

CRANFIELD UNIVERSITY

Besant Bhakta

**NOVEL ICE PROTECTION SYSTEM SUITABLE FOR
UNMANNED AERIAL VEHICLE WING**

School of Engineering

PhD

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2005

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**NOVEL ICE PROTECTION SYSTEM SUITABLE FOR
UAV COMPOSITE WINGS**

Dr. Robert Jones

1 Feb 05

"This thesis is submitted in partial fulfilment of the requirements for the Degree of PhD."

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ABSTRACT

The existence of an atmospheric icing threat to aircraft operating in moist, low altitudes at temperatures below freezing is well known.

The objective of this study was to develop an aircraft ice protection system suitable for a UAV composite wing. Conventional wing leading edge ice protection systems were examined and found to either, necessitate significant electrical power, or were costly with respect to system mass.

A low cost and low power technology capable of protecting the UAV wing leading edge was identified. It was proposed that a commercial magnetostrictive actuator fitted to and in direct contact with the non-airflow wing surface would provide mechanical impulses to break the ice-wing surface bond.

Assuming the accreted ice was of a form expected of test points in the FAR/JAR Appendix C flight-icing envelope. Computational simulations demonstrated that pairs of magnetostrictive actuators acting in unison at a 0.3m span spacing, and deployed along the upper and lower wing leading edge surfaces at around the 7% chord coordinate, would successfully break the ice-surface bond. It was estimated that for a medium endurance UAV of Predator B class, the proposed system power requirements is 500W at a 45kg total system mass.

The proposed system would be more competitive than conventional systems if the use of consumer grade electronics and control systems, harnessing etc. were permitted, together with the removal of system redundancy and fail-safe provision requirements necessary for manned aircraft.

Further work would require the demonstration of a physical de-icing installation in the icing wind tunnel.

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CHAPTER 1: INTRODUCTION

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1. INTRODUCTION

1.1. Project Objectives

The objectives of this study was to develop an aircraft ice protection system suitable for an Unmanned Aerial Vehicle by:

- Identifying a suitable low cost/ low power technology capable of protecting the UAV wing leading edge.
- Demonstrating the feasibility of this low power/ low cost technology by means physical and computational simulation.
- Understanding the potential advantages and limitations of such a system on the effectiveness on UAV mission.

1.2. Background

Ice accretion occurs when an aircraft strikes cloud water droplets at temperatures below freezing. Super cooled water droplets are inherently unstable, and will generally crystallise on impact with the aircraft surface as shown in Figure 1. Ice accretion has a profound effect on the flow of air over aerofoil surfaces.



Figure 1: Ice Accretion On Wing Leading Edge (OHC Twin Otter)¹

Ice presents the following threats to aircraft safety:

- Increased drag on wings, stall speed increases
- Structural damage due to ice shedding
- Change in airflow distribution to engine causing stall or flame out
- Icing behind flaps & spars causing control problems.

¹ : Courtesy NASA Glenn Research Centre. <http://icebox-esn.grc.nasa.gov>.

1.2.1. Atmospheric Icing Environment

Source: FAA Icing Handbook DOT/FAA/CT8-88/B-1 Vol. 1 (Mar 91)

The super cooled cloud water droplets are present in the clouds, when they impinge on the aircraft surface as shown in Figure 2. The droplet size and population are unstable, and some droplets grow in size at the expense of others. This growth influences the rate of aircraft icing through the increase in collection efficiency.

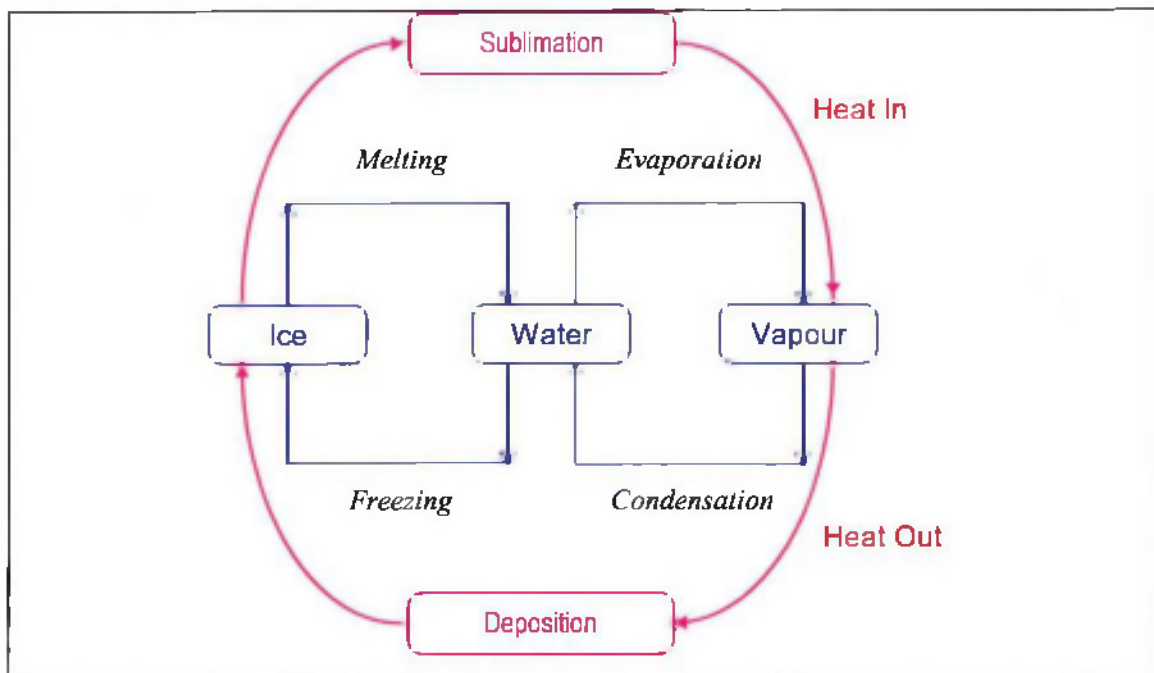


Figure 2: Physical Changes To Cloud Water Droplets. *Source: FAA Icing Handbook DOT/FAA/CT8-88/B-1*

There are 2 mechanisms by which the cloud droplets grow:

a) Coalescence:

The different fall velocities of the cloud droplets cause coalescence into larger droplet sizes. With most clouds, a vertical air current results in the large droplets sweeping up the smaller droplets, and taking more time to fall. The longer the time spent in the cloud, the larger the droplets grow (particularly in mountain and hilly terrain's where there are strong updrafts) to raindrop sizes. This is not so in still air conditions, where the large droplets have less time to capture the smaller droplets in their paths.

Cloud water droplets are acted upon by three forces: (1) the gravitational force; (2) buoyancy force and (3) the drag force. When a droplet falls at speed, the gravity balances the difference between buoyancy and gravity. This means the droplet rapidly reaches terminal velocity. For small droplets, the terminal velocity is very low. Larger droplets will have a greater terminal velocity, thus allowing impact and coalescence with smaller droplets.

b) Growth in Ice Crystals:

Here the coexistence of ice crystals and super cooled water droplets in the cloud result in the tendency for the ice crystals to grow at the expense of smaller water droplets. This is the case at freezing temperatures, when the saturation vapour pressure over the ice surface is less than that over the liquid surface, resulting in a pressure force over the ice surface being less than that over a liquid surface. The outcome of this is there is a pressure force driving water molecules from the water into ice.

1.2.2. Characteristics of Aircraft Icing

Understanding the physics of aircraft icing permits determination of the area of the aerofoil to be protected, the rate of ice collection, and therefore the rate at which ice must be removed.

1) Some of the physical variables affecting the aircraft ice accretion rate and resultant ice shape are²:

- a) Cloud liquid water content (0.1g/m^3 to 2.8g/m^3) which produce larger "horn-like" features on the forward aspect of wing leading edge with increasing liquid water density. This can be attributed to larger quantities of water freezing.

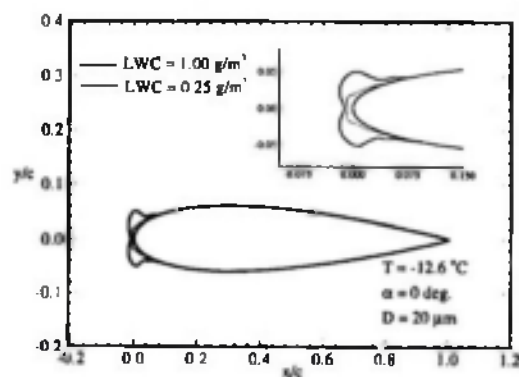


Figure 3: Effect of Liquid Water Content of Ice Shape. *Source: Brahimi et al*

- b) Droplet sizes (0.02mm to 0.15mm) which tend to move the "horn like" feature aft of the leading edge with increasing droplet size. This is due to greater "splash".

²: "Effect of Super Cooled Large Droplets on Ice Accretion Characteristics", Brahimi M.T; Tran P.; Chcaron D.; Tezak F. & Paraschivaiu I., AIAA-97-0306 (Jan 97).

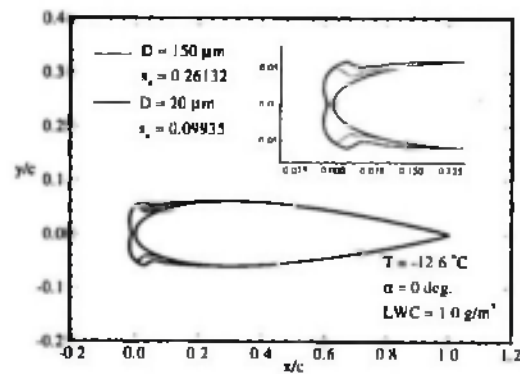


Figure 4: Effect of Droplet Diameter on Ice Shape. *Source: Brahimi et al*

- c) Air temperature (0°C to -30°C) reduction results in droplets increasingly freezing on contact and building the ice horn forwards.

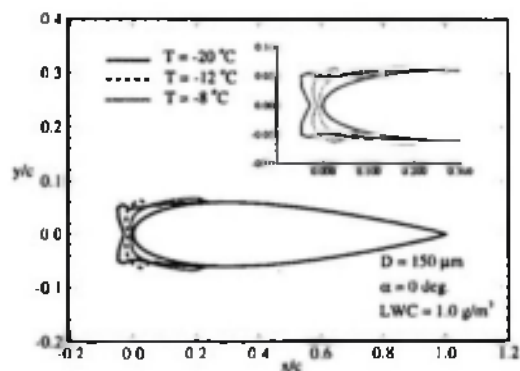


Figure 5: Effect of Air Temperature On Ice Shape. *Source: Brahimi et al*

- d) Altitude (ground to 7000m) has the effect of increasing the probability of encountering super cooled water droplets.

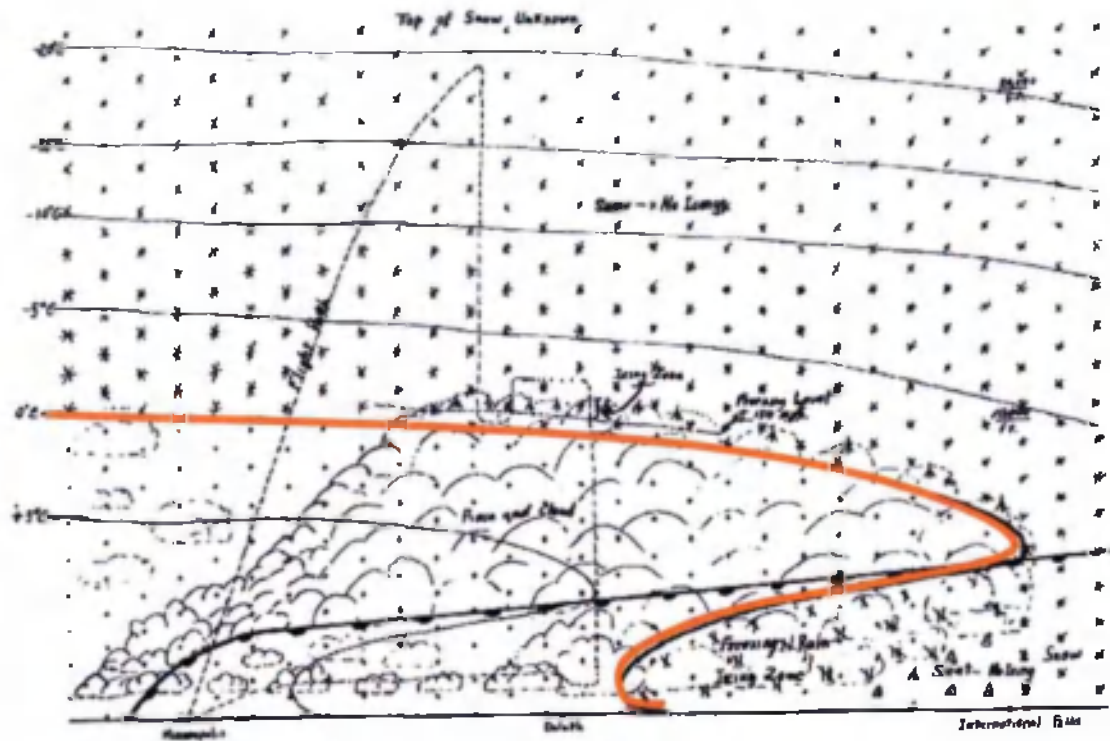


Figure 6: Orange Line Indicating Demarcation between Icing Threat Region & Ice Clear Regions from Ground to 6100m, "Cross-Section of Warm Front Cloud System over Minnesota", 15Mar45 at 9:30.³

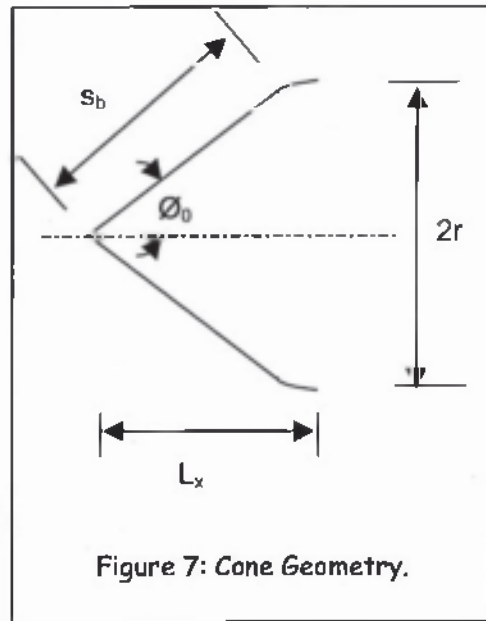
e) Aerofoil Geometry

Determination of the impingement limits will depend on the complexity of the intercepting aerofoil geometry. At its simplest for a cone, the impingement limits can be evaluated as follows:

³: NACA Technical Note No. 1392, Icing Zones in A Warm Front System With General Precipitation, W Lewis (Jul47).

$$\text{Cone Frontal Area } A = \pi s_b \cdot r = \pi L_x^2 \sin^2 \phi_0 \dots\dots$$

Equation 1



Where S_b = impingement limit, ϕ = cone angle, L_x = chord length being considered.

The limits can then be used to determine how far aft of the leading edge the surface requires protection.

f) Aircraft velocity.

The effect of forward velocity shifts the temperature at which icing starts to occur due to kinetic heating.

The Water Catch of a body is defined as the amount of water (or ice) that impinges on the aircraft surface. It is expressed in lb/hour or g/hour. For a 2D body (e.g. a wing profile), the water catch can be expressed in terms of a unit span.

The fundamental equation for water catch⁴ (kg/s/m) of span is:

$$M = v A(LWC)E_m \dots\dots\dots$$

Equation 2

where v = free-stream velocity (m/s - true air speed)

A = aerofoil projected height (m)

LWC = cloud liquid water content (g/m³)

E_m = water catch efficiency (i.e. a function of velocity, aerofoil shape and thickness for a given volume median droplet size, ambient temperature and thickness)

Data has been published to determine E_m value for a variety of aerofoils⁵.

A simplified parameter called the "Modified Inertia Parameter, K_0 " has since been developed⁶. This method was used to generate a set of K_0 universal plots of water catch efficiency (E_m) and upper and lower limits of ice impingement for aerofoils, engine inlets, and various body shapes.

⁴: "Rotor Blade Electro-thermal Ice Protection Design Considerations", SAE Aerospace Information Report AIR1667 (Jul96).

⁵: "Determination of rate, area & distribution of impingement of water drops on various aerofoils from trajectories obtained on the differential analyser", Guibert A.G, Janssen E. & Robbins W.M -Add' I, (Apr 49).

⁶ : "Rime Ice Accretion & Its Effect on Aerofoil Performance", Bragg, M.B, NASA CR165599 (Mar82).

An example of this is shown in Figure 8, where the water catch efficiency (E_m) for aerofoils, engine inlets and other geometry shapes is plotted against the modified inertia parameter K_0 . Each geometry shape is represented by a single curve. The plots are generally within $\pm 10\%$ of the original data for the Reynolds number range 100 - 250.

The modified inertia parameter K_0 factor can be determined by the Equation 3.

$$K_0 = \left(\frac{d_{med} \cdot 2v \cdot \rho_w}{C \cdot \mu_0} \times R^{-\frac{2}{3}} \right) - \left[\left(\frac{\sqrt{6}}{R} \right) \cdot \text{Arc tan} \left(\frac{R^{\frac{1}{3}}}{\sqrt{6}} \right) \right] \dots\dots\dots$$

Equation 3

- Where
- v = free stream velocity (m/s)
 - μ_0 = air viscosity at temperature (kg/m s)
 - d_{med} = water droplet diameter (μ m)
 - R = Reynolds number = $\frac{\rho \cdot D_{med} \cdot v}{\mu_0}$
 - C = aerofoil chord length (m)
 - ρ_w = droplet density (kg/m³)
 - ρ = ambient air density (kg/m³)

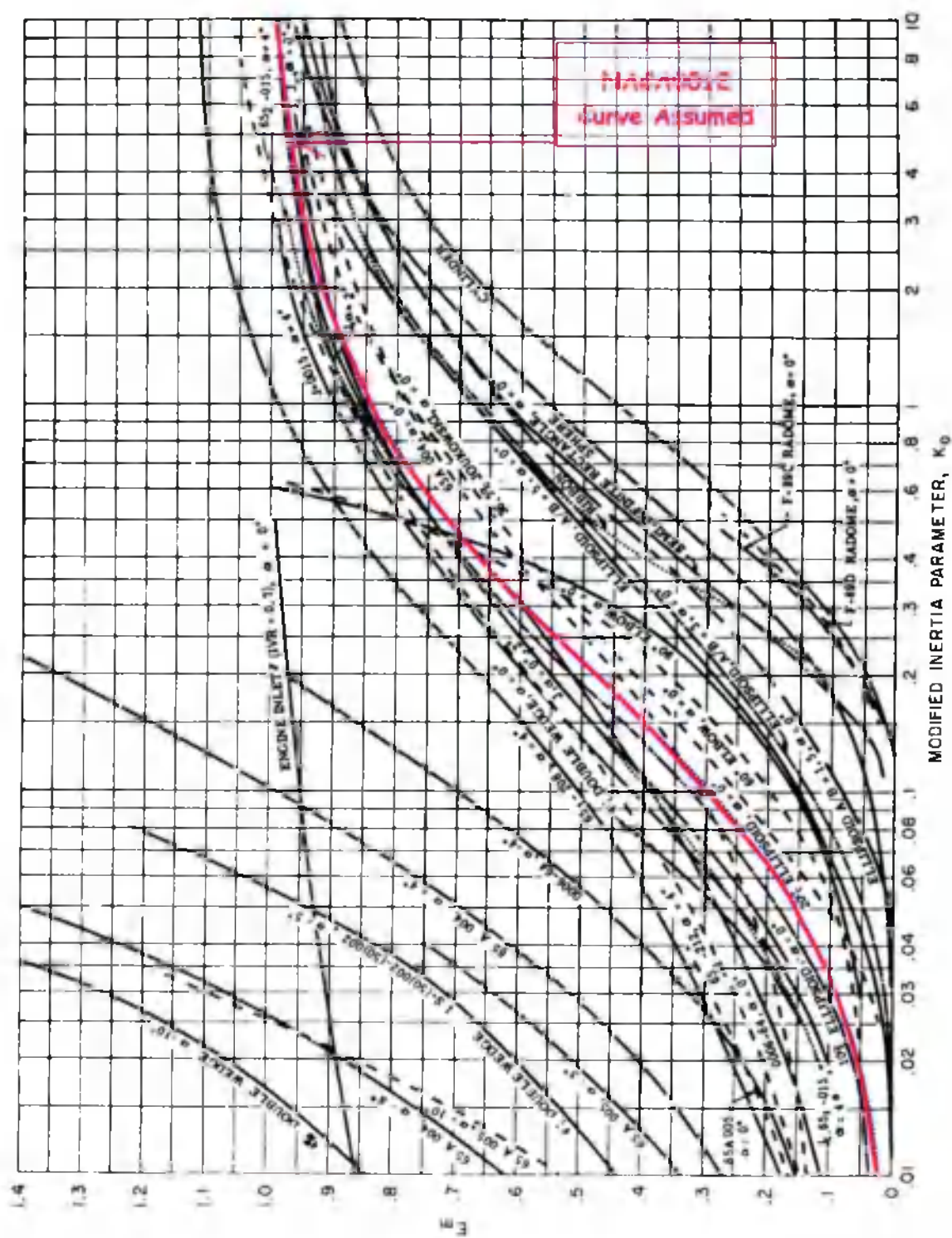


Figure 8: Water Catch Efficiencies For Various Aerofoils and Geometric Shapes. *Source: SAE Aerospace Information Report AIR1667 (Jul96).*

Industry practice for analysis and experimentation is NACA0012 wing profile exposed to icing conditions with a liquid water content (LWC) 1g/m^3 . The NACA0012 wing profile is a relatively simple, symmetric profile, which reduces geometric effects from calculations. The 1.0g/m^3 is a convenient design case LWC value also ensures the majority of "FAR23/Appendix C-continuous maximum" icing encounters (Stratiform cloud) at the UAV operation altitudes are covered, in addition to it being a useful standard for comparing the performance of various ice protection systems. As a result, a significant database of case studies is available for reference.

Using atmospheric condition where the liquid water content is 1g/m^3 , a NACA0012 wing with a span of 1m at 1800m in a free stream velocity of 41m/s, a modified inertia parameter of 14.74 is obtained. Calculations are shown in Appendix A1. With reference to Figure 8, an ice accretion Collection Efficiency of 0.9 is obtained. This in turn suggests, the mass of water intercepted per span meter is approximately 4.1g/m/s . A UAV wing ice protection system would therefore need to cope with this maximum volume of super cooled water droplets.

g) The forms of icing on an aircraft wing can be classified into two distinct forms:

i) "Horn like" or "glaze" ice - see Figure 9.

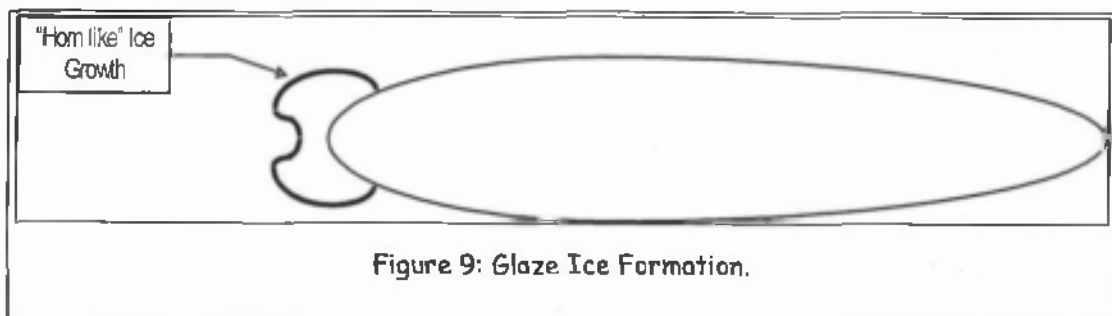
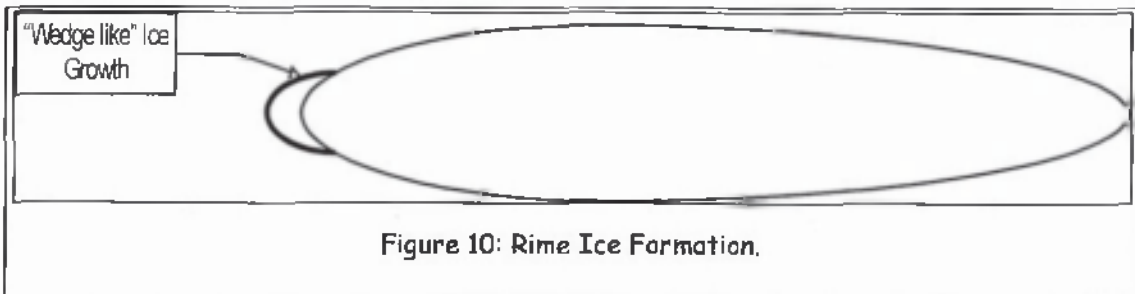
(1) Stratiform clouds: prevalent in winter, occurs at higher ambient temperatures (-7°C to 0°C), long duration icing encounters (1 to 500km)

(2) Cloud water content is highest, large water droplets, more numerous, LWC: 0.3 to 0.8g/m^3

(3) Large quantities of latent heat of fusion liberated resulting in the raising of surface temperatures. This permits further water crystallisation. Water droplets spread on the aerofoil surface before cooling and crystallising, causing the ice formation to be a greater extent on the wing surface

(4) Specific density ~ 0.6 to 0.9g/cm^3

(5) Drag can increase to between 200 and 300%



ii) "Wedge like" or "rime" ice - see Figure 10.

(1) Cumuliform clouds: prevalent in the summer, occurs at lower ambient temperatures ($< -10^\circ\text{C}$), short intense icing encounters (0.3 to 9.5km)

(2) Cloud water content is lower, small size water droplets, less numerous than above, LWC: 0.3 to 2.8g/m^3 (tropical conditions to 8g/m^3)

(3) Water droplets crystallise on impact with aerofoil surface

(4) Specific density ~ 0.2 to 0.6g/cm^3

(5) Drag can increase up to 10%

iii) *Mixed Conditions* - in this condition, varying amounts of rime ice, glaze ice, running water (due to electro-thermal heating for instance) can also prevail.

h) *Wet Snow* - Wet snow (near freezing temperatures) tends to cling to exposed surfaces and is likely to form a narrow hard ridge on the stagnation line.

i) *Freezing Rain or Drizzle* - precipitation of large water droplets particularly in mild conditions results in flow on the aerofoil surface of most of the impinging water. Freezing is not instantaneous in these conditions.

1.2.3. Effect of Icing On Aircraft Performance

The effects on aerodynamic performance on a commercial transport and business jet aerofoils with varying levels of glaze ice shapes were examined by Addy et al ⁷. No equivalent data was found for UAV based aerofoils. Ice shapes based on FAR/JAR 25 Appendix C test conditions were developed from icing tunnel tests. These shapes were transferred and converted into a wooden block pattern and bonded to the aerofoil. A force balance was then used to measure forces normal to and parallel to the aerofoil chord, and a wake survey used to measure aerofoil section drag. A pitot-static probe measured free-stream conditions. Absolute and differential transducers determined total and static pressures respectively.

⁷: "Modern Aerofoil Ice Accretions", Addy Jr H.E; Potapczuk, M.G; Sheldon D.W, AIAA-97-0174 (Jan 97).

Following repeatability checks and wind tunnel corrections, the following glaze ice shapes at varying cloud exposure times (Business Jet - clean 0min, 2min, 6min & 22.5min/ Commercial - clean 0min, 2min, 4.9min, & 18.5min) were obtained and displayed in Figure 11 and Figure 12 below. These figures show the cross-section profile of the accreted ice shapes at -0.8°C total air temperature.

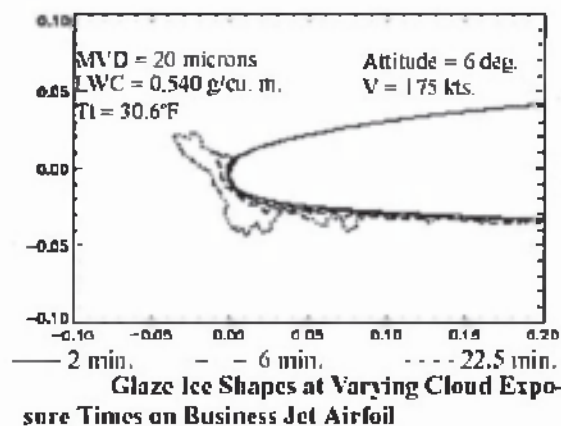


Figure 11: Glaze Ice Shapes V Exposure Times For Business Jet Aerofoil. *Source: Addy Jr et al.*

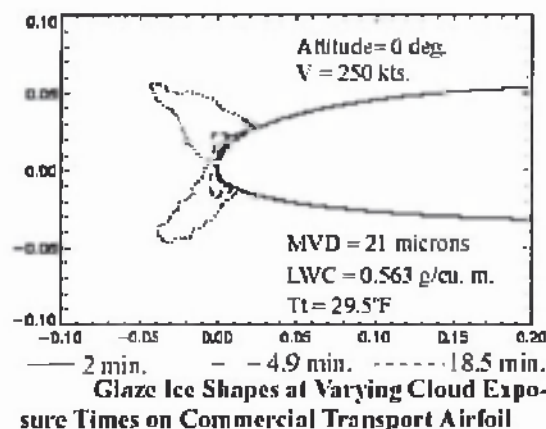


Figure 12: Glaze Ice Shapes V Exposure Times for Commercial Transport Aerofoil.

Source: Addy Jr et al.

Focusing on the business jet, the graph in Figure 13 demonstrates the reduction in the measured lift coefficients when compared with an ice-free (0 minutes) aerofoil. Measured lift reductions observed at 2 minutes was 17%, 6minutes was 21% and at 22.5minutes at 30% respectively.

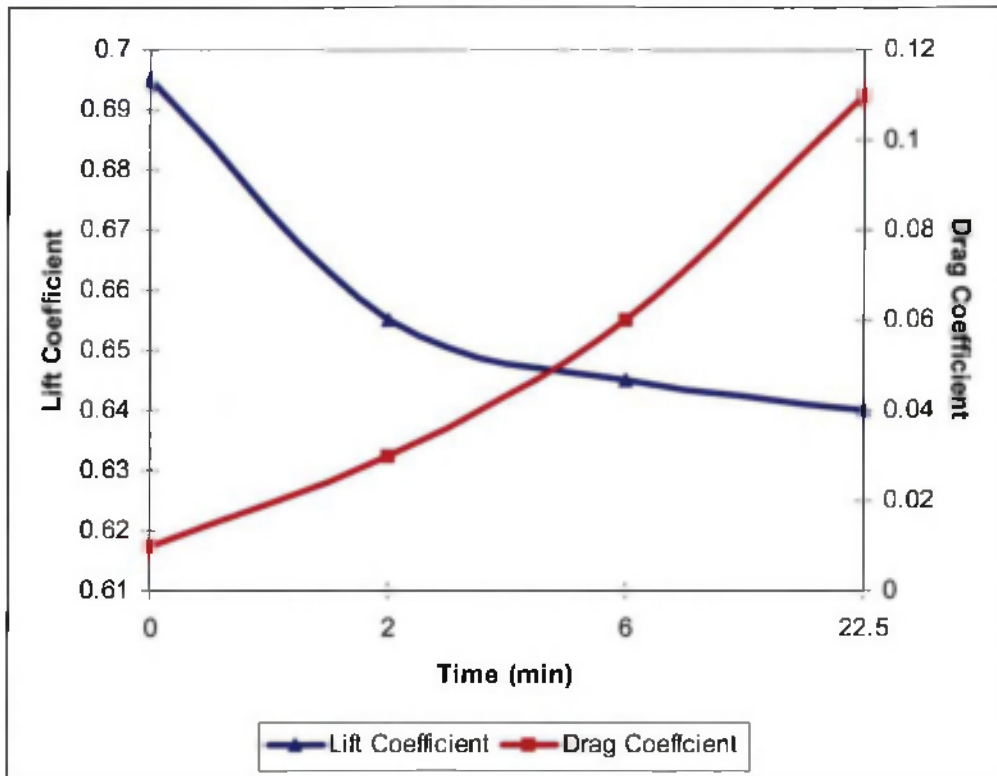


Figure 13: Lift & Drag Coefficients For Business Jet Aerofall 6° Attitude, -0.8°C @ 175 knots. *Source: Addy Jr et al.*

Temperature has a corresponding effect on lift and drag coefficients, where the warmer icing conditions generate ice shapes, which are detrimental to wing aerodynamic performance.

This is shown in Figure 14 below where the clean wing lift and drag coefficients are compared with those of the wing contaminated after 6 minutes into an icing encounter with rime ice (-10.9°C), mixed ice (-6.2°C) and glaze ice (-0.9°C) respectively.

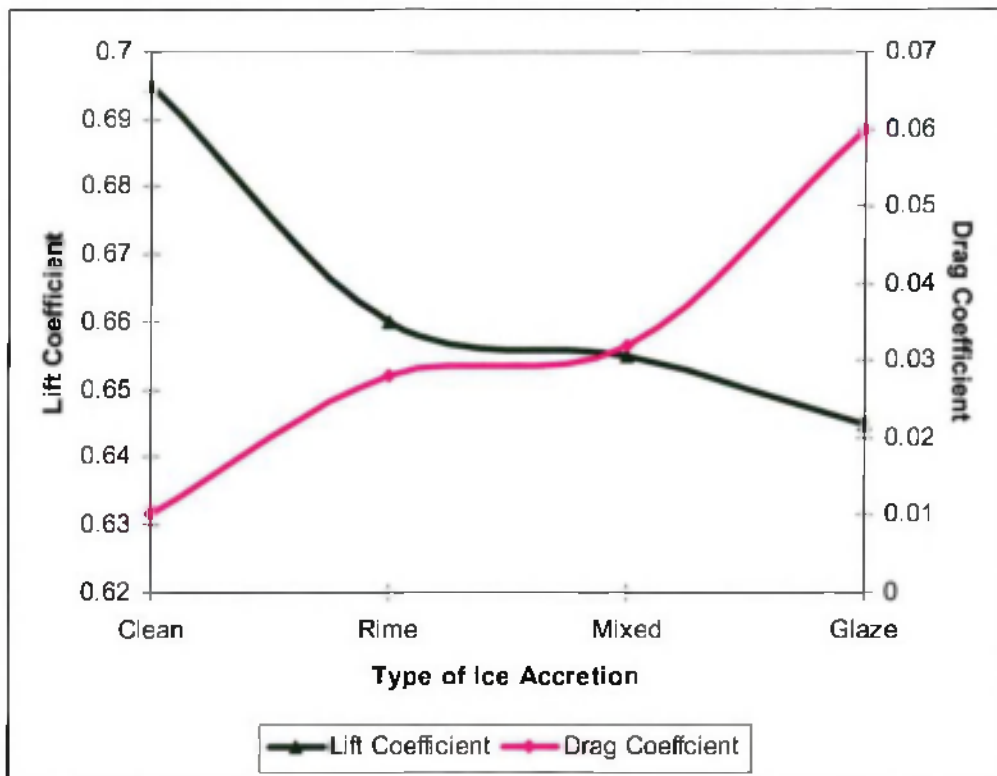


Figure 14: Lift & Drag Coefficients For Various Ice Shapes. *Source: Addy Jr et al.*

The paper does suggest that much of the decrease in lift occurs early on in the icing encounter, which tends to level off. Drag however continues to increase strongly with time.

1.3. Unmanned Aerial Vehicles (UAV)

1.3.1. Description of UAV System

(ref: <http://www.fas.org/irp/program/collect/uav.htm>)

There are varieties of UAV systems in the market place. The aircraft are usually employed by the military in reconnaissance, surveillance, and target acquisition roles. In general, three classifications of UAV system exist based on their operational ranges: Close (<50km), Medium range (<200km) and Endurance (>200km). Within these classifications, there are overlapping mission specific categories such as Tactical, Medium Altitude, and High Altitude aircraft. For this aircraft icing study, the former two classes are of interest.

An example of the "tactical" class is Phoenix UAV system primarily operated by UK forces, where it is used by the Royal Artillery as a primary Battlefield Surveillance and Targeting System⁸ - see Figure 15.



Figure 15: Phoenix UAV in flight (Photo: courtesy of DERA, UK)

The Phoenix is able to provide real time surveillance of the battlefield in most weather conditions. The UAV delivers target identification and position data to the command centre and artillery units. It has been involved in NATO peacekeeping missions, notably in Kosovo. The Phoenix has flown over 700 flights, making it one of the most successful and reliable UAV's in operation.

The vehicle is a modular construction made from composites including Kevlar Carbon fibre and Glass fibre reinforced plastic with sections of Nomex Honeycomb panels.

The Phoenix is able to fly on a pre programmed course or can be piloted from the vehicle controller. The payload of the Phoenix is carried out in a pod mounted on the underside of the fuselage.

Table 1: PHOENIX UAV STATISTICS

Length	3.8m
Wingspan	5.5m

⁸ : Reference <http://www.army-technology.com/projects/phoenix/index.html>

Table 1: PHOENIX UAV STATISTICS

Maximum launch weight	175kg
Mission payload weight	50kg
PROPULSION	
Motor	WAE 342, two stroke, flat twin fuel injection, 19kW
Propeller	Two blade fixed pitch wooden propeller, 780 mm
Generator capacity	Plessey 900 watts
PERFORMANCE	
Maximum speed	55km/hr
Flight endurance	4 hours
Radius of operation	50 km
Maximum altitude	2,750m
Launch & Recovery	Truck-mounted hydraulic catapult, parachute - airbag

An example of medium altitude aircraft is Predator⁹ operated by US Air force and displayed in Figure 16.

This aircraft has a greater capability in terms of payload and mission capabilities to that offered by the Tactical system. The aircraft is designed for rapid turnaround in data to aid the robust targeting cycle. The instrumentation on-board includes colour nose camera, synthetic aperture radar, satellite data link, laser designation etc. It also can be armed with two laser guided hellfire anti-tank missiles. The aircraft is operated beyond the line of sight. With respect to this study, the aircraft does possess an aircraft icing mitigation system for the main wings.

Table 2: PREDATOR UAV STATISTICS

Length	8.22m
Wingspan	14.8m
Maximum launch weight	1020kg

⁹: <http://www.ga.com> & www.af.mil/factsheets

Table 2: PREDATOR UAV STATISTICS

Mission payload weight	512kg
PROPULSION	
Motor	Rotax 914 four cylinder, 75kW (101hp)
Propeller	Two blade fixed pitch wooden propeller, 780 mm
Generator capacity	4000 watts (estimated)
PERFORMANCE	
Maximum speed	212km/hr
Maximum flight endurance	40 hours
Radius of operation	741 km
Maximum altitude	7,620m
Launch & Recovery	Requires Hard Surface Runway



Figure 16: Predator UAV system (Photo: courtesy of General Atomic, USA)

1.3.2. Regions Where An Icing Encounter Is Likely

It has been reported in the literature¹⁰ that adverse weather conditions have an impact on the air vehicle and sensors performance. Two Predator B aircraft have been reported lost during operations in Kosovo, and aircraft icing were sighted as contributing factors.¹¹ Suggested solutions include rapid descent in icing conditions to reduce the effects of any substantial ice accretion, and training to accustom operators to meteorological effects.

¹⁰: http://www.fas.org/irp/doddir/udaf/concops_uav/part02.htm Para 3.8

¹¹: http://www.fas.org/irp/program/collect/docs/n19991223_992294.htm

Figure 17¹² displays highlighted regions identified as intermediate zones, cold and severe cold zones in the world. Aircraft flying within these zones carry increasing risk of ice contamination of surfaces, particularly at surveillance attitudes and durations expected of UAV systems. UAV operations in these areas would require consideration of ice protection or mitigation.

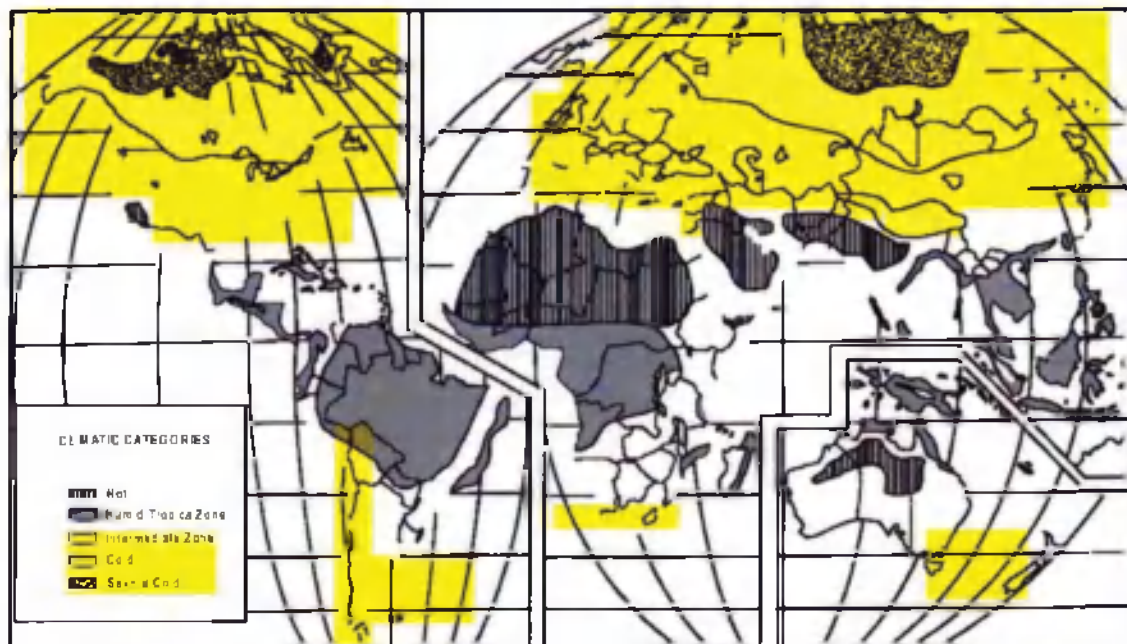


Figure 17: Highlighted Regions identified as "Intermediate, Cold, & Severe Cold" Zones Carry Risk of Aircraft Icing.¹²

It should be noted that icing threats to aircraft are not confined to high latitudes highlighted in Figure 17.

¹² : US Military Standard For Environmental Considerations Mil-STD-810F, Appendix C, Fig C-1.

1.3.3. Power, Performance & Weight Limitations Specific to UAV Systems.

As can be seen in Table 1 and Table 2, the aircraft have modest power generation capacity, much of which would be deployed in supporting the payload and remote piloting system. Thus, electrical power is at a premium and any additional power generation capacity would need to be balanced against the benefit an ice protection system would bring to the mission.

To determine the benefits of such an ice protection system, a first order model study based of additional fuel requirements and drag penalty is shown in the Appendix A.

1.3.4. Advantages For UAV Systems With A Wing Ice Protection System.

With reference to Mil-STD-810F¹² Para 2.3.1 climatic conditions and Figure 17, the regions deemed "intermediate and cold" zones, the probabilities of occurrence of extreme low temperatures below -51°C as a percentage of total hours in the most extreme low temperature month is 20%. This figure seems realistic when the UAV operating altitude is also considered.

Accident report released by the US Air Force⁹ indicated the loss of a RQ-1 Predator Aircraft on 18th of April 1999 in Kosovo was partially due was the aircraft experiencing icing.

A separate study by the US Defence Airborne Office¹³, states, "...Due to the worst winter in 40 years in Europe, little flying was accomplished as prevailing weather conditions were below flight minimums...". This statement is despite this Predator aircraft configuration having an installation of a de-icing system that disperses ethylene glycol over the wings.

The author has seen a study on ice accretion effects on the Phoenix aircraft¹⁴. Due to confidentiality requirements, no reference can be made to this report. However, the study does demonstrate an icing threat exists and mission capabilities are compromised.

It is clear from the above investigations that an ice protection system fitted to UAV systems would enable the aircraft missions to be undertaken in adverse weather conditions, particularly where there is a high probability of an icing encounter. It would also meet the objectives of military planners of requiring "continuous 24-hour target area presence".

1.4. Report Structure & Contents

1.4.1. Literature Survey

A literature survey of public, commercial, and military databases and discussions with acknowledged experts within the field was undertaken. Areas of particular interest that were investigated were:

- Examination of the icing threat to aircraft and specifically UAV.

¹³: "Modelling & Simulation Employed in the Predator UAV Programme", Israel K.R (Maj Gen), 7 Mar 97. <http://www.fas.org/irp/agency/dara/product/predms.htm>

¹⁴ : "Phoenix Icing Trials Data", Usher. M.C, (Mar 00), DERA Report Ref. AS/SID/350/1/24

- Review current aircraft ice protection systems and their suitability for UAV's.
- Survey alternative technologies that could be used for UAV wing ice protection. These technologies could be derived from existing commercial, industrial or consumer products and refined for UAV application.

The information gathered was collated and summarised with the view to meeting the project objectives defined in section 1.1 above.

1.4.2. Physical Experimentation & Computational Simulation.

The literature survey identified a technology suitable for a UAV wing leading ice protection system. The programme of work then:

- Described the UAV wing ice protection system functioning with the alternative novel technology.
- Studied the behaviour of the novel ice protection system on a flat composite panel.
- Developed a FEA model of the wing and verify the accuracy of the model as a means for simulation.
- Take a known icing condition as defined within FAR/JAR requirements, and simulate the effect on a UAV wing using CFD.
- Extracted data from the simulated icing encounter and apply the aerodynamic effects of the ice accretion on a FEA model of an iced wing.
- Simulate the action of the novel technology on the ice-contaminated wing.

1.4.3. Discussion & Conclusion

This section of the report will summarise the results obtained and observations made, and use the information to:

- Determine the effectiveness of the novel ice protection system on UAV wing.
- Consider the value of the novel ice protection device to the effectiveness of UAV system mission.

CHAPTER 2: LITERATURE SURVEY

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2. LITERATURE SURVEY

The survey of literature aims to examine existing aircraft ice protection systems, alternative aircraft ice protection methods and technologies that maybe applicable to aircraft ice protection.

2.1. Existing Aircraft Ice Protection System

A number of aircraft ice protection systems exist in the market place. All have advantages and limitations in their application and effectiveness. The primary limitations are generally associated with power consumption and system weight. Source for the following descriptions is FAA Aircraft Icing Handbook DOT/FAA/CT-88/8-1 Vol. II.

Ice protection systems can be classified as either de-ice systems or anti-ice systems. The de-ice method allows ice to accumulate to a fixed cross-sectional thickness before being removed by breaking (pneumatic de-icing) or melting (thermal de-icing) the ice-surface bond. The anti-ice method prevents ice from bonding to the surface by maintaining the surface above freezing (thermal anti-icing) or depressing the ice freezing point (fluid anti-icing).

Either ice protection method is suited for UAV wing leading edges.

2.1.1. Pneumatic De-Icing.

Pneumatic boot surface de-icing systems remove ice accumulations mechanically by alternatively inflating and deflating tubes in a boot covering the surface to be de-iced. Inflation of the tubes under the accreted ice breaks the ice into particles and destroys the ice bond to the surface. Aerodynamic forces then remove the ice.

Normal practice is to allow ice to accrete to a thickness of around 12mm before the de-ice system is turned on. The thickness does depend on the type of boots, leading edge contour, aircraft speed, atmospheric conditions and drag characteristics of the surfaces that have accreted ice.

To function effectively, pneumatic boot system should inflate and deflate rapidly - typically 5 to 6 seconds to reach full pressure. In moderate icing conditions, a 60 second cycle is suggested. In light icing conditions, a 3 - 4 minute cycle is suggested to allow ice to accrete.

Pneumatic boot de-icing systems operate on clean turbine engine bleed from dry air pumps. The primary components of a pneumatic system are: the elastomeric boots, a regulated pressure source, an air distribution system, solenoid, check and relief valves, air filters, control switches and timer, electrical interfaces including fuses, circuit breakers etc.

The air pumps deliver pressure as a function of altitude. The regulated pressure source ensures the expansion of all the tubes in the system to the design limits and expansion times.

The vacuum source is required to secure a positive and rapid deflation. The application of a vacuum also assures the tubes remain deflated during non-icing flight conditions to minimise aerodynamic penalties.

Pneumatic boot de-icing system can be added to existing aircraft with minor modification and expense.

Advantages of this pneumatic system for a UAV are:

- System weight and power requirements are minimal.
- Low cost systems appropriate for smaller aircraft.
- Low servicing, maintenance, and replacement costs.

- Can be added to existing surfaces without alterations to aerofoil etc.

Limitations with this pneumatic system for a UAV are:

- Air pumps need to be close to the area requiring ice protection to minimise weight. On some aircraft, there may be little room or power for such equipment. Alternatives may be air drawn from the engine compressor, which for the Predator class of aircraft is some distance away from the wings.
- There is an aerodynamic drag penalty with the system, particularly on wings. This can be minimised by recessing the leading edge surfaces to off-set the boot thickness.
- A degree of skill is required by the aircraft controller for safe and effective pneumatic boot operation. Actuation when the ice is too thin (either activating the system too early or too rapidly) can result in the formation of ice bridges. Thus, attention is required to visually judge whether the cycle time continues to be correct as icing conditions change. This is particularly difficult for an autonomous and remotely controlled aircraft.
- Ice particles shed by pneumatic boots can be large enough to damage aft mounted engines or propellers. Turbofan and turbojet engines are the most vulnerable. Turboprop engines with particle separating inlet ducts are the least vulnerable.
- Pneumatic boot materials deteriorate with time, and are sensitive to contamination and mechanical damage (e.g. Scuffing). This demands for regular inspection.

FAA Aircraft Icing Handbook¹⁵ suggests for a FAR23/JAR23 class (e.g. small single engine aircraft equivalent to Predator B UAV), a wing and tail de-icing equipment requires ~250W of power and has a complete system mass of 12.7kg (of which 3.6kg is ancillaries). Commercial literature from BF Goodrich¹⁶ claims power as little as 100W for a FAR23 aircraft for the same equipment weight.

2.1.2. Hot Air Anti-Icing.

This method of ice protection prevents ice forming on the aerofoil surface by ducting hot air span wise on the internal surfaces of the surface to be protected. Due to the class of aircraft being investigated here, the method of ice protection is considered unsuitable for UAV largely due to location of the engine compressor with respect to the wings requiring protection.

Bleeding air from the engine compressor is not suited for an aircraft with neither the power nor the spatial arrangement to pipe hot duct air. In addition, the aircraft fuselage and wings are made from lightweight, polymer composites, which can degrade at the hot air temperatures (even when mixed with cooling air).

¹⁵ : FAA Aircraft Icing Handbook DOT/FAA/CT-88/8-1 Vol. II.

¹⁶: "Advanced PIIP for Aircraft": Martin C., Putt J.; Journal of Aircraft Vol.29, No.4 (Aug 92).

2.1.3. Electro Thermal Anti-Icing or De-Icing.

Source: FAA Aircraft Icing Handbook DOT/FAA/CT-88/8-1 Vol.II

Electro-thermal ice protection systems apply heat to the area to be protected i.e. the direct impingement region. The applied heat is used to evaporate or prevent the impinging water droplets from freezing (anti-ice protection system). Alternatively, the heat is used to melt and de-bond the accumulated ice (de-ice system). The equipment for electro-thermal systems include:

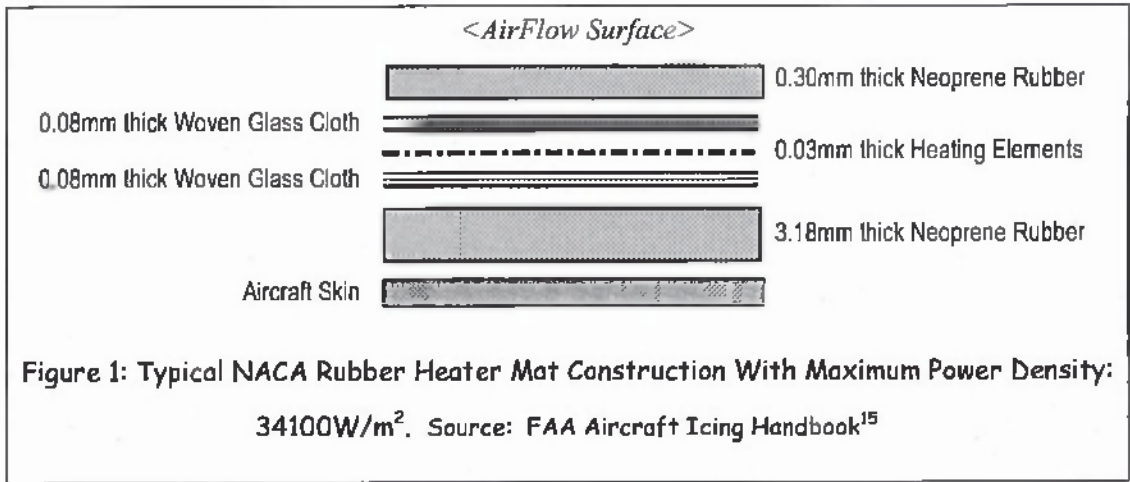
- Source of electrical energy (e.g. power generators or auxiliary power units).
- Wiring system to distribute the energy to the surfaces being heated.
- Resistance heaters for areas to be anti-ice or de-iced.
- Temperature sensors at the heated surface.
- Control unit (manual or automatic) to monitor icing conditions and heat supplied to the surfaces.
- Display panel to inform crew of the system operation.

Electro-thermal systems use electrical heater elements (foil, film, resistance wire, mesh etc.) embedded in fibreglass composite, rubber laminate, or metallic fabrication to heat the surface. These elements are insulated on either side with a suitable dielectric, with power leads discreetly welded or soldered through the non-airflow side. There are a number of general configurations found in practice. The construction details will vary slightly from manufacturer and installation position on the aircraft.

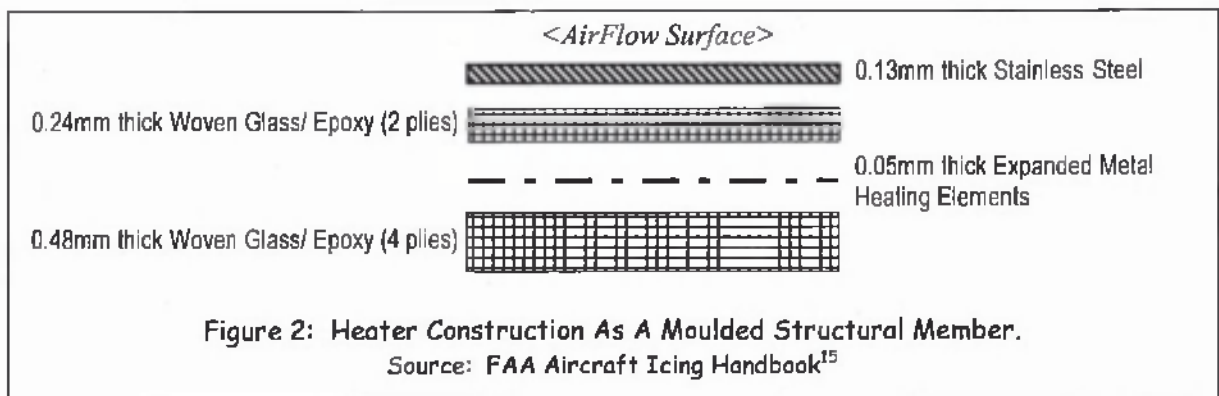
Heater elements can be laid span wise (element from tip to root) or chord wise (around the leading edge) with power density variations in span or chord wise directions. Chord wise heaters are generally activated in sections from tip to root, while span wise heaters are activated around the chord starting at the stagnation region. Combinations of these two arrangements can also be used.

The extent of heater mat coverage depends on the shape of the aerofoil. With most ice accretion-taking place at the leading edge (around the stagnation zone) along the span, heater mat coverage is typically between 40% and 80% of the wingspan, and between 15% and 25% of the wing chord.

Figure 1 shows a typical NACA rubber based heater mat construction bonded to the aerofoil surface. The heater mat can be retro-fitted or supplied with the aircraft. Due to the materials employed, the maximum power output is generally in the region of 34100W/m².



For UAV systems, the heater mat can be part of the structural member of the aircraft composite wing structure. As such, it will carry loads associated with the structure. As shown in Figure 2, the construction is typically part of the wing and engine inlet fabrication stage. Heater output is generally around 24800 - 31000W/m². Due to the thermal insulation nature of the composite construction, there is a time lag between system activation and wing surface achieving the required temperature to anti-ice or de-ice the surface. This can be addressed with more electrical power for instance.



Further variation on these construction configurations have the heater pads installed or bonded on the non air-flow side of the aerofoil. All configurations apply equally to anti-ice or de-ice systems. They all have the heating elements encased in an electrically non-conductive material to insulate the elements. Encasing the elements also reduces the damage due to impact, environmental attack etc. Separately temperature sensors can be placed behind the encased heater elements to monitor surface temperatures.

With reference to Figure 1, this heater pad configuration can be installed to existing aircraft surfaces by attaching flexible wrap-around pads to the structure with adhesive or mechanical fixings i.e. rivets. When installed on the airflow side, these heater pads have small effect on the aircraft aerodynamic performance.

The integrally moulded solution displayed in Figure 2 provides no effect to the aerodynamic performance. As part of the fibre reinforced composite wing construction, this method can be considered as a low system weight solution. However, most composite or rubber pad solutions for high-speed applications require a metal over-shoe to reduce rain erosion and hail damage. This means the design has to cater for the additional weight and heat loss.

Advantages of an electro-thermal de-icing system for a UAV system are:

- Electro-thermal systems have the ability to apply heat effectively to where it is required. This is particularly advantageous in regions of the aircraft away from hot air sources.
- De-ice systems consume less power than anti-ice or hot air based systems.
- Noise, fatigue and development risks are low.
- Low servicing, maintenance, and replacement costs.
- The system can be added to existing specific types of surfaces. However, alterations to airframe will be required.

Limitations of the system for UAV applications are:

- Power consumption can be very high, as there is significant heat loss to the structure and environment.
- Additional power generating capacity means additional weight and fuel consumption.
- With de-icing systems, a parasitic drag associated with ice accumulation must be accounted for.

- Electromagnetic interference (EMI) of other aircraft electronic components & systems have to be considered, particularly if there is frequent system switching.

This last limitation is considered a significant problem for UAV systems with highly sensitive surveillance equipment. This has been addressed by some manufacturer's with the development of specialised heating mats¹⁷.

More efficient thermal energy transfer using nickel coated carbon fibre heaters (Perkins¹⁸), and more sophisticated de-icing control algorithms (Giamati¹⁹) will improve the attractiveness of electro-thermal de-icing to UAV system. Perkins paper claims their heater pad system would require less power. Giamati proposed method of a de-icing control method that could manage the power demand on the aircraft system.

FAA Aircraft Icing Handbook¹⁵ suggests for FAR23/JAR23 class (small single engine aircraft equivalent to Predator B UAV), a wing and tail de-icing equipment requires ~1800W of power and has a complete system mass of 9.9kg (of which 5.4kg is ancillaries). Anti-icing electro-thermal systems for wings & tail would exceed power available for the Predator and Phoenix class of UAV.

¹⁷ : "Electro-Thermal Ice Protection Of A Surveillance Aircraft Radome", Smith. J; Cox & Co. AIAA-96/0388.

¹⁸ : "Thin, Lightweight, Electro-thermal Ice Protection - New Technology Supplementing Existing Ice Protection", Perkins P.J; Hastings O.; Rieke, W., NASA-Lewis AIAA-96/0391 paper.

¹⁹ : "Electro-thermal De-Icing System", Giamati M.; BF Goodrich; EU Patent 0749894

2.1.4. Electro-Expulsive De-Icing.

A number of inventions uses electromagnetic actuators mounted within the aerofoil of an aircraft to effect de-icing.

One form²⁰ has each actuator include conductive strips fabricated on a flexible dielectric sheet. The flexible dielectric sheet is then rolled into an elongated tube and flattened to the actuator shape. The axis of the winding coils is coincident with the longitudinal axis of the elongated tube. The elongated shape of the tube constrains the coil windings from displacing along the longitudinal axis.

Upon application of an electric current to each actuator in turn, an electromagnetic force is induced which causes the flattened tube to expand to an oval shape. This expansion causes an impulsive force to be applied to the semi-rigid aerofoil surface. The aerofoil surface distends and then recoils. Any ice accreted on the aerofoil surface has the surface bond broken by the distension and recoil action. Ice is then expelled or removed by the aerodynamic forces.

The main advantage claimed is that the actuators have low voltage, and current requirements due to the high number of coil windings used, and so is more energy efficient than electro-thermal de-icing systems.

²⁰ : "Electro-expulsive De-Icing System", Ingram R.; Codner G., Geradi J., Innovative Dynamics, WO Patent 96/37406.

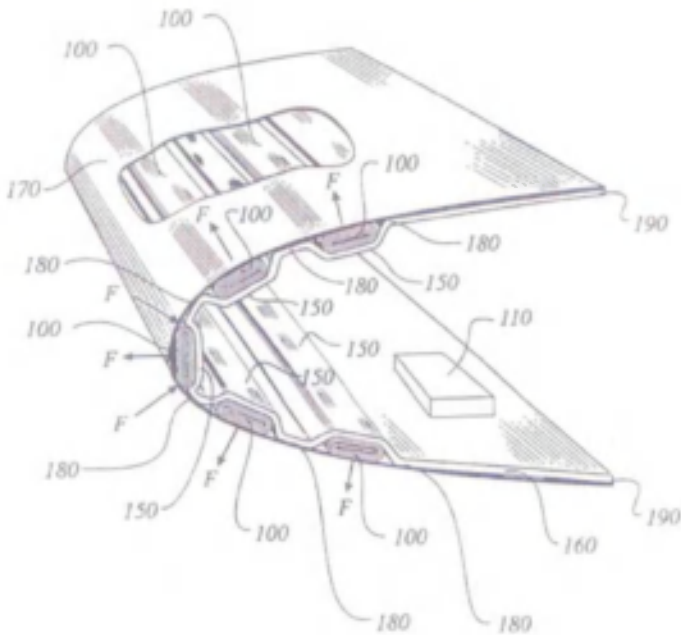


Figure 3: Electromagnetic Expulsion De-Icing System. Source: WO Patent 96/37406

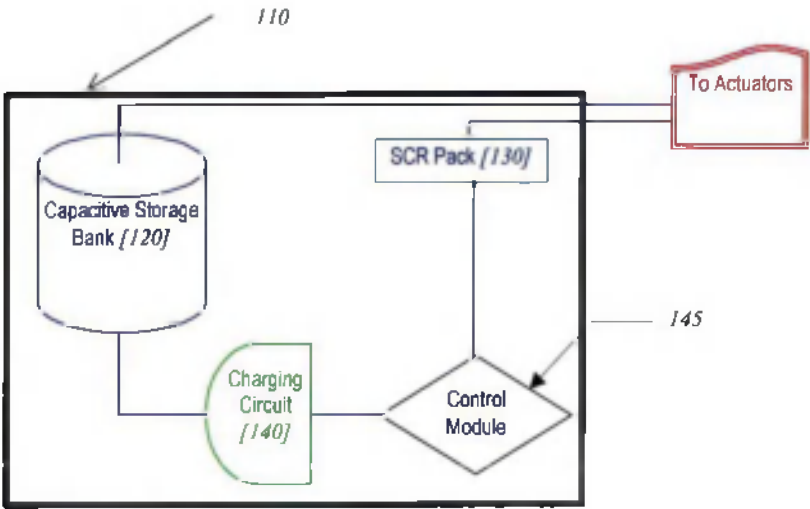


Figure 4: Driver Electronics. Source: WO Patent 96/37406

Figure 3 and Figure 4 show the de-icing system as mounted within the aerofoil of the aircraft. The system also consists of actuators [100] and driver electronics [110]. The driver electronics box houses the capacitive storage bank [120], a silicon control rectifier (SCR) [130], a charging circuit [140], and a control module [145]. Some or all of the driver electronics can be mounted or integrated within the aerofoil or an aerofoil channel section [150]. Alternatively, all the driver electronics can be mounted within the aircraft cabin.

The actuator is made up of a conductive strip fabricated on a flexible dielectric sheet. A detailed image of the actuator construction is shown in Figure 6 as item [300]. The dielectric sheet [320] with the conductive strip [310] is wound into an elongated coil and flattened to the actuator shape. The length of tube is sufficient with respect to the generated electromagnetic force to provide a lateral restraint, which prevents the coil windings from displacing. The elongated tube longitudinal axis is in line with the coil axis windings.

The tube is selectively bonded to the corrugated channels of a housing [160]. The housing can be integrated within the leading edge structure, or be shaped to conform to the aerofoil structure. If fitted to the aerofoil, the housing can be bonded to the aerofoil surface with an adhesive or, preferably mechanically fastened at support points [180] and fixed at points [190].

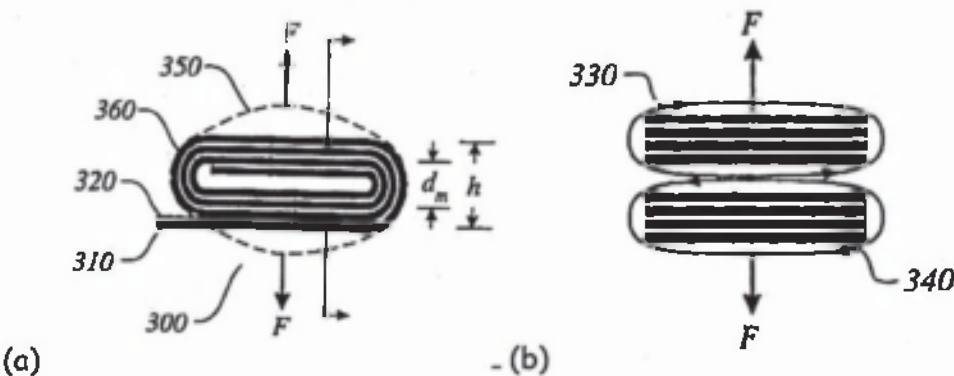


Figure 5: Actuator Cross-section In Expulsion Mode (a) & Coil Detail. Source: WO Patent 96/37406

For very sharp leading edges, fracture proof plastic shims can be bonded to conform to the profile. The actuator and housing are covered with a protective semi-rigid sheet [170] such as stainless steel, titanium, aluminium, fibre composite etc. The inventor prefers this semi-rigid sheet to be electrically conductive.

In operation, the semi-rigid sheet can flex and lift off from support points. The number and location of support points is dependent on aerodynamic load characteristics. However, the upper portion of the aerofoil is in a low-pressure state, which can result in "auto-inflation". To compensate for this, a void is fabricated into the construction. This allows the semi-rigid sheet to deflect and vibrate to effect de-icing in the chord wise directions

The operation of the actuator [300] is shown in Figure 5. The de-icing operation begins with the charging circuit charging the capacitive storage banks. When icing is detected on the aerofoil, the control module triggers the SCR, which in turn causes an instantaneous current to be discharged through the actuator coils. The control module activates each actuator in succession. The interaction between the current in the coil windings results in an impulsive electromagnetic force that causes the coil to rapidly expand from its flattened shape to a more oval shape.

This expansion exerts an impulsive force on the semi-rigid sheet surface, which in turn distends and recoils. The recoil action expels any ice accretions. Following actuation, the coil returns to its original shape due to the coil elasticity (winding sides return to the original condition) and the inward tension exerted by the semi-rigid sheet.

Figure 5(b) shows the current in the top windings flow in one direction and the current in the bottom winding in the opposite direction. The resulting magnetic fields [330] and [340] within each bundle produce a repulsive force that distends each coil half along a direction perpendicular to the current flow forming the oval shape [350].

For each coil assuming equal thickness of upper & lower windings, the actuation force F (N/m) is:

$$F = 4 \times 10^{-7} \cdot \frac{L N^2 I^2}{b} \left[\tan^{-1} \left(\frac{b}{d_m} \right) - \left(\frac{d_m}{2b} \ln \left(\frac{d_m^2 + b^2}{d_m^2} \right) \right) \right]$$

where L = coil length (m)

b = conductive strip width (m)

d_m = mean separation between wiring bundles (m)

I = coil current (Amps)

N = number of turns in the coil

This approximation is only valid for the mean separation d_m is much greater than the thickness of the conductive strip.

Natural flight icing trials of an electro-expulsive de-icer system was described in a paper by Weisend²¹. The de-icer system is based on a NASA patent²² had been installed on a De Havilland DHC-6 turbo prop aircraft. The paper also described the icing tanker trials of McDonnell Douglas F/A-18 jet fighter with an Electro-Expulsive De-Icer fitted to the engine inlet.

The development of the electro-expulsive de-icer has allowed the mechanical removal of very thin layers of ice. Table 1 shows comparative power requirements for different types of ice protection systems for the McDonnell Douglas F/A 18 Fighter Aircraft.

²¹ : "Electro-Expulsive De-Icer Flight Tests", Weisend N.A, BF Goodrich Aerospace, 15th European Rotorcraft Forum (Sept 99).

²² : "Electro-Expulsive Separation System", Haslim L.A., US Patent 4,690,353 (Sep 89).

Table 1: McDonnell Douglas F/A 18 Comparative Power Requirements	
Ice Protection System	Power Required (W)
Thermally evaporative anti-ice	6100
Thermal running wet anti-ice	39000
Thermal de-ice	4800
Mechanical De-Ice Electro-Expulsive	175

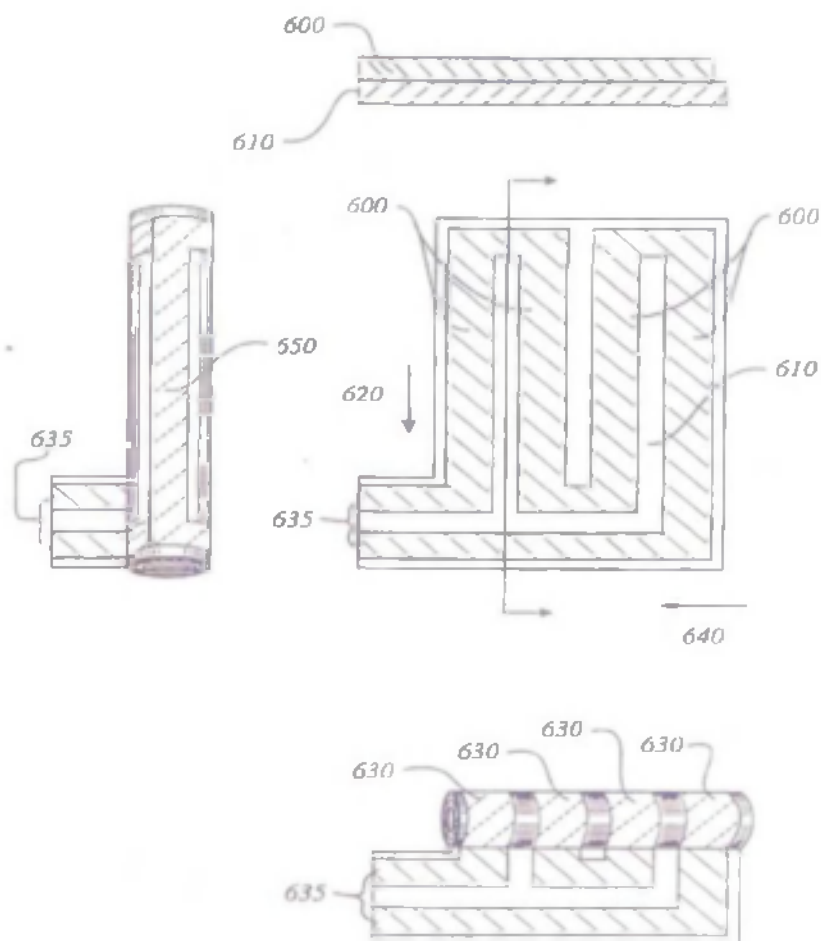


Figure 6: Actuator Construction In Detail. Source: WO Patent
96/37406

The electro-expulsive de-icer uses the principle of electro-magnetic repulsion triggered, when high amperage current is pulsed in two closely spaced parallel conductors. If the electrical currents are in opposing directions, an electro-repulsive force is generated which acts to move the two conductors apart as shown in Figure 7. The force of multiple sets of parallel conductors in two layers can be directed to an ice-covered surface to break the ice.

The ice protection device is applied over the airfoil surface and is constructed from the following. A base material layer consisting of a set of closely spaced flat, parallel electrical conductors is bonded to the airfoil surface. A second layer containing a set of parallel electrical conductors lie opposite to and are electrically isolated from the base layer conductors. The isolation layer also allows the conductor layers to move apart when electrically energised.

The action of the high current pulse through the conductive layers causes the outer surface conductors to deflect rapidly which causes the expulsion of any outer surface ice into small pieces.

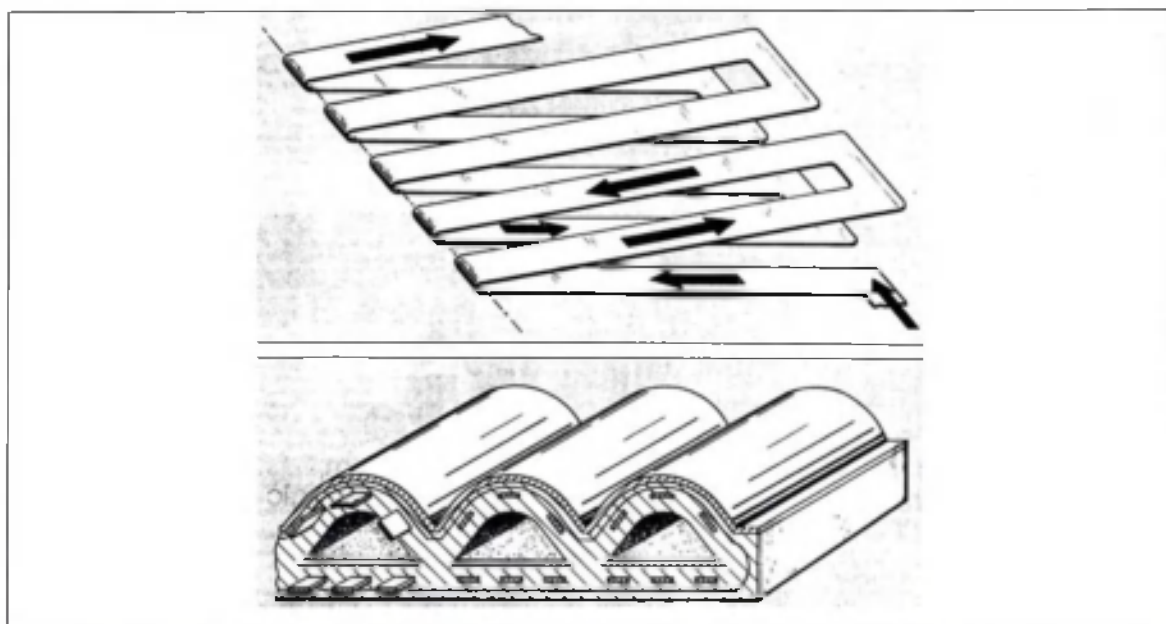
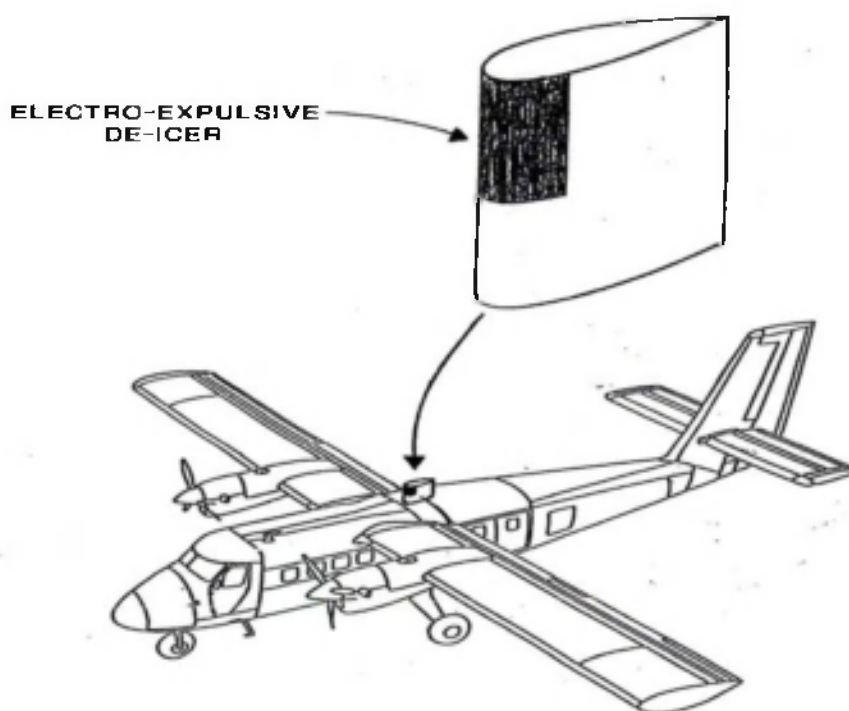


Figure 7: Electrical Pulse Causes Separation Of Conductors. Source: US Patent 4,690,353

The power source for the current pulse is the capacitor bank, which receives its charge from a low voltage converter. To minimise the energy required being stored and therefore the weight, the de-icer is divided into de-icing segments. A controller directs the electrical power from the energy storage and monitors the pulse discharges through the de-icing segments using a distributor.

For the "proof of concept" flight-tests, all system operating components were combined into a single controller.

A test airfoil was installed on a NASA-Lewis DHC-6 aircraft as shown in Figure 8. The test panel was a non-standard, symmetrical airfoil (305mm chord, 22% maximum thickness, 13mm leading edge) which could be extended or retracted vertically in-flight thorough the DHC-6 fuselage ceiling.



DEHAVILLAND DHC - 6

Figure 8: DeHavilland DHC-6 Flight Icing Trials. Source: Weisend²¹

The leading edge was of elastomer/ fabric construction and contained 2 off 203mm long electro-expulsive span wise de-icing segments, that butted at the leading edge centreline. The de-icer was bonded to the airfoil surface and had power leads extended through the airfoil to an operating container located in the cabin.

Four natural icing flights at test temperatures between -11°C to -0.5°C , the cloud liquid water content 0.1 to $0.75\text{g}/\text{cm}^3$ and flight speed ranges from 198 to 237km/h. The thinnest ice consistently removed was 0.5mm and the thickest ice removed was 10.2mm. This range of ice thickness removed is larger than that observed with pneumatic de-ice systems.

No published data is available on the system mass. The author suspects the total system mass will be little different from that calculated for Electro-Impulsive De-Icing system discussed in section 2.1.5 below. The reasoning behind this statement is the dominant constituent in the EEDI system mass are the large banks of energy storage capacitors and power harnesses, rather than the expulsion blankets bonded to the aerofoil. Thus for a FAR23/JAR23 class (small single engine aircraft equivalent to Predator B UAV), an EEDI wing and tail de-icing equipment requires ~500W of power and has a complete system mass of 28kg (of which 5.4kg is ancillaries).

2.1.5. Electro-Impulsive De-Icing.

The physical form of the Electro-Impulsive De-Icing²³ (EIDI) system as installed on an aluminium leading edge is shown in Figure 9. Flat wound coils of copper ribbon wire are placed just inside the leading edge of the wing's skin, with a small gap separating the skin and coil.

When installed on a Kevlar composite leading edge, the EIDI configuration is in the form of a "bumper coil" design. Here a spring loaded foam "bumper" is placed between the coil and skin. As with the metal skin leading edge, a small gap is maintained between the bumper and coil. The coil (s) are placed at a given span wise station that is dependent on the size and shape of the leading edge. The coils can be supported either by the front spar (as shown here) or on the ribs or mounted to the skin.

The coils are connected by low resistance, low inductance cables to a high voltage capacitance bank. A remote signal to a silicon-controlled rectifier ("thyristor") enables energy to be discharged through the coil. Discharge of the capacitor through the coils creates a rapidly forming and collapsing electromagnetic field, which induces eddy currents in the aluminium skin or bumper skin.

²³ : "Structural Dynamics of Electro-Impulse De-Icing on the Lear Fan Leading Edge "; Paramasivam T. & Zumwalt G., Wichita State Univ. SAE paper 850917 (Apr 85).

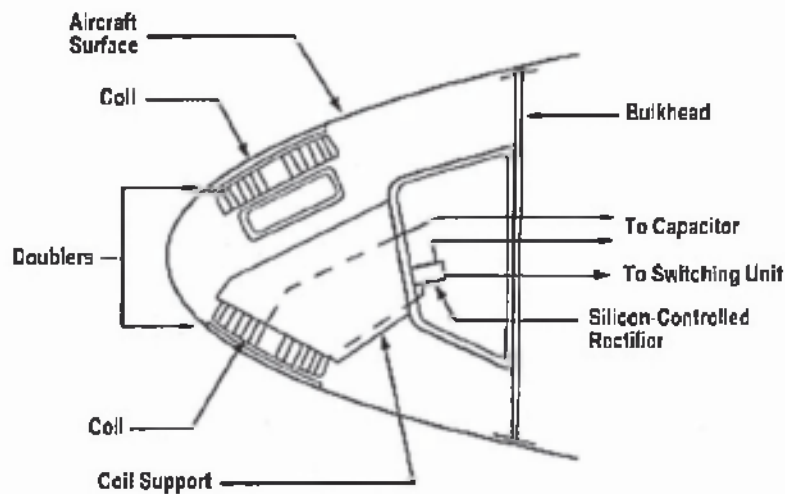


Figure 9: Impulse Coil Installation On Aluminium Wing Leading Edge. Source: Paramasivam T. & Zumwalt G²³

The fields resulting from the current flow in the coils and skin creates a repulsive force of several hundred pounds magnitude for only a fraction of a millisecond. A small amplitude (high acceleration) of the skin acts to shatter, de-bond and expel the ice. 2 or 3 such impulses performed sequentially separated by the time required to recharge the capacitors, allows ice to accumulate to optimum thickness for removal.

Figure 9 shows unalloyed aluminium "doubblers" bonded to the skin opposite the coil. These doublers have a slightly larger diameter than the coils and a thickness less than the minimum required for adequate eddy current conductance. Composite and non-metallic leading edges require a similar procedure.

The basic EIDI circuit is illustrated in Figure 10. The current waveform and resultant skin displacement is shown in Figure 11.

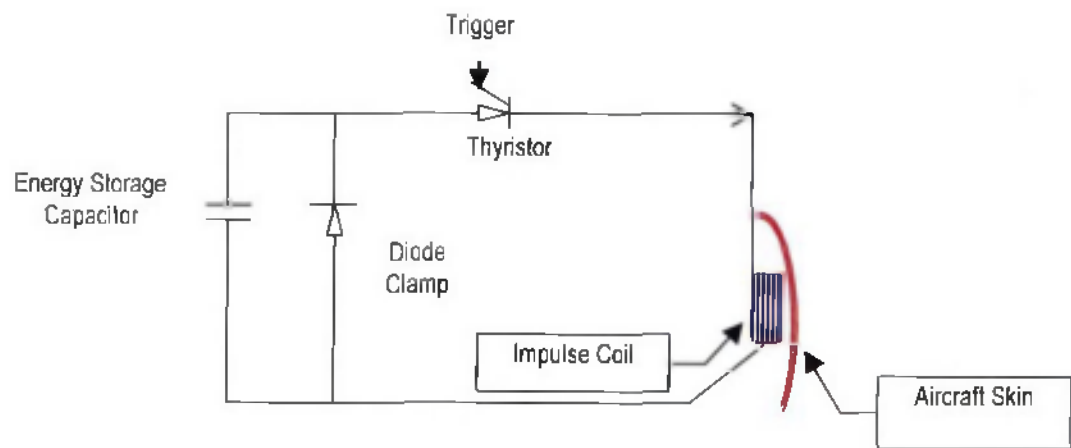


Figure 10: Basic EIDI Circuit. Source: Paramasivam & Zumwalt²³

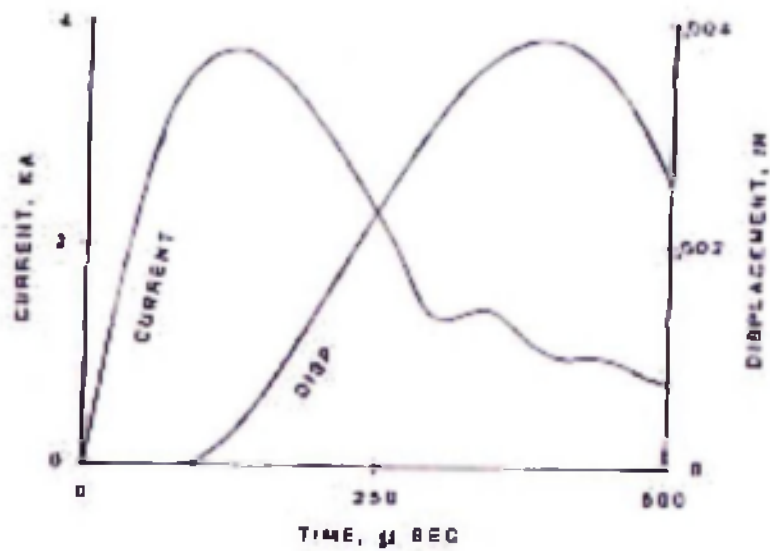


Figure 11: Typical Coil Current & Skin Displacement. Source: Paramasivam & Zumwalt²³

The normal force-time response on a metallic target (2024-T3 aluminium) due to an impulse from the coil is displayed in Figure 12. The experimental determination of this impulse force was performed with the initial capacitor energy at 20% of that required for de-icing. If extrapolated to the forces used in de-icing, the 2135N force is applied (equivalent to 0.36N-sec impulse for 800Volts and 600μF capacitance).

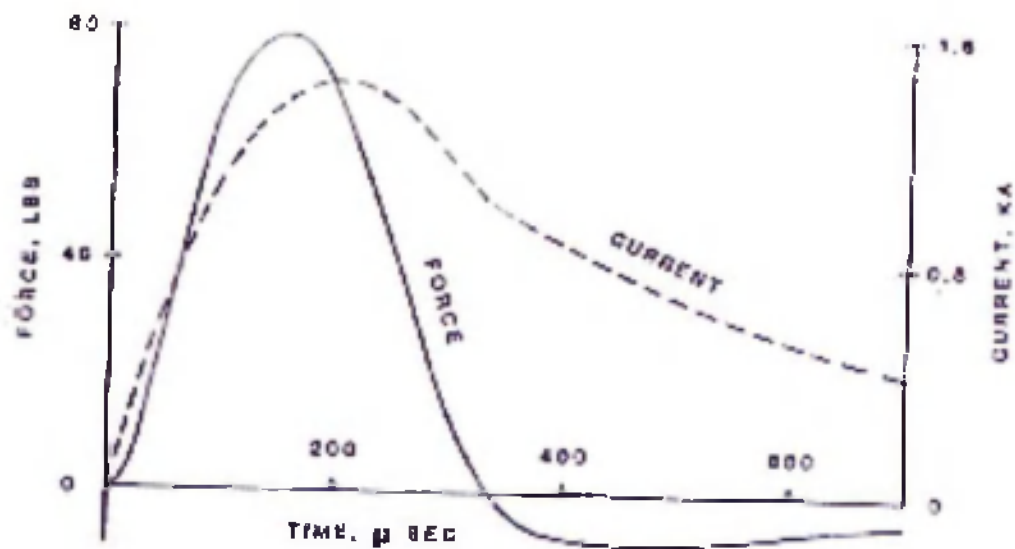


Figure 12: Normal Force Against Time. Source: Paramasivam & Zumwalt ²³

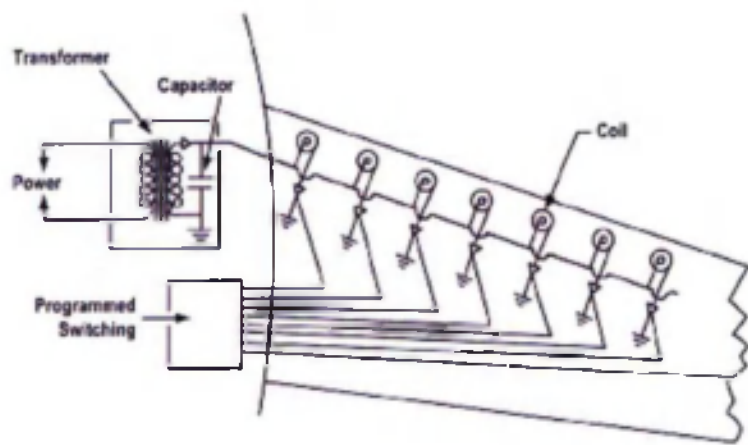


Figure 13: EIDI System In A Wing. Source: Paramasivam & Zumwalt ²³

Figure 13 shows a wing with the coil placed at a 0.4m span wise separation, all supplied with a single power unit. The paper claims the power requirements are equivalent to that required for the same size aircraft landing lights.

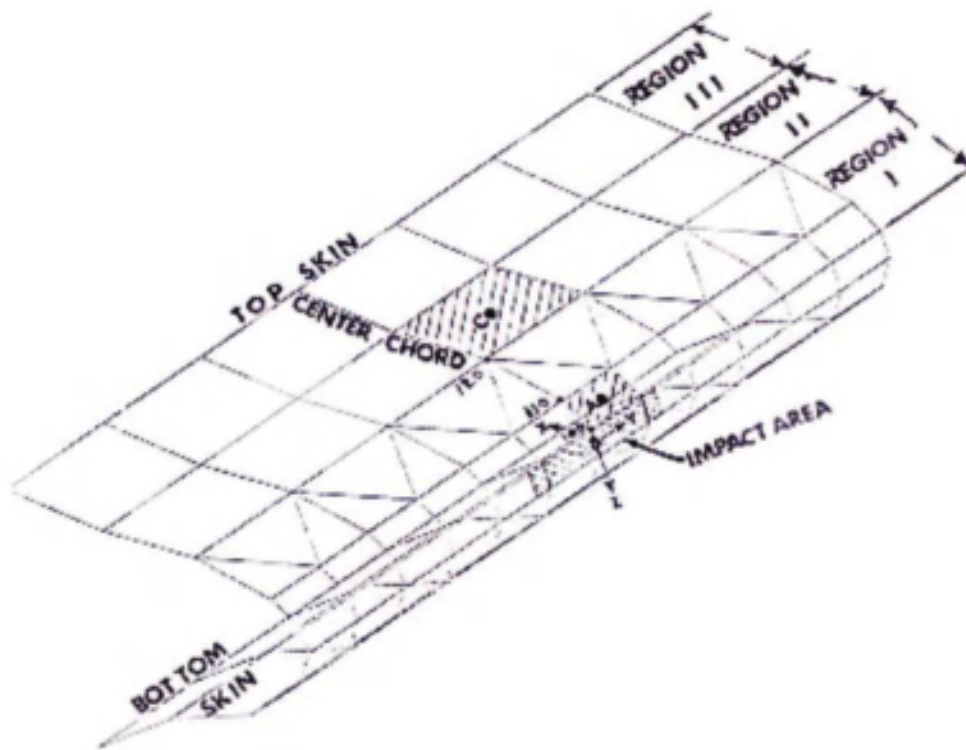
Analytical Study

Figure 14: MSC/NASTRAN FE Model Of LEAR FAN Kevlar Leading Edge. Source:

Paramasivam & Zumwalt²³

The transient response of the Lear fan Kevlar leading edge due to an EIDI system was investigated using MSC/NASTRAN. The 0.30m leading edge FE model with constant cross-section was developed. This corresponds to a 1.0m actual test piece. The leading edge model was divided into 3 regions with differing skin thickness, orientation, and fibre lay-ups.

The FE mesh model is illustrated in Figure 14. Region I representing the surface profile from the leading edge apex to 50mm aft has 10 layers of Kevlar fabric orientated at $0^\circ/\pm 45^\circ/90^\circ$ at a total thickness of 2.6mm.

Region II extends 50mm aft of the Region I to the trim, and is made up of 6 layers of Kevlar fabric orientated $0^\circ/\pm 45^\circ/90^\circ$ but at a total thickness of 1.6mm.

Region III starts at the aft trim and ends at the wing forward spar. Region III laminate is made up of 10 layers of Kevlar fabric orientated at $0^\circ/\pm 45^\circ/90^\circ$ at a total thickness of 2.6mm.

Material properties were defined. The FE model was constrained in all 3 displacement degrees of freedom at all nodes along the forward spar, and constrained in in-plane displacements along the in-board and outboard edges. Perpendicular rotations of the top and bottom skin surfaces were also constrained.

Points A and C are located on the top surface. Points B and D are on the bottom surface and are directly beneath A and C respectively. Point A and B are 12.7mm off the chord centre line i.e. bumper nose contact area. Points C and D are located in Region II and are about 25mm off the chord centre line of the bumper-nose contact area.

A forcing function (2135N in 300 μ s) as defined in Figure 12 was applied near the mid-span stagnation point at the leading edge, and the transient responses in terms of grid displacements, velocity, acceleration, and panel stresses obtained. For example, Figure 19 and Figure 15 show the panel stresses and strains in top skin at point C.

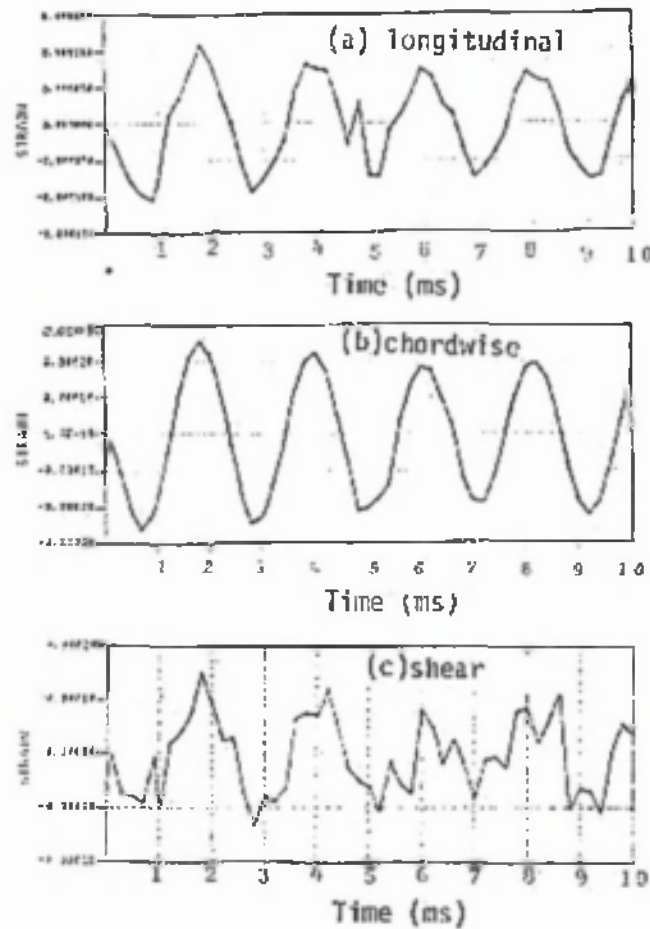


Figure 15: Panel Strains At Point C. Source: Paramasivam & Zumwalt ²³

Laminate ply failure/ strength analysis (Tsai-Hill failure theory) was used to plot the failure envelope for various ply configurations and different regions. A failure envelope example can be seen in Figure 18, where the maximum induced stresses for the Region I is indicated by the large dark dot is much smaller than the skin stress allowable envelope. This suggests the impulse loads are much smaller than that required to generate first ply failure.

Lab Studies

The theoretical transient responses from above were compared with those developed experimentally from an actual 1.0m leading edge test specimen. Here the leading edge top and bottom faces were strain gauged as per the FE model was prepared and subject to an impulse load as shown in Figure 17 and Figure 16.



Figure 17: Lear fan Leading Edge & EIDI Bumper Coil. Source: Paramasivam & Zumwalt ²³



Figure 16: Side View of Leading Edge With EIDI Bumper Coil Installation.

Source: Paramasivam & Zumwalt ²³

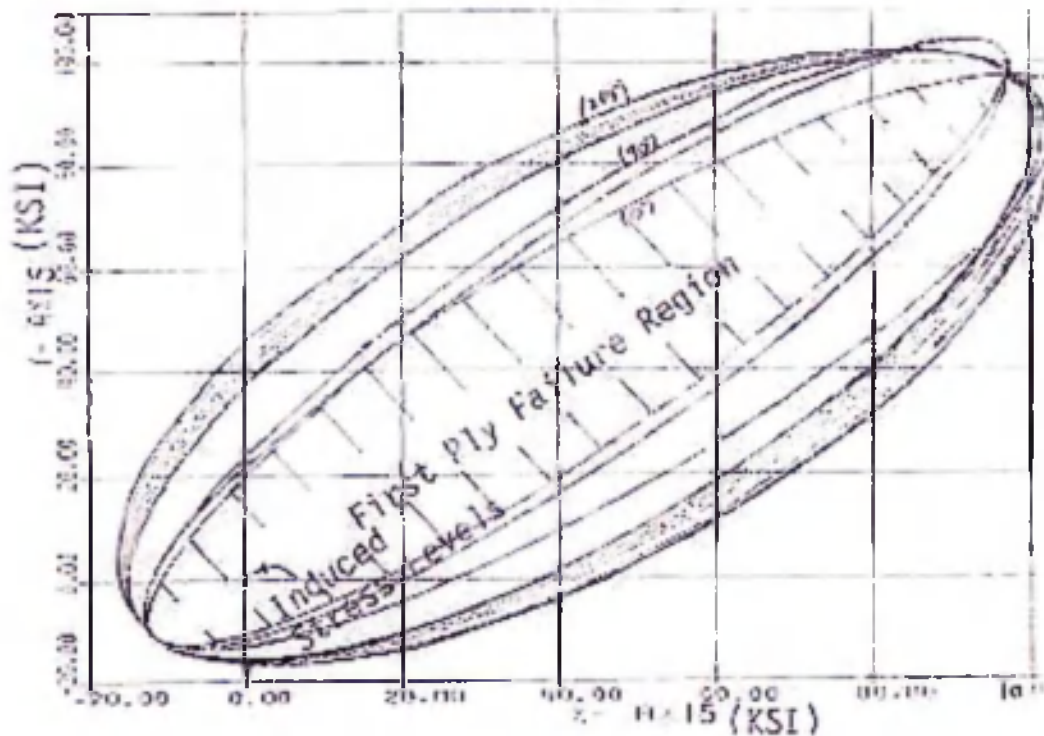


Figure 18: Strength Envelope For Region I. Source: Paramasivam & Zumwalt ²³

The aft skins top and bottom of the Kevlar leading edge were screwed into a rigid channel section running span wise and simulating an actual wing spar. Two bumper coils were mounted on the channel, and were located at quarter span from either end. The bumper and impulse coils were just behind the leading edge tip. The complete test piece was rigidly mounted.

The bumper coils were connected to the de-icing circuit and strain gauge circuit completed. Testing of the leading edge involved suddenly discharging the high voltage through the coils either separately or with the 2 coils in series. The hammer like blows on the leading edges were measured for various energy levels of the coil impacts.

Good agreement was found in the frequency correlation between the NASTRAN FE model and the test specimen as shown in TABLE 2. Although predicted and measured strains were found to agree with the longitudinal strain example at Point C in Figure 20.

TABLE 2: COMPARISON OF PREDICTED & MEASURED FREQUENCIES		
MODE NO	NASTRAN (Hz)	TEST SPECIMEN (Hz)
1	307	299
2	391	345
3	468	446
4	672	760
5	912	956
6	1102	1175
7	1383	1325

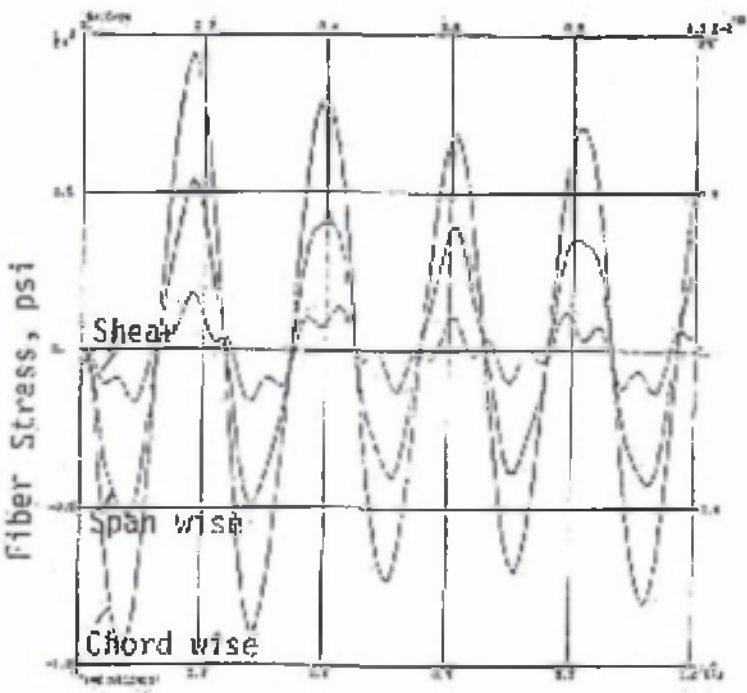


Figure 19: Panel Stresses At Top Skin Point C. Source: Paramasivam & Zumwalt²³

However, other measurements showed some considerable variation from predicted outcomes. In particular, surface acceleration and displacement values were found to deviate from prediction by almost a factor of 10 as shown in the example in Figure 21.

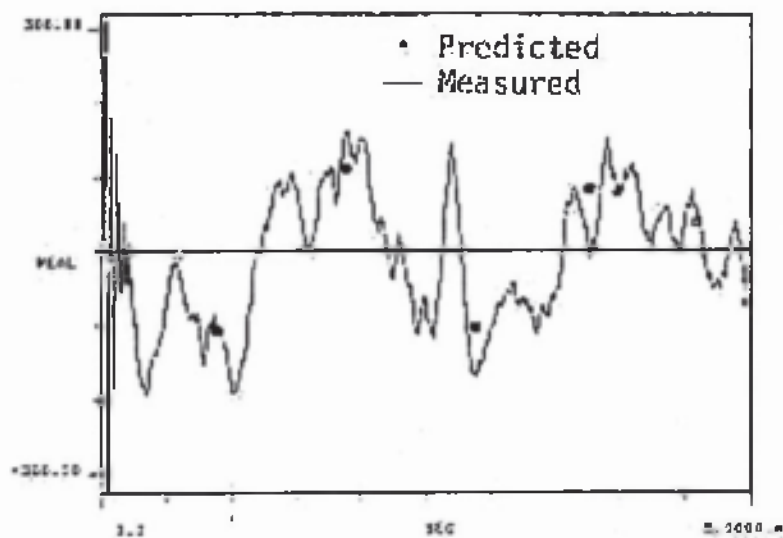


Figure 20: Predicted & Measured Longitudinal Strains At Point C

Predicted and Measured Displacement At Point A. A number of reasons for the discrepancy are obvious given the complexity of the leading edge geometry. It was found that the NASTRAN model was stiffer than the test model particularly around the nose at point A. The lab test showed highly damped responses throughout the leading edge section and not only at the bumper nose area. Thus, the NASTRAN model required further refinement.

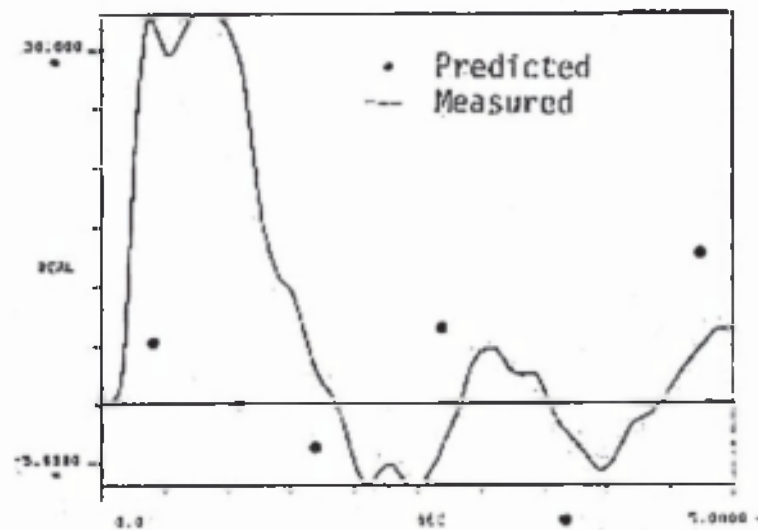


Figure 21: Predicted & Measured Displacement At Point A.

Source: Paramasivam & Zumwalt ²³

The test model used a different target material (1175 aluminium) from that of the FE model (2024-T3 aluminium). The forcing function was derived from the latter target material, and studies have indicated that different target material alloys will generate dissimilar forcing functions.

Hence, further work to improve the prediction model was necessary.

Icing Tunnel Testing

The 1.0m Lear Fan Kevlar leading edge was alternatively painted black and bare in quarter spans and extending up to the middle of the bumpers. Coils were located at the bare and painted junction surfaces. The purpose of the paint was to study the effects of ice property adhesion and its subsequent removal.

The leading edge was mounted vertically in the NASA Lewis Icing Research Tunnel and electro-impulsion tests were conducted at various energy levels in different icing conditions and angles of attack.

Figure 22 shows the Kevlar leading edge at a 4° angle of attack, -4°C, 14µm droplet size, LWC 0.8g/m³ at 10 minutes. Figure 23 displays the partially cleaned leading edge after the 2 coils were impulsed simultaneously once. Figure 24 shows the completely cleaned edge after the 3 impulses, with each impulse being 3 seconds apart.

Table 3: Icing Test Data On Lear Fan Leading Edge							
Run	A	Volts	Capacitance (µF)	Coil Firing Sequence	T _p (µS)	Total Energy (J/ft)	Results
0	4°		Baseline Ice Accretion To Be Removed.				See Figure 22.
1	0°	1000	400	1,1,1,2,2,2	125	424	Fair, some ice on upper surface
2	2°	1100	400	1,1,1,2,2,2	125	512	Good, light residue leading edge
3	4°	1200	400	1,1,1,2,2,2	125	610	Not quite clean
4	4°	1100	400	1,1,1,2,2,2	125	512	As per run 2
5	4°	1550	200	1,1,1,2,2,2	90	509	Not as good as run 2
6	4°	900	600	1,1,1,2,2,2	165	514	Poorer than run 2
7	4°	1200	400	$\frac{1}{2}, \frac{1}{2}, \frac{1}{2}$	150	305	Poor- see Figure 23.
8	4°	1400	400	$\frac{1}{2}, \frac{1}{2}, \frac{1}{2}$	170	415	Good ice removal - see Figure 24.

Table 3: Icing Test Data On Lear Fan Leading Edge						
Run	A	Volts	Capacitance (μ F)	Coil Firing Sequence	T_p (μ S)	Total Energy (J/ft)
Results						

A: angle of attack.

T_p : time to reach peak current value.

Coil sequence $\frac{1}{2}$ indicates that 2 coils were wired in series & fired simultaneously.

All tests at air temperature -4°C ; LWC: 0.83g/m^3 ; $14\mu\text{m}$ droplet size; air speed 71.5m/s for 10 minutes.

Even though ice was not removed effectively under some combinations of angle of attack, energy levels, and impulse sequence, there was no distinction between painted and bare Kevlar surfaces in terms of removing or retaining accreted ice.

As can be seen from the test data in Table 3, only some combinations of energy levels with either separate or simultaneous impulses along particular angles of attack were favourable for removing accreted ice. The de-icing system condition that managed to remove ice in all temperature conditions and angles of attack required 1100V and $400\mu\text{F}$ capacitance of energy.

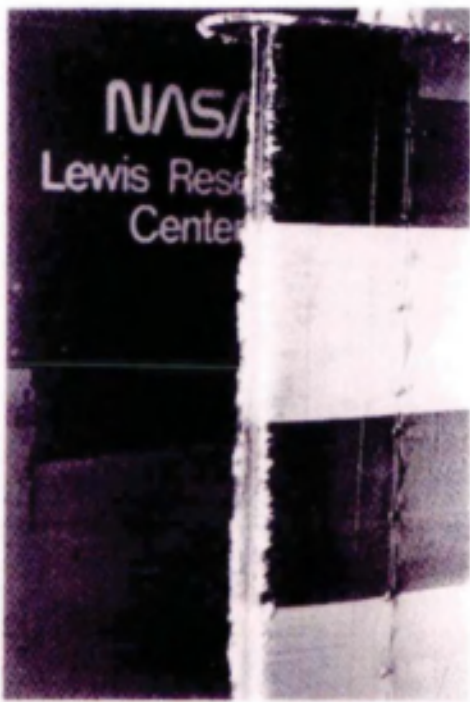


Figure 22: Ice Accretion On Leading Edge.
Source: Paramasivam & Zumwalt ²³



Figure 23: Partially Cleaned Leading Edge.

Source: Paramasivam & Zumwalt²³

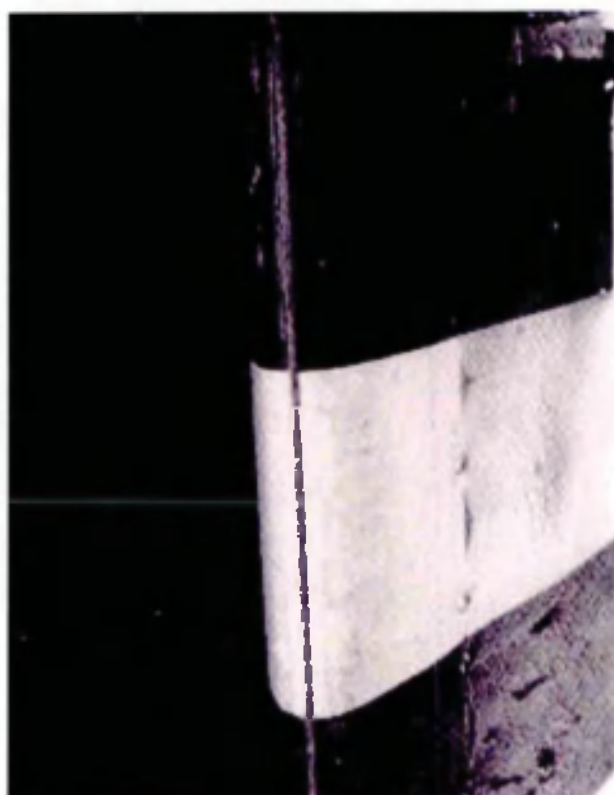


Figure 24: Cleaned Leading Edge. Source: Paramasivam & Zumwalt²³

Advantages of the EIDI system are:

- Low power required ~500W for 6 actuator, propeller passenger carrying aircraft. EIDI system power consumption is less than that required for hot air or electro thermal anti-ice systems.
- Reliable de-icing. Ices in all forms are expelled, with only light residual ice remaining after the impulses. Ice thicknesses from 8mm to 26mm have been consistently shed.
- Non-intrusive in the air stream, hence no aerodynamic penalty.
- Weight comparable to other de-icing systems.
- Low maintenance, theoretically, since there are no moving parts. However coils, capacitors, and diodes can fail.
- No run-back refreezing occurs.
- UAV operator skill and judgment required to operate the system are minimal in that no threshold ice-thickness is required for turn-on.
- System can include self-test circuitry.

• Limitations of the EIDI system are:

- It has had limited use, possibly due to what are perceived as "complex" design requirements. The equipment also has to be incorporated at the wing manufacturing stage.
- It is not an anti-icing system, so some ice will be present over most of the aircraft leading edges during flight in icing.
- Outside the aircraft, the discharges may be quite loud, resembling a light gunshot. This maybe a disadvantage for a UAV systems approaching target zones at very low altitude.

FAA Aircraft Icing Handbook¹⁵ suggests for a FAR23 class (small single engine aircraft equivalent to Predator B UAV), a wing and tail de-icing equipment requires ~500W of power and has a complete system mass of 28kg (of which 5.4kg is ancillaries).

2.1.6. Fluid Ice Protection.

Source: FAA Aircraft Icing Handbook DOT/FAA/CT-88/8-1 Vol.II

Fluid ice protection systems operate on the principle that the surface to be protected is coated with a fluid (typically a glycol based fluid) that acts as a freezing point depressant. When super cooled water droplets impinge on a surface, they combine with the glycol fluid to form a mixture with a freezing temperature below the temperature of the ambient air. The mixture then flows aft under the influence of the boundary layer, and is either evaporated or shed from the trailing edge of the surface.

The glycol fluid is distributed onto the surface leading edge to be protected by pumping it under pressure through porous material and channelling along the leading-edge grooves - see Figure 25. The use of the freezing point depressant fluid can provide anti-icing or de-icing protection for the surface onto which the fluid is distributed.

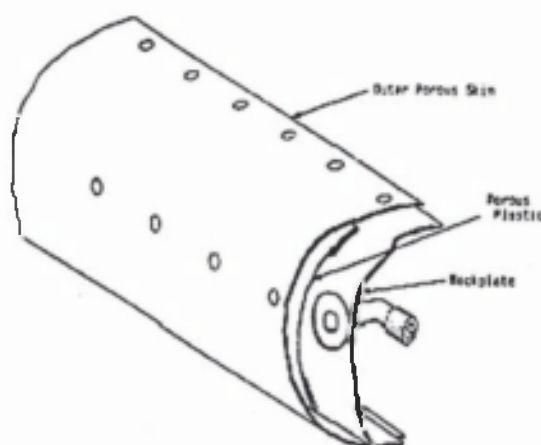


Figure 25: Typical Leading Edge Construction For Fluid Ice Protection System. Source: FAA Aircraft Icing Handbook¹⁵

In light to moderate icing conditions, the system is turned on before, or upon encountering the icing conditions. Then providing a continuous liquid supply in sufficient quantity at the point of maximum impingement, the freezing point of the water-fluid mixture is below the local air temperature, and no ice will form. Thus, the system functions in anti-icing mode.

If icing conditions become too severe, there may be insufficient fluid flow to prevent ice formation. When this occurs, ice will begin to form at the aerodynamic stagnation point. As the fluid continues to be pumped onto the surface, the ice is unable to bond to the aerofoil surface. The ice will grow until aerodynamic forces are sufficient to sweep it off the wet surface into the air stream.

Recent designs utilise adhesive bonding and tend to extend beyond the minimum working region, often forming the complete leading edge. In such cases, the choice of material thickness and some construction features are usually influenced by structural strength requirements. The most recent development is the use of laser-drilled titanium for the porous leading edge, which offers significant weight reduction and improved resistance to impact damage.

FAA Aircraft Icing Handbook¹⁵ suggests for FAR23/ JAR23 class (small single engine aircraft equivalent to Predator B UAV), a wing and tail fluid ice protection equipment requires ~100W of power and has a complete system mass of 18kg and up to 27kg of fluid). The limitation of this system is the amount of fluid in the reservoir.

Typical FAR23/JAR23 rules for small aircraft are to be safe for operation at 0.5g/m^3 LWC icing conditions for continuous icing for 45 minutes. Whilst the time limit is appropriate for a piloted aircraft, where pilot responses and manoeuvres can be tailored to suit the conditions. This clearly is insufficient for UAV where mission durations in icing conditions can be several hours. The UAV operator may not be aware of the aircraft is in icing conditions until handling feedback suggests increased drag. At this stage, ice contamination of the wings may not be the immediate consideration to the UAV operator. At this stage, aircraft safety could be compromised.

2.2. Summary of Conventional Ice Protection Systems For UAV Wing Application

Assuming FAR23/JAR23 rules apply to tactical and medium endurance UAV, the estimated power and complete system (wing & tail ice protection) with no redundancy, and system activation by an ice detector. [Key: O.E. = "Original Equipment" fitted at place of aircraft manufacture"]

Ice Protection System	Estimated Power (W)	Complete System Mass (kg)	Advantages	Disadvantages
Pneumatic Boot De-Icing	250	12.7	Proven, reliable technology Easy retro-fit or fit as O.E.	Significant aerodynamic penalty during ice build-up.
Hot Air	Not Applicable		Bleeding air from a distant compressor & ducting hot air to polymer composite wings precludes its application on UAV wings.	
Electro-Thermal De-Icing	1800	9.9	Proven, reliable technology Easy retro-fit or fit as O.E.	Large power requirements Aerodynamic penalty during ice build-up
Electro-Expulsive De-Icing	500	28	"Concept" flight tested Fit as O.E.	Large mass penalty Aerodynamic penalty during ice build-up

Ice Protection System	Estimated Power (W)	Complete System Mass (kg)	Advantages	Disadvantages
Electro-Impulsive De-Icing	500	28	"Concept" flight tested Fit as O.E.	Large mass penalty Aerodynamic penalty during ice build-up
Fluid Anti-Icing	100	45	Proven, reliable technology Fit as O.E.	Large mass penalty when not in use. System effectiveness only whilst fluid available

This summary strongly suggests that conventional ice protection systems are not entirely suited to UAV systems. Use of any of the above technologies encumbers the UAV designer with solutions and limitations, particularly with aerodynamic performance, power, and system mass. The intent of this study therefore is to address UAV specific requirements.

2.3. Current Solutions to UAV Icing Threats

The presence of ice accretions on the wing leading edge presents an aircraft designer with a significant challenge. UAV wings are designed to provide the aircraft with lift to support long endurance flight with the minimum of fuel consumption. The wing geometry provides an efficient flow crucial to sustaining lift. UAV designer has also spatial geometry, power, and mass limitations.

Any UAV mission is unlikely to take place in dry warm temperature conditions only. An aircraft on an endurance flight has a high probability of encountering localised icing conditions for which the UAV operator may not be aware of. If aware of a potential icing threat, the following options available are:

- Curtail the mission.

The UAV operator on being aware of an icing threat to the aircraft, may curtail the duration of the mission, seek an alternative route or abort the mission and return the aircraft to base. The first and last options reduce the effectiveness of the aircraft.

- Fly above icing cloud.

Flying above the icing cloud may be an option, for some UAV aircraft. However, for many systems, this is not an option as the aircraft may exceed it's flight ceiling.

If the surveillance target is within icing cloud conditions, flying above the target may obscure the photographic sensors.

- Risk the safety of the aircraft to meet mission requirements.

In battlefield conditions, the UAV operator may have no choice but risk the aircraft for vital intelligence.

UAV Aircraft with installed Ice Protection system.

At the time of writing, the Predator class of UAV has fluid de-icing system incorporated. Phoenix is not known to have any ice protection system for its wings. It is not known if other UAV aircraft have ice protection systems fitted.

2.4. Patent and Applicable Technology Survey

2.4.1. Ice Phobic Coatings.

Ice phobic coatings have the potential of reducing the strength of the ice-airfoil surface bond. Less energy is then required to remove ice and the removal is cleaner. Anderson and Reich²⁴ investigated the performance of coatings for reducing the adhesion of ice to an aerofoil surface and found the following conclusions:

- Trials at NASA Lewis Icing Research Tunnel and BF Goodrich Icing Wind Tunnel suggest that the amount of ice and the shape of the ice accreted in the icing wind tunnel are highly dependent on the tunnel and cloud conditions, and not on the surface on which the ice is accreted. However, ice shape is dependent on surface effects, such as water beading and run-back.
- If a coated surface becomes covered with an ice layer, subsequent accretion occurs on the ice surface and, is independent of the substrate characteristics.
- Hydrophobic coatings do not necessarily equate to ice-phobic coatings.

²⁴ : "Test of the Performance of Coatings for Low Ice Adhesion", Anderson D.N., NASA Lewis, Reich A.D, BF Goodrich, AIAA97-0303.

The conclusions were based on a series of trials carried out at NASA and BF Goodrich using a multi-blade rig methodology. A mounting stand holding up to 15 test blade specimens was located in the horizontal centre of the icing tunnel test sections. Un-coated reference specimens were used to compare the adhesive strength of the coated specimens.

At completion of the icing spray run, a hand-held digital force meter was used to apply a load to the ice edge to shear it from the face of the test blade. The force measurements were manually recorded. The adhesive strength was determined by dividing the recorded force by the face area and applying a stress correction factor. Results of these studies is collated in Table 4, below.

Both NASA and BF Goodrich tunnels had an average impact ice adhesive strengths for un-coated aluminium in the range of $7 - 9 \times 10^5 \text{N/m}^2$. Results of the trials with these coatings suggest the Dow "Anti-stick" appeared to reduce the adhesion of ice compared with the un-coated surface. All the other coats showed either increased ice adhesion, or no affect when compared to un-coated aluminium.

Although the PTFE based paints (Surface Seal) are strongly repellent to water, the high pore density was suspected to encourage strong mechanical bonding with ice. As can be seen in Table 4, the coatings had no effect on either the quantity of ice accreted or its appearance. This phenomenon has been verified by others²⁵.

Anecdotal evidence of trials of coated surfaces by NASA Lewis indicate that while ice adhesion to greased surfaces appeared to be nearly zero, if aerodynamic forces held the ice in place, ice would accrete normally.

²⁵ : "Investigation Of Surface Water Behaviour During Glaze Ice Accretion" - Hansman Jr. R.J. & Turnock S.R., J. Aircraft Vol. 26 No. 2 (Feb89).

Studies using high magnification images²⁶ have shown that water coalesces into beads during glaze ice accretion. If the beading process is altered, the final ice shape is expected to alter as well. In this study, the lack of a surface effect can be explained by the fact that once a thin layer of ice has formed over the model, water impinging and freezing on this surface will only see this ice surface. Any surface coatings are totally masked.

Loughborough²⁷ studied static-ice adhesion to surfaces found that the adhesion strength increased linearly with decreasing temperatures. He also showed silicone grease reduced the adhesion of ice, but a small layer of silicone was stripped away with each ice removal. Thus, silicone could not act as a permanent coating. In addition, a poor correlation was shown to exist between low ice adhesion and water repellence.

Further studies by others²⁸ present contradictory evidence. Ice accretion temperature was found to have had no effect over ranges -22°C to about -5°C . A linear decrease in adhesive shear stress was however observed from -5°C to 0°C . In addition, adhesive shear stress was shown to increase slightly with increasing airspeed and increasing droplet size.

²⁶ : "Experimental Evidence For Modifying The Current Physical Model For Ice Accretion On Aircraft Structures" - Olsen W.A. & Walker E., NASA TM87184 (1987).

²⁷ : "The Physics Of The Mechanical Removal Of Ice From Aircraft" - Loughborough D.L., Aeronautical Engineering Review Vol.2 (Feb 52).

²⁸ : "Interface Influences Upon Ice Adhesion To Airfoil Materials" - Reich A., AIAA94-0714 (Jan 94).

Multiple ice removal (to simulate repeated de-icing) showed the unstable nature of the coatings. In some coatings such as silicone grease, ice adhesion shear strength increased with repeated ice removals, whilst others showed the reverse. Rain erosion was shown to increase ice adhesion significantly, even after short exposure times. Thus, durability needs to be demonstrated for repeated ice removal and, for rain impingement at aircraft speeds.

Table 4: Coating Trials With Aluminium Substrates At NASA/ IRT Except Where Noted

COATING	QUANTITY	FORCE TO REMOVE ICE COMPARED WITH UN-COATED ALUMINIUM
<u>76mm Diameter Cylinder</u>		
PPG Surface Seal	1	No apparent difference
Deft Teflon Paint	1	No apparent difference
BF Goodrich Icem	1	No apparent difference
Dow Anti-stick	1	Easier to remove ice, less residual
Dow #2	1	No apparent difference
Freecom Ceram-Kote 54	1	Harder to remove ice
CHD	1	No apparent difference
Dynamic 101W	1	Easier to remove ice, less residual
Dynamic 104	1	Somewhat easier to remove ice
<u>Stacked 25mm Diameter Cylinders</u>		
Dynamic Coatings	4	Two coatings: lower force; Two coatings about the same
Zebco Sales	3	For all coatings: no apparent difference
Novecon Tech.	6	For all coatings: no apparent difference
Microclean	1	No apparent difference
<u>Flat Plate</u>		
Univ. of Virginia Diamond Like Coatings	3	All three coatings: force was up to 50% lower
<u>Stacked 25mm Diameter Cylinders</u>		
Du Pont	6	For all six coatings: typically 50% lower
Hartman	4	Approximately the same force as un-coated
<u>Multi-Blade (10 Blades)</u>		
Du Pont Krytox Grease	1	Lower force than for un-coated
Scotch 5490 PTFE film	1	Lower force than for un-coated
Scotch 5480 PTFE film	1	Approximately the same force as un-coated
Du Pont TE-3447	1	Approximately the same force as un-coated
Du Pont TE-30	1	Approximately the same force as un-coated

Table 4: Coating Trials With Aluminium Substrates At NASA/ IRT Except Where Noted

COATING	QUANTITY	FORCE TO REMOVE ICE COMPARED WITH UN-COATED ALUMINIUM
<u>Multi-Blade (15 Blades) @ NASA/</u> Icing Research Tunnel		
Kaliardos	4	All four coatings: same or higher force required
Line Chemical	2	Both coatings: same or higher force required
ECC	9	Two coatings: 30 - 50% of un-coated force Seven coatings: same force as un-coated
<u>Multi-Blade (15 Blades) @ BFG</u> /Icing Wind Tunnel		
Line Chemical	6	All six coatings: same or higher force required
Line Chemical	4	All four coatings: same or higher force required
ECC	3	All three approximately same as un-coated
Du Pont	11	Two coatings: 30 - 50% of un-coated force Nine coatings: same force as un-coated
Du Pont	12	One coating: 30 - 50% of un-coated force Eleven coatings: same force as un-coated

2.4.2. Hybrid aircraft ice protection systems.

The technology is a variant of the Electro-Impulse De-Icing system called "Electro-magnetic Expulsion De-Icing" system (EMEDS) developed by Innovative Dynamics Inc., and is marketed worldwide by Cox & Co. of New York, USA²⁹.

The system uses an electro-thermal running wet anti-ice system at the leading edge section of the chord is required to be kept free of ice. Runback water freezes aft of the heated zone, where any thermal system would be inefficient. This is because of the low surface wetness factor associated with the runback rivulet flow structure.

²⁹ : "Surface Roughness Due To Residual Ice In The Use Of Low Power De-Icing Systems" - J. Shin; Bond, T.; AIAA93-0031.

In this hybrid system, a mechanical de-icing system is used at these locations to deflect the airfoils' semi-rigid skin, and periodically remove the frozen runback ice accumulation (and/ or the frozen directly impinged water droplets). Cox & Co. calls this de-icing system a "Low Power De-Icing" system (LPDI), which works on the principle of opposing electro-magnetic fields resulting from opposing current flow in adjacent conductors. Typical constructions of the hybrid system concept are shown in Figure 26.

As per Electro-impulse systems, there exists a minimum level of ice thickness accumulation before a particular system can fully clean the surface during a de-icing cycle. The minimum "critical ice thickness" varies from system to system, and depends on the type of ice on the surface (i.e. rime or glaze) and the location of the ice to be removed (i.e. leading edge or downstream). The use of this type of system does require an assessment of the level of residual ice that can be tolerated on the surface. Aircraft with sensitive foils require that no ice be allowed to accumulate on the leading edge. For UAV systems, some ice accretion may be tolerated which may make this hybrid system suitable for main wings.

Typically, LPDI systems are capable of removing ice build-up as thin as 1.3mm and as thick as 50.8mm. This ice thickness can be controlled by varying the de-icing cycle time. This information together with aerodynamic data can be used to determine the extent of the leading edge heating zone.

The system is mounted internally, and therefore does not affect airflow. The system consists of a De-Icing control unit, and an energy storage bank (containing capacitors and electro-mechanical actuators). The EMEDS contains the actuator that imparts a large force on the inside surface of the aerofoil skin in a very short duration, when current is discharged through the capacitors. The force deflects the skin, which shatters the ice build-up. The De-Icing control unit controls the firing sequence of the actuators and de-ice cycle timing. The De-Icing control unit can also control the heater power output.

The actuators are mounted span wise, and built into a composite material housing that constitutes a substantial part of the leading edge structure. As per Figure 26, 3 actuator rows are located chord wise in the structure. Tests have shown that the number of actuators can be reduced to two without affecting performance as also shown in Figure 26.

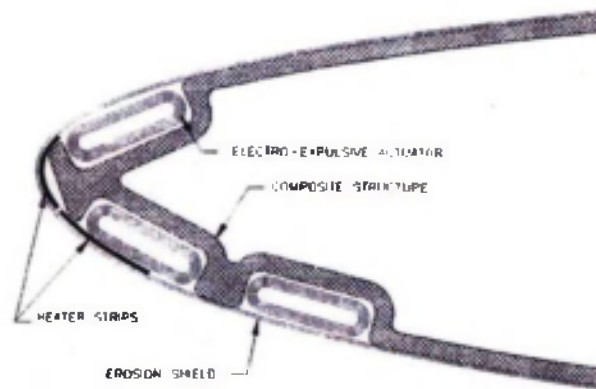


Figure 1 | Schematic of Hybrid System Tested

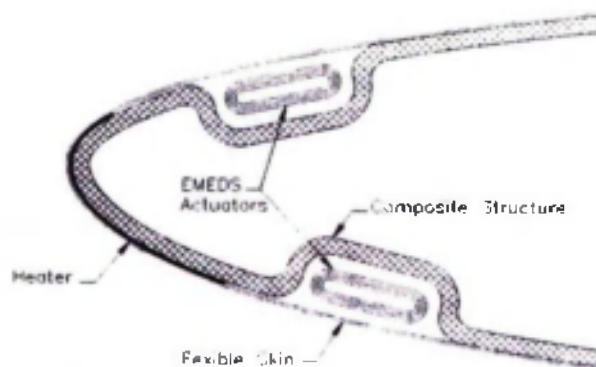


Figure 2 Current Hybrid System Layout

Figure 26: Schematic of Hybrid System. Source: Shin & Bond ²⁹

2.4.3. Digital loudspeaker technology.

A UK Corporation³⁰ "1...Limited" define a Digital Loudspeaker as a loudspeaker that takes digital signals (from a CD for example) as its input, emits digital pulses of sound as its output, and which contains no electrical Digital to Analogue Converter; that task is left to the ear of the listener to perform.

Conventional analogue loud speakers based on moving coils are inefficient and convert only 1% of their electrical power into sound. Thus, a 1-Watt acoustic output needs approximately 100 Watts of electrical input drive power. A vast proportion of the power is dissipated as heat within the loudspeaker. In addition, heavy gauge cables are required to carry this power to a heavy power amplifier. A heavy and bulky cabinet is necessary to support the speaker.

A digital loudspeaker utilises requires only digital input signals, these can be transmitted to the speakers by wireless or infra-red with no loss of quality, using modern error correcting systems. This leads to smaller, lighter, and more efficient loudspeakers that are driven by conventional batteries. This also aids portability, which supports UAV system applications.

The basis of this technology is driven by advances in solid-state actuator technology, specifically piezoelectric and magnetostrictive materials.

³⁰; Source: http://www.cambridgenetwork.co.uk/pooled/articles/BF_TECHHART/1...Limited, Cambridge, UK (2004)

2.5. Proposed Novel Ice Protection Solution: Magnetostrictive De-Icing System.

One of the drawbacks of piezoelectric materials for actuator applications is that a relatively large (several hundred volts) excitation voltage is required. This limits its use in some applications, particularly for UAV systems where power is at a premium. An alternative type of actuator is based on the magnetostrictive effect. Magnetostrictive materials convert magnetic energy into mechanical energy and vice versa. When a magnetostrictive material is magnetised, it exhibits a change in the length per unit length (the Joule effect). The length change is accompanied by a transverse change in dimensions. The effects are the basis of the magnetostrictive actuator.

In most magnetostrictive alloys, typical dimensional changes are of the order of $100\mu\text{L}/\text{L}$. "Giant" magnetostrictive alloys such as Terfenol-D ($\text{Tb}_x\text{Dy}_{1-x}\text{Fe}_2$), Metglass (amorphous Fe_xSi_x) and ferromagnetic shape memory materials such as FePd , $\text{Tb}_{0.4}\text{Fe}_{0.6}/\text{Fe}_{0.5}\text{Co}_{0.5}$ and NiMnGa exhibit strains greater than $1000\mu\text{L}/\text{L}$ at maximum magnetic field saturation than their piezoelectric counterparts for a given input power.³¹ Another potential advantage is that magnetostrictive actuators can be driven by external magnetic fields, thus removing the need for large driving voltages. Other advantages include maximum stress the actuator can safely withstand are displayed in Table 5 when compared with the more established piezoelectric devices.

³¹: Source: <http://etrema-usa.com/core/terfenold/> ETREMA Products, USA (2003)

Table 5: Approximate Characteristics of Mechanical Actuators

Note this table compares a material property based devices (Piezoelectric & Magnetostrictive). Approximations were made by "normalising" the applied current, coil windings, & induced force to the active device geometry to determine the comparative properties.

Actuator Type	Piezoelectric	Magnetostrictive
Max. Actuation Stress (MPa)	1 - 9	90 - 200
Max. Actuation Strain (L/L)	2.3×10^{-5} - 3×10^{-5}	2.25×10^{-3} - 1×10^{-3}
Elastic Modulus (GPa)	44 - 300	40 - 200
Max. Power Density (W/m ³)	9×10^7 - 5×10^8	1×10^8 - 7×10^8
Max. Frequency Range (Hz)	1390 - 55560	83330

Magnetostrictive source: <http://etrema-usa.com/core/terfenol-d/> ETREMA Products. USA (2003).

Piezoelectric source: <http://piezo.com/prodstacks1.html> Piezo Systems Inc. USA (2005).

It is envisaged that the evolutionary steps in this process will be similar to those experienced with the formulation of thick-film piezoelectric materials. The combination of thick-film and silicon technologies will therefore lead to a powerful and economic solution for new types of MEMS actuator. These material alloys are proposed for use for part of a UAV wing de-icing actuator.

2.5.1. Proposed Actuator Description.

The proposed actuator consisting of a drive rod clamped to a fixed plate and mounted either within a current carrying coil, or for these demonstrations permanent magnets. The drive rod is manufactured from a magnetostrictive or ferromagnetic shape memory material (GM) e.g. Terfenol-D. The permanent magnets and drive rod are electrically and environmentally isolated and held in-situ by being set in a polymeric resin. The setting and curing of the resin slightly compresses the GM rod and constrains the magnets from moving apart as shown in Figure 27. A series of actuators are placed on the spar element of the wing section.

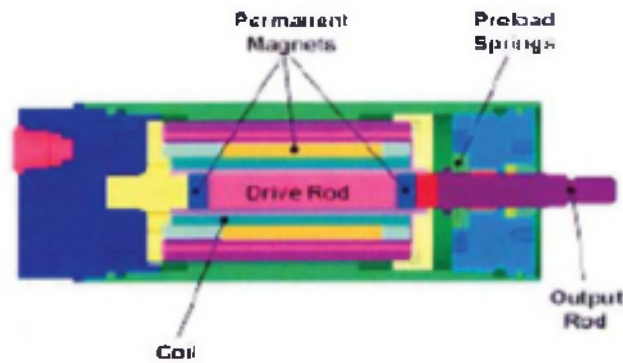


Figure 27: Cross-section Sketch of Actuator Mechanism

Variations on this actuator design include the elastomeric mat with embedded magnetostrictive or ferromagnetic shape memory material particulates, and multi-layer laminates of magnetostrictive film and soft magnetic layers. Both variations present possible solutions for more difficult or inaccessible regions of an aerofoil application and have been declared in US Patent 51030384. In addition, these actuator forms could enable the actuator to be used as an ice detector. These themes are not explored in this document.

2.5.2. Proposed Magnetostrictive De-Icing System Design.

Proposed design has the free end of the GM rod is in direct contact with the internal aerofoil surface as shown in Figure 28. The number and spacing of the actuators are dependent on the shape/ profile of the aerofoil to be protected. The actuators are powered in a set of series and parallel circuits along the span of the aerofoil. The activation of the actuators takes place in the form of balanced chord-wise pairs of top and lower surface actuators. This ensures the surface reaction forces cancel out, and the net force acts on the aerofoil surface.

The circuit feed for the actuators consists of rectifiers, capacitors, transformers, and thyristors with a separate UAV pilot or automated system controller triggered by an ice detector. Conventional power and control systems utilised by EIDI or EEDI systems form the remaining portion of the system.

2.5.3. Proposed De-Icing System Operation

Proposed de-icing operation with this magnetostrictive device is as follows. Ice is allowed to accrete onto the airflow surface until it reaches an undesirable thickness. Upon de-icing system activation by the UAV pilot or ice detectors, the current pulse is triggered. This has the effect of inducing a large strain in the GM rod. This in turn causes a small displacement of the local region of the aerofoil non-airflow surface.

The accreted ice together with the airflow surface is briefly accelerated away by this localised displacement. However, the equally rapid removal of the displacement loading causes the aerofoil surface to return to its relaxed state. During this relaxation, the ice-aerofoil surface bond is broken. Several of these impulses are required to excite the aerofoil surface, to enable the accreted ice to de-bond and shatter. Aerodynamic forces then result in the removal of the shattered ice shards.

Working in upper and lower surface pairs, the actuators are cyclically activated from wing root to wing tip. The pulse widths for an effective de-icing operation are dependent on available power, the severity of the icing threat, and wing structural and damping properties. It is expected the pulse widths would range from the ultrasonic (20kHz) to the auditory (1Hz) with energies exceeding 500 Joules.

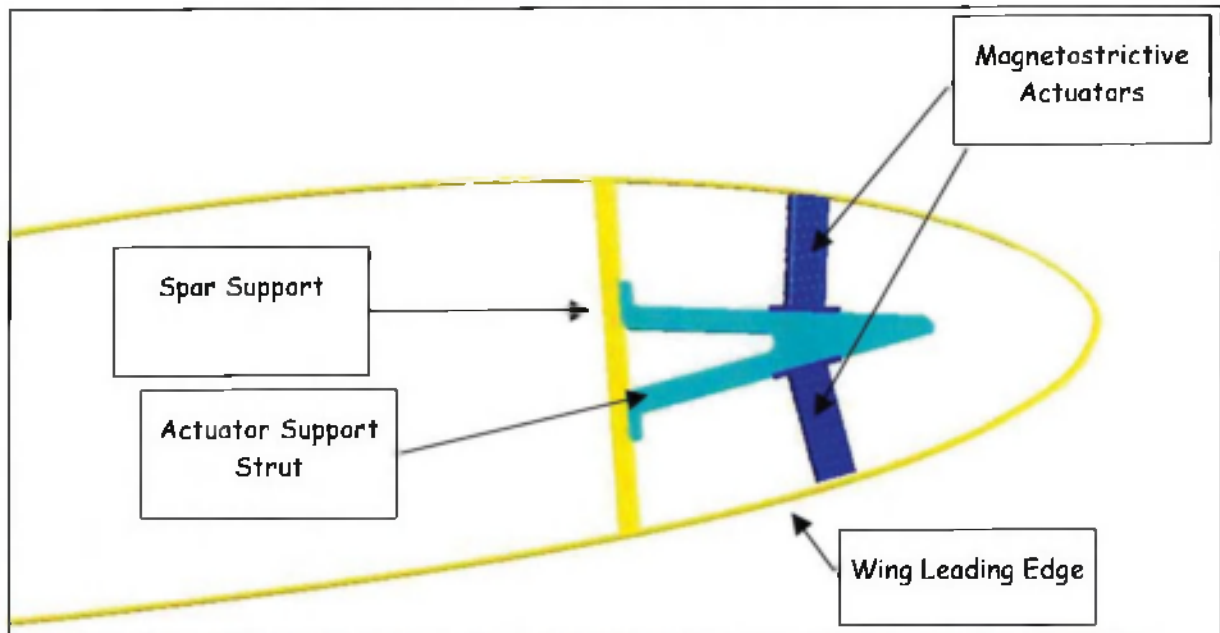


Figure 28: Proposed Wing Application Of Magnetostrictive Actuator For De-Icing

The action of the proposed system is based on a variation of the electro-impulsive system discussed in section 2.1.5 above. As stated earlier, the basic circuit shown in Figure 29 will be an imitation of the EIDI circuit shown in Figure 10, with a wing installation as shown in Figure 13. The proposed design is proposed to be modular, allowing rapid installation and removal in the field by UAV system technicians. This ensures the aircraft is not carrying superfluous systems and devices on mission where the probability of an icing encounter is very small.

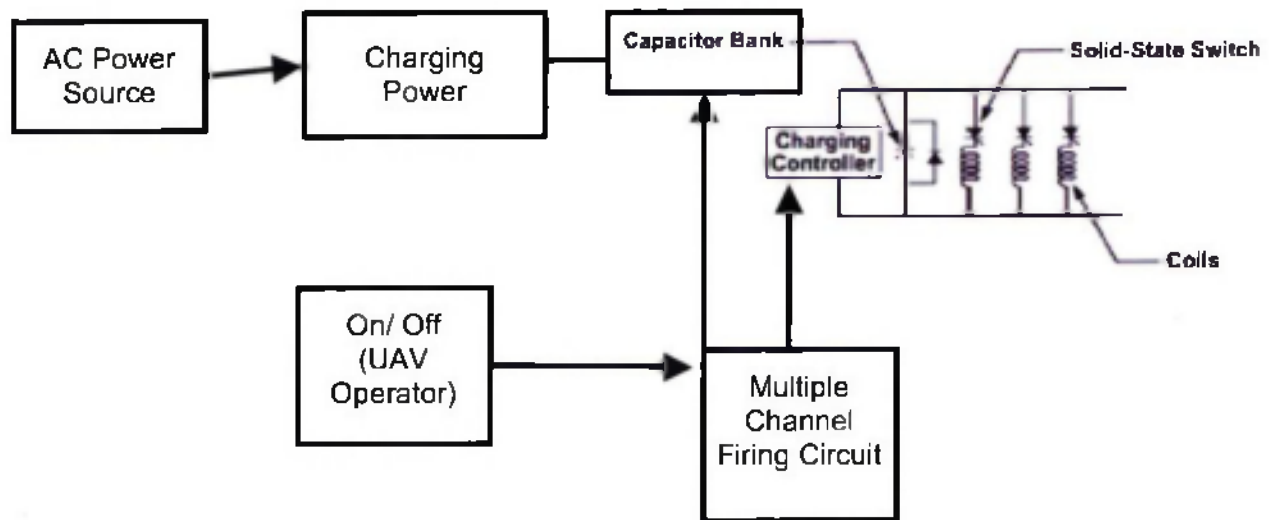


Figure 29: Proposed Magnetostrictive De-Icing Circuit

2.5.4. Possible System Advantages.

- 1) Giant magnetostrictive materials (GMM) technology ~ 5 - 10 years old.
 - a) Uses rare earth metals in alloy composition in the "solenoid core" rather than standard nickel iron composition.
 - b) Core does not move but "expands" and "contracts" under magnetic field influence.
 - c) Expansion & contraction is 1000 times greater than piezo-electric based materials and 10000 times that of standard nickel iron alloys.
 - d) Applied loads may be significantly higher as conversion of magnetic force into displacement is many times more efficient than that of standard nickel iron alloys.
- 2) The actuator does come into contact with non-airflow surface. The surface material or local wing geometry is not a strong variable on actuator de-icing performance as in EIDI or EEDI systems.

- 3) Alternatives to GMM are piezo-electric core actuators (cheaper, but not as efficient) & GMM film (covered by existing patents).
- 4) Actuators are currently being developed for and targeted at mobile phone and high performance entertainment markets. Thus as a high volume, consumer commodity item, costs are expected to be relatively low per unit item. This is beneficial to UAV systems, which are intended to be low cost systems.
- 5) Price tag per main wing system extrapolated from customised "Consumer" devices ~£60K. This estimated cost is assumed production volumes for a wing and tail de-icing system consisting of 10-fitted actuators/wing, software based controller (+no redundancy), charge circuit, harnesses etc. that are incorporated into the wing at manufacturing stage.
- 6) Advantages of proposed GMM based system for UAV's are:
 - a) Greater power and mass efficiencies can be obtained when compared to existing ice protection systems.
 - b) Commercial development is further driving to reduce device size, weight, and power requirements.
 - c) Device/ system is software driven, with accompanying self-diagnosis & fault detection. Removal of standard redundancy requirements reduce the software development costs. This is considered necessary for low cost, modular design necessary for UAV applications.
 - d) Device may be sensitive enough to act as an icing sensor/ detector without requiring additional hardware. Ice accreting on the airflow surface would change the local stiffness of the wing, which would translate into a change in the local dynamic response. Further work is required to determine if this concept can be converted into a reliable form of ice detection.

7) Disadvantages of proposed GMM based system for UAV's are:

- a) The GMM actuator technology is unproven for de-icing, although the de-icing method is well established.
- b) The de-icing system needs to be considered early within the design process, and incorporated within the wing and tail plane manufacture.
- c) EMI during capacitor discharge maybe a serious problem to other aircraft systems, and cannot be easily masked from enemy detection systems.

CHAPTER 3: EXPERIMENTAL & COMPUTATIONAL METHODS

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3. EXPERIMENTAL & COMPUTATIONAL METHODS.

To establish this proposed de-icing technology is suited for the task, data was sought from manufacturer's and quotations for a system that could serve as demonstrator de-icing system was sought. Due to the inordinate cost of a customised "on-off" demonstrator system, a commercially available actuator in the form "sound bug" was purchased. This would serve as a low cost demonstrator platform, from which adequate understanding of the de-icing mechanism could be obtained at a minimum of investment in fixture and tooling.

To demonstrate the magnetostrictive actuator as a basis for de-icing of wing leading edges for UAV systems, a number of physical experiments and computational simulations were undertaken.

3.1. Composite Flat Plate-Actuator Interaction Study.

The easiest means of determining the actuator's interaction with a composite UAV wing, was to experiment and analyse it's function with the simplest representation of a composite UAV wing, held in an unconstrained i.e. "free-free" state.

The "free-free" method of holding a test article in space with little to restrain it from movement, ensures the effects of constraint does not influence the test article's dynamic properties. This "free-free" test is undertaken by suspending the test item with elastic cords or bungee. The test article has strain gauges or accelerometer devices bonded to the surface to measure a physical parameter such as local strain or acceleration. A means of exciting the test part such as striking it with a calibrated hammer is then carried out. Data acquisition software capture the strain gauge or accelerometer signals, and converts and manipulates the data into a time history form.

The choice of "free-free" allows minimal investment in test fixtures and provides valuable test data for further studies with mathematical tools such as Finite Element Analysis.

To simplify understanding, test, and analysis of the proposed method of de-icing, initial analysis and tests were performed on a simple flat plate. This allowed verification of the correlation between the analytical techniques to be applied to the more complex real aircraft wing structures, on which tests were not possible during this research, with tests performed on the simple structure.

The wing laminate construction is proprietary to the manufacturer. A composite laminate of the approximate thickness and lay-up was prepared by taking a microscopic cross-section from the UAV wing. The information was used to "reverse engineer" a composite sandwich laminate construction. A flat plate laminate test specimen replicating the wing laminate build was then prepared, whose geometry, materials, lay-up, and mechanical properties were known.

As stated above, the high cost of purchasing customised actuator system, the testing was initially carried out using a "Sound Bug"³². A Sound Bug is a small TERFENOL-D actuator that is encased within a polymer housing and has the appearance similar to that of a PC mouse - see Figure 1. The actuator as displayed in Figure 2 was designed to be linked to music sources such as a CD player. The actuator is then attached to a flat surface such as a car window, work tables etc. by means of a suction pad. On activation, the surface behaves as a loudspeaker for the CD player.

³²: Sound Bug is manufactured by Newland Technologies, Hull, England.
<http://www.newland-technologies.co.uk>



Figure 1: "Sound Bug" Supplied by Newland Technologies, Hull, England.

The Sound Bug is powered by 3 AAA batteries and connects to any standard audio jack found on Walkmans, Mini Disks and computers. An individual Sound Bug can produce a maximum output of -75db. A pair of Sound Bugs linked together but, attached to different surfaces provides the user with a stereo sound effect.

In order to demonstrate the de-icing philosophy, the Sound Bug was chosen due to its easy availability and price to perform the tests. The Sound Bug, stripped of its enclosure was the cheapest Magnetostrictive device in the market place. However, this is reflected in the quantity of TERFONEL-D used in actuator. A criticism of the Sound Bug device here can be its the lack of power. Unfortunately, this reservation was only observed when physical testing took place.

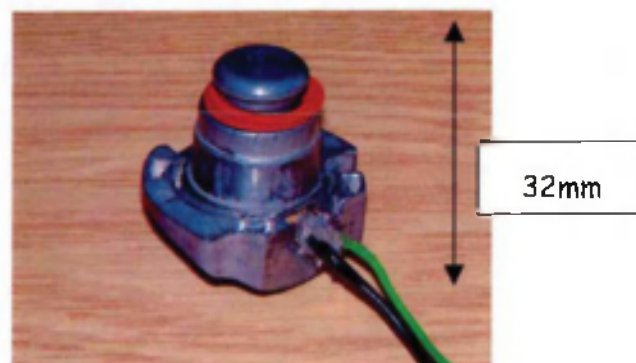


Figure 2: Actuator taken from a Sound Bug Device ~mass: 56g

3.1.1. Mathematical Modelling of Flat Plate-Actuator Interaction Test.

Before physical testing, a study was undertaken to determine the likely natural frequencies to be expected. This ensures better control of the test input and output parameters, and confirms an understanding of the underlying physics.

Roark's³³ suggests for a "free-free" beam, the natural frequency for the nth mode is

$$f = \frac{K_n}{2\pi} \cdot \sqrt{\frac{E \cdot I \cdot g}{w \cdot L^4}} \quad \text{Equation 1: Beam Natural Frequency Calculation}$$

where K_n is mode number vector, E is Elastic modulus, I is area moment of inertia, w is the load including beam mass and L is the length of the beam. The parameters employed in the calculations are shown in Appendix A4.

With reference to Appendix A4, the predicted first 5 natural frequencies for the composite test panel are:

1. 262.3Hz
2. 722.5Hz
3. 1417Hz
4. 2342Hz
5. 3501Hz

³³: "Roarks' Formulas for Stress and Strain", Case 4a and Table 36, Young W.C, 6th Edition, McGraw-Hill Publications.

3.1.2. Modelling Assumptions.

The above-predicted natural frequencies for the composite test panel are based on the following modelling assumptions:

- The material properties employed in this study were based on data from commercially available woven glass-epoxy fabric and aramid honeycomb core. The average mechanical properties of the glass-epoxy material used are considered valid for this construction i.e. within a small statistical error. Those error margins are dependent on the variables described below.
- The study ignores the effects on the mechanical properties of ambient temperature, moisture ingress, and panel flaws due to manufacturing imperfections. The amount of resin in the finished composite is not always as expected, but the fibre volume is fixed. This leads to an actual layer thickness that is slightly different to the nominal one. Thus, the position of the fibres through the laminate is affected and hence bending properties.
- The lay-up is idealised with perfect transfer of loads across the discrete layers. In reality, load transfer between plies is far from perfect, and will vary due to laminate build, geometry, and manufacturing processes employed.

- The modulus of the laminate was determined using Classical Laminate Theory (A description of the theory is not necessary for the discussion here). These calculations assumed the classical laminate theory would generate a reasonably accurate Effective Modulus and Strength for a multi-ply, multi-orientated glass-epoxy laminate (see Appendix A7). This entails the input of the average mechanical properties into LAP laminate analysis software³⁴, within which the calculations were carried out. A brief description of the software is provided below.

LAP (Laminate Analysis Program) is a software tool for the design and analysis of composite material laminates. The Classical Laminate Theory solution algorithms employed utilises the basic material engineering properties at the layer level, definition of stacking sequences, and definition of loadings (in-plane and bending loads, deformations and hygrothermal effects) or constraints. Material stiffness properties can be constant (i.e. Linear), or functions of stress or strain (i.e. Non-Linear). Results thus obtained are displayed interactively on multiple windows at the user's request.

The critical material allowables used in the LAP calculation are:

Material : Woven glass/epoxy fabric.

Elastic Modulus in 11 direction $E_{11} = 2.39\text{E}+010 \text{ N/m}^2$

Elastic Modulus in 22 direction $E_{22} = 2.42\text{E}+010 \text{ N/m}^2$

Shear Modulus in 12 direction $G_{12} = 5.66\text{E}+009 \text{ N/m}^2$

Poisson's Ratio $\nu_{12} = 0.194$

Thermal expansion coefficient in 11 direction $\alpha_{11} = 1.332\text{E}-005 / ^\circ\text{C}$

Thermal expansion coefficient in 22 direction $\alpha_{22} = 1.444\text{E}-005 / ^\circ\text{C}$

Hygrothermal coefficient in 11 direction $\beta_{11} = 0.011$

Hygrothermal coefficient in 22 direction $\beta_{22} = 0.011$

Elastic Modulus $E_{\text{resin}} = 5\text{E}+008 \text{ N/m}^2$

Poisson's Ratio $\nu_{\text{resin}} = 0.3$

Nominal fibre volume fraction $V_{f \text{ nom.}} = 0.5$

³⁴ : LAP version 4.0 - laminate analysis software obtained from <http://www.anaglyph.co.uk>

Where E is the elastic modulus, G is the Shear modulus, ν is Poisson's ratio, α is the thermal expansion coefficients, β is the resin moisture coefficients, V_f is fibre volume fraction. 11, 22, 12 values represent fabric warp, weft and cross-weave directions respectively,

The software predicts the effective modulus of the laminate is:

Effective Stiffness:

Membrane

$$E_{11} = 2.18237e+010 \text{ N/m}^2$$

$$E_{22} = 2.18237e+010 \text{ N/m}^2$$

$$G_{12} = 7.12696e+009 \text{ N/m}^2$$

$$\nu_{12} = 0.26971$$

The E value was used in the above calculation in section 3.1.1 above

3.1.3. "Free-Free" Composite Flat Test Plate Preparation.

Initial trials with a replica of the wing composite sandwich build with a honeycomb core, found the strain responses were barely detectable despite significant amplification. An alternative monolithic construction (Balanced 6 ply lay-up orientations: $[0^\circ/+45^\circ/90^\circ/90^\circ/+45^\circ/0^\circ]$) allowed measurements to be recorded.

Using a balanced symmetrical 6-fabric ply lay-up, a woven glass-epoxy composite test panel was manufactured. The cure process is proprietary to Dunlop Aerospace and is not required for the discussions here.

Upon curing the composite test panel, three strain gauges were bonded to the top surface. The three strain gauges were bonded in different positions on the test panel, one of which can be seen in Figure 3.

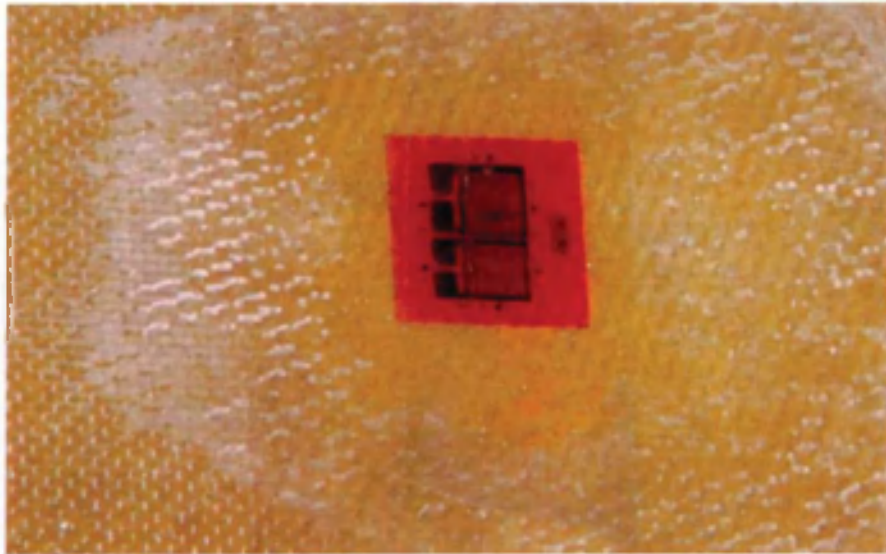


Figure 3: Strain Gauge Bonded To Test Panel (Prepared Area Visible)

3.1.4. Test Equipment Used In "Free-Free" Plate Study.

Equipment used in frequency testing:

- Frequency source laptop
- 6 channel amplifier
- BNC to serial converter
- 3 bridge completion circuits
- Magnetostrictive actuator
- Test rig
- Insulated wire
- 3 AAA batteries
- Actuator control module.

The first step was to support the composite test panel within the free-free rig consisting of a steel box frame from which the test panel was supported with Bungee cord.

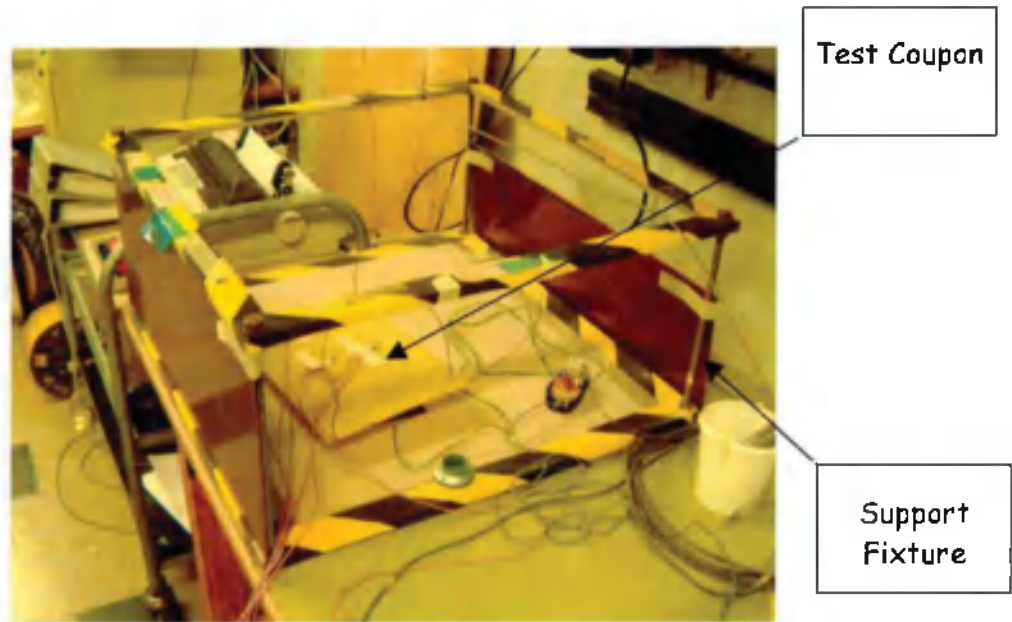


Figure 4: Composite Test Panel Supported In Test Rig

The length of these cords ensured the composite test panel was suspended over the actuator, and in direct contact with the actuator. The choice of bungee allowed the panel to move with very little constraint.

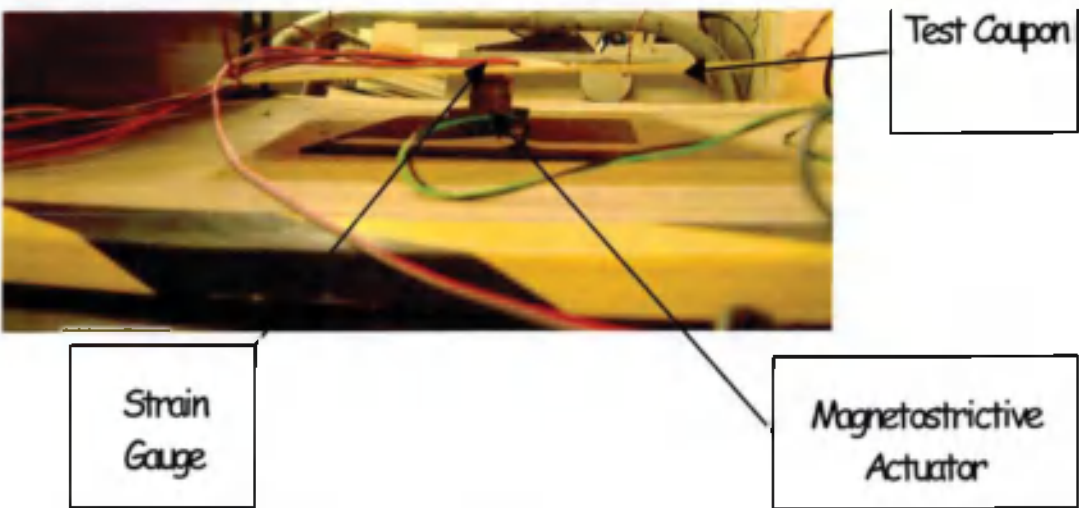
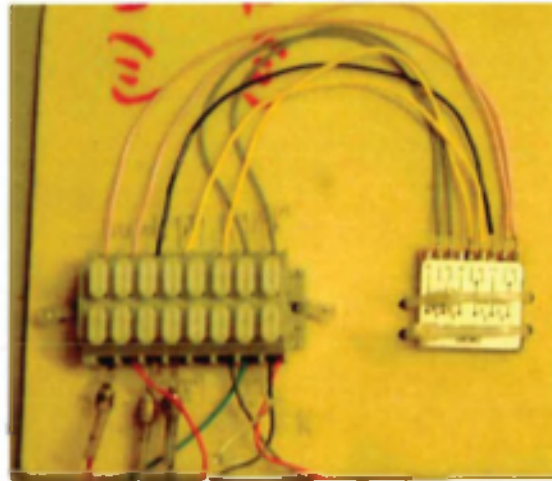


Figure 5: Test Panel Suspended Over Actuator

Once the test panel was suspended over the actuator, the strain gauges were connected to the Wheatstone bridge completion circuits. Within this circuit, a strain gauge was either resistively parallel with another strain gauge in the same orientation, or in series with a calibration resistor.

Figure 6: Bridge Completion Module



Upon the strain gauges being linked to the bridge completion circuit, the amplifier was connected. The important operation of calibrating the strain gauges could then commence. The aim of the calibration was to "balance" the bridge circuit i.e. matching potential drop across the strain gauges and resistors in the "null-strain" condition. An imbalance would have affected measured results. This task was simplified by the use of amplifiers with built-in circuit balance functions.

When the system was balanced, a data acquisition laptop computer was linked to the amplifier output via a BNC to serial converter. In parallel, the Magnetostrictive actuator was connected to its own control module. This module in turn linked to a second computer laptop. This second computer was used to produce the input signal using a "Free-ware" program called "SigJenny"³⁵. The program uses the sound card of a computer to produce an output signal. The frequency of the signal can be selected from within the program, while the amplitude is determined by the volume on the sound card.

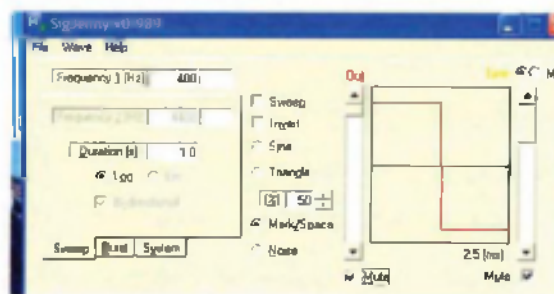


Figure 7: "SigJenny" Sound Signal Generator Program

The actuator was used in conjunction with its control unit (See Figure 8) provided with the Sound Bug because it supplied the actuator with the correct power rating. In using the supplied hardware and software, the supplied control module ensured the actuator did not exceed its designed operating parameters.

Due to hardware limitation within the amplifier, only one channel from the amplifier was available for measurements. For each frequency tested, the amplified signal was recorded by the data acquisition computer. Two strain gauges were used together to form a single bridge. This meant only one bridge module was used in a half bridge set-up.

³⁵: "SigJenny" software available from <http://www.natch.co.uk/downloads/SigJenny.html>.



Figure 8: Sound Bug Control Unit



Figure 9: Experimental Set-up

Due to the earlier trials, testing commenced at 100Hz. After a sample measurement was taken, the input frequency was then be increased by 10Hz until a maximum of 700Hz. To obtain clear results for each frequency, the input signal was started 5 seconds before the strain gauge response was recorded. Throughout the tests, the amplitude/volume was to be kept constant at 75db.

The response sample taken at each frequency step lasted 2 seconds. Within this two-second period, 24000 readings were taken. This was calculated to create clear result with no chance of aliasing occurring in the results.

Each frequency sample was then saved on the data acquisition laptop in an independent file format '*ASD.'

3.1.5. Data Processing.

For each frequency test, the raw strain gauge response data was processed into a form that could be employed later within the FEA study. Much of the following steps take place automatically within the mathematical algorithms built into the data acquisition software. In general, the following processes are carried out:

- The software, analyses the signals for the strain gauge, and employs electronic band pass filters to separate and quantify each frequency using the Fast Fourier Transform (FFT). The signal is digitised into varying levels against time. A sine and cosine function is performed on these points, which will produce the quantities of the frequencies during the sampling period.
- The sine waves of the different frequencies can be summed to form a complex waveform. The components of these different frequencies are of fixed amplitude and are continuous sine waves. The result is periodic, repetitive, and predictable spectrum.

- The resultant waveforms will be in root mean square (rms) value of the acceleration, typically expressed in its squared form as g^2_{rms} . The resolution of this analysis is 1Hz i.e. 1Hz bandwidth. Thus, the spectral output reported is g^2_{rms}/Hz that describes the average power seen in 1Hz bandwidth at any particular frequency.

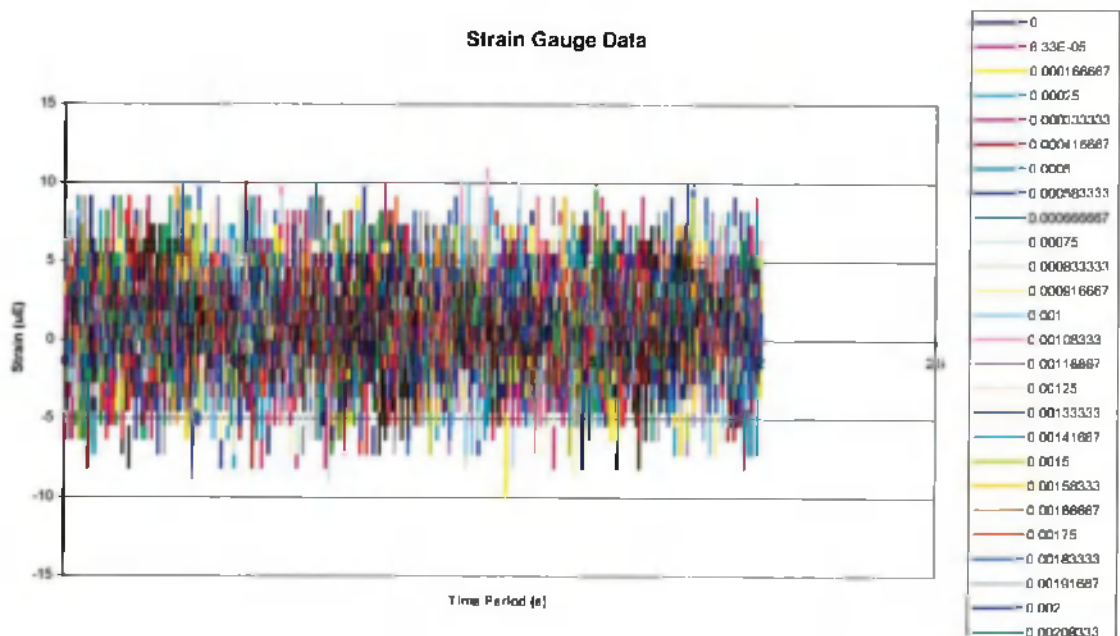


Figure 10: Strains observed at 110Hz

Typical raw strain gauge output produced in the physical test generated resembles a "random noise spectrum", an example of which is shown in Figure 10. This output is difficult to work with. An alternative output from the software was able to convert the strain output into root mean square power output at each frequency, and a number of identifiable response signals were then produced. This is shown in Figure 11.

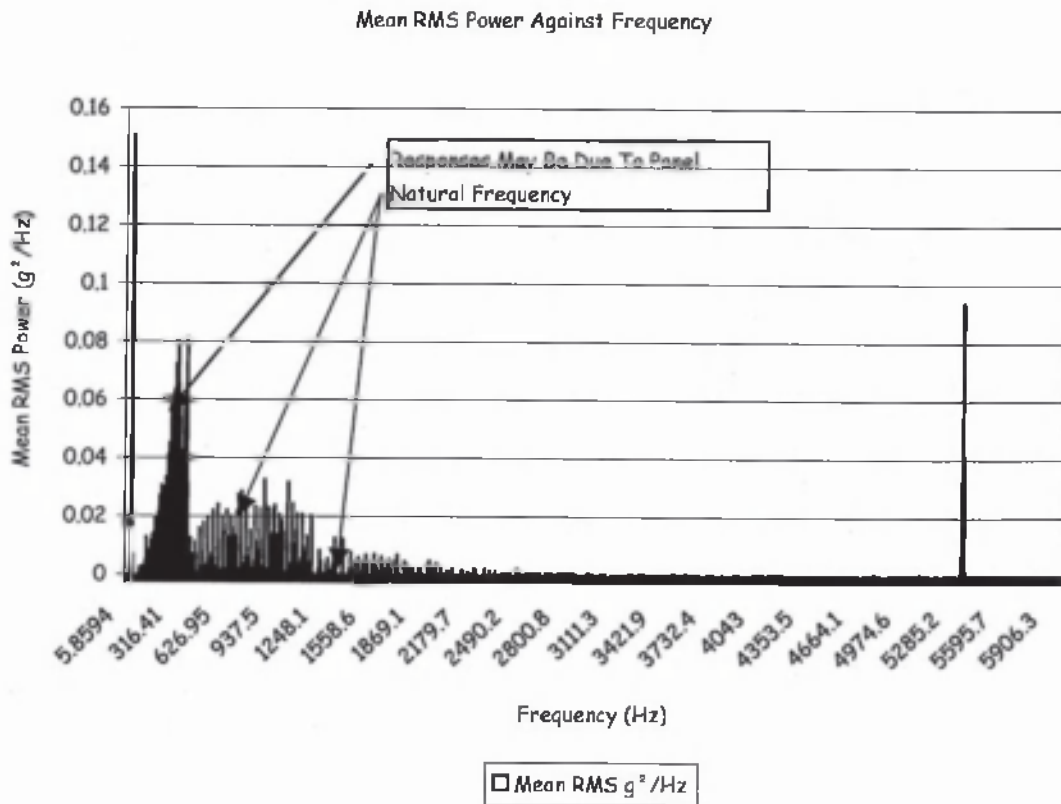


Figure 11: Strain Signals converted To RMS Power v Frequency

The observed strain - time vibration spectrum displayed in Figure 10 is now represented in the frequency domain by a power spectral density function as demonstrated in Figure 11.

To determine the local force on the test panel then, Newton's second law remains valid i.e.

$$Force (N_{rms}) = mass (kg) \times acceleration \left(\frac{m}{s^2} \right)_{rms}$$

or

$$Force (kgf_{rms}) = mass (kg) \times acceleration (g)_{rms}$$

Equation 2: Determination of Applied Force_{RMS}

To obtain the acceleration G_{RMS} necessary to establish the force on the panel applied by the actuator. The overall root-mean-square (RMS) value is equal to the square root of the area under the curve. An integration procedure by Irvine³⁶ was used to determine the area beneath the curve. The power spectral density (psd) then represented as a series of continuous segments, where each segment is a straight line on a log-log plot.

Assuming that point 1 at frequency f_1 and point 2 at frequency f_2 , each has an amplitude a_1 and a_2 respectively in G^2_{rms}/Hz .

$$\Delta dB = 10 \cdot \log \left[\frac{a_2}{a_1} \right]$$

The slope between the two coordinates (f_1, a_1) and (f_2, a_2) in a log-log plot.

$$\text{slope } N = \frac{\log \left[\frac{a_2}{a_1} \right]}{\log \left[\frac{f_2}{f_1} \right]}$$

and area under segment 1 is

$$\text{area} = \int_{f_1}^{f_2} \left[\frac{y_1}{f_1^n} \right] \cdot f^n \cdot df$$

The data obtained from the physical test was processed in MS Excel spreadsheet in Appendix B.

The results obtained for this data is shown below in Figure 12.

³⁶ : Irvine, T., "Integration Of The Power Spectral Density Function", Mar00.
<http://www.vibrationdata.com>.

RMS Power Density Conversion To Acceleration G_{RMS}

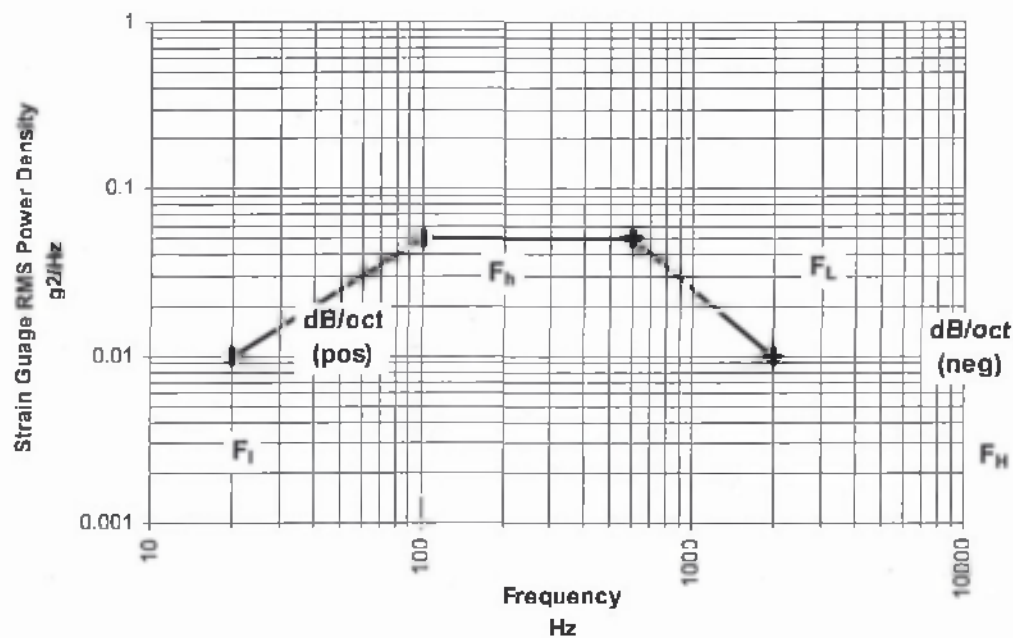


Figure 12: RMS Power Density Conversion To Acceleration G_{RMS}

3.1.6. "Free-Free" Flat Plate Study: Prediction V Test Data.

Panel natural frequencies predicted by the beam calculation are (calculations shown in Appendix A4:

Mode	Composite Panel Estimated Natural Frequencies in "Free-Free" Mode (Hz)
1	262.3
2	722.5
3	1417
4	2342
5	3501

This calculated data is useful in that panel strain responses at specific frequency regions could be attributable to the panel natural frequencies i.e. as shown in Figure 11. These panel responses can be removed from the results data to determine the panel strain.

3.2. Validation of FEA Wing Model By "Free-Free" Modal Testing.

The above actuator-flat plate interaction study in Figure 12 suggests the actuator can deliver an even magnitude impulse force over that range of frequencies from 100Hz to 700Hz. By tuning the actuator, an impulse load can be delivered at a frequency more suited to the dynamic properties of the wing, and the de-icing function being sought.

The dynamic properties for the Phoenix UAV wing could be measured experimentally using "Free-Free" modal test. The objective of the study was to determine the natural frequencies of an actual wing, and build a finite element model (FEA) replica of this wing. If the FEA model were to produce a similar modal response, then confidence in the stiffness and therefore predicted results is assured.

It is outside this thesis to discuss the detailed mathematics employed in the Finite Element Analysis Method (FEA). However, a general context sensitive description is provided here. A more detailed description is provided by Chillery³⁷.

FEA assumptions typically start with a body in equilibrium i.e. having zero acceleration or a result of all external applied forces equal to zero. This approach is labelled static analysis. However, all real physical structures when subjected to loads or displacements, behave dynamically. The additional inertia forces, from Newton's second law, are equal to the mass times the acceleration. If the loads or displacements are applied very slowly then the inertia forces can be neglected, and a static load analysis can be justified. Hence, dynamic analysis is a simple extension of static analysis.

³⁷ : Chillery M., "Fundamentals of Numerical Techniques for Static, Dynamic and Transient Analyses", Benchmark magazine, Apr04. <http://www.nafems.org>.

In addition, all real structures potentially have an infinite number of displacements. Therefore, the most critical phase of a structural analysis is to create a computer model, with a finite number of massless members (mesh) and a finite number of joint (node) displacements that will simulate the behaviours of the real structure.

The mass of a structural system, which can be accurately estimated, is lumped at the nodes. In addition, for linear elastic structures, the stiffness properties of the members, with the aid of experimental data, can be approximated with a high degree of confidence. However, the dynamic loading, energy dissipation properties, and foundation (boundary conditions) for many structures are difficult to estimate.

Most structures are in a continuous state of dynamic motion because of random loading such as wind, vibrating equipment, or human loads. These small ambient vibrations are normally near the natural frequencies of the structure, and are terminated by energy dissipation in the real structure. If the frequency of an applied load is significantly lower than the first natural frequency of the structure, the computationally less intensive and simpler static analysis would be valid. If not, a vibration analysis is required, which can determine whether the structure is likely to resonate in response to the load.

Special instruments (e.g. accelerometers) attached to the structure can easily measure the motion, and hence if the structure is in resonance. Ambient vibration field tests are often used to calibrate computer models of structures and their foundations.

Only modal analysis, which does not consider damping, is considered here. The aim of a modal analysis is to find the frequency values and displacement shapes of the natural frequencies of the structure. The analysis calculates the Eigenvalues, which are the squares of the natural frequencies. Eigenvectors, which describe the maximum displaced shape the structure has when excited at this frequency was also determined.

The Eigenvectors does not give the actual values of displacement, but rather the relative displacements of each node in the model; hence the term "mode shape". The displacement response of a structure to a specific forcing frequency would comprise some contribution to all the Eigenvectors in varying proportions, dependant on the value of forcing frequency and the amount of damping for each mode.

3.2.1. UAV Physical Wing Description.

A damaged starboard wing from a crashed Phoenix UAV was supplied to the author for experimental purpose by UK Ministry of Defence research agency DERA (now QinetiQ). The 1570mm span, tapered wing is a hollow composite construction of unknown asymmetrical profile (possibly NACA65-012). The chord length at the wing root is 775mm. The wing tip was removed before receipt by the author. Remaining outboard edge has chord length of 556mm.

The airflow surface skins appeared to be constructed from a sandwich consisting of woven glass- epoxy fabric (at least 2 plies on both faces)/ honeycomb core (5mm thick). Internal spar and supporting struts are an aluminium alloy fabrication that has been riveted to the skins. The aileron together with the internal mechanisms controlling the aileron had also been removed. The wing mass is 0.622kg.

The wing was repaired using a Dunlop Aerospace proprietary fabric and room temperature cure resin system.

3.2.2. Description of Modal Test Method.

The philosophy of the "Free-Free" testing has already been discussed above. This approach ensures the presence of any physical constraint and in particular, the constraint's location does not influence the first few natural frequencies observed. The data generated with this study will be used to verify the UAV wing FEA model behaves in a dynamically representative way to the actual physical specimen. A "like for like" similar FEA study replicating the physical test conditions was also carried out. This is discussed below in section 3.2.6 below.

For this particular study, if the wing first or second modes are near the operating frequency of the actuator mechanisms, some vibration amplification may be possible. In "normal" circumstances, this phenomenon would be avoided, as the wing may resonate uncontrollably, and damage the component. In addition, vibration would also result in increased wear and in flight, additional wing aerodynamic flutter. However, it is the author's contention, that if wing resonance by means of vibration amplification is used carefully, it may assist in the de-icing process.

Due to the specialist nature of the software and hardware, Dunlop Aerospace dynamic specialists³⁸ carried much of the test work out, under direction from the author.

3.2.3. Wing Modal Test Set-up

The testing consisted of 3 stages:

- creating a geometric model of the test item,

³⁸ : Tim Parker and Paul Smith, Dynamics Laboratory, Dunlop Aerospace, Coventry CV6 4AA. E-mail: tim.parker@dunlop-aerospace.com.

- fitting the instrumentation to the test specimen and,
- acquiring the frequency response functions (FRF) required to carry out the modal analysis.

Creating Geometric Model

The wing geometry was created within the LMS Coda-X Geometry module by importing a fully parametric model from the Unigraphics CAD system. The creation of a geometric model was required to allow the mode shapes calculated in the Modal Analysis to be animated. The model is a simplified wire frame of the upper wing structure, which has sufficient nodes to accurately describe all the mode shapes.

Due to hardware limitations, a limited number of accelerometer channels were available. To make best use of the available number of input channels, only the upper face was modelled. The approximate symmetry of the wing profile, allowed a reasonable assumption that most of the observed modes in the upper breeze surface will have a mirror image in the lower airflow surface mode shapes. In hindsight, there are many more asymmetric modes shapes, than symmetric mode shapes, and such data would be not available. However, as a FEA model verification exercise, the test data derived thereof would suffice.

At each node position, a PCB single axis accelerometer was bonded to the wing upper surface. The acceleration of the structure could then be measured in a single axis direction. To allow for the sloping surfaces, the angles of the local nodal coordinate system for each accelerometer was changed. The model can be seen in Figure 13.

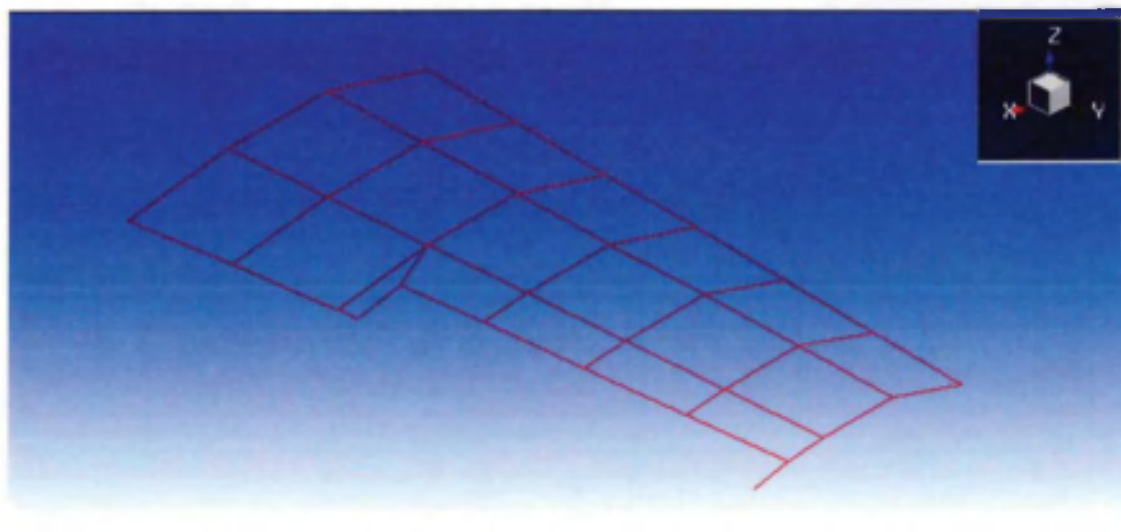


Figure 13: Geometric Model of the UAV Wing.

Instrumentation & Set-up

Care was taken to ensure the wires from the accelerometers were held clear of the structure to prevent unwanted vibrations being fed back into the structure. Any loose parts on the structure were taped up to prevent them rattling or generating erroneous signals. The data was acquired using the Hewlett-Packard VXI and LMS Cada-X Impact Testing Monitor using a bandwidth of 1024 Hz.

To obtain the FRF functions, measurements of the wing structure's response to a known force input was required. A PCB impact hammer supplied the force input. A force transducer in its tip measured the input force. The PCB single axis accelerometers glued onto the structure measured the response of the structure to the force input.

Test Procedure

The structure was suspended from bungee cords in a Free-Free arrangement as shown Figure 14. The wing surface was tapped at various locations with a PCB impact hammer.

Averaged FRF functions were calculated from 10 measurements and the resolution of 0.5 Hz was checked using Nyquist diagrams. A matrix of FRF functions was built up which was then used in the Modal Analysis.

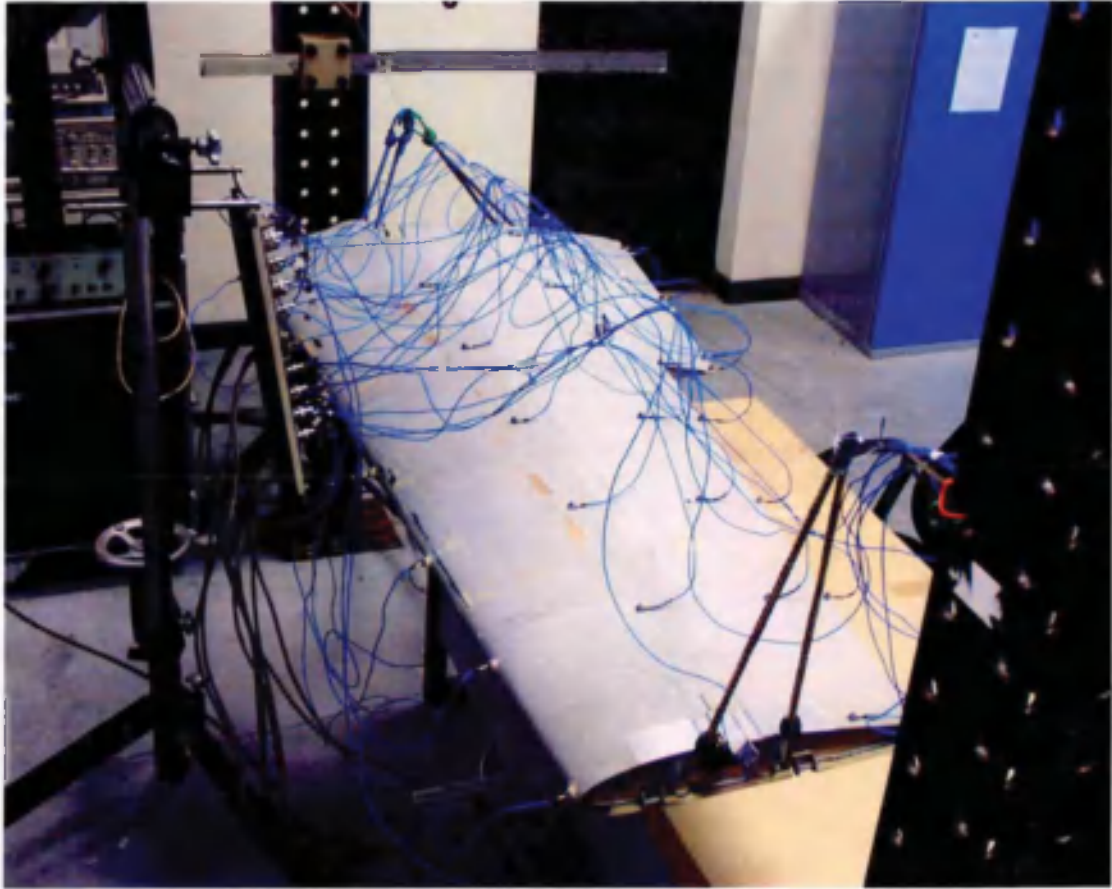


Figure 14: Modal Test Arrangement For The UAV Wing.

3.2.4. Data Analysis & Processing.

The LMS Coda-X Modal Analysis module was then used to calculate the modal parameters, resonance frequency, damping, and mode shapes. The wing was analysed and a number of modes were extracted. The analysis software used a Stabilisation diagram to show the frequencies at which the modes can be found. The modes appear as columns of red "s" symbols. Figure 15 shows the stabilisation diagram for the frequency bandwidth of 12 - 120 Hz.

There are about 6 modes present for the structure within the selected bandwidth. This is indicated by a column of red stable equations occurring at the same frequency. A number of validation tools were used to determine the quality of the extracted modes.

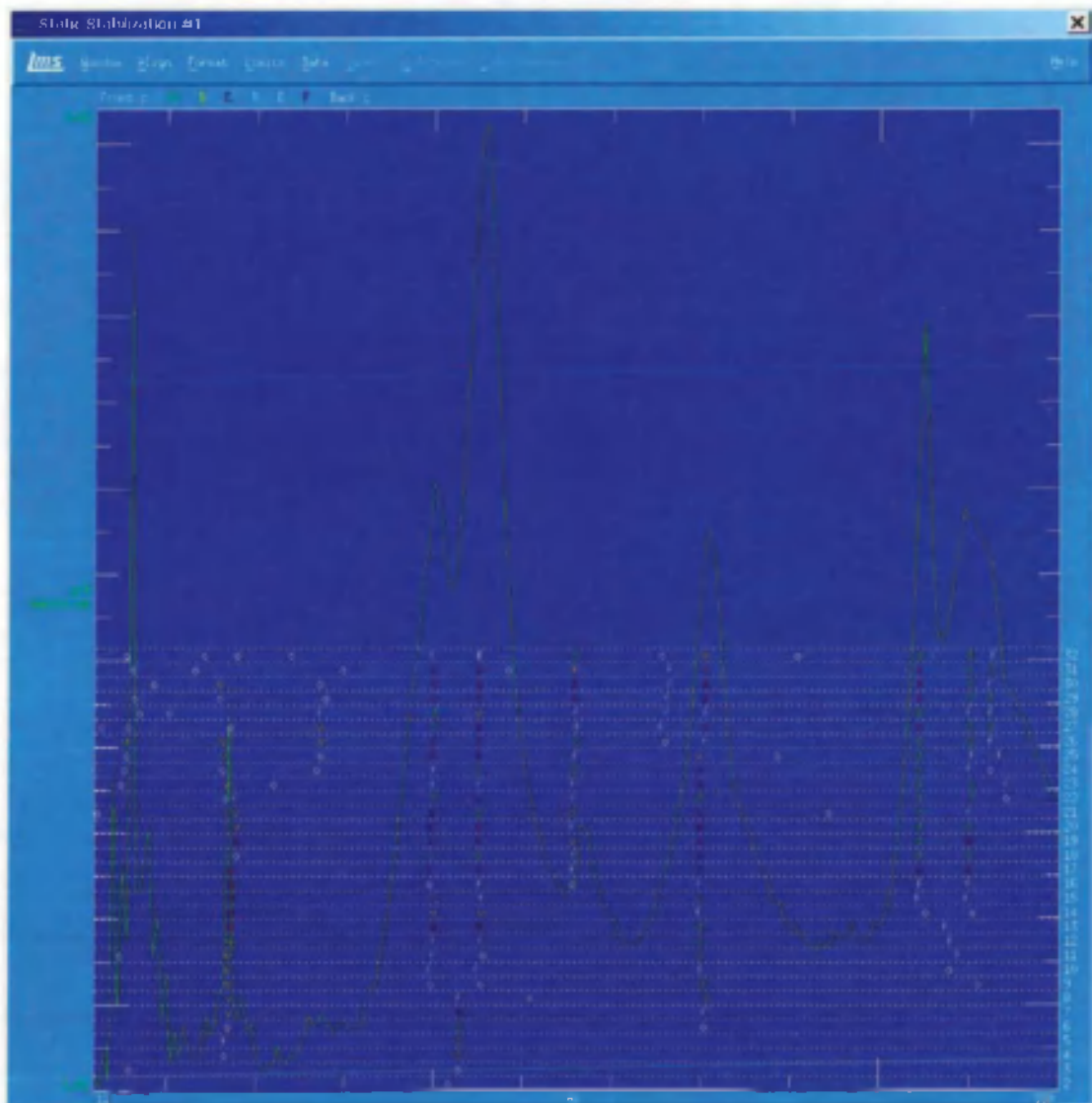


Figure 15: Stabilisation Diagram For A Bandwidth Of 12 - 120 Hz Showing Sum Of FRFs (Green).

Validation

A number of validation tools were used to ensure that the modes were real structural modes of the wing.

Mode Phase Deviation (MPD):

For a real structural mode, all the nodes on the structure should be moving in phase with each other. The MPD gives the level of phase scatter of all the degrees of freedom on the structure. The phase scatter should be as close to 0 degrees as possible.

Mode Complexity (MOV):

The Mode Complexity is based on the theory that when mass is added to a structure its resonance frequency should decrease. The MOV value is a weighted percentage of all the degrees of freedom on the structure that do show a decrease in the natural frequency. The MOV value should be as near to 100 % as possible.

The results show the modes extracted during the analysis and the results from the validation.

3.2.5. Test Determined Modal Shapes & Frequencies.

Table 1 shows the frequency of the modes calculated using the Time Domain MDOF curve fitter for the wing. Also shown are the MPD and MOV values for each mode.

Table 1: Modal Test Results For UAV Wing				
Mode	Frequency (Hz)	MPD (Deg)	MOV (%)	Ref Fig
1	55.4	4.79	98.72	Figure 16
2	66.2	7.11	96.91	Figure 17
3	80.9	1.36	100.00	Figure 18
4	104.6	5.54	99.36	Figure 19
5	110.4	5.59	99.31	Figure 20
6	138.6	6.57	99.66	Not Shown
7	153.1	5.19	98.79	Not Shown

Table 1: Modal Test Results For UAV Wing				
Mode	Frequency (Hz)	MPD (Deg)	MOV (%)	Ref Fig
8	160.4	3.79	99.49	Not Shown
9	192.3	6.40	100.00	Not Shown
10	208.6	6.97	98.36	Not Shown

Table 1 shows that all the modes have low MPD values and high MOV values.

Plots of the first 5 Mode shapes are displayed below (Note the reported discrepancy between Mode figure value and the caption description is due to the removal of unstable modes).

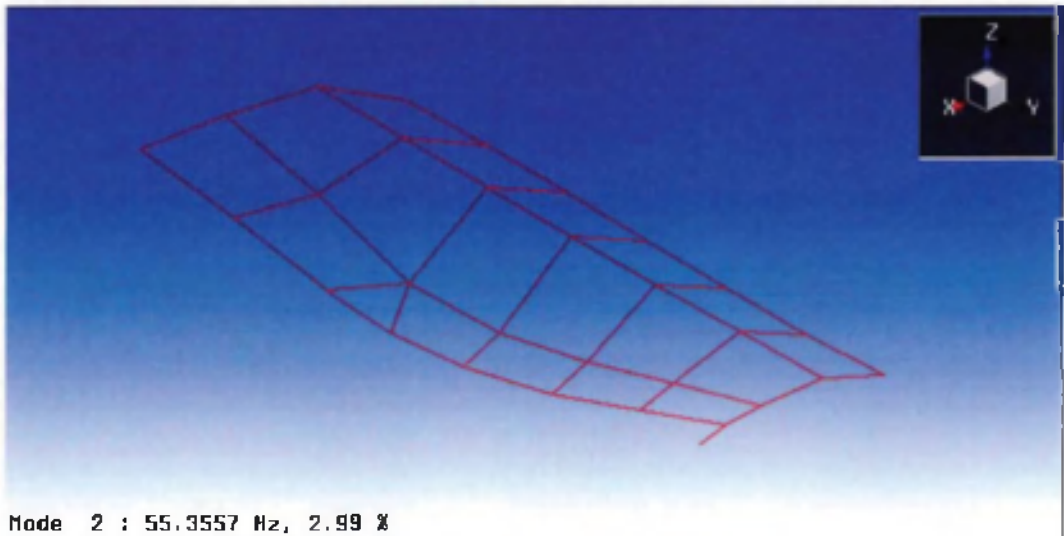


Figure 16: Mode 1 At 55.4Hz

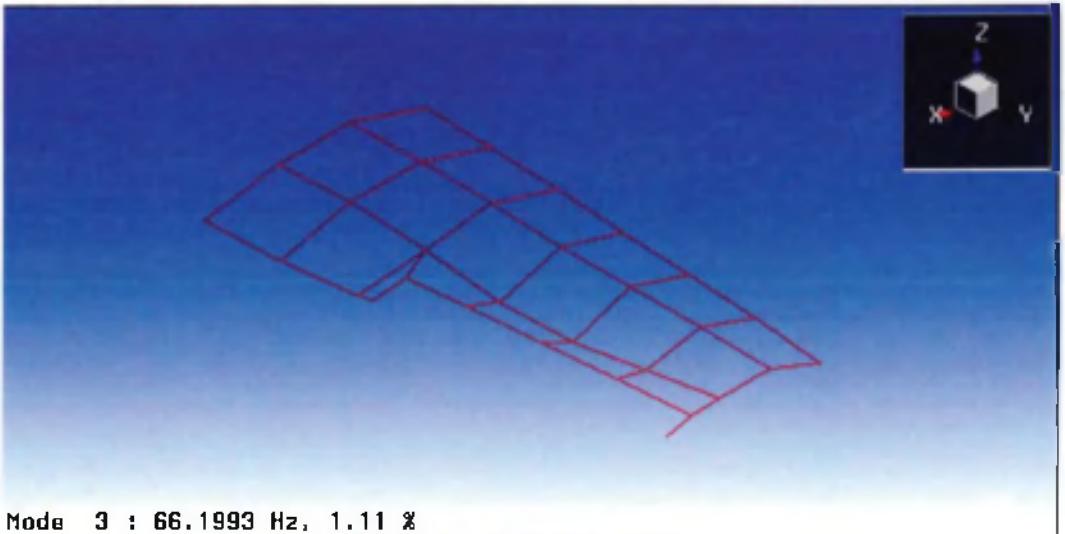
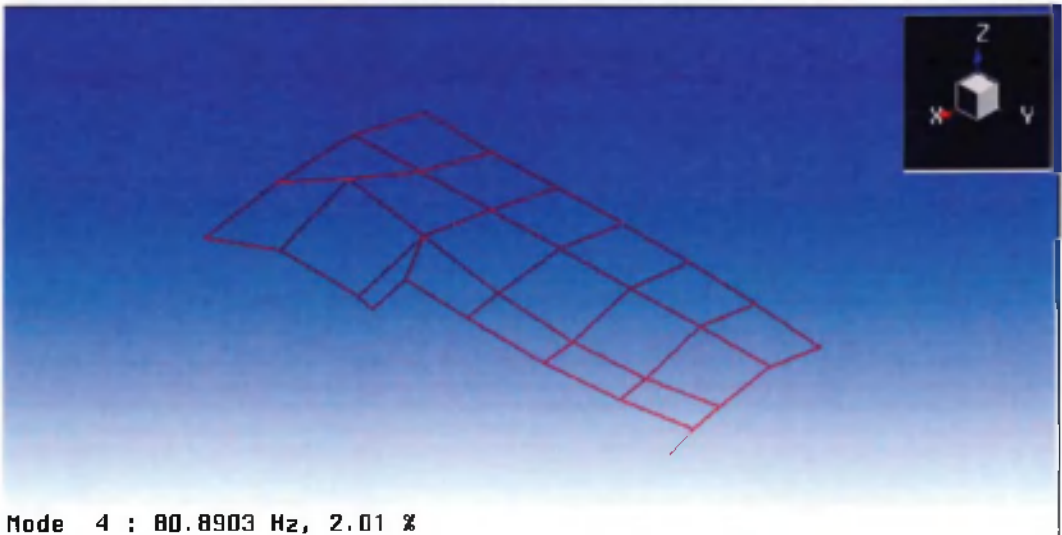


Figure 17: Mode 2 At 66.2Hz

Figure 18: Mode 4 at 80.9Hz



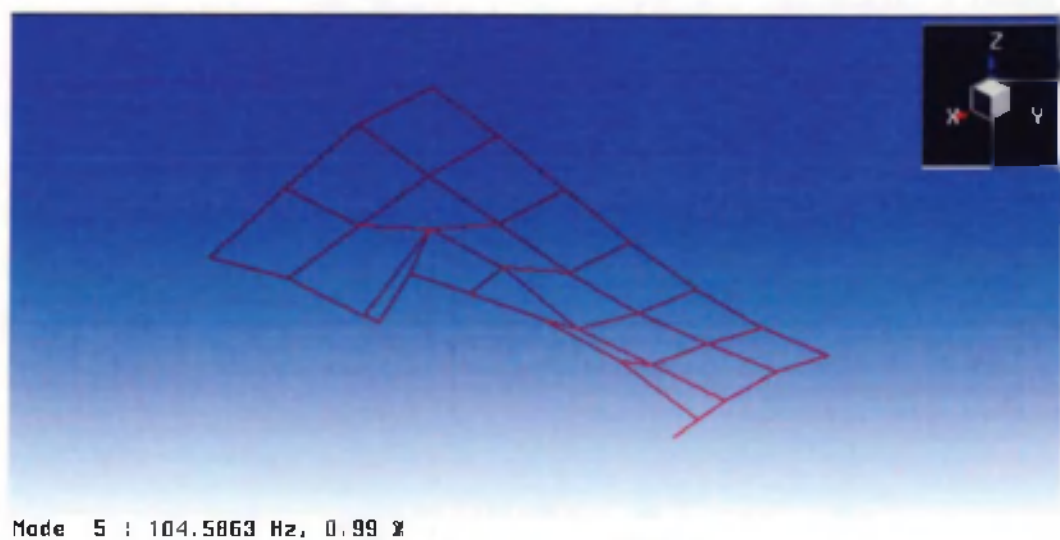


Figure 19: Mode 4 at 104.6Hz

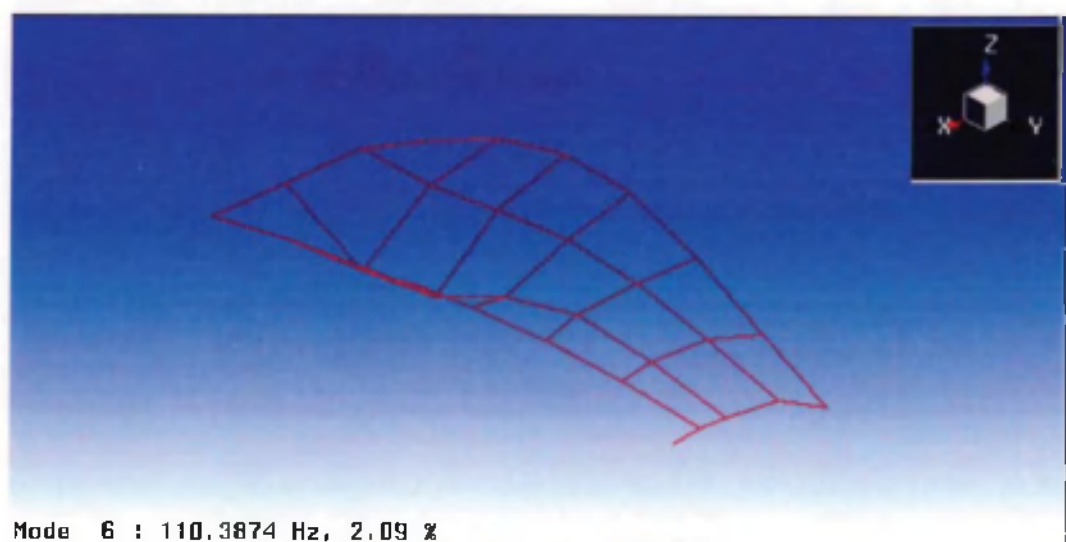


Figure 20: Mode 5 At 110.4Hz

3.2.6. MSC PATRAN/ NASTRAN FEA Model Set-up.

A description of the finite element analysis method has been given above in section 3.2 above and by Chillery³⁷. Although the actual physical wing specimen is asymmetrical in profile, the ice accretion, and related aerodynamic data available for NACA0012 profile wings are more extensive³⁹. Replicating some of these icing and aerodynamic conditions using the NACA0012 wing section, ensures the FEA based simulation of these conditions provides realistic results. This further builds confidence in the FEA model responses to mechanical and environmental stimulations of the proposed de-icing system.

A CAD model was generated using Abbott & Doenhoff⁴⁰ NACA0012 profile coordinates. This model was imported into FEA pre-processor MSC.PATRAN v2004, where the FEA mesh, load conditions, boundary constraints, and material properties are defined. To solve problems, a solver FEA code MSC.NASTRAN v2004 is utilised in this study. The results files thus generated are imported and processed by MSC.PATRAN to produce resultant plots and graphical simulations.

3.2.7. Input Parameters & Assumptions.

The FEA mesh model produced with MSC PATRAN is shown in Figure 21 below.

The mesh model had the following properties:

No. of Nodes: 1963

³⁹ : "Experimental and Computational Ice Shapes and Resulting Drag Increase for a NACA0012 Airfoil", Shin J. and Bond, T.H, NACA TM-105743.

⁴⁰ : "Theory of Wing Sections", Abbott, I.R. & Von Doenhoff, A.E., Dover Publications Jun 58.

No. of Elements: 2230 (Quad4 & Tri3 elements shell elements; metallic materials: PSHELL element property - homogenous definition; composite materials: PCOMP element property - laminate definition)

FEA Model mass: 537g

Coordinate [0,0,0] at wing root, +X: aft; +Y: Up; +Z: Outboard

Centre of Gravity: X: 0.350m; Y: -0.4m; Z:0.783m

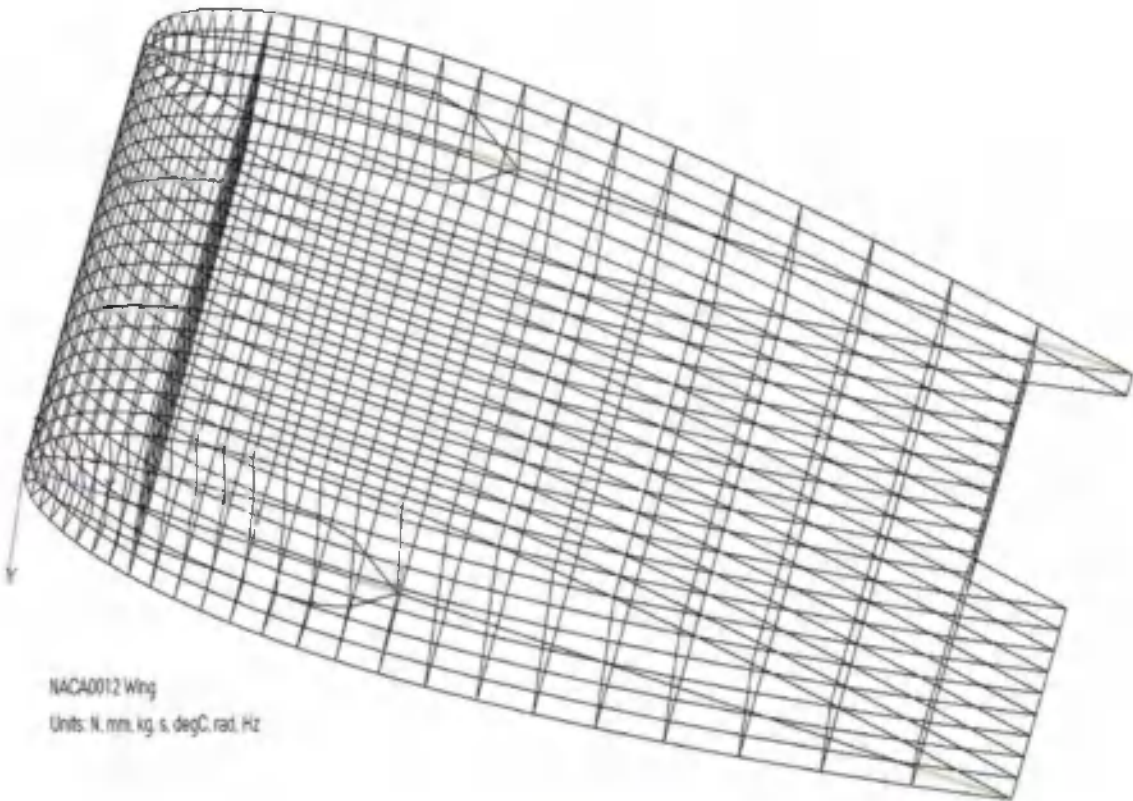


Figure 21: FEA Model of NACA0012 Wing Representing Phoenix UAV Wing

The material properties used are as declared in section 3.1.2 above.

Table 2: Composite Material Properties		
Engineering Property	Woven glass/epoxy	Honeycomb
Elastic Modulus 11 (MPa)	23900	1
Elastic Modulus 22 (MPa)	24200	1
Poisson Ratio 12	0.194	0.3
Shear Modulus 12 (MPa)	5660	1

Table 2: Composite Material Properties		
Engineering Property	Woven glass/epoxy	Honeycomb
Shear Modulus 13 (MPa)	5500	14.5
Shear Modulus 23 (MPa)	5500	24.8
Ultimate Tensile Strength (MPa)	214.2	1
Ultimate Compressive Strength (MPa)	352.3	1
Ultimate Shear Strength (MPa)	71.3	1
Bonding Shear Stress (MPa)	7.8	0.35
Density (g/cm ³)	1.91	0.0489
Nominal Ply Thickness (mm)	0.28	5
Thermal Expansion Coefficient (x10 ⁻⁶ m/m°K)	13.32	Use facing sheet
Thermal Expansion Coefficient - Through Thickness (x10 ⁻⁶ m/m°K)	47.7	19.4
Structural Damping Coefficient	0.1	---
Room temperature "S-basis" properties.		

The composite lay-up of the wing airflow surfaces are:

- 0° Glass-epoxy fabric
- 90° Glass-epoxy fabric
- 0° Honeycomb
- 90° Glass-epoxy fabric
- 0° Glass-epoxy fabric

The spars and struts are assumed to have been developed from 1mm thick aluminium alloy BS L72 in T3 temper condition clad sheet fabrication. Mechanical properties are as indicated in used in ESDU series 00932 Metallic Materials Handbook. Values used in the FEA study are:

MECHANICAL PROPERTIES (LONG TRANSVERSE DIRECTION)	BS L72
ESDU series 00932, Volume 4, Section 6	(T3 Temper)
Allowable Ultimate Tensile Stress, F_{tu} (MPa)	401
Allowable Tensile Yield Stress At Strain 0.002, F_{ty} (MPa)	254
Allowable Compressive Yield Stress At Strain 0.002, F_{cy} (MPa)	230
Allowable Ultimate Stress In Pure Shear, F_{su} (MPa)	233
Elongation, e (%)	13

MECHANICAL PROPERTIES (LONG TRANSVERSE DIRECTION)	BS L72
ESDU series 00932, Volume 4, Section 6	(T3 Temper)
Elastic Modulus In Tension (Below Proportional Limit), E (GPa)	72
Elastic Modulus In Compress (Below Proportional Limit), E _c (GPa)	66
Shear Modulus G, (GPa)	27
Poisson's Ratio, μ	0.33
Thermal Expansion Coefficient α (10^{-6} /°C)	23
Density, ω (g/cm ³)	2.79
Structural Damping coefficient	0.03
Room temperature "B-basis" Properties	

For the "Free-Free" study, no constraints or boundary conditions were applied to the model. The analysis method employed was NASTRAN SOL103 (Normal Modes Analysis). The mass calculation set to "coupled", and a damping figure was assigned to the composite and metallic materials used.

The output file "Real Eigenvalue Extraction" method used was Lanczos for which 35 desired roots were sought. The initial solver condition commenced at 1.0Hz. This allowed the solver to ignore the rigid body motions usually reported at around 0Hz or very small values. This practice is computationally less costly.

3.2.8. Output Mode Shapes & Frequencies.

Results files were imported in MSC.PATRAN and a table of the first 20 modal frequencies produced - see Table 3 below. The table also compares the FEA predicted modal frequencies with the natural frequencies observed in the physical wing. The FEA corresponding mode shapes are also displayed, with the lower airflow surface removed for clarity.

In line with Mil-STD 810F Method 514.4, Annex B Para 2.7.6, similarity can be argued *"..if resonant frequencies are within ±5% of each other"*. The FEA model reported modal analysis when compared with the measured "free-free" physical test data developed are listed in Table 3.

Table 3: NASTRAN Predicted UAV Wing Modal Frequencies

FEA Calculated Mode No.	FEA Calculated Eigenvalue	FEA Calculated Frequency (Hz)	Similarity to Test Frequency (Hz)+ Ref Figure	Wing Developed Test Mode No.
1	139.27	1.88		
2	149706.30	61.58 (Figure 22)	55.4 (Figure 16)	2
3	248988.00	79.42 (Figure 23)	80.9 (Figure 18)	3
4	270631.10	82.80		
5	280644.30	84.31		
6	440647.70	105.65 (Figure 24)	104.6 (Figure 19)	4
7	465278.10	108.56		
8	473732.00	109.54 (Figure 25)	110.4 (Figure 20)	5
9	495802.80	112.07		
10	578618.10	121.06		
11	595455.90	122.81		
12	702739.40	133.42		
13	769169.60	139.58	138.6	6
14	774076.40	140.03		
15	783850.40	140.91		
16	809391.60	143.19		
17	839896.10	145.86	153.1	7
18	924438.90	153.02		
19	1059822.00	163.85	160.4	8
20	1081024.00	165.48		

Comparison between test data developed modal frequencies and those reported by the FEA model demonstrates significant correlation between the FEA model and actual wing test specimen. This gives strong confidence in the quality of the FE analysis being undertaken, and the results obtained from it.

The number of modal frequencies reported by the FEA is significantly larger. This is not unexpected given the number of nodes and elements deployed in the FEA model when compared to the sparseness of the nodes employed in test specimen, and all which are located in the upper aerofoil skin.

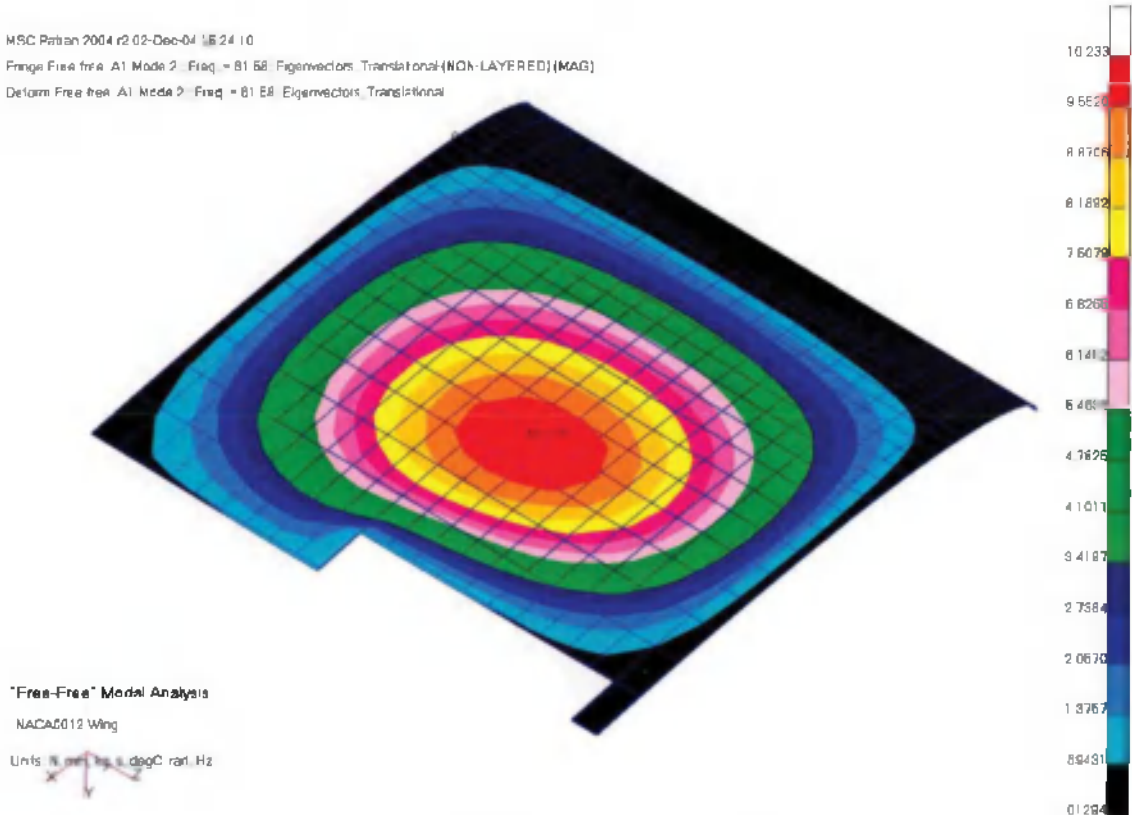


Figure 22: Mode 2 at 61.6Hz

MSC Patran 2004 r2 02-Dec-04 16:26:19
Fringe Free-free A1 Mode 3 Freq = 79.418 Eigenvectors Translational(NON LAYERED) (MAG)
Deform Free-free A1 Mode 3 Freq = 79.418 Eigenvectors Translational

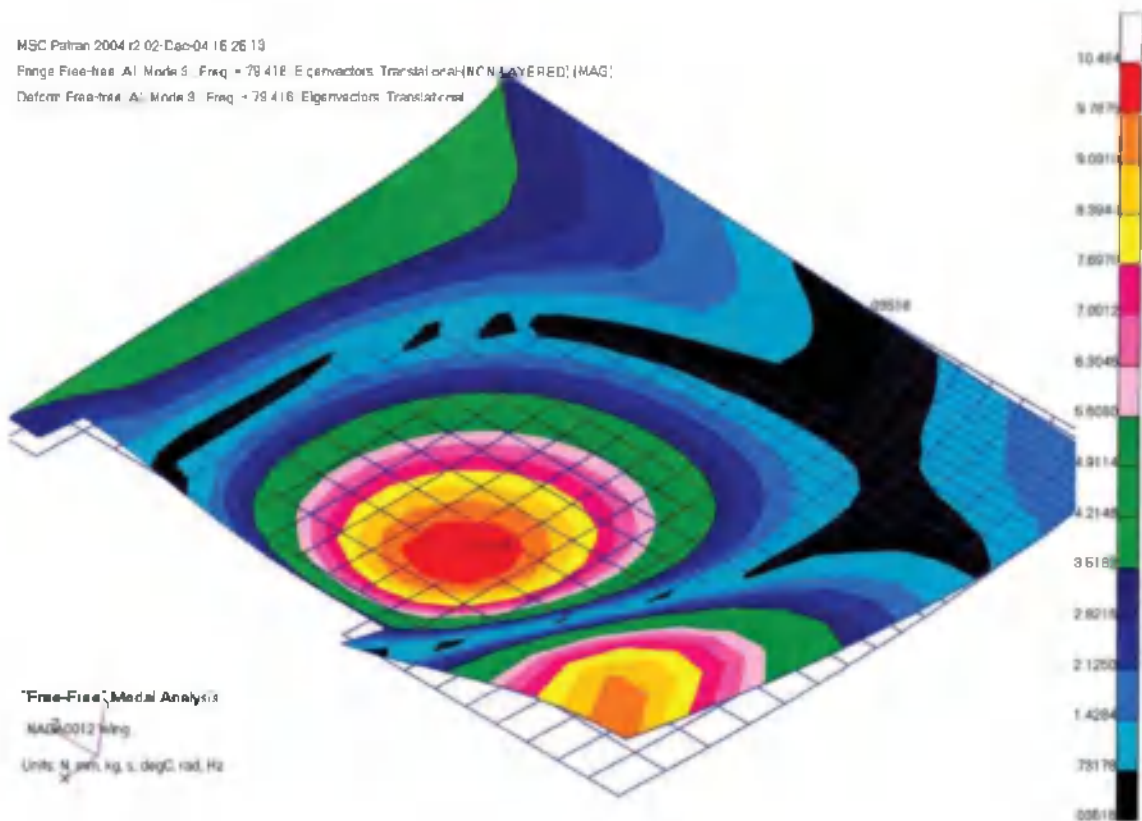


Figure 23: Mode 3 at 79.4Hz

MSC Patran 2004 r2 02-Dec-04 16:28:45
Fringe Free-free A1 Mode 6 Freq = 105.65 Eigenvectors Translational(NON LAYERED) (MAG)
Deform Free-free A1 Mode 6 Freq = 105.65 Eigenvectors Translational

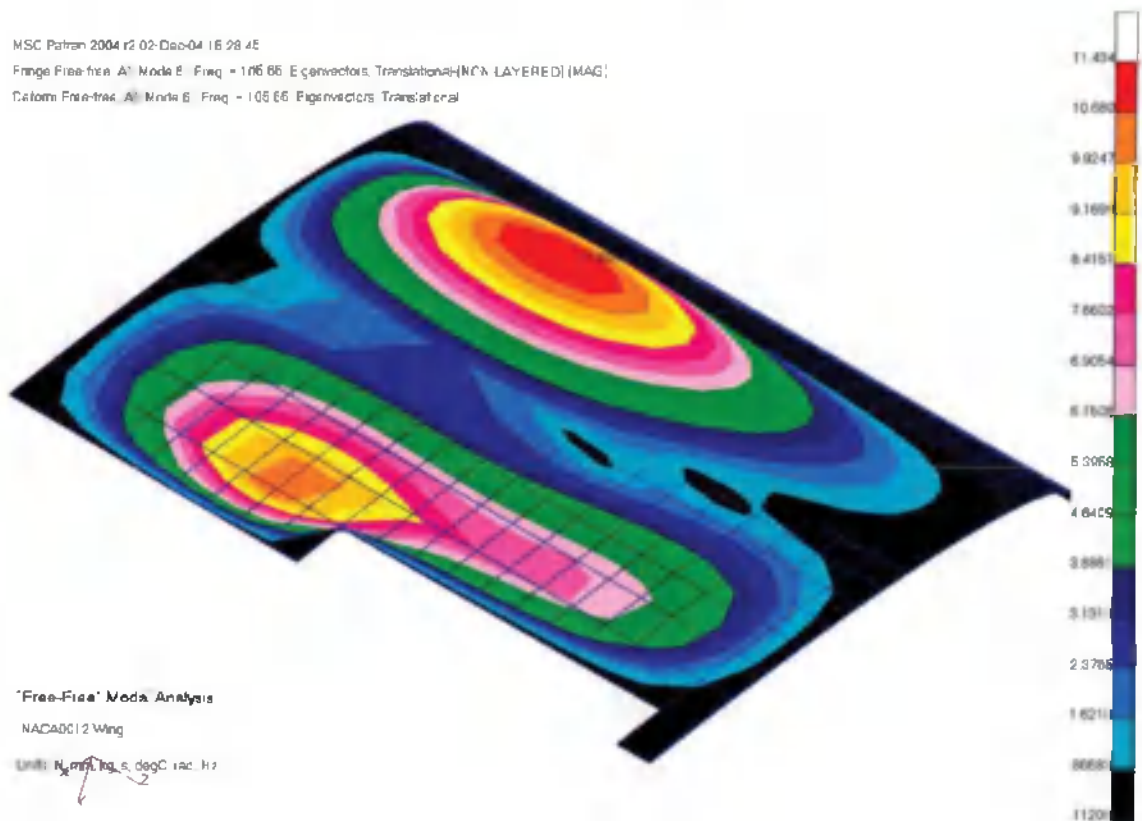


Figure 24: Mode 6 at 105.6Hz

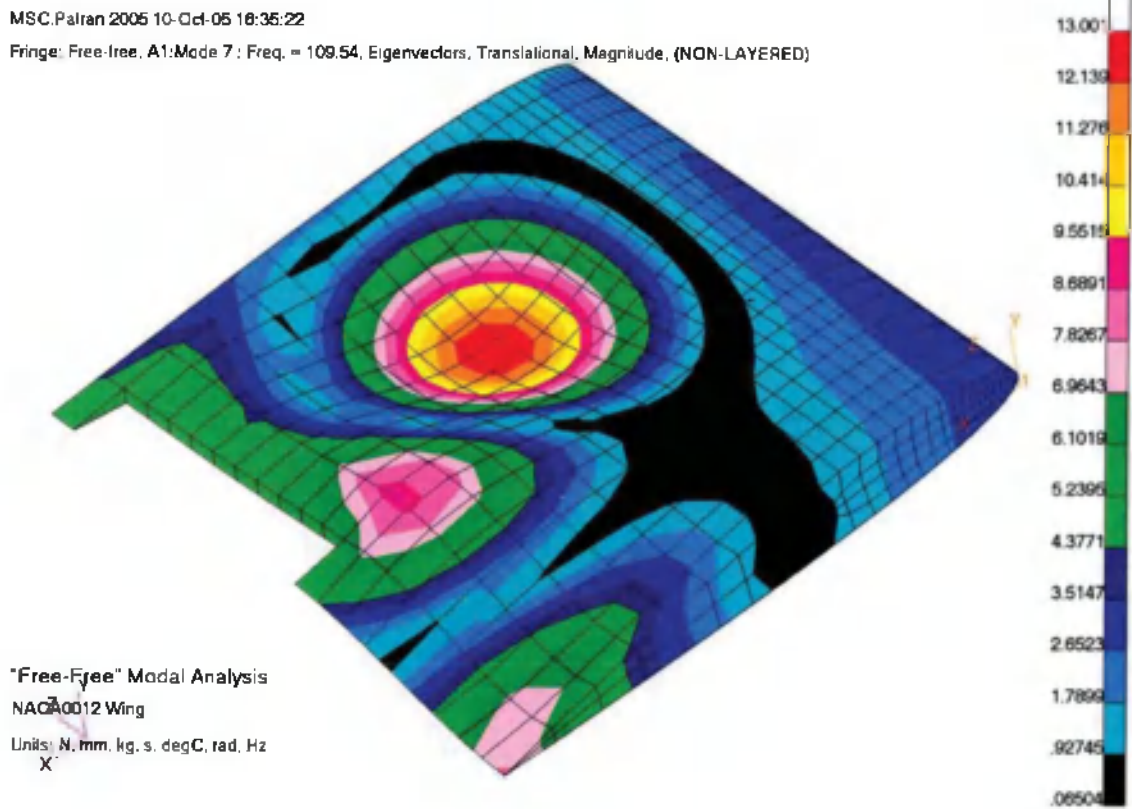


Figure 25: Mode 7 at 109.5Hz

3.3. Simulation of UAV Wing In Icing Conditions Using CFD.

The ideal approach to determining the effect of ice contamination on the UAV wing would be deliberate flights into natural icing conditions. As this method is costly and physical environmental conditions often indeterminate, the more usual approach is placing the wing specimen in an icing wind tunnel. Within the icing tunnel, the experimental variables such as air speed, temperature, angle of attack, water droplet density etc. are more controllable, and hence the effectiveness of any ice protection system deployed.

An icing wind tunnel test programme had been planned for this study, in the Cranfield University Icing Research Wind Tunnel. At the time of writing, construction work had been incomplete⁴¹. The test piece would have been the Phoenix wing shown on Page 97.

In the absence of the physical test opportunity, a FORTRAN based mathematical model (TRAJICE) and two-dimensional computational fluid dynamic (Fluent) exercise was conducted in its place. Verification of the results generated would be compared with ice accretion profiles published in the public domain. Data extracted (e.g. static pressure) from this study, would be used as a load condition in the later mechanical de-icing FEA study.

3.3.1. Input Parameters & Assumptions For Mathematical Models.

The following flight parameters were used in this study were specific points derived from the FAR23/JAR 23 Appendix C flight icing envelope, that any aircraft may encounter.

:

Table 4: Computational Flight & Aerodynamic Conditions

a) Dry Air (Ice Free) Conditions:	
Air speed: 41m/s	Altitude: 1800m
Ambient air temperature: 263.14 °K	Angle of attack: 0°
Ambient air density: 1.032kg/m ³ (ideal gas)	C _p = 1006.43J/kg °K
Ambient air viscosity: 1.734 x 10 ⁻⁵ kg/m s	Thermal conductivity: 0.0242 W/m °K

⁴¹ : E-mail: David Hammond (Cranfield University) to various individuals including the author, Oct04.

Table 4: Computational Flight & Aerodynamic Conditions

b) 15minutes Icing Cloud Conditions (Glaze Ice):

Air speed: 41m/s	Altitude: 1800m
Ambient air temperature: 268.14 °K	Angle of attack: 0°
Ambient air density: 1.032kg/m ³ (ideal gas)	Cp = 1006.43J/kg °K
Ambient air viscosity: 1.734 x 10 ⁻⁵ kg/m s	Thermal conductivity: 0.0242 W/m °K
Liquid water content: 1g/m ³	Median Water Droplet: 50µm
Collection efficiency: 0.9	Wetness factor: 0.7

b) 10minutes Icing Cloud Conditions (Rime Ice):

Air speed: 41m/s	Altitude: 1800m
Ambient air temperature: 263.14 °K	Angle of attack: 0°
Ambient air density: 1.032kg/m ³ (ideal gas)	Cp = 1006.43J/kg °K
Ambient air viscosity: 1.734 x 10 ⁻⁵ kg/m s	Thermal conductivity: 0.0242 W/m °K
Liquid water content: 1g/m ³	Median Water Droplet: 50µm
Collection efficiency: 0.9	Wetness factor: 0.7

3.3.2. TRAJICE (FORTRAN Code) Ice Shape Predictions.

TRAJICE is a FORTRAN program developed by UK Defence Research Agency⁴². The program predicts the two-dimensional ice accretion profile, for a given aerofoil geometry. The program requests a series of environmental parameters such as those listed in Table 4. Additional information sought includes the icing encounter duration, calculation time steps, water droplet collection efficiency, convective heat transfer coefficient etc.

⁴² : Now called "QinetiQ Plc", Farnborough, Hampshire GU14 6TD.

For the ice accretion calculation⁴³, the aerofoil surface is divided into a number of equi-spaced elements at which the surface temperature and water freezing fraction is evaluated using the standard heat balance method (a 1st order study is displayed in Appendix A1 and A3). The local ice growth rate is calculated from the freezing fraction and an ice profile is obtained by extrapolating the growth rates to the time spent in icing.

An assumption within this program is that the ice growth is normal to the aerofoil surface for glaze icing (i.e. ambient temperatures between 0°C and -7°C) as the water droplets will flow before freezing. Ice growth is assumed to parallel to the aerofoil for rime type icing conditions (i.e. temperatures lower than -10°C). The justification for this that in rime ice conditions, water droplets freeze on impact.

Usefully a standard program demonstration and tutorial run includes glaze and rime icing runs for a NACA0012 aerofoil. Alteration of a few input parameters to reflect those listed above in Table 4, time steps of 1 minute, and modified inertia parameter $K_0=10.48$ (as determined in Appendix A1) were used. An ice profile geometry file was out put by the TRAJICE code, consisting of x,y coordinates of the ice and aerofoil. This geometric point data was imported into Unigraphics CAD software for conversion into NURBS curves for display.

For glaze ice condition (-5°C), the 15minute ice accretion profile predicted by TRAJICE is shown in Figure 26:

⁴³ : "TRAJICE 2 - A combined water droplet trajectory & ice accretion prediction program for aerofoils", Gent R.W, Royal Aerospace Establishment Report TR90054 (Nov90).



Figure 26: Glaze Ice Profile Predicted By TRAJICE software.

For rime ice condition (-10°C), the 10-minute ice accretion profile predicted by TRAJICE is shown in Figure 27



Figure 27: Rime Ice Shape Predicted By TRAJICE software

The icing exposure time was based FAA Document AC29.877 advisory statements for inadvertent flight into icing conditions, where the pilot has delayed activation of the ice protection system by 5 minutes.

3.3.3. Ice Shapes Observed In The Icing Wind Tunnel Trials.

The results obtained were compared with those observed in the literature.

Examination of Chapter 1-Figure 5 and icing trial papers by Gent⁴³, Al-Khalil & Irani⁴⁴ amongst others report glaze and rime ice shape profiles in comparable physical icing conditions can now be made. The physical ice shapes were developed from observations in icing wind tunnel trials.

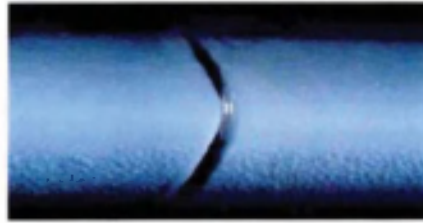


Figure 28: Rime Ice Accretions Observed In An Icing Wind Tunnel at 54m/s, Liquid Water Content 0.7g/m^3 , Ambient Temperature -11°C .

TRAJICE predicted rime ice shape shown in Figure 27 is similar to that recorded for rime ice conditions in the icing wind tunnel trials in Figure 28.

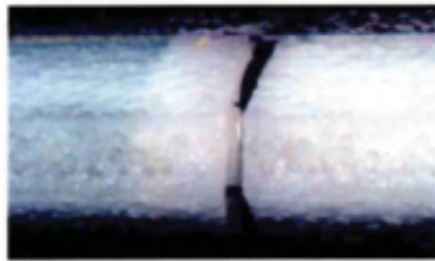


Figure 29: Glaze Ice Condition Observed in Icing Wind Tunnel⁴⁴ at 54m/s, Liquid Water Content 0.7g/m^3 , Ambient Temperature -5°C .

Similarly, TRAJICE predicted glaze ice shape shown in Figure 26 is similar to that recorded for glaze ice conditions in the icing wind tunnel trials in Figure 29. This gives confidence in the further work undertaken using these profiles.

⁴⁴: "Icing Simulation and Testing at the Cox Icing Wind Tunnel", Al-Khalil K. & Irani E., Cox & Co, New York. NASA/TM-2003-212395 (Jan03)

3.4. Predicted Aerodynamic Performance Of An Ice Contaminated UAV Wing.

The ice profiles were imported into Fluent and a mesh developed around the profile. The objective was to determine the pressure profile on the UAV wing, and thence the effectiveness of the proposed de-icing system in breaking the ice-surface bond, and overcoming the aerodynamic pressure. Conditions could exist, whereby despite the breaking of the ice-surface bond, ice was still retained on the wing by aerodynamic pressure.

3.4.1. FLUENT CFD Model Set-up.

Computational Fluid Dynamics (CFD) is a computational technology that enables the study the dynamics of things that flow. Using CFD, a computational model representing a system is built and a fluid flow is applied to this virtual prototype. The software outputs a prediction of the dynamics of the fluid. CFD not only predicts fluid flow behaviour, but also the transfer of heat, mass, phase change (such as in freezing or boiling) etc. Fluent is a commercial CFD computer program uses finite element mesh method to simulate the fluid flow. Here Fluent is used to determine the two dimensional pressure profile across a wing section with and without ice contamination.

A 2D mid-span wing profile was extracted from the Unigraphics CAD model. A mesh model was developed within GAMBIT (pre-processor software for Fluent CFD code). The mesh grid generated was imported into Fluent v6.1 CFD code. The aircraft flight and atmospheric conditions parameters were applied to model. This research paper is fortunate in that a Fluent tutorial exists for undertaking aerodynamic studies with a NACA0012 wing profile. Thus, it was a simple case of adapting the tutorial exercise file and geometry to the purposes of this UAV de-icing study. Thus, the Trajice predicted ice profiles were easily imported from Unigraphics CAD system and a mesh generated.

3.4.2. CFD Model Input Parameters & Assumptions.

The objective of the study was to examine the aerodynamics of a clean (i.e. ice free) wing and an ice contaminated aerofoils following 15minutes and 10minutes exposure to Table 4 icing conditions.

As stated earlier, ice contaminated surface profile was developed within a FORTRAN 2D ice accretion prediction code. The accreted ice shapes predicted by the code were imported into GAMBIT and a mesh suitable for CFD was generated. The ice accretion shape profiles were also imported into MSC. PATRAN for mesh generation for the mechanical de-icing FEA study.

3.4.3. Performance of Ice Free Wing Section.

Fluent CFD code model for a clean or ice free NACA0012 wing is shown in Figure 30.

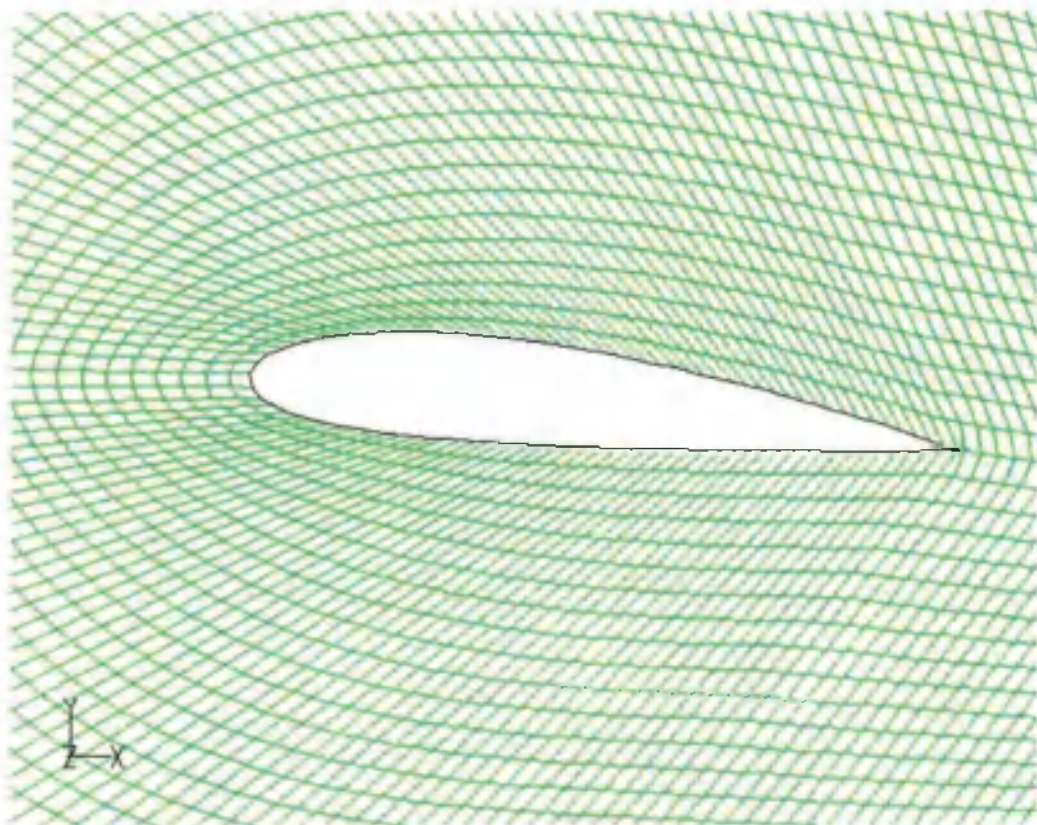


Figure 30: Fluent Mesh Model Grid Set-up

Clean (ICE Free) Mesh model properties:

No. of cells: 81230	No. of nodes: 81302
No. of faces: 162532	No. of partitions: 1
Solver conditions: Segregated, Implicit, Steady state.	
Velocity formulation: Absolute	Gradient: cell based
Pressure-Velocity coupling: simple	

The input parameters declared in Table 4 were used in this CFD model. The velocity profile predicted by the CFD code is displayed in Figure 31.

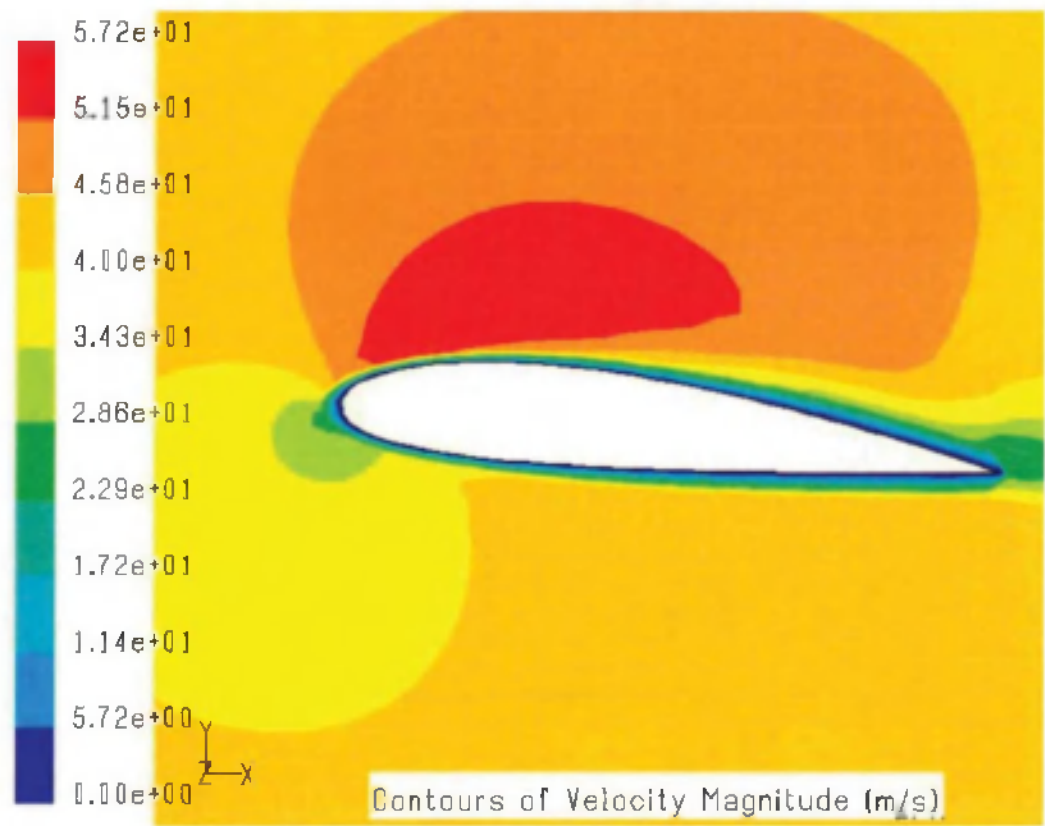


Figure 31: Velocity Profile At 0° Angle of Attack

Of real interest to the de-icing study, is the pressure profile across the wing, which contributes to loading of the wing. This can be observed as a contour plot in Figure 32 or graphically as in Figure 33.. It can be seen that the Fluent CFD static pressure predictions are in line with hand calculations based expectations shown in Appendix A2.

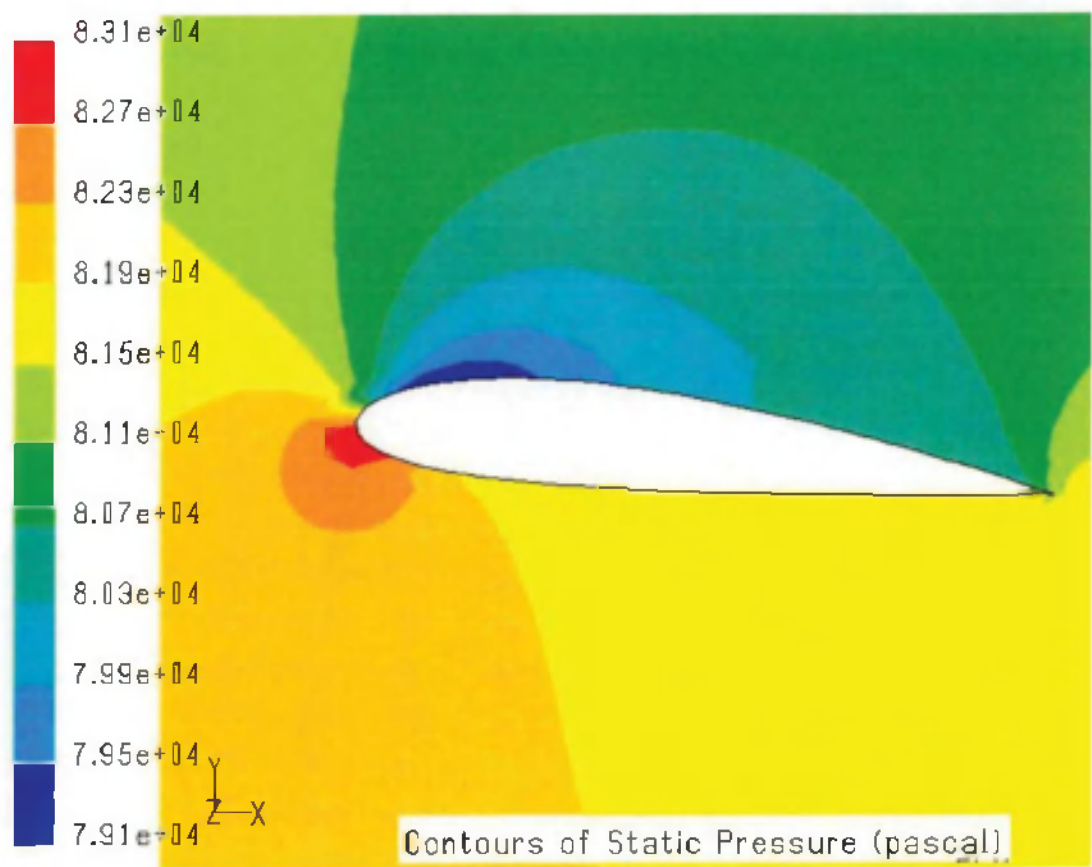


Figure 32: Clean (Ice Free) Wing - Static Pressure Contour

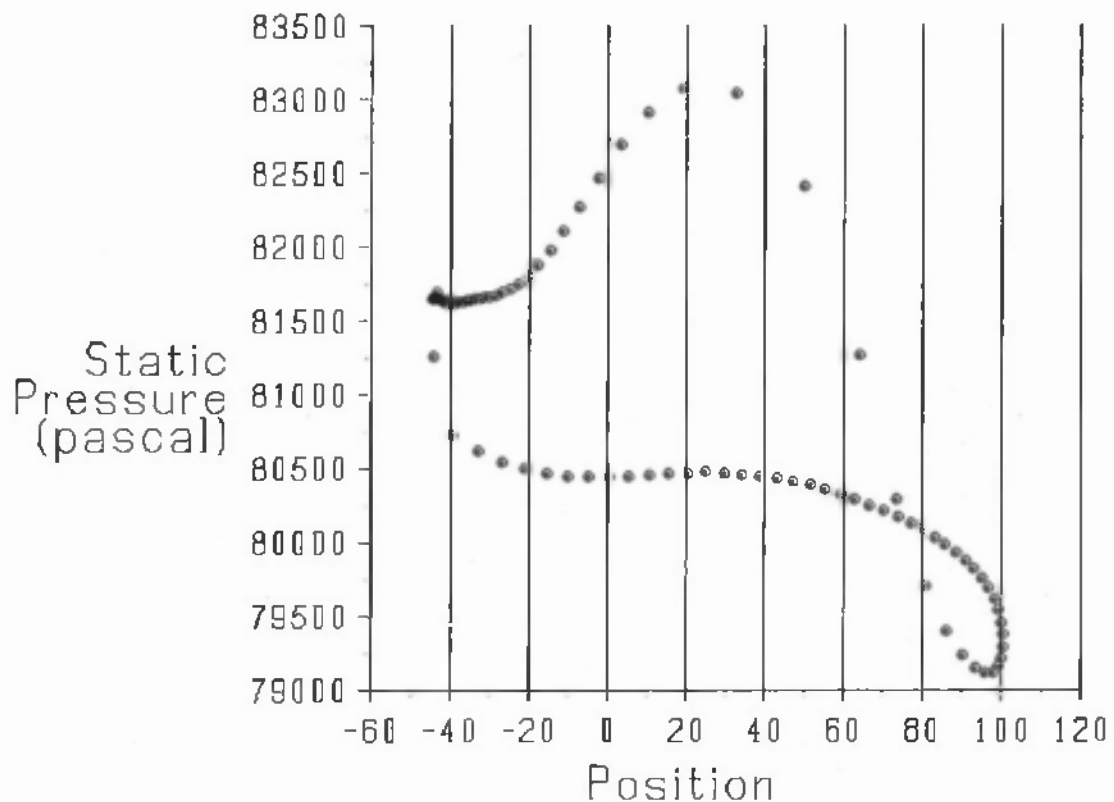


Figure 33: Graph of Static Pressure V Chord Position (%) Across the Clean (Ice Free) Wing Profile.

Figure 33 suggests the maximum and minimum pressure is observed just aft of the leading edge. The drag and lift coefficients reported by the clean (ice free) wing Fluent study following 30 iterations are displayed in Figure 34. The figures predicted by Fluent are also in line with NACA Report No.647⁴⁵ where wind tunnel test data suggests drag and lift coefficients for NACA0012 wing of comparable physical parameters were obtained.

⁴⁵ : NACA Report 647 "Test of NACA009, 0012 & 0018 Airfoils", H.J Goett & W K Bullivant

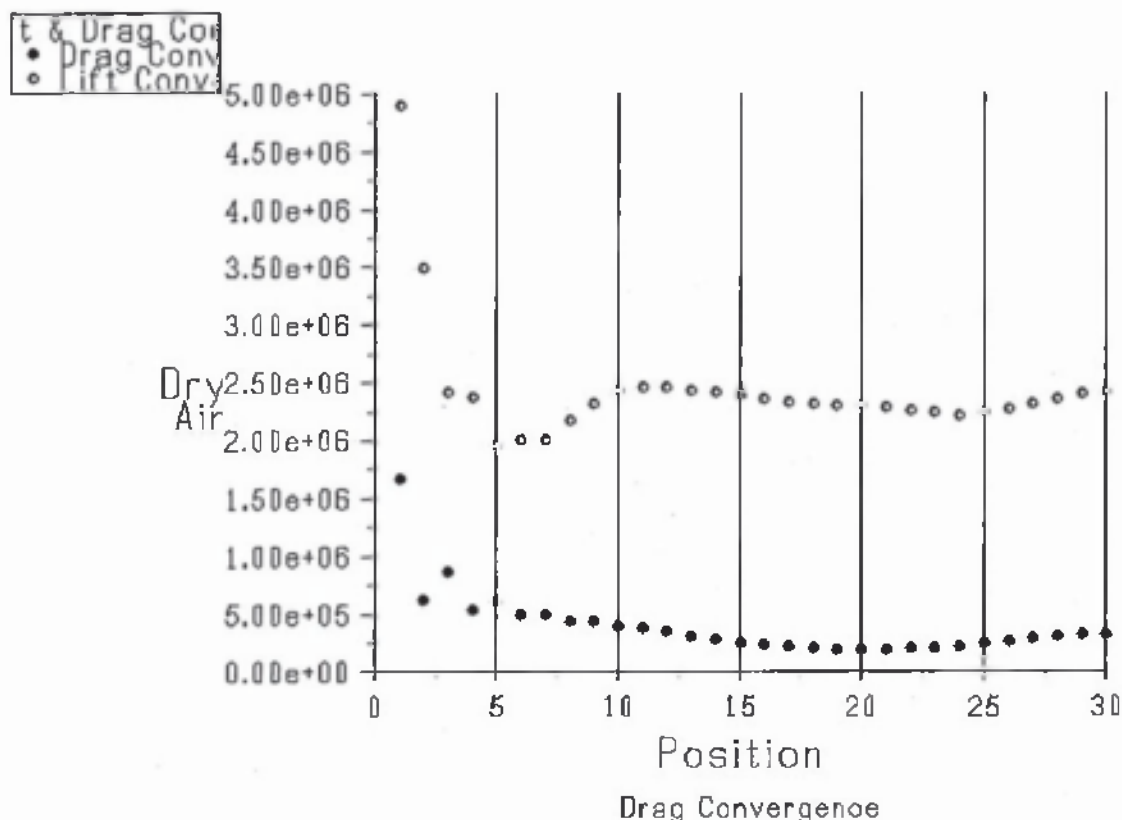


Figure 34: Clean (Ice Free) Wing Lift & Drag Coefficient Predictions By Fluent

This gives further confidence the Fluent CFD pressure profiles for the ice contaminated wings would be realistic.

3.4.4. Performance of Ice Contaminated Wing Section

The ice contaminated surface profiles were meshed and a solution derived for the glaze and rime ice conditions respectively.

The ice profile captured from TRAJICE for the rime ice condition is described in Figure 35.

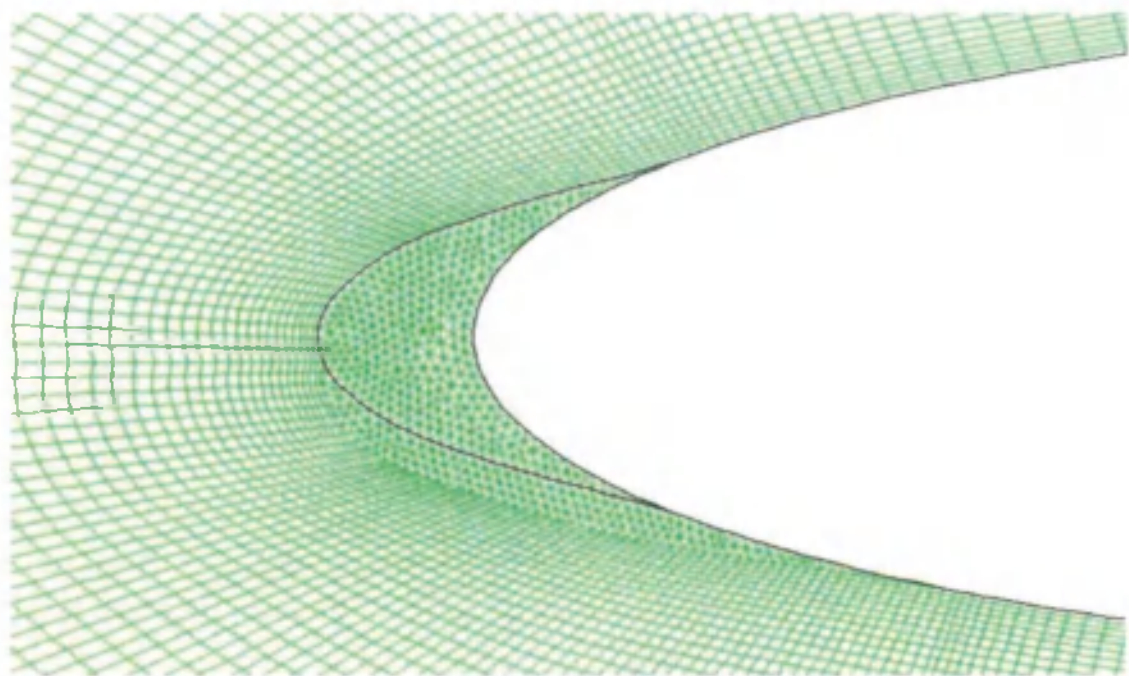


Figure 35: Fluent Rime Ice Mesh Model.

The model variables used in addition to that Table 4 declared in were:

Rime Ice Mesh Model Properties:	
No. of cells: 81230	No. of nodes: 81375
No. of faces: 162606	No. of partitions: 1
Solver conditions: Segregated, Implicit, Steady state.	
Velocity formulation: Absolute	Gradient: cell based
Pressure-Velocity coupling: simple	

The effect of the rime ice shape on the static pressure on the wing was calculated and displayed in Figure 36.

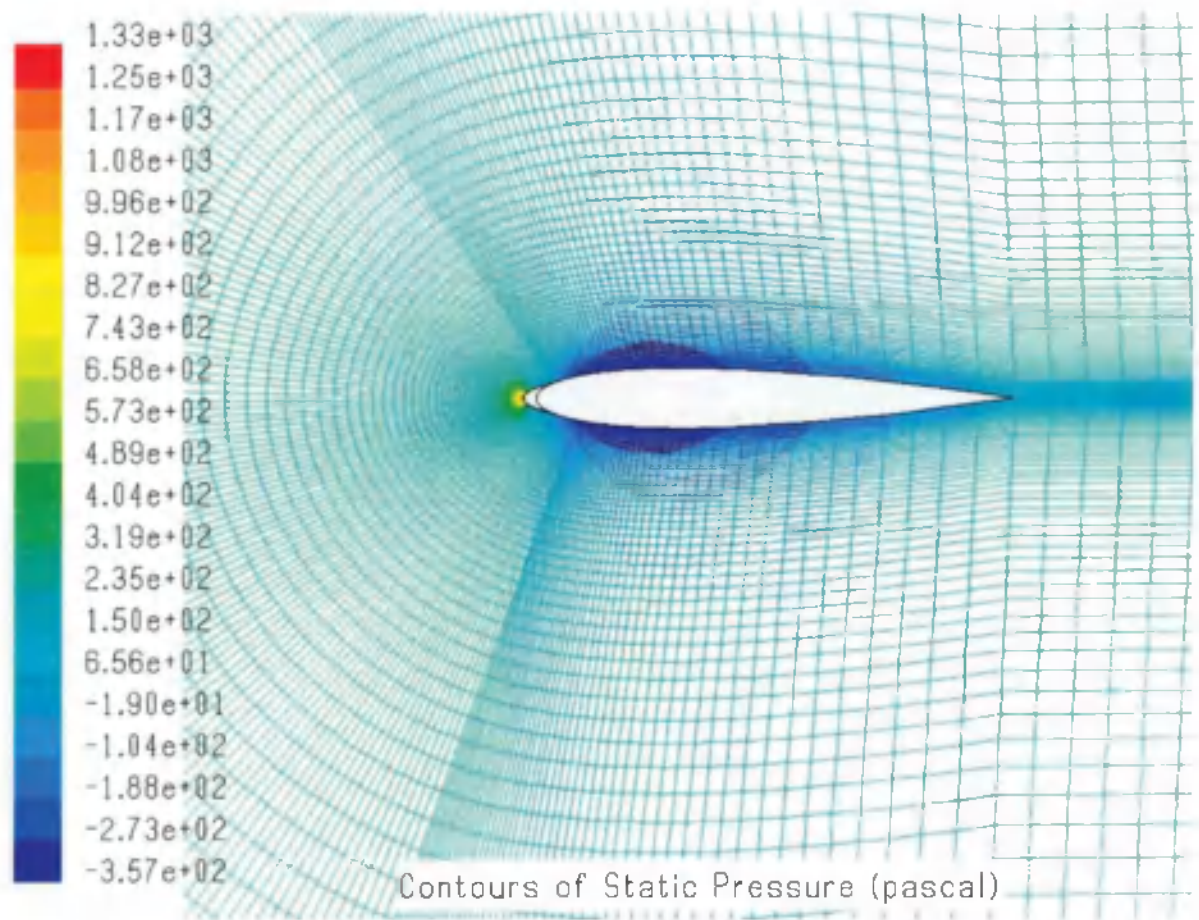


Figure 36: Static Pressure For Rime Ice Condition As Predicted By Fluent

Figure 36 and Figure 38 show the accreted rime ice generates a large stagnation pressure and corresponding stationary air at the tip of the rime ice accretion. The changes are significant when compared to the clean wing results, and demonstrate the parasitic nature of the ice accretion.

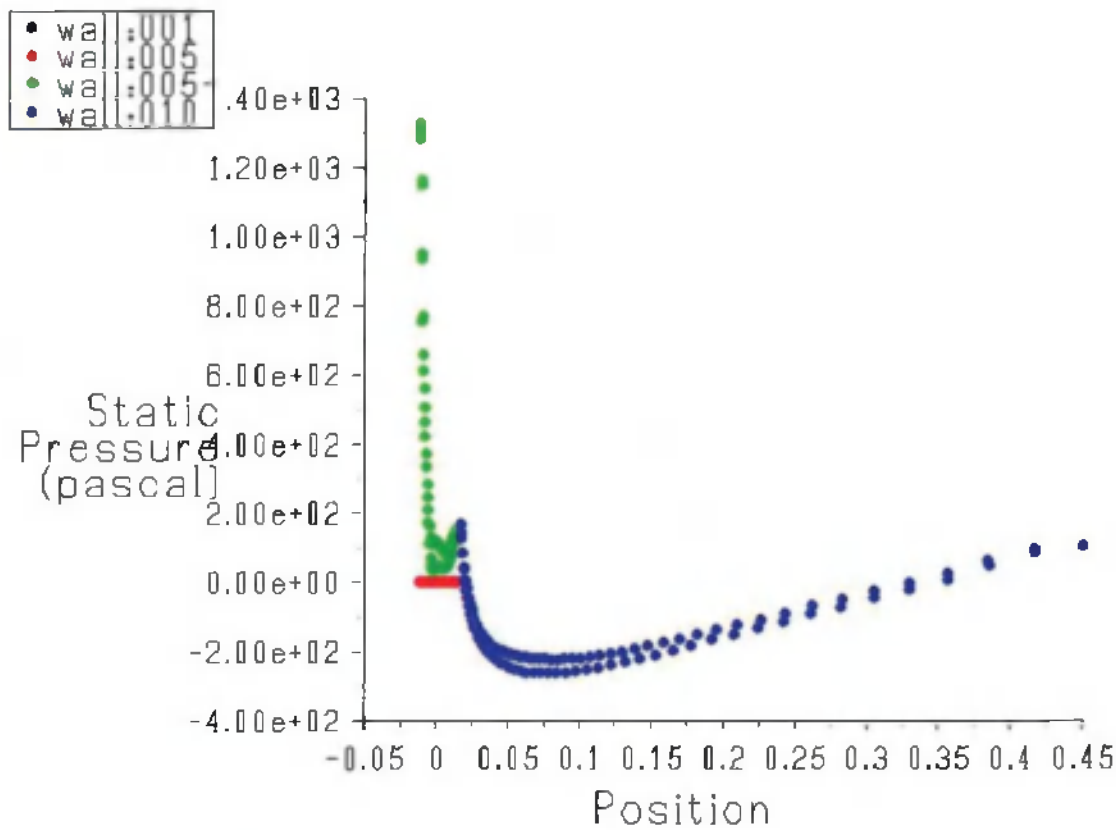


Figure 37: Static Pressure As A Function of Chord Position (%) As Predicted By Fluent

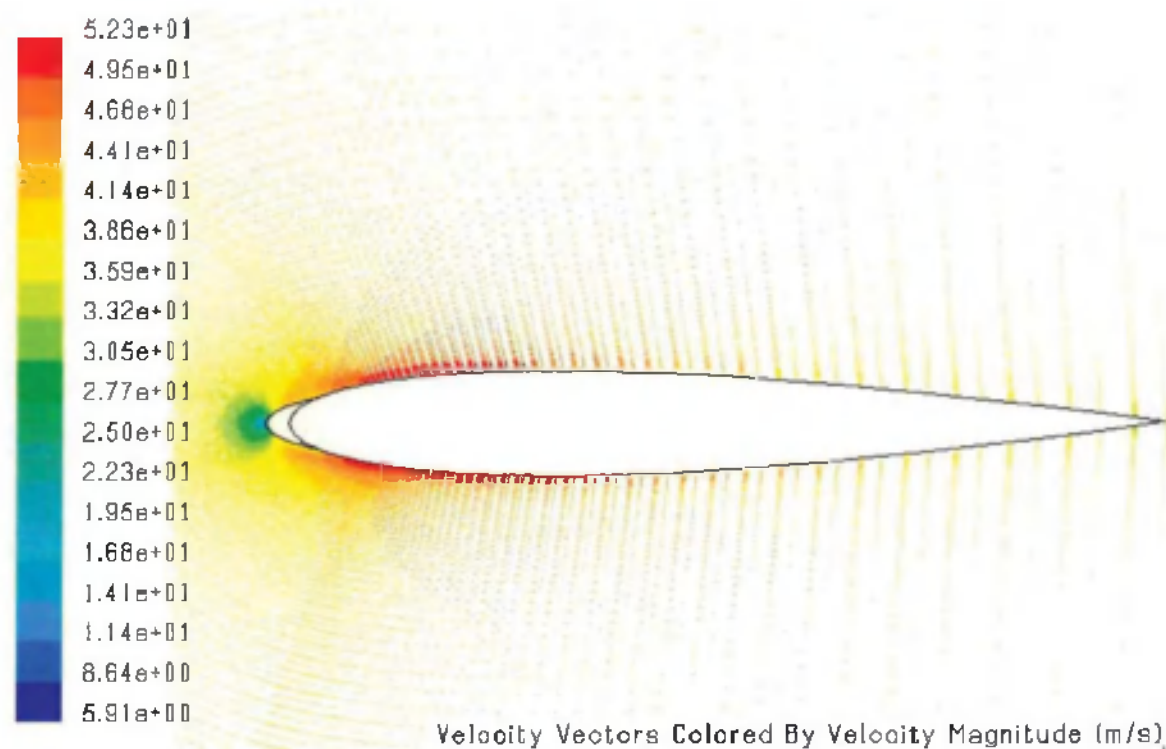


Figure 38: Velocity Profile For Rime Ice Condition As Predicted By Fluent

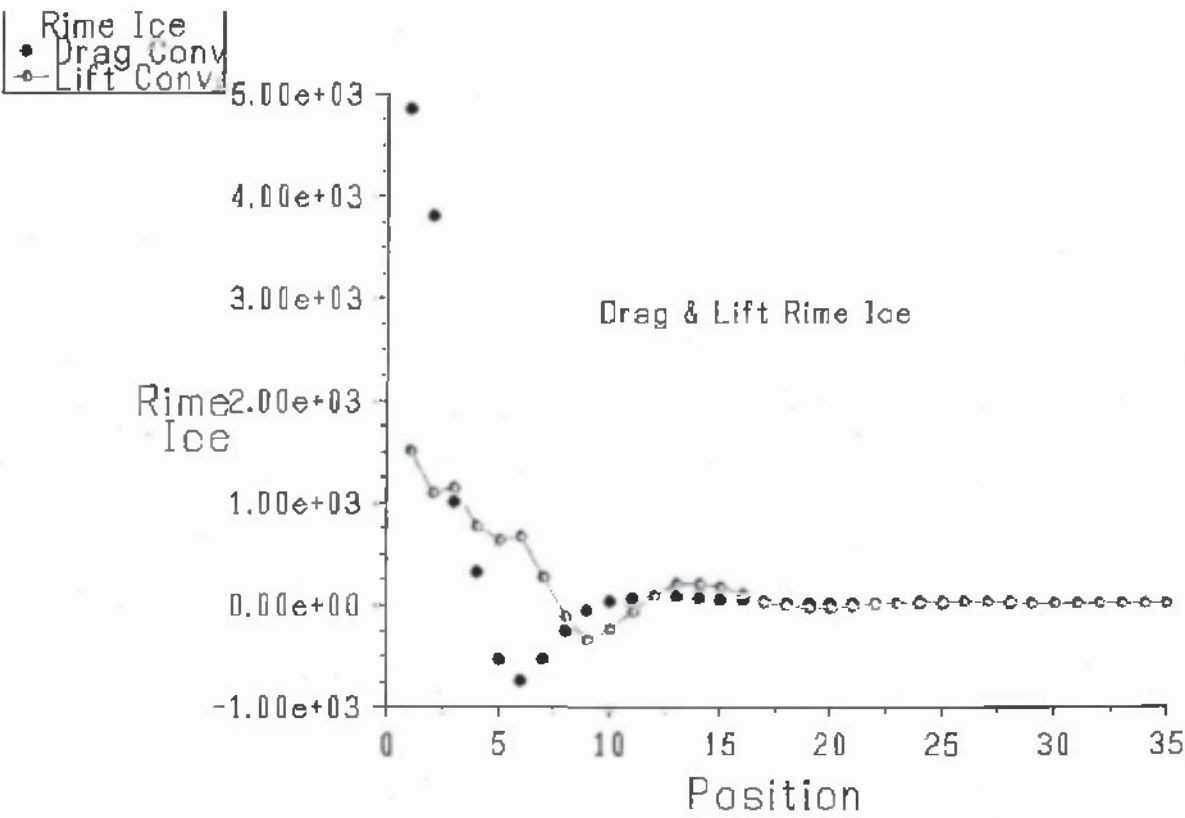


Figure 39: Lift and Drag Coefficients For Rime Ice Predicted By Fluent As A Function of Chord Position (%)

The ice profile captured from TRAJICE for the glaze ice condition is described in Figure 40. The model variables used in addition to that Table 4 declared in were:

Glaze Ice Mesh Model Properties:	
No. of cells: 15000	No. of nodes: 15320
No. of faces: 303206	No. of partitions: 1
Solver conditions: Segregated, Implicit, Steady state.	
Velocity formulation: Absolute	Gradient: cell based
Pressure-Velocity coupling: simple	

The effect of the glaze ice shape on the static pressure on the wing was calculated and displayed in Figure 41 and Figure 42. The changes are considerable when compared to the clean wing and rime ice results, and demonstrates the parasitic nature of the ice accretion.

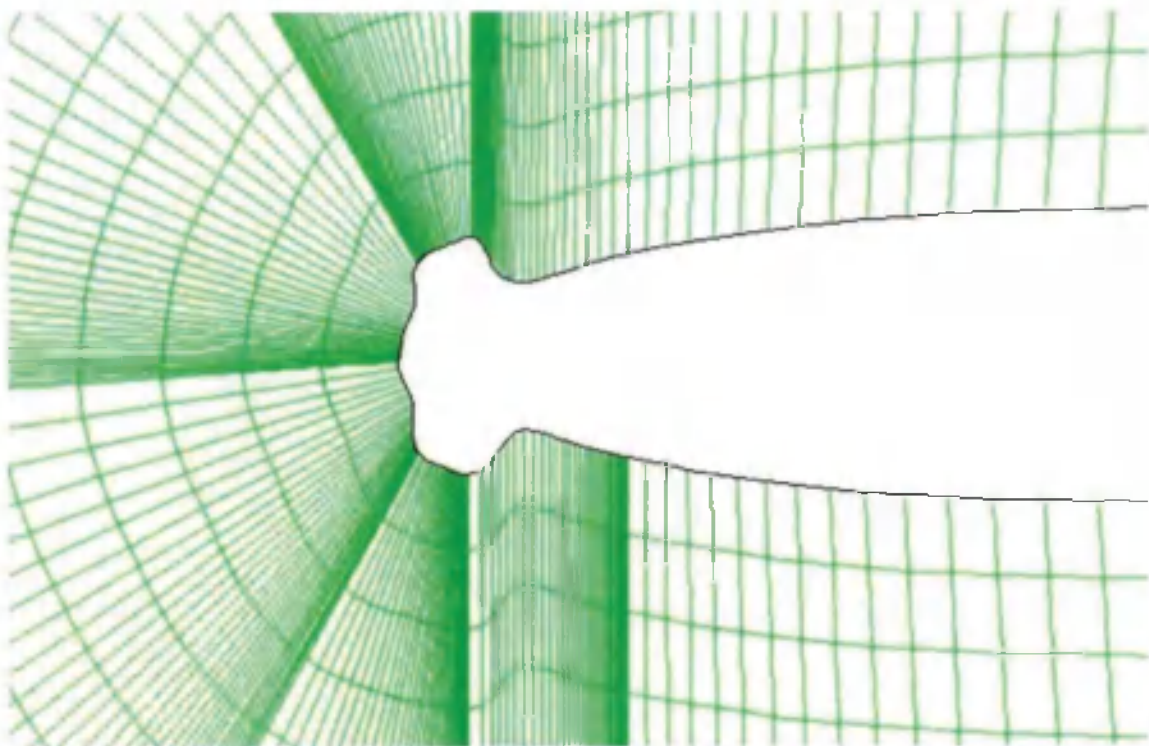


Figure 40: Fluent Glaze Ice Mesh Model

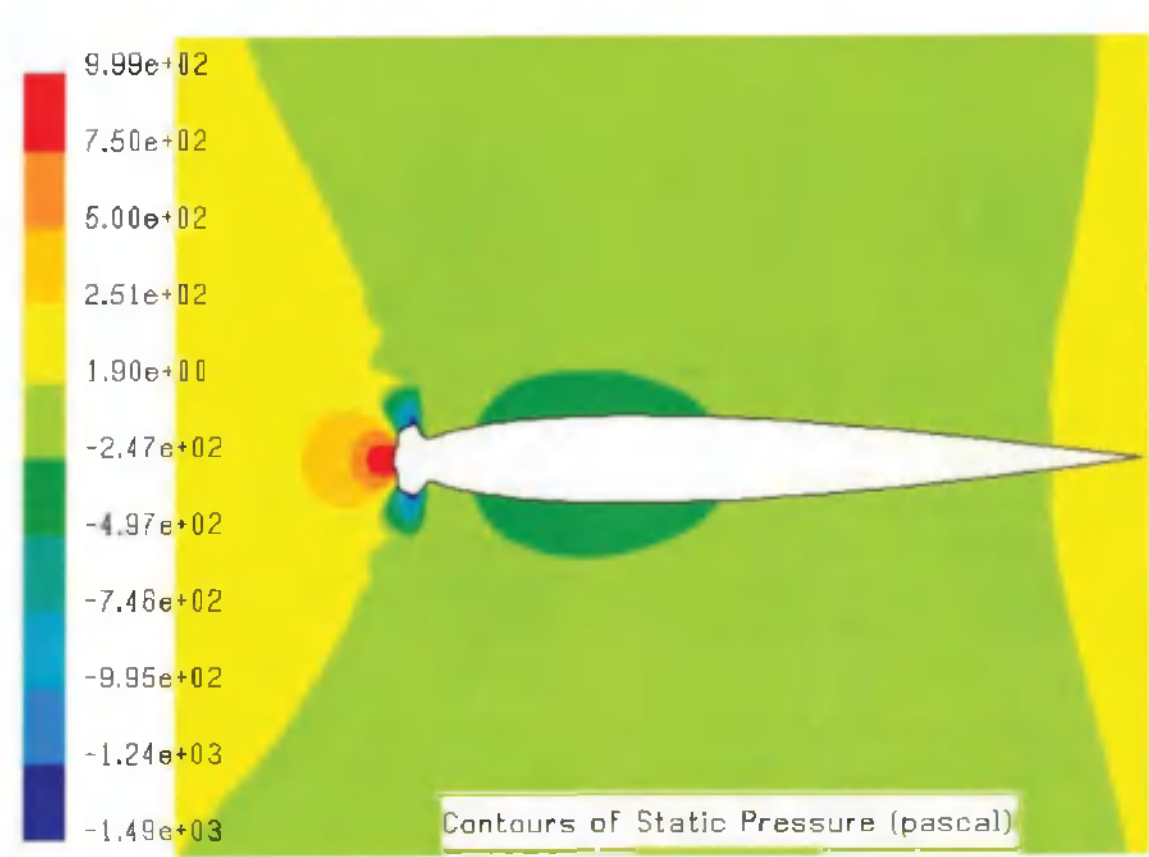


Figure 41: Glaze Ice Static Pressure Prediction By Fluent

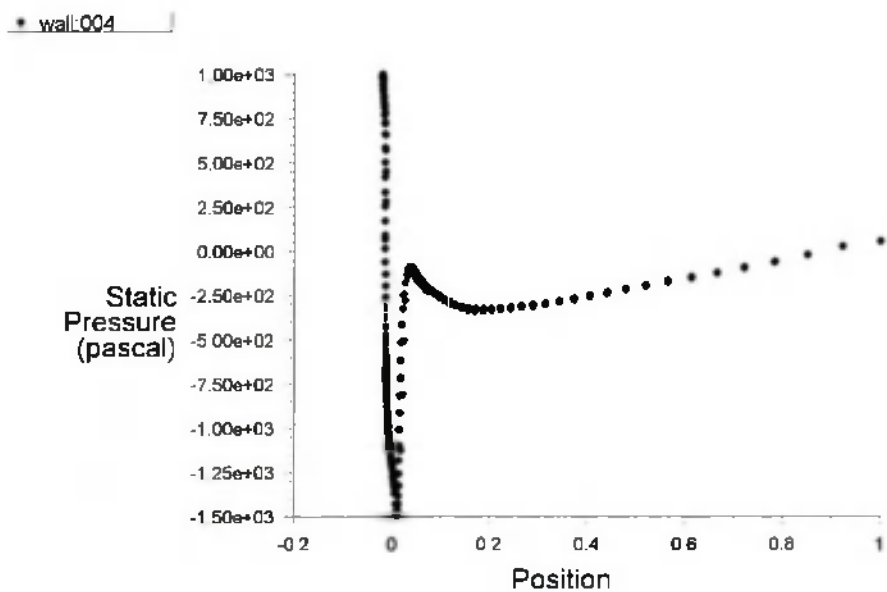


Figure 42: Static Pressure As A Function of Aerofoil Ordinates

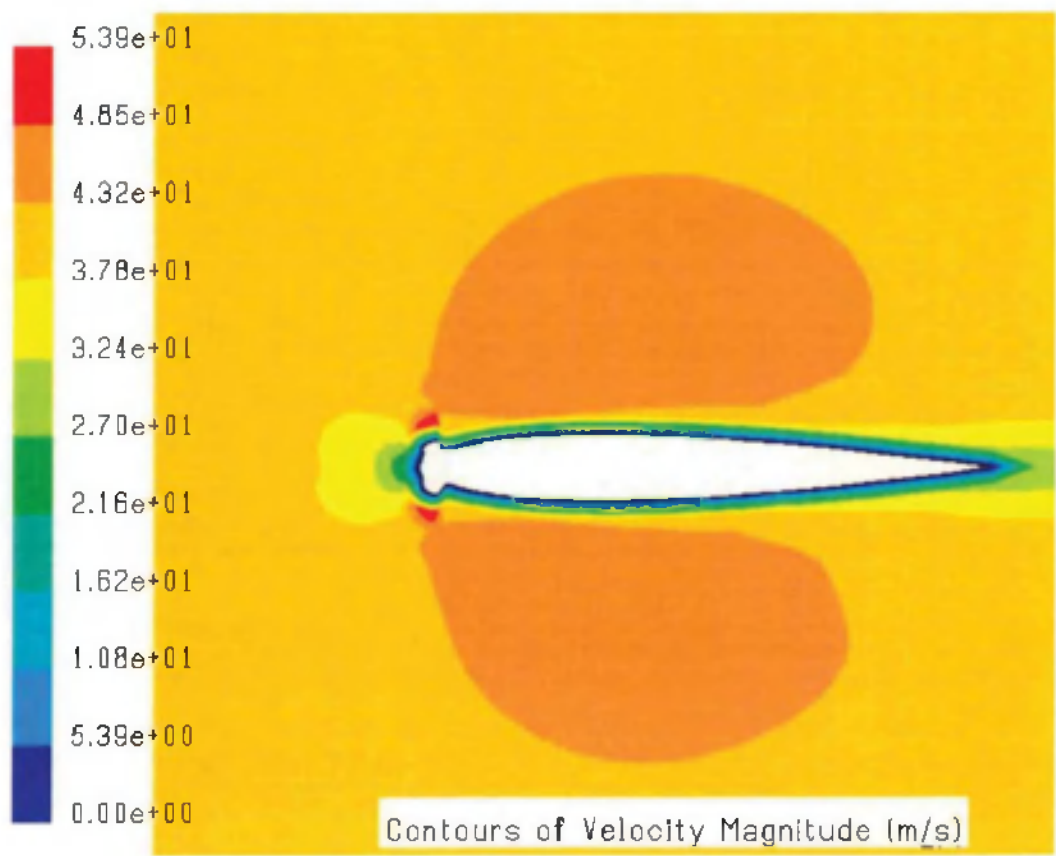


Figure 43: Velocity Profile For Glaze Ice As Predicted By Fluent

3.5. Wing Ice Profile: Test Data V TRAJICE-FLUENT CFD Predictions.

The ice shape profiles predicted by TRAJICE are in line with test data and user manual accompanying the software. The ice profiles predicted also correspond to ice shapes described in Chapter 1-Figure 5 of this report.

3.6. Computational Simulation of Novel Ice Protection System On An Ice Contaminated UAV Wing.

To simulate the performance of the proposed ice protection system on an ice wing, a FEA study was conducted using MSC.PatranV2004 and MSC.NastranV2004 software.

3.6.1. FEA Model Description & Set-up.

As discussed in section 3.4.1 above, the two dimensional (2D) ice shape profiles developed in FORTRAN based TRAJICE predictions were converted into NURBS curves within Unigraphics CAD software. These 2D profiles were imported into MSC.Patran and 2Dsurface ice profile created. The surface profile was then meshed using Triangular elements (Nastran element reference: TRIA3). Spring elements were then used to "bond" the triangular elements to the wing surface FEA model. The triangular elements were then extruded along the span of the wing model. Spring elements representing the ice-surface adhesion bond were used. The determination of the spring constants for shear and peel were determined as displayed in Appendix A6.

3.6.2. FEA Model Input Parameters & Assumptions.

The following parameters were used within the "Rime" Ice FEA model. It should be noted although the geometry of the clean wing was unchanged from previous study in 3.3.2, the mesh model had been refined to capture the rime ice detail developed within the Fluent CFD study. Thus, the number of nodes and elements within the wing had increased. The ice was represented as solid elements.

Rime ICE FEA Model Parameters	Number
<u>Wing</u>	
No. of Nodes (Including spar)	3590
No. of Elements (NASTRAN ref: Shell element topology Quad4/ Tria3)	3599
No. of Spring Elements (representing ice-surface band)/ 3 per surface node for each translational degree of freedom	1517
<u>Rime Ice Body</u>	
No. of Nodes	1440
No. of Elements (NASTRAN ref: Solid element topology Wedge6)	1839
<u>Glaze Ice Body</u>	
No. of Nodes	980
No. of Elements (NASTRAN ref: Solid element topology Wedge6)	521

The engineering material properties used for the wing skin and spar were displayed in Table 2 above.

Ice mechanical properties used were derived from publications^{24, 27} and Appendix A6:

Ice Mechanical Property	Value
Shear Modulus	9000MPa
Poisson's Ratio	0.33

3.6.2. FEA Model Input Parameters & Assumptions.

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No. of Nodes	980
No. of Elements (NASTRAN ref: Solid element topology Wedge6)	521

The engineering material properties used for the wing skin and spar were displayed in Table 2 above.

Ice mechanical properties used were derived from publications^{24, 27} and Appendix A6 (It should be noted that the ice and the ice-surface mechanical properties will vary depending on environmental exposure. Thus the values used here are considered "typical"):

Ice Mechanical Property	Value
Shear Modulus	9000MPa

Ice Mechanical Property	Value
Poisson's Ratio	0.33
Density	920kg/m ³
Shear Strength	0.7MPa
Shear Strain (estimated)	7.78E-05
Ice Peel Strength (calculated)	45 x 10 ⁶ N/mm

3.6.3. FEA Clean Wing "Baseline" Study

To determine the stresses in the wing in it's "base line" condition i.e. due to it's own natural frequency, a modal analysis was carried out with the FEA wing clamped at it's root. The first 3 mode shapes and frequencies predicted by MSC. NASTRAN (analysis code SOL 103) are shown below:

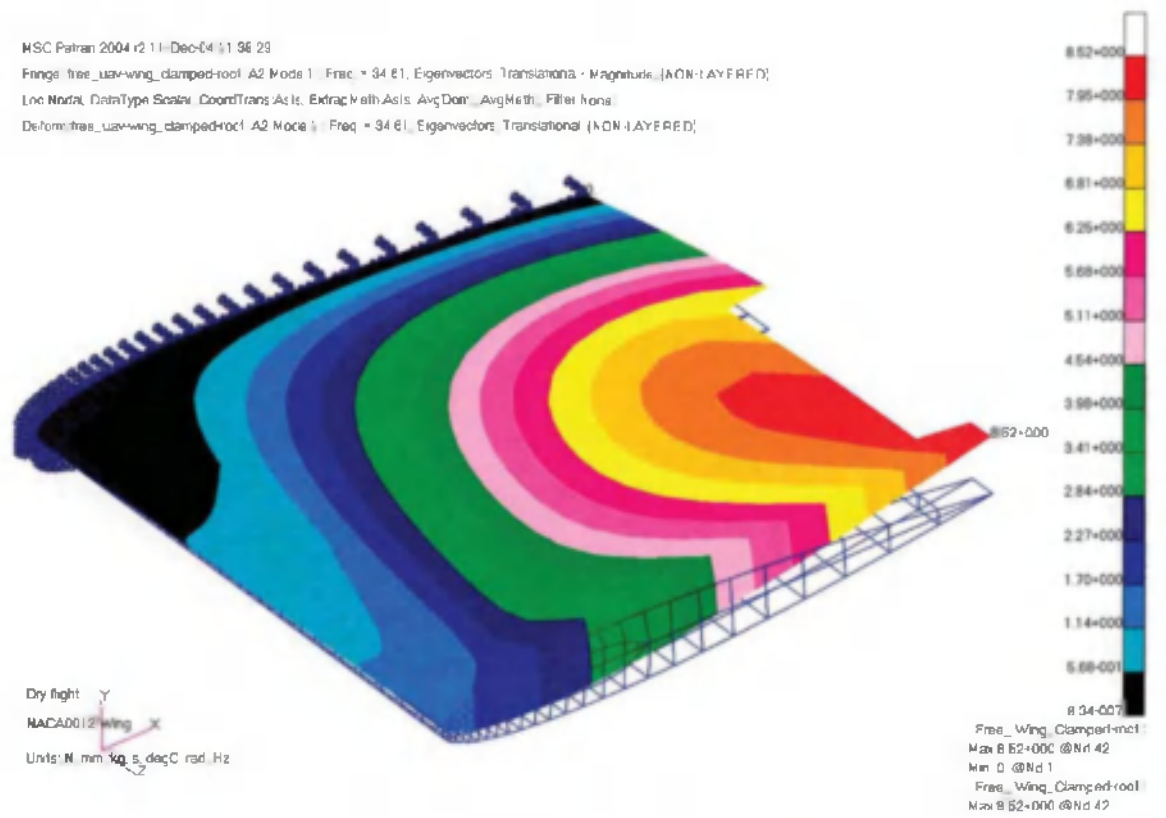


Figure 44: First Mode (34.6Hz) for Wing Clamped at The Root

Ice Mechanical Property	Value
Density	920kg/m ³
Shear Strength	0.7MPa
Shear Strain (estimated)	7.78E-05
Ice Peel Strength (calculated)	45 x 10 ⁶ N/mm

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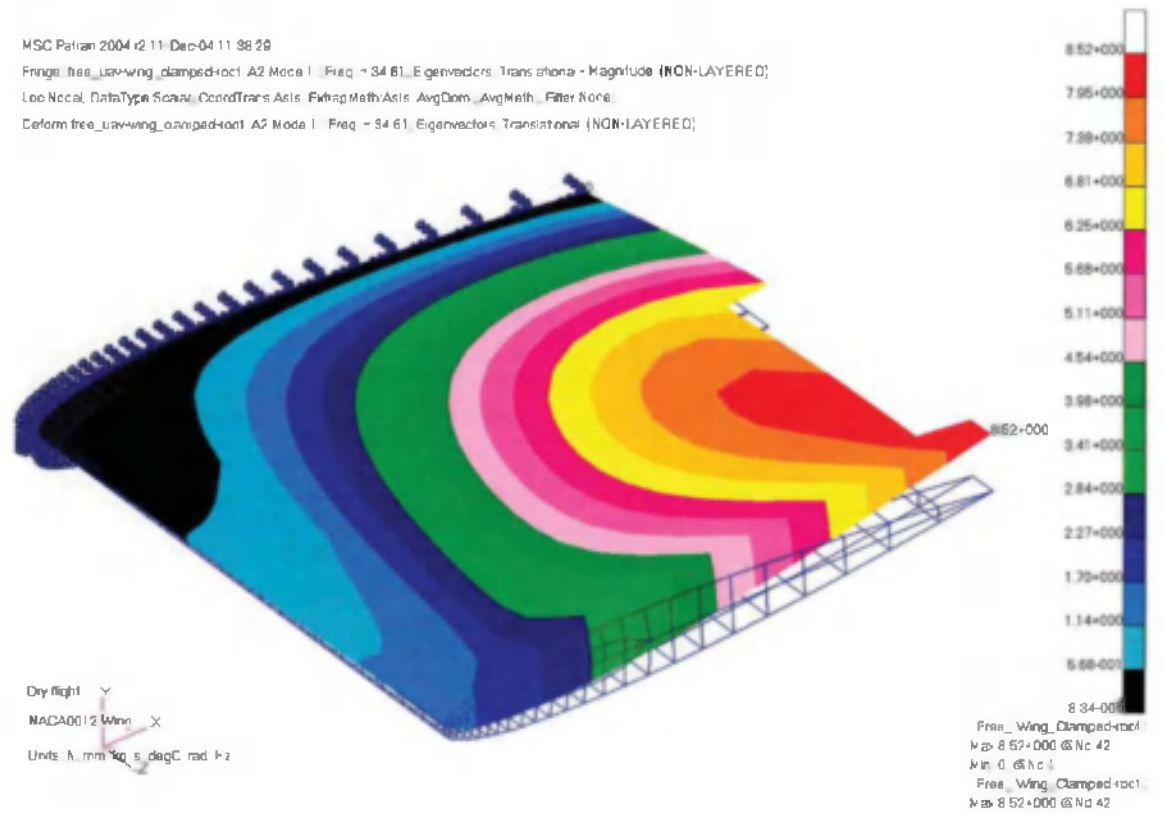


Figure 44: First Mode (34.6Hz) for Wing Clamped at The Root

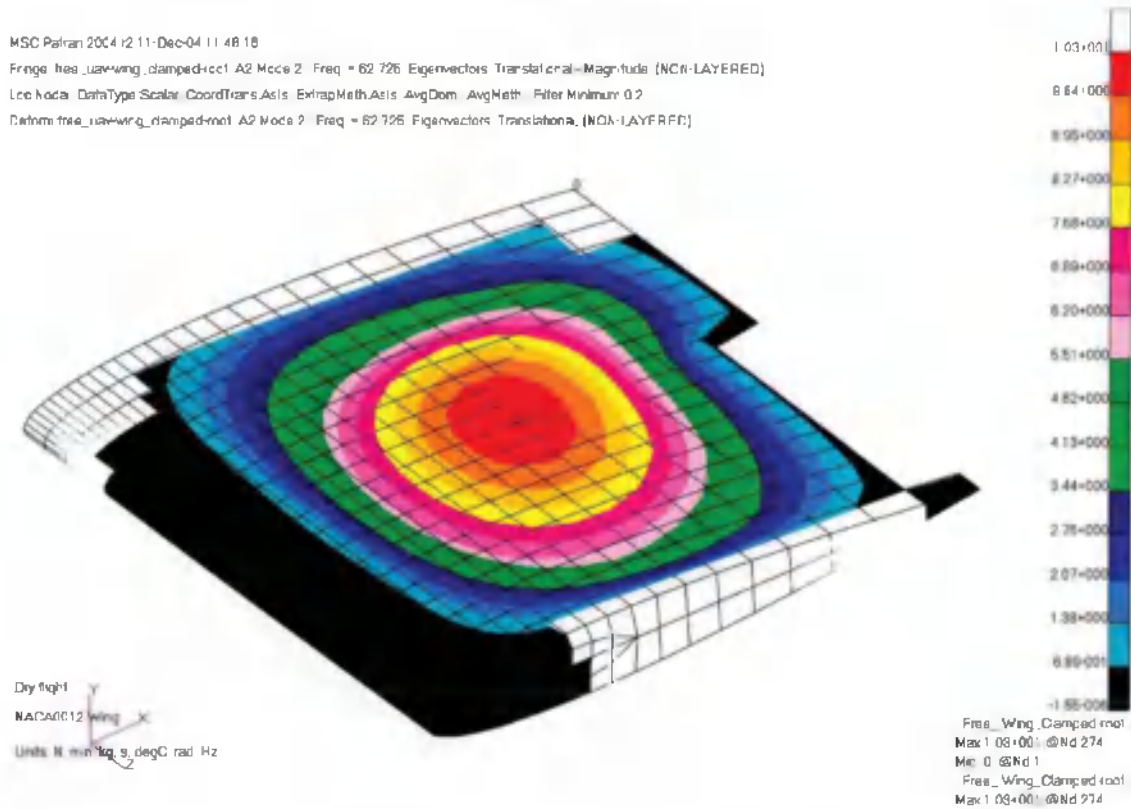


Figure 45: Second Mode (62.7Hz) for Wing Clamped at the Root

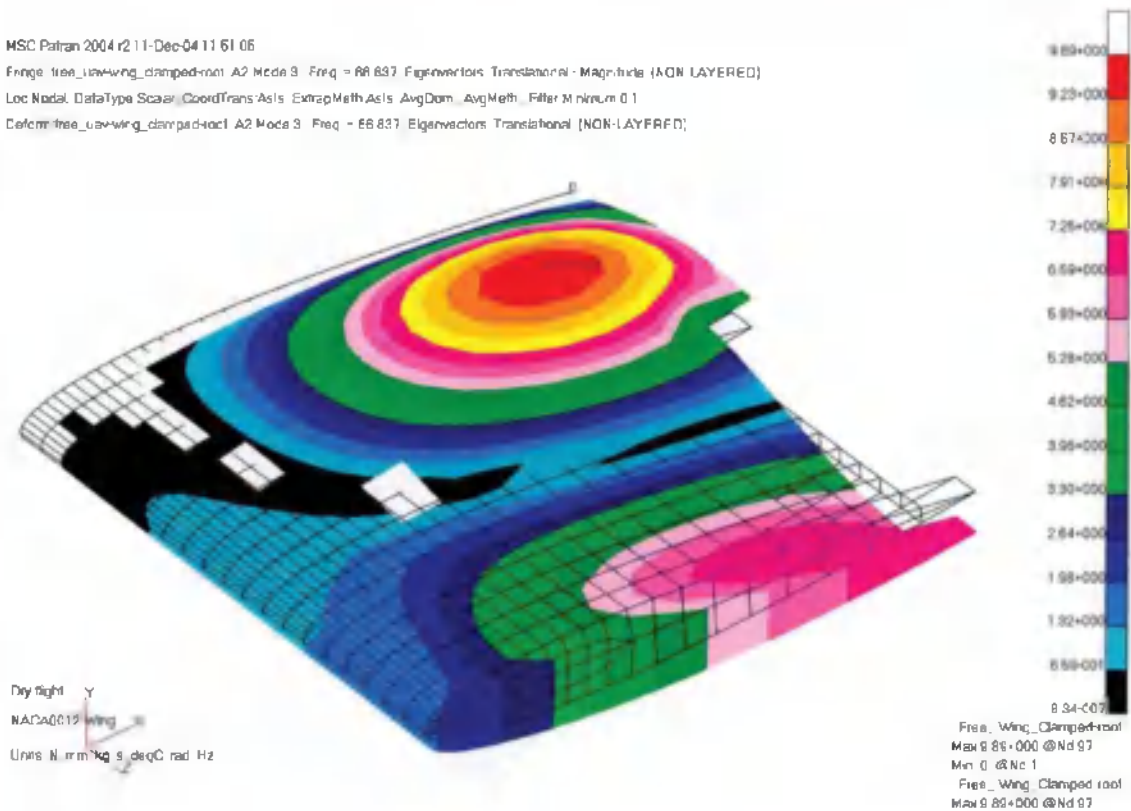


Figure 46: Third Mode (66.8Hz) for Wing Clamped at the Root

To obtain typical strains in the wing structure due to modal excitation, a 2g forcing frequency (range 1 to 100Hz) was applied to a point at the wing tip (NASTRAN analysis code: SOL 111). This would simulate a flight loading, and provide an insight into the likely strains in the structure before the proposed de-icing operation. For the first 3 modes, the maximum principle, minimum principle and maximum shear strain values observed for all composite layers were:

Mode (Frequency)	Maximum Principle	Minimum Principle	Maximum shear
1 (34.6Hz)	9.74E-4	-9.69E-4	9.63E-4
2 (62.7Hz)	2.64E-3	-2.66E-3	2.65E-3
3 (66.8Hz)	2.8E-3	-2.79E-3	2.8E-3

The strain distribution for mode 1 is shown below:

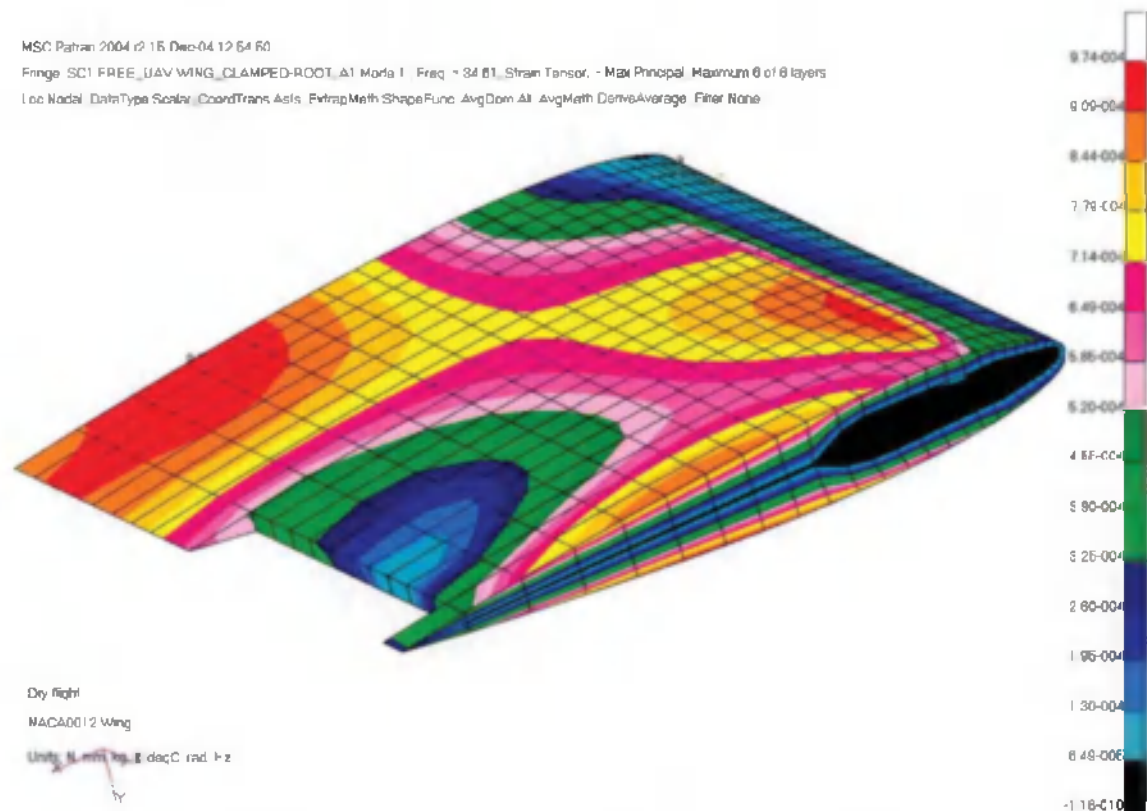


Figure 47: Strain Distribution At the First Natural Frequency (34.6Hz)

3.6.4. FEA Wing Contaminated With Rime Ice

The wing leading edge section of the FEA model of the 10 minute rime ice contamination at 0° angle of attack is shown in Figure 48. The FEA model rime ice variables used were declared in the 3.3.1 above, and profile imported from Fluent CFD model.

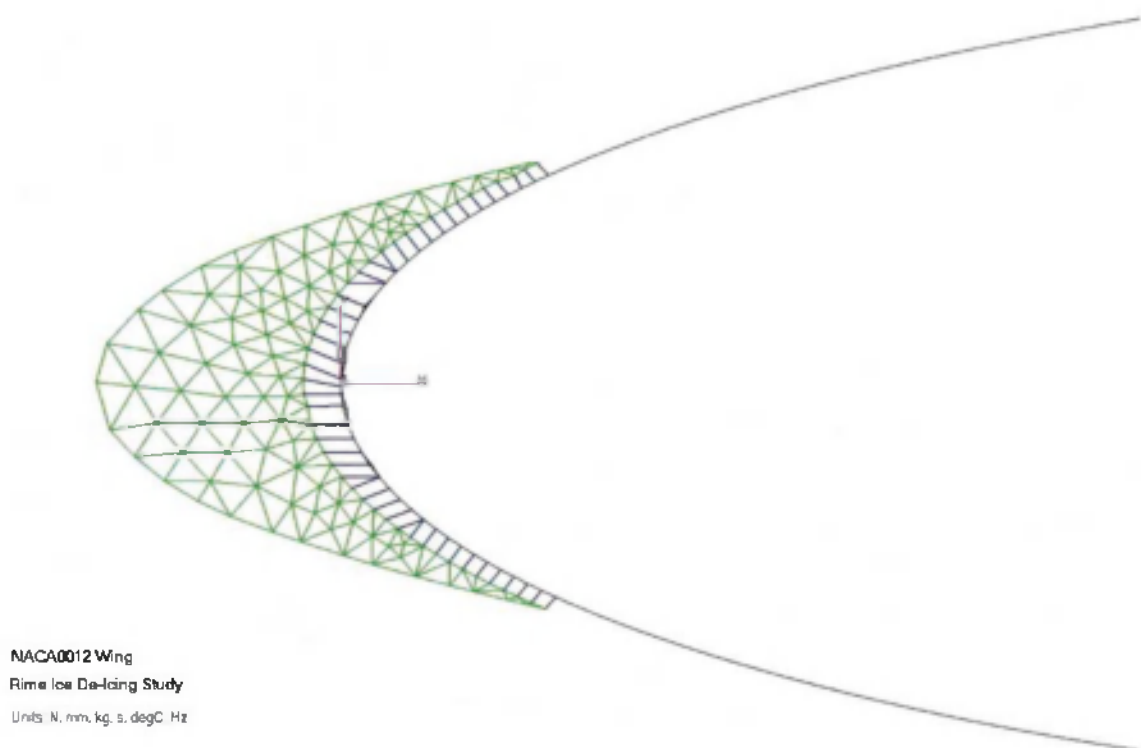


Figure 48: FEA Model of Rime Ice Contamination of The UAV Wing

3.6.4.1 Strains In the Rime Ice Contaminated Wing

The primary forces on a rime ice contaminated wing in flight conditions were determined here. Knowledge of this "base-line" or pre-load condition was necessary to ensure that impulses applied by the proposed de-icing system do not damage the structural integrity of the wing.

The first natural frequency of the composite wing contaminated with rime ice at 0° angle of attack was determined using the MSC NASTRAN solver (analysis code: SOL 103). The first natural frequency prediction and shape is displayed in Figure 49.

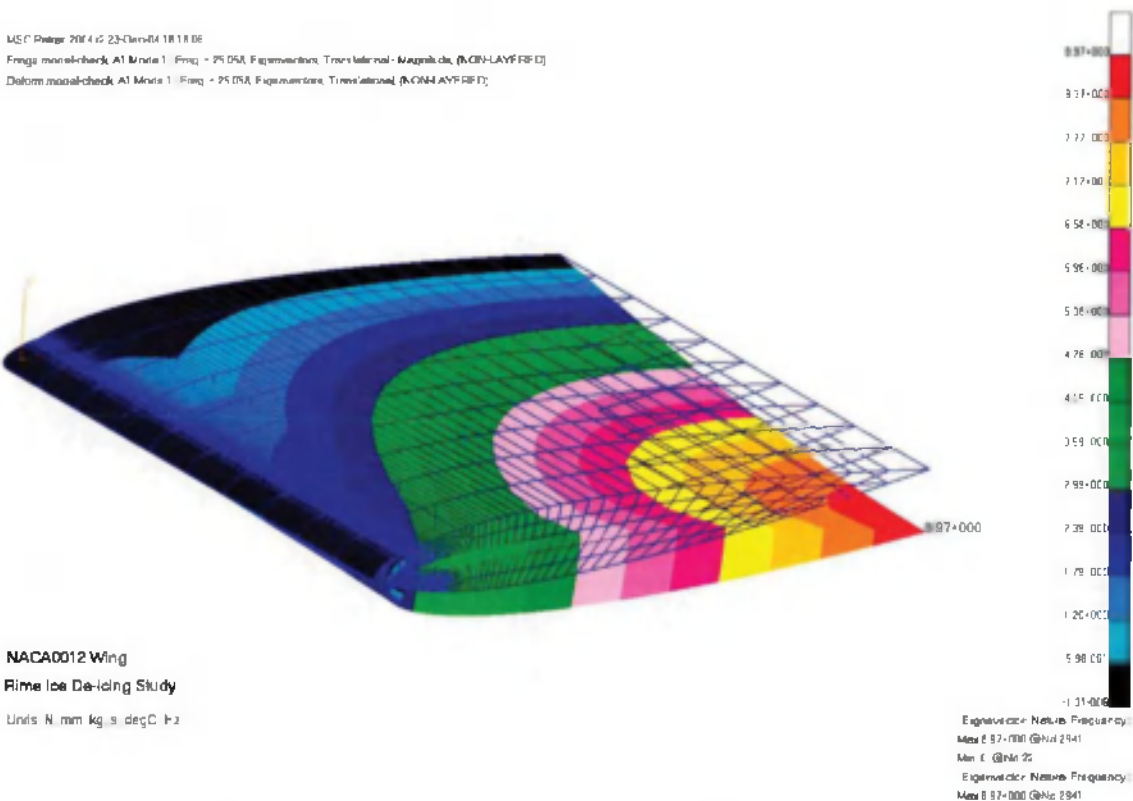


Figure 49: Rime Ice Contaminated Wing - First Natural Frequency (25.1Hz) Prediction

Strains in the wing contaminated with rime ice and subject to the aerodynamic pressure as derived by the CFD study is displayed in Figure 50.



Figure 50: Strains In The Rime Ice Contaminated Wing Subject To Aerodynamic Pressure

Strains In Rime Ice Contaminated Wing Due To Aerodynamic Pressure			
Component	Maximum Principle	Minimum Principle	Maximum Shear
Wing	2.20E-5	-2.22E-5	2.21E-5
Rime Ice	1.02E-5	-3.86E-6	2.41E-5

Based on the calculated shear strain for ice, the above study suggests rime ice will not disintegrate in the simulated static flight conditions. This complements observations made by the author in icing wind tunnel trials, where the cold conditions were sufficient to retain the rime ice accretions. No ice shedding was observed which implies rime ice is sufficiently well adhered to the wing surface, and is not disturbed by the aerodynamic pressure or modal excitation.

3.6.4.2 Proposed De-icing System Function In Rime Ice Conditions.

The above rime ice FEA model was modified to simulate the de-icing actuation. De-icing actuator positions were chosen at approximately 7% of the chord length. This is at the extreme edge of the rime ice accretion as shown in Figure 51 below. The choice was driven by the need to also meet de-icing requirement of the more aerodynamically onerous glaze ice condition.

The impulse profile was derived from the flat plate actuator studies undertaken earlier, scaled up 1000 times to reflect the minimum impulse requirement of 2500N. The author has confidential data from the actuator supplier, that such a actuation mechanism is viable.

A MSC NASTRAN "Transient response" (SOL 112) was set-up, with the impulse actuation undertaken in upper and lower surface pairs. The impulses were time steps of 0.00474 seconds from wing root to wing tip. The entire wing de-icing undertaken for the 3 pairs of actuators in 0.01896 seconds. Each actuator was simulated as a single node through which the impulse force was directed normal to the wing surface. The actuator spacing was centred at 30%, 50%, and 75% of span length in order to capture actuator optimum spacing arrangements.

However, this analysis route failed to capture strain distributions correctly, and a linear static (SOL 101) run was utilised instead. Thus, the displacement values reported are based on a "quasi-static" application of force at selected actuator locations. In practice, the actuation force is a highly, non-linear transient event and would occur extremely rapidly, say 250Hz (to avoid wing natural frequencies) then actuator displacement is of the order of 1.01mm. This well within the capability of the actuator.

Strain and displacement levels were extracted from the results file and graphical output displayed in Figure 52 and Figure 53.

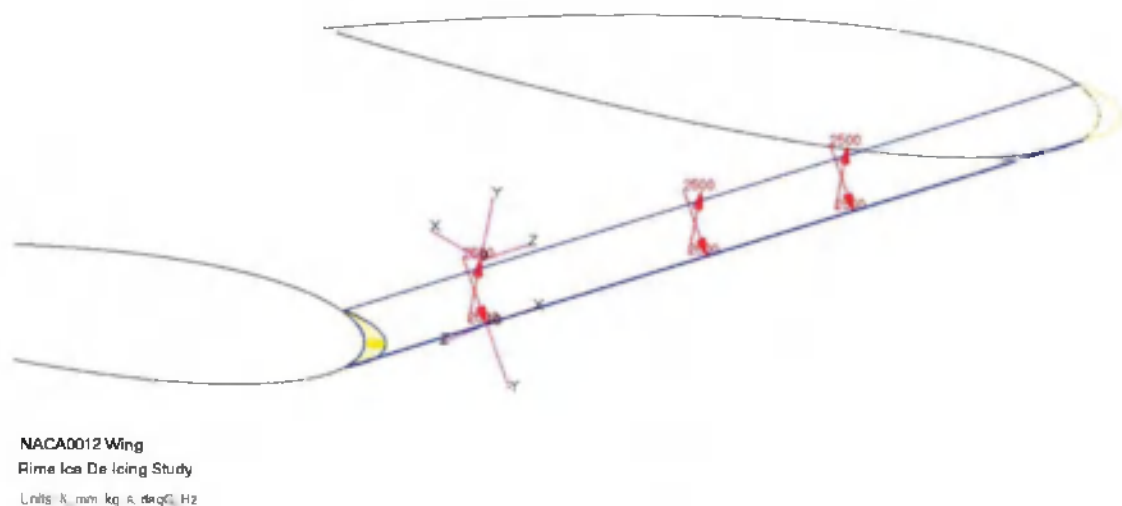


Figure 51: Actuator Positions along Trailing Edge of Rime Ice Accretion (blue line)

MSC NASTRAN predicted the following strains would be in the wing and ice models. The distribution of the strain field is best displayed as a displacement plot shown in Figure 52.

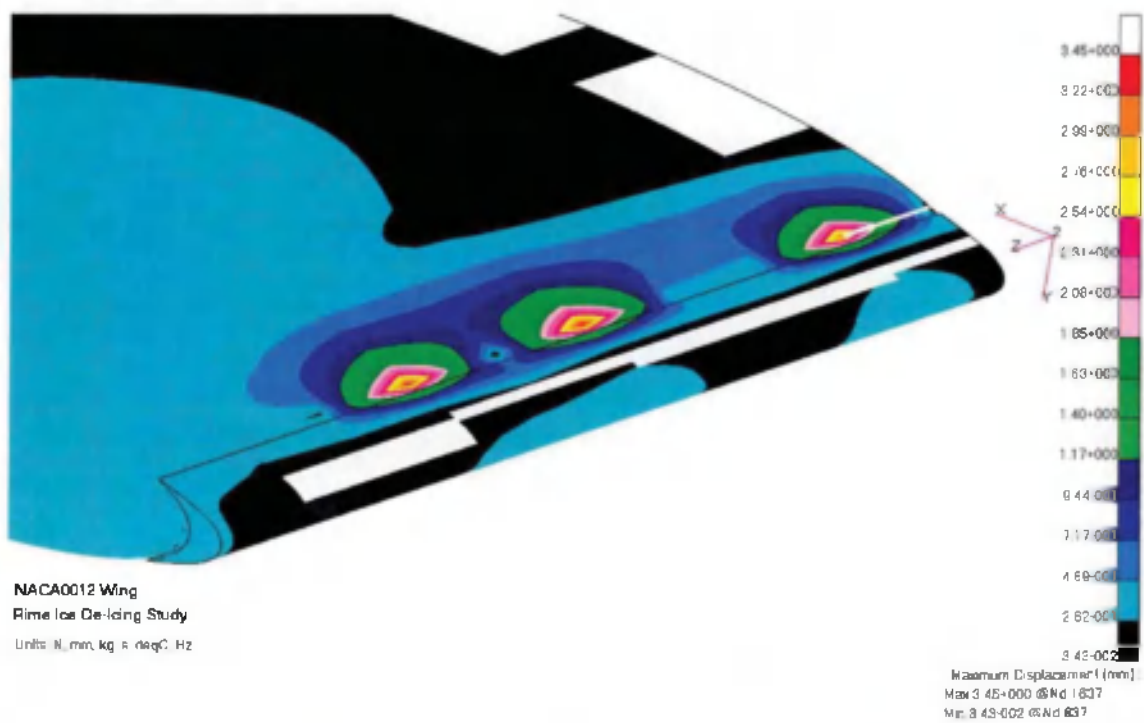


Figure 52: Displacement Plot of Actuator De-Icing Cycle For Rime Ice Contaminated Wing.

Maximum Strains In Rime Ice Contaminated Wing Due To Aerodynamic Pressure + Proposed De-Icing Actuation			
Component	Maximum Principle	Minimum Principle	Maximum Shear
Wing	4.23E-03	-4.15E-03	4.19E-03
Rime Ice	4.03E-03	-3.93E-03	3.98E-03

The maximum strains values reported for the ice exceeds calculated ice shear strains, suggesting the most of the ice bond will be broken, and ice could be lost from the wing surface.

The influence of actuator spacing was obvious, in that the closely spaced actuators provided a more uniform strain distribution as displayed by the red banding in Figure 53. It can also be noted that regions indicated in black (i.e. leading edge ice accretions) have strain values less than the calculated shear strain necessary to break the ice. Thus, the model suggests ice accretions may remain on the leading edge of the wing surface. These could be clarified by the development of an ice crack model, or more robustly by physical test.

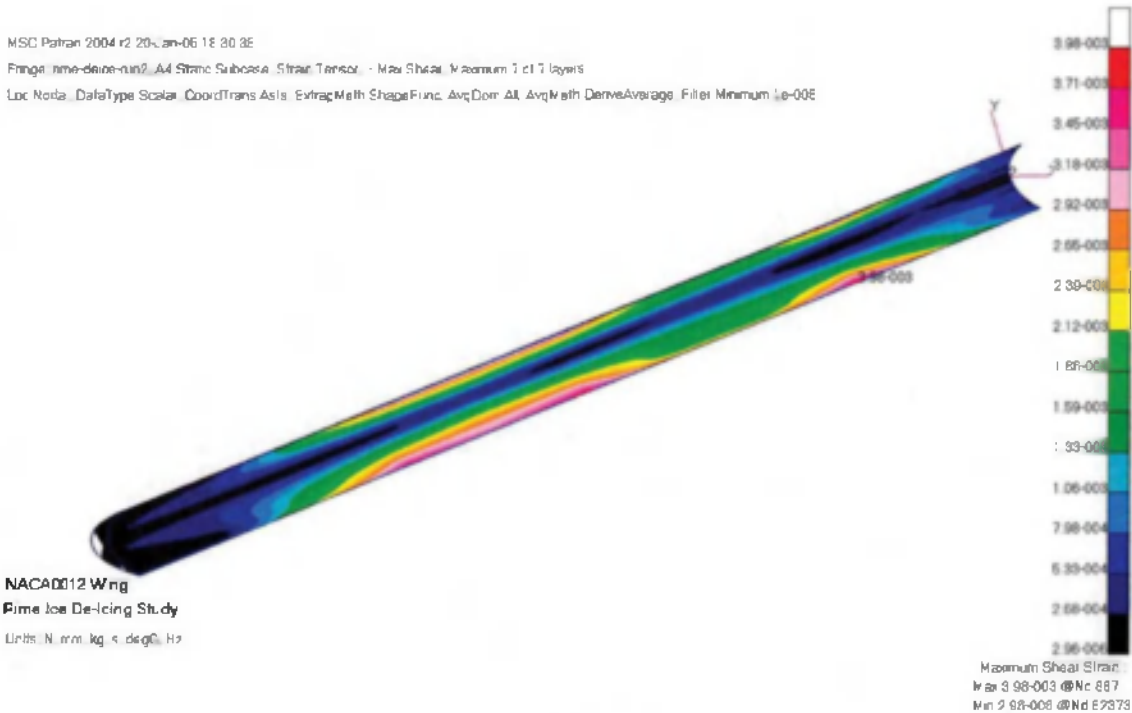


Figure 53: Strain Field At the Rear Face of Rime Ice Accretion

3.6.5. FEA Wing Contaminated With Glaze Ice

The wing leading edge section of the FEA model of the 15 minute glaze ice contamination at 0° angle of attack is shown in Figure 48. The FEA model glaze ice variables used were as declared in the 3.3.1 above, and profile imported from Fluent CFD model.

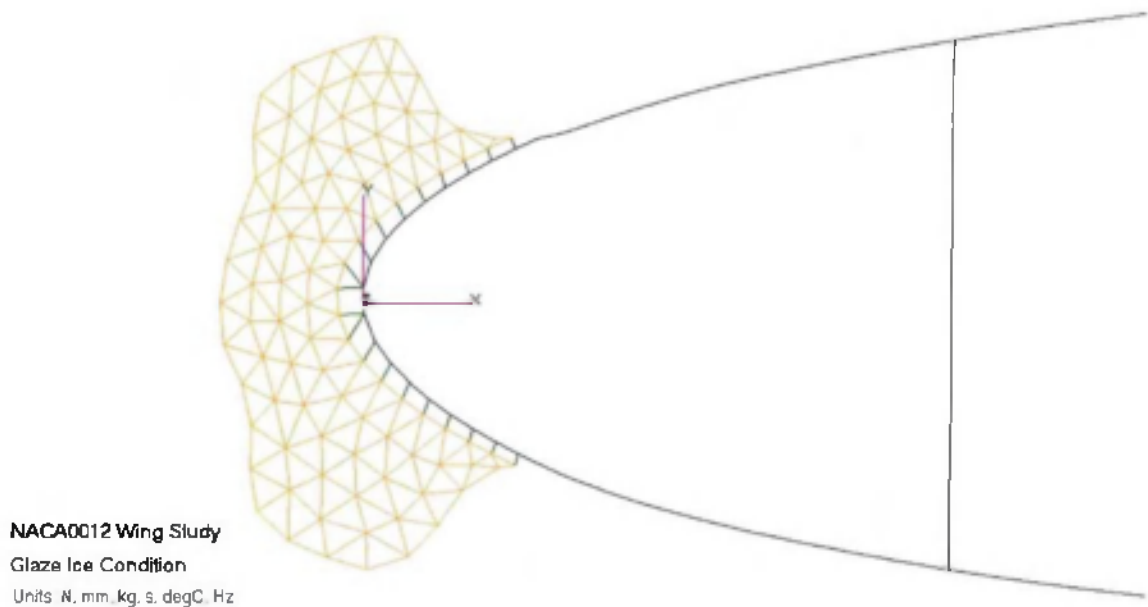


Figure 54: FEA Model of Glaze Ice Contamination of The UAV Wing

3.6.5.1 Strains In the Glaze Ice Contaminated Wing

As with the rime ice study, the primary forces on a glaze ice contaminated wing in flight conditions are determined here. The first natural frequency of the composite wing contaminated with glaze ice at 0° angle of attack was determined using the MSC NASTRAN solver (analysis code: SOL 103). The first natural frequency prediction and shape is displayed in Figure 49.

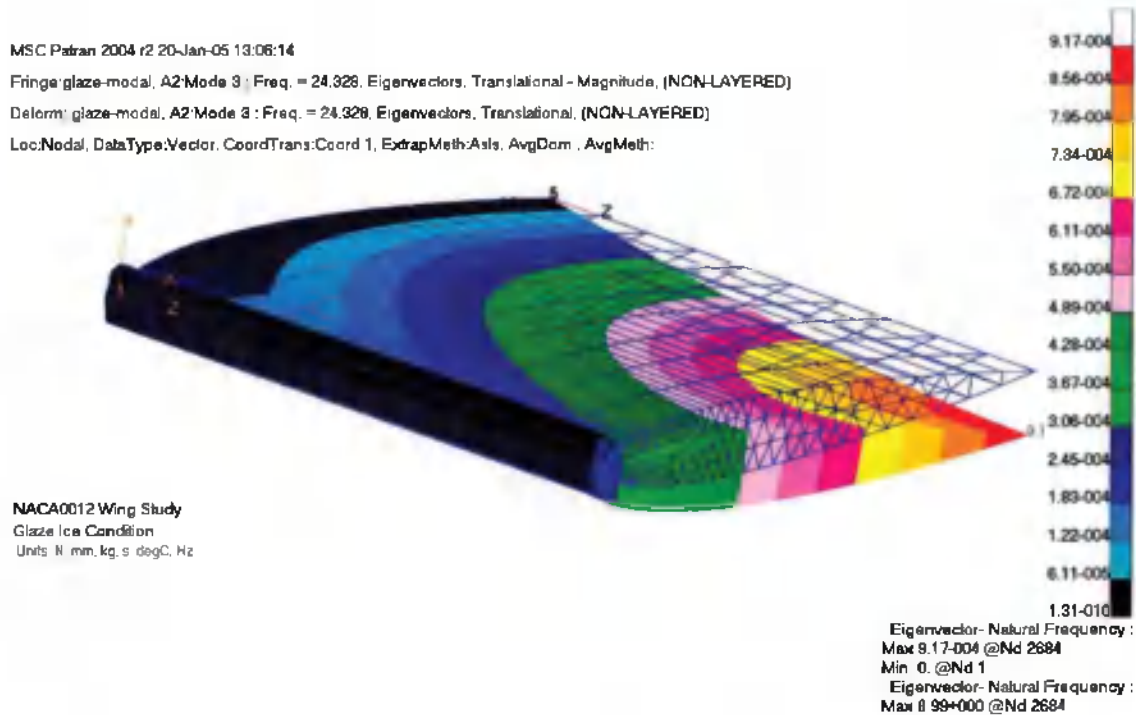


Figure 55:Glaze Ice Contaminated Wing - First Natural Frequency (25.1Hz) Prediction

Strains in the wing contaminated with glaze ice and subject to the aerodynamic pressure as derived by the CFD study are displayed in Figure 50.

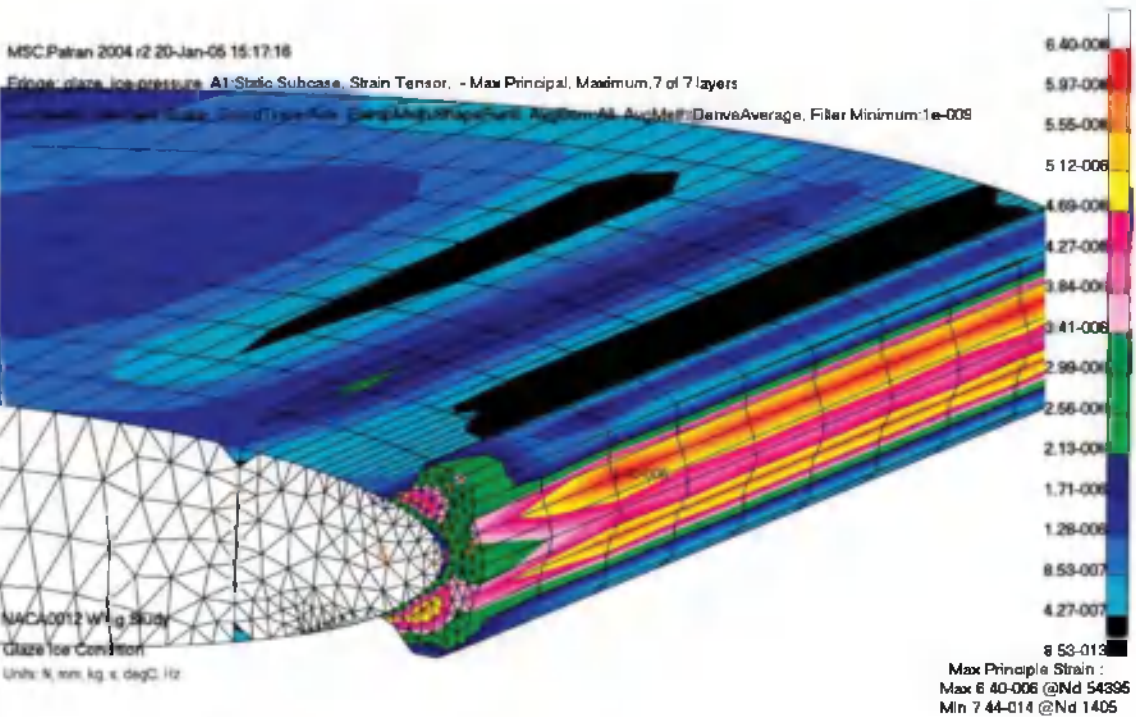


Figure 56: Strains In The Glaze Ice Contaminated Wing Subject To Aerodynamic Pressure

Strains In Glaze Ice Contaminated Wing Due To Aerodynamic Pressure			
Component	Maximum Principle	Minimum Principle	Maximum Shear
Wing	3.93E-06	-7.45E-06	5.31E-06
Glaze Ice	6.40E-06	-1.36E-05	9.36E-06

Based on the calculated shear strain for ice, the above study suggests glaze ice will not disintegrate in the simulated static flight conditions. This may be unrealistic and could be the result of FEA modelling limitation where the root of the wing is modelled as being infinitely stiff and not permitting wing flexing that is observed in flight conditions.

In icing wind tunnel test conditions, the author has witnessed self-shedding of localised small pieces of glaze ice, which can be attributable to stress fracture as a result of any of load combinations (aerodynamic loads and/or modal excitation). In addition, adhesion strength in warmer glaze ice conditions are lower than that of the colder glaze ice.

3.6.5.2 Proposed De-icing System Function In Glaze Ice Conditions.

The glaze ice contaminated wing model was then modified to simulate actuator de-icing. Actuator positions were as displayed in Figure 51. As described errors generated using NASTRAN transient response approach lead to the linear static simulation route instead (SOL 101).

The results obtained from the MSC NASTRAN simulation are displayed below, with the displacement plot provided Figure 57.

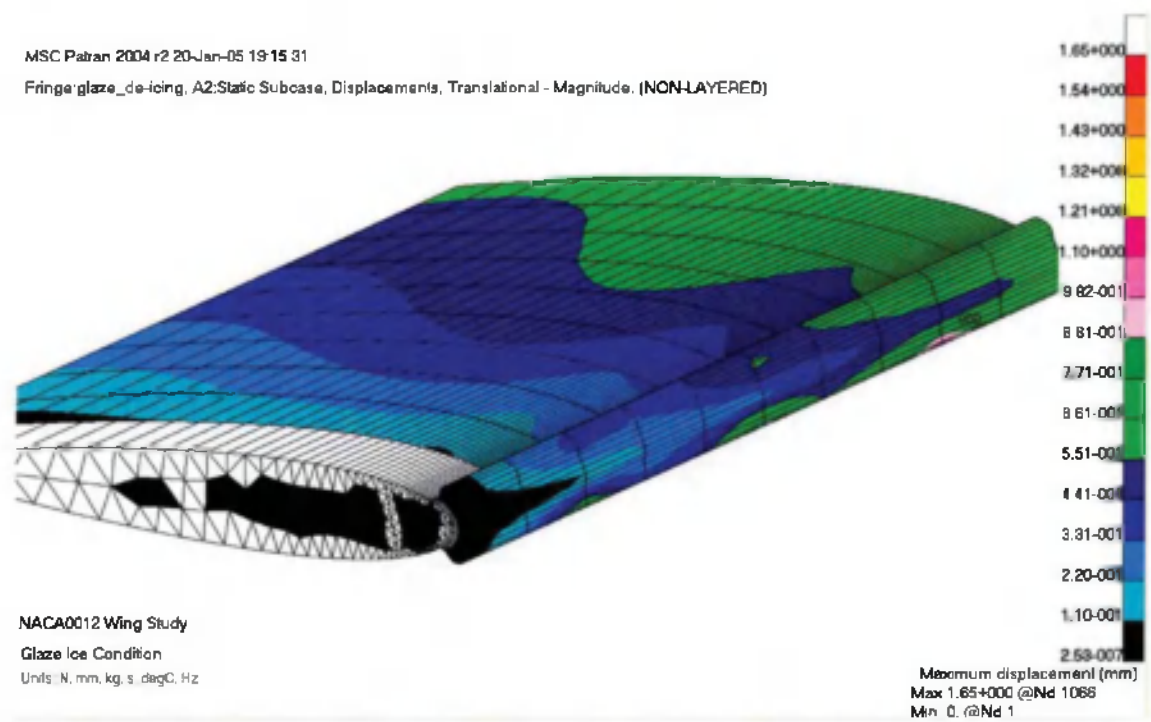


Figure 57: Displacement Plot Of Actuator De-Icing Cycle For Glaze Ice Contaminated Wing

A more detailed plot of the cross-sectional strain distribution predicted by MSC NASTRAN is shown from the rear view of the ice in Figure 58.

Within the constraints of the analytical approach, this study does suggest the optimum actuator positions were obtained for the breaking of the ice-wing surface bond, within the minimum predicted strains in the ice (displayed in pale blue) being greater than the calculated shear strain of ice. The maximum strain values are displayed in the table below.

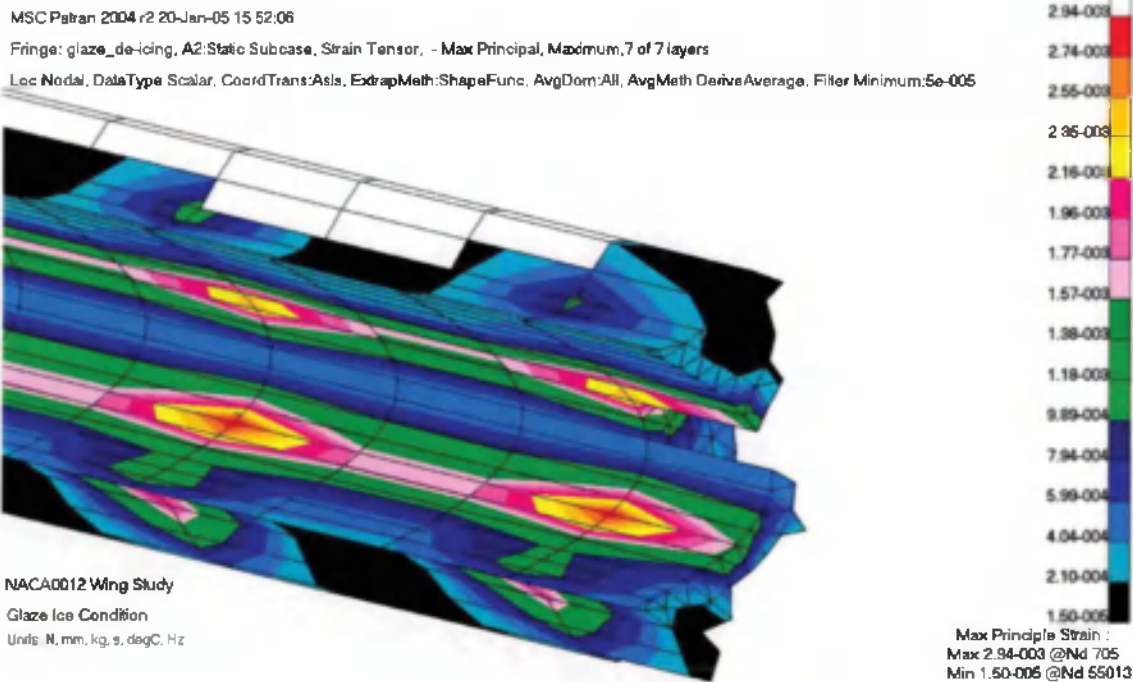


Figure 58: Strain Field in Cross Section of the Glaze Ice Accretion Following Actuator De-Icing

Strains In Glaze Ice Contaminated Wing Due To Aerodynamic Pressure and Proposed De-Icing Actuation

Component	Maximum Principle	Minimum Principle	Maximum Shear
Wing	2.94E-03	-3.94E-03	3.07E-03
Glaze Ice	2.94E-03	-3.94E-03	3.33E-03

CHAPTER 4: DISCUSSION

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4. DISCUSSION

UAV's have increasingly become sophisticated and costly systems that are prized assets for intelligence gathering agencies and the armed forces. Future trends suggest UAV operations will grow to be an important part of general aircraft flight patterns in wider regions of the world.

4.1. Project Objectives

The objectives of this study was to develop an aircraft ice protection system suitable for an Unmanned Aerial Vehicle by:

- Identifying a suitable low cost/ low power technology capable of protecting the UAV wing leading edge.
- Demonstrating the feasibility of this low power/ low cost technology by means physical and computational simulation.
- Understanding the potential advantages and limitations of such a system on the effectiveness on a UAV mission.

The following sections demonstrate how these objectives were met.

4.2. Mission Requirements

UAV wings are designed to provide the aircraft with lift to support long endurance flight with the minimum of fuel consumption. The wing profile provides an efficient flow crucial to sustaining lift. Any surveillance mission is unlikely to take place in fair weather conditions only.

The existence of an atmospheric icing threat to aircraft and specifically UAV operating in moist, low altitudes for long periods has been elaborated in Chapter 1 and 2.

Chapter 1 provided a general description of atmospheric conditions that present an icing threat to aircraft. Super cooled water droplets and ice crystals is described and the likely regions of the world where there is a high probability of an aircraft encountering the phenomena.

The physics of ice accretion on aircraft surfaces was described. Aircraft variables such as airspeed, angle of attack, the geometry of the exposed surface together with environmental variables such water droplet size, cloud liquid water content, ambient air temperature, and altitude affect the ice profile grown on the wing leading edge.

Literature and personal observations demonstrate the accreted ice found on aircraft, are of varying compositions of 2 basic crystalline forms of ice. Rime ice formed at temperatures less than -10°C and glaze ice found at temperature ranges between 0°C and -7°C . Both types of ice have different mechanical and physical properties, which require wing and ice protection system to cater for.

From an aerodynamic drag/lift point of view, glaze ice presents a direct threat to aircraft handling qualities. From an ice removal perspective, rime ice is more difficult to remove due to the stronger mechanical bond between ice and wing surface.

The presence of ice accretions on the wing leading edge presents the UAV designer with a significant challenge. UAV wings are designed to provide the aircraft with lift to support long endurance flight with the minimum of fuel consumption. The wing geometry provides an efficient flow crucial to sustaining lift.

It is a high probability, the aircraft on an extended period of flight will encounter icing conditions for which the UAV operator will have little foreknowledge of, or more commonly will be forced to fly the aircraft through it.

If aware of the meteorological conditions that may result in aircraft icing, the UAV operator may choose to delay the mission or fly around it. An alert to the UAV operator can be made by having an ice detector fitted to the aircraft. This would provide the operator with options to abort the mission, fly above the icing condition, choose a different route etc.

4.3. Existing Ice Protection Technologies

If the UAV designer chose to deploy an ice protection system in addition to the ice detector, Chapter 2 provided a survey of existing ice protection systems available for installation. The assumption being made here is the UAV has a suitable wing profile, available power, and capacity. This issue is fundamental, given the limitations of power and load carrying capacity of the aircraft.

Assuming the decision is made to make power and capacity available for an ice protection system, the designer is faced with a choice of proven aircraft systems. It would be obvious to the designer on a limited power budget, the anti-icing systems based on electro-thermal heating were not suited to wing leading edges. Based on power and weight, the de-icing routes seem the most power and weight efficient at a modest compromise of aerodynamic drag during ice build up.

For turboprop engine aircraft, the system of choice has traditionally been pneumatic de-icing. For aircraft with the available power, the wing leading edges were fitted with pneumatic boots. For effective operation, visual judgement is used to activate and correct cycle times. This facility is difficult on a remotely piloted vehicle where the operator may not be able to observe the system's de-icing operation.

The main disadvantage of pneumatic de-icing is the large performance penalty due to deliberate ice accretion and on inflation. The ice accretion has to be of significant thickness to be effectively removed. For wings designed for efficient flow characteristics, the airflow is disturbed significantly. It also requires the aircraft to have suitable pneumatic installation, which may already be present for driving flaps and other wing devices. There is a drive towards "all electric" aircraft, for which, UAV are the ideal demonstrator platforms. Part of the drive to all electric aircraft is to reduce the system complexity, and therefore aircraft mass.

Electro-thermal de-icing does offer the route towards all electric aircraft. The system does require significant electric power and corresponding increased power generation capacity. This has an impact on fuel consumption and effective range of the aircraft. Electro-thermal de-icing requires ice accretions to reach minimum thickness to achieve clean removal. This ice accretion thickness is smaller than that necessary for pneumatic boot systems, and therefore an improvement on pneumatic systems in relation to the aerodynamics penalty.

Electro-thermal de-icing seems suitable for UAV wings. In particular, if the heater pads are incorporated within the composite wing skin laminate. However, power consumption, and the additional power generation capacity and its attendant fuel and mass penalties tend to steer the designer away.

Hot air based systems bleeding air from the engine compressor is not suited for an aircraft with neither the power nor the spatial arrangement to pipe hot duct air. In addition, the aircraft fuselage and wings are made from lightweight, polymer composites, which degrade at the hot air temperatures (even when mixed with cooling air).

Electro explosive de-icing systems (EEDI) seem suited for UAV wing application. The application has successfully been demonstrated on F/A18 Engine Inlets, and power consumption appears to be low. No system mass for a FAR 23/ JAR 23 standard aircraft (equivalent to medium and long endurance class UAV) was available. However it's total system mass and power consumption may be competitively similar to the magnetostrictive de-icing system proposed in this study.

Electro-impulsive de-icing (EIDI) has similar de-icing mechanism to that of the proposed magnetostrictive de-icing system in that the wing surface is subject to a large impulsive force. The means of effecting this impulse differs. The electro-impulsive system induces an Eddy current in a metallic disk bonded to the wing surface. The collapsing magnetic field presents a displacement that causes ice-surface bond to break. Experimental evidence from NASA/Kansas University trials suggest the system works effectively. Typically, three de-icing cycles where impulse forces greater than 2000N was applied to successfully remove ice accretions. The system's power consumption was low (~500W for 6 actuator passenger carrying aircraft), the total system mass appears to be considerable. A significant proportion of the system's mass was the heavy coils and solenoids utilised in the actuators. Power losses due to dielectric heating and cable resistances reduce efficiencies. A number of the issues associated with pneumatic boots such as the aerodynamic penalties are equally valid here.

Fluid anti-icing system is currently employed on the Predator B class UAV. As ice mitigation system, the glycol fluid lowers the freezing point of the accreted ice preventing the ice from freezing and bonding to the aerofoil surface. It is highly effective for this UAV application.

However, the main disadvantages are the mass of fluid that is required to be transported until the icing encounter (and therefore it's fuel consumption penalty), and its limited life (typically 60 minutes). For medium and long endurance UAV's where mission times can be long as 8 hours and more, a 60 minute ice protection life is a significant limitation.

4.4. Alternative Technologies

The observations above suggest conventional wing ice-protection requirements are at best adequate for this class of medium and long endurance UAV aircraft. This study examined if alternative technologies were available that were suited for this class of aircraft. The simplest, lightest and requiring the least amount of energy was to prevent ice from bonding to the wing surface in the first place i.e. application of an ice phobic coating. Chapter 2, section 2.3 collated experimentally evidence of various commercially available coatings in preventing ice from adhering to a simulated wing surface. The icing wind tunnel based trials by NASA and others showed that most ice phobic coatings were of limited durability. The effectiveness of the coatings on preventing ice from bonding to the wing surface was rapidly reduced with each mechanical ice removal.

For UAV wing surfaces, the reduction in the ice-surface bond strength did not prevent ice accretions from developing in the first place. Thus, the all-important efficient airflow could not be assured with the use of ice phobic coatings. However, it may aid the existing ice protection system, by reducing the energy required to remove the ice.

If ice formations cannot be prevented, perhaps the use of existing conventional ice protection systems can be hybridised to reduce power consumption and system mass. Chapter 2, section 2.4.2 examined this approach, with the development of hybrid electro-thermal anti-ice and electro-expulsive de-icing system. This takes advantage of both systems' strength.

The anti-ice "running wet" electro thermal system prevents ice formation by keeping the local surface temperature above freezing. The impinging water droplets run back onto an aerodynamically less sensitive region of the wing leading edge. This water is allowed to freeze and then routinely removed by the low energy electro-expulsive system.

This system is an effective compromise between the needs of the UAV wing for efficient airflow and the power requirements to achieve such a goal. The surface roughness of this "runback" ice is likely to be less than that of surface roughness of ice developed from direct impingement. This in turn may reduce aerodynamic drag losses, but the transition to turbulent flow would creep forwards.

The hybrid system does have a greater system mass, since both systems need to be catered for and controlled. For UAV wing applications, this hybrid system has definite advantages over the existing fluid ice protection system in terms of mission longevity.

Common to electro-mechanical de-icing systems is the need to apply a large impulse force on the wing surface to break the ice-surface bond, and break the monolithic ice into fragments. The aerodynamic forces then remove this fragmented ice. This impulse force is translated into small-localised displacements of the wing surface. Chapter 2, Section 2.4.3 showed audio loudspeakers perform, a similar function on the speaker cone. Advances in materials technology, allowed the replacement of solenoid coil system with smaller, energy efficient piezoelectric and now magnetostrictive actuators. Energy losses due to the power cable resistances, coil, and solenoid heating were eliminated. With this in mind, a transfer of technology from audio speakers to aircraft wing leading edges was made.

4.5. Proposed Magnetostrictive De-icing System

At the heart of the proposed novel ice protection system described in Chapter 2, Section 2.5 is the magnetostrictive actuator. This actuator uses the magnetostrictive phenomena (present in all materials) to translate an applied electromagnetic field into a large expansion of the actuator material's geometrical length (~1-2mm). The rod tip in contact with the wing surface applies an impulse force (~0.002 seconds). This large impulse breaks the ice-surface bond by rapid expansion/contraction and breaks up the accreted ice into fragments. Magnetostrictive materials have a magnetic field "saturation" level beyond which further increases in the geometric size or strength of the magnetic field does not cause further expansion of the magnetostrictive rod. To overcome this limitation, a larger number of these small discrete devices can be employed. The use of "smart" electronics and device distributions will ensure uniform ice removal.

The author argues this is a more competitive solution to that presented by electro-expulsive (EEDI), electro-impulsive (EIDI) and hybrid systems for the following reasons:

- Magnetostrictive devices convert the applied magnetic field into localised surface displacement more efficiently than EIDI because the actuator tip is directly in contact with the wing surface. Electro impulsive systems require a metallic target in which an Eddy field induces. Only the magnetic field generated within the metallic target surface area or volume is utilised. In terms of device mass, the EIDI based metallic target/solenoid coil system is significantly greater than the solid-state magnetostrictive actuator.

- Electro-expulsive systems rely on coincident magnetic fields induced by the current to cause copper conductive strips laid parallel to each other to move apart. Part of the applied current is lost as thermal energy as the current overcomes resistance. Magnetostrictive materials are ceramic electrical insulators. No currents are induced, thus thermal losses are smaller. In addition, parallel strips of copper conductors coils covering the entire span and part of leading edge chord, together with harnesses and connectors are likely to be heavier than the proposed system.
- At the wing leading edge tip, EIDI devices cannot be accommodated, whereas EEDI present wing geometry changes to the designer. Electro-thermal heater pads address this issue for hybrid systems but require significant power. The proposed magnetostrictive devices are small enough to be fitted within the nose region.

The main disadvantage with the proposed magnetostrictive de-icing system is the need for a minimum ice thickness to build up before removal. Practical experience may demonstrate that the smaller devices may provide greater coverage, and in the turn remove ice thickness less than required by conventional systems.

4.6. Proof of Concept Study

To demonstrate this technology is suited for the task, data was sought from manufacturer's and quotations for a system that could serve as demonstrator de-icing system was sought. Due to the inordinate cost of a customised "on-off" demonstrator system, a commercially available actuator in the form "sound bug" was purchased. This would serve as a low cost demonstrator platform, from which adequate understanding of the de-icing mechanism could be obtained. As seen in Chapter 3, Section 3.1, the commercial device stripped of it's housing could easily be handled with the minimum of investment in fixture and tooling.

The easiest means of determining its interaction with a composite UAV wing, was to experiment and analyse it's function with the simplest representation of the wing i.e. a flat composite wing, held in an unconstrained i.e. "free-free" state. The wing laminate construction was not available. However, a sample laminate test specimen whose geometry, materials, mechanical properties, and lay-up was known, and was constructed to replicate wing build.

A composite laminate of the approximate thickness and lay-up was prepared by taking a microscopic cross-section from the UAV wing. A basic manual analysis was carried out initially (shown in Chapter 3, Section 3.3.1) and the first five natural frequencies of the composite panel determined. These calculations assumed the classical laminate theory would generate a reasonably accurate Effective Modules and Strength for a multi-ply, multi-orientated glass-epoxy laminate (see Appendix A6).

Classical Laminate theory employs basic material engineering properties at individual ply level and together with the laminate definition and loading arrangements. The material values input and assumptions used are listed in Chapter 3, Sections 3.1.2. and 3.2.7. The material properties employed in this study were based on data from commercially available woven glass-epoxy fabric and aramid honeycomb core.

"LAP" is a Windows based software consisting Classical Laminate theory algorithm. The input, calculation, and output files are listed in Appendix A7. The software permitted the effective stiffness i.e. general modulus of the laminate to be determined. Upon determination of the modulus of the test specimen, the geometrical dimensions were flat test piece noted, and estimate made of the first 5 natural frequencies of the test panel determined using Roark's formulas for stress and strain (Appendix A4). Knowledge of these frequencies ensured that any strong responses measured during the test could be filtered from the measured frequency response spectrum.

"Free-free" flat panel test set up described in Section 3.1.3 enabled the general interaction between actuator and composite laminate to be observed. A square wave signal was produced by "Sigjenny" software and output through the PC soundcard was fed to the actuator in direct contact with the suspended composite panel. The strain response against frequency was captured by a PC following suitable amplification and conversion (Chapter 3, Section 3.1.5). Test data was then analysed and processed into a root mean square (rms) power density against frequency in Appendix B.

The data was further investigated, and the g_{rms} acceleration against frequency estimated for the actuator in contact with a composite panel. The power density root mean square against frequency graph showed the peak strain response in the composite laminate at approximately 283Hz and 304Hz, allowing for the elimination of these responses from the measured natural frequencies. If strain gauge rms power densities are "converted" to acceleration rms (assuming a direct linear relationship between strains and displacement) by an integration method by Irvine, an almost flat line response is shown from 100Hz to 700Hz. This suggests the actuator could deliver an even impulse force over that range. By tuning the actuator, an impulse load can be delivered at a frequency more suited to the dynamic properties of the wing and the de-icing function being sought.

The dynamic properties for the Phoenix UAV wing were measured experimentally in Section 3.2. The objective of the study was to determine the natural frequencies of an actual wing, and build a finite element model (FEA) replica of this wing. If the FEA model were to produce similar modal response, then confidence in the stiffness and therefore the FEA predicted results is assured.

"Free-free" modal test set up of the Phoenix wing, and subsequent processing of accelerometer data is described in Chapter 3, Sections 3.2.3 and 3.2.4 respectively. The mode shapes and frequencies captured are shown in Chapter 3, Section 3.2.5.

A parallel study using MSC PATRAN to develop the Phoenix wing FEA model, and MSC NASTRAN to "solve" the modal analysis provided mode shapes at frequencies close to those determined experimentally was described in Chapter 3, Sections 3.2.6 and 3.2.7. This gave confidence that "dynamically", the Phoenix FEA model was strongly representative of the physical wing specimen.

This statement was supported by examination of the test mode frequencies and mode shapes with the MSC NASTRAN FEA predictions (Section 3.2.8). Seven mode shapes and frequencies predicted by MSC NASTRAN coincided with physical test determined modal shapes and frequencies, using Mil-STD-810F definition for modal similarity.

It had been envisaged during conceptual planning of this study, icing wind tunnel experiments would be undertaken to first determine the icing threat faced by a UAV wing test specimen (e.g. Phoenix UAV wing), in various icing conditions, and second, physically deploy and demonstrate an ice protection system that suited a UAV wing.

As neither funding for the proposed ice protection system nor, the Cranfield University icing tunnel were available in time for this study, a computational simulation was undertaken instead. To determine the extent of ice accretion and shape of the ice to be removed, TRAJICE-2D ice accretion simulation software was used. This FORTRAN code developed by UK Ministry of Defence Research Agency (DERA now QINETIQ) had an icing tunnel experimentally verified case study based on NACA0012 based wing profiles (Section 3.3.1).

Adapting this case study to UAV wing flight profile meant a change in the input variables such as airspeed, angle of attack, altitude, ambient air temperature and liquid water content that would be expected by UAV aircraft. A number of studies using time steps of 5, 10 and 15 minutes were investigated at alternative variables of 0°, 4° and 8° angles of attack, liquid water content of 0.5g/m³ and 1g/m³, and 80 knots and 100knots airspeed.

Only the cruise condition 0° angle of attack, liquid water content 1 g/m^3 , 80 knot airspeed for -5°C (15 minutes glaze ice condition) and -10°C (10 minute rime ice condition) were reported here. This was due to the supporting literature, and verifiable test evidence that was available for these specific conditions. It was also argued that the 0° angle of attack flight condition was likely to be the most prevalent flight attitude through out the mission envelope (Chapter 3, Section 3.3.2).

A "sanity" check of the data items used was undertaken in Appendix A1, by means of an energy and mass balance analysis. The proof is by induction, provided additional confidence. The heater mat intensities at the leading edge of the wing were in line with FAA Icing Handbook published data.

The commonly encountered (at 0° angle of attack) ice shape profiles of the two ice crystalline forms (rime and glaze) were exported from TRAJICE-2D in form of datum coordinates. These coordinates were imported into Fluent CFD. CFD models were developed based on the two TRAJICE profiles (Section 3.4.1). Mesh models created allowed the aerodynamic effects of a clean wing versus ice-contaminated wings to be calculated. Of particular interest, was the pressure profile on the aircraft wing. The CFD predictions for the clean wing were verified as of "reasonable correlation" with NACA developed wind tunnel test data.

The two dimensional pressure contours were exported from Fluent CFD into MSC PATRAN. Within which the TRAJICE-2D geometry had already been imported and meshed (Section 3.6.1). With the application of suitable materials and boundary conditions (Section 3.6.2), the strains in the ice-contaminated wing were calculated. Without a de-icing system load, the FEA models predicted the ice accretions were stable, and were not severely stressed by aerodynamic pressures or modal excitement (Section 3.6.3).

The proposed de-icing system entailed the application of an impulse load. This is a dynamically transient condition. Moving from the wing root to the wing tip, pairs of actuators would consecutively apply a force to aerofoil surface over a very short interval.

4.7. Results of Concept Study

It was hoped that the FEA simulation would replicate this de-icing action by analysing one pair of actuators interacting with the surface ice, and then repeat the analysis for the next pair of actuators, whose starting condition was the final ice strain condition of the previous pair of actuators. It had become apparent that MSC NASTRAN was unable to support this approach. Alternative software MSC MARC was better suited for this step approach. Access to this FEA simulation software code was unavailable to the author.

As the determination of the optimum impulse level and actuator positions was the central piece of information being sought, a linear static analytical approach using MSC NASTRAN was used instead. This approach was considered valid, as ice is a brittle material, whose behaviour is considered linear elastic up to the point of fracture with no discernable yield point. The ice fracture takes place and is induced at very small displacements, which a fundamental axiom for linear static analysis. However, it should be noted a wide range of ice-surface adhesion bond strengths are possible, thus the last statement is correct for this condition.

Following a number of iterations where variables such as actuator span distance, actuator chord location, angle of attack, impulse magnitude, water droplet diameter, an optimum impulse load of 2500N at a 5% chord distance and span distance 30mm apart provided strain levels and strain distributions that would assure ice-surface bond is broken across the leading edge. Actuator locations was favoured towards 0° of angle of attack, as this was perceived to be the dominant flight mode (Chapter 3, Sections 3.6.4 and 3.6.5).

The results obtained are of the same order of magnitude as that determined experimentally by the NASA/ Kansas University consortium elector-impulsive de-icing (EIDI) trials discussed in Chapter 2, Section 2.1.5. albeit with a Kevlar composite wing leading edge.

Scrutiny of the photographs within the paper of the NASA/Kansas wing test specimen (Chapter 2, section 2.1.5) indicates no core was present within the wing leading edge. This differs from the FEA model representation of the Phoenix UAV wing where aramid honeycomb core was present. The damping characteristics of a honeycomb core sandwich panel are likely to be significantly greater than that of a monolithic Kevlar composite panel at the same frequency. The damping losses (i.e. loss of energy as a result of material/ molecular inefficiencies) were sufficiently greater to suggest greater impulse forces were necessary to remove the ice.

A further possible limitation is the linear static analysis route chosen. As the event is a dynamic one, the base wing structure may not react quickly enough to absorb some of the impulse forces as is suggested by this study. In which case more of the impulse energy would be transmitted to the ice accretion. This can be verified by physical test.

Supporting calculations in Appendix A8, demonstrate that if the actuation impulse force was modelled and applied in the frequency domain, say at 400Hz, the observed aerofoil displacements are significantly smaller than that suggested by the linear static study in Chapter 3, Figure 52 and Figure 57. The magnitude of any surface displacement is also proportional to the mass of ice accreted i.e. the exposure time in icing conditions. The earlier the de-icing system is activated, the greater the aerofoil displacements would be observed as less ice mass is required to be removed.

To actuate such large aerofoil surface movements requires the device to be able to convert the small actuator piston displacements (microns) into larger aerofoil surface movements (millimetre) i.e. provide up to x1000 piston stroke.

Traditional devices such conventional pneumatic and hydraulic actuators exist in the market place for converting small displacements into larger stroke movements. The incompressibility of fluids is the underlying physical principle in these machines. Adaptation of such devices into compact forms suited for attachment to small thickness UAV wing sections has not been considered in detail in this study.

Perhaps one route towards compact design could be to use the magnetostrictive actuator to compress a pre-compressed spring. The magnetic field is removed to de-activate the magnetostrictive material. The spring releases it's compression energy to drive a piston in contact with the non-airflow surface to provide the impulse to de-ice the wing surface. A second magnetostrictive actuator, together with the reaction force then drives the piston back to pre-compress the spring.

For application on a UAV wing, the aircraft designer would want further information on a number of the following issues discussed in the sections below.

4.8. Proposed Magnetostrictive System Advantages

A UAV designer may ask what the advantages are of the proposed magnetostrictive de-icing system over conventional systems.

The author contends this proposed de-icing system presents significant advantages in terms of mission effectiveness at relatively low economic and fuel burn costs. The system can be considered "low power" if considered equivalent to the EIDI system. The justification for this statement is that similar impulse loads are involved in de-icing operation. Differences such as impulse induced by non-contact means (EIDI) rather than the direct contact route employed by the magnetostrictive system do not change the charging and control circuits common to both systems.

The author asserts the proposed magnetostrictive actuators are more discrete devices that are more efficient at converting electrical energy into surface displacement than the more conspicuous EIDI coil/target arrangement or, the parallel conductor configuration used by EEDI. Arguments have been presented above to support this statement. Based on this reasoning, the power requirements are expected to be ~500W in line with the power used on EIDI systems.

If a suitable stable ice phobic coating were employed as well, the mechanical impulse energy required to break ice-surface bond would also be lowered, reducing the system power consumption. It is envisaged the coating would be renewed before each UAV mission.

In terms of system mass, the proposed magnetostrictive de-icing system is likely to be of equivalent mass to EIDI or EIDI systems or the existing Predator B fluid ice protection system of 27kg (estimated). Although the magnetostrictive actuators are smaller and lighter, more actuators are likely to be employed if magnetic field saturation is the limitation.

This can be of advantage for UAV wings where the smaller radius of curvature rule out EIDI based systems. Appendix A5 provides a fuel consumption study with the existing the de-icing system on the Predator B aircraft. The suggestion in this primitive study is that icing contamination of the wing is costly in terms of fuel burn. The study is "primitive" or first order as it does not allow for changing flight conditions or decrease in the fuel mass throughout the mission duration.

A positive feature of the proposed system is the system is non-intrusive in the air stream. Hence no aerodynamics penalties when compared with pneumatic de-icing systems. The proposed magnetostrictive system can also be considered low maintenance, as there are no moving parts. However, diodes, capacitors, and actuators can fail.

In operation, there is no heating of the wing surface. Thus, there is no "run-back" ice refreezing aft of the ice protection system. For UAV wing applications, it is anticipated that an ice detector will be fitted to each system installation. This is considered necessary, as the UAV operator may not be aware of the local meteorological conditions, and therefore when to activate the ice protection system. The magnetostrictive actuator is a very sensitive device, which may through wing surface contact to be able to detect ice accretions. This concept is outside the remit of this study.

The controller logic circuit will have a self-test capability. Upon installation and activation will be able to inform the UAV controller of any defects. This provides the UAV operator with sufficient data to continue the mission, plot alternative routes, or abort the mission if aircraft safety were to be compromised.

4.9. Proposed Magnetostrictive System Limitations

Limitations with the proposed systems that cannot be resolved without physical testing and operational experience are:

- Fatigue life - Fatigue life of the system and its mounts on the aircraft wing. FAA Icing Handbook published laboratory trials with EIDI found no fatigue damage following accelerated testing equivalent to 20-year aircraft lifetime. A similar result is likely with magnetostrictive de-icing system.
- Electromagnetic interference (EMI) - the creation and collapse of large magnetic fields due to currents of hundreds of volts may cause interference with other electronic components on the aircraft. In particular, on-board navigation, communication and control equipment, and the sensitive antennas used for electronic signal eavesdropping. Current evidence with EIDI has failed to detect "appreciable" interference with metallic aerofoils. Physical tests would be necessary to verify if electromagnetic interference is an issue. Any EMI enclosures fitted to the system are likely to add mass and therefore fuel consumption penalties, and delivered system costs to the structure.

- Lightning strikes - As the UAV wing structure is primarily composite, the proposed magnetostrictive de-icing system is a primary electrical path into the aircraft structure. Additional lighting paths may be necessary, which add further to the system mass.
- System Failure and Redundancy - for a small, pilot less aircraft, the failure of the proposed de-icing systems would not result in the loss of human life (assuming an aircraft crash would be a safe distance from humans). It however is costly in terms of financial and mission penalties. The aircraft designer may wish to consider building redundancy into the system e.g. more actuators then are necessary, by-pass/redundancy circuits and duplicated control mechanisms. However, these all add to the final system cost and the economics of the operation.
- An alternative to system redundancy is in the provision of a failure warning system. This permits the UAV operator to determine a course of action. Unlike system redundancy, provision of a failure warning system is a single "one off" cost at the development stage. It carries negligible fuel penalties, but improves aircraft life.

4.10. System Economics

Analysis in Appendix A5 indicates conservative fuel losses due to ice accretion on the existing Predator B UAV class aircraft. The study assumes the glaze ice or rime ice accretions are present as the result of a single icing cloud encounter, and it takes place after the glycol de-icing fluid has been used up.

The analysis is a first order study, where fuel consumption is assumed to be constant with respect to time.

The analysis suggests up to 44 kg of fuel is utilised in glaze ice conditions. The assumption is the aircraft acquires and continues to fly at 90% aerodynamic efficiency. By comparison, rime ice conditions are more benign with fuel losses up to 22kg of fuel.

In reality, this will vary with loading, aircraft manoeuvres, etc. The ice accreted is likely to be a mixture of both ice forms rather than the simple single phase forms examined in this report.

Fuel consumption efficiency will improve with cruise time, as the more fuel is used up and the aircraft gets lighter. However, if the wings were contaminated with ice, the drag penalties would reduce aircraft endurance, is balance by the fuel consumption improvements obtained with a lighter aircraft.

In terms of system purchase costs, it is likely that the system without redundancy is likely to be cheaper than any conventional ice protection system to implement. This statement is based on the use of consumer grade electronics and control systems, harnessing etc. Although this category of electronics does not have the quality specifications (which invariably raise the system non-recurring costs) as well as the qualification/ redundancy/ airworthiness approvals necessary for fitment to manned aircraft, it ought to be acceptable for UAV. However, no airworthiness directives prevail on this issue.

4.11. Installation

Installation of the system is best accomplished as original equipment installed at the factory. This approach is necessary as the small radius of curvature and the wing manufactured (assuming Phoenix UAV composite wing build configuration is similar for other UAV aircraft) as a one-piece item with no exterior joints. A technician would have a great deal of difficulty installing the actuators, power harnesses, and controls by hand in the field without specialised support tools.

An alternative approach is to make the leading edge section i.e. forward of the spar a separate sub-assembly to that of the main wing body. This "modular" approach has the advantage of the UAV operator having a "choice" of field fitting the leading edge with ice protection system or not. The installation can be made "fool proof" i.e. designed to prevent incorrect assembly. The ice protection controller, power/sensor/control harnesses are integrated within the mating face to allow rapid assembly/disassembly from the aircraft. Additional wing strengthening will be necessary to ensure the fit meets load requirements.

The UAV operator can fit the ice protection system when an icing threat is perceived as likely, and remove the ice protection based wing leading edge when it is considered air icing is unlikely. This improves overall fuel economics of the aircraft.

The wing leading edge itself needs to be designed with magnetostrictive de-icing in mind as part of the early considerations in the UAV wing design. In addition to the "modular" design approach described above, the wing surface structure needs to be more compliant. This can entail designing a more flexible wing leading edge to enable the actuator devices to transfer impulse forces more effectively.

A combination of physical experimentation in icing wind tunnels, and correlations/iterations using finite element analysis tools would be required to optimise actuator positions and de-icing effectiveness over a broad range of icing conditions. However, a more flexible wing leading edge will compromise structural and performance characteristics of the wing, and therefore aircraft handling. Aircraft handling may be affected by changes in the natural frequencies of the wing leading to increased vibration. Bird strike impact resistance will similarly be compromised and even reduced.

Early consideration in the design of the wing also allows the actuator design to be optimised to ensure required piston strokes perform effective de-icing. The efficiency of the device and therefore system requires secure and sound mounting arrangements.

5. CONCLUSION

The objectives of this study was to develop an aircraft ice protection system suitable for an Unmanned Aerial Vehicle by:

- Identifying a suitable low cost/ low power technology capable of protecting the UAV wing leading edge.

It was clear that an anti-icing system i.e. keeping the surface clear of ice was costly with respect to system power demands (if electro-thermal route is chosen) or limited life (if the existing fluid anti-icing system is used).

Magnetostrictive actuators were identified as a means of de-icing wing leading edges. The technology uses low cost consumer grade electronics to effect the de-icing of the wing leading edge. This contributed to lowering of system costs if airworthiness directives were to exist permitting the use of non-aircraft grade components for UAV systems.

- Demonstrating the feasibility of this low power/ low cost technology by means physical and computational simulation.

Proof of concept studies using actuator interacting with a composite test specimen suggest the actuator can function as a means of delivering an impulse to the wing surface.

FEA model developed and verified by "free-free" modal means as a good representation of the Phoenix UAV wing demonstrated as suitable platform for simulating interaction between a "virtual" magnetostrictive de-icing system and an ice contaminated wing leading edge.

The rime and glaze ice profiles that the proposed magnetostrictive de-icing system would be required to remove was confirmed as accurate representations of the contamination observed in icing wind tunnel. The aerodynamic pressures on the ice profiles, and the resulting loading on the remaining wing was also obtained.

Simulations with the virtual magnetostrictive de-icing system of pairs of actuators fitted on upper and lower surfaces at equal span distances showed wing-ice surface bond was broken for both rime and glaze ice conditions. Within the constraints of the analysis, the optimum impulse load and location was 2500N at the 7% chord position, with actuator span spacing at 0.3m for 0° angle of attack.

- Understanding the potential advantages and limitations of such a system on the effectiveness on a UAV mission.

Power demands for the system is not expected to exceed 500W at a overall system mass that is comparable to conventional electro mechanical de-icing (without redundancy). The proposed magnetostrictive de-icing system is estimated to have a system mass equivalent to the Predator B UAV fluid ice protection system (system mass 27kg + 18kg of de-icing fluid). This makes the proposed magnetostrictive system competitive in relation to the existing Predator B fluid ice protection system, without the fluid system operational time limit disadvantage.

System limitations were identified. Of particular concern was the effect on fatigue life of the base wing structure and electromagnetic interference. The former is of interest with respect to structural integrity of the wing. The latter is of issue with the sensitive electronic equipment used on the aircraft. Both have to be addressed by physical test.

6. FURTHER WORK

Computational simulations provide opportunities to rapidly assess system behaviour under a variety of environmental and physical stimuli. However, the modelling process and the quality of the results are highly dependent on the assumptions made and the input properties.

Further work should entail the installation of the proposed magnetostrictive de-icing system on an actual wing, such as the Phoenix UAV wing, and observe system function in icing wind tunnel conditions.

To undertake this physical test, further work will required to:

- ◆ Design/ develop a composite aerofoil structure with magnetostrictive devices fitted. The mechanism of attachment to the wing and efficient transfer of impulse force requires development.
- ◆ Design/ develop support equipment & test rigs are necessary to hold the wing specimen within the icing tunnel.
- ◆ Supplementary studies should investigate computational models for optimising the actuator positions at varying angles of attack, droplet diameter, water droplet density, ambient temperature conditions and air speed etc.

Additional work will be necessary for the optimisation of the actuator function.

These include changing the parameters such as:

- ◆ Fixed voltage, fixed electrical pulse width, variable current against surface displacement ($\therefore$ force imparted on surface) and ambient temperature.
- ◆ Fixed current, fixed electrical pulse width, variable voltage against surface displacement ($\therefore$ force imparted on surface) and ambient temperature.
- ◆ Study of surface ice and power optimisation through de-icing cycles against surface area to determine minimum ice shedding forces and shed ice sizes.

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Chapter 4

No new references used.

9. APPENDICES

Appendix A: Calculation Files to support CFD/ FEA studies

- A1: Determination of Rate of Ice Collection
- A2: Verification of CFD Generated Static Pressure Values
- A3: Energy-Mass Balance Check
- A4: Estimation of Flat Plate natural Frequencies
- A5: Fuel Usage Calculation For Predator B UAV
- A6: Ice-Surface Spring Constants Calculation For FEA De-Icing Model
- A7: Composite Laminate Modulus Calculation For Actuator-Flat Plate Interaction
- A8: Conversion of FEA Derived Linear Static Displacement To Equivalent Frequency Domain Actuator Displacement During De-icing.

Appendix B: Actuator-Composite Strain Gauge Data

Appendix C: MSC. NASTRAN bulk data file of FEA Wing De-Icing Models

APPENDIX A: MANUAL CALCULATIONS TO SUPPORT FEA/ CFD ASSUMPTIONS

A1: Determination of Rate of Ice Collection

Ref: SAE Aerospace Information Report AIR1667 (Jul96)

Liquid Water Content	$LWC := 1 \cdot \frac{\text{gm}}{\text{m}^3}$
Aerofoil Projected Height	$A := .11 \cdot \text{m}$
Free stream Velocity	$v_0 := 41.16 \cdot \frac{\text{m}}{\text{s}}$
Chord length	$C := 0.9144 \cdot \text{m}$
Water Droplet Median Diameter	$d_{med} := 0.05 \cdot \text{mm}$
Air Viscosity at 1800m	$\mu := 0.9688 \cdot (17.894 \times 10^{-6}) \cdot \frac{\text{kg}}{\text{m} \cdot \text{s}}$
Ambient Air Density At 1800m	$\rho := 0.8424(1.2256) \cdot \frac{\text{kg}}{\text{m}^3}$
Water Droplet Density at 0°C	$\rho_w := 1 \cdot \frac{\text{kg}}{\text{m}^3}$
Wing Span	$\text{span} := 1.57 \cdot \text{m}$
Reynolds number for the water droplet	$R := \frac{\rho \cdot d_{med} \cdot v_0}{\mu}$ $R = 122.57$
Modified Inertia Parameter	$Ko := \left[\frac{d_{med} \cdot 2 \cdot v_0 \cdot \rho_w}{C \cdot \mu} \cdot R^{\left(-\frac{2}{3}\right)} \cdot 1 \cdot \text{m} \right] \cdot \left[\left(\frac{\sqrt{6}}{R} \right) \cdot \text{atan} \left(\frac{1}{R^{\frac{1}{3}}} \right) \right]$ $Ko = 10.5$
Therefore, with reference to SAE1677 chart	
Collection Efficiency	$Em := 0.9$
Mass of water intercepted per span metre is	$W_m := v_0 \times LWC \times A \times Em$

Mass Collection appears reasonable due to span length and speed of aircraft.

$$W_m = 4.07 \frac{\text{gm}}{\text{m} \cdot \text{s}}$$

Ref: Wind Tunnel determination $C_p=1.15$ at $v_0=26.52\text{m/s}$, Angle of Attack= 0° , $R=3.3 \times 10^6$, Aspect Ratio=6, Fig. 8, NACA Report No. 647, "Tests of NACA 0009, 0012, & 008 Airfoils In The full-scale Tunnel, Goett. H.J, Kenneth Bullivant W.

A2: Verification Of CFD Generated Static Pressure Values

Ref: Aerodynamics For Engineering Students, Houghton E.L. & Carpenter P.W.
4th Ed, Butterworth Heinemann

The airspeed for the condition being considered in the CFD model was 41m/s. At this speed, the air can be considered to be incompressible. From Bernoulli's equation for incompressible flow, the Total pressure - static pressure = dynamic pressure, where

Standard Pressure at 1800m

$$P_0 := 81490 \text{ Pa}$$

$$\text{dynamic_pressure} := \frac{1}{2} \cdot \rho \cdot v_0^2$$

$$\text{dynamic_pressure} = 874.56 \text{ Pa}$$

$$\text{Static_pressure} := P_0 - \text{dynamic_pressure}$$

$$\text{Static_pressure} = 80615.44 \text{ Pa}$$

Speed of sound at 1800m:

$$\gamma := 1.4$$

$$c_w := \left(\gamma \cdot \frac{\text{Static_pressure}}{\rho} \right)^{\frac{1}{2}}$$

Mach No:

$$c = 330.63 \frac{\text{m}}{\text{s}}$$

$$M := v_0 \div c$$

$$P := \frac{P_0}{\left[1 + \left(\frac{1}{5} \cdot M^2 \right) \right]^{\frac{7}{2}}}$$

$$P = 80612.09 \text{ Pa}$$

Values appear reasonable for a first order calculation and assumptions. CFD study independently verifies Static Pressure at $8.25 \times 10^4 \text{ Pa}$

Ref: Wind Tunnel determination $C_p=1.15$ at $v_0=26.52\text{m/s}$, Angle of Attack= 0° , $R=3.3 \times 10^6$, Aspect Ratio=6, Fig. 8, NACA Report No. 647, "Tests of NACA 0009, 0012, & 008 Airfoils In The Full-Scale Tunnel, Goett. H.J, Kenneth Bullivant W.

P₀ PA2 $\gamma = 1.4$

A3: Energy-Mass Balance Check

*Ref: British Standard 36 100 Part 2, Section 3.10 (Jan74): Impact Icing
"Ice & Water Film Growth", Myers. T & Hammond. D, Int. J of Heat & Mass
Transfer Jul98*

Temperature of the surface of a body travelling through the air is increased because heat is generated in the boundary layer by the action of viscosity. In clear air, the increase in temperature above t_1 , the temperature at the edge of the boundary layer where the velocity v , is

for the case of laminar flow: (for turbulent flow, the exponent of $Pr = 1/3$)

Prandtl number at 0°C & 0.6kPa Vapour Pressure

$$Pr := 13.67$$

Specific Heat Capacity for aircraft
constant pressure

$$air_heat_capacity := 1.014 \times 10^3 \frac{J}{kg \cdot K}$$

Joules equivalent

$$Joules := \frac{12 \cdot kg \cdot m^2}{s^2 \cdot J}$$

Gravitational acceleration

$$grav := 9.81 \frac{m}{s^2}$$

Free stream Air Temperature at 1800m

$$Freestream_air_temp := 263.14 \cdot K$$

Velocity at boundary layer

$$v1 := 0.01 \frac{m}{s}$$

$$temp_difference := \sqrt[2]{\frac{1 - \left(v1^2 \cdot \frac{1 - Pr^{\frac{1}{2}}}{v0^2} \right)}{2 \cdot grav \times Joules \times air_heat_capacity}} \frac{m}{s^2}$$

$$temp_difference = 0.01 \cdot K$$

$$Surface_temperature1 := Freestream_air_temp + temp_difference$$

$$Surface_temperature1 = 263.15 \cdot K$$

- Latent Heat of Evaporation

Water Vapour Pressure at
0.01°C

$$\text{latent_evaporation} := 2248 \times 10^3 \frac{\text{J}}{\text{kg}}$$

$$\text{saturation_vapour_press} := 0.006112 \times 10^5 \text{ Pa}$$

$$\text{saturation_vapour_press} = 611.2 \text{ Pa}$$

$$\text{water_press_at_temp} := 44.4 \text{ Pa}$$

$$\text{Latent_temp} := \frac{0.622 \cdot \text{latent_evaporation} \cdot (\text{saturation_vapour_press} - \text{water_press_at_temp})}{\text{Static_pressure} \times \text{air_heat_capacity}}$$

$$\text{Latent_temp} = 9.7 \text{ K}$$

$$\text{surface_temperature} := \text{Surface_temperature1} - \text{Latent_temp}$$

$$\text{surface_temperature} = 253.45 \text{ K}$$

Energy Balance

- Heat Loss Due To Convection:

Heat transfer Coefficient Ice:

$$\text{heat_transfer_coeff} := 2.2 \cdot \frac{\text{W}}{\text{m}^2 \cdot \text{s} \cdot \text{K}}$$

$$\text{convection_heat} := \text{heat_transfer_coeff} \cdot (\text{surface_temperature} - \text{Freestream_air_temp})$$

$$\text{convection_heat} = -21.31 \frac{\text{kg}}{\text{s}^4}$$

- Heat Loss Due Sublimation

At an aerofoil temperature < 0°C, Latent heat of Sublimation

$$\text{saturated_vapour_over_ice} := 0.01 \text{ Pa}$$

$$\text{Latent_sublimation} := 23 \cdot \frac{\text{J}}{\text{kg}}$$

$$\text{ice_capacity} := 2059.91 \cdot \frac{\text{J}}{\text{kg} \cdot \text{K}}$$

$$q_s := \frac{\text{saturated_vapour_over_ice} - \text{saturation_vapour_press}}{\text{ice_capacity} \times P_0}$$

$$\text{sublimation_heat} := \text{heat_transfer_coeff} \times \text{Latent_sublimation} \times q_s$$

$$\text{sublimation_heat} = -0 \frac{\text{kg}}{\text{s}^4}$$

- Heat Loss Due To Evaporation

At an aerofoil surface temperature > 0°C, wetness factor is

$$\text{wetness} := 0.7 \quad q_e := \frac{\text{water_press_at_temp} - \text{saturation_vapour_press}}{\text{air_heat_capacity} \times P_0}$$

$$\text{evaporation_heat} := \text{heat_transfer_coeff} \times \text{latent_evaporation} \times q_e \times \text{wetness}$$

$$\text{evaporation_heat} = -23.75 \frac{\text{kg}}{\text{s}^4}$$

- Heat Loss Due To Warming of Trapped Impinging Water

Specific heat capacity of water $\text{water_heat_capacity} := 4187 \frac{\text{J}}{\text{kg} \cdot \text{K}}$

$$\text{surface_temp_change} := \text{surface_temperature} - \text{Freestream_air_temp}$$

$$\text{impingement_heat} := \frac{W_m \times \text{water_heat_capacity} \times \text{surface_temp_change} \times v_0}{\text{span} \times A}$$

$$\text{impingement_heat} = -39394.8 \frac{\text{kg}}{\text{s}^4}$$

- Heat Gained Due to Water Fusion

Latent Heat of Fusion $\text{Latent_fusion} := 3349 \frac{\text{J}}{\text{kg}}$

$$\text{fusion_heat} := W_m \cdot v_0 \cdot \frac{\text{Latent_fusion} + (\text{ice_capacity} \cdot \text{surface_temp_change})}{\text{span} \times A}$$

$$\text{fusion_heat} = -16128.92 \frac{\text{kg}}{\text{s}^4}$$

- Heat Gained Due Frictional Heating

$$\text{Recovery_factor} := 0.88$$

$$\text{friction_heat} := \frac{\text{heat_transfer_coeff} \times \text{Recovery_factor} \times v_0^2}{2 \times \text{water_heat_capacity} \times \text{Joules}}$$

$$\text{friction_heat} = 0.03 \frac{\text{kg}}{\text{s}^4}$$

• Heat Gained Due To Kinetic Energy

$$KE_Heat := \frac{Wm \cdot v^2}{2} \cdot \frac{1}{m \cdot s}$$

$$KE_Heat = 3.45 \frac{kg}{s}$$

$$Heat_loss := convention_heat + sublimation_heat + evaporation_heat$$

$$Heat_loss = -45.06 \frac{kg}{s}$$

$$Heat_gain := impingement_heat + fusion_heat + friction_heat + KE_Heat$$

$$Heat_gain = -55520.24 \frac{kg}{s}$$

$$Heat\ lost\ per\ unit\ span\ area\ per\ second \quad Total_heat := Heat_loss + Heat_gain$$

$$Total_heat = -55565.3 \frac{W}{m^2 \cdot sec}$$

Heat energy is being lost for the wing leading edge. This suggests that energy would be required within the system to bring it into equilibrium i.e. prevent ice formation. This confirms an ice protection system. The value derived using this first order calculation has used many assumptions, but appears reasonable. The statement is supported by the commercial practice literature, that an electro-thermal evaporative system would require ~43,400W/m² (Ref: FAA document DOT/FAA/CT-88/8-2 Chap III, Section 2.0, Para 2.3.3.2, power density of 28W/in²)

A4: Estimation of Flat Plate Natural Frequencies

This calculation file estimates the natural frequencies expected of the composite flat plate test coupon tested and discussed in the Main Report section 3.1. There are number of methods available within Roark referenced below for flat plates. However the "Free-Free" boundary condition precludes their use here, although it can also be argued the "simply supported" flat panel study would provide a fairer approximation.

Ref: *Roark's Formulas for Stress and Strain, Case 4a and Table 36, 6th Edition*

Using LAP software it is possible to calculate the effective stiffness i.e. Young's Modulus of the composite test coupon, using the known ply mechanical properties. The LAP study (See Addendum A) uses classical laminate theory to develop this moduli value that is here.

Test coupon dimensions and properties

Modulus of elasticity:	Modulus = $2.182 \times 10^{10} \cdot \frac{\text{N}}{\text{m}^2}$
Test Piece thickness:	thickness = 0.00168·m
Length of beam:	Length = .25·m
Gravitational acceleration:	gravity = $9.81 \cdot \frac{\text{m}}{\text{sec}^2}$ Area := Length·thickness
Area	Area := Length·thickness Area = 0 m ²
Moments of inertia	$I_1 := \frac{1}{12} \cdot \text{Length} \cdot \text{thickness}^3$ $I_1 = 0 \text{ m}^4$
Nominal uniform load:	loadw = $1 \cdot \frac{\text{N}}{\text{m}}$

The formula for f_n , the natural frequency for
the n^{th} mode of vibration, is:

$$f_n = \frac{Kapa_n}{2 \cdot \pi} \sqrt{\frac{\text{Modulus} \cdot I_1 \cdot \text{gravity}}{\text{loadw} \cdot \text{Length}^4}}$$

The values of Kappa for the first five values of n are put into a vector:

$$Kapa := \begin{pmatrix} 22.4 \\ 61.7 \\ 121 \\ 200 \\ 299 \end{pmatrix}$$

$$f := \frac{Kapa}{2 \cdot \pi} \sqrt{\frac{\text{Modulus} \cdot I_1 \cdot \text{gravity}}{\text{loadw} \cdot \text{Length}^4}}$$

$$f = \begin{pmatrix} 262.3 \\ 722.49 \\ 1416.87 \\ 2341.94 \\ 3501.2 \end{pmatrix} \text{ Hz}$$

These are the first four natural resonant frequencies:

$$f_1 = 722.49 \text{ Hz}$$

$$f_2 = 1416.87 \text{ Hz}$$

$$f_3 = 2341.94 \text{ Hz}$$

$$f_4 = 3501.2 \text{ Hz}$$

A5: Fuel Usage Calculation For Predator B UAV

Assuming straight and level flight only, then the fuel consumption for the Predator aircraft, with the existing configuration (1), with 10min rime ice accretion encounter (2) and with with 15min glaze ice accretion encounter (3):

Reference: <http://www.fas.org/irp/program/collect/uav.htm>

$$\text{Max_range} = 730 \cdot \text{km}$$

$$\text{Max_speed} = 129.64 \cdot \text{kph}$$

$$\text{Max_Take_off_Mass} = 1020 \cdot \text{kg}$$

$$\text{Aircraft_Empty_Mass} = 528 \cdot \text{kg}$$

$$\text{Aircraft_Pay_Load} = 206 \cdot \text{kg}$$

$$\text{Paraffin_density} = 0.89 \frac{\text{kg}}{\text{m}^3}$$

$$\text{Fuel_capacity} = 302 \cdot \text{kg}$$

$$\text{cruise_fuel\%} := 8\%$$

$$\text{time_on_target} = 8 \cdot \text{hr}$$

$$\text{Total_mission_time} = 9.5 \cdot \text{hr}$$

To determine the fuel costs, it is necessary to separate out fuel consumption usage throughout the flight envelope. For FAR/JAR23 size aircraft based fluid de-icing system, FAA Aircraft Icing Handbook DOT/FAA/CT-88/8-2, Chapter III, Section 3, suggests

$$\text{de_icing_fluid_mass} = 18 \cdot \text{kg}$$

$$\text{de_icing_sys_mass} = 27 \cdot \text{kg}$$

Aircraft effective mass minus the de-icing system is

$$\text{Empty_mass} := \text{Aircraft_Empty_Mass} - \text{de_icing_sys_mass}$$

$$\text{Empty_mass} = 501 \cdot \text{kg}$$

Aircraft effective payload minus the de-icing fluid is

$$\text{Effective_payload} := \text{Aircraft_Pay_Load} - \text{de_icing_fluid_mass}$$

$$\text{Effective_payload} = 188 \cdot \text{kg}$$

Assume on board electrical power is derived entirely from fuel based generation, then percentage of fuel used for power is:

$$\text{power_generation} := 15\% \times \text{Fuel_capacity}$$

$$\text{power_generation} = 45.3 \cdot \text{kg}$$

It can be considered that a fixed quantity of fuel is required at take-off and landing of a fully laden aircraft:

$$\text{take_off} := 5\% \times \text{Fuel_capacity}$$

$$\text{take_off} = 15.1 \cdot \text{kg}$$

Assuming the ratio of cruise time to target site and time at target site is

$$\text{cruise_time} := \text{Total_mission_time} - \text{time_on_target}$$

$$\text{cruise_time} = 1.5 \text{ hr}$$

$$\text{mission_time} := \text{Total_mission_time} - \text{cruise_time}$$

$$\text{mission_time} = 8 \text{ hr}$$

Average speed to and from target site, is likely to be close to maximum aircraft speed.

$$\text{cruise_distance} := \text{Max_speed} \times \text{cruise_time}$$

$$\text{cruise_distance} = 194.46 \text{ km}$$

Likely fuel usage split between cruise and mission in dry, ambient flight condition:

$$\text{fuel_for_flight} := \text{Fuel_capacity} - (\text{power_generation} + \text{take_off})$$

$$\text{fuel_for_flight} = 241.6 \text{ kg}$$

Therefore for cruise condition, % of fuel used for cruise

$$\text{cruise_fuel} := \text{cruise_fuel\%} \times \text{fuel_for_flight}$$

$$\text{cruise_fuel} = 19.33 \text{ kg}$$

and assuming icing encounter is during target surveillance/ mission; laminar flight efficiency in dry air=100%, flight efficiency in rime air=90%, flight efficiency in glaze air=80%

$$\text{flight_efficiency} := \begin{pmatrix} 100\% \\ 90\% \\ 80\% \end{pmatrix}$$

$$\text{icing_inefficiency} := 1 - \text{flight_efficiency}$$

Thus fuel available for target surveillance/ mission

$$\text{mission_fuel} := \text{flight_efficiency} \cdot (\text{fuel_for_flight} - \text{cruise_fuel})$$

$$\text{mission_fuel} = \begin{pmatrix} 222.27 \\ 200.04 \\ 177.82 \end{pmatrix} \text{ kg}$$

Dry Air Flight
Rime Ice Flight
Glaze Ice Flight

Losses due to icing conditions- dry air; rime and glaze ice respectively

$$\text{fuel_loss_due_icing} := \text{icing_inefficiency} \cdot (\text{fuel_for_flight} - \text{cruise_fuel})$$

$$\text{fuel_loss_due_icing} = \begin{pmatrix} 0 \\ 22.23 \\ 44.45 \end{pmatrix} \text{ kg}$$

Dry Air Flight
Rime Ice Flight
Glaze Ice Flight

A6: Ice-Surface Spring Constants Calculation For FEA De-Icing Model

Determination of Shear Spring and Peel Spring Constants For FEA
model Ice-Surface Adhesive Bond For UAV Wing Leading Edge.
Then Calculation of the Adhesive Bond Reserve Factor from FEA
model output.

Calculation ref:

"Finite Element Modelling of an Adhesive in a Bonded Joint"

Farhad Tahmasebi (Jul 99) <http://femci.gsfc.nasa.gov/index.html>

Ice-surface data ref: Reich & Anderson AIAA97-0303 (6 Jan 97)

Element Area $\text{Area} := 17.91 \text{ in}^2$ Ice-Adhesive Thickness $\eta := 0.1 \text{ in}$

Poisson's Ratio $\nu := 0.1$

Ice Adhesive strength after 17 removals Adhesion_variability_factor := 0.5

Adhesive Shear Modulus at -18°C $G := \left(9 \times 10^3 \frac{\text{N}}{\text{mm}^2} \right) \text{Adhesion_variability_factor}$

For All Node Positions:

Shear spring constant K_{shear} , in both lateral directions:

$$K_{\text{shear}} := \frac{\text{Area} \cdot G}{\eta}$$

$$K_{\text{shear}} = 20471130 \frac{\text{N}}{\text{mm}}$$

For Internal Node Regions

Peel spring constant k_{peel} , though thickness by estimation of the Young's Modulus E
for the ice-surface adhesive:

$$E_{\text{internal}} := 2 \cdot G \cdot (1 + \nu)$$

$$E_{\text{internal}} = 9900000000 \text{ Pa}$$

$$k_{\text{peel_internal}} := \frac{\text{Area} \cdot E_{\text{internal}}}{\eta}$$

$$k_{\text{peel_internal}} = 45036486 \frac{\text{N}}{\text{mm}}$$

For Edge Node Regions

Peel spring constant K_{edge} , is determined from the effective Young's Modulus E_{edge} where:

$$E_{edge} := \frac{E_{internal} \cdot (1 - \nu)}{1 - \nu - (2 \cdot \nu^2)}$$

$$E_{edge} = 10125000000 \text{ Pa}$$

$$K_{peel_edge} := \frac{\text{Area} \times E_{edge}}{2 \cdot \eta}$$

$$K_{peel_edge} = 23030021.25 \frac{\text{N}}{\text{mm}}$$

For Corner Node Regions

Peel spring constant K_{corner} , is determined from the effective Young's Modulus E_{corner} where:

$$E_{corner} := \frac{E_{internal}}{1 - \nu^2}$$

$$E_{corner} = 10000000000 \text{ Pa}$$

$$K_{peel_corner} := \frac{\text{Area} \times E_{corner}}{4 \cdot \eta}$$

$$K_{peel_corner} = 11372850 \frac{\text{N}}{\text{mm}}$$

These figures will now be used to bond the ice solid to the wing surface models

*A7: Composite Laminate Modulus Calculation For
Actuator-Flat Plate Interaction Study*

Study to determine the effective STIFFNESS or MODULUS of the glass-epoxy flat plate to support natural frequency calculations.

Data File : C:\My Documents\Thesis\Appendix A flat-plate-study.ld

Units : m N deg °C %

Volume fraction correction method : Rule of Mixtures

Failure tracked by : Tsai-Wu Tsai-Hill Hoffman Max.stress Max.strain

Material : DMS436_Type1

Woven 181 glass/epoxy fabric allowables.

E11	=	2.39e+010	N/m ²
E22	=	2.42e+010	N/m ²
G12	=	5.66e+009	N/m ²
ν12	=	0.194	
α11	=	1.332e-005	/°C
α22	=	1.444e-005	/°C
β11	=	0.011	
β22	=	0.011	
E resin	=	5e+008	N/m ²
ν resin	=	0.3	
Vf nom.	=	0.5	
Vf act.	=	0.5	
S11T	=	2.142e+008	N/m ²
S22T	=	1.9707e+008	N/m ²
S11C	=	3.523e+008	N/m ²
S22C	=	3.568e+008	N/m ²
S12	=	7.13e+007	N/m ²
F12	=	auto	
BFS k	=	(undefined)	
BFS Ø	=	(undefined)	
BFS β	=	(undefined)	
Density	=	1910	kg/m ³
Cost	=	(undefined)	

Safety Factors for Strength :

11 (T)	=	1.5
22 (T)	=	1.5
11 (C)	=	1.5
22 (C)	=	1.5
12 (S)	=	1.5

Selected Lay-up : Flat-Plate_Test-Coupon_Lay-up
Flat Plate Test Coupon being subject to acoustic loading from magnetostrictive actuator.
Lay-up is symmetric Total number of layers = 6
Total thickness: nominal = 0.00168 m actual = 0.00168 m
Curing Temperature = 160 °C Room Temperature = 23 °C
Unnotched Compressive Strength based on Budiansky-Fleck model
KIC = (undefined)

Layer	material name	thickness (nominal, m)	angle (deg)
- - - top - - -			
1 :	DMS436_Type1	0.00028	0
2 :	DMS436_Type1	0.00028	45
3 :	DMS436_Type1	0.00028	90
- - mid-plane - -			

Selected Loading : Unit Loading to Determine Stiffnesss (ref: Operating Conditions)
Assuming minimum operating & wet conditions (Forces & moments are unitary figures to determine effective stiffness)

			order	steps
X Load	= 10	N/m	1	1
Y Load	= 10	N/m	1	1
XY Load	= 10	N/m	1	1
X Moment	= 10	N	1	1
Y Moment	= 10	N	1	1
XY Moment	= 10	N	1	1
Top Temperature	= -40	°C		
Bottom Temperature	= -40	°C		
Moisture Content	= 3	% weight		
Allowable Strain and Curvature range :				
ex	= (undefined)			
ey	= (undefined)			
exy	= (undefined)			
lox	= (undefined)		weight =	1
ky	= (undefined)		weight =	1
loxy	= (undefined)		weight =	1
Safety Factors for failure :				
11 (T)	= 1			
22 (T)	= 1			
11 (C)	= 1			
22 (C)	= 1			
12 (S)	= 1			

ABD Matrix :

N/m
N

N
N m

3.95402e+07	1.06643e+07	-43664	0	0
1.06643e+07	3.95402e+07	-43664	0	0
-43664	-43664	1.19734e+07	0	0
0	0	0	9.38758	2.37943
0	0	0	2.37943	9.46974
0	0	0	-0.00798759	-0.00798759

Inverse ABD Matrix :

m/N
1/N

1/N
1/(N m)

2.72748e-08	-7.35617e-09	7.26386e-11	0	0
-7.35617e-09	2.72748e-08	7.26386e-11	0	0
7.26386e-11	7.26386e-11	8.35192e-08	0	0
0	0	0	0.11377	-0.0285863
0	0	0	-0.0285863	0.112782
0	0	0	0.000253192	0.000250259

Effective Stiffness :

	Membrane	Flexural	
Exx	= 2.18237e+010	2.22447e+010	N/m²
Eyy	= 2.18237e+010	2.24394e+010	N/m²
Gxy	= 7.12696e+009	6.80098e+009	N/m²
vxy	= 0.26971	0.25126	
vyx	= 0.26971	0.25346	
EIxx	=	8.7897	N m
EIyy	=	8.86662	N m
GJ	=	10.7492	N m

Expected Strength :

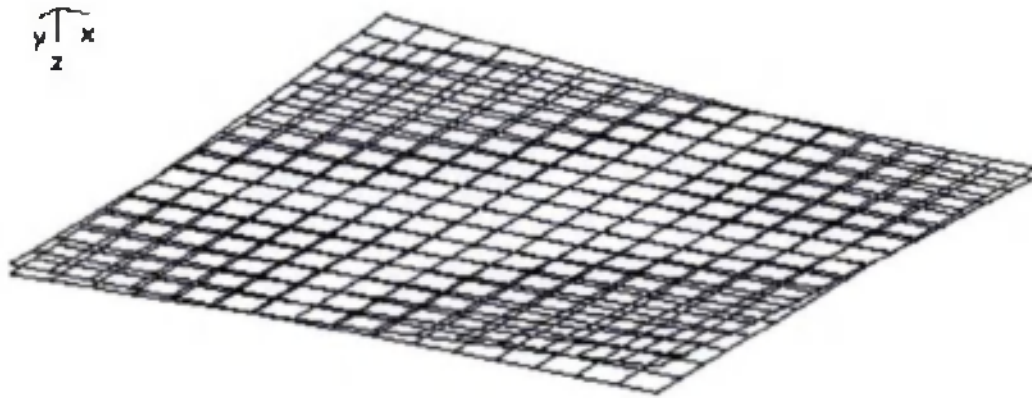
+Nx :	load at first ply failure	= 274480 N/m	(layers 1-6)
	ultimate load	= 274480 N/m	
+Ny :	load at first ply failure	= 274480 N/m	(layers 1-6)
	ultimate load	= 274480 N/m	
-Nx :	load at first ply failure	= 348078 N/m	(layers 2,5)
	ultimate load	= 377550 N/m	
-Ny :	load at first ply failure	= 348078 N/m	(layers 2,5)
	ultimate load	= 377550 N/m	
Nxy :	load at first ply failure	= 149805 N/m	(layers 1-6)
	ultimate load	= 149805 N/m	

Displaced shape

Components : Curing + Hygro + Thermal + Mechanical

Scaled displacements for a 1 m square element.

Magnification factor : 0.0537 View Angles : H 60° V -70°



A8: Conversion of FEA Derived Linear Static Displacement To Equivalent Frequency Domain Actuator Displacement During De-Icing Study

With reference to "Building Better Products with FEA", Vince Adams & Abraham Askenazi, equations for "simple harmonic motion" are:

For a given frequency F and time period T,

- Displacement = amplitude $\times \sin(2\pi fT)$
- Velocity = $\frac{d}{dT}$ Displacement = amplitude $\times (2\pi f) \times \cos(2\pi fT)$
- Acceleration = $\frac{d^2}{dT^2}$ Displacement = amplitude $\times (2\pi f)^2 \times \sin(2\pi fT)$

where amplitude = peak to peak displacement of wing surface.

The MAXIMUM value of displacement, velocity & acceleration occur when the trigonometric function are equal to 1 (unity). Then the above equations become:

- Max Displacement = amplitude
- Max Velocity = $2\pi f \times \text{amplitude}$
- Acceleration = $(2\pi f)^2 \times \text{amplitude}$

The RMS Power Density plot displayed in Chapter 3, Figure 12 shows that at the actuator is able to apply a maximum impulse force from 100Hz to 700Hz. If we consider the 4 actuator applying 2500N at 3 possible actuation frequencies of 200Hz, 400Hz and 600Hz, the equivalent displacement in the frequency domain can be calculated below.

$n := 1..3$ $no_of_actuators := 4$ $actuator_force := 2500 \cdot N$

Frequency range being considered:

$frequency_n := n \cdot 200 \cdot Hz$

$frequency_n =$ Hz

200
400
600

Chapter 3 considered a wing exposed to rime ice conditions for 10minutes and glaze ice exposure for 15minutes.

$$p := 1..3$$

$$\text{exposure_time}_p := 5 \cdot p \cdot \text{min}$$

$$\text{exposure_time}_p =$$

5	min
10	
15	

Assuming 90% conversion of intercepted water droplets into ice, Appendix A1 estimated the rate of mass of water intercepted per span metre as

$$W_m = 4.07 \frac{\text{gm}}{\text{m}\cdot\text{s}} \quad \text{for a wing span of} \quad \text{span} = 1.57 \text{ m}$$

Thus , the ice mass accreted against a 5, 10 and 15minute exposure is:

$$\text{icemass}_{\text{exposure_time}_p} := W_m \cdot \text{span} \cdot \text{exposure_time}_p$$

$$\text{icemass}_{\text{exposure_time}_p} =$$

1.92	kg
3.84	
5.76	

If the ice needed to be removed by the actuator is equally shared between the 4 actuators, the the mass of ice to be removed by each actuator is:

$$\text{mass_of_ice}_{\text{exposure_time}_p} := \frac{\text{icemass}_{\text{exposure_time}_p}}{\text{no_of_actuators}}$$

From Newton's Laws: acceleration=Force/mass, then

$$\text{acceleration}_{\text{exposure_time}_p} := \frac{\text{actuator_force}}{\text{mass_of_ice}_{\text{exposure_time}_p}}$$

Adams & Askenazi have shown the relationship in the frequency domain between amplitude and acceleration and frequency. Therefore,

$$\text{amplitude}_{\text{frequency}_n, \text{exposure_time}_p} := \frac{\text{acceleration}_{\text{exposure_time}_p}}{(2 \cdot \pi \cdot \text{frequency}_n)^2}$$

As amplitude is twice the displacement from the neutral point, then

displacement

frequency

exposure_time

p

=

amplitude

frequency

exposure_time

p

2

Hence for a 2500N actuation force at 200Hz, the actuator displacement against exposure times of 5min, 10min & 15min is:

displacement

200

exposure_time

p

=

1.65	mm
0.82	
0.55	

Similarly at 400Hz:

displacement

400

exposure_time

p

=

0.41	mm
0.21	
0.14	

And at 600Hz

displacement

600

exposure_time

p

=

0.18	mm
0.09	
0.06	

IF the de-icing system is activated earlier into the icing encounter say, 5minutes, then actuation force at higher frequencies translates into larger surface displacements than at say 15minutes.

The mechanical advantage required of the actuator mechanism becomes proportionately smaller at higher frequencies. This in turn could allow the device to be of a more discrete geometry and therefore more suited to the wing de-icing application.

Appendix B.xls- Example Strain Output

nCode data file converted by program DTA from file :
Total number of points = 1024
Sample rate = 0.170667
X-axis base value = 0; X-axis units = Hz.
X-axis increment = 5.85938
Y-axis label = RMS Power; axis units = $\mu\text{E}^2 \cdot \text{Hz}^{-1}$

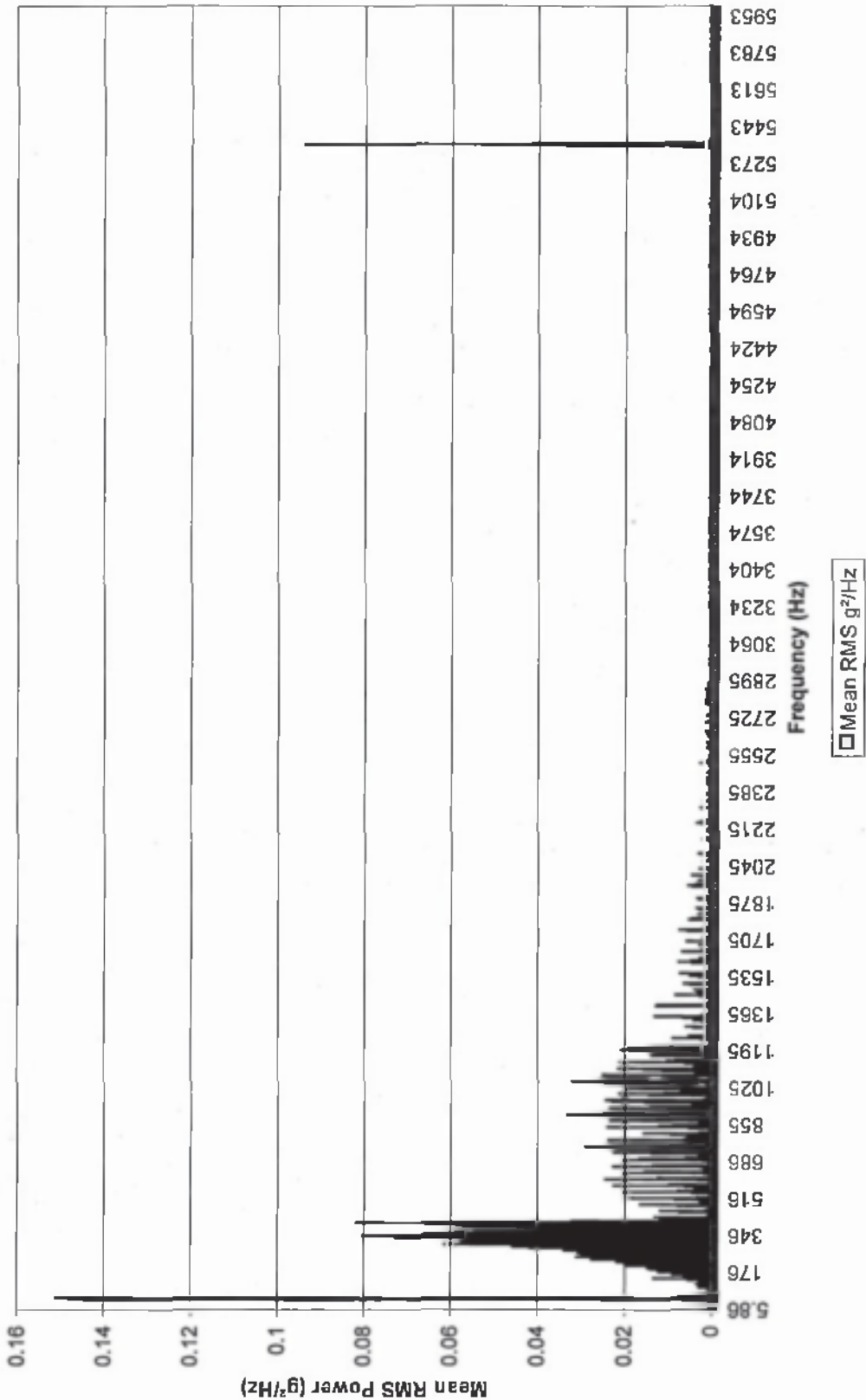
RMS Power					
Frequency	Test01	Test02	Test03	Test04	Test05
0	0.000125035	7.58811E-05	0.00004669	7.19984E-05	7.20E-05
5.85938	0.00022494	0.000134329	0.000135334	0.000161402	0.000161402
11.7188	0.000126231	9.69574E-05	0.000128805	0.000128828	0.000128828
17.5781	0.000119551	0.000104	0.000123323	0.000128415	0.000128415
23.4375	0.000134712	0.000103072	0.000126823	0.000109322	0.000109322
29.2969	0.000146387	0.00012446	0.000138494	0.000153459	0.000153459
35.1563	0.000271578	0.000204899	0.000246831	0.000208921	0.000208921
41.0156	0.00593041	0.00605713	0.00623022	0.00598614	0.00598614
46.875	0.149933	0.148845	0.149379	0.147032	0.147032
52.7344	0.156953	0.154213	0.153881	0.152277	0.152277
58.5938	0.00725867	0.00710196	0.0071286	0.00688076	0.00688076
64.4531	0.000293705	0.000315361	0.000212756	0.000253954	0.000253954
70.3125	0.000158689	0.000174054	0.000203604	0.00016755	0.00016755
76.1719	0.000187791	0.000108724	0.000117571	0.000124641	0.000124641
82.0313	0.000192631	9.63296E-05	0.000102471	9.68695E-05	9.69E-05
87.8906	0.000219847	0.000106304	9.61938E-05	0.000102735	0.000102735
93.75	0.0154061	0.000200525	0.000274987	0.000202519	0.000202519
99.6094	0.0675005	0.000870813	0.000486076	0.000399899	0.000399899
105.469	0.0191368	0.0417625	0.000282406	0.000285355	0.000285355
111.328	0.000105903	0.0787599	0.00480978	0.000141228	0.000141228
117.188	0.000107146	0.00880512	0.0877077	7.64743E-05	7.65E-05
123.047	0.000135134	0.000143837	0.0763234	0.000202821	0.000202821
128.906	0.000121775	0.000147537	0.00273714	0.000513246	0.000513246
134.766	9.65366E-05	0.000147648	0.000180293	0.078322	0.078322
140.625	9.64819E-05	0.000168715	0.000169522	0.198181	0.198181
146.484	0.000246254	0.000308382	0.000213742	0.0313302	0.0313302
152.344	0.000340475	0.000316677	0.000222317	0.000363397	0.000363397
158.203	0.000202189	0.000143325	0.000136999	0.000189286	0.000189286
164.063	0.000154179	0.0001225	0.000115586	0.000134969	0.000134969
169.922	0.000104004	0.00010881	0.00010733	0.000126289	0.000126289
175.781	0.000129496	0.000106052	0.000143422	0.000131675	0.000131675
181.641	0.000139816	0.000117452	0.000215074	0.000131434	0.000131434

Appendix B.xls- Example Strain Output

nCode data file converted by program DTA from file :
Total number of points = 1024
Sample rate = 0.170667
X-axis base value = 0; X-axis units = Hz.
X-axis increment = 5.85938
Y-axis label = RMS Power; axis units = $\mu E^2 \cdot Hz^{-1}$

Test24	Test25	Test26	Test27	Test28	Test29
0	0	0.000408725	0.000303016	0.000251681	0
0	0	0	0	0.000278619	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0.000557897	0
0.00609945	0.00620138	0.0061377	0.0061679	0.0063981	0.00640329
0.14448	0.150534	0.145263	0.147127	0.144686	0.147594
0.146761	0.153402	0.147789	0.149434	0.146604	0.148579
0.00661053	0.00689721	0.00656302	0.00665979	0.00647213	0.00651976
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0.000298988	0
0.000559852	0.000615326	0.000701411	0.000461127	0.000400677	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0
0	0	0	0	0	0.000498064
0	0	0	0	0	0.000552334
0	0	0	0	0	0
0	0.000837659	0	0.00052202	0	0
0	0	0	0.00116365	0.000286912	0
0	0	0	0.00043159	0	0
0	0	0	0	0.00114902	0
0	0	0	0	0.000619096	0
0	0	0	0	0	0
0	0	0	0	0	0

Mean RMS Power Against Frequency



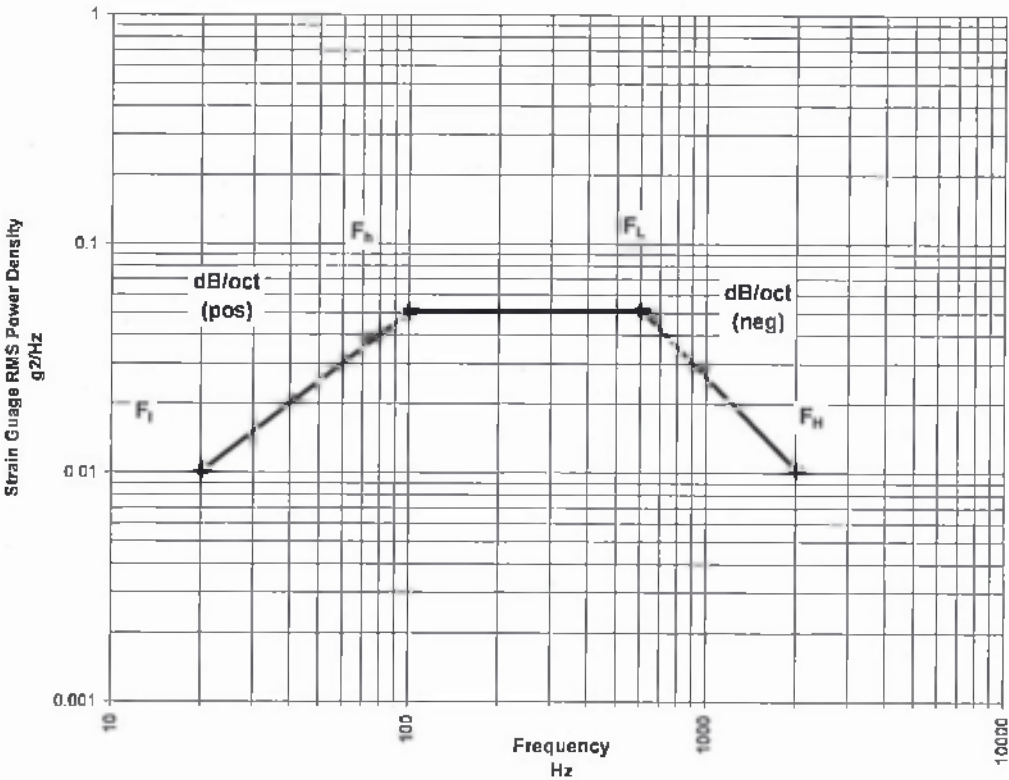
Appendix B xls/Freq v Avg

Frequency	Mean RMS						Slope	Acceleration
Hz	g ² /Hz	FREQ(Hz)	ASD(G ² /Hz)	dB	OCT	dB/OCT	AREA	Grms
5.85938	0.000112	5.85938	0.000112	*	*	*	*	*
11.7188	5.01E-05	11.7188	5.01E-05	-3.49	1.00	-3.49	0.00	0.02
17.5781	6.67E-05	17.5781	6.67E-05	1.24	0.58	2.13	0.00	0.03
23.4375	5.32E-05	23.4375	5.32E-05	-0.99	0.42	-2.37	0.00	0.03
29.2969	5.88E-05	29.2969	5.88E-05	0.44	0.32	1.36	0.00	0.04
35.1563	0.000214	35.1563	0.000214	5.62	0.26	21.37	0.00	0.05
41.0156	0.00621	41.0156	0.00621	14.62	0.22	65.73	0.01	0.11
46.875	0.147627	46.875	0.147627	13.76	0.19	71.43	0.28	0.53
52.7344	0.150758	52.7344	0.150758	0.09	0.17	0.54	1.16	1.08
58.5938	0.006808	58.5938	0.006808	-13.45	0.15	-88.50	1.42	1.19
64.4531	0.000158	64.4531	0.000158	-16.34	0.14	-118.82	1.43	1.20
70.3125	5.36E-05	70.3125	5.36E-05	-4.70	0.13	-37.45	1.43	1.20
76.1719	0.000109	76.1719	0.000109	3.10	0.12	26.85	1.43	1.20
82.0313	8.43E-05	82.0313	8.43E-05	-1.13	0.11	-10.58	1.43	1.20
87.8906	4.6E-05	87.8906	4.6E-05	-2.63	0.10	-26.42	1.43	1.20
93.75	0.000646	93.75	0.000646	11.47	0.09	123.20	1.44	1.20
99.6094	0.002599	99.6094	0.002599	6.05	0.09	69.14	1.44	1.20
105.469	0.002082	105.469	0.002082	-0.96	0.08	-11.68	1.46	1.21
111.328	0.002748	111.328	0.002748	1.21	0.08	15.45	1.47	1.21
117.188	0.003168	117.188	0.003168	0.62	0.07	8.34	1.49	1.22
123.047	0.002526	123.047	0.002526	-0.98	0.07	-13.98	1.51	1.23
128.906	0.000196	128.906	0.000196	-11.10	0.07	-165.42	1.51	1.23
134.766	0.005116	134.766	0.005116	14.17	0.06	220.88	1.52	1.23
140.625	0.012945	140.625	0.012945	4.03	0.06	65.66	1.57	1.25
146.484	0.005947	146.484	0.005947	-3.38	0.06	-57.36	1.62	1.27
152.344	0.005517	152.344	0.005517	-0.33	0.06	-5.77	1.66	1.29
158.203	0.00863	158.203	0.00863	1.94	0.05	35.69	1.70	1.30
164.063	0.007922	164.063	0.007922	-0.37	0.05	-7.08	1.75	1.32
169.922	0.011918	169.922	0.011918	1.77	0.05	35.04	1.80	1.34
175.781	0.01023	175.781	0.01023	-0.66	0.05	-13.56	1.87	1.37
181.641	0.013561	181.641	0.013561	1.22	0.05	25.87	1.94	1.39
187.5	0.014411	187.5	0.014411	0.26	0.05	5.77	2.02	1.42
193.359	0.013741	193.359	0.013741	-0.21	0.04	-4.66	2.10	1.45
199.219	0.019319	199.219	0.019319	1.48	0.04	34.35	2.20	1.48
205.078	0.014628	205.078	0.014628	-1.21	0.04	-28.88	2.30	1.52
210.938	0.019846	210.938	0.019846	1.32	0.04	32.60	2.40	1.55
216.797	0.021472	216.797	0.021472	0.34	0.04	8.65	2.52	1.59
222.656	0.020375	222.656	0.020375	-0.23	0.04	-5.92	2.64	1.62
228.516	0.027301	228.516	0.027301	1.27	0.04	33.91	2.78	1.67
234.375	0.020424	234.375	0.020424	-1.26	0.04	-34.51	2.92	1.71
240.234	0.030725	240.234	0.030725	1.77	0.04	49.79	3.07	1.75
246.094	0.026538	246.094	0.026538	-0.64	0.03	-18.30	3.23	1.80
251.953	0.029671	251.953	0.029671	0.48	0.03	14.28	3.40	1.84
257.813	0.030355	257.813	0.030355	0.10	0.03	2.98	3.57	1.89
263.672	0.027965	263.672	0.027965	-0.36	0.03	-10.98	3.74	1.93
269.531	0.033603	269.531	0.033603	0.80	0.03	25.15	3.92	1.98

Appendix B.xls/Freq v Avg

275.391	0.030604	275.391	0.030604	-0.41	0.03	-13.08	4.11	2.03
281.25	0.041419	281.25	0.041419	1.31	0.03	43.27	4.32	2.08
287.109	0.045394	287.109	0.045394	0.40	0.03	13.38	4.58	2.14
292.969	0.044407	292.969	0.044407	-0.10	0.03	-3.27	4.84	2.20
298.828	0.061028	298.828	0.061028	1.38	0.03	48.33	5.14	2.27
304.688	0.048588	304.688	0.048588	-0.99	0.03	-35.33	5.46	2.34
310.547	0.057187	310.547	0.057187	0.71	0.03	25.75	5.77	2.40
316.406	0.050548	316.406	0.050548	-0.54	0.03	-19.87	6.09	2.47
322.266	0.056044	322.266	0.056044	0.45	0.03	16.93	6.40	2.53
328.125	0.072378	328.125	0.072378	1.11	0.03	42.73	6.78	2.60
333.984	0.066849	333.984	0.066849	-0.35	0.03	-13.52	7.18	2.68
339.844	0.079852	339.844	0.079852	0.77	0.03	30.76	7.61	2.76
345.703	0.065546	345.703	0.065546	-0.86	0.02	-34.77	8.04	2.83
351.563	0.058285	351.563	0.058285	-0.51	0.02	-21.03	8.40	2.90
357.422	0.054089	357.422	0.054089	-0.32	0.02	-13.61	8.73	2.95
363.281	0.039499	363.281	0.039499	-1.37	0.02	-58.20	9.00	3.00
369.141	0.035396	369.141	0.035396	-0.48	0.02	-20.63	9.22	3.04
375	0.042629	375	0.042629	0.81	0.02	35.54	9.45	3.07
380.859	0.056682	380.859	0.056682	1.24	0.02	55.33	9.74	3.12
386.719	0.058065	386.719	0.058065	0.10	0.02	4.75	10.07	3.17
392.578	0.065449	392.578	0.065449	0.52	0.02	23.96	10.43	3.23
398.438	0.08139	398.438	0.08139	0.95	0.02	44.29	10.86	3.30
404.297	0.031745	404.297	0.031745	-4.09	0.02	-194.16	11.17	3.34
410.156	0.000601	410.156	0.000601	-17.23	0.02	-829.87	11.22	3.35
416.016	0.011478	416.016	0.011478	12.81	0.02	625.80	11.24	3.35
421.875	0.01269	421.875	0.01269	0.44	0.02	21.61	11.31	3.36
427.734	0.000701	427.734	0.000701	-12.58	0.02	-632.17	11.33	3.37
433.594	0.000129	433.594	0.000129	-7.35	0.02	-374.38	11.33	3.37
439.453	0.000411	439.453	0.000411	5.04	0.02	260.13	11.34	3.37
445.313	0.007158	445.313	0.007158	12.40	0.02	649.06	11.35	3.37
451.172	0.011439	451.172	0.011439	2.04	0.02	107.98	11.40	3.38
457.031	0.001093	457.031	0.001093	-10.20	0.02	-547.74	11.43	3.38
462.891	0.000114	462.891	0.000114	-9.82	0.02	-534.34	11.43	3.38
468.75	0.000138	468.75	0.000138	0.84	0.02	46.48	11.43	3.38
474.609	0.007451	474.609	0.007451	17.31	0.02	966.03	11.44	3.38
480.469	0.016142	480.469	0.016142	3.36	0.02	189.64	11.51	3.39
486.328	0.002164	486.328	0.002164	-8.73	0.02	-499.12	11.55	3.40
492.188	0.000395	492.188	0.000395	-7.39	0.02	-427.59	11.56	3.40
498.047	0.002653	498.047	0.002653	8.27	0.02	484.67	11.56	3.40
503.906	0.008112	503.906	0.008112	4.85	0.02	287.68	11.59	3.40
509.766	0.018107	509.766	0.018107	3.49	0.02	209.04	11.66	3.42
515.625	0.003007	515.625	0.003007	-7.80	0.02	-472.90	11.71	3.42
521.484	7.23E-05	521.484	7.23E-05	-16.19	0.02	-993.10	11.72	3.42
527.344	5.75E-05	527.344	5.75E-05	-1.00	0.02	-61.74	11.72	3.42
533.203	0.004168	533.203	0.004168	18.60	0.02	1166.94	11.72	3.42
539.063	0.020076	539.063	0.020076	6.83	0.02	432.95	11.78	3.43
544.922	0.007427	544.922	0.007427	-4.32	0.02	-276.91	11.86	3.44
550.781	0.001993	550.781	0.001993	-5.71	0.02	-370.34	11.88	3.45
556.641	0.003558	556.641	0.003558	2.52	0.02	164.88	11.90	3.45
562.5	0.005755	562.5	0.005755	2.09	0.02	138.29	11.93	3.45

RMS Power Density Conversion To Acceleration G_{RMS}



• NOTE: All values in the AREA column are cumulative and therefore the Gms values and the AREA values are for the area under the entire curve

\$ NASTRAN input file created by the MSC MSC.Nastran input file
 \$ translator (MSC.Patran 12.1.060) on November 27, 2004 at 11:09:11.
 \$ Direct Text Input for File Management Section
 \$ Normal Modes Analysis, Database

SOL 103

TIME 600

\$ Direct Text Input for Executive Control

CEND

SEALL = ALL

SUPER = ALL

TITLE = MSC.Nastran job created on 15-Jul-04 at 11:57:17

ECHO = NONE

MAXLINES = 99999999

\$ Direct Text Input for Global Case Control Data

SUBCASE 1

\$ Subcase name : Free-free

SUBTITLE=Free-free

METHOD = 1

VECTOR(PLOT, SORT2, REAL) = ALL

STRAIN(PLOT, SORT2, REAL, VONMISES, FIBER, SGAGE) = ALL

\$ Direct Text Input for this Subcase

BEGIN BULK

PARAM POST 0

PARAM AUTOSPC YES

PARAM COUPWASS 1

PARAM WTMASS 9800.

PARAM SNORM 20.

PARAM PRTHMAXIM YES

BEGR 1 1.

40 0

\$ Direct Text Input for Bulk Data

\$ Elements and Element Properties for region : spar

PSHELL 1 1 1.

1 1

\$ Pset: "spar" will be imported as: "psHELL.1"

CQUAD4 1992 1 750 1839 1844 751

CQUAD4 1993 1 1839 1840 1845 1844

CQUAD4 1994 1 1840 1841 1846 1845

CQUAD4 1995 1 1841 1799 788 1846

CQUAD4 1996 1 751 1844 1849 752

CQUAD4 1997 1 1844 1845 1850 1849

CQUAD4 1998 1 1845 1846 1851 1850

CQUAD4 1999 1 1846 788 789 1851

CQUAD4 2000 1 752 1849 1854 753

MASS

1
PAGE 2

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0 NASTRAN EXECUTIVE CONTROL ECHO

SOL 103
TIME 600
\$ DIRECT TEXT INPUT FOR EXECUTIVE CONTROL
CEND

1 MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17
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NOVEMBER 27, 2004 MSC.NASTRAN 9/15/04

0
0 CASE CONTROL ECHO

COMMAND	COUNT
1 SEALL = ALL	
2 SUPER = ALL	
3 TITLE = MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17	
4 ECHO = NONE	
5 MAXLINES = 999999999	
6 \$ DIRECT TEXT INPUT FOR GLOBAL CASE CONTROL DATA	
7 SUBCASE 1	
8 \$ SUBCASE NAME : FREE-FREE	
9 SUBTITLE=FREE-FREE	
10 METHOD = 1	
11 VECTOR(PLOT, SORT2, REAL) =ALL	
12 STRAIN(PLOT, SORT2, REAL, VONMISES, FIBER, SCAGE) =ALL	
13 \$ DIRECT TEXT INPUT FOR THIS SUBCASE	
14 BEGIN BULK	
INPUT BULK DATA ENTRY COUNT = 4126	
TOTAL COUNT= 4108	

0
0

MODEL SUMMARY

NUMBER OF GRID	POINTS = 1254
NUMBER OF CQUAD4	ELEMENTS = 1270
NUMBER OF CTRIA3	ELEMENTS = 10

1 MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17
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0 *** USER INFORMATION MESSAGE 7555 (GMTSTD)
FINITE ELEMENT GEOMETRY CHECK RESULTS EXCEED TOLERANCE LEVELS FOR THE FOLLOWING ELEMENTS. METRIC VALUES THAT EXCEED
TEST TOLERANCE LIMITS ARE IDENTIFIED BY ONE OF THE FOLLOWING FLAGS PLACED TO THE RIGHT OF THE METRIC VALUE.
"++++" FOR TEST RESULTS EXCEEDING TOLERANCES. INFORMATIONAL ONLY. PROBLEM SOLUTION CONTINUES. (DEFAULT FLAG)
"IGNR" FOR TEST RESULTS EXCEEDING TOLERANCES. INFORMATIONAL ONLY. PROBLEM SOLUTION CONTINUES.
"WARN" FOR TEST RESULTS EXCEEDING TOLERANCES. INFORMATIONAL ONLY. PROBLEM SOLUTION CONTINUES.
"FAIL" FOR TEST RESULTS EXCEEDING TOLERANCES. SEVERE ERROR. PROBLEM SOLUTION TERMINATES.
USER ACTION: USE THE GEOMCHECK (EXECUTIVE CONTROL STATEMENT) KEYWORD-VALUE TO CHANGE TOLERANCE VALUES IF DESIRED.
A MAXIMUM OF 100 SKEW ANGLE (SA) TOLERANCE LIMIT VIOLATIONS WILL BE IDENTIFIED BY THE FLAG "++++"
A MAXIMUM OF 100 MAX INT. ANGLE (IA) TOLERANCE LIMIT VIOLATIONS WILL BE IDENTIFIED BY THE FLAG "++++"
1 MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17 NOVEMBER 27, 2004 MSC.NASTRAN 9/15/04
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0 TOLERANCE LIMITS ARE: SKEW = 10.00, IA(MAX) = 160.00 (++++ = LIMIT VIOLATED)
ELEMENT TYPE ID SKEW ANGLE MAX INT. ANGLE
TRIA3 2154 7.78 +++++ 90.00
TRIA3 2155 7.78 +++++ 90.01
TRIA3 2238 7.78 +++++ 91.77
TRIA3 2239 7.79 +++++ 88.43
1 MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17 NOVEMBER 27, 2004 MSC.NASTRAN 9/15/04
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E L E M E N T G E O M E T R Y T E S T R E S U L T S S U M M A R Y									
ELEMENT TYPE DETERMINANT	SKEW ANGLE	TAPER RATIO	INTER. ANGLE	INTER. ANGLE	MAXIMUM SURFACE/FACE	WARP FACTOR	OFFSET RATIO	LENGTH RATIO	EDGE POINT
QUAD4	0	0	0	0	0	0	N/A	N/A	N/A
N/A									
TRIA3	4	N/A	N/A	0	0	N/A	N/A	N/A	N/A
N/A									

N/A IN THE ABOVE TABLE INDICATES TESTS THAT ARE NOT APPLICABLE TO THE ELEMENT TYPE AND WERE NOT PERFORMED.

FOR ALL ELEMENTS WHERE GEOMETRY TEST RESULTS HAVE EXCEEDED TOLERANCES,

1 TRIA3 ELEMENT ID 2238 PRODUCED SMALLEST SKEW ANGLE OF 7.78 (TOLERANCE = 10.00).
MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17
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0 *** USER INFORMATION MESSAGE 7310 (VRCPRN)
ORIGIN OF SUPERELEMENT BASIC COORDINATE SYSTEM WILL BE USED AS REFERENCE LOCATION.
RESULTANTS ABOUT ORIGIN OF SUPERELEMENT BASIC COORDINATE SYSTEM IN SUPERELEMENT BASIC SYSTEM COORDINATES.

SUBCASE/		LOAD		T1		T2		T3		R1		R2		R3	
DAREA	ID	TYPE													
0	1	FX	0.000000E+00	----	----	----	----	----	----	----	----	0.000000E+00	----	0.000000E+00	----
		FY	----	0.000000E+00	----	----	----	0.000000E+00	----	0.000000E+00	----	----	----	0.000000E+00	----
		FZ	----	----	----	----	0.000000E+00	0.000000E+00	----	0.000000E+00	----	0.000000E+00	----	----	----
		MX	----	----	----	----	----	0.000000E+00	----	0.000000E+00	----	----	----	----	----
		MY	----	----	----	----	----	----	----	----	----	0.000000E+00	----	----	----
0		MZ	----	----	----	----	----	----	----	----	----	----	----	0.000000E+00	----
		TOTALS	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00
		*** SYSTEM INFORMATION MESSAGE 6916 (DFMSYN)													
		DECOMP ORDERING METHOD CHOSEN: BEND, ORDERING METHOD USED: BEND													
		*** USER INFORMATION MESSAGE 5010 (LNCILD)													
0		STURM SEQUENCE DATA FOR EIGENVALUE EXTRACTION.													
		TRIAL EIGENVALUE = 4.095786D+02, CYCLES = 3.220985D+00 NUMBER OF EIGENVALUES BELOW THIS VALUE = 6													
		*** USER INFORMATION MESSAGE 5010 (LNCILD)													
		STURM SEQUENCE DATA FOR EIGENVALUE EXTRACTION.													
		TRIAL EIGENVALUE = 3.947842D+01, CYCLES = 1.000000D+00 NUMBER OF EIGENVALUES BELOW THIS VALUE = 5													
0		*** USER INFORMATION MESSAGE 5010 (LNCILD)													
		STURM SEQUENCE DATA FOR EIGENVALUE EXTRACTION.													
		TRIAL EIGENVALUE = 2.038010D+06, CYCLES = 2.272078D+02 NUMBER OF EIGENVALUES BELOW THIS VALUE = 46													
		*** USER INFORMATION MESSAGE 5010 (LNCILD)													
		STURM SEQUENCE DATA FOR EIGENVALUE EXTRACTION.													

TABLE OF SHIFTS: (LNNRIGL)		FREQUENCY, CYCLES		# EIGENVALUES BELOW		# NEW EIGENVALUES FOUND	
SHIFT #	SHIFT VALUE						
1.	4.0957861E+02	3.2209854E+00		6		36	
2.	3.9478416E+01	9.9999994E-01		5		0	
3.	2.0380101E+06	2.2720782E+02		46		5	

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5.786181E+05	5.954559E+05	7.716579E+02	1.228132E+02	1.000000E+00
11 11				
5.954559E+05	7.027394E+05	8.382955E+02	1.334189E+02	1.000000E+00
12 12				
7.027394E+05	7.691696E+05	8.770231E+02	1.395826E+02	1.000000E+00
13 13				
7.691696E+05	7.740764E+05	8.798161E+02	1.400271E+02	1.000000E+00
14 14				
7.740764E+05	7.838504E+05	8.853533E+02	1.409083E+02	1.000000E+00
15 15				
7.838504E+05	8.093916E+05	8.996619E+02	1.431856E+02	1.000000E+00
16 16				
8.093916E+05	8.398961E+05	9.164584E+02	1.458589E+02	1.000000E+00
17 17				
8.398961E+05	9.244389E+05	9.614775E+02	1.530239E+02	1.000000E+00
18 18				
9.244389E+05	1.059822E+06	1.029476E+03	1.638463E+02	1.000000E+00
19 19				
1.059822E+06	1.081024E+06	1.039723E+03	1.654770E+02	1.000000E+00
20 20				
1.081024E+06	1.101813E+06	1.049673E+03	1.670606E+02	1.000000E+00
21 21				
1.101813E+06	1.125306E+06	1.060804E+03	1.688323E+02	1.000000E+00
22 22				
1.125306E+06	1.166296E+06	1.079952E+03	1.718796E+02	1.000000E+00
23 23				
1.166296E+06	1.220415E+06	1.104724E+03	1.758222E+02	1.000000E+00
24 24				
1.220415E+06	1.315314E+06	1.146871E+03	1.825303E+02	1.000000E+00
25 25				
1.315314E+06	1.330925E+06	1.153657E+03	1.836102E+02	1.000000E+00
26 26				
1.330925E+06	1.401763E+06	1.183961E+03	1.884332E+02	1.000000E+00
27 27				
1.401763E+06	1.448899E+06	1.203702E+03	1.915751E+02	1.000000E+00
28 28				
1.448899E+06	1.462911E+06	1.209509E+03	1.924993E+02	1.000000E+00
29 29				
1.462911E+06	1.511390E+06	1.229386E+03	1.956629E+02	1.000000E+00
30 30				
1.511390E+06	1.589516E+06	1.260760E+03	2.006562E+02	1.000000E+00
31 31				
1.589516E+06	1.607291E+06	1.267790E+03	2.017750E+02	1.000000E+00
32 32				

1.607291E+06	1.640658E+06	1.280982E+03	2.038587E+02	1.000000E+00
33	33			
1.640658E+06	1.657702E+06	1.287518E+03	2.049148E+02	1.000000E+00
34	34			
1.657702E+06	1.704630E+06	1.305615E+03	2.077951E+02	1.000000E+00
35	35			
1.704630E+06	1.800306E+06	1.341755E+03	2.135469E+02	1.000000E+00
36	36			
1.800306E+06	1.885603E+06	1.373173E+03	2.185472E+02	1.000000E+00
37	37			
1.885603E+06	1.932238E+06	1.390049E+03	2.212332E+02	1.000000E+00
38	38			
1.932238E+06	1.949820E+06	1.396359E+03	2.222375E+02	1.000000E+00
39	39			
1.949820E+06	1.998703E+06	1.413755E+03	2.250061E+02	1.000000E+00
40	40			
1.998703E+06				

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SUBCASE

1 MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17

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SUBCASE

MAXIMUM DISPLACEMENTS

SUBCASE/ DAREA ID	T1	T2	T3	R1	R2	R3
0	1	5.2408042E+00	9.1076433E-04	5.5686478E+00	9.8564348E+00	1.0334483E+01
0	2	7.6519632E-01	1.0229798E+01	1.9155750E-01	1.4094098E-01	1.0184740E+00
0	3	8.8299239E-01	1.0519322E+01	3.2346073E-01	6.7376337E+00	3.2511215E+00
0	4	3.1248546E-01	9.4251003E+00	5.8490497E-01	1.1758414E+01	6.2527781E+00
0	5	6.4456958E-01	1.0972821E+01	2.8534502E-01	4.2193145E-01	1.8827344E+00
0	6	1.2065343E+00	1.1433287E+01	8.2375288E-02	2.7356315E-01	1.2110318E+00
0	7	7.1254224E-01	1.190104E+01	3.9236268E-01	3.0572324E+00	1.3122472E+00
0	8	7.6331574E-01	1.2993406E+01	1.1011273E-01	1.9604263E+00	2.5882668E+00
0	9	1.0538906E+00	1.2104548E+01	9.2373729E-02	1.7024185E-01	1.1662104E+00
0	10	1.3140578E+00	1.1512202E+01	1.9902369E-01	5.2468455E-01	1.5393500E+00
0	11	1.7206181E+00	1.2512876E+01	1.5694772E-01	3.7306330E+00	1.6731527E+00
0	12	1.4139779E+00	1.2071901E+01	5.0041592E-01	1.0395486E+01	5.2059002E+00
0	13	1.6659369E+00	1.1454293E+01	5.0138909E-01	3.3036716E+00	1.8211915E+00
0	14	1.2935158E+00	1.2800183E+01	2.2145300E-01	4.2438674E-01	1.2704270E+00

0	15	1.3201318E+00	1.2544381E+01	4.0431085E-01	8.8889790E+00	2.6592956E+00	2.4045687E+01
0	16	1.3304698E+00	1.3042578E+01	1.2503649E-01	1.7428926E-01	1.6075547E+00	1.0485506E+00
0	17	1.3389552E+00	1.2198185E+01	2.8780338E-01	4.7591419E+00	2.8067992E+00	3.2510651E+01
0	18	1.3733151E+00	1.3230835E+01	8.9338601E-02	2.3411249E-01	1.8510491E+00	4.9319407E-01
0	19	9.6602041E-01	1.3261151E+01	3.1431100E-01	6.8481398E+00	1.4275994E+01	1.4275994E+01
0	20	1.5199513E+00	1.3754130E+01	9.3490042E-02	3.002958E-01	1.8457091E+00	4.3440053E-01
0	21	1.2945977E+00	1.3613601E+01	9.9958032E-02	4.4707873E-01	1.9423704E+00	2.5591488E+00
0	22	1.2200956E+00	1.3324661E+01	5.3332752E-01	2.7332654E+00	2.7931705E+00	3.6834457E+01
0	23	1.0992471E+00	1.2695412E+01	7.0547655E-02	1.1653555E-01	8.8846469E-01	1.5152778E-01
0	24	2.0025637E+00	1.3765011E+01	3.7006605E-01	8.3413382E+00	3.0737481E+00	1.7096483E+01
1	MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17						
PAGE		12					

NOVEMBER 27, 2004 MSC.NASTRAN 9/15/04

SUBCASE

MAXIMUM DISPLACEMENTS

SUBCASE/ DAREA ID	T1	T2	T3	R1	R2	R3
0 25	1.1481447E+00	1.3357345E+01	7.5875238E-02	2.7119359E-01	2.2534339E+00	7.2732019E-01
0 26	2.0250950E+00	1.4316995E+01	2.8857285E-01	6.9608269E+00	2.6141276E+00	2.0742977E+01
0 27	1.5252187E+00	1.3073984E+01	2.6351857E-01	3.6181779E+00	2.1906266E+00	2.0793240E+01
0 28	1.3417424E+00	1.2456515E+01	6.0739583E-01	9.6078129E+00	3.4602306E+00	5.3672195E+01
0 29	1.5963758E+00	1.4005592E+01	9.9118114E-02	3.3494720E-01	1.8209181E+00	1.4838123E+00
0 30	1.3105774E+00	1.4756025E+01	3.1535706E-01	5.9793792E+00	2.7706811E+00	4.0605564E+01
0 31	1.7948925E+00	1.6109528E+01	9.9508204E-02	4.2978624E-01	2.4253976E+00	7.1931010E-01
0 32	1.4638301E+00	1.4538096E+01	1.0003304E-01	2.9604402E-01	2.4797621E+00	1.2599914E+00
0 33	1.0494508E+00	1.2403088E+01	1.2894355E-01	3.0117840E-01	7.7712363E-01	4.9479803E-01
0 34	1.5145944E+00	1.3620959E+01	4.3682033E-01	8.9276762E+00	3.3847144E+00	1.2449486E+01
0 35	1.4806631E+00	1.3353104E+01	5.1032370E-01	4.7157974E+00	2.6709237E+00	5.8027225E+01
0 36	1.6683429E+00	1.4502959E+01	6.1003774E-01	1.0444496E+01	3.0203192E+00	4.1392300E+01
0 37	1.8351510E+00	1.4751471E+01	1.3786058E-01	5.6336039E-01	2.0003262E+00	1.3689501E+00
0 38	1.6475418E+00	1.5148776E+01	6.7870215E-02	2.6176944E-01	2.2756140E+00	7.4094582E-01
0 39	2.0675008E+00	1.3819255E+01	4.7693160E-01	1.1353372E+01	2.8530040E+00	1.9374908E+01
0 40	1.5540913E+00	1.4517945E+01	2.6065260E-01	1.9743574E+00	2.1039271E+00	7.3545051E+00
MSC.NASTRAN JOB CREATED ON 15-JUL-04 AT 11:57:17						
NOVEMBER 27, 2004 MSC.N.						

NOVEMBER 27, 2004 MSC.NASTRAN 9/15/04

SUBCASE

0 SEID PEID PROJ VERS APRCH ***** ANALYSIS SUMMARY TABLE *****
 SENSITIVITY SEMG SEMR SEKR SELG SELR MODES DYNRED SOLLIN PVALID SOLNL LOOPID DESIGN CYCLE

0 0 1 1' ' T T T T T T F T 0 0 -1 0

OSEID = SUPERELEMENT ID.
 PEID = PRIMARY SUPERELEMENT ID OF IMAGE SUPERELEMENT.
 PROJ = PROJECT ID NUMBER.
 VERS = VERSION ID.
 APRCH = BLANK FOR STRUCTURAL ANALYSIS. HEAT FOR HEAT TRANSFER ANALYSIS.
 SEMG = STIFFNESS AND MASS MATRIX GENERATION STEP.
 SEMR = MASS MATRIX REDUCTION STEP (INCLUDES EIGENVALUE SOLUTION FOR MODES).
 SEKR = STIFFNESS MATRIX REDUCTION STEP.
 SELG = LOAD MATRIX GENERATION STEP.
 SELR = LOAD MATRIX REDUCTION STEP.
 MODES = T (TRUE) IF NORMAL MODES OR BUCKLING MODES CALCULATED.
 DYNRED = T (TRUE) MEANS GENERALIZED DYNAMIC AND/OR COMPONENT MODE REDUCTION PERFORMED.
 SOLLIN = T (TRUE) IF LINEAR SOLUTION EXISTS IN DATABASE.
 PVALID = P-DISTRIBUTION ID OF P-VALUE FOR P-ELEMENTS
 LOOPID = THE LAST LOOPID VALUE USED IN THE NONLINEAR ANALYSIS. USEFUL FOR RESTARTS.
 SOLNL = T (TRUE) IF NONLINEAR SOLUTION EXISTS IN DATABASE.
 DESIGN CYCLE = THE LAST DESIGN CYCLE (ONLY VALID IN OPTIMIZATION).
 SENSITIVITY = SENSITIVITY MATRIX GENERATION FLAG.

The following PARAM values were set in the Control File:

PARAM,NCCOMPS,0
 PARAM,PREGEST,NO

Note: These values may have been changed by PARAM statements in Bulk Data or Case Control.

***** END OF JOB *****

<Removed to reduce print file size>

CQJAD4	1030	2	1070	1565	1528	1071	-179.984
CQJAD4	1031	2	1071	1528	1491	1072	-179.984
CQJAD4	1032	2	1526	1489	1491	1528	-90.
CQJAD4	1033	2	1822	1785	1787	1824	-89.9964
CQJAD4	1034	2	1785	1748	1750	1787	-90.
CQJAD4	1035	2	1748	1711	1713	1750	-90.
CQJAD4	1036	2	1711	1674	1676	1713	-90.
CQJAD4	1037	2	1674	1637	1639	1676	-90.
CQJAD4	1038	2	1637	1600	1602	1639	-90.
CQJAD4	1039	2	1600	1563	1565	1602	-90.
CQJAD4	1040	2	1563	1526	1528	1565	-90.

\$ Referenced Material Records

\$ Material Record : aluminium_iso SI_mm
\$ Description of Material : Date: 30-Jun-94
MAT1 1 70000. .33 2.7-6 2.32-5 Time: 16:52:56 .03

\$ Material Record : Glass-epoxy

\$ Description of Material : Date: 15-Jul-04
MAT8 2 23900. 24200. .194 5660. 5500. 5500. 1.91-6 Time: 12:13:11
23. 214.2 352.3 197.07 356.8 71.3

.1

\$ Material Record : Honeycomb

\$ Description of Material : Date: 12-Aug-04
MAT8 3 1. 1. .3 1. 24.3 14.5 4.89-8 Time: 11:40:17

\$ Nodes of Group : wing-geometry

GRID	1	1.	0.	0.	0.	0.	
GRID*	2			.902286			.0133569
*	0.						
GRID*	3			.902266			.0133596
*	.05						
GRID	4	1.	0.	.05			.0133622
GRID*	5			.902245			
*	.1						
GRID	6	1.	0.	.1			.0133649
GRID*	7			.902224			
*	.15						
GRID	8	1.	0.	.15			.0133675
GRID*	9			.902204			
*	.2						

GRID	10	1.	0.	.2	.0133702
GRID*	11			.902183	
*	.25				
GRID	12	1.	0.	.25	

<Removed to reduce print file size>

GRID	2144	.418635	0.	1.	
GRID	2145	.295122	0.	1.	
GRID	2146	.260031	0.	1.	
GRID	2147	.227943	0.	1.	
GRID	2148	.197089	0.	1.	
GRID	2149	.168107	0.	1.	
GRID*	2150	.141461			-3.77479-4
*	1.				
GRID*	2151	.0462312			0.
*	1.				
GRID*	2152	.0327776			0.
*	1.				
GRID*	2153	.0208244			0.
*	1.				
GRID*	2154	.0611032			0.
*	1.				
GRID*	2155	.292657			-.0323917
*	1.				
GRID*	2156	.0710691			0.
*	1.				
GRID*	2157	.0852707			0.
*	1.				
GRID	2158	.902287	0.	1.	
GRID	2159	.33226	0.	1.	

\$ Loads for Load Case : Free-free
\$ Referenced Coordinate Frames
ENDDATA 42164aa2