmes undertaken since the late 1960’s.
The NASA research centre at Langley undertook the first research programme beginning in 1967 and continuing until 1973. The investigations were all high-speed wind-tunnel tests and were reported in five reports (Refs. 5, 6, 7, 8, 9). The tests involved generic swept-wing full aircraft models, to which large-scale damage was simulated by removing all or part of the wing, horizontal tail or vertical fin. In addition to the basic lift, drag and pitching moment coefficient data produced, some limited control effectiveness data were also obtained. The investigations were undertaken over a range of incidence and side-slip angles, with no attempts being made to trim the models. The investigations were intended to serve as an aid in determining the probability of kill for aircraft. However, the simulated damage was very simplistic and not very representative. Typical damage was the removal of the outer 30% to 50% of one wing, or the partial/full removal of the fin or horizontal tail. No attempts were made at testing more detailed and realistic damage, although a large number of aircraft configurations / Mach numbers were tested (ranging from Mach 1.41 to 4.63). Tests were undertaken at such high Mach numbers as a result of the US interest in high-speed combat aircraft at that time.

The reports presented the numerical data obtained and did little in the way of analysis. No flow visualisation techniques were employed, and no explanation of the flow mechanics were offered. The conclusions presented were drawn directly from the data obtained, and were very superficial, "... removing the leading- or trailing-edge portion of one wing panel, or removing the entire wing panel, lead to a decrease in both lift-curve slope and maximum lift-drag ratio." (Ref. 5).

There was then a nine year gap before any further work was undertaken on the subject at NASA Langley. However, in this second programme the test runs were again simplistic high-speed tests (Mach 1.57) of generic aircraft models with various percentages of the outer wing and tail removed. The conclusions claimed that "...major damage to the wing, up to the point of the complete removal of one wing and major damage to the horizontal tail may be sustained without necessarily causing the loss of the aeroplane or pilot." (Ref. 10). An interesting claim, which no doubt, would need further verification in the low speed region. However, this did provide the first insight into the relative ‘robustness’ of the wing in generating lift, in that significant amounts of simulated damage may be sustained by the
lifting surfaces before they prove to be totally ineffective.

During the period between the Langley research, work was undertaken elsewhere on providing methods of predicting drag increments for damaged aircraft (Ref. 11). Data of this type were deemed valuable in calculating the maximum range of an aircraft with structural damage resulting from a nuclear blast. Given that such an aircraft might sustain only minimal structural damage, this study attempted to quantify mission-essential aerodynamic characteristics. There were two basic types of damage modelled; (i) increased skin friction drag as a result of thermal damage to aircraft outer skins (e.g. blistered or peeling paintwork, delaminated surfaces and exposed honeycomb), and (ii) blunted leading edges and removed wing panels.

Reference 11 also noted that literature on the subject of damaged wings was "... practically non-existent". No experimental investigations were undertaken, instead, computer based skin-friction drag calculations were developed with data extrapolated from available information on "... unusual leading edges". In part, the expected drag estimates were also developed from earlier drag prediction data (Ref. 12), which considered the effects of relatively small protrusions into surface flows and tolerable irregularities due to aircraft manufacturing processes. The method presented then suggested that the data could be scaled by the ratio of damaged to undamaged areas, to obtain a final aircraft drag increment. However, given the nature of the extrapolations and assumptions made, together with the lack of any supporting experimental data, the method suggested must be treated with caution. Drag increments were the only consideration and no attempts were made to predict possible lift loss or pitching moment changes.

In 1977, AGARD reported on an investigation into the transient, or 'impulsive' effects resulting from the close proximity detonation of a high-explosive missile warhead (Ref.13). However, only the impulse-drag resulting from the pressure wave impinging on the wing was considered.

Whilst the above references were attempting to develop computer-based models to predict damage induced drag increments, in 1976 the NASA Ames Research Centre undertook a
low-speed investigation of a full-size damaged aircraft in their 40ft x 80ft wind tunnel (Ref. 14). The aircraft used was a McDonnell Douglas A-4B with a standard fuselage and three sets of wings. In the first wing holes were cut and detachable cover plates fitted. By removing one or more of the plates, fourteen different simulated damage cases could then be investigated, each representing the hydraulic-ram effect of a panel ‘blown out’. The other two wings tested were damaged by actual gunfire at a U.S. Air Force range, using 25mm and 30mm calibre shells. Aerodynamic forces and moments were taken for the aircraft in the undamaged and damaged configurations, by varying the incidence between -4° and +22°. Variations were made in tail incidence, aileron, slats and flaps for the different test runs. The data obtained in the tests generated over 130 pages of graphs, however, no flow visualisation was undertaken and no analysis was published. It is understood that the investigation was undertaken on behalf of McDonnell Douglas, and that the results were supplied to them for evaluation. Despite direct requests, they were not able to provide any resulting analysis reports.

However, from the test data presented in Ref.14, it was seen that with through-damage of a small percentage-chord diameter, no noticeable loss in lift coefficient, $C_L$, was observed up to within 3° of stall onset (+12°). Additionally, drag increases were minimal (approximately 5% at minimum drag coefficient, $C_d$) and only slight reductions in pitching moment, $C_m$, were observed over the -4° to +22° incidence range tested. When progressively increasing the damage size, trends became clearly identifiable. As size increased so did the magnitude of lift loss. With relatively large damage present, in addition to significant reductions in lift, the onset of stall was seen to have been delayed some 2° to 3° (See Fig.1.6), which corresponds with findings to be discussed later. Minimum and induced drag increments were also seen to increase with size, and an incremental reduction in pitching moment was observed. However, with such a highly 3-dimensional wing subjected to ‘cumulative’ damage of different sizes, applied at different chordwise and spanwise locations, drawing further general conclusions on the measured effects or mechanisms present was not possible.

In 1979, the University of Texas at Austin undertook a research programme on behalf of the US Air Force Vulnerability Assessment Group (Ref. 15). The initial intention of their work was to investigate numerically, the aeroelastic failure mechanisms involved with damaged wings. However, it was again identified that “... virtually no experimental studies have been
Figure 1.6 Lift & Drag Coefficients; Undamaged & Damaged. (Ref. 14)
performed to aid the investigation on the aerodynamic modelling of damaged lifting surfaces.” Following this finding, it was intended to undertake an experimental investigation prior to computer modelling. However, the use of a full-size T-38 horizontal tail plane at relatively low Reynolds numbers resulted in inaccurate results. This was attributed to unsteady pressures due to leading edge separation effects. Consequently, only their unverified computer model results were presented. Given the level of Computational Fluid Dynamics (CFD) modelling at the time, various software limitations were identified by the report’s authors. However, it was suggested that little variation in chordwise pressure distribution was likely to occur either upstream or downstream of the damage hole, and that the calculated “... pressure disturbance is largely attenuated in approximately one hole width on either side of the damage area.” These findings are not substantiated by the results contained within this thesis, where it will be shown experimentally that significant surface pressure changes occur not only upstream and downstream of damage, but also to a chordwise location in excess of five hole radii either side of the damage hole. These discrepancies may be attributed to the limitations of the CFD modelling methods used in 1979.

The following year, the Austin research team obtained both experimental surface pressure distributions and lift and drag coefficient changes as a result of damage (Ref. 16). Although, the effects of leading edge separation had continued to prevent accurate measurements, key observations included; modest lift losses (up to 10% just prior to stall), significant drag increments, and importantly, spanwise pressure disturbances extending across the span of the model, contradicting their 1979 CFD findings.

It is believed that the Cornell Aeronautical Laboratory Inc. undertook some work on damaged aerofoils in 1952. A series of four reports were produced (Refs. 17, 18, 19 and 20). As a consequence of their age, the reports are no longer available, despite direct approaches to the technical library of the organisation concerned. However, these four reports were cited in Refs. 15 and 16, from which the following details have been obtained. Tests were undertaken on an aerofoil with various types of simulated gunfire damage, at Mach numbers of 0.3, 0.7 and 0.85. The resulting changes in lift and drag coefficients were normalised with respect to the area of damage.
Damage holes were 10% and 15% chord size in diameter, with simulated pettaling effects in the form of raised edges to form a "scoop" or a "spoiler" lip. However, Ref. 15 questioned the validity of the results due to the excessively large size of the modelled edges, which it claimed were much larger than any possible full-size edge distortions. It is worth noting that the tests were undertaken on a hollow aerofoil, which would have correctly simulated the effects of through-flow, between the lower and upper surfaces.

Following the conclusion of the high-speed model testing at NASA Langley in 1982, no further publications were made on the subject until 1993, when wind-tunnel tests were undertaken by Leishman on a “two-dimensional” section of a UH-60A Black Hawk helicopter main rotor blade (Ref. 21). A 24.1%c (c = chord) circular hole was cut through the section, centred at the 0.602c position. Changes in lift, drag and pitching moment characteristics were measured at Reynolds numbers of 1x10^6 and 2x10^6, and limited surface flow visualisation was undertaken with wool-tufts. The conclusions drawn were; (i) significant surface flowfield effects were found in the region of the damage (ii) flow separation was initiated at the upstream leading edge of the hole, (iii) aerodynamic coefficient characteristics were significantly degraded, and (iv) a method of predicting lift curve slope reduction was proposed. However, the results were based on only the one size of hole at a single location. Additionally, the tests were conducted on a relatively small wing section with a half-span-to-damage-radius ratio less than 5. The results of this thesis indicates that a wing damaged with a single hole has a highly three-dimensional flowfield, with spanwise effects extending beyond 5 damage radii. Consequently, the results of Ref.21 would probably have been influenced by the presence of the model end-plates. Such end-plates would have restricted the spanwise flowfield effects, especially at high angles of attack. Thus it is believed that the predictive techniques developed are only applicable to the damaged wing/end-plate configuration tested. If extrapolated to wings with unrestrained three dimensional damage-flow, significant prediction errors might be expected.

Additionally, Ref. 21 also proposed that the location of the forward edge of the damage fixed the position of flow separation, and thus allowed Kirchhoff’s theory to predict a reduction in lift-curve-slope. This is also disputed by the results gained here, which showed that the location of the separation point was often upstream of the forward edge of the damage, and
varied with both damage size and incidence.

Further tests were undertaken by Robinson & Leishman on advanced helicopter rotor sections and reported in Refs 22 and 23. In these References, investigations were undertaken on SC1095 and SC1095-R8 rotor sections subjected to a variety of different damage conditions. These sections were constructed with honeycomb centres, giving an effect similar to the solid wings tested here. The tests considered 19.3%c holes located at two chordwise locations (0.65c and 0.16c), together with more ‘realistic’ attempts at simulating damage. These involved the removal of a triangular section of the trailing edge, holes with surrounding upper skin removal, holes with petalling effects and actual test specimens subjected to live-fire ballistic damage.

Standard coefficient plots against incidence were used to illustrate the effects of damage. The changes due to the single hole cases indicated reductions in lift curve slopes by up to 20%, reductions in the maximum lift coefficient attainable and delays in the onset of stall. Significant drag and pitching moment changes were also measured. It was deduced that the effects of damage were more complex than just area loss alone. Attempts at re-normalising damage coefficient values indicated that it was not possible to reconcile changes by accounting for the amount of wing area removed. No alternative method was proposed. Attempts were made to compare the results from the different forms of damage (circular hole, simulated trailing edge removal and live-fire damage) for the two sections tested. Whilst a detailed comparison of the coefficient results was made, no clear characteristic trends were identified. Flow visualisation and surface pressure data indicated the presence of flow separation at the forward edge of the hole, together with a significant region of separation extending in a chordwise and spanwise direction. The extent of the separation region was noted as increasing with incidence. It appeared that the tests were undertaken in the same wind tunnel with a similar set-up to that used in Ref.21. Given this, and the flow visualisation illustrations, it may have been that the surface flows of Refs 22 and 23 were also influenced by the presence of the model end-plates. Unfortunately the images of surface flow patterns were limited in detail and only provided support in terms of general flow-structure to the results presented here. No discussion of the surface flow structures was presented, other than to state that it was complex in nature.
1.6 General Scope of the Investigation

The topic of this thesis is the experimental investigation into the effects of simulated battle-damage on the low-speed aerodynamic characteristics of a two dimensional wing. Both penetration and fragmentation effects were considered. The changes in aerodynamic characteristics were assessed in terms of both overall wing performance changes and detailed flow structure. A constant wing geometry was used throughout.

By identifying the key flow mechanisms involved and quantifying the resulting effects, the thesis then presents techniques for predicting simulated-damage characteristics. Such methods could be used in future vulnerability assessments of aircraft.

1.7 Programme Objectives

(1) To devise techniques to physically model both penetration and fragmentation battle damage effects.

(2) To devise experimental methods to investigate the aerodynamic effects of battle damage in terms of force and pressure coefficients together with flow-visualisation of both surface and local flow fields.

(3) To determine the effects of both types of battle damage on wing aerodynamic characteristics.

(4) To investigate the influence of key damage and experimental variables.

(5) To develop successful prediction techniques which may be used in vulnerability assessments.
CHAPTER 2. EXPERIMENTAL FACILITIES AND TECHNIQUES

2.1 Introduction

The following chapter describes the wind tunnel facilities used for the experimental investigation. Together with a summary of the model manufacturing process and test procedures adopted.

With the exception of recent work undertaken to reduce the working section flow turbulence levels, no significant experimental testing had been undertaken in the wind tunnel for a number of years. The author's contribution was to upgrade the overall test facilities to the necessary standards and develop the investigative procedures that were required to undertake the research. The techniques required were force and pressure measurements, as well as flow visualisation using both paint and smoke.

2.2 Tunnel Test Facilities

The experimental research contained in this thesis was undertaken entirely in the Loughborough University A.A.E.& T.S. Department's low-turbulence wind tunnel. This was an 'Open return' type wind tunnel, as illustrated in Fig. 2.1. The working section had a constant cross-section area of $0.45m \times 0.45m$, and a maximum working velocity of $38m/s$. The recent flow improvements had reduced the working-section turbulence level to less than 0.1%.

Test wings were mounted horizontally in the working section. Full-span wings were used throughout, with a wall clearance of 1.5mm each side. This prevented the force-balance from 'grounding' whilst attempting to preserve two-dimensional flow. The gap was within the tunnel wall boundary layer thickness, and flow visualisation tests indicated no significant leakage effects. Two-dimensional flow across the (undamaged) wing-span was confirmed by 'quality-check' pressure measurements (see Section 2.5.4).
The test wings were mounted on four streamlined struts. These were recessed into the lower surface of the wing, allowing the pivot-points to be i) located at 0.25c and 0.75c locations, and ii) keep them out of the tunnel airflow. The loads generated were measured by a three component Aerotech Model 528 Force Balance located directly beneath the working section, see Figure 2.2. The balance was contained in an air-tight compartment to prevent leakage into the working section through the strut clearance gaps in the tunnel floor hatch. Incidence changes were achieved by the manual adjustment of the incidence mechanism on the force-balance. Repeatability tests of the mechanism indicated an accuracy of ±0.09° (i.e. to 5' of a degree). All force measurements were within the full scale range of the balance with the only exception of high incidence missile damage drag forces. Incidence changes were performed without the need to stop the tunnel. The tunnel speed was governed by a...
dedicated control unit, which maintained a constant dynamic head with the model at a fixed incidence. Test techniques, discussed later (section 2.4.2), maintained the dynamic head to within 0.3mm H₂O of the required setting (80 mm H₂O for Reynolds Number = 5.0x10⁵ tests).

2.3 Wind Tunnel Wing Models

2.3.1 Wing Geometry

At the outset of the research it was decided that all forms of damage would be applied to a single wing geometry. Both planform and aerofoil section characteristics would be constant throughout all testing, with only the form of damage varying. After careful consultation with the project sponsor, it was decided that a NACA 641-412 aerofoil section would be used as the test section. This was because it was (i) a well known section with known characteristics at low Reynolds Numbers, (ii) capable of providing a sufficient ‘thickness to chord ratio’ to facilitate internal modelling and pressure tapping, and (iii) could be representative of possible future military aerofoils (see Global Hawk’s high aspect ratio wing, Figure 2.3). The test wing was of constant section over the full-section span (0.45m) with zero taper and twist. The chordlength was 0.2m, except for models used to verify Reynolds Number and wing area effects (in these cases the chordlength was 0.1m). Running at test speed a Reynolds Number

Figure 2.3 Teledyne Ryan’s Global Hawk

22
of 5.0x10^5 was obtained with a chord length of 0.2m. Note that NACA 641-412 aerofoil geometry is included in Appendix A.

Internal wing construction was a key factor for consideration. It was decided that all standard damage test cases would be undertaken on hollow wings. This attempted to simulate the realistic internal cavity structure of an aircraft wing. Hollow wings were achieved by manufacturing the models from two skins, which when bonded together would leave an internal void. Additionally two spars were incorporated for additional realism; a forward-spar at the 20%c position and a rear-spar at the 65%c position, see Figure 2.4.

![Figure 2.4 Hollow Wing Internal Construction](image)

2.3.2 Wing Model Manufacture

By the very nature of the project, it was intended from the outset that different forms of simulated battle-damage conditions were to be investigated. Due to the nature of the damage, it was not possible to simulate the different cases on the same wind-tunnel wing model. This necessitated the production of a number of ‘identical’ test wings, with the term identical implying both geometric and aerodynamic similarity. To this end, a method of manufacture was developed to produce ‘identical’ models. The method involved a moulded fibre-glass composite construction.

From the defined NACA 641-412 aerofoil specification it was possible to calculate detailed profile co-ordinates for both the upper and lower surfaces at 1%c chordwise steps. These were ‘spline fit’ with a least-squares curve, within AUTOCAD (Version 11). The results were then passed to a computer controlled milling machine. This was then used to cut
accurate ‘section patterns’, from which a master mould was cast. The accuracy of the process was limited by the resolution of the profile calculation, the milling machine tolerance, and wear on the cutting tool. To illustrate, at the centreline maximum thickness location of 40% of the profile thickness definition was 23.924mm, whilst the actual measured value was 23.895mm, a difference of 0.029mm (0.0145%).

The mould, made from thermally-stable epoxy-resin, was made in two halves; one upper and one lower. Each half of the outer-surface master mould was then used to make moulds suitable for manufacturing the upper and lower wing ‘skins’. The skins were to be exactly 2mm in thickness (0.01c) and included mouldings for the two internal spars, each of which was also 2mm thick. The support mounting recesses were also included in the moulding.

To produce each wing (see Figure 2.5), a mixture of glass-fibre and resin was laid up in each of the upper and lower skin moulds. Once removed from the moulds, the mounting holes were drilled accurately with the aid of a jig, ensuring their exact location on the chordline at locations of 25% and 75%. The two separate skins were then bonded together at their leading edge and spar locations. With the two skins held in position by the original master mould, the trailing edge was then cast. This bonded it to both skins and ensured a high degree of accuracy. Once removed from the moulds, each wing was cut to the required spanwise length, fitted with end-plugs and given a final polish.

In addition to the hollow wing models, two of solid construction were also required. To manufacture these, the two outer-surface moulds were used alone to cast the solid models. When removed from the moulds, they were cut to the required spanwise length and polished.

A total of twenty hollow wings were produced using this method. Following completion and before being used, each one was subjected to validation acceptance checks. Both the geometry and aerodynamic characteristics of each one were then measured and compared to a standard wing (see Section 2.4.4). Only two models were found to be unacceptable and were not used for testing.
2.4 Aerodynamic Force Measurement

2.4.1 Force Measurement Facilities

The force-balance used was specifically upgraded for this research project, and constituted the first item in the Force Data-Acquisition-System (D.A.S.). Both the hardware and software were developed specifically for the project by the author. The force D.A.S. simultaneously measured the balance forces and the tunnel dynamic-head, and recorded the values on an IBM compatible 486-DX Personal Computer (PC). The main stages of the Force D.A.S. are illustrated by Fig. 2.6.

The force balance output consisted of three analogue voltages, each representing the three components; lift, drag and pitching moment. In addition, the tunnel dynamic head pressure transducer also produced an analogue voltage. These four channels were then sampled by a
16-bit CIL Analogue to Digital Converter (A.D.C.). The D.A.S. software running on the Pc computer performed the following; pause for a 5 second settling period (to allow the balance to reach a steady-state value), then digitally sample and measure the four voltage channels. The design of the Force D.A.S. was such that data sampling took place at a rate of 100 data measurements per data-cycle. The mean average value was then taken for each cycle. This was then repeated at a rate of 2 Hz, which was twice the cut-off frequency of the force-balance’s low-pass filters. Data measurement cycles continued at a rate of 2 Hz for 15 seconds to give a final time averaged value. These data measurement timings were defined following a series of force measurement optimisation tests. Note that the data sampling ‘interlaced’ all four channels to give simultaneous time averaged measurements. The discrete voltage values were then converted into the corresponding dimensional quantities, using the calibration equations supplied by the balance and pressure transducer manufacturers. The final values were then written to a text file on the Pc’s hard drive, suitable for post-processing at a later time.

Static load and pressure calibration checks undertaken on the data acquisition system as a whole indicated an accuracy of ±0.0014, ±0.0001 and ±0.0001 for lift, drag and pitching.
moment coefficients respectively. These values were well within the factory-quoted accuracy of the balance alone.

2.4.2 Experimental Test Procedures & Techniques

Incidence Trimming
The struts used to mount the wing models on the balance were also the means of setting the wing incidence. Given that the same fixed-length supports were used throughout the test programme, the exact incidence of each individual model was dependent on the location of the internal pivot points. Although these points had been located using a jig, it was found that slight inaccuracies occurred. These resulted in the hole centre being up to ±0.35mm off the chordline, which in turn produced up to a ±0.2° error in the model’s incidence.

To correct any such error the zero-lift incidence of each undamaged model was adjusted to match that of the accepted definition (see Appendix A) of the NACA 641-412 aerofoil. This was quoted as being -2.8°. An accurate inclinometer was attached to the wing via a mounting-jig contoured to the upper surface profile. By an iterative process the models incidence was then adjusted within the ±0.2° error band, to the correct position. This ensured the correct model incidence for both the undamaged and damaged force measurement tests.

Force Measurement Test Run Procedure
Each force-measurement test run consisted of two parts; wind-off tares and wind-on measurements. The procedure was identical for both undamaged and damaged wings.

(i) Wind Off Tare Run
Initially, the model was traversed through the full test incidence range with the tunnel speed at zero. Incidence was set by hand using the balance mounted angular scale. The Force Data Acquisition System (D.A.S.) measured and recorded the wind-off ‘tare’ force values for lift, drag and pitching moment.

(ii) Wind On Test Run
Firstly, the tunnel was set to the required test dynamic head value. The Force D.A.S. then
recorded the wind-on force values and dynamic head at each individual incidence point. Again incidence setting was done by hand. Throughout the duration of the test run the tunnel dynamic head was adjusted by hand so as to stay within 0.3mm H₂O of the required setting. This ensured that the Reynolds Number varied by less than 1x10⁵, from the nominal value of 4.98x10⁵.

Now, by subtracting wind-off tare values from wind-on measurements, pure aerodynamic loads were calculated. Additionally, to ensure the accuracy of data, each test run was undertaken twice. Data from the second run were used only to provide confirmation of the results from the first set. No form of results averaging was undertaken, and only the first data run values were subsequently processed.

2.4.3 Wind Tunnel Data Corrections

Having calculated the pure aerodynamic loads measured by the Force D.A.S., the corrected wing ‘free-stream’ coefficient values were then calculated using the following corrections.

Mounting Strut Influence Corrections
Due to the strut mounting method used to support the model in the working section, the aerodynamic forces measured by the balance also included components resulting from the struts. As the struts were common to all tests, it was possible to subtract their contributions directly from the measured aerodynamic loads.

Tunnel Wall Coefficient Corrections
Due to the constraining effect of the wind tunnel walls on the flow around the wing, corrections were applied to the tunnel-based coefficients to calculate the equivalent ‘free-stream’ values. The commonly used wind-tunnel corrections were defined by ESDU (Ref. 24). However, these contain assumptions and simplifications (e.g. limited chord to height ratios) to allow their general application to different configurations of working section. The original unsimplified equations were published in AGARDograph 109 (Ref. 25). It is these detailed equations which have been used here. It was felt that these would be more appropriate as they are not limited by the assumptions made by ESDU.
Goldstein (Ref. 26) developed correction equations in terms of a power series of the chord to height ratio \((c/h)\), suitable for larger than usual values of \(c/h\), thickness to chord ratio \((t/c)\), camber or lift coefficient over a wide incidence range \((\alpha)\).

They give incidence correction, \(\Delta \alpha\), as:

\[
\Delta \alpha = \frac{\pi^2}{96} \left(\frac{c}{h}\right)^2 \left(2\alpha + D_1\right) + \frac{\pi^4}{92160} \left(\frac{c}{h}\right)^4 \left(-2\alpha + 20D_2 - 21D_4\right)
\]...

Lift coefficient correction, \(\Delta C_L\), as:

\[
\Delta C_L = -\frac{\pi a_1 c_L}{96} \left(\frac{c}{h}\right)^2 + \frac{7\pi^3 a_1}{30720} \left(\frac{c}{h}\right)^4 \left[3C_L + 2\pi(2\alpha - D_1 + D_2 + D_3)\right]
\]...

and Pitching moment correction, \(\Delta C_m\), as:

\[
\Delta C_m = \frac{-\Delta C_L}{4} + \frac{7\pi^5}{61440} \left(\frac{c}{h}\right)^4 \left(2\alpha + D_2\right)
\]...

where

\[
D_n = \frac{4}{\pi} \int_0^1 \frac{Z e}{c} \sin \theta \sin \theta d\theta
\]

when in equation (4), for distance \(x\) from the leading edge:

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

**Blockage Correction Factor**

The total blockage correction factor \((\varepsilon_B)\) applied is the summation of the solid blockage factor \((\varepsilon_S)\) and wake blockage factors \((\varepsilon_W)\), each calculated independently, for non-bluff bodies;

\[
\varepsilon_B = \varepsilon_S + \varepsilon_W
\]...

Each factor was calculated as follows.
i) Solid Blockage $\varepsilon_s$
Goldstein’s solid blockage equations were extended by his thickness profile approach (which accounted for the wings finite thickness) to include terms up to the fourth power of ($c/h$), calculated for the individual aerofoil’s profile. Thus the blockage factor (at zero incidence) was given by:

$$\varepsilon_{s0} = \frac{\Pi 4}{6h^2} + \frac{\Pi^3}{960} \left(\frac{c}{h}\right)^4 \int_0^\pi \frac{Z_c \cos^4 \theta}{\cos \theta} \, d\theta$$

...(7)

where $\theta$ is defined in equation (5).

A simple calculation was then used to adapt the theory to deal with solid blockage of an aerofoil at a given incidence. Batchelor (Ref. 27) showed that the increase in solid blockage factor is proportional to the square of the incidence. Thus giving the relationship between the zero incidence factor, and that at an incidence (in radians), as:

$$\varepsilon_s = \varepsilon_{s0} \left[1 + 1.1 \left(\frac{c}{h}\right) a^2\right]$$

...(8)

ii) Wake Blockage $\varepsilon_w$
Wake blockage is related to the measured body drag coefficient. The definition for wake blockage has been derived in different ways by many authors, and for incompressible flow is accepted as;

$$\varepsilon_w = \frac{1}{4} \left(\frac{c}{h}\right) C_d$$

...(9)

Wake Blockage Gradient
Calculations involving the effects of wake blockage on the local velocities around a model have been developed. Unlike solid blockage effects, wake effects are not symmetrical longitudinally about the model, but increase along the chord length, ie. a longitudinal velocity gradient exists. The resulting buoyancy effect imposes additional drag force on the model;
\[ \Delta C_d = -C_d \varepsilon \]  

...(10)

This effect is the only change in the drag coefficient value.

**Kinetic Pressure Correction**

Given the previous equations for solid and wake blockage corrections, the total blockage factor is used to correct the stream kinetic pressure. This was done by the development of the factor 'G', which was applied to all aerodynamic coefficients;

\[ G = \frac{1}{1 + 2\varepsilon_B} \]  

...(11)

thus, the free-stream values are given by;

\[ \alpha_f = \alpha + \Delta\alpha \]  

...(12)

\[ C_{Lf} = G(C_L + \Delta C_L) \]  

...(13)

\[ C_{df} = G(C_d + \Delta C_d) \]  

...(14)

\[ C_{mf} = G(C_m + \Delta C_m) \]  

...(15)

Some equation sources indicated that the correction methods were not necessarily accurate for values of \( c/h > 0.3 \). Given that the models used here had a value of \( c/h = 0.44 \), the results obtained indicated that the corrections were acceptable (see Section 2.4.4 below).

To illustrate the magnitude of the overall tunnel corrections applied to the undamaged wing, the following lists the corrections to incidence, \( C_L \), \( C_d \) and \( C_m \) at test incidences of \( 0^\circ \) and \( +14^\circ \). Percentage changes in terms of the uncorrected coefficients are also given;

<table>
<thead>
<tr>
<th>( \alpha )</th>
<th>( C_L )</th>
<th>( C_d )</th>
<th>( C_m )</th>
</tr>
</thead>
<tbody>
<tr>
<td>( 0^\circ )</td>
<td>( 0.0002^\circ )</td>
<td>0.0141 (4.8%)</td>
<td>0.00019 (2.4%)</td>
</tr>
<tr>
<td>( +14^\circ )</td>
<td>( 0.5341^\circ )</td>
<td>0.0955 (7.6%)</td>
<td>0.00786 (6.0%)</td>
</tr>
</tbody>
</table>
2.4.4 Data Validation, Accuracy and Repeatability

Comparison with Published NACA 641-412 Data
A 'definitive' wing model was established firstly to validate the corrected 'free-stream' coefficient values obtained, and secondly to establish a 'control' model (Model 003) against which all subsequent manufactured models were compared. The experimental results obtained here were at a Reynolds Number of 500,000, whilst the published data (Ref. 28) for the NACA 641-412 aerofoil were at a Reynolds Number value of 700,000 (the closest match available). The results are illustrated in Figures 2.7 to 2.9.

Figure 2.7 illustrates both the experimental and published lift curve slopes. The published NACA 641-412 (undamaged) lift curve slope was quoted as 0.1011, here the calculated corrected free-stream value was 0.1000, an error of -1.09%. Drag results (Fig. 2.8) were acceptable, although a rapid drag increase at increased lift levels was seen. This may be explained by the experimental results lower Reynolds number value. Additionally, the published drag data might underestimate the true $C_d$ values as they were obtained by wake traverse rather than direct balance measurement. Pitching moment differences were minimal at low negative incidences, but increased with incidence (Fig. 2.9). This may be explained by the trailing edge being inaccurately modelled on the 200mm chord models used, where the last 1.5%c was missing (resulting from manufacture limitations). Given the section profile, it might then be expected that reduced pitching moment would occur at high positive incidence values.

Coefficient Repeatability from the Same Model
The process of measuring both undamaged and damaged wings required that the same model be removed and replaced within the tunnel between runs. Repeatability checks were undertaken to prove that this process did not unduly influence the test results. These indicated that for the same model that had been removed and re-installed, the resulting errors fell within the following bounds over the full incidence range tested;

- Lift coefficient : $\pm 0.0043$
- Drag Coefficient : $\pm 0.0005$
- Pitch Coefficient : $\pm 0.0023$
Fig 2.7  $\frac{d(CL)}{d(\alpha)}$ Comparison with Published Data

Model 003  Published data
Lift curve slopes:  (0.1000)  (0.1011)
ReyNo. = 500,000  ReyNo. = 700,000
Fig 2.8 Drag Comparison with Published Data

![Graph showing drag comparison]

- Model 003
  - ReNo. = 500,000

- Published data
  - ReNo. = 700,000
Fig 2.9 Pitch Comparison with Published Data

![Graph showing pitch comparison with published data.](image-url)

- Model 003, $ReyNo. = 500,000$
- Published Data, $ReyNo. = 700,000$
Incidence Traverse Hysteresis

Tests quantifying the effects of incidence traverse direction on hysteresis (between the two bounds of $+14^\circ$ and $-10^\circ$), indicated that the maximum variation in coefficient values, were within the following bounds:

- Lift coefficient: $\pm 0.0050$
- Drag Coefficient: $\pm 0.0007$
- Pitch Coefficient: $\pm 0.0040$

Model to Model Repeatability

Following the determination of the ‘definitive’ wing model, all subsequent models were then compared against its characteristics and fell within the following tolerances:

- Lift Coefficient tolerance: $\pm 0.010$
- Drag Coefficient tolerance: $\pm 0.001$
- Pitching moment Coefficient tolerance: $\pm 0.004$

These tolerances reflected the acceptable spread of characteristics obtained from the eighteen wing models accepted. The models which failed to exhibit acceptable characteristics fell well outside of at least one of the above ranges, and were rejected for use in the test programme.

Results from four wing models are presented to illustrate the repeatability of coefficient values between accepted models. The four were chosen at random from the eighteen used in the test programme. The results were compared over the range $-6^\circ$ to $+8^\circ$, avoiding data scatter due to stall.

The following values indicate the greatest difference measured between the four coefficient data-sets recorded:

- Lift coefficient: $< 0.0091$
- Drag Coefficient: $< 0.0018$
- Pitch Coefficient: $< 0.0051$
These results are illustrated by Figures 2.10 to 2.12. Whilst the repeatability between models for lift and drag coefficients indicate relatively small differences, pitching moment showed greater variations between models. It is believed that this is primarily as a result of (i) variations in friction at the wing's pin-jointed mounting locations, and (ii) slight geometric differences between the models. Figure 2.10 also lists the lift curve slope values obtained from each of the four example test models. It was found that all the gradient values were within 0.0008 per degree of each other.

**Definition of Model Baseline Characteristics**

It must be noted that in order to assess the aerodynamic effects due to simulated battle damage, it was the change in the coefficient values between the undamaged and damaged conditions which were of primary interest.

Although the acceptance tests on each model ensured that they were all similar aerodynamically, in order to ensure maximum results accuracy each model's undamaged coefficient values were recorded to define individual 'base-line' characteristics.

### 2.4.5 Damaged Wing Data Presentation

With the 'base-line' characteristics identified, the model was removed from the wind tunnel and the particular form of simulated damage was applied. The model was then remounted on the balance. The strut-mounting method used preserved the datum incidence setting for the particular wing model. Re-testing of the model was then undertaken, to measure the new damaged-wing coefficients.

The effects of the damage were then calculated as; \(d[C_L]\), \(d[C_d]\) and \(d[C_m]\), the changes in lift, drag and pitching moment coefficient values at a given incidence.

\[ d[C_L] = C_{L\text{, damaged}} - C_{L\text{, undamaged}} \]

\[ d[C_d] = C_{d\text{, damaged}} - C_{d\text{, undamaged}} \]

\[ d[C_m] = C_{m\text{, damaged}} - C_{m\text{, undamaged}} \]
Fig 2.10 Undamaged Wing Repeatability Checks; Lift

Lift curves for different models:
- Model 003: Slope = 0.1000
- Model 007: Slope = 0.1000
- Model 012: Slope = 0.1006
- Model 018: Slope = 0.0998

Reynolds Number = 500,000
Fig 2.11 Undamaged Wing Repeatability Checks; Drag

Model Reynolds Number = 500,000
Fig 2.12 Undamaged Wing Repeatability Checks; Pitch

Reynolds Number = 500,000


2.5 Wing Surface Pressure Measurements

2.5.1 Pressure Tapping Equipment

**Wing Model**

A single wing model was used to obtain static pressure data on both the upper and lower surfaces. Pressure coefficient values, $C_p$, were calculated as:

$$C_p = \frac{p_t - p_0}{\frac{1}{2} \rho V^2}$$

where:

- $p_0$ = Static pressure of undisturbed flow
- $p_t$ = Local surface tapping static pressure

In the undamaged state, chordwise pressure distributions were obtained at two spanwise stations. Firstly, along the proposed damage centreline which coincided with the wing centreline, and secondly at a spanwise distance of 5R, where $R=20\%c$ damage radius, see Figure 2.13. A total of 47 tapping locations were used (24 upper and 23 lower surface) at each spanwise station, with measurements made over the full incidence range, $-10^\circ$ to $+14^\circ$ in $2^\circ$ steps. Note that no tappings were possible beyond 0.925c due to insufficient thickness of the trailing edge. The locations of the pressure tappings are given in Table 2.1.

![Figure 2.13 Undamaged Wing Pressure Tapping Stations](image)

In the damaged configuration, the wing chordwise pressure distributions were obtained at five spanwise stations; the damage centreline, 0.5R, 1.5R, 2.5R and 5R (see Figure 2.14).
Tunnel-based Pressure Data Facilities
The pressure tapped wing was mounted on the four streamlined struts. However, these were purely for support, and no force measurements were recorded. The pressure tapping tubes were moulded into the upper and lower surfaces of the wing and all emerged at one end of the model. At this end a circular end plate was attached to allow the wing to be mounted in the tunnel. The end plate fitted flush against the tunnel wall and maintained an airtight seal, see Figure 2.15. With this configuration the tunnel flow was not disturbed by exposed pressure tubings, and the wing incidence might be varied over the full range of values required.

Figure 2.15 Pressure Tapped Wing Mounted in Tunnel (Plan view)
The pressure-tapping tubes exited the model and were attached to a scanivalve, as seen in Fig. 2.16. The scanivalve output was measured by a Setra 239 low-pressure transducer, with a F.S.D. of ±381 mm H₂O. The voltage output of which was fed into the 16-bit CIL Analogue to Digital Converter (A.D.C.). The output from the independent tunnel dynamic head pressure transducer was input on a separate A.D.C. channel. The Pressure D.A.S. software then sampled these two input channels, using the same sampling method employed by the Force D.A.S., to produce time averaged interleaved values for the scanivalve channel and tunnel dynamic head. The process was then repeated for each scanivalve pressure channel in turn. This also required the software to control the scanivalve positioning mechanism via an intermediate control unit.
When all pressure tapping channels were evaluated, voltage values were then converted into the corresponding dimensional quantities using the two pressure transducer calibration equations, and the final values were again written to a text file on the PC’s hard drive. Pressure calibration checks indicated an accuracy of ±0.27mm H₂O (±0.33% tunnel dynamic head).

2.5.2 Test Procedures

Incidence Trimming
When the pressure tapped wing had been mounted within the tunnel, the incidence was finely adjusted to ensure zero-lift at -2.8°. This was confirmed by integrating the surface pressure $C_p$ values.

Test Run Procedure
The following procedure was followed for each pressure-measurement test run. Due to the fixing method of the end mounting plate, the wing incidence was not adjustable during a test run. Thus each run considered a single incidence setting for a particular damage case,
measuring both upper and lower surface pressures at a single spanwise station.

The test procedure was as follows;

(i) Firstly, wind-off tares were recorded for each of the two pressure transducers.
(ii) The tunnel was run up to a constant speed, to give the required tunnel dynamic head pressure.
(iii) The D.A.S. software initiated the scanivalve, by setting it to the first pressure tapping channel.
(iv) A settling time period of 2 seconds allowed full equalisation of the tapping and pressure transducer tubes, and also the pressure transducer to reach the required deflection value.
(v) The D.A.S. then measured and recorded interlaced values for both the tunnel dynamic head and pressure tapping channel, time averaged over 10 seconds.
(vi) The D.A.S. cycled through the above steps (iv) and (v) for all 47 surface pressure tapping channels, recording the results in a text file.
(vii) The tunnel was stopped, before adjusting the incidence setting to the next value.

The above procedure was repeated for each incidence setting required, in the range -10° to +14°.

Initial pressure tests indicated the optimum settling and measurement times defined above for the pressure transducer / scanivalve configuration used.

2.5.3 Data Processing

From the surface pressure tappings and tunnel dynamic head measurements taken, the local surface pressure coefficient \( C_p \) values were obtained. However, no satisfactory correction methods have been developed to date, to correct surface \( C_p \) values for the constraining effects of wind tunnel walls. It is commonly accepted that such pressure data is presented 'as measured' on the test model, i.e. uncorrected. This approach was adopted throughout the \( C_p \) results presented here.
2.5.4 Pressure Data Accuracy

Pressure Coefficient Repeatability

The repeatability of pressure coefficient data was considered for both the damaged and undamaged cases, for all pressure tapping locations. The error was defined as the difference between \( C_p \) values recorded at the same tapping location on three independent test runs. Values were recorded for all tapping locations, at incidences values with (i) minimal lift \( 0^\circ \), (ii) maximum lift with stable pressures, i.e. pre-stall, at \( +10^\circ \) and (iii) maximum lift with unstable pressures, i.e. post-stall, \( +14^\circ \).

The error bands were found to increase over the three incidences tested, with the maximum value occurring post-stall. Below lists the largest \( C_p \) error seen over all 47 tapping locations, and at an example location of 26%\( c \) (used in later comparisons):

<table>
<thead>
<tr>
<th>Incidence</th>
<th>Error Band (all tappings)</th>
<th>Error Band (26%( c ) location)</th>
</tr>
</thead>
<tbody>
<tr>
<td>( 0^\circ )</td>
<td>0.0086</td>
<td>0.0009</td>
</tr>
<tr>
<td>( +10^\circ )</td>
<td>0.0387</td>
<td>0.0015</td>
</tr>
<tr>
<td>( +14^\circ )</td>
<td>0.1176</td>
<td>0.0405</td>
</tr>
</tbody>
</table>

Figure 2.17 illustrates the repeatability of the results, for this example, taken along the centreline of 10%\( c \) quarter-chord damage, at \( +10^\circ \) incidence. This damage case is considered here to illustrate the \( C_p \) repeatability of damage flow.

Spanwise Uniformity of Results

Undamaged spanwise variations in pressure data were found to be slightly greater than the general repeatability of data at the individual tapping locations. By way of comparison with the above \( C_p \) error band values, comparing undamaged data from the 26%\( c \) location at the two spanwise locations indicated a \( C_p \) difference of 0.0118 at \( +8^\circ \). This was greater than the identified error band, indicating that the differences due to spanwise location appear to be a genuine effect due to small geometric variations in the model. Figure 2.18 illustrates the differences between the complete centreline \( C_p \) results and those taken at a spanwise distance of 5 damage radii, at \( +8^\circ \) incidence.
Fig 2.17 Cp Repeatability (10% quarter-chord hole at +10 deg incidence)

ReyNo. = 500,000

Run A          Run B
Fig 2.18 Cp Spanwise Variations (Undamaged wing at +8 deg incidence)
2.6 Flow Visualisation Techniques

Flow visualisation tests were also undertaken. The aim was to gain an insight into the actual flow mechanisms present. This would aid in the understanding and interpretation of the quantitative results.

Experimental flow visualisation was undertaken using two specific techniques; smoke released into the local flow field, and surface flow visualisation paint. Both of these techniques proved successful methods of illustrating the damage flow-structure.

2.6.1 Local Flow-field Smoke Visualisation

A smoke generator was used to produce a continuous supply of smoke, which was released into the local flow field in the region of the damage. Introducing the smoke into the flow at different points illustrated, (i) the propagation of the ‘through-damage’ flow, and (ii) the flow characteristics within the damage wake;

Through-Damage Flow Smoke Injection
Firstly, smoke was fed from the smoke generator, via a smoke reservoir, into the central box section of the hollow wing. This allowed the flow through the damage to be seeded with smoke, see Figure 2.19. By lighting the flow from above, together with reduced ambient lighting conditions, it was possible to illuminate the smoke particles within the damage flow.
Still photography was undertaken using a conventional Single Lens Reflex (SLR) camera taking long-exposure photographs. Both side and rear camera viewpoints were used to photograph the flow-field, see Figure 2.20. This allowed the study of the flow-structure in both chordwise and spanwise directions.

![Diagram of plan view of flow direction with rear and side viewpoints labeled](image)

**Figure 2.20 Smoke Flow Visualisation Camera View Points**

**Damage Wake Smoke Injection**

To obtain an insight into the structure of the flow immediately down stream of the damage hole, smoke was injected from the wing surface into the damage wake flow. This was undertaken by feeding smoke back through a pressure tapping at each of two chordwise locations in turn, 67%c & 50%c (See Fig 2.19). Both tapping positions were in line with the damage centre. The relative pressure between the smoke and the tunnel air flow was low so as not to unduly influence the structure of the damage wake. With this arrangement, only the side viewpoint was used to photograph the flow-field.

**2.6.2 Surface Flow Visualisation**

Experimental investigation of the surface flow patterns considered the flow directions, separation regions and stagnation lines, etc. of both the damaged and undamaged test wings. The standard technique of flow-visualisation ‘paint’ was used. A mixture of white titanium dioxide powder, linseed-oil and paraffin was combined, and painted uniformly over the wing surface. The tunnel was immediately run to test speed. Once a steady flow pattern had been achieved, photographs were taken through the upper tunnel viewing hatch, see Figure 2.21a.
In addition detailed notes and sketches were taken by the author. By photographing through the viewing-hatch it was possible to record the effects with the tunnel running, i.e. with the flow still acting on the wing. Photographs of the wing's lower surface flow-visualisation were obtained by the use of a mirror and zoom lens on the camera, see Figure 2.21b.

![Figure 2.21 Upper Surface Flow Visualisation Photography](image_url)

**Figure 2.21 Upper Surface Flow Visualisation Photography**
CHAPTER 3. SENSITIVITY OF TEST AEROFOIL TO SIMULATED GUNFIRE DAMAGE.

3.1 Introduction

This chapter firstly considers the method of simulating gunfire damage to a wing. Various characteristics of actual damage are examined before outlining the key basic assumptions used in modelling. The results of the qualitative and quantitative investigations into the aerodynamic characteristics of the damaged wings are then presented.

Both gunfire damage chordwise location and size were examined in the parametric studies. The results are discussed in terms of both flow mechanisms and changes in lift, drag and pitching moment coefficients.

3.2 Methods of modelling Gunfire Damage.

Current UK defence standards indicate that up to 40% of battle-damage to aircraft will be as a result of anti-aircraft gunfire (Ref. 3). Given the large array of variables involved with gunfire damage, the range of damage which may occur to a wing is extensive. However, in order to study the effects of such damage it was necessary to reduce the large number of damage forms into a small number of representative types, which could be investigated with the minimum number of test runs.

Due to the lack of previous experimental studies, no experimental techniques were readily available to help in quantifying gunfire damage at the outset of the project. Following consultations with British Aerospace (Military Aircraft Division) and referring to the relevant Military Standards (References 1, 3), it was possible to categorise the most important basic features of damage threats and characteristics. This allowed a systematic approach to the investigation and led to the definition and evaluation of a number of key variables. This in turn, allowed parametric studies to be undertaken into some of the most important variables. The following text outlines the key variables and assumptions identified.
3.2.1 Damage Occurrence
Damage occurrence measures the number of times the wing has been hit. Actual gunfire occurrences may be a single hit, or a number of closely located hits. However, in this chapter only the effects of a single hit will be considered. (Two damage hits will be discussed later, in Chapter 5.)

3.2.2 Extent of Penetration
Given the different calibre’s and types of anti-aircraft artillery in use, the extent to which damage extends through the aircraft structure can vary significantly. However, given that the High-Explosive shell is considered to be the most common threat (Refs. 1 and 4) and that wing structures are relatively lightweight, it was assumed that damage would extend completely through the wing structure, giving a ‘through-hole’. Damage of this type was accepted to be one of the most common types of damage encountered (Ref. 15). The term ‘through-hole’ implies both an entry and exit hole, one on each wing surface. This was believed to be the ‘worst-case’ situation where there was a flow path through the wing structure from the lower to upper surface, or vice-versa. With a hole in only a single surface, no such path would be present, hence there would be no flow through the damage to disrupt the upper surface pressure distribution. Here the assumption was made that damage would be of the ‘through’ type with an entry/exit hole on both surfaces.

3.2.3 Direction of attack
The direction from which an anti-aircraft artillery shell approaches a wing can vary significantly. However, there were a few key assumptions which were used to help simplify the problem. Firstly, attack directions of ‘ahead and below’ or ‘above from the rear’ are generally the most common, the former from ground based anti-aircraft emplacements and the latter from aircraft mounted cannon (Ref. 29), as illustrated by Figure 3.1. Now, given the assumption that the extent of penetration was ‘through’ the target, this results in the same form of damage for either direction when entry and exit holes were assumed to be of equal size (to be discussed later). Thus, the range of attack direction can be reduced to an angle between 0° and 90° to the chord line. Studies into attack direction probability (Ref. 30) showed that the distribution of attack angles was not uniform over this range, but biased towards higher angles. The test simulations varied the approach direction and speed.
of an aircraft against anti-aircraft positions, to obtain the predicted impact angles. The findings indicated that attack directions between $70^\circ$ and $80^\circ$ degrees were the most common (see Figure 3.2).

In addition, military aircraft aerofoil sections are relatively 'thin', i.e. the thickness to chord ratio can be less than 6%. Consequently, even at relatively shallow attack angles, the difference in chordwise location of upper and lower surface holes would only be a few percent of the chord.
Thus, given the bias in attack direction towards 90° to the chordline, and the relatively small differences resulting from a variation from 90°, a key assumption was made that all damage would have entry and exit holes at the same chordwise location, i.e. with an attack direction of 90° to the chordline. This resulted in the upper surface hole directly above the lower surface hole. The holes were located on the model wing centreline to avoid asymmetric loading of the tunnel balance and to give maximum spanwise displacement from any possible side-wall interference effects.

3.2.4 Damage Shape
The many different types of anti-aircraft artillery in use can produce a variety of forms of damage. The factors influencing the damage encompass not only the key variables considered here, such as the attack direction, but also a larger number of smaller variables. Live-fire testing (Ref. 31) has shown that the type of fuse used or the velocity on impact, may vary the size of damage. The tests also indicated that the most common damage shape was circular. Whilst this shape might be expected from a non-explosive armour-piercing shell, the results showed that similar damage was also obtained from a high-explosive shell, where the shell’s “fragment spray zone” passed through the surface, see Figure 3.3. Thus in modelling the gunfire damage for the current tests, it was decided that a fair assumption would be to model circular shaped damage holes.

![Figure 3.3 Damage from a Contact-Fused H.E. Shell](image)

Further, characteristics of a damaged aircraft skin will depend on the skin’s material. In general, at the point where the projectile passes through a metallic skin, the extreme shear forces effectively ‘punch’ a hole through, removing the material to form a hole. In addition,
extensive cracking and deformation of the surrounding material leads to the effect known as petalling (see Figure 3.4a). This is known to be a highly random effect, with the petalling of the surface in the direction of the shell trajectory, i.e. into the structure at the entry-hole and into the airflow at the exit hole.

In the case of a skin manufactured from composite materials, i.e. carbon-fibre, at the point where the projectile/spray passes through the skin, the extreme shear forces again ‘punch’ a hole through. However, the petalling effect does not occur (Ref. 29)(Ref. 32)(Ref. 33) with composites. Although some delamination may occur around the damage edge, the surface remains fundamentally flat and does not protrude into the airflow, see Figure 3.4b.

![Figure 3.4 Skin Damage Characteristics](image)

For the current tests it was decided that the holes used to model the damage would not attempt to simulate any forms of petalling effect. This decision was based on the following;

(i) When petalling does occur, it is highly random in shape. Removing this variable from the test programme increased the generality and applicability of the results.

(ii) The anticipated continued application of composite materials in wing manufacture reduces the need to consider metallic-structure petalling.

(iii) Where petalling does occur in real-life it is typically less than 0.6% of an aircraft wing chord (Ref. 29), and so would be very hard to reproduce accurately on the models.
3.2.5 Damage Size

Actual survivable damage resulting from gunfire damage may extend up to 1000mm in diameter (Ref. 29). Damage size was to be a key variable in the parametric studies, and can be expressed in terms of a percentage ‘diameter to chordlength’ value. It is possible for this to vary from 100%c for a large shell at the wing tip, down to 5%c for a small shell at the wing root (Ref. 29). Also, the same size damage to a main wing or canard may give significantly different percentage chord sizes.

When selecting a realistic range of gunfire damage sizes to model, the structural strength of the wind tunnel models were also considered. The final range of damage sizes were defined as 10%c to 40%c (percentage chord), in 10%c increments. This gave a wide baseline of damage-size parametric tests, with the possibility of interpolation to intermediate sizes within this range.

3.2.6 Location of Damage

As indicated previously, all gunfire damage was located along the centreline of the two-dimensional wing model. Thus, the location of the damage was reduced to a one-dimensional problem; a value of $x/c$, where $x$ was the distance from the leading edge to the damage hole centre and $c$ the wing chord.

Given the nature of the gunfire threat, the wing may be damaged at any point along its chord. However, air-to-ground studies (Ref. 30) indicated that the distribution of the frequency of hits with chordwise location was not uniform. Figure 3.5 illustrates the findings that the leading edge was the region most likely to be hit. However, this should be treated with caution, as the threat direction considered was only from a ground based ‘below and ahead’ position. Air-to-air combat would be more likely to result in damage towards the trailing edge.

Noting the above points, four locations were chosen; the leading edge (L.E.), quarter chord ($c/4$), half chord ($c/2$) and trailing edge (T.E.) positions. Gunfire damage modelling was
centred at each of these points in turn. The points were chosen as they included the two extremes of the leading and trailing edges, as well as two ‘mid-chord’ values biased towards the front half of the chord. The pressure distribution around a typical aerofoil will be such that a pressure peak would develop on the upper surface and move forward with increasing incidence. Intuitively, this would imply that the forward half of the chord was the most likely to be susceptible to the effects of damage. Figure 3.6 is a plan view schematic illustrating the

Figure 3.5 Frequency of Hits as a Function of Chordwise Location
(Air to Ground Attack)

Figure 3.6 Gunfire Damage Location Schematic
The tests were undertaken separately at each of the four points, with the intention that any predictive techniques would be capable of interpolation to intermediate chordwise locations.

3.2.7 Summary of Simulated Damage Characteristics

To summarise the above assumptions; simulated gunfire damage was modelled on wind tunnel wing models by a smooth circular hole through both upper and lower wing surfaces, centred on an axis normal to the chordline. The hole axis was at the wing mid-span position, at one of four chordwise locations. The hole size and chordwise locations considered are illustrated in the test matrix Table 3.1.

<table>
<thead>
<tr>
<th>Damage Location</th>
<th>Damage Sizes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Leading Edge (L.E.)</td>
<td>10%c 20%c 30%c 40%c</td>
</tr>
<tr>
<td>Quarter-chord (c/4)</td>
<td>10%c 20%c 30%c 40%c</td>
</tr>
<tr>
<td>Half-chord (c/2)</td>
<td>10%c 20%c 30%c 40%c</td>
</tr>
<tr>
<td>Trailing Edge (T.E.)</td>
<td>10%c 20%c 30%c 40%c</td>
</tr>
</tbody>
</table>

Table 3.1 Gunfire damage modelling parametric test matrix
3.3 Leading Edge Damage; A Qualitative & Quantitative Investigation

3.3.1 Damage Cases Considered

As indicated in section 3.2.6 leading edge damage takes the form of a semi-circular ‘notch’ in the leading edge. Given the 3-box construction of the hollow wing, with the 1%c thick front spar position at 20%c, none of the leading edge damage cases broke through this spar.

3.3.2 Smoke Injection Flow Visualisation

Using the smoke injection technique described earlier, it was possible to seed smoke into the wing either side of the leading edge damage location (See Figure 3.7, points A & B). The 40%c L.E. damage case is used here to illustrate the flow characteristics. This 40%c damage case was the only one possible to test with smoke, however, it was believed to be representative of all leading edge damage cases considered.

![Plan View Diagram](image)

Figure 3.7 Leading Edge Damage – Smoke Injection Points

Figures 3.8(a) and (b) illustrate the development of the flow field at incidences of 0° and +12°. Note that due to uneven smoke injection at points A and B vortices of uneven strength were observed. This results only from the limitations of the flow visualisation technique used and does not imply any asymmetry in the vortices present. The dominant flow mechanism was a pair of contra-rotating vortices, observed at all incidences. Each vortex core started at the point where the damage intersected the leading edge. At low incidences, the size of the
Figure 3.8 Smoke visualisation of 40%c leading edge damage (from rear) ReyNo.=500,000
vortex pair appeared small, and the vortex core remained close to the surface (following the surface profile) over the full chordlength (Figure 3.8(a)). At near zero lift incidences, vortex pairs were seen on both the upper and lower surfaces. However, with increased incidence, the vortex pair were seen only on the upper surface and appeared larger. The vortex cores moved further away from the upper surface and no longer followed the surface profile (Figure 3.8(b)). Observations of the vortices indicated their continued interaction with the surface flow up to the trailing edge, at which point they detached from the wing. Figure 3.9 summarises the effects observed.

(a) Low Incidence

Small vortices

Vortices remain close to surface

(a) High Incidence

Vortices move away from surface

Enlarged vortices

Figure 3.9 Vortex-pair From Leading Edge Damage
3.3.3 Surface Flow Visualisation

Variation of surface flow with incidence.

Surface flow visualisation results from leading edge 20%c damage, are illustrated by figures 3.10(a) and (b) at incidences of 0° and +12° respectively (20%c damage is presented for consistency with later results).

Considering the damage at 0° incidence, a laminar separation bubble on the upper wing surface was located between 50%c and 65%c. This was a consistent characteristic of the aerofoil in both the undamaged and damaged state at this incidence. The separation bubble was seen in the mid-chord region up to +2°, before moving rapidly forward to give a short leading edge laminar separation bubble by +4°.

Here at 0°, as the oncoming flow approached the damaged leading edge, it was brought rapidly to rest before 'spilling' over onto the upper and lower surfaces. The resulting wake remained fully attached to the surface as far as the trailing edge. The wake can clearly be seen to have ‘cut’ through the laminar separation bubble where it occurred. The spanwise extent of the flow disruption was clearly visible, and by the trailing edge the wake’s span was approximately 2.1D (D = damage diameter) across.

The above characteristics were observed for positive lift incidences up to +8°. However, from +10° onwards, surface flow patterns changed characteristics. In these cases, the wakes were dominated by large regions of separation and reverse flow (see Figure 3.10(b)). Immediately behind the damage location, a region of flow separation was observed. Beyond this to the trailing edge, a region of significant reverse flow developed, i.e. flow moving towards the leading edge. This region expanded in a spanwise direction becoming more ‘triangular’ in shape, and by +10° the wake width was 4.1D at the trailing edge. Either side of the reverse flow region ‘flow dividing lines’ were seen. These indicated where the vortices generated at the leading edge (and seen in the smoke visualisation) interacted with the surface flow and showed the spanwise extent of the reverse-flow region. As both dividing-lines extended rearwards, they ended with two contra-rotating vortices on the surface close to the trailing edge, generated where the vortices detached from the wing (as observed in the smoke tests).
Figure 3.10 Surface flow characteristics (20% L.E. damage) ReyNo.=500,000
The strength of the reverse flow and associated vortices were observed to increase with incidence up to the maximum test incidence of +14°. Note that the undamaged wing at +14° was seen to be in the advanced stages of stall, with significant separation extending rearwards from the leading edge. However, with damage present, the flow outboard of the wake did not show any indication of stall-inducing separation. It would appear that the wake entrained the outboard surface flow, and thereby delayed separation.

**Variation of surface flow with damage size.**

By varying the diameters of the leading edge damage it was possible to examine variations in the wake due to damage size alone, i.e. at similar incidence values. Figures 3.11 (a) to (d) illustrate all four damage size results, +2° incidence is presented as an example case.

In general, the results obtained indicated that the basic characteristics of the damage wake remained similar for all damage sizes tested, although the width of the damage wake increased in line with the increase in damage diameter. It was also noted that as damage diameter was increased, a region of separation leading to reverse flow developed along the wake centreline. These observations may be explained as follows; Given that each vortex core was centred at the point where the damage intersected the leading edge, and each vortex defined the outer edges of the damage wake, it would be expected that the wake-width would increase with damage diameter. As the distance between the vortices increased due to damage size increments, so the region of separation developed into one of reverse flow along the wake centreline, as seen in Figures 3.11 (c) and (d).

### 3.3.4 Leading Edge Damage Effects on Aerodynamic Coefficients

Lift, drag and pitching moment characteristics are illustrated in Figures 3.12 to 3.14. \( C_L \) values for the 10%c damage case showed very little variation from the undamaged case up to +8°. However beyond this, damaged \( C_L \) values were greater than those of the undamaged wing. At +10° the measured increase was 0.0167, which was approximately four times greater than the previously identified lift coefficient error band. Thus, this was a real effect, which may be explained by the stall-delaying effect of the vortex pair on the upper surface flow.
Figure 3.11 Upper Surface Flow Visualisation (Leading Edge Damage, all +2deg)
Reynolds Number = 500 000
The effects of 20%c, 30%c and 40%c damage were only a slight reduction in lift curve slope at low incidences. This tied in with the flow visualisation results, where the damage wake remained attached to the surface. At higher incidences lift losses were more pronounced. In all three cases, $C_L$ values indicated the trend of 'premature stall' followed by a recovery in lift curve slope, up to the maximum incidence tested. For 20%c damage, this occurred around $+10^\circ$, which corresponds with the flow visualisation results, where the wake detached and resulted in significant separation. As incidence was then increased further, the entrainment of the flow outboard of the wake resulted in the subsequent recovery in lift curve slope. The incidence at which 'premature stall' was seen decreased to $+8^\circ$ for 30%c and 40%c damage. The latter resulting in the greatest reduction in lift curve slope, before subsequent lift curve slope recovery.

Pitching moment characteristics for the 20%c damage (Figure 3.13) also indicated a significant change from the undamaged characteristics by $+10^\circ$ incidence. This coincided with the "premature stall" noted above. The same effects were seen to occur for the other damage sizes tested, although as size was increased, so the magnitude of the effects increased and the incidence at which it was seen decreased. The 30%c and 40%c results indicated that pitching moment was more sensitive to large scale leading edge damage, with significant differences from the undamaged case occurring at lower incidences.

Finally, drag coefficient results are illustrated by Figure 3.14. It can be seen that the minimum drag value increased with damage size. This was probably as a result of normal pressure drag on the frontal area of the damaged leading edge, i.e. as damage size increased so the frontal area of the damage also increased. Also it was seen that as $C_L$ was increased, so the $C_d$ penalty also increased dramatically. This can be explained by the increasing amounts of upper surface area affected by the wake separation region.
Fig 3.12 Leading Edge Damage Lift Loss (10%c to 40%c)

Reynolds Number = 500,000
Fig 3.13 Leading Edge Damage Pitching Moment Changes (10%c to 40%c)

Reynolds Number = 500,000
Fig 3.14 Leading Edge Damage Drag Increments (10% c to 40% c)

Reynolds Number = 500,000

\[
C_d \quad \alpha
\]

\[
C_L \quad \text{Undamaged} \quad 10\% c \quad 20\% c \quad 30\% c \quad 40\% c
\]

-1 -0.5 0 0.5 1 1.5
3.4 Mid-Chord Damage; A qualitative & Quantitative Investigation

3.4.1 Damage Cases Investigated

This section considers the effects of simulated gunfire-damage located at the quarter and half chord positions. In this context, both locations will be termed ‘mid-chord’ damage. The results showed similar flow characteristics and aerodynamic coefficient trends for both locations. The following text will concentrate on quarter-chord damage results to illustrate these effects.

3.4.2 Flow Structures Identified by Flow-Visualisation Techniques

The flow visualisation techniques used in the investigation identified two different forms of flow structure, both of which result from flow through the simulated gunfire damage hole. In the following sections, the terms ‘weak-jet’ and ‘strong-jet’ will be used to identify these two flow structures. This use of the description ‘jet’ is adopted from investigations into Jets-in-Crossflows, a subject with which comparisons will be discussed later in this chapter.

3.4.2.1 Weak-Jet Flow Observations

Smoke visualisation of 20% quarter-chord damage, when viewed from the side, indicated the following. At near-zero lift, i.e. -2°, negligible flow through the damage hole was observed. At incidences other than that giving near-zero lift, flow through the damage was visibly issuing from the surface (Figure 3.15). Over the positive lift incidence range, the flow direction was from the lower to upper surface. When generating negative lift, the through flow direction was reversed. The characteristics of the ‘weak jet’ were that the flow exited at the rear edge of the hole, and was immediately bent over. Given the lack of penetration into the upper-surface flow, the jet through the damage appeared to have a velocity component normal to the chord significantly less than that of the freestream. The bent-over flow, becoming the damage wake, was then seen to remain attached to the upper surface. The direction of the surface flow was confirmed as being in the freestream direction by the injection of smoke into the wake from the upper surface. Figure 3.16 gives a schematic representation of the flow structure identified.
When viewed from directly downstream (Figure 3.17), the weak-jet was "encapsulated" by the freestream flow, which was displaced away from the surface. The wake was seen to be small, both in terms of 'vertical height' and 'spanwise width'. At this incidence the 'vertical height' of the wake was approximately 23% of its 'spanwise width'.

Surface flow-visualisation indicated that the surface flow patterns were similar for all weak-jet cases. Figure 3.18, taken at +4°, indicates clearly the typical surface flow features observed; firstly two lines of flow separation, known here as the 'forward' and 'secondary' separation lines. These can be used to determine the extent of the through-damage flow over the upper surface.
Figure 3.17 Rear View Smoke Visualisation (20% c/4, 0°)

Figure 3.18 'Weak-jet' Surface Flow Visualisation (10% c/4, 0°)
At the point where the surface flow meets the jet an adverse pressure gradient was created, causing separation ahead of the damage hole, i.e. the forward separation line. Between this and the front of the damage hole lay; firstly a vortex, located between the forward and secondary separation lines. This was seen to wrap around to form a ‘horse-shoe’ vortex (terminology as used by jets-in-crossflow investigations). Secondly, a region of ‘upstream reverse-flow’ where flow through the damage (at a greater pressure relative to the surface flow), pushed upstream from the damage hole. Flow in this region was subsequently entrained rearwards, and added to the damage wake.

At the rear edge of the damage hole were two vortex centres observed to be contra-rotating, these were indicated by the white concentrations of paint. The exact position of the vortex centres varied with jet strength; moving forward around the damage edge with increased incidence, but always remaining symmetrical about the hole centreline.

Downstream of the damage, the wake clearly cut through the laminar separation bubble (as seen with leading edge damage). Complementing the smoke tests above, the surface flow patterns also showed that the weak-jet wake remained attached to the surface, moving in the freestream direction. Detailed experimental observations indicated that a velocity gradient existed across the wake from the centreline outwards, such that the outer edge’s velocity was greater than that of the centre (this was found in all the damage sizes tested, where the jet flow was seen to be weak, i.e. attached to the surface). Figure 3.19 illustrates the above weak-jet flow characteristics.

The above observations were for 10%c quarter-chord damage. For this case, it was found that the weak jet characteristics were observed on the lower surface throughout the negative lift range, -10° to -2°, and from -2° to +6° in the positive lift range. Moving to 20%c quarter-chord damage cases, it was found that; (i) the incidence range over which weak-jet characteristics were observed was reduced to -4° to 0°, and (ii) the surface flow structures observed remained similar to those seen for 10%c damage. Increasing damage size to 30%c and 40%c resulted in no further observations of weak-jet characteristics.
3.4.2.2 Strong-Jet Flow Observations

Smoke visualisation of 20% quarter-chord damage, indicated that from 0° upwards different jet characteristics were observed. These will be referred to as ‘strong-jet’ characteristics. The jet was no longer immediately bent over on exiting the hole, but instead, penetrated further into the flow above the upper surface (Figure 3.20). This increased penetration into the freestream flow resulting in the jet ‘detaching’ from the surface, causing a separated region. This region existed between the jet and upper wing surface, extending from immediately behind the damage to the trailing edge. Within this region the flow was highly three dimensional with significant reverse flow within the wake, as illustrated by injecting smoke from the upper surface in Figure 3.20. Figure 3.21 illustrates the flow mechanisms identified from detail observations.

When viewed from downstream, the size of the strong-jet wake was seen to increase between 0° and +4°, with the ‘vertical height’ increasing at a faster rate than the ‘spanwise width’, becoming approximately semi-circular in shape by +4°. Beyond +4°, the wake continued to increase in size, but remained semi-circular in shape. By +10° the damage jet became visible through the obscuring smoke within the wake (Figure 3.22), and by +12° the jet was more discernible. From the downstream viewing point, the jet was seen to ‘fan-out’ in a spanwise
Surface flow-visualisation indicated that the strong-jet cases had some similar surface flow patterns to those seen for weak-jets. Figure 3.23 illustrates the 10\%c quarter-chord surface flow pattern at +10° incidence. The main surface flow features were as follows. Firstly, both forward and secondary separation lines were again seen upstream of the damage. Although the surface flow immediately upstream of the damage was turbulent, the separation lines showed little change in position, relative to the damage, from those of the weak-jets. (Figure 3.24 illustrates the forward-flow from the damage leading edge using smoke, for a strong-jet
Figure 3.22 Rear View Smoke Visualisation (20%c c/4, +10°)

Figure 3.23 ‘Strong-jet’ Surface Flow Visualisation (10%c c/4, +10°)
case. The smoke within the jet can clearly be seen forward of the damage hole.) Moving rearwards, in Fig 3.23, the track of the separation lines and the associated horse-shoe vortex increased in a spanwise direction. This significant increase in wake size indicated the extent of the large region of reverse flow beneath the detached strong-jet.

Figure 3.24  Forward Separation of Oncoming Flow (20%c c/4, +6°)

Figure 3.23 also showed that the strong-jet characteristics included the two centres of the contra-rotating vortices located at the damage edge. However, these had moved further forwards around the edge of the damage from the previous weak-jet locations. The dividing line between wake and undisturbed wing surface flow ended with two large contra-rotating vortices located close to the trailing edge. From the observations and flow patterns photographed, it appeared that the wake entrained fluid from the surrounding upper surface flow, and around the trailing edge from the lower wing surface. The entrainment of upper surface undisturbed flow into the wake appeared to reduce the onset of trailing edge separation, which had been seen on the undamaged wing.

Approximately 5%c to 10%c behind the damage was seen a small region of sluggish flow, however, the remainder of the wake clearly shows significant reverse flow, confirming the
smoke results illustrated by Figure 3.20. Detailed observations indicated that the reverse velocity was greatest at the trailing edge. As the surface flow was entrained forwards from the trailing edge, the velocity component along the surface was seen to significantly reduce, which suggests the presence of an adverse pressure gradient. Figure 3.25 illustrates the strong-jet characteristics discussed above.

![Diagram of flow characteristics](image)

**Figure 3.25 'Strong-jet' Flow Characteristics**

The above observations were for 10%\(c\) quarter-chord damage, where strong-jet characteristics were only observed at incidences of \(+8^\circ\) and greater. Increasing the quarter-chord damage size to 20%\(c\), strong-jet characteristics were then seen in the range \(-10^\circ\) to \(-6^\circ\), and from \(+2^\circ\) onwards. In terms of the actual surface flow characteristics, the 20%\(c\) damage 'strong-jet' flow patterns (Figure 3.26) remained similar to those seen for 10%\(c\) damage, although the separation region immediately behind the damage increased in size, at the expense of a smaller region of reverse flow.

For 30%\(c\) and 40%\(c\) damage, only strong-jet characteristics were seen over the incidence range tested. With the exception of near zero lift incidence, where no jet was seen due to the lack of flow through the damage.
3.4.3 Half-Chord Damage Flow Structure Characteristics

Flow visualisation techniques undertaken on half-chord damage cases indicated similar strong and weak-jet flow characteristics to those identified for quarter-chord damage. However, it was noted that the incidence range over which the 10%\(c\) half-chord weak-jet was observed was slightly less than that seen for the quarter-chord results, being \(-8^\circ\) to \(+4^\circ\). Outside this range, strong-jet characteristics were observed. It is believed that the relatively slower surface velocity at the half-chord location resulted in an increased ratio of jet through-flow to surface velocity, which in turn increased the propensity of a strong jet occurring. Increasing the half-chord damage to 20%\(c\) resulted in strong-jet characteristics throughout the incidence range tested.
3.4.4 Surface Pressure Data

3.4.4.1 Damage Centreline Pressure Data

The wing centreline pressure coefficient, $C_p$, profiles for both the undamaged and $10\% c/4$ damaged upper and lower surfaces at $-2^\circ$ are illustrated by Figure 3.27. Force measurements indicated that $-2^\circ$ was near the zero lift case. The gap in the damage results was where the damage hole cut through the surface, hence there were no centreline $C_p$ values measured at this point. It was seen that there was very little difference between the majority of the two surface profiles. For the damaged case, on the upper surface there was a slight reduction in magnitude of the upstream negative $C_p$, becoming more significant just prior to the forward edge of the hole, whilst the rear showed a reduction of similar magnitude in the local $C_p$ value. The lack of any significant effect confirms that there was negligible flow through the damage, and the $C_p$ variations at the upper edges resulted from a relatively small amount of flow being entrained into the damage and internal cavity. As this flow re-emerged, it was then accelerated around the hole edge.

Figure 3.28 illustrates the pressure distributions at $0^\circ$. It can be seen that the characteristics identified above, at $+2^\circ$, became more pronounced, with the upper surface $C_p$ in the region forward of the damage showing a greater reduction in negative value.

The pressure difference at the damage location between undamaged upper and lower wing surfaces suggests that with damage present, air would flow through the hole from lower to upper surface. The change in damaged lower surface $C_p$ at the forward hole edge resulted from the surface flow accelerating as it was drawn around the edge and through the hole. At the exit hole, the 'blockage' effect of the emerging through-flow gave a positive pressure increment as the surface flow was retarded (as seen in the earlier flow visualisation), hence the reduction in negative $C_p$ observed. As the flow exited the hole, flow visualisation indicated that it was bent over immediately, thus the localised increase in negative $C_p$ at the rear edge as the flow was accelerated around it. The subsequent recovery of $C_p$ in the latter 60% of the chord length reflects the flow returning to undisturbed surface flow velocities.

The pressure distributions continued to reflect the above characteristics as incidence was
Figure 3.27 \(c/4\) 10\%c Pressure Profile (Mid Span); \(-2\) alpha

![Figure 3.27 Pressure Profile](image-url)
Figure 3.28  c/4 10%c Pressure Profile (Mid Span) ; 0 alpha
Figure 3.29  c/4 10%c Pressure Profile (Mid Span) ; +6 alpha
increased up to +6° (Figure 3.29). However, at this incidence the negative pressure peak in
the wake had moved progressively rearwards from the damage rear edge, by approximately
10%c. Continuing towards the trailing edge, the damaged $C_p$ profile returned to
approximately the undamaged values. This was consistent with both the smoke and surface
flow visualisation results, which indicated that for these test conditions, the damage jet
remained attached up to +6°.

When increased to +8°, the upper surface continued to show reductions in negative $C_p$
between the leading edge and damage. However, different $C_p$ characteristics were now seen
within the wake.

Downstream of the damage, $C_p$ values peaked at around the 45%c position (but at less
negative values than seen for +6° results). Also, the profile no longer returned to undamaged
values by the trailing edge, see Figure 3.30. This was consistent with the flow visualisation,
which indicated that the jet had changed from the weak to strong type by +8°. As incidence
was further increased to +10°, the effects of separation on the $C_p$ profile continued, with the
maximum negative $C_p$ values moving rearwards to the 58%c position, as seen in Figure 3.31.
Note that with the strong-jet wake separation after the maximum value, $C_p$ values remained at
significantly greater negative values than for the undamaged case.

Considering the lower surface, throughout the undamaged positive lift range, the damaged $C_p$
values were similar to those of the undamaged values. Whilst the leading edge stagnation line
on the lower surface did not show any indication of moving, the subsequent pressure recovery
was more rapid as the surface flow was drawn into, and accelerated through the damage. The
rear edge of the damage showed a consistent localised positive $C_p$ increase. This indicated the
presence of a stagnation line located at the rear edge of the damage. This was the dividing
point between flow going through the damage and flow going over the remainder of the
lower surface. For the flow continuing over the lower surface, the subsequent $C_p$ profile
indicated less positive values. This may be attributed to the upper surface reverse flow
entraining fluid from the lower surface around the trailing edge and back onto the upper
surface.
Figure 3.30  c/4 10%c Pressure Profile (Mid Span); +8 alpha
Figure 3.31  c/4 10% c Pressure Profile (Mid Span) ; +10 alpha
Over the negative incidence range from zero lift up to the point of negative stall, the flow direction through the damage was reversed, and hence the upper and lower surface trends were also reversed. Throughout the range tested, the lower surface damaged $C_p$ profiles were only those seen for the weak jet. No strong-jet with separation and reverse flow was observed, even up to the negative stall, confirming the flow-visualisation results.

Enlarging the quarter-chord damage to $20\%c$, indicated similar weak and strong-jet $C_p$ characteristics to those identified for the $10\%c$ damage. However, as indicated by the $20\%c$ damage flow-visualisation, the incidence at which the $20\%c$ strong-jet characteristics were first observed, was seen to have reduced to $+2^\circ$ in the positive, and $-6^\circ$ in the negative, lift range.

3.4.4.2 Damage Spanwise Pressure Data

$C_p$ Pressure Profile Graphs

Direct plotting of the five spanwise $C_p$ profiles illustrated the relative change in $C_p$ values from the undamaged case at each of the spanwise stations. Figures 3.32 and 3.33 illustrate all five sets of $c/4$ $20\%c$ damage data, together with the undamaged profile, at incidences of $-2^\circ$ and $+8^\circ$ respectively. Spanwise data were only obtained for the $20\%c$ damage case.

At $-2^\circ$, the near zero-lift incidence, a direct comparison of the results indicated that all five profiles showed little variation from the undamaged case over the majority of the chord. The exception was seen at the centreline and $0.5r$ stations, just downstream of the damage where the previously identified negative $C_p$ peak occurred on both surfaces. As these results are for $20\%c$ damage, so slightly more flow may be entrained within the hole, resulting in slightly greater $C_p$ peaks than seen earlier for the $10\%c$ damage in Figure 3.27. The peaks seen on both surfaces died out in the chordwise direction within approximately $5\%c$ of the hole rear edge. In the spanwise direction, both peaks again decayed rapidly between the centreline ($0r$) and $0.5r$ stations. By $1.5r$ both upper and lower surface profiles had returned to approximately the undamaged state.

Considering the $+8^\circ$ results (Figure 3.33), all spanwise profiles showed features consistent with both the significant reductions in upper-surface pressure peak suction, and the
Figure 3.32 20% c/4 Spanwise Pressure Profile; -2 alpha

![Graph showing pressure distribution along the spanwise direction with labels for Undamaged, CL, 0.5r, 1.5r, 2.5r, and 5.0r.]
Figure 3.33 20% c/4 Spanwise Pressure Profile; +8 alpha
development of the damage wake region.

The reductions in upper surface peak suction varied with spanwise displacement from the damage. Both centreline and 0.5\( r \) \( C_p \) profiles were similar and reflected the pressure recovery resulting from flow through the hole. At spanwise distances of 1.5\( r \) to 5\( r \), both upper and lower \( C_p \) profiles were seen to recover towards the undamaged case, however, within this spanwise region neither profile recovered fully. For example, on the upper surface at a tapping location of 26\( \% c \), it was seen that at the 5\( r \) station the damaged \( C_p \) was still 0.1889 less than its corresponding undamaged centreline value. It is important to note that this discrepancy was not an effect of model spanwise variation in the undamaged profiles. As seen in Chapter 2, the undamaged \( C_p \) profile at this 5\( r \) location (26\( \% c \)) was found to have a difference of only 0.0118 from the corresponding centreline value. The evidence of an effect up to 5\( r \) suggests that the spanwise effects of the damage are significantly greater than that suggested by flow-visualisation alone.

**Upper & Lower Surface \( C_p \) Contours**

Whilst illustrating the \( C_p \) data by a series of chordwise profile plots may reflect certain aspects of the pressure field, this form of representation does not adequately reflect the three dimensional effects. In order to achieve this, the spanwise data are presented in the form of pressure contours. Data from the matrix of upper and lower surface pressure tappings were used to generate contour lines of locations with the same \( C_p \) value (contour intervals were 0.05). This was done by linear interpolation calculations, performed using the \( P_c \)-based data processing package ‘MATLAB’ (Ref. 34). The contour plots were then superimposed on top of digitised flow-visualisation photographs. Note that in the immediate spanwise vicinity of the damage, pressure data were not available between the centreline (0\( r \)) and 1.5\( r \).

The first incidence to be considered was that of near-zero lift, -2° (Figure 3.34). From the surface flow visualisation it appeared that significant surface flow disruption was present, extending rearwards from the hole to the trailing edge. However, it was found that the \( C_p \) field disruption was limited and localised. Immediately downstream, of the rear edge of the damage the pressure peak was seen as a close grouping of contours. However, beyond this, the peak soon died out. Only a slight variation in \( C_p \) values occurred upstream of the damage.
Figure 3.34 Upper Surface (20% c/4, -2deg.)

Contour increment
$C_p = 0.05$
Figure 3.35 illustrates the corresponding lower surface flow-visualisation and $C_p$ pressure distribution. Here, there was slightly greater deflection of the pressure contours around the damage location than seen in the upper surface case. Although again, the $C_p$ variation in the damage wake remained limited, and the identifiable spanwise extent corresponded well with the spanwise extent of the visualised disruption.

Considering now the upper surface strong jet at $+4^\circ$ incidence, Figure 3.36. It can clearly be seen that there was significant $C_p$ disruption over the entire pressure tapped region. The contours were no longer parallel with the leading and trailing edges at 5r, indicating that the influence of the damage on the upper pressure field extended beyond this spanwise-station.

The 'separation line' was previously suggested as the line indicating where flow from the leading edge met through-flow from the damage. Such separation would be expected to occur along a line of constant $C_p$. This was seen to be the case in these results immediately upstream of the damage, see point A Figure 3.36. It is believed that the constant $C_p$ contours did continue along the separation lines towards the separation bubble location, although this was not reflected in the $C_p$ contours plotted. The 'jagged' $C_p$ contours are believed to be misleading, and due to the limitations of the tapping matrix resolution and the linear interpolation of the data. Considering further down-stream, the $C_p$ field did not indicate any pressure discontinuity between the damage wake region and the outer non-wake region (Fig 3.36, B). Instead, the $C_p$ contours appeared re-orientated at ninety degrees to the interface between the two regions.

At $+4^\circ$ the surface flow characteristics indicated that a 'strong jet' effect was present. As explained earlier, the centre-line $C_p$ data indicated that the resulting separated wake contained a 'less pronounced' negative peak. This peak was seen to develop some 15-20%c downstream of the damage rear edge, and moved rearwards as incidence was increased. Now, using the pressure contours, it was seen (Fig. 3.36, point C) that at $+4^\circ$ the peak region was approximately circular in shape, the width of the wake, and centred on the damage centreline.

With the incidence increased to $+8^\circ$ (Figure 3.37), the influence of the damage on the flow can again be seen to extend well beyond the region covered by the pressure tapping matrix.
Figure 3.35 Lower Surface (20% c/4, -2deg.)

Contour increment
Cp = 0.05
Figure 3.36 Upper Surface (20% c/4, +4deg.)
Again pressure contours were seen to follow the general direction of the forward separation line (Fig. 3.37, region D). Additionally, the centre of the negative peak within the separated wake not only moved rearwards, but also divided in two, each moving outwards in a spanwise direction (point E). From inspection of similar plots at all incidences, it appears that there was a close similarity between the position of the two vortex centres as indicated by the surface flow visualisation, and the two indicated $C_p$ pressure peaks. Where differences between flow-vis and pressure centres were indicated, this may have resulted from the limited pressure tapping resolution and use of linear interpolation. Point F indicates the separation point, within the damage wake, where the reverse surface-flow (identified by smoke-tests) separated as a result of the adverse pressure gradient.

Figure 3.38 illustrates results for the lower surface $C_p$ field, here at $+8^\circ$ incidence, where again the contours were seen to ‘deflect’ around the damage hole. It was seen that the deflection of surface flow into, and through, the damage had less of an effect on the lower than upper-surface $C_p$ field.

Finally, figure 3.39 indicated that with flow passing through the damage and onto the lower surface at $-6^\circ$, the strong-jet flow features identified on the upper surface can now be seen on the lower surface at negative incidences. The contours were again deflected around the damage (point G), and a ‘peak’ developed within the damage wake (point H). Again $C_p$ results indicated that disturbances extended beyond that indicated by flow-visualisation and the $5r$ spanwise station.

**Overall Changes in Wing Pressure Field Due to Damage**

Denoting the upper surface pressure coefficient at a given chordwise & spanwise location as $C_{p\text{ Upper}}$ and the corresponding lower surface pressure coefficient as $C_{p\text{ Lower}}$, the pressure coefficient differential, $\Delta C_p$, was calculated as:

$$\Delta C_p = C_{p\text{ Upper}} - C_{p\text{ Lower}}$$

Now, $\Delta C_p$ values can be found for both the damaged and undamaged wing at the same wing
Figure 3.37 Upper Surface (20%c c/4, +8deg.)
Figure 3.38 Lower Surface (20%c c/4, +8deg.)
Leading Edge

Figure 3.39 Lower Surface (20% c/4, -6deg.)

Contour increment
Cp = 0.05
location. By subtracting the undamaged $\Delta C_p$ value from the damaged value, the change in pressure coefficient differential, $\Delta C_p$, due to the presence of damage, was calculated:

$$
d[\Delta C_p] = \Delta C_p \text{Damaged} - \Delta C_p \text{Undamaged}
$$

A $d[\Delta C_p]$ value of zero indicates that the damage caused no change in the overall pressure field acting on the wing at the given point. The experimental data were then used to generate contours of $d[\Delta C_p]$ over the wing.

Figure 3.40 illustrates the general findings of such $d[\Delta C_p]$ contour plots for the wing with 20% $c/4$ damage. In general it was seen that two distinct regions could be identified, one with positive and the other with negative changes in the $C_p$ field acting on the wing. Here at $+8^\circ$, the positive region extended from the leading edge over the first 60%$c$, and was where $\Delta C_p$ values were less negative, i.e. lift loss occurred. Beyond this, and extending as far as the trailing edge, the negative region showed $\Delta C_p$ values being more negative, i.e. generating greater lift (compared with the undamaged wing). Also to be noted was the rapid variation in leading edge pressure field. This occurred as a result of the difference between the undamaged and damaged leading edge pressure profiles in the first 5%$c$.

![Figure 3.40 d[\Delta C_p] Contours Over Wing Surfaces (+8°)](image-url)
3.4.5 A Comparison of Wing and Flat-plate Jet Mechanics

From the above wing damage results, it was seen that a form of jet developed on the suction side of the wing as a result of the flow through the damage hole. Considering the upper wing surface only, it may be suggested that the wing characteristics observed were similar in nature to those identified for a flat-plate ‘jet in crossflow’. In such a situation a jet emerges from a flat-plate into an oncoming free-stream flow. Whilst accepting that the forces driving the two systems are very different, a number of key characteristics were found to be similar. The following text will firstly outline published findings for flat-plate jet-in-crossflow tests, before drawing a comparison with observations of the damaged-wing jet effects.

3.4.5.1 Published Characteristics of Jets-in-Crossflows

Extensive research into the characteristics of jets in cross-flows has been undertaken for many years, mainly with a view to VSTOL (Vertical and Short Take-Off and Landing) aircraft applications. In experimental tests a uniform jet exits a nozzle on an infinite plate into a uniform cross flow. The velocity of the jet may be greater or less than that of the crossflow.

Given a typical experimental set-up (Figure 3.41), a number of key variables are apparent. The most important are; (i) the ratio of jet velocity to free-stream velocity, \( R = \frac{V_{jet}}{V_{freestream}} \), (ii) the jet angle relative to the flat plate, and (iii) the jet nozzle shape. The majority of the flat-plate results considered here are limited to circular crossflow jets, emerging at

![Figure 3.41 Typical 'Jet in Cross-flow' Experimental Set-up](image-url)
different R values and jet angles. It was assumed in such tests that the flat plate was of infinite size, and that the jet flow had a uniform velocity within the pipe.

In general, published data relates to tests where the velocity of the jet was significantly greater than that of the cross flow, as found in VSTOL applications. It has been found that the flow characteristics of such tests with ‘high jet ratios’ (typically R ≥ 2), was significantly different from flow at ‘low velocity ratios’ (typically R ≤ 0.5). Experimental results for both cases are summarised in Ref. 35, and it is from this source that the following summary was compiled;

... the complicated nature of the jet in a crossflow is illustrated in Figures 3.42 (a) and (b), where the composite pictures of the flow development are presented for the velocity ratios R=0.5 and 2 respectively.

The most obvious feature of the jet in crossflow is the mutual deflection of both jet and crossflow. The jet is bent over by the cross-stream, while the latter is deflected as if it were blocked by a rigid obstacle, the difference being that the jet interacts with the deflected flow and entrains fluid from it. In the case of the small velocity ratio (R=0.5), the flow behaves as if a partial, inclined ‘cover’ were put over the front part of the exit hole, causing the jet streamlines to start bending while still in the discharge tube and the jet to bend over completely right above the exit and also to lift up the oncoming flow over the bent-over jet. In the case of the higher velocity ratio (R=2) the jet is only weakly affected near the exit and penetrates into the cross-stream before it is bent over. In both cases, wake regions with very complex three-dimensional flow patterns form in the lee of the jet. In these regions, the longitudinal velocity accelerates and the conservation of mass requires fluid to move from the sides towards the plane of symmetry. Very close to the wall a reverse-flow region forms, and cross-stream fluid has been observed to enter this region, travel upstream and then to be lifted upwards by the jet fluid and to be carried downstream together with it. Unlike the flow in two-dimensional situations, the flow does not recirculate in this highly three-dimensional case.

The vorticity characteristics of a jet in crossflow will now be discussed. An important feature of this flow is the deflection of the streamlines in the x- and z- directions and the associated
reorientation and generation of vorticity. Particularly striking is the presence of streamwise vorticity downstream of the exit, which is contained in the secondary motion formed by two counter-rotating vortices and gives the bent-over jet a kidney shape. Considering the vorticity of the approach flow ensuing from its interaction with the jet. New vorticity is generated at the interface between the jet and crosstream. The vortex lines are then bent around the jet and form a horseshoe vortex similar to that found when a boundary layer is
deflected around an obstacle, for example by a cylinder mounted on a flat plate. In addition, the oncoming boundary layer separates upstream of the jet as indicated in Figures 3.42 (a) and (b).

The vorticity present causes most of the entrainment of cross-stream fluid into the wake region and into the deflected jet. There is also some evidence given by Ref. 36, that the horseshoe vortex is swept into the 'hollow' of the kidney-shaped jet field underneath the bound vortex.

Further explanation of the jet development from circular to the kidney-shape was given by Reference 37, which cited the commonly used explanation of potential flow pressure distributions around a rigid circular cylinder as a driving mechanism. This explanation states that for a circular jet, the potential flow pressure distribution (close to the nozzle exit) may be compared to that around a rigid circular cylinder. For such potential flow, stagnation points \( (C_p=1) \) are located upstream \( (\theta=0^\circ) \) and downstream \( (\theta=180^\circ) \) and minimum pressures at the lateral edges \( (\theta=90^\circ \) and \( 270^\circ) \). Consequently, a jet would be expected to spread laterally into an oval shape. Taken together with the crossflow shearing along the lateral edges, the jet develops a kidney shaped cross-section. Moving downstream the shearing forces develop and drive a vortex pair within the jet.

The penetration of circular jet paths into cross-flows has been investigated many times (including Refs. 38,39,40,41,42), Ref. 42 gives results from tests with jet velocity ratios in the range 2 to 10, reproduced here as Figure 3.43. This reference also noted that at low values of \( R_j \) (i) the pressure field at the jet exit resulted in flow angle deflection towards the rear of the nozzle exit, and (ii) the jet appeared to remain attached to the plate surface upon exiting the nozzle. This attachment of the bent over jet was also noted in Ref. 43.

Flow visualisation tests (Ref. 44) illustrated the flow structure present both ahead and in the wake of a rectangular jet into crossflow. Figures 3.44 (a) and (b) are reproduced from this source. Although jet angles are 90° and 45°, both illustrations clearly illustrate similar separation lines to those observed on the damaged wings. Reference 44 stated that these separation lines were the signature of a horseshoe-type vortex system, which were seen at other velocity ratios and jet angles. Two strong contra-rotating vortices were also noted in the
wake of the jet, and that they generated a lower pressure region with reverse flow, into which surrounding freestream fluid was entrained.

At high jet angles, the wake region was clearly dominated by the presence of the two vortices. However, when the jet angle had been decreased, the wake region increased in width and extended considerably in a streamwise direction. Additionally, the strength of the vortices declined, and both were seen to have moved rearwards and outwards immediately behind the jet exit. The distance from jet to primary separation line, normalised with respect to nozzle geometry, was also measured and found to be a function of velocity ratio and jet angle. This distance was found to be insensitive to either the freestream Reynolds Number or jet Reynolds Number.

Pressure distributions on flat plates around the jet exits have been well documented (Refs 45,
of which the most commonly cited publication was Fearn & Weston (Ref. 47). The accepted method of presenting such results was to subtract the jet-off pressures from the jet-on pressures to give the increment due to the jet as a pressure coefficient \( C_p \):

\[
C_p = C_{p\text{-jet-on}} - C_{p\text{-jet-off}}
\]

The results showed that for flat plates there are generally three identifiable regions within the pressure distribution, the details of which depend on the value of the jet velocity ratio \( R \):

- **Upstream of the jet**, where ‘blockage’ of the freestream flow occurred, a positive pressure region resulted.
- **The second region** was generally located either side of the jet, and exhibited negative pressures resulting from the acceleration of the freestream flow around the jet, and subsequent entrainment.
- **The third region**, was a distortion of the negative pressure region in a rearwards direction, due to the close proximity of the deflected jet downstream of the jet exit. All three regions were found to vary in shape and extent with the value of \( R \). Contour plots of \( C_p \) reflect the change in position of the three regions as the value of ‘\( R \)’ varied. Below \( R \approx 5 \), the plots were known as ‘snail-plots’ (see Fig. 3.45, where \( R = 2.8 \)), as they
...have the shape of a snail where the \( C_p = 0 \) contour is analogous to the antenna, the positive contours represent the head, and the negative contours represent the shell. At greater \( R \) values the shell grows and the antenna droops to the ground as the head becomes smaller." (Ref. 37)

It has been noted above, that the angle at which the jet emerged from the surface was seen to vary with jet velocity ratio, as the freestream bends the jet over. The effect of varying this jet angle has been investigated by References 43, 48 and 49. It was found that on a ‘snail’ plot, as the deflection angle was decreased, so the negative pressure region moved away from the ‘side’ of the nozzle to a position further downstream of the jet exit. Thus for a flat-plate crossflow jet, a clear link was established between the location of the negative pressure region and the relative strength of the jet.

![Figure 3.45 Contours of constant \( C_p \), \( R=2.8 \) (Ref. 47)](image)

3.4.5.2 Comparison of Flat-plate and Damaged-wing Jet Effects.

In all the above published tests, the jet emerged from a flat plate into the crossflow. Now, in the case of wing damage-flow, the jet emerged from the wing surface, which was clearly not
a flat plate. However, it is suggested here that the flow characteristics of the damage-jet were similar to those of the flat-plate tests, although modified as a result of: (i) differing driving mechanisms and (ii) the presence of the curved wing surface. This suggestion results from a number of key similarities in the flow mechanisms and pressure fields of both cases.

In the case of both wing and flat-plate jets, flow visualisation indicated that there were similarities in surface flow patterns. Considering Figures 3.18, 3.23, and 3.44; in both cases oncoming surface-flow was seen to separate upstream of the jet, producing ‘separation lines’. Flow from the jet pushed upstream in the region between the jet and secondary separation line. The surface flow was deflected up and over the jet-flow, generating vorticity just behind the forward separation line. The vorticity was then seen to be swept around the jet, forming the horse-shoe shaped vortex, before being entrained into the jet wake.

Considering the general jet flow structures observed for wing-damage (Figures 3.19 & 3.25), there appeared to be similarities with the flat-plate characteristics identified by Foss (Figure 3.42(a) and (b)). In the case of both the wing weak-jet and flat-plate at R<2, the pressure field at the hole exit was seen to influence the jet flow. For both, the jet-flow emerged from the rear of the hole, as if with a ‘partial inclined cover over the front of the exit’, before being bent over immediately and remaining attached to the surface. The resulting wing wake was swept aft, with no significant enlargement. Flat plate R<2 results also indicated relatively small jet wakes. However, although at R=0.5 (Fig. 3.42(a)) some reverse flow was indicated immediately downstream of the hole. As this was not observed for the wing weak-jets, this might indicate that the equivalent R value was less that 0.5.

The characteristics seen for the wing strong-jet were similar to those of the flat-plate for R>2. For both cases, the jets no longer exited at the rear of the hole, but instead penetrated further into the crossflow and remained detached from the surface. In the case of the flat-plate, the jet caused significant reverse flow and two strong contra-rotating vortices (Figure 3.44(b)). For the wing, extensive reverse flow was also seen for the strong-jet wakes together with contra-rotating vortices (Figure 3.23). In both cases the wakes were highly complex 3-D flows entraining freestream fluid, part of which formed the reverse flow which was then lifted up by the jet and carried downstream. The strong-jet’s wake vortices were located further
downstream, possibly as a result of the wing upper surface $C_p$ profile and the presence of the wing trailing edge.

The published flat-plate analyses indicated that the effects of Reynolds Number (based on jet exit conditions or freestream velocity) were secondary to jet velocity ratio $R$. (Ref. 37). This was also found to be the case for the wing damage tests, where no significant variations were observed due to changes in the chord-based Reynolds number (see Section 3.6.1).

Considering the pressure field measurements, it has been seen that flat-plate results were characterised in terms of ‘snail plots’ (Figure 3.45). The three regions on the plots resulted from blockage induced deceleration and acceleration, together with wake separation. For the wing, the equivalent plots were of $d(C_{p_{\text{upper}}})$, which was the change in upper-surface pressure coefficients as a result of damage, and calculated by:

$$d(C_{p_{\text{upper}}}) = (C_{p_{\text{upper}}})_{\text{Damaged}} - (C_{p_{\text{upper}}})_{\text{Undamaged}}$$

Figure 3.46, illustrates a contour plot of $d(C_{p_{\text{upper}}})$ values, for 20%$c$ quarter-chord damage at $+8^\circ$. It was found that similar forms of ‘snail plots’ were not produced by changes in the wing upper surface pressure distribution. The $C_p = 0$ contour did not intercept the damage hole, but was instead located at approximately the 56%$c$ position, running roughly parallel to the
leading edge. Forward of this line, upper-surface negative pressure losses were indicated, reflecting the spanwise pressure relief which occurred along the forward 50%c as a result of the wing being effectively 'punctured'. Whilst aft of the $C_p = 0$ contour, values showed negative pressure increases, i.e. improved upper-surface local suction. Similar characteristics were also reflected in the results obtained at lower incidence values.

In summary, it appears that the interpretation of the presented flow mechanics from smoke and surface visualisation techniques were very similar for both the wing damage and flat-plate jets. However, the surface pressure changes measured were significantly different, for the damage sizes tested and incidence range considered.

3.4.6 Variations in Aerodynamic Coefficients Resulting from Quarter-Chord Damage

For ease of comparison, the format of the results was amended, indicating now the change in wing coefficient ($d[C_L]$, $d[C_d]$ and $d[C_m]$), rather than just the coefficient magnitudes at a given incidence. The values were calculated by:

\[
d[C_L] = C_{L\;\text{damaged}} - C_{L\;\text{undamaged}} \\
d[C_d] = C_{d\;\text{damaged}} - C_{d\;\text{undamaged}} \\
d[C_m] = C_{m\;\text{damaged}} - C_{m\;\text{undamaged}}
\]

Lift changes, $d[C_L]$ (Fig. 3.47), for all quarter-chord damage cases showed reductions in $C_L$ magnitude over the full incidence range (i.e. negative $d[C_L]$ in the positive lift range, and positive $d[C_L]$ in the negative lift range). The 'premature stall' and subsequent recoveries seen for leading edge damage larger than 10%c, were not seen for this quarter-chord damage, instead a more progressive loss of lift resulted from increments in incidence, at all damage sizes.

Considering 10%c lift in the positive lift range up to +6°, a gradual but small incremental trend in loss was seen, remaining less than -0.02. This corresponded with the incidence range identified for the weak-jet characteristics, with minimal flow disruption and the damage jet
remaining attached to the surface. However, from +8° onwards the 10%c damage ‘strong’ jet was observed (see Point A, Fig 3.47). Associated with this form of jet, was a more rapid increase in $d[CL]$ values for +8° and +10°. Correspondingly, the rate of $d[CD]$ change with incidence also increased dramatically (Point B, Fig. 3.48) from +8° onwards, and similarly, the pitch $d[Cm]$ trend was also seen to (Point C, Figure 3.49) give significantly greater negative values from +8° upwards. Over the negative incidence range, all three coefficients showed no sudden changes in damage trends, which coincided with the weak jet flow-characteristics observed throughout.

Increasing the damage size to 20%c, resulted in greater effects on the coefficient values. This was expected due to the larger damage size. It was also found that the transition from weak to strong jet identified by flow visualisation, again coincided with significant changes in coefficient values. Flow visualisation indicated that for 20%c damage; the change between jet types occurred between 0° and +2°. The changes seen in coefficients from +2° onwards (Points X, Y and Z), reflect this and showed a more rapid rate of lift loss and drag increase with incidence.

Noticeably though, with 20%c and greater damage, changes in pitching moment were approximately linear with incidence (up to the onset of stall at +10°). This corresponded to the pressure field changes highlighted previously in Figure 3.40, where $d[ΔC_p]$ values showed surface pressure differential, $ΔC_p$, reductions over the forward section of the wing and increases over the aft. Given that moments were calculated about the quarter-chord location, it was therefore consistent that net negative pitching moment effects were observed.

Thus, in general, trends for all coefficient results ($d[CL]$ , $d[CD]$ and $d[Cm]$) indicated that jet type was an important factor, with differing characteristics for weak and strong jets. The results were seen to have greater magnitude and increased rates of change when strong jets were present. In hollow wing cases, some degree of internal flow would have occurred, giving rise to a number of effects on the coefficient changes. Of primary importance was that part of the through-flow would have circulated within the wing internal box cavity, before emerging from the exit hole. This would have influenced internal pressures, and may have resulted in reduced jet exit angles. At the very least, the through flow would have driven
internal circulation. It may also be expected that these effects would be influenced by the type of jet present, leading in part to the changes observed in $d[C_l]$, $d[C_d]$ and $d[C_m]$ results. Whilst it has not been possible to identify the effects of internal flow, it has been seen that strong jets give rise to significantly increased areas of upper surface separation, which may in part explain the significant increments in $d[C_d]$ values observed in figure 3.48.

It is important to note the observation that incidence ranges for the weak and strong-jets varied with damage size:

<table>
<thead>
<tr>
<th>$c/4$ Damage Jet-type Incidence Ranges</th>
</tr>
</thead>
<tbody>
<tr>
<td>Strong</td>
</tr>
<tr>
<td>--------</td>
</tr>
<tr>
<td>10%c</td>
</tr>
<tr>
<td>20%c</td>
</tr>
<tr>
<td>30%c</td>
</tr>
<tr>
<td>40%c</td>
</tr>
</tbody>
</table>

For a given damage size, it was seen that the transition from a weak to strong jet was dependent on the 'flow-rate' through the hole. The jet was generated by the pressure differential across the hole. Thus, as incidence was increased, so pressure differential increased and hence jet velocity increased (relative to the wing freestream velocity). This equated to the previously discussed flat-plate findings of crossflow jet increments in velocity ratio $R$. Similarly, damage at different locations (at a fixed incidence), resulted in different pressure differentials due to the chordwise variation in pressure coefficient. It was also seen that at a given incidence (e.g. $+4^\circ$ for $c/4$ results), the type of jet could also depend on the size of the damage present.

In the cases of jets-in-crossflow, the jet velocity ratio was an experimental parameter that could be varied by changing either the freestream velocity or the jet velocity. In the crossflow cases discussed, no interdependency existed between these two values. Each one could be set separately from the other, to give the resulting jet characteristics of interest. However, in the case of the damaged wing, the ratio of freestream to through-flow was seen to be a function of incidence, damage location and damage size, which together influenced the type of jet present. The exact nature of this function is undoubtedly complex and worthy of further detailed investigation.
Figure 3.50 Half-chord Damage; d[CL]

Figure 3.51 Half-chord Damage; d[Cd]

Figure 3.52 Half-chord Damage; d[Cm]
3.4.7 Half-Chord Damage Aerodynamic Coefficient Changes

For completeness, the coefficient changes resulting from half-chord damage (10%c to 40%c) have been included, Figures 3.50 to 3.52. Although having smaller magnitudes of effects, the results showed similar trends to those identified for quarter-chord damage. This would have resulted from less of a reduction in the upper surface pressure peak, as the damage was located further away downstream. Again, incidences over which weak and strong jet effects were observed in flow visualisation tests, corresponded with those changes seen in the lift, drag and pitching moment coefficients.
3.5 Trailing Edge Damage; A Qualitative & Quantitative Investigation

3.5.1 Surface Flow Visualisation

Figures 3.53 and 3.55 illustrate the effects of 20% c damage located at the trailing edge position at +2° and +8°. These incidence values were chosen as representative test points.

It was seen that the trailing edge damage had little effect on the wing surface flow characteristics, for damage up to 20% c, over the incidence range tested. At low incidences, the pressure differential between the upper and lower surfaces at the trailing edge was minimal. Consequently, there was little if any through-flow. On reaching the damage, the surface flow separated in the same way as if it had reached the trailing edge (Figure 3.53). As incidence increased, so the flow passing through the damage also increased. The greater through-flow resulted in the deflection of the upper surface flow (similar to that seen for mid-chord damage), which caused premature-separation ahead of the damage forward edge (Figure 3.54). When this happened within the already separated trailing edge flow, as seen at +8°, no significant effects were seen (Figure 3.55). However, when the separation occurred upstream of an enlarged damage hole, the normal undamaged wing separation line was seen to deflect forward. In this situation a curved separation line formed, as seen in Figure 3.56 for

![Figure 3.53 20% c Trailing Edge Damage, Upper Surface (+2°)](image-url)
40%c damage at +8°. In such cases, it was found that the chordwise position of this premature separation was seen to move forward as incidence was increased.

![Diagram](image_url)

**Figure 3.54** Premature Separation Due to Trailing Edge Damage.

![Image](image_url)

**Figure 3.55** 20%c Trailing Edge Damage, Upper Surface (+8°)
3.5.2 Trailing Edge Damage Effects on Aerodynamic Coefficients

Results of wing coefficient tests are seen in Figures 3.57 to 3.59. Firstly, from the results it was seen that the effects of trailing edge damage are significantly less than those of leading edge or mid-chord damage. The minimum loss in lift, \( d[C_L] \), (Figure 3.57) was seen to occur around \(-8^\circ\) for all damage sizes. As incidence was increased from this point, all damage cases showed a progressive increase in lift-loss up to \(+8^\circ\) (the onset of stall). For a given incidence prior to stall, the results clearly indicated that the larger the damage size, the greater the lift loss. However, even with the largest damage size, the resulting lift-losses were significantly less than those for damage of the same size at the leading edge or mid-chord. This was due to; (i) the damage being in a region of low surface pressure differential, and (ii) disruption of the freestream flow took place downstream of the trailing edge, thereby having little effect on the surface flow over the wing.

Figure 3.58 shows that the drag increments \( d[C_d] \) due to the trailing edge damage were significantly less than those seen resulting from damage at the other locations tested. The \( d[C_d] \) increments identified, resulted from a combination of base pressure drag on the forward
Figure 3.57 Trailing Edge Damage; $d[CL]$

Figure 3.58 Trailing Edge Damage; $d[Cd]$

Figure 3.59 Trailing Edge Damage; $d[Cm]$
edge of the damage, and increased separation seen either side of the damage in the earlier flow visualisation photographs.

On the undamaged wing, as incidence was increased beyond +4°, separation was seen to develop from the trailing edge on the upper surface. Associated with this development, was the rapid rise in undamaged $C_d$ values. With the damage located at the trailing edge, as incidence was increased, so the small drag increments caused by the damage were swamped by the normal onset and development of the separation. Hence, the reduction in magnitude of $d[C_d]$ beyond +4° indicated in Figure 3.58. However, as incidence was increased to +10° and beyond, there were significant reductions in $d[C_d]$ as a consequence of the removed surface on which the stalled-flow would otherwise have incurred drag.

Note, that similar characteristics were also identified for the negative incidence stall onset at -8°, which corresponded with the development of rapid lower surface trailing edge separation seen in flow-visualisation at -8°.

Figure 3.59 indicates damage effects on pitching moment, $d[C_m]$. As seen with lift loss, the minimum effects of the damage occurred at -8°. Beyond this, there was a trend for the effects to increase with incidence, up to the onset of stall. The damage resulted in a consistent positive increment in pitching moment. Again pitching moment variation was related to damage size. These findings were consistent with the removal of a portion of trailing edge of this aerofoil section; i.e. given the undamaged pressure profiles, the trailing edge would have been expected to contribute to the overall negative (nose-down) pitching moment. Removing such a section would result in a net decrease in the amount of negative pitching moment generated, as seen in the $d[C_m]$ results.

3.6 Reynolds Number and Internal Modelling Effects

In addition to the main experimental investigation into the effects of gunfire damage, brief assessments were undertaken to firstly identify the influence of Reynolds number on damage characteristics, and secondly determine possible differences as a result of internal wing
construction.

3.6.1 Reynolds Number Influence on Damage Characteristics

The previous sections considered results from tests undertaken at a Reynolds Number of 500,000. In relative terms this is a low Reynolds Number at which to undertake wind-tunnel tests. It was therefore necessary to ascertain the applicability of the results at higher Reynolds Numbers.

By varying the tunnel speed, it was possible to change the Reynolds Number of the test. A value of 500,000 was obtained with the tunnel running at just under maximum velocity. Thus only by reducing the tunnel speed could a significantly different Reynolds Number be obtained. The value was reduced to 250,000. By reducing the value, any possible Reynolds Number dependent effects were more likely to become apparent (due to increased boundary layer thickness).

Figures 3.60 to 3.62 compare the results obtained for 20%e quarter-chord damage at both Reynolds Number values. The data indicated that although some variations occurred, mainly when approaching stall (incidence ≥ +6°), the magnitudes and trends of the incremental effects were largely independent of Reynolds Number over the range tested. The differences observed between the results may be explained by;

(i) At a lower Reynolds number, with the dynamic pressure and hence force magnitudes reduced, so proportionally greater inaccuracies may be expected. This would be equally applicable for both wing force measurements and in the determination of support-strut corrections.

(ii) Slight differences in side-wall ‘leakage’ effects, particularly at high incidence values.

As identified previously, the effects of damage were dependent on the location of the damage and the extent of through-flow. As such, ‘normal’ boundary layer characteristics which are more susceptible to variations in Reynolds Number played a less significant role. As the current tests have indicated no significant variations, it is suggested that results at higher Reynolds Numbers are unlikely to vary significantly from those presented here.
**Figure 3.60 Reynolds Number Influence on d(\text{CL})**  
20\% Quarter Chord Damage

**Figure 3.61 Reynolds Number Influence on d(Cd)**  
20\% Quarter Chord Damage

**Figure 3.62 Reynolds Number Influence on d(Cm)**  
20\% Quarter Chord Damage
3.6.2 Influence of Wing Construction on Damage Characteristics

Typical wind tunnel models are of solid construction. However, as discussed earlier, the models used for the above investigations were hollow in nature to represent the construction of actual aircraft wings. With hollow wings, through-flow may have been expected to drive circulation within the internal cavity structure. Tests were undertaken to ascertain whether internal aerofoil construction influenced $d[C_L]$, $d[C_{d}]$ and $d[C_{m}]$ values.

The results of quarter-chord 40%$c$ gunfire damage cases are illustrated by Figures 3.63 to 3.65. This extreme damage condition was used to illustrate the trends observed throughout the damage cases considered. Lift loss results indicated that in general, $d[C_L]$ trends were similar for both types of construction. However, in general, the magnitude of solid wing $d[C_L]$ values were seen to be greater than those of hollow wings. The differences were seen to become more pronounced at higher incidences, and further increased with damage size. It is suggested that with hollow construction, internal flow may have caused a slight negative $C_p$ increment around the hole on the internal side of the lower surface, and a positive $C_p$ increment on the underside of the upper surface. This would have led to an additional positive net lift increment. As incidence and damage size increased, this would have been expected to become greater. With solid construction, the effect would not have been present and hence the greater negative $d[C_L]$ values measured.

Trends in $d[C_L]$ values observed were similar for both types of internal construction, although solid construction cases were seen to exhibit a $d[C_L]$ increment. This increment appeared approximately constant over incidence range between positive and negative stall conditions. Finally, $d[C_{m}]$ values were found to be similar for both types over the majority of the incidence range, with minimally greater $d[C_{m}]$ values observed for solid tests just prior to stall. Overall, the differences observed between the two types of construction clearly indicated that for accurate damage modelling, wind tunnel models must reflect the true nature of full-scale wing construction. As aircraft wings are primarily hollow in construction, this confirmed the decision to use hollow wings for this investigation, ensuring both the accuracy and realism of the results obtained.
CHAPTER 4. INTERPOLATION TECHNIQUES FOR SIMULATED GUNFIRE DAMAGE.

4.1 Introduction

This chapter outlines predictive interpolation techniques for use in determining the aerodynamic characteristics of the wing with simulated gunfire damage of differing size and location to that tested experimentally. The three different ‘regions’ of gunfire damage considered previously (leading edge, mid-chord and trailing edge) are each discussed separately as a result of their differing characteristics.

The applications of the following techniques, when used in the development of aerodynamic vulnerability analyses, would be to;

(i) provide methods of interpolation to intermediate damage sizes and locations with some degree of accuracy, and
(ii) provide methods which themselves could be applied to wings of differing section, given the availability of the necessary data.

It must be noted that the values identified for constants in the following sections are specific to the aerofoil geometry tested here. Different aerofoil sections are likely to result in different values to those identified. However, by using the proposed methods, the size of any experimental damage coefficient dataset required to determine such values could be reduced significantly from that tested in this investigation.

4.2 Damage Conditions with Minimal Effects

The experimental evidence presented in the previous chapter indicated that the type of jet resulting from the ‘though flow’ determined the extent of the damage effects. Weak jets were seen to result in relatively small changes in $C_L$, $C_d$ and $C_m$, whilst strong jets gave significantly greater changes. However, it was also seen that at certain incidences the
through-flow was small, if not negligible. This, it was concluded, resulted from a near zero pressure differential between the upper and lower surfaces at the damage location.

From the undamaged, two-dimensional, upper and lower surface $C_p$ distributions it has been shown (Section 3.4.4.2) that the pressure coefficient differential, $\Delta C_p$, between the two surfaces could be calculated. Hence, for the undamaged wing, the $\Delta C_p$ values at each damage centre location (c/4, c/2 and T.E.) may be determined over the incidence range tested. These values are illustrated by Figure 4.1. Note that trailing edge values have been approximated by 90%c values. It was found that for each of the locations there were incidence values where the pressure differential was approximately zero. These were -2.5°, -4.5° and -8° for the quarter-chord, half-chord and 0.90c locations.

![Figure 4.1 Undamaged Pressure Differential vs Incidence at damage locations.](image)

Experimental $d[C_L]$ values (Figures 3.47, 3.50 and 3.57) indicated that for damage centred at a given location, all damage sizes tested gave approximately zero lift loss, at approximately the same incidence. For quarter-chord, half-chord and trailing edge damage, the incidence values were found to be approximately -2°, -4°, and -8° respectively. The findings indicated that (given the 2° incidence increment of coefficient values), there was a good match between
the incidence value at which the pressure differential was seen to be zero, and the incidence at which minimum lift loss occurred, for damage at each location tested.

Hence it appears that for all damage sizes tested; firstly, damage lift loss is related to the undamaged pressure differential, $\Delta C_p$, at the damage centre location. Secondly, minimum lift loss may be achieved when the incidence is set to the value which would have given a zero $\Delta C_p$ value at the damage centre location. Finally, given that the pressure differential between the surfaces would have been expected to vary along the chord over the diameter of the damage, this proved to have little influence on the consistency of the findings across all damage sizes tested.

Likewise, away from stall, the smallest drag increments for a given damage size and location were observed at the same incidences as those seen for the minimum lift losses. Although, minimum $d[C_d]$ values were seen to increase with damage size. These effects were seen at -2° and -4° for quarter-chord and half-chord damage respectively, as seen in figures 3.48 and 3.51. Even in the case of trailing edge damage, where $d[C_d]$ values were small (Figure 3.58), all damage sizes indicated a reduction in $d[C_d]$ at the same incidence. This was found to be -8° for trailing edge damage.

Finally, minimum $d[C_m]$ values, for all damage sizes, were also seen to be related to zero pressure differential at the damage location (Figures 3.49, 3.52 and 3.59). However, both quarter-chord and half-chord damage zero $d[C_m]$ incidence values were approximately +1.5° greater than for their minimum $\Delta C_p$ values, occurring at approximately -1° and -3°. Only trailing edge damage indicated minimum $d[C_m]$ at the same incidence as damage location minimum $\Delta C_p$. Observation of changes in the upper and lower surface $C_p$ results do not indicate any changes which may explain these $d[C_m]$ observations.
4.3 Leading Edge Damage Effects

4.3.1 Lift Loss Interpolation

The coefficient values for the wing with leading edge damage were previously presented in Figures 3.12 to 3.14. This section considers the resulting \( d[C_L] \) values and illustrates that it is possible to correlate the loss in lift to the percentage wing area removed by the leading edge damage.

Given the previous definition of change in lift coefficient \( d[C_L] \), as:

\[
d[C_L] = C_{L \text{ damaged}} - C_{L \text{ undamaged}}
\]

this can be normalised by \( dA \), the ratio of wing area removed by damage to total undamaged wing area:

\[
dA = \frac{(\text{Damage semi-circular area})}{(\text{Total undamaged wing area})}
\]

The values of \( d[C_L] / dA \) for the four damage sizes tested, are plotted against incidence in Figure 4.2. The data from all three larger damage sizes approximates to the same curve, between \(-6^\circ\) and \(+6^\circ\), with the two largest sizes remaining closely matched up to \(+14^\circ\). Recalling section 3.3.4, it was shown that premature stall for 20\(c\%) \) damage occurred later than that seen for the other two damage sizes, with 30\(c\%) \) and 40\(c\%) \) premature stalls occurring at similar incidences.

The earlier results for changes in \( C_L \) values (Figure 3.8) indicated that only a slight loss in lift was caused by 10\(c\%) \) damage. However, when plotting the 10\(c\%) \) values of \( d[C_L] / dA \), it was found that the effect per unit area removed was significantly greater than that of the larger damage sizes. It was also seen that from \(+8^\circ\), the values of \( d[C_L] / dA \) become positive, reflecting the earlier observations that the damage triggered vortices on the leading edge, which at high incidences re-energised the upper surface flow and reduced separation, thereby delaying the onset of stall.
Figure 4.2 L.E. Damage; $\frac{d[CL]}{dA}$

Figure 4.3 L.E. Damage; $\frac{d[Cd]}{dA}$

Figure 4.4 L.E. Damage; $\frac{d[Cm]}{dA}$
The above results indicate a method with some degree of accuracy, for interpolation of lift losses between -6° and +6° for damage sizes between 20%c and 40%c diameter. For damage sizes between 10%c and 20%c, the accuracy of any interpolation would be reduced. However, given the relatively small losses for these cases, interpolated $C_L$ values would remain good approximations.

4.3.2 Drag Increase Interpolation

The drag coefficients associated with leading edge damage were presented in Figure 3.13. From these results it was found that the drag increase was progressive with damage size. The same approach was then taken to normalise drag coefficient changes as with lift, i.e. given $d[C_d]$, where:

$$d[C_d] = C_{d damaged} - C_{d undamaged}$$

this was then normalised by $dA$.

Thus, the values of $d[C_d] / dA$ were calculated and the results plotted for the four damage sizes tested, Figure 4.3. It was found that for all the damage sizes tested, the values of $d[C_d] / dA$ closely matched over the incidence range -6° to +8°. Beyond +8°, the differing stall characteristics become apparent. At negative stall, with the exception of 10%c damage, it was found that the correlation continued. Thus, from these results it would be possible to interpolate drag increments for intermediate damage sizes, at incidences away from stall.

4.3.3 Pitching Moment Interpolation

The changes in leading edge pitching moment coefficients were previously presented in Figure 3.9. As with lift and drag coefficients, $d[C_m]$ was normalised by $dA$.

$$d[C_m] = C_{m damaged} - C_{m undamaged}$$
The results are illustrated by Figure 4.4. It was seen that a close match between 30\%c and 40\%c results existed between -8° and +6°. This indicated that the predominant factor influencing the change in pitching moment coefficient was the damage size. Between -4° and +4°, 20\%c results also closely related to those found for 30\%c and 40\%c damage. These results indicated similar characteristics, up to the onset of premature stall.

The \( d[C_m] \) results for 10\%c damage have been included for completeness, however they have a greater degree of scatter over the incidence range tested. The \( C_m \) results (Figure 3.13) indicated that up to +8°, the 10\%c results observed were within the error band identified for \( C_m \) values, and as such should be treated with caution. Further, given that the 10\%c \( d[C_m] \) values were small and that normalisation was by a small damage area, scatter and error in the results might be expected.

4.4 Mid-Chord Damage Effects

4.4.1 Lift Loss Interpolation

For circular shaped damage with a chord diameter ratio \( d/c \), the non-dimensional area of damage \( A_d \) is given by:

\[
A_d = \frac{\pi \left( \frac{d}{c} \right)^2}{4}
\]

...(4.1)

The assumption is made that the non-dimensional area of the wing surface affected by the damage, \( A_{loss} \), will be a function of the following:
(i) The non-dimensional area of damage, \( A_d \)
(ii) Average pressure differential reduction over the region of the wing effected by the damage, \( \Delta C_p \text{loss} \)
(iii) An aerofoil section and through-flow direction related value, \( \mu \)
Thus, $A_{\text{loss}}$ may be written as:

$$A_{\text{loss}} = \mu \cdot A_d \cdot \Delta C_{p_{\text{loss}}}$$

$$= \frac{\mu \cdot \pi \left( \frac{d}{c} \right)^2}{4} \Delta C_{p_{\text{loss}}}$$  \hspace{1cm} ... (4.2)

By defining the lift loss resolved normal to the chord as $d[C_L]_N$, the change in resolved lift loss may be equated to the product of the pressure differential 'lost' and the non-dimensional area affected $A_{\text{loss}}$, i.e.;

$$d[C_L]_N = A_{\text{loss}} \cdot \Delta C_{p_{\text{loss}}}$$

$$= \frac{\mu \pi \left( \frac{d}{c} \right)^2}{4} \left( \Delta C_{p_{\text{loss}}} \right)^2$$  \hspace{1cm} ... (4.3)

Now, the value of $\Delta C_{p_{\text{loss}}}$ results from;

i) An average pressure differential value over the area removed by the damage hole. This may be approximated by the $C_p$ differential at the hole centre, $\Delta C_{p_{\text{UD}}}$.  
ii) Additional $C_p$ changes in the pressure field surrounding the damage, represented by an average change value, $\delta C_p$.  

Thus the total average pressure coefficient loss may be represented by;

$$\Delta C_{p_{\text{loss}}} = \Delta C_{p_{\text{UD}}} + \delta C_p$$  \hspace{1cm} ... (4.4)

Resulting in

$$d[C_L]_N = \frac{\mu \pi}{4} \left( \frac{d}{c} \right)^2 \left( \Delta C_{p_{\text{UD}}} + \delta C_p \right)^2$$  \hspace{1cm} ... (4.5)
Assuming that the values of $\mu$ and $\delta C_p$ remain constant for a given damage size $d$.

Now, from equation (4.5) and taking

$$m_1 = \frac{1}{2} \left( \frac{d}{c} \right) \sqrt{\mu \pi} \quad \text{... (4.6)}$$

it is possible to write;

$$\sqrt{d|C_L|_N} = m_1 \Delta C_{pUD} + m_1 \delta C_p \quad \text{... (4.7)}$$

From the experimental $d[C_L]$ values, $d[C_L]_N$ values were calculated for quarter and half-chord damage. Thus, it was possible to determine the values of $m_1$ and $\delta C_p$ from graphs of $\sqrt{d|C_L|_N}$ vs the undamaged pressure differential values at the damage centre locations, $\Delta C_{pUD}$. The value of $m_1$ is given by the gradient of a straight line fit, and $\delta C_p$ from the intercept on the x-axis. Figures 4.5 and 4.6 illustrate such plots for $d=20\%c$ damage at the quarter chord location. Note that different $m_1$ and $\delta C_p$ values were obtained for the positive and negative $\Delta C_{pUD}$ ranges. This primarily resulted from the difference in through-flow direction and differing pressure fields on the upper and lower surfaces. Note also that the straight-line fit does not include the stall data points.

Making the assumptions that (i) $\mu$ is constant for all values of $d$ (but different for the positive and negative $\Delta C_p_{UD}$ range), and (ii) it would be expected that for no damage, i.e. $d=0$, the value of $d[C_L]_N$ would also be zero. Then the values of $m_1$ from each damage case, when plotted against damage size $d/c$, would be expected to give a straight line, passing through the origin.
Figure 4.5 $\sqrt{d[C_{LN}]}$ vs Pressure Differential (c/4 20%c, Negative $\delta C_{pu}$ range)

Figure 4.6 $\sqrt{d[C_{LN}]}$ vs Pressure Differential (c/4 20%c, Positive $\delta C_{pu}$ range)
Figure 4.7  $m_1$ values for damage cases tested (Negative $\Delta C_{P_{\alpha}}$ range)

Gradient $m_2 = -1.292$
Figure 4.7 shows such a plot of \( m \) values for the negative \( \Delta C_p \) range. Although some data scatter is present, from the gradient of this graph, \( m_2 \), an average value of \( \mu \) (in this case for the negative \( \Delta C_p \) range) may be determined from:

\[
\mu = \frac{4m_2^2}{\pi} \tag{4.8}
\]

From the experimental data available this gave a value of \( \mu = 2.125 \) for the negative \( \Delta C_p \) range of all quarter and half chord cases tested.

From graphs such as Figures 4.5 and 4.6, the values of \( \delta C_p \) are taken from the x-axis intercepts. Here, -0.037 and 0.242 for the positive and negative \( \Delta C_p \) ranges respectively. Note, that the value of \( \delta C_p \) was found to vary with damage size and location. The values are illustrated by Figure 4.8 for the negative \( \delta C_p \) range for quarter and half-chord cases. Although differing, there were similarity in both the trends and magnitudes of both damage cases. Thus, for a given damage size \( d/c \), and interpolating to the corresponding values of \( \mu \) and \( \delta C_p \), it would be possible to calculate an approximation for \( d[C_L]_N \), and hence \( d[C_L] \), from the undamaged \( \Delta C_p \) value at the damage centre. However, the method is limited to the mid-incidence region, away from positive and negative stall.

![Figure 4.8 Average \( C_p \) Change Value \( \delta C_p \), (Negative \( \Delta C_p \) range)](image-url)
4.4.2 Drag Increase Interpolation

Drag increments from mid-chord damage may result from; (i) surface flow separation, as shown, caused separated regions and re-circulation, resulting in increased pressure drag, and (ii) Damage pressure drag, where the exposure of the wing internal structure, on which the through-damage flow impinged, resulted in pressure-drag contributions.

Neither of these contributions have individually shown any direct correlation with the total drag increments measured. Thus it was believed to be the combined effects of these factors which together resulted in the total drag increments seen. However, an interpolation method has been determined which has shown that it is possible to correlate the drag increment $d[C_d]$ with the lift loss $d[C_L]$.

By plotting quarter-chord $d[C_d]$ vs $d[C_L]$ throughout the negative $d[C_L]$ range, it was possible to identify two distinct characteristics. Figure 4.9 illustrates the general observations seen, with 10%c and 30%c as examples. The other damage sizes showed similar results, but are not illustrated here. Firstly, the weak-jet data points for each damage size tended to form a cluster of data points located close to the origin. This reflected the small drag increments present when through-flow was limited. The data within this cluster showed no clear trends. Secondly, results for strong jets were seen to emerge from the general cluster of data around the origin. Away from the stall, all data points associated with strong jets gave an approximately linear relationship, see Figure 4.9. This was found to be the case for both quarter and half-chord damage results.

It was found that different damage sizes resulted in different gradient, $\frac{d[C_d]}{d[C_L]}$, values (Figure 4.9), and different $d[C_d]$-offset values, i.e. intercept on the y-axis. Although negligible net flow through the damage has been shown to occur at zero lift-loss, drag still resulted from the interaction between the wing surface flow and the hole edges.

The values of $\frac{d[C_d]}{d[C_L]}$ and $d[C_d]$-offset for the quarter-chord damage cases are shown in
Figure 4.9 $\frac{d[CL]}{d[Cd]}$; $c/4$, negative $d[CL]$ range

Gradients: $10\%c$ $-0.537$ $30\%c$ $-0.198$
Figure 4.10. From this type of graph it is possible to interpolate to values for intermediate damage sizes, and hence determine $d[C_d]$ values.

![Graph](image)

Figure 4.10  $d[C_d]/d[C_L]$ & $d[C_d]$-offset vs. Damage size (c/4 damage)

4.4.3 Pitching Moment Interpolation

It was found that no clear pitching moment effects may be identified from the 10%c damage results weak-jet. For 20%c, 30%c and 40%c c/4 damage in the range $-2^\circ$ to $+10^\circ$, the $d[C_m]$ values were approximately linear and of a similar magnitude (Fig 3.49). This was also found to be true for the half-chord damage results (Figure 3.52). Thus, when interpolating to different damage sizes at the same damage location, over the positive incidence range, it would be fair to use a mean value based on 20, 30 & 40%c results and assume an approximately linear variation between locations, provided a strong-jet had been identified, from flow-visualisation tests. The negative incidence range indicated a greater level of data scatter. In the absence of any further data, the above linear interpolation between mean incidence point values is the only suggested method for this range at this time.
4.4.4 Validation Test Case

An independent validation test case was used to assess the accuracy of the suggested mid-chord damage interpolation techniques. This involved damage of diameter 24%c centred at the quarter-chord location, a damage case not used in the development of the techniques.

As outlined previously in section 4.4.1, values of $\mu$ were calculated for mid-chord damage from the experimental data, and $\Delta C_p$ values for specific damage sizes must be interpolated from data in Figure 4.8. For this validation case, where $d/c=0.24$, the associated values were:

<table>
<thead>
<tr>
<th>$\Delta C_p$ range</th>
<th>$\Delta C_p$ range</th>
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<tbody>
<tr>
<td>+0.13</td>
<td>-0.18</td>
</tr>
</tbody>
</table>

These values were then substituted into equation 4.5:

$$d[C_L]_N = \frac{\mu \pi}{4} \left(\frac{d}{c}\right)^2 \left(\Delta C_{pUD} + \delta C_p\right)^2$$

together with the known undamaged pressure differential, $\Delta C_{pUD}$, for each incidence. This determined the change in resolved lift loss $d[C_L]_N$, from which the value of $d[C_L]$ was calculated.

A comparison of calculated and experimental $d[C_L]$ results are presented in Figure 4.11, and showed a good match in the range $\pm 8^\circ$. Within this range, the $d[C_L]$ error ranged from 0.0006 at $0^\circ$ to 0.0309 at $+8^\circ$. At incidences outside this range, a significant mismatch became apparent, with the calculated results being excessively pessimistic. However, these points resulted from the onset of stall and lay outside of the range for which the method was stated as being applicable.
Figure 4.11 Predicted $d[CL]$ : $c/4$ 24% c damage

Figure 4.12 Predicted $d[Cd]$ : $c/4$ 24% c damage

Figure 4.13 Predicted $d[Cm]$ : $c/4$ 24% c damage
Calculated drag values are shown in Figure 4.12. The method suggested required \(d[C_L]\) values in order to calculate \(d[C_d]\) values. The \(d[C_L]\) values used were those calculated by the above method. Comparing calculated and experimental \(d[C_d]\) values between 0° and +10°, a good match was found. However, beyond +10° the calculated drag increments were again over pessimistic. This resulted from both the compounding of calculated \(d[C_L]\) errors used to calculate \(d[C_d]\), and the identified problems of predicting values in the stall region.

However, a more significant problem was identified, in that the calculated minimum drag increment, at -2°, was -0.00007 instead of the experimental value of 0.00359. This indicated that while the suggested method was sufficient to approximate the larger drag penalties incurred when the strong-jet separations occur, it was not accurate enough to give small drag increments due to the damage with little or no flow through it. Further detailed investigations are required to find an accurate method of determining such values. Over the remaining negative incidence range where positive \(\Delta C_p\) values were found at the damage centre, the predication method was also found to be inaccurate, and under-estimated the drag increments for negative lift.

The pitching moment interpolation technique was found to generate accurate calculated values of \(d[C_m]\), see Figure 4.13. Over the incidence range -2° to +12° the \(d[C_m]\) error ranged between 0.0004 and 0.0020. The negative lift range was slightly less accurate. However over the stall characteristics, the calculated results compared well with the experimental values obtained.

4.5 Trailing Edge Damage Effects

4.5.1 Lift Loss Interpolation

Scaling \(d[C_L]\) values (shown in Figure 3.57) by the area of the undamaged wing, \(S_{UD}\), equates to the lost lift force normalised by the dynamic head, \(Q\);

\[
d[C_L] \cdot S_{UD} = \frac{\text{Damage Lift Loss}}{Q} \quad \text{... (4.9)}
\]
where
\[ \text{S}_{\text{UD}} = \text{Span} \cdot \text{chord} \]

Consider a trailing edge region of the wing, full span and defined in the chordwise direction by the damage diameter, i.e. the region extends from the c-(d/2) position rearwards to the trailing edge (See Figure 4.14).

By integrating the undamaged \( C_p \) pressure distribution for this trailing edge region, it was possible to obtain a local normal-force coefficient \( C_{N\text{local}} \) for the region alone (e.g. in the case of 40%c damage, extending from \( x/c=0.8 \) rearwards). Now, scaling this \( C_{N\text{local}} \) value by the region’s area, gave an undamaged local force (normal to the surface) normalised by \( Q \);

\[ C_{N\text{local}} \cdot S_{\text{TE region}} = \frac{\text{Local Normal Force}}{Q} \quad \ldots \quad (4.10) \]

where;
\[ S_{\text{TE region}} = \text{Area of trailing edge region considered} \]

By plotting the results of the normalised damage lift-loss force against local normal force (for all trailing edge damage sizes tested) it was found that an approximately linear relationship existed. As the span was a common factor, this was removed, and by specifying
Figure 4.15 Lift loss as a function of T.E. Normal coefficient

Gradient = 0.717

40%c damage 30%c damage 20%c damage 10%c damage

*   O   □   △
damage diameter in its non-dimensional form \( \left( \frac{d}{c} \right) \), the results were best illustrated by plotting \( d[C_L] \) against \( \left( \frac{d}{c} \right) C_{N_{\text{local}}} \) (Figure 4.15). The stall related data points were excluded, and such high data scatter might be expected due to inaccuracies in the integrated pressure values. From this graph it can be seen that for any damage size within the range tested, the value of T.E. \( d[C_L] \) may be approximated by the following relationship:

\[
d[C_L] = 0.717 \ C_{N_{\text{local}}} \left( \frac{d}{c} \right)
\]

... (4.11)

Note that the accuracy of this approach will be considered in section 4.5.4.

4.5.2 Drag Increase Interpolation

In comparison with the drag increments resulting from leading edge and mid-chord damage, \( d[C_d] \) increments for trailing edge damage were found to be an order of magnitude smaller, as discussed earlier in Section 3.5.2.

Clear trends in the data were seen, as illustrated by Figure 4.16. At moderate incidence values between \(-6^\circ\) and \(+4^\circ\), there was an approximately linear relationship between damage size and magnitude of drag increment \( d[C_d] \). Considering two specific incidence points within this range, Figure 4.17 indicated the near linear relationship. Such a relationship might be used to interpolate to intermediate damage size values.

4.5.3 Pitching Moment Interpolation

When looking at changes in pitching moment, a similar technique to that employed in the assessment of trailing edge lift losses was found to be successful in linking undamaged wing surface pressure coefficients with damage \( d[C_m] \) values.
From the undamaged surface pressure $C_p$ values over the region $(c-d/2)$ to $c$, a local pitching moment $C_m$ value (about the quarter-chord) was calculated. This value will be defined as $C_m_{local}$. From the previous experimental tests, the values of $d[C_m]$ were known. By plotting values of $d[C_m]$ against $(\frac{d}{c})C_m_{local}$, excluding stall related points, it was found (Figure 4.18)
Figure 4.18 Pitching Moment Change as a function of T.E. $C_m$

Gradient = -0.217

40% c damage 30% c damage 20% c damage 10% c damage

* ○ □ △
that for any damage size within the range tested, the value of $d[C_m]$ may be approximated by the following relationship:

$$d[C_m] = -0.217 C_m \text{ local} \left( \frac{d}{c} \right) \quad ... (4.12)$$

**4.5.4 Validation Cases**

Given the degree of scatter within Figures 4.15 and 4.18, the accuracy of the $d[C_L]$ and $d[C_m]$ calculation methods was checked. By using the undamaged wing pressure data and equations 4.11 and 4.12, $d[C_L]$ and $d[C_m]$ values were calculated for the 20\%c and 40\%c trailing edge damage cases. Both the actual experimental results and the calculated results are presented for comparison in Figures 4.19 and 4.20. It can be seen that in the incidence range away from stall, there was a good match between the experimental and calculated results, both in terms of the data trends and magnitudes. The minimal effects (as discussed earlier) were also correctly calculated as occurring at -8°. Where stall effects were seen in the experimental results, the calculated results differed significantly, as expected.

![Figure 4.19 T.E. Damage; Experimental vs Calculated $d[C_L]$](image-url)
Figure 4.20  T.E. Damage; Experimental vs Calculated $d[C_m]$
CHAPTER 5. THE EFFECTS OF SIMULATED MULTI-HIT GUNFIRE DAMAGE

5.1 Introduction

This chapter considers the effects of damage from more than one simulated gunfire damage hole. It was believed that some degree of correlation might exist between the effects of single hole damage and multi-hole damage. A simple method of summing the lift losses, drag increments and pitching moment changes is presented. The results are then compared with corresponding experimental data, and the method's accuracy is assessed.

5.2 Superposition Prediction Method

The method proposed for calculating the effects of lift loss, drag increase and pitching moment changes from more than one gunfire damage hole was one of 'superposition'. By combining the effects of the individual single-hole damage cases, it is suggested that the effects of the multi-hole damage might be calculated with some degree of accuracy.

The method calculates the predicted total lift ($d[C_L]'$), drag ($d[C_d]'$) and pitching moment ($d[C_m]'$) multi-hole changes, by adding together the single-hole values of $d[C_L]$, $d[C_d]$ and $d[C_m]$. For example, consider two-hole damage of 20% at the quarter-chord and 20% at the half chord. Given that the individual lift loss for each case is known, the predicted total lift loss could be calculated by;

$$d[C_L]' = d[C_L]_{20\%c/4} + d[C_L]_{20\%c/2}$$

and similarly,

$$d[C_d]' = d[C_d]_{20\%c/4} + d[C_d]_{20\%c/2}$$

$$d[C_m]' = d[C_m]_{20\%c/4} + d[C_m]_{20\%c/2}$$

(where subscripts denote damage cases.)
Whilst considering only two damage locations here, this method might be extended to cover a greater number of damage locations, all located along the same spanwise station.

5.3 Experimental Investigation

To assess the superposition prediction method, an experimental investigation was undertaken. The multi-hole damage was restricted to two-hole damage cases. The investigation of two, rather than three or more holes was intended to provide an insight into any interference effects that may occur between two such holes. With three or more holes, it would not have been possible to assess the individual interference between any two holes. Additionally, the structural integrity of the test wing with three damage locations at the same spanwise station was considered. Assessing the risk to model and tunnel equipment, it was believed that with three hole damage, wing-model failure was a real possibility.

In order to assess the proposed prediction method, the actual location and size of the damage applied to the wing was restricted to a combination of test data previously collected on the single-hole damage cases. Tests considered damage of the same size at both locations, as it was probable that actual multi-hit battle-damage would result in holes of similar sizes.

The experimental tests undertaken considered various combinations of 10%c, or 20%c, damage at each of the four locations identified in Section 3.2.6. However, only three of the cases tested will be presented here, as illustrative examples of the principles identified. The individual holes within each two-hole case considered had already indicated significant values of \( d[C_l] \), \( d[C_d] \) and \( d[C_m] \) on their own. These were leading edge, quarter and half-chord damage cases. Trailing edge damage results indicated similar trends, although the results are not included here.

The damage combinations presented here are ;
5.4 Comparison of Experimental & Predicted Results

5.4.1 Multi-Hole Mid-chord Damage

Firstly, considering two-hole 10%ch damage at both quarter-chord and half-chord locations, the results of both experimental and predicted values are shown in Figures 5.1, 5.2 and 5.3. Additionally, for discussion purposes the effects of the single hole cases are also included.

In general, the trends of the predicted summation values (d[C_L]', d[C_d]' and d[C_m]'), matched closely those measured experimentally (d[C_L], d[C_d] and d[C_m]). However, the accuracy of the predicted values were found to vary over the incidence range tested. At small incidences, the predictions matched very closely to those measured, for example d[C_d]' at +4° had an error of only 4%. For lift loss, the range of accurate predictions was from -8° to +10°. Whilst for drag increments and pitching moment changes, the range was slightly reduced, -8° to +6°. Within this range, two-hole d[C_m] values appeared to ‘average out’ the data scatter previously seen from the single-hole cases. Thus the above incidence range (-8° to +6°) showed a negligible interference effect.

In the positive lift range, it was found that from +8° onwards, both d[C_d] and d[C_m] experimental results were significantly less than those predicted from superposition. Thus the experimental effect of the two damage holes was to produced a negative interference effect. It

<table>
<thead>
<tr>
<th></th>
<th>Case 1</th>
<th>Case 2</th>
<th>Case 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>L.E.</td>
<td></td>
<td></td>
<td>20%</td>
</tr>
<tr>
<td>c/4</td>
<td>10%</td>
<td>20%</td>
<td>20%</td>
</tr>
<tr>
<td>c/2</td>
<td>10%</td>
<td>20%</td>
<td></td>
</tr>
<tr>
<td>T.E.</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Figure 5.1 $d[CL]$ Values - c/4 10%c & c/2 10%c Damage

Figure 5.2 $d[Cd]$ Values - c/4 10%c & c/2 10%c Damage

Figure 5.3 $d[Cm]$ Values - c/4 10%c & c/2 10%c Damage
was seen that this negative interference coincided with the onset of the 'strong-jet' wake characteristics for the c/4 single-hole damage, between +6° and +8°.

Up to +8° incidence, either both or only the c/4 (single-hole) damage would have resulted in weak-jets, with attached wakes. As c/4 weak-jets have been shown to have no significant influence on the pressure field downstream from the damage, it is suggested that with two weak-jets present, the lift loss of each is relatively unaffected by the presence of the other. Although c/2 damage develops into a strong-jet by +6°, both it and the weak c/4 jet continue to contribute to the total effects measured. However, by +8° the c/4 damage jet would be expected to have become strong, and developed it's own reverse flow wake. This would then have overlaid and effectively replaced the existing c/2 wake. Additionally, the pressure field which would normally be expected within the c/4 damage wake would be reduced by flow though the c/2 hole. This was seen as a reduction in reverse flow from the trailing edge in the flow visualisation (Figures 5.4 and 5.5). Hence, the net result was an actual reduction from the combined single-hole cases, i.e. a negative interference effect. Compared with the d[Cd] results, interference on d[C_L] values appeared to have been delayed by 4°, so that they were seen to occur from +12° incidence upwards. This would indicate that interference effects on d[C_L] were weaker than for d[Cd].

Over the negative lift range, similar negative interference effects were observed beyond -8°. Once again this coincided with the onset of the c/4 strong jet on the lower surface beyond -8°.

5.4.2 Effects of Enlarged Mid-Chord Damage

Given the above predicted vs experimental characteristics, moving to larger damage at the same locations (20%c at both quarter and half-chord locations) indicated the same trends were present.

Firstly, c/4 attached flow from a weak-jet was expected over the range -4° to 0°, as indicated by the single hole damage case. Thus within this incidence range, a good match between predicted values and experimental values was expected. This was found to be the case (see
Figure 5.4 Upper Surface (10\%c c/4) damage, +8°

Figure 5.5 Upper Surface (10\%c c/4 + 10\%c c/2) damage, +8°
Figures 5.6, 5.7 and 5.8). Additionally, as found with 10% damage, $d[C_L]$ values continued to be correctly approximated for a further 4° (at +4° the difference between prediction and experiment was only 0.011, approximately 6%).

Secondly, c/4 'strong-jet' wakes were expected to result in negative interference effects outside this -4° to 0° range (-4° to +4° for $d[C_L]$). This was confirmed by 20% damage, which showed that experimental results were significantly less outside this range, e.g. by +8°. The summation value was 0.064 whilst the actual experimental value was 0.033, an error of 94%.

5.4.3 Effects of Varying Damage Location

Further tests were undertaken with 20% damage located at different damage locations. These were the leading edge and quarter-chord positions. The comparison between predicted and experimental values can be seen in Figures 5.9, 5.10 and 5.11.

For all coefficients, the trends predicted matched well to those found experimentally. A high degree of prediction accuracy was seen over the range defined by a leading edge weak-jet, i.e. between -10° and +6° incidence. Lift $d[C_L]$ predicted values were seen to be accurate over most of this incidence range. As were $d[C_d]$ and $d[C_m]$ results, which were then seen to exhibit greater inaccuracies in the range where a strong jet was expected from the leading edge location (from +8° and greater). As found previously, lift loss accuracy was maintained for approximately a further 4° up to +10°, an error of only 0.0004 at +10°, although beyond this errors were not unduly significant 0.0281 at +12°. Drag increments became inaccurate at negative stall, and lift losses were inaccurate from -6° and less.

Together with the previous c/4 and c/2 tests, these results presented indicated that the damage location which defined the onset of interference effects (and hence the overestimation of the superposition method), was that of the forward-most damage location. Previously, the c/4 damage characteristics were seen to control the c/4 and c/2 damage combination. Now, the
combination of LE and c/4 damage is seen to be controlled by the LE damage flow. Similar effects were also seen for the other 2-hole damage combinations tested.

Thus to summarise, the results gained indicated that summation of single-hole damage characteristics by the 'superposition method' can be used to predict the effects of two-hole damage. The magnitudes of the effects predicted matched closely to those found experimentally over a restricted portion of the full incidence range. This restricted range was found to be dependent on the damage jet types observed, and could itself be predicted from the single hole results. A correlation has been identified relating the forward-most damage jet characteristics to the accuracy of the superposition method. Where the superposition method failed to accurately predict the experimental values, the trends were generally correct, but the predicted magnitudes consistently overestimated the effects of two hole damage for the cases considered here. However, this may not be the situation for all hole combinations.

Note that the multi-hole damage was only considered for damage at the same spanwise location. It is not suggested that this method could be used for damage located at different spanwise locations. In such circumstances the damage wakes would not be expected to combine or overlay each other in the same way. Additionally, the spanwise pressure losses might be expected to combine with some degree of interference, not necessarily the same as those identified above. Combinations of spanwise located damage are one of the possible lines of further investigation discussed later.
CHAPTER 6. THE CHARACTERISTICS OF SIMULATED MISSILE DAMAGE

6.1 Introduction

The key characteristics of missile warheads, together with their method of damage propagation, have already been discussed in Chapter 2. This chapter outlines the approach taken to quantify such missile fragment damage so that it could be applied to a test wing. The key characteristics are discussed and the basic modelling assumptions are presented.

The aerodynamic effects of such damage were investigated, and the consequences of varying two key variables; fragment density and damage hole size, were considered. As with previous gunfire damage testing, both the measurement of aerodynamic coefficient changes and surface flow visualisation techniques were undertaken. No surface pressure measurements were made.

6.2 Methods of Modelling Missile Damage

The geometry of missile engagements on target aircraft may vary significantly, due to a large number of factors. However, the resulting damage remains broadly similar in most cases. This results from the fact that all missile warheads are fundamentally similar, as outlined in Chapter 1, generating a ‘beam’ of high velocity fragments which become incident on the aircraft’s structure.

As with previous gunfire damage cases, no experimental missile damage modelling techniques were readily available at the outset of this research. Indeed, the literature survey indicated that where as some limited modelling of gunfire damage had been undertaken in the past, this was not the case for missile damage. Not one aerodynamic investigation into missile damage appeared to have been previously published.

With reference to British Aerospace technical specialists, and current Military Standards, the basic features of missile damage were identified and quantified (Refs. 2, 3, 29, 50). The
following text outlines the variables used in the subsequent missile damage parametric study.

6.2.1 Extent of Damage Area

Given the nature of a missile engagement, where the warhead detonation may be anywhere from 5m to 50m away from the aircraft, the extent of damage over the aircraft structure may vary extensively. In the majority of cases, large portions of the wing and fuselage may sustain fragmentation damage. In order to model such widespread damage and to generate ‘uniform’ effects over the entire wing, the method of simulating missile-damage was applied over the full area of the model. As there were no results of prior testing available, for any such simulated damage, it was believed that this would define ‘baseline’ characteristics.

6.2.2 Fragmentation Grid Pattern

Generally, warhead designs are optimised to give the maximum chance of causing damage in all directions. Following the detonation of the high-explosive charge, the warhead casing bursts in a predefined manner. As the casing breaks-up, a large number of similarly sized fragments are propelled outwards from the missile’s longitudinal axes. (See Figure 6.1) This generates an expanding cylindrical beam of fragments. The optimisation of the design ensures that approximately equal numbers of fragments are propelled in all directions, i.e. there is a uniform distribution. Where the beam of fragments intersects with the aircraft structure, this forms a regular and uniform distribution of fragmentation holes per unit area. This is a widely accepted assumption and is supported by a large number of live-fire test results (Refs. 2, 3, 29).

The number of fragment holes per unit area, within the region of damage is commonly defined as the ‘fragment density’, \( \rho_f \). Fragment density was a key variable in this test programme. With reference to British Aerospace specialists (Ref. 29) and current Military Standards (Ref. 3), a range of fragment densities was chosen to reflect potentially survivable damage cases. These were \( \rho_f \) values of 10, 20 and 40 fragments per m\(^2\). However, these were full-scale damage values applicable to actual aircraft. To apply the equivalent number of
holes to the test wing a $\rho_f$ scaling factor was required, relating the model chord size (200mm) to a full-size wing chord. It was decided that a full-size mean chord value of 2m would be assumed. From this it was then possible to determine a value for model grid spacing; $a$. This value defines the distance between damage centres (see Figure 6.2) in both the chord and spanwise directions, and was determined by:

$$a = \frac{\text{model chord}}{\text{full-size chord}} \sqrt{\frac{1}{\rho_f}}$$  \hspace{1cm} (6.1)

Now, given the assumption of a regular uniform hole grid pattern, the placing of the grid on the wing planform had to be defined. The location of the holes relative to each other and to
the oncoming flow required careful consideration. The orientation of the grid could have been between 0° and 360° relative to leading edge. However, when at 0° or multiples of 45° orientation, holes downstream would sit directly in the wakes of those upstream. It was felt desirable, that the holes should be arranged such that their centres were distributed uniformly upstream of the trailing edge, with the minimum number of holes placed directly downstream of others. This would then give equally spaced wake centrelines along the trailing edge. This was achieved by a small rotation of the grid relative to the leading edge by an angle \( \theta \). The value of \( \theta \) was calculated from the value of hole density \( \rho_r \), the grid spacing value and the wing chord length (see Appendix B).

Assuming a row had the centre of its first hole just at the leading edge, then the maximum number of holes possible occurred along that row. The grid rotation angle \( \theta \) was required to place the first hole in this row (Figure 6.3, row X+1) directly upstream of the last hole in the preceding row (hole N, row X). In this way, all other rows (having less than or equal to the maximum number of holes possible) had their centres distributed uniformly upstream of the trailing edge (See Appendix B).
Given the methods for calculating hole spacing ‘a’ and grid rotation angle ‘θ’, each full-scale fragment density resulted in the following model wing values:

<table>
<thead>
<tr>
<th>ρf (m²)</th>
<th>Grid spacing, a (mm)</th>
<th>Rotation angle, θ (°)</th>
</tr>
</thead>
<tbody>
<tr>
<td>10/m²</td>
<td>31.6</td>
<td>9.5°</td>
</tr>
<tr>
<td>20/m²</td>
<td>22.4</td>
<td>6.3°</td>
</tr>
<tr>
<td>40/m²</td>
<td>15.8</td>
<td>4.7°</td>
</tr>
</tbody>
</table>

6.2.3 Extent and Direction of Penetration

Missile fragment velocities can be up to approximately 2000m/s. Given the relatively light weight construction of aircraft wings, live-fire tests have shown that fragments may pass straight through wing structures and result in ‘through-hole’ damage, aerodynamically the ‘worst-case’ situation. Through-hole damage was chosen to simulate missile damage here, not only to reflect the above characteristics, but also to remain consistent with the earlier simulated gunfire damage tests.

As with gunfire damage, the direction of penetration of the fragments was modelled as
normal to the plan-view plane of the wing, i.e. at 90\(^\circ\) to both chordline and leading edge. This resulted in the upper surface holes being located directly above those in the lower surface.

6.2.4 Fragmentation Hole Shape and Sizes

Experimental investigations (Refs. 50 & 51) have shown that in tests designed to reproduce the effects of missile fragment impacts on aircraft structures, the resulting damage was typically circular in shape, of a size similar to that of the missile fragment, and with negligible petalling around the edges. These characteristics were also found to vary little with fragment velocity in tests conducted over the range 541 m/s to 1666 m/s. Consequently, circular shaped holes were used here to simulate the damage of each missile fragment.

Damage size was to be a key variable in the parametric studies. Using the circular representation of the resulting hole, it remained appropriate to continue using the percentage ‘diameter to chord length’ value as the measure of damage size. In all cases, the hole sizes were constant throughout the grid.

Following the release of Reference 29 and detailed discussions with British Aerospace specialists, damage hole diameters of 2.5\%, 4\%, 6\% and 8\% were selected as being appropriate test cases, i.e. corresponding to typical hole diameters on the chosen 2m full-size representative wing chord.

6.2.5 Summary of Simulated Fragment Damage Characteristics

The following summarises the test matrix. Note that it was not physically possible to model 8\% damage at a fragment density equivalent to 40/m\(^2\) (i.e. damage holes with diameters of 8\%, located 7.9\% between centres.)

<table>
<thead>
<tr>
<th>Fragment Damage Hole Diameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.5%</td>
</tr>
<tr>
<td>10 / m(^2)</td>
</tr>
<tr>
<td>20 / m(^2)</td>
</tr>
<tr>
<td>40 / m(^2)</td>
</tr>
</tbody>
</table>
6.3 Damage Effects on Aerodynamic Coefficients

6.3.1 Lift Loss First Order Effects

Despite the presence of complex surface flow characteristics and the development of upper surface separation (to be discussed below), it was found that to a first order approximation, lift loss measurements for missile damage at a given incidence was related to the percentage wing area removed.

The changes in lift loss for the different damage cases is summarised by Figure 6.4, which plots $d[C_L]$ values against incidence. It can be seen from this figure that all the test cases exhibited similar trends to differing magnitudes. The results confirm that the least lift loss resulted from the smallest damage size at the lowest fragment density (2.5%c at 10/m²), and the greatest loss from the largest damage size at the highest density (6%c at 40/m²). The figure also indicated that some damage cases resulted in similar lift losses, for example, 6%c at 10/m², 4%c at 20/m² and 2.5%c at 40/m². At +8° these cases all resulted in $d[C_L]$ values of -0.4570 ± 0.0035. These cases were found to have similar area losses (percentage of...
undamaged wing area removed) of 10.5%, 9.6% and 8.1% respectively. By plotting the values of $d[C_L]$ against percentage wing areas lost for a given incidence, a relationship was seen across the range of test cases. Figure 6.5 illustrates the $+8^\circ$ results.

![Figure 6.5 Lift Loss vs. Percentage Wing Area Lost ($+8^\circ$)](image)

All cases of missile damage produced reduced lift curve slopes up to the maximum incidence tested ($+14^\circ$) and evidence of delayed stall. Figure 6.6 is included here to illustrate the characteristics seen. Lift curve slopes were seen to depend on the amounts of wing area removed. As with individual $d[C_L]$ values, to a close approximation, the removal of similar percentages of area resulted in similar lift curves for different combinations of $\rho_t$ and damage size (see 6% at 10/m$^2$, 4% at 20/m$^2$ and 2.5% at 40/m$^2$ results Figure 6.6).

6.3.2 Lift Loss Second Order Effects

In addition to the above first order effects, it was found that the details of the lift curve slopes (Figure 6.6) were influenced by the complex interactions of attached and separated surface flow structures, and their variations with damage size and density. Principally, there were two forms of lift curves seen over the incidence range tested; those with a ‘partial’ stall followed by a recovery in gradient, and those with a more consistent positive gradient. These will be discussed below.
Figure 6.6  Missile Damage Lift Loss (20/m²)

\[ C_L \]

\[ \alpha (\text{deg.}) \]

Clean  2.5\%c  4\%c  6\%c  8\%c  (6\%c 10/m²)  (2.5\%c 40/m²)

Reynolds Number = 500,000
6.3.2.1 Lift Curve Slope with Partial Stall & Recovery

The $C_L$ results showed a reduced lift curve slope with a 'stall like' reduction in gradient for the three cases of damage at a low $\rho_f$ and damage diameter (2.5%c and 4%c at $\rho_f=10/m^2$ and 2.5%c at $\rho_f=20/m^2$). The partial stall was seen to occur at the same incidence as the undamaged wing, approximately $+10^\circ$. Increasing incidence beyond $+12^\circ$, the lift curve slope gradient was found to recover (See 2.5%c results in Figure 6.6).

Examining the flow visualisation for these cases, it was found that at all incidence values tested, the upper surface flow was in a chordwise direction. Additionally, three different regions could be identified in the flow, the extent of each was seen to vary with incidence. The first, was located between the leading edge and the first row of holes, where the surface flow appeared relatively unaffected by the presence of the holes. Figure 6.7 illustrates the effect for 2.5%c damage ($\rho_f=10/m^2$) at $+8^\circ$. As was seen for the undamaged wing, the separation bubble remained located at the leading edge (except where intersected by two holes). The second region was seen to start at the location of the first row of holes. Within this region, the clearest effects were seen for those holes on the first row. From detailed examination it was seen that the flow mechanisms present were very similar to those seen for gunfire damage, but on a smaller scale (See Figure 6.8). Forward separation lines were identifiable, where oncoming surface flow separated as a result of flow emerging through the hole. The collection of flow-visualisation mixture at the hole rear edges, also indicated the presence of contra-rotating vortex pairs, as seen for the larger gunfire damage. The majority of second region holes appeared to have characteristics similar to the previously identified weak-jets, i.e. wakes attached to the surface and little spanwise increase in wake width. Importantly, flow visualisation indicated no interaction between the flows of each hole. Indeed the majority of the surface flow, located in the gaps between the holes, appeared relatively unaffected by the wakes, and exhibited attached chordwise boundary layer growth (Figure 6.7 - point A). It should be noted that this was seen to occur in all three 'partial stall and recovery' cases, where, for each the ratio of 'inter-hole gap to damage radii' was greater than 12. (Inter-hole gap = grid spacing – hole diameter). Progressing rearwards beyond the first few rows, it was seen that the accumulating attached wakes from the damage resulted in a progressive reduction in the spanwise width of the unaffected areas.
Figure 6.7 Upper Surface Flow Visualisation (2.5%c at 10/m², +8 deg.)

Figure 6.8 Enlargement of individual fragment flow (2.5%c at 10/m², +2 deg.)
It is suggested that areas of ‘suction’ were able to develop in the first and second regions, see Figure 6.9. These areas would be what remained of the ‘two-dimensional’ pressure peak seen on the undamaged wing (Figures 3.29 to 3.31). The suction areas developed as a result of the relatively undisturbed flow between the holes. Note also that gunfire weak-jets were found to give relatively small changes in $C_p$ in the local flow field either side of damage (See upper-surface $C_p$ contour plot for 20%c c/4 weak-jet, Figure 6.10).

![Diagram showing regions of attached boundary layer growth](image)

**Figure 6.9 Regions of Attached Boundary Layer Growth**

![Contour plot of $C_p$](image)

**Figure 6.10 Gunfire Damage Weak-jet $C_p$ Contours (Upper Surface, c/4 20%c, 0°)**
The flow visualisation indicated that the combined wakes in the second region separated from the surface to form a third region. As seen in figure 6.7, the forward edge of this region was ambiguous, but was seen to move forward with increasing incidence (approx. 90%c, 60%c and 40%c locations for +2°, +8° and +12° respectively). Within this region, flow visualisation indicated only a small amount of through-flow, as little paint collected at the hole edges. In fact, within the entire third region, the surface flow appeared very sluggish with little if any movement (Note the lack of disturbance in the spanwise ‘bands’, produced when applying the paint). As incidence increased, it has already been noted that the partial stall effect occurred at the same incidence as the undamaged stall. It is suggested that this resulted from the collapse of the suction areas located between the holes. Flow visualisation also indicated that at the same incidence the surface flow separation point moved to the leading edge (See Figures 6.11(i) and (ii)). These combined effects resulted in the partial stall seen in the lift curve slope.

Following the partial stall, it was observed that flow through the majority of the holes remained minimal, although it appeared strongest for those holes located closest to the
leading edge (Figure 6.11(ii)). It was seen that together, the combined through-flow was sufficient to energise the upper surface flow and give some chordwise flow direction in the separation region up to the maximum incidence tested. This inhibited a ‘full’ stall and gave the lift curve slope recovery seen in Figure 6.6.

6.3.2.2 Lift Curve Slopes of more Constant Gradient.

With the exception of the three damage cases discussed above, the remainder of the cases tested exhibited reduced lift curve slopes with no partial stall effects. These damage conditions were of increased $\rho_t$ and hole diameter, e.g. 6% c at $\rho_t = 20/\text{m}^2$ results in Figure 6.6. Flow visualisation indicated that at low incidences, flow through the holes was limited. For the first few rows, the combination of boundary layer growth and cumulative wake development, previously identified as the first and second regions, was again observed (Point B - Figure 6.12).

When incidence was increased, so too did the amount of damage through-flow, which in turn increased the size of the separation around the individual holes. Also, due to increases in $\rho_t$ and/or damage diameter, the relative size of the gaps between the holes was reduced. (The ratio of ‘inter-hole gap to damage radii’ was now less than 12.) Together, this resulted in the joining together of the separation regions around each hole in the forward-most row of holes (See point C, Figure 6.13). This was seen to occur at relatively low incidences, $+4^\circ$ in one case. Not only did the separation inhibit any development of a suction peak, but also ‘forced’ the upper surface flow to separate at the first row (across the span of the wing). Over the remainder of the chord the previously observed third region of sluggish surface flow was not seen. Instead, the increased through flow produced surface flow in a chordwise direction, over the remaining surface. This flow would have been comprised of the combined wakes of the upstream holes, which remained attached to the upper surface. Unfortunately, surface pressure data were not available for missile damage cases, and it has not been possible to identify the details of the flow-field mechanics. However, it can be said that the effects are to delay a ‘full’ stall beyond $+14^\circ$, and to continue to generate a reduced level of lift.
Figure 6.12 Upper Surface Flow Visualisation (6%e at 20/m², 0 deg.)

Figure 6.13 Upper Surface Flow Visualisation (6%e at 20/m², +8 deg.)
Figure 6.14 Missile Damage $C_d$ values (Range of cases)

Undamaged 2.5% at 10/m2 4% at 20/m2 6% at 40/m2

Reynolds Number = 500,000
6.3.3 Drag Increments

In all missile damage cases, $C_d$ increased *significantly* over the entire incidence range tested. This resulted from both the large numbers of surface discontinuities (i.e. holes) and the formation of the extensive regions of surface separation. Figure 6.14 illustrates the range of $C_d$ values seen, with both the least and most severe missile damage cases tested. Compared with the undamaged wing minimum-lift $C_d$ of 0.0056, the least severe case (2.5%c at 10/m$^2$) resulted in a minimum-lift $C_d$ of 0.0113, a 102% increase. Whilst the worst case (6%c at 40/m$^2$) resulted in a corresponding value of 0.0724, a 1190% increase. Also, the damage reduced the range of $C_L$ between the drag rises.

Figure 6.15 indicates the drag increments $d[C_d]$ vs. incidence for the $\rho_f=20/m^2$ damage cases tested. Firstly, it was seen that for a fixed $\rho_f$, $d[C_d]$ values increased with hole size. This was consistent with the results found for gunfire damage. Secondly, $d[C_d]$ increased with incidence and exhibited the same trends for all cases. Indeed, similar curves were seen for each hole size, with each effectively translated up the $d[C_d]$ axis by different amounts.

For a constant hole size, $d[C_d]$ increments with incidence could be explained by increasing through-flow resulting in greater separation on the exit surface. With holes in the range 4%c to 8%c ($\rho_f=20/m^2$) showing similar surface separation from the forward most row of holes.

![Figure 6.15 Drag Increments for 20/m$^2$ Missile Damage](image)
(as discussed previously), it was not surprising that Figure 6.15 exhibited similar \( \Delta C_d \) gradients for these cases. This was seen to be true for similar forward separations at different \( \rho_f \) values. However, when some degree of undisturbed surface flow had been observed (as a result of surface separation not being 'forced' at the first row of holes), a reduced \( \Delta C_d \) gradient resulted above 0°. This was found to be the case for the damage which had previously indicated the partial stall effect followed by recovery (as for 2.5% at 20/m²).

In all cases the minimum drag occurred at -2°. This was where the least average through-flow would be expected (based on undamaged \( C_p \) values), and hence damage induced separation would be minimal. Plotting the minimum drag, it was found that for all cases tested, the values had an approximately linear relationship with percentage wing area lost (Figure 6.16). Varying incidence, it was found that \( \Delta C_d \) remained related to the total wing area removed by the damage, although the nature of the relationship varied with incidence (Figure 6.16).

![Figure 6.16 Drag Increase vs. Percentage Area Lost](image)

**Figure 6.16 Drag Increase vs. Percentage Area Lost**

### 6.3.4 Pitching Moment Effects

Figure 6.17 indicates \( C_m \) results for the \( \rho_f = 20/m^2 \) damage tests, and have been included here for completeness. The differences observed between the undamaged and damaged cases result from the combined effects of removed wing area, upper surface separation and suction area development.
Figure 6.17 Missile Damage Pitching Moment (20frag/m²)
It was found that a clear trend existed, in that the larger the damage size (for a given $\rho_f$) the larger the effect on $C_m$ values. At negative lift incidences, it was found that the greater the hole size, the less $C_m$ varied with incidence, and the closer the results became to zero $C_m$. For low positive incidences, both 2.5%c and 4%c results gave similar $C_m$ profiles, which were in themselves not too dissimilar to the undamaged wing. Whilst 6%c and 8%c results showed greater differences, reflecting the more significant changes seen at negative incidences. Interestingly, it was found that moving to higher positive incidences, 4%c, 6%c and 8%c results tended towards similar values. This may be attributable to the separation of the upper surface flow from the forward-most row of holes for these cases. It was found that similar variations with damage size were exhibited at all $\rho_f$ values tested, with the largest diameter holes at the highest $\rho_f$ giving the closest results to zero $C_m$. It might have been expected that after the partial stall seen for 2.5%c damage, $C_m$ values would have fallen in line with the other results. However this did not occur. Without the aid of surface pressure data for missile damage, this effect is difficult to explain, and in general it is not possible to comment further on these $C_m$ results.

6.4 Effects of Wing Construction on Missile Damage Characteristics

As with gunfire damage, the consequences of different internal wing construction were considered for missile damage. Findings indicated that the differences were found to be of a similar order of magnitude for both missile and gunfire damage. Typical missile results obtained are illustrated by Figures 6.18 to 6.20, which are for 4%c at $\rho_f=20/m^2$.

In general, it was found that the effects of damage on the solid wing coefficients followed very similar trends to those seen for the hollow wing, although the magnitudes of the effects were found to differ. Figure 6.18 illustrates that the solid wing had both the greatest rate of $d[C_L]$ change with incidence, and the greatest positive and negative $d[C_L]$ values (an 11.2% increase from -0.4813 to -0.5352, at $+10^\circ$ incidence).

Unlike gunfire damage, the differences between solid and hollow aerofoil $d[C_d]$ values were seen to increase with incidence (Figure 6.19). This may have been as a result of solid wing
Figure 6.18 Lift loss due to 4%c (20/m2) Missile Damage

\[ \text{Reynolds Number} = 500,000 \]

Figure 6.19 Drag increase due to 4%c (20/m2) Missile Damage

\[ \text{Reynolds Number} = 500,000 \]

Figure 6.20 Pitch variation due to 4%c (20/m2) Missile Damage

\[ \text{Reynolds Number} = 500,000 \]
through-flow influencing the development of the upper surface separation region. Possible increases in exit-flow from holes near the leading edge may have resulted in an increased propensity for separation, and thus increased $d[C_d]$ values. The exit-flow increase may have resulted from the inability of the through-flow to circulate internally within the wing and exit from a hole further 'downstream' than the entry hole.

$C_m$ measurements for both solid and hollow wings indicated similar effects, with solid construction again giving slightly greater $d[C_m]$ changes. Similar effects were again seen for damage of differing hole sizes.

6.5 Interpolation Method for Intermediate Damage Cases

Earlier, it was shown that to a first order, a relationship existed between $d[C_L]$ and wing area removed for the different combinations of $\rho F$ and damage size tested. Figure 6.5 illustrated results for $\pm 8^\circ$ incidence. It can be seen that within these $\pm 8^\circ$ results, certain cases did not reflect the overall trend. For example, 6% at 20/m$^2$ removed a slightly greater percentage wing area than 4% at 40/m$^2$, however, the change in $C_L$ was actually less. This was repeated at other incidence values, indicating that this was not an effect of data scatter, but a consequence of $\rho F$ and hole size.

Given the limited amount of experimental data gathered, it was not possible to identify the exact relationship between $d[C_L]$ and $\rho F$, hole size or percentage area removed. However, it was found that by contour plotting $d[C_d]$ results against hole diameter (%c) and $\rho F$, it was possible to interpolate to intermediate $d[C_L]$ values. Figure 6.21 illustrates such a $d[C_L]$ contour plot for $\pm 4^\circ$. The plot was based on all the available experimental $d[C_L]$ data point values at that incidence (The data point for 4%c at 20/m$^2$ has been illustrated). The contour plot was then produced by the application of linear interpolation techniques and plotting routines within the Pc based data-processing package MATLAB (Ref. 34). No curve fitting was applied. Similar plots were obtained at other incidences. As can be seen, by using these plots interpolation was possible to give $d[C_L]$ values for intermediate $\rho F$ and damage diameter.
values within the ranges 10 to 40/m² and 2.5% to 8% respectively. For example, 6% damage at 25/m² might be expected to result in a d[C_L] value of -0.552 at +4°.

Drag increments can be illustrated in a similar manner, as seen in Figure 6.22. For the above example, it can be seen that a d[C_d] value of 0.0748 might be expected at +4°. Similarly, for pitching moment (Figure 6.23), interpolation gives a value of d[C_m] equal to +0.022. Note, that as a result of 8% at ρ_t = 40/m² data being unobtainable, interpolated contours for +0.030 and greater were restricted to diameters less than 6% in size. If required, such data would have to be extrapolated from 8% data at ρ_t values of 10/m² and 20/m², or additional experimental data points added.
Figure 6.21 Contour Plot of $d[C_L]$ values (+4deg.)

NB: Linear interpolation used
Figure 6.22 Contour Plot of \(d[C_d]\) values (+4deg.)

NB: Linear interpolation used
Figure 6.23 Contour Plot of $d[C_m]$ values (+4 deg.)

- Hole diameter (%c)
- Fragment density (frag/m²)

NB: Linear interpolation used
CHAPTER 7. CONCLUSIONS

1. From the results of single-hole gunfire damage, the following conclusions were drawn;
   (i) For leading edge damage, the dominant flow mechanism was seen to be a pair of contra-rotating vortices close to the wing surface with a surface flow wake extending rearwards from the damage over the full chord length of the wing. At low incidences, the wake remained attached, with little if any effect on the surface flow either side of the wake. There was only a slight reduction in lift curve slope. At a critical incidence the surface flow detached, resulting in an enlarged region of separation and reverse flow. $C_L$ and $C_m$ values indicated a 'premature stall' followed by recovery. $C_d$ values were seen to increase across the incidence range tested. Increasing damage size increased the magnitudes of the coefficient changes measured, and reduced the critical incidence at which the wake detached from the surface.

(ii) From the locations tested, quarter-chord and to a lesser extent half-chord locations were found to be the most sensitive to damage. This may be attributed primarily to the effects of through-flow on the upper surface pressure distribution. Through-flow, driven by the pressure differential between the two surfaces was seen to take one of two forms. Firstly, a 'weak jet' forming an attached wake with minimal localised $C_p$ changes. $C_L$, $C_d$ and $C_m$ changes were relatively small. Increased incidence, or damage size, resulted in a 'strong jet' effect. Through-flow penetrated further into the crossflow, resulting in the detachment of the oncoming surface flow and development of a wake region of separation and reverse flow. Strong-jet coefficient effects indicated significantly greater lift-loss, drag increment and pitching moment changes than those seen for the weak jet. Lift losses resulted primarily from the reduction in pressure peak upstream and either side of the damage, whilst the $C_p$ distribution within the detached wake region resulted in significant $C_d$ increments and $C_m$ variations.

A number of similarities were identified to those of flat plate jets in crossflows. These primarily concerned both the surface and local flow-field mechanisms, whilst surface pressure changes observed were found to be significantly different. Importantly, it was
found that effects on upper surface $C_p$ values extended up to and beyond a spanwise distance of 5 damage radii from the damage centre.

(iii) Trailing edge damage was found to have the least detrimental effects on wing coefficient values. With little pressure difference between the upper and lower surfaces, no downstream surface flow to disturb and only minimal separation ahead of the damage, changes in $C_L$, $C_d$ and $C_m$ were minimal. The magnitudes of the effects seen increased consistently with damage size.

2. Analysis of the experimental gunfire damage data led to the development of a set of empirical relationships, which related damage location and size to the changes measured in $C_L$, $C_d$ and $C_m$;

(i) Results indicated that for each damage location tested, minimum changes in lift, drag and pitching moment coefficients were directly related to the incidence at which the undamaged pressure differential, between the upper and lower surfaces, at the damage centre was seen to be approximately zero.

(ii) For leading edge damage it was found that up to the point of premature stall, coefficient changes could be scaled by the damage percentage wing area. This scaling can be used to interpolate values for intermediate sizes of damage.

(iii) Quarter and half chord damage results led to the development of a lift-loss interpolation process based on the undamaged pressure differential at the damage centre location. The drag increments were shown to be related to the strength of jet observed, with strong-jet results being significantly greater than those of the weak-jet. Strong-jet drag increments were found to exhibit an approximately linear relationship with lift-losses. Finally, pitching moment changes for all damage sizes were seen to have similar linear variations with incidence, when strong-jet characteristics were observed. Comparisons of interpolated results with an independent validation test case indicated a good match over the majority of the incidence range for lift-loss and drag increment results, and over the whole incidence range for pitching moment changes.
(iv) Trailing edge data indicated that $C_L$ and $C_m$ changes were proportional to the product of the non-dimensional damage diameter and the undamaged trailing edge region normal force, and pitching moment, respectively. Whilst $C_d$ increments showed an approximately linear relationship with non-dimensional damage diameter alone. A good match between calculated and experimental results gave confidence in the applicability of the methods.

3. It was found that by combining the effects of single-hole characteristics, it was possible to predict the characteristics of the wing with two holes. Results indicated that the magnitudes of the effects predicted matched well to those found experimentally, over a restricted incidence range. This range extended up to the incidence at which the forward-most single-hole indicated the change from weak to strong-jet characteristics. Outside of this range, the superposition method consistently overestimated the changes in lift, drag and pitching-moment.

4. Simulated missile damage results indicated the following;

(i) In all cases tested, significant $C_L$ losses, $C_d$ increments and $C_m$ changes were measured. To a first order approximation, both lift and drag increments at a given incidence were related to the percentage wing area removed.

(ii) All cases failed to show fully stalled characteristics up to the maximum incidence tested. This was attributed to combined though-flow effects exerting control on the upper surface flow and energising the flow field.

(iii) Low fragment densities and smaller damage sizes resulted in a complex surface flow structure made up of boundary layer growth, attached wakes and detached surface flow. Individual hole patterns reflected similar flow mechanisms to those seen on a larger scale for gunfire damage cases. Increased fragment density and hole size resulted in upper surface separation at the first row of holes at low incidences.

(iv) From the results gathered over the test matrix, it was shown that changes in $C_L$, $C_d$ and $C_m$ results could be presented in the form of contour plots, from which interpolation to intermediate damage sizes or fragment densities was then possible.
5. Tests on the influence of Reynolds Number changes on the results indicated that the magnitudes and trends of the effects on coefficients were largely independent of Reynolds Number over the range tested, and were unlikely to vary significantly at higher values.

6. Investigations into the influence of internal model construction on both gunfire and missile damage indicated that observed trends in coefficient changes were similar for both hollow and solid wings. However, the magnitudes of the effects were found to be smaller for hollow wings than for solid wings (for both gunfire and missile damage). This indicated that for accurate damage modelling, wind tunnel models must reflect the true nature of the full scale wing construction. This confirmed the decision to use hollow wings for the investigation, ensuring both the accuracy and realism of the results.
CHAPTER 8. RECOMMENDED FURTHER WORK

It is recommended that further work is undertaken to vary some of the assumptions made during the initial investigations and to extend the tests undertaken so far;

(i) Unequal Entry & Exit Hole Sizes
The method of modelling gunfire damage assumed that entry and exit holes would be of the same diameter. In some cases, damage holes could have exit holes larger than entry holes. The results have shown clearly that the effects of damage change significantly with damage size. Such tests could investigate the relative importance of both upper and lower hole sizes.

(ii) Variation in Penetration Angle
Throughout, damage has been modelled using the basic assumption that the penetration angle, i.e. direction of attack, would be at 90 degrees to the chord line. This gave through-holes with entry and exit holes at the same chordwise location. The mechanism for driving flow through the aerofoil damage has been shown to be related to the pressure differential between the upper and lower surface damage locations. Tests should be undertaken to investigate combinations of gunfire damage with entry and exit holes located at different chordwise locations.

(iii) Sensitivity of Results to Damage Shape
Both gunfire and missile damage holes were modelled as circular in shape throughout. However, delamination effects, or the loss of panels, etc. might result in damage of a different shape; rectangular or diamond for example. By testing damage of different shapes, yet maintaining the same percentage area lost, it would be possible to gain an insight into 'read across' applicability of the data.

(iv) Missile Damage Surface Pressures
Upper and lower surface pressure distributions for missile damage cases would provide additional data relating to the coefficient changes and flow visualisation tests undertaken so far. This might confirm the suggested development of a limited pressure peak in the cases of minimal damage, and provide data on the upper surface pressure distribution within the
regions of separation.

(v) Internal Cavity Pressures
Pressure tapping of the internal box cavity of the hollow aerofoil would give an indication of the aerodynamic loads present on the internal surfaces of the wing. Such data could be used to give an indication of the loading on wing-panels when damaged. High loading might result in additional sections of panelling being ‘torn-off’ which would lead to secondary damage of a larger size than that caused by the initial explosive/fragmentation damage.

(vi) Sensitivity of Results to Aerofoil Profile
Throughout this investigation a ‘typical’ aerofoil (NACA 641-412) has been tested. It has been identified that the results gained are influenced by the pressure distributions across the aerofoil profile. It is suggested that further tests on a different profile should be undertaken, to investigate further the applicability of the empirical relationships presented here.

(vii) Span wise Distribution of Gunfire Damage
Multi-hit gunfire damage should be extended to include damage separated in a spanwise direction. Surface pressure data has shown that spanwise influences may be extensive. Further tests could investigate minimum spanwise separation distances, such that the effects of holes may be treated as aerodynamically independent. Where hole separations are less than the minimum required, it should be possible to assess inter-hole interference effects.

(viii) Partial Wing Coverage of Missile Damage
Missile damage tests considered the uniform distribution of damage over the full wing span. Further tests might have only a limited span of the wing subjected to simulated missile damage, with the remainder undamaged. The interference effects present in the transition region between areas of missile damaged and undamaged wing might be assessed, and developed into a form of strip-theory.

(ix) Detailed Experimental Flow-field Measurements
Detailed flow-field measurements could be made by traversing hot-wire or pressure probes through the region of flow effected by the damage. The field of interest would extend from
within the hollow aerofoil damage through to the free-flow region, both above and below the aerofoil. Tests would involve one or two key damage configurations.

(x) CFD Prediction Calculations
Initial progress in this area was not realistic due to the complexity of the flow, and required detailed experimental work to be undertaken prior to commencement. However, if internal cavity pressure data and detailed flow-field measurements were available for validation purposes, it may be possible to make an attempt at predicting damage effects using Computational Fluid Dynamics (CFD) codes.

(xi) Finally, any further work must attempt to extend the damage characteristics from the current damaged section of a two-dimensional infinite aerofoil, to a finite three-dimensional wing. By studying a half-wing, the presence of wing-tip effects, taper and sweep could be investigated. Tests might include both the spanwise and chordwise distribution of gunfire and missile damage.
REFERENCES


20. "Investigation of Characteristics of Airflow Within Aircraft Structures - Phase B(1)"
Cornell, V-699-D-17, 1952.


28. L.K. Loftin, H.A. Smith, “Aerodynamic Characteristics of 15 NACA Airfoil Sections at Seven Reynolds Numbers from 0.7x10^6 to 9.0x10^6”, National Advisory Committee for Aeronautics, Technical Note 1945


APPENDIX A - NACA 64₁-412 AEROFOIL DETAILS

The NACA 64₁-412 aerofoil section was used throughout the test programme. The section was based on section requirements as indicated by British Aerospace (Military Aircraft) technical specialists.

Section Data

At Reynolds Number = 700 000.

\[
\begin{align*}
\text{Design } C_L & = 0.4 \\
C_{L,max} & = 1.19 \\
\alpha_0 & = -2.8^\circ \\
C_{d,min} & = 0.007 \\
C_{m,c/4} & = -0.08 \\
\end{align*}
\]

Profile Generation

The aerofoil section is produced by combining the following thickness profile and camber line.

- Thickness Profile : NACA 64₁-012
- Camber line : 40% of a = 1.0
NACA 641-412 Model wing section stations and ordinates given in percent of aerofoil chord.

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APPENDIX B - Determination of Missile Damage Grid Angle.

The following outlines the method used to determine the required missile damage grid rotation angle $\theta$, required to correctly orientate the uniform grid of holes relative to the aerofoil leading edge. Each value of $\rho_f$ tested had a corresponding value of $\theta$.

Assuming,

Full size chord $= 2000mm$
Model Chord $= 200mm$
$\rho_f = 10/m^2$

Equation (6.1) gives the model grid spacing value as;

$$a = 31.6mm$$

Now, to calculate the grid angle, $\theta$, an iterative process is required.

Firstly, an initial value for $N$, the maximum number of holes in a row is calculated. On the first iteration, the row length is set equal to the model chord length;

$$N = \text{Integer} \left\lfloor \frac{\text{Row length}}{a} - 0.5 \right\rfloor + 1 \quad \text{... (B.1)}$$

$$N = 7$$

Given that the distance from the first to $N^{th}$ hole along such a row is given by $a(N-1)$, to place the first hole in the row (Figure B1, point A) directly upstream of the last hole in the preceding row (point B);

$$\theta = \tan^{-1} \left( \frac{a}{a(N-1)} \right) \quad \text{... (B.2)}$$

$$\theta = 9.5^\circ$$
Having rotated the row AC by $\theta$, the next iteration determines if the number of holes along AC has increased;

Calculate the new row length, from intercept on leading edge to intercept on trailing edge;

$$\text{Row AC Length} = \frac{\text{Model Chord}}{\cos \theta} \quad \cdots \text{(B.3)}$$

$$= 202.76 \text{ mm}$$

Now, recalculating the number of holes on the new length using equation (B.1) gives;

$$N = 7$$

i.e. the number of holes on the row has not increased due to it's rotation.

Thus, 7 holes are located on the row, rotated at an angle $\theta = 9.5^\circ$.
APPENDIX C. Publications from this project.


Further publications from this project are currently planned.