WIND TUNNEL STUDIES EMPLOYING HIGHER ORDER STATISTICS TO DETECT ICING INDUCED UPSETS

Woutijn J. Baars¹, Ronald O. Stearman², and Charles E. Tinney³

¹Graduate Student, w.j.baars@student.tudelft.nl
²Professor, aerial@mail.utexas.edu
³Assistant Professor, cetinney@mail.utexas.edu

Department of Aerospace Engineering and Engineering Mechanics
The University of Texas at Austin, Austin, TX, 78712, USA

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Abstract. Several years of earlier research for the Air Force, related to the impact that war head induced damage had on the aeroelastic integrity of lifting surfaces and in turn the resulting upset of the complete aircraft, prompted a current look at similar aeroelastic events that might be triggered by damage of the aircraft due to icing. This possible aeroelastic impact, on the aircraft, due to icing damage is not a very commonly explored area of research. Although seldom studied, icing can also significantly impact the aircraft’s aeroelastic stability and hence the overall aircraft stability and control. In this latter context, classical flutter events of the lifting surfaces and controls can occur due to ice-induced mass unbalance or control force reversal. Also, a loss of control effectiveness or limit cycle oscillations of the controls and lifting surfaces may appear, due to significant time dependent drag forces introduced by separated flow conditions imposed by the ice accumulation. Two commonly observed aircraft icing induced stability upset scenarios were selected to research. The first scenario involves an elevator limit cycle oscillation and a resulting loss of elevator control effectiveness. The second upset is related to a violent wing rock or unstable Dutch Roll event. A plausible failure mechanism is proposed for both types of upsets.

1 INTRODUCTION

Ice accumulation on aircraft can have a major impact on stability and control of the aircraft and can result in irreversible upsets. The structural ice formation on the leading edges of wings and control surfaces initiate significant regions of unsteady flow. [1] These regions disrupt the flow over the aerodynamic surface, thereby altering its performance. This makes ice formation a significant threat to aircraft safety. In the period of 1990 - 2000, a total of 3,230 aircraft accidents were recorded by the Air Safety Foundation (AOPA). Twelve percent were related to icing. [2] Studies on ice-related accidents of small general aviation aircraft revealed that in many cases even the most experienced pilots do not have more than 5 to 8 minutes to escape the harmful icing conditions before their aircraft experiences violent upsets. The majority of the accidents, for one make and model studied, occurred during the
approach and landing phase [3–5], where the aircraft is flying at a larger angle of attack, $\alpha$, compared to cruise flight. The ice accumulates in cruise flight but the effects on stability and control remain mostly unobserved. When changing the attitude of the aircraft to a higher angle of attack, the ice induces unsteady flow phenomena that can upset the aircraft.

After reviewing two NTSB Safety Recommendations (2004,2006) [4, 5], Airworthiness Directives from the FAA [3], and two pilot reports presented in the work of Endruhn et al. (2006) [6], two aircraft destabilizing, fluid-induced, mechanisms have been identified for study. The upsets can occur on small general aviation aircraft when ice accumulation is present on the leading edges of the elevator horn balances as well as residual ice remaining on the wing and stabilizers after a few boot deicing cycles. The horn balances are critical aerodynamic control surfaces not equipped with anti- or de-icing devices in full-scale aircraft applications. The first scenario involves a limit cycle oscillation (LCO) of the elevator caused by an ice-induced unsteady shearlayer flow-structure interacting with the relative flexible horn balance. This interaction appears to be non-linear in nature. [7–9] In the work of Baars et al. (2009) [10] an effort is undertaken to identify these linear and non-linear coupling mechanisms using Higher-order spectra (HOS) analysis techniques [11–16]. The first studied scenario can result in a loss of elevator control effectiveness. The second upset mechanism results in a violent wing rock or unstable Dutch Roll event caused by flowfield interactions between the ice-induced separated flowfield behind the elevator horn balance and the fuselage cross flow. The plausible failure mechanisms are presented in this paper.

2 AIRCRAFT CHARACTERISTICS RELATIVE TO ICING

Part 23 FAA Federal Aviation Regulation (FAR) certified aircraft are considered in the category of the smaller general aviation aircraft that are mostly propeller driven vehicles with features somewhat different than found on the larger Part 25 FAR aircraft that are more often turbine powered and much higher performance aircraft. Although Part 23 aircraft are certified to fly in icing through 14 CFR Part 23, this 14 CFR Part 23 references the 14 CFR Part 25 appendix for icing certification, so the icing requirements are equivalent, however, some differences are found between Part 23 and 25 aircraft. In a Part 23 aircraft, for example, the flight control systems are generally activated directly by pilot manual input and are reversible, not of the hard hydraulic irreversible control systems found typically on fighter aircraft and some of the larger Part 25 aircraft. The implication being that the general aviation control surfaces can be activated by hand through a manual oscillation or movement of the trailing edge of the control. This cannot be done on aircraft with irreversible controls. From the point of view of aircraft upset events, an aerodynamic input external to the pilot can in the Part 23 aircraft, with reversible controls, force the controls into an action that could overpower the pilot control input, upsetting the aircraft through an external aerodynamic gust or hinge moment input not initiated by the pilot. This could, in turn, upset the stability and control or the aeroelastic features of the total aircraft.
2.1 Elevator Horn Balance

Most of the reversible control surface systems employ a combined aerodynamic and mass balance horn, incorporated in an external control surface as indicated in figure 1 and 2.

![Image](image1.png)

Figure 1: 'shoulder', drip plate and elevator horn balance on the LHS of a small general aviation aircraft.\(^1\)

![Image](image2.png)

Figure 2: (a) Side view from tip. Schematic of LHS elevator horn balance. (b) Bottom view. Schematic of LHS horizontal stabilizer and location of elevator, horn balance, and simulated ice accumulation.

Mass balancing alleviates low flutter speeds associated with control surfaces found in reversible control systems. In the de-icing philosophy of ice protection on horn balances no inflatable de-icing boots are employed since their activation causes significant control hinge moment variations upsetting the aircraft trim conditions in cruise flight. However, when using the TKS anti icing systems\(^2\) no such problems exist and aerodynamic control horns are currently using the TKS ice protection system successfully.

2.1.1 Ice Accumulation Characteristics

In this section a brief outline is presented of the ice accumulation on the horizontal tail and the flow disturbances caused by it. For a more complete description, the reader is referred to the work of Baars et al. (2009) [10]. When the aircraft is flying through icing conditions, ice accumulates on all frontal exposed surfaces of the aircraft. On the leading edges of the wings and stabilizers a commonly found ice horn protrudes normal to the surface. Figure 3 shows an ice formation on a tail leading edge of a full scale NASA test aircraft.

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\(^1\)picture Cessna Aircraft Company (http://www.cessna.com/caravan/grand-caravan/grand-caravan-gallery.html, retrieved 14 March 2009)

\(^2\)Tecalemit Killfrost Sheep-bridge-Stokes (TKS) is an advanced anti- and de-icing system that squeezes ethylene glycol-based fluid through laser drilled porous titanium panels attached over the airfoil leading edges.
Unique wind tunnel experiments were conducted at the NASA Glenn Research Center icing research tunnel by Wilson (1967) [18] on a horizontal tail shielded horn balance in icing conditions. One of the experiments was performed at a free stream velocity of 90 m/s and in sub zero temperatures (−10°C). The horizontal stabilizer was set at α = 0° while the horn balance was deflected by δ = −4°. The mean droplet diameter (MDD) and the liquid water content (LWC) were 15 and 1.2, respectively. The experiments showed significant accumulation of ice in the form of ice horns, on both the leading edge of the stabilizer and the lifting surface of the horn balance. An illustration of the ice formation after a 5 and 7 minute exposure to known icing conditions is shown in figure 4a and 4b. After 8 minutes the horn balance started to show significant sub-critical-flutter events or LCO’s. Wind tunnel studies performed by Tate and Stearman (1986) [9], using simulated icing, have demonstrated this LCO behavior. Actually, this control horn LCO was first observed on a Fokker FR-1 brought to the front during during World War I in September of 1917. [8].

Several studies have been performed on the actual effects of simulated ice accretions on the leading edges of airfoils. [19–24] A schematic indicating the flow features around an airfoil with ice formation is shown in figure 5. The air flow around the ice horn on the lifting surface of the airfoil separates due to an adverse pressure gradient. Downstream of the ice horn a primary eddy and corner eddy is present followed by a reattachment region and a vortex shedding event from the shear layer. [19] The characteristics are consistent with the well-studied backward-facing step flows such as in review by Eaton & Johnston (1981) [25] and as presented in the study by Bradshaw & Wong (1972) [26].

The studies conducted by Gurban & Bragg (2006) [20] on a NACA 0012 airfoil (Re = 1.8 · 10^6) with leading edge ice simulations showed an increase in mean reattachment length (the length of the separation bubble), L_R, with increasing angle of attack. At α = 0° the reattachment point was located at 0.13c, while at α = 8° the separation bubble extended over the full chord of the airfoil. [19, 20] In the study by Broeren et al. (2004) [22] it was
indicated that the ice simulation caused a large increase in $L_R$ at $\alpha = 6^\circ$. Furthermore, Bragg & Khodadoust (1992) [24] concluded, based on an experiment on a NACA 0012 airfoil ($Re_c = 1.5 \cdot 10^6$) with leading edge ice simulation, that it is likely that at $\alpha = 6^\circ$ the bubble is highly unstable. At $\alpha > 6^\circ$ the flow is unable to overcome the adverse pressure gradient, resulting in an intermittent reattachment of the flow or no reattachment at all. This results in a bubble bursting phenomenon that can initiate a premature airfoil stall across the entire span (the stall angle of attack for a clean NACA 0012 airfoil is $\alpha_{stall} = 16^\circ$ [27]). [19, 23] Khodadoust (1988) [23] performed a study where ice horn simulations were placed on both the lifting- and non-lifting surface. The separation bubble on the lifting surface was having a significant larger size than the bubble at the non-lifting surface. Similar results were found by Busch et al. (2008) [28] who concluded that the horn at the non-lifting surface is less critical, because small variations of this horn make only a difference in the drag coefficient and only at low angles of attack. Therefore, the experimental studies were conducted with only one ice shape attached to the lifting surface of the elevator horn balance leading edges.

A qualitative wind tunnel visualization study was conducted at The University of Texas at Austin [29] to validate the occurrence of unsteady flow events behind the ice accumulation. Figure 6 and 7 show the coherent type of span wise vorticity structures and the flow separation over the horn balance for $\alpha = 0^\circ$ and $\delta = -8^\circ$. It was observed that the two dimensional span wise vorticity signatures were still present although highly 3d flow effects were observed at the stabilizer tip.

Figure 5: Schematic of surface separation bubble on an airfoil with leading edge ice accumulation, duplicate from Gurbacki et al. (2004) [21].

Figure 6: Flow visualization of a 1:10 scale LHS horizontal stabilizer with elevator horn balance (side view), $\alpha = 0^\circ$, $\delta = -8^\circ$, $Re_c = 1.7 \cdot 10^5$ [29].
Unsteady flow features resulting from the ice induced separation bubble were experimentally investigated by Gurbacki & Bragg (2004) [21] on a NACA 0012 airfoil. The first unsteady flow feature, quantified by a Strouhal number based on the free stream velocity $V_\infty$ and the reattachment length $L_R$ of $St_{L_R} = f \cdot V_\infty / L_R = 0.53 - 0.73$ [21], is associated with the shear layer vortex structures (vortex movement in and aft of the shear layer) and is referred to as the regular mode. This mode was found by spectral analysis of time dependent surface pressure measurements at the chordwise position where the separation bubble reattached. Likewise, spectral analysis of lift and moment coefficients captured by a three component balance system revealed the second unsteady flow feature, corresponding to a Strouhal number of $St_{L_P} = f \cdot V_\infty / L_P = 0.0048 - 0.0101$ [21], where $L_P = c \cdot \sin(\alpha)$ is the projected airfoil height. This mode is referred to as the low-frequency mode and is often associated with shear layer flapping. In this study the linear and non-linear coupling between the two unsteady measurements was not analyzed. The unsteady quasi-harmonic ‘motion’ of the structure can actually be a result of a non-linear lock-in of the structure with the unsteady quasi-harmonic, ice-induced, flow features as outlined by Baars et al. (2009) [10].

2.2 Wing Root Leading Edge Shoulder

A wing root vortex is originated at the wing root/fuselage junction. This vortex is the result of the trailing edge flow at the wing root that has the tendency to flip over to the other side of the fuselage due to the relative low pressure on top of the fuselage. Therefore, the two wing root vortices have the same orientation as the well-known wing tip vortices. In addition, for some small general aviation aircraft the strength of the wing root vortices is increased by the vortices trailing off from the so-called ‘shoulders’ at the wing root leading edge/fuselage junction as indicated in figure 1 and 8. It is believed that those shoulders are put on to generate extra vortical lift, because an elliptical shaped body, placed under a slight angle with respect to the incoming flow, generates a vortex.

Water tunnel studies have been performed by Stearman et al. (2006) [30] on a representative 1:32 scale aircraft model to confirm the existence of the wing root vortex pair. Similar studies were performed at the low-speed wind tunnel facility where quantitative stream wise velocity contours were acquired with a total head pressure rake. [30] The vortical regions show up as low pressure zones. At $\alpha = 0^\circ$ the vortex pair would trail below the horizontal stabilizers, as indicated in figure 9a. When increasing the angle of attack to $\alpha = 6^\circ$, the vortex pair
Figure 8: Shoulder on the RHS of the aircraft wing/fuselage junction.

would trail at the same level as the location of the horizontal stabilizers, shown in figure 9b. For $\alpha = 10^\circ$ the vortex pair is clearly located above the horizontal stabilizer and next to the vertical tailplane as shown in figure 9c. Furthermore, it was observed that when the aircraft is in a conventional landing approach or climbing mode the vortex pair would be above the horizontal stabilizer and next to the vertical stabilizer. [6]

![Figure 9: Total pressure ratio $1 - p/p_\infty$, $Re_{\text{wing}} = 85,000$, (a) $\alpha = 0^\circ$ (b) $\alpha = 6^\circ$ (c) $\alpha = 10^\circ$. [30]](image)

The last important characteristic are the drip plates mounted above the rear doors of the fuselage as indicated in figure 1. Those drip plates act as gurney flaps relative to the cross flow, to ensure that the separation points of the cross flow around the fuselage are fixed. The cross flow, $V_c$, occurs when the aircraft is flying at an angle of attack as schematically indicated in figure 10. The trailing edge vortical unsteadiness originating from the wing root is further aggravated when the fuselage chines or drip plates, holding these root vortices to the fuselage, also ice up and thereby blunting the sharp aerodynamic chines, which can then no longer hold these two wing root trailing vortices to the fuselage. This will produces two alternately moving separation points.

![Figure 10: Fuselage cross flow schematic.](image)
3 PROPOSED ICE-INDUCED DESTABILIZING MECHANISMS

This chapter starts off with the identification of multiple destabilizing or failure mechanisms related to icing for a representative small general aviation aircraft. Two failure mechanisms are selected for study, that are introduced by pilot and witness reports. The failure mechanisms are discussed in detail in the last two paragraphs.

3.1 Weibull Failure Analysis

A Weibull Failure Analysis of published NTSB icing accident data, for a representative small general aviation aircraft, was accomplished employing the Weibull Smith commercially available software. This software has one of several analyses options known as the Multi-Weibull (MW) subroutine. This was developed for the case where the accident data input is known to have multi-failure modes embedded within it. Identifying the presence and approximate number of failure modes embedded within the input data set, can be estimated by first assuming the presence of only one failure mode in the input data set and running a traditional single Weibull Failure Analysis. A careful review of the resulting Weibull Reliability and Unreliability output data plots will then illustrate different groupings of failure modes through the so-called ”Dog Legs” in these plotted functions. That is, the data will group in what appears as different segmented straight lines defined by smaller numbers of output data points. Each grouping of data points can then be connected by jogs or ”Dog Legs” in the output data plots. The number of such straight line segments is a first estimate of the number of failure modes. Even though the NTSB accident data has been prescreened to include only accidents occurring in the presence of atmospheric icing conditions in practice, several different upset or failure modes could involve icing induced tail stall, icing induced wing stall or malfunction of different elements of the deicing systems, for example.

The object of the MW analysis is to determine the Weibull fit lines from one general data set involving mixed failure modes without requiring initial input data censoring or categorization into different data failure sets. The MW input data analysis option determines the Weibull fit lines from this one, general mixed, data set involving multiple failure events, which have all occurred while the aircraft is flying in icing. The resulting MW analysis output would, in essence, be different Weibull fit lines first identifying Infant Mortality or Early Failures as well as Random Failures (Weibull line slope less than or equal to 1.0) followed by wear out (Weibull slope greater than 1.0). The general perspective of the Reliabilities for each observed failure mode, are produced by individual straight lines, which when taken together give a resultant curved line on the overall Weibull graph simulating the entire range of product life as described by the well known ”bath-tub curve”. This model of the failure rate or hazard rate of the human life cycle was first adopted by the insurance companies. It was later adopted to model the Reliability of mechanical systems. [31]

An estimate of the overall Reliability of this representative general aviation aircraft, when flying in icing conditions, is shown when estimated by the MW approach in figure 11a, demonstrating an estimated four failure modes in the 28 data point set. The slope $\beta$ of the
first and most critical Weibull line is 0.60 which is seen to lie well within the Infant Mortality or Early Failures region of the hazard rate or "bath-tub curve". The Characteristic Life $\eta$ for this first most critical failure mode is approximately 4,365 hours. This characteristic life is quite low for this class of aircraft. The other three failure modes lie outside the Infant Mortality failure zone into the so-called Wear Out region. Finally, a typical Weibull Failure Analysis using a single line slope estimate, produces the Reliability function illustrated in figure 11b. The resulting "Dog Legs" in this Reliability estimate of the 28 point data set suggest four failure modes as well. The slope $\beta$ of this single Weibull line approaches the random failure slope of 1.0. The estimated value is close to this, being 1.08 as shown in figure 11b. Since multi-failures are admitted, the single line analysis is suggesting that any of the multiple failures could occur randomly across any one of the possible failure modes.

3.2 Pilot and Witness Reports of Two Failure Modes Selected for Study

Of the four possible failure modes suggested in the Weibull Failure Analysis, two were chosen for analysis that were also identified in pilot reports to the FAA/NTSB. One of the failure modes is initially triggered with the aircraft in a normal cruise attitude with no apparent significant icing condition. The upset occurred by a sudden onset of a LCO of the elevator, creating a "fluttering" of the elevator control column. This LCO of the elevator stalled the aircraft and pitched it over into a dive. A complete elevator control effectiveness then occurred in the dive with no way for the pilot to recover. In a last desperate move the pilot activated the de-icing system which initiates the tail de-icing first. The removal of the horizontal tail ice re-established the elevator control effectiveness and averted a fatal accident. A brief description of this ice-induced upset is presented in the following pilot report.
Narrative, Upset Number I: Acft. was in light rime ice at 9,000 ft M.S.L. The wings and windshield were showing light rime ice accumulation, but not enough to warrant turning the boots on. The pitot static and prop heat were already on. The aircraft yoke started to flutter and almost immediately the aircraft stalled and pitched over into a dive. The elevator would not respond to any pilot elevator input, but only to pilot rudder and aileron input. I turned the pneumatic boots on while in the dive and regained elevator ctl. at approximately 4,000 ft M.S.L. I then proceeded to climb to 7,000 ft M.S.L. where I remained for the rest of the flight at a temp. of +2 degs. C. The main reason I wanted to rept. this is that similar circumstances occurred to me approx. 14 months ago. It would appear that the tail is accumulating more ice or is unable to carry as much ice as the main wing.

Another pilot working for the same air freight company that reported the above two upsets, had a similar upset making a total of three similar upset events reported by the same freight carrier. In summary, this upset event is triggered by an elevator limit cycle initiation for an aircraft in normal cruise attitude, which had no significant icing encounters based upon what the pilot observed in the way of only light ice deposits on the wings and windshield. This limit cycle event pitched the aircraft over into a dive where all elevator control was lost. Recovery back to an effective elevator control was only achieved after the removal of ice on the horizontal tail, averting three fatal accidents in this situation. There were two major unknowns to the pilots that experienced this upset scenario. First, in November 1991 the FAA and NASA sponsored an International Tailplane Icing Workshop where it was established that the tailplane is generally a more efficient collector of ice than the wing because it generally has a smaller leading edge radius than the wing. There have been reports of ice accretion on the tailplane 3 to 6 times thicker than ice on the wing. The tailplane icing was therefore more extensive than the pilots realized by monitoring only the wing and windshield. Secondly, a difficulty occurred because little is known about the icing of vortex generators and the exact mechanism as to how icing can disable their function. This item will be clarified in the analysis of this first upset.

A second class of upset has been experienced when the aircraft enters into a climbing attitude which might be the case when encountering an icing situation he wishes to fly above. Elevator horn icing is a strong possibility here due to the 3:1 to 6:1 rapid tail plane icing compared to the wing icing, which is obviously easier for the pilot to see, but is not a good indicator of the overall aircraft icing state. The crew and pilot usually can’t see the tail, and typical de-icing system pressure sensors provide no indication of unshed ice. Further complicating this issue is the fact that no boot de-icing systems are employed on control horns, because they introduce a time-varying control hinge moment during the activation that upsets the aircraft’s trim conditions. The control horns are thus essentially unprotected for aircraft with de-icing systems. This we feel is one of the key issues in this upset. The following narrative of this upset was extracted from ten on sight witness statements submitted to the NTSB.

Narrative, Upset Number II: Ten witnesses submitted written statements and their locations were plotted on a chart. Several of these witness observations were nearly identical. Witness 1 observed the airplane emerge from a 300 foot ceiling. "It was rocking very badly from side to side. It rocked 2 to 3 times before diving 100 to 150 feet onto the highway and skidded into the lake." Witness 2 saw the airplane "bank to the left and then bank to the right, then bank
to the left, then took a hard bank to the right. The cockpit of the plane and the right wing hit the road.... (And the airplane) slid into the lake on its back and landed approximately 30 feet into the water.” Witness 3 who was sitting in her office saw the airplane’s "right wing go down sharply. It came back up and the left wing tipped down sharply. The left wing came up and the (airplane) flipped completely over.”

A more quantitative identification of the violent wing rock problem can be found in the NTSB Safety Recommendation (2006) [5] where a Russian aircraft was equipped with both a cockpit voice recorder as well as a flight data recorder that have quantitatively identified significant wing rock excursions on the order of ±40 degrees. Conversations found on the voice recorder definitely identify the presence of significant icing just prior to the wing rock upset.

3.3 Upset No. I: Loss of Elevator Control Effectiveness Through Icing

The category I upset described in the previous section caught the pilot off guard as he was not acquainted with the fact that the tail plane icing accumulates 3 to 6 times thicker over time than on the wing. By considering a free body diagram of the elevator control horn as a rectangular typical section slender wing configuration, where drag caused by icing may be significant, one can rationalize a theoretical model of the form employed by Petre and Ashley (1976) [33] for investigating drag effects on wing fluttering or limit cycling as shown in figure 12. The reader is referred to [6,9] for more details on the derivation of the Van der Pol control horn equation of motion given by Eq. (1):

\[
\beta'' - \delta \left( a - b \beta^2 \right) \beta' + \left[ \bar{k}_0 - 2 \delta \left( a - b \beta^2 \right)^2 \right] \beta = 0,
\]

where \( \beta \) is a non-dimensional \( \phi \), \( a = (\varepsilon - 1) \), \( b = \varepsilon \eta/2 \), \( \eta \) a non-dimensional shear layer geometry, and \( \delta \& \bar{k}_0 \) are non-dimensional parameters. For a complete derivation and description

Figure 12: Aeroelastic model for the shielded slender elevator horn balance in a Gaussian approximation wake velocity field \( u(z, t) \), duplicate from Endruhn et al. (2006) [6].
of the model the interested reader is referred to Tate and Stearman (1986) \cite{9}. Usually, the drag force element will only be significant when icing is present. It will manifest its presence in an aerodynamic stiffness element that is most likely time dependent in character. This brings in a possible Mathew-Hill character to the equations and a possibility for a quadratic non-linearity to the already non-linear form of the equations. \cite{7} From figure 12 it is easy to rationalize why a LCO could be triggered for a sheltered control horn. That is, for the proposed model it is well known that the equation of motion, Eq. (1), can exhibit relaxation oscillations (i.e. pulsing of the elevator horn, some times felt by the pilot in icing conditions) or a steady LCO occurring possibly at slightly higher velocities or for different icing parameters. This latter event is the case for the discussed category I upset event. For slender rectangular wings, the aerodynamic loads are impulsive in character and concentrated at the leading edge of the aerodynamic horn. By looking at figure 12, simple statics tells us that for slender body airloads, which act at the horn balance leading edge, the shear layer aerodynamics wants to suck the leading edge of the elevator horn out of the sheltering pocket of the stabilizer tip. Only sufficient elevator bending and torsional stiffness in the neighborhood of the horn geometry, friction in the control circuit, pilot input, and a balancing or weather vaining opposing force from the remaining elevator input can help prevent this from happening. Once the elevator horn pops out of the sheltering stabilizer pocket, the aerodynamic forces on the leading edge of the horn, based on slender body theory, will vanish once the leading edge is exposed to uniform flow. The elastic restoring forces move the horn back, producing a natural limiting amplitude and hence a LCO repeating oscillation.

To further understand the loss of elevator control issue, reference is made to an excellent book on the Cessna Single Engine development story by William D. Thompson, Chief of Flight Test and Aerodynamics during the Caravan Development Days. \cite{34} In one part of the book he comments on the Grand Caravan C208B development history which answers a question commonly asked by the layman and experts alike concerning why a row of vortex generators are added to the stabilizers just ahead of the elevator hinge line. The following is Thompsons answer to that question: "A unique problem of marginal nose-down elevator power was observed in transitional out-of-trim flight evaluations. This was alleviated by a single row of vortex-generators on the top surface of the horizontal tail just ahead of the gap between the stabilizer and elevators." Since most of these aircraft icing accidents occur in the landing glideslope approach and flaring maneuver, and little is known about the influence of icing on vortex generators, a study was initiated to investigate this issue. A lack of elevator nose down authority during a landing maneuver could give rise to an upsetting event during this critical transitional phase of flight.

Another study was initiated at The University of Texas to obtain some insight on the influence of icing on vortex generator performance. Very limited hints of hand written comments found in flight test formal reports for small general aviation aircraft, for example, indicated that the vortex generators located along the elevator hinge line did not seem to ice up to any degree of significance. That is, the 3:1 to 6:1 tail-icing:wing-icing growth rate does not seem to hold in the limit of smaller radii lifting surfaces such as vortex generators. It seem that when there is vanishing frontal area for ice to deposit on it will not build up. Then the question arises what mechanism is at work, if any, for ice to nullify the benefits of vortex generators? Results
from an experimental wind tunnel study conducted at The University of Texas at Austin [35] related to this question are shown in figure 13. A total head rake was run along the trailing edge of the stabilizer of our 1:10 scale model with vortex generators to check the presence of the spanwise vorticity generated by the vortex generators. Figure 13a (top) illustrates a clean stabilizer where the generated vorticity of each vortex generator is clearly seen in good detail. When simulated residual plus intercycle icing is applied to the stabilizer, as shown in figure 13b, a complete annihilation of the organized vorticity from the vortex generators is observed as shown in figure 13a (bottom). An examination of figure 14 suggests that an overlay of a spanwise vorticity, shed off the icing generated bubble shear layer, is interacting with the orthogonally vorticity created by the vortex generators. In essence, the orthogonal overlay of these two distinct vorticity fields annihilate each other. This action then nullifies the benefit of the vortex generators creating a loss of elevator pitch down control authority induced by icing conditions. In summary, if a LCO is occurring, driven by the elevator iced up horns, as in the case of the category I upset, a much stronger spanwise bound vorticity is being shed which will certainly annihilate the benefit of the vortex generators. A complete loss of elevator control authority is to be expected.
3.4 Upset No. II: Wing Rock

The category II “wing rock upset” described earlier under the pilot and witness reports has been observed in a number of accidents by several witnesses and has been experienced by at least one surviving pilot. [3–6] Seeking the opinion of an Aerospace Engineering expert on the subject, G. J. Hancock (1995) [36] states in his book: “A key ingredient in wing rock is the loss of Dutch Roll damping”. According to Hancock, wing rock can occur in at least three different forms which are itemized below:

1. Wing rock can occur in flow regimes where lag effects are not prominent, this form of wing rock is repeatable.
2. Wing rock can occur in flow regimes where lag effects due to onset of flow breakdown and reattachment, including vortex breakdown and reassembly, are prominent. This form of wing rock is sometimes random in occurrence with variations in amplitudes. Forebody vortices can play an influential role in this type of wing rock.
3. Persistent small amplitude irregular oscillations in roll can be generated at high subsonic speeds by asymmetric fore and aft movements of shock waves on the upper surface of the wing.

The first type of wing rock is most likely the one that is influencing the small general aviation aircrafts considered in this study, since their vortices are not developed from a fore body but the wing leading edge, implying that no prominent phase effects exist. This would allow a quasi-steady analysis with the dynamic terms and the airplane inertias in the equations of motion playing a less important role. It has also been observed in aircraft with well damped stability and control modes that the precise value of their three major inertias are not so critical to the study of their wing rock phenomenon.
The EA-6B Prowler, a twin-engine, electronic warfare aircraft, is exposed to stability and control upsets that were related to the pair of wing root vortices. Visualization studies were performed by Jordan, Hahne, Masiello and Gato (referenced by Bertin & Smith (1989) [37]) indicating that a pair of vortices was generated at the wing root leading edge. At low angles of attack those vortices were located below the horizontal stabilizers, as presented in figure 15a. At angles of attack below the stall angle the vortex pair trailed at around the same location below the horizontal stabilizers due to the wing downwash. However, at angles of attack close to the stall angle, where the downwash effect of the wing was significantly reduced by the flow separation over the wing, the vortex pair was located above the horizontal stabilizers next to the vertical stabilizer as indicated in figure 15b. In case of a slight side slip, i.e. due to a gust, the vortex pair would flip over to one side of the vertical stabilizer, as indicated in figure 15c. The vertical stabilizer may now be exposed to a net force acting to the right hand side. This has a direct impact on the directional stability of the aircraft by causing a yaw motion that can result in the stall of the left wing, because the aircraft’s attitude was already close to the stall angle, causing a roll motion and thereby initiating the wing rock event.

![Figure 15: Directional destabilizing mechanism of the wing root vortex pair on the EA-6B. [37]](image)

It is claimed that the EA-6B and the small general aviation aircraft having those wing root leading edge shoulders, have similar vortex interactions with the aircraft tail planes. This is claimed to be due to the fact that both aircraft trailing wing root vortices originate from their wing root leading edges. In section 2.2 the pair of wing root leading edge vortices was introduced. Those wing root vortices are then shown to trail off from this wing root leading edge passing over the wing and ending up next to the vertical stabilizer as was shown in figure 9.

![Figure 16: (a) Pressure transducer mounted on the elevator horn balance, (b) Cable-mounted model in the wind tunnel. [35]](image)
It was also concluded that a resulting violent wing rock motion occurred when the wing trailing vortex pair broke loose from the aerodynamic chines resulting in a fuselage cross flow vortex shedding motion, as was presented in figure 10. A study has been conducted at The University of Texas at Austin on a 1:10 scale, powered, radio-controlled, dynamic-cable-mounted, 6 DOF model, shown in figure 16b. The observed fuselage cross flow shedding frequency of approximately 1Hz locked-in with the unsteady separated flow frequencies of 10Hz and 20Hz over the elevator horn balances, caused by the simulated ice accumulation, in a combination resonance mode. To deal with the non-linear aspects of these vortex dynamic interactions, Higher Order Spectra (HOS) analysis techniques [11–16] were used in the signal processing employed in the wind tunnel testing. Although the ice-induced unsteady flow phenomenon had different frequencies (measured using a pressure transducer near the elevator horn balance as 10Hz and 20Hz, see figure 16a) than the Dutch Roll wing rock frequency (measured using accelerometers at 1Hz), it was concluded based upon cross-bicoherency-squared analysis, that a lock in of these three separate frequencies occurred in a quadratic sense, only when icing was present on the horn balance leading edges. [6] Figure 17 is a plot of this estimated cross-bicoherency-squared function illustrating these events by demonstrating spectral peaks at coordinates of 10Hz and 1Hz as well as at 20Hz and 1Hz.

\[
\Omega_f \approx |\omega_i \pm \omega_j|/k, \quad k = 1, 2, 3...and (i \neq j)
\]

Figure 17: Cross-bicoherency-squared function of the pressure transducer and z-component of the accelerometer when ice was simulated on the elevator horn balance leading edges. [6] \(\Omega_f\) indicate acceleration and \(\omega\) indicate pressure.

Highspeed camera screenshots, presented in figure 18, indicate that the wing root vortices become very violent when the lock-in occurs, as they flip over to the other side. In this study by Endruhn et al. (2006) [6] it was also shown that if the aerodynamic chines or drip plates were extended back along the fuselage to about 0.5m (in full scale size) forward of the leading edge of the horizontal stabilizers, the wing rock instability was suppressed. In essence, the highspeed video taping of this event also indicated that the extended chines actually hold the vortices down and away from the vertical stabilizer, so no cross flow can induce yawing and rolling of the aircraft as shown in figure 15c of the EA-6B aircraft.
4 CONCLUDING REMARKS

Two known general aviation aircraft icing upset scenarios were investigated to determine if some insight could be obtained as to the actual mechanisms causing the upsets. One upset involved an elevator horn ice-induced limit cycle oscillation of the elevator followed by a complete loss of elevator control authority. The second upset investigated involved an iced up elevator control horn inducing a violent wing rock or unstable Dutch Roll response. Both of these problems were investigated by employing wind tunnel testing as the primary investigative tool.

It was demonstrated that the loss of elevator control authority encountered in the first upset was due to a spanwise vorticity shedding off the shear layer of the separation bubble, caused by leading edge icing, which will occur even in the absence of a limit cycle oscillation. This vorticity is approximately orthogonal to the vortex generator vorticity which is incorporated to overcome a loss of elevator nose down trim authority of relative long-fuselage aircraft. The orthogonal overlay of two vortex fields, of the appropriate relative strength and wave length, will destroy both vorticity fields. This was demonstrated experimentally in the wind tunnel, figure 9. When the elevator is in a state of limit cycle oscillation the continual shedding of the bound vorticity of the surface will utterly destroy any vorticity produced by the row of vortex generators on that surface. The obvious solution to this problem is to employ an anti-icing system on the elevator as well as to any aerodynamic horn balance. Some commercial aircraft are now successfully employing the TKS technology on aerodynamic control horn balances.

Wind tunnel studies were employed to investigate the Dutch Roll induced instability, also triggered by an iced up elevator aerodynamic control horn balance. Two potential areas proved to show promise for alleviating this problem. First again, the anti icing procedures are recommended for the aerodynamic control horn balances to avoid rapid ice build-ups on tail planes that always seem to occur if no ice protection is employed here. Since this upset involves the two wing root fuselage trailing edge vortices, which are employed to enhance the

Figure 18: Highspeed camera screen shots showing the violent pair of wing root vortices. The orange boxes highlight the tuft, attached to the farside of the fuselage just behind the trailing edge, flipping over to the nearside.
aircraft vortex lift through aerodynamic chines, thought to be drip plates for passenger rain protection, these should also be anti-ice protected. Otherwise the chines will become blunted by the icing process and the trailing vortices will no longer be held by the blunted chines. In addition, wind tunnel studies indicated that when the chines, that are even ice free, are extended by about two feet full scale they will hold these trailing vorticies and will not allow a wing rocking event.

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5 REFERENCES


