

Large-Eddy Simulation of an Iced Swept Wing: A Perspective on the Aerodynamic Effect of Triggering Resolved Turbulence

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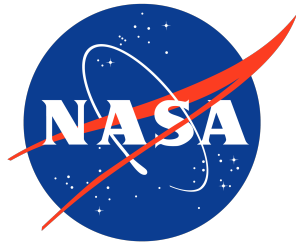
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Abstract

Analyzing iced wing aerodynamics is important for ensuring aircraft safety. However, simulating iced configurations accurately and efficiently can be challenging due to the complex geometries and flow topologies involved. To investigate these challenges, wall-modeled large-eddy simulation of a swept wing with leading-edge ice is performed using an unstructured Voronoi mesh paradigm and compared to experimental results. The study focuses on an 8.9% scale model of the CRM65 swept wing, a 65% scale version of the Common Research Model (CRM) geometry, featuring both high-fidelity and smooth leading-edge ice shapes at a Mach number of 0.18 and a Reynolds number per mean aerodynamic chord of 1.6×10^6 . For the high-fidelity ice shape, good agreement with the experiment is obtained using a moderate level of mesh resolution. For the smooth ice shape, a much finer mesh is required to obtain a similar level of accuracy. However, comparable accuracy is achieved at similar mesh resolutions for the high-fidelity ice and smooth ice when roughness is added to generate resolved turbulence near the leading edge. For the smooth ice, using a simple equivalent sand-grain roughness estimate to mimic experimental grit roughness results in a factor of approximately 30 reduction in computational cost for similar accuracy compared to simulating the smooth ice without roughness. Finally, simulations of the tripped wing without ice are used to quantify the predicted aerodynamic degradation due to icing across an angle of attack sweep.

Nomenclature

b	=	span, m
C	=	Sutherland temperature $\left(\mu_\infty = \frac{C_S T_\infty^{3/2}}{T_\infty + C}\right)$ [3], K
C_D	=	drag coefficient
C_L	=	lift coefficient
C_M	=	pitching moment coefficient
C_S	=	Sutherland law constant, $\text{kg m}^{-1} \text{s}^{-1} \text{K}^{-0.5}$
c	=	chord, m
c_p	=	specific heat of air at constant pressure, $\text{J kg}^{-1} \text{K}^{-1}$
d	=	distance below the top of the roughness where the mean flow is zero, m
k_{Ra}	=	average surface roughness, μm
k_{RMS}	=	root-mean-square surface roughness, μm
k_s	=	equivalent sand-grain roughness height, μm
$k_{s,\text{Ra}}$	=	equivalent sand-grain roughness height based on average surface roughness, μm
$k_{s,\text{RMS}}$	=	equivalent sand-grain roughness height based on root-mean-square surface roughness, μm
L_s	=	mesh length-scale parameter, mm
M	=	Mach number
p_∞	=	freestream pressure, Pa
q_∞	=	freestream dynamic pressure, Pa
Re	=	Reynolds number per mean aerodynamic chord
T_∞	=	freestream temperature, K
u_τ	=	friction velocity, m s^{-1}
$u_\parallel$	=	tangential velocity at the matching location, m s^{-1}
u_∞	=	freestream velocity magnitude, m s^{-1}
x	=	chordwise location, m
y	=	spanwise location, m
y_{match}	=	wall-model matching location, m
y_0	=	scale that is similar to the equivalent sand-grain roughness height, μm
y^+	=	normalized wall distance $\left(y^+ = \frac{y_{\text{match}} u_\tau}{\nu}\right)$
ΔC_i	=	numerical minus experimental aerodynamic coefficient, where $i \in \{D, L, M\}$
Δt	=	time step size, s
Δx	=	minimum cell size, mm
α	=	angle of attack, $^\circ$
γ	=	ratio of specific heats
κ	=	von Kármán constant
μ_∞	=	freestream dynamic viscosity, $\text{kg m}^{-1} \text{s}^{-1}$
ν	=	kinematic viscosity, $\text{m}^2 \text{s}^{-1}$
ρ_∞	=	freestream density, kg m^{-3}

1 Introduction

Understanding the aerodynamic performance effects of ice buildup during flight is important to mitigate the negative impacts of in-flight icing, including reduced lift, nonlinear changes in pitching moment, and increased drag [4]. Computational approaches for understanding flows over iced geometries are attractive as they can be financially cheaper and less resource intensive compared to traditional wind tunnel experiments and flight tests. Furthermore, experimental strategies are not immune to uncertainty. For example, in [5], it is highlighted that the iced swept wing studied in the present work is associated with experimental uncertainties in the aerodynamic loads of up to or just exceeding 9% at an angle of attack of 4° [6]. Computationally, iced configurations are challenging to simulate because they can require that high-quality meshes be built around intricate geometries and involve complex flows with separated regions and a range of length scales. Stebbins et al. [7] notes that Reynolds-averaged Navier-Stokes (RANS) methods are commonly used to simulate iced flows (e.g., [8, 9]) due to their relatively cheap computational cost and historical legacy. However, they have difficulty predicting aerodynamic loads for separated flows, which are common for high angles of attack and iced geometries [10–12]. The focus of the present work is on aerodynamic performance predictions of a swept wing with fixed leading-edge ice shapes. This approach is adopted to separate the challenge of simulating the generation and evolution of an ice shape from the challenge of predicting its aerodynamic performance. It is reasonable to consider these challenges separately because the physical time associated with generating the ice shapes considered in this work is on the order of several minutes [13], while the aerodynamics flows associated with these same ice shapes typically reach a statistically stationary state in a fraction of a second of physical time.

As an alternative to RANS, scale-resolving approaches like wall-modeled large-eddy simulation (WMLES), along with other hybrid approaches, have been introduced as a computationally tractable means of time-accurately resolving large-scale turbulent structures while modeling subgrid-scale features. Xiao et al. [14, 15] performed pioneering work applying WMLES to iced airfoils. They noted the remarkable ability of WMLES to predict flow separation and resolve the Kelvin-Helmholtz instability associated with the ice-induced separated shear layer. Similar results were obtained by Wong et al. [16] for a NACA 23012 airfoil with EG1164 horn ice accretion. This case was also used as a validation test for Voronoi WMLES in [1]. Bornhoft et al. [17] studied a NACA 23012 airfoil with several different ice shapes and observed that WMLES accurately captured the aerodynamics of glaze ice shapes with large roughness scales, but that special consideration must be taken for rime ice shapes when the length scales of the roughness are sub-grid.

Recent WMLES studies have focused on three-dimensional ice shapes formed on a straight wing with the NACA 23012 airfoil section [18] and an 8.9% scale version of the CRM65 swept wing, a 65% scale version of the Common Research Model (CRM) geometry [1, 5]. Hereafter, the 8.9% scale version of the CRM65 swept wing will be referred to as the CRM65 wing. Note that the CRM geometry was initially created for the AIAA Drag Prediction Workshop [19–22]. The CRM65 wing with leading-edge ice buildup is an attractive validation case for numerical methods in part because it has been studied in detail experimentally [13, 23–25]. Numerical studies of the iced CRM65 wing have focused on two three-dimensional ice shapes: a high-fidelity shape representative of realistic ice accretion, along with a smooth variant of the same shape in which the local maximum cross-sectional profiles are smoothly lofted across the wing span [26]. The development and study of the smooth shape is to compare and validate ice shapes created by icing simulation tools that cannot reproduce the spanwise variability associated with the high-fidelity shape [27]. For instance, NASA’s LEWICE3D software combines a three-dimensional droplet trajectory algorithm with a two-dimensional strip strategy for predicting ice accumulation, which produces ice

shapes that approximate realistic ice accretion at discrete perpendicular profiles but lack detailed three-dimensional features [28]. Another example is ONERA’s IGLOO3D icing suite, which uses a predictor-corrector approach that is unable to capture highly three-dimensional spanwise features, since this approach poorly models shadowing effects involving droplet collection that significantly contribute to the formation of spanwise ice structures on swept wings [29].

Initial computational studies by Stebbins et al. [30, 31] focused on using improved delayed detached-eddy simulation and RANS to predict the aerodynamics of the CRM65 wing with the smooth ice shape. Good agreement was observed for the aerodynamic loads for the lower angles of attack considered, but under-prediction was noted at the higher angles with some discrepancy in the surface pressure profiles. Subsequently, Ozcer et al. [32] used RANS to simulate the CRM65 wing with high-fidelity ice across a comprehensive angle of attack sweep and noted fair agreement with the aerodynamic loads at an affordable computational cost by slightly increasing the value of a closure constant used within the shear stress transport (SST) turbulence model such that the predicted aerodynamic loads fell between those predicted by the baseline $k-\omega$ and standard SST turbulence models. Recently, Bornhoft et al. [5] investigated WMLES of the CRM65 wing with both the high-fidelity and smooth ice shapes using an unstructured Voronoi approach. They observed that the high-fidelity ice triggers bypass transition near the leading edge, which is accurately simulated by WMLES. This resulted in good agreement with the experimental aerodynamic loads, surface pressure profiles, and wall shear stress contours. However, the smooth ice was hypothesized to introduce laminar boundary layers and laminar-to-turbulent transition effects, which violate the fundamental assumption of a turbulent boundary layer required by WMLES. To address this, [5] introduced additional mesh resolution and modeled the smooth ice shape with a no-slip boundary condition, which resulted in good agreement with the aerodynamic loads and pressure profiles for the smooth ice. However, they emphasized that this procedure significantly increased computational cost and benefited from information obtained experimentally to assist with mesh resolution requirements and boundary condition placement.

The objective of this work is to investigate and validate WMLES for simulating the aerodynamics of the CRM65 wing with both high-fidelity and smooth leading-edge ice, using an unstructured Voronoi mesh paradigm. Specifically, numerical results are compared against experimental aerodynamic loads, surface pressure profiles, and oil flow visualizations. A simple roughness modeling approach is investigated to trigger resolved turbulence near the leading edge of the smooth ice. This strategy will be shown to remove the restrictive mesh and explicit time step size requirements associated with simulating the very thin boundary layer near the leading edge of the smooth ice while achieving comparable accuracy to the high-fidelity ice cases on similar mesh levels. In addition, simulations of the wing without ice are performed to provide a quantitative baseline for predicting the aerodynamic degradation due to icing. The remainder of this paper is organized as follows: details about the CRM65 test case are provided in Sec. 2, computational methodology is outlined in Sec. 3, simulation results are presented in Sec. 4, and conclusions are discussed in Sec. 5.

2 Problem Description

This work investigates the aerodynamic performance of a CRM65 wing with high-fidelity and smooth leading-edge ice shapes. These ice shapes were initially conceived in the context of experimental work [24, 25, 27]. A two stage procedure was used to generate the ice shapes. In the first stage, portions of the wing were placed inside the Icing Research Tunnel located at NASA Glenn Research Center at conditions facilitating ice buildup. Specifically, due to the size of the wing, three wing sections based on inboard, midboard, and outboard regions of the CRM65 wing were used to generate three leading-edge ice shapes [33]. The test sections were also truncated in the streamwise direction; however, the leading-edge wing size was maintained. The resultant ice accretions were then digitized using a three-dimensional electronic scanning procedure. In the second stage, the three sections of scanned ice accretions were combined using interpolation and extrapolation to define a representative ice shape over the full span of the wing, printed using additive manufacturing techniques, and affixed to the leading-edge of a wind tunnel model for experimental data collection [26]. Two ice shapes are investigated in the present work: a “high-fidelity” shape that was constructed to preserve as many details from the original scans as possible and a “smooth” shape that took the high-fidelity shape and smoothly lofted it over the full span, guided by the largest cross-sectional features along the span. Representative sections of these two ice shapes are shown in Fig. 1. The high-fidelity ice shape exhibits highly three-dimensional features known as scallops along the span that are absent for the smooth ice shape due to the smoothing and lofting procedure. Geomagic Wrap [34] was used to correct non-manifold artifacts in the original ice triangulations, resulting in the watertight triangulations utilized in this work to obtain the presented numerical results.

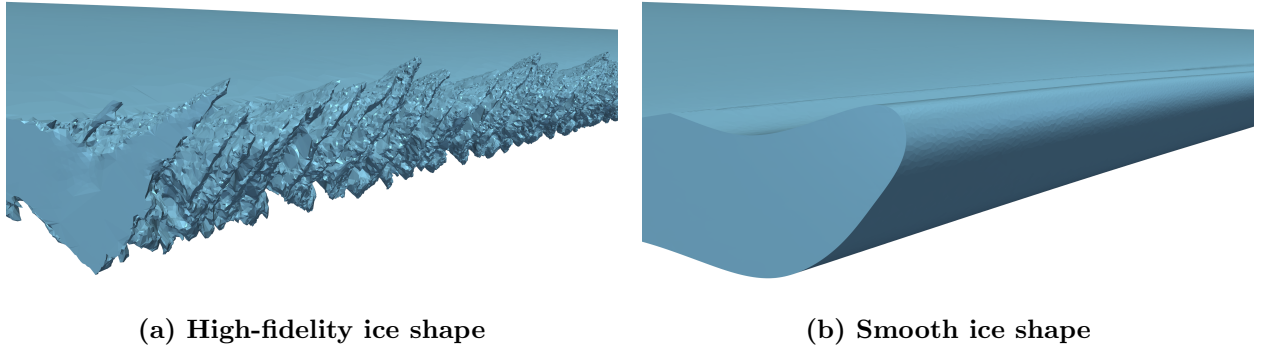


Figure 1: Renderings of the CRM65 wing with high-fidelity and smooth leading-edge ice buildup.

The experimental data used in the present work was obtained at the Walter H. Beech Wind Tunnel at Wichita State University (WSU) in Wichita, Kansas [24]. The WSU wind tunnel was used to perform tests at low Reynolds numbers from 0.8×10^6 to 2.4×10^6 and the ONERA F1 wind tunnel was used to verify that the WSU results applied to higher Reynolds numbers up to 12×10^6 [24]. Table 2 summarizes the flow conditions of interest, which were chosen in part because they match those of the WSU wind tunnel used for a tunnel-to-tunnel comparison [24]. A correction procedure was applied to the experimentally measured aerodynamic data to correct for the walls of the wind tunnel [24, 27, 35]. Table 3 provides the geometric information for the CRM65 wing. The present work compares aerodynamic loads, surface pressure profiles, and surface oil flow visualizations to WMLES predictions. For the surface pressure profiles, Fig. 2 and Table 4

detail the pressure tap locations used to collect information about experimental surface pressure distributions.

Table 2: Flow conditions for the CRM65 wing.

Variable	Value
Re	1.6×10^6
M	0.18
q_∞	2068.43 Pa (0.3 psi)
T_∞	288.87 K
u_∞	61.34 m s^{-1}
p_∞	91 200.62 Pa
ρ_∞	1.10 kg m^{-3}
c_p	$1005 \text{ J kg}^{-1} \text{ K}^{-1}$
γ	1.4
C_S	$1.45 \times 10^{-6} \text{ kg m}^{-1} \text{ s}^{-1} \text{ K}^{-0.5}$
C	110 K

Table 3: Summary of geometric parameters for the CRM65 wing [13, 23, 25].

Wing Parameter	8.9% Scale (WSU Model)	
b	60.00 in	1.524 m
Mean aerodynamic chord (MAC)	16.67 in	0.423 m
Planform area	865.3 in^2	0.558 m^2
Volume	1069 in^3	0.0175 m^3
Aspect ratio [*]	8.3	
Taper ratio	0.23	
Root chord	27.00 in	0.686 m
Tip chord	6.19 in	0.157 m
Root twist	4.4°	
Tip twist	-3.8°	
Sweep angle (1/4-chord)	35°	
Sweep angle (leading edge)	37.2°	
Location of rotation center [†]	$x = 19.37 \text{ in}, y = 0 \text{ in}, z = 0 \text{ in}$	$x = 0.492 \text{ m}, y = 0 \text{ m}, z = 0 \text{ m}$
Location of moment center [†]	$x = 23.87 \text{ in}, y = 0 \text{ in}, z = 0 \text{ in}$	$x = 0.606 \text{ m}, y = 0 \text{ m}, z = 0 \text{ m}$
Location of MAC (1/4-chord) [†]	$x = 17.49 \text{ in}, y = 0 \text{ in}, z = 0 \text{ in}$	$x = 0.444 \text{ m}, y = 0 \text{ m}, z = 0 \text{ m}$

^{*} Whereas the other parameters in this table are defined specifically for this geometry, the aspect ratio is defined for a full aircraft configuration using the formula $(2 \times \text{semispan})^2 / (2 \times \text{area of one wing})$.

[†] (0, 0, 0) is the wing root-section leading edge at zero angle of attack.

Experimentally, the smooth ice shape was tested in two configurations: with and without 60-grit roughness. The motivation for including grit roughness comes from the Federal Aviation Administration (FAA) Advisory Circular 25-25A, which specifies a roughness height of 3 mm as guidance for developing ice accretions during aircraft performance and handling certification tests [36]. The advisory stipulates that simulated ice accretions should be representative of natural icing in terms of location, general shape, thickness, and texture. When no specific texture is substantiated, the advisory provides a default roughness level based on a full-scale aircraft. This default was scaled

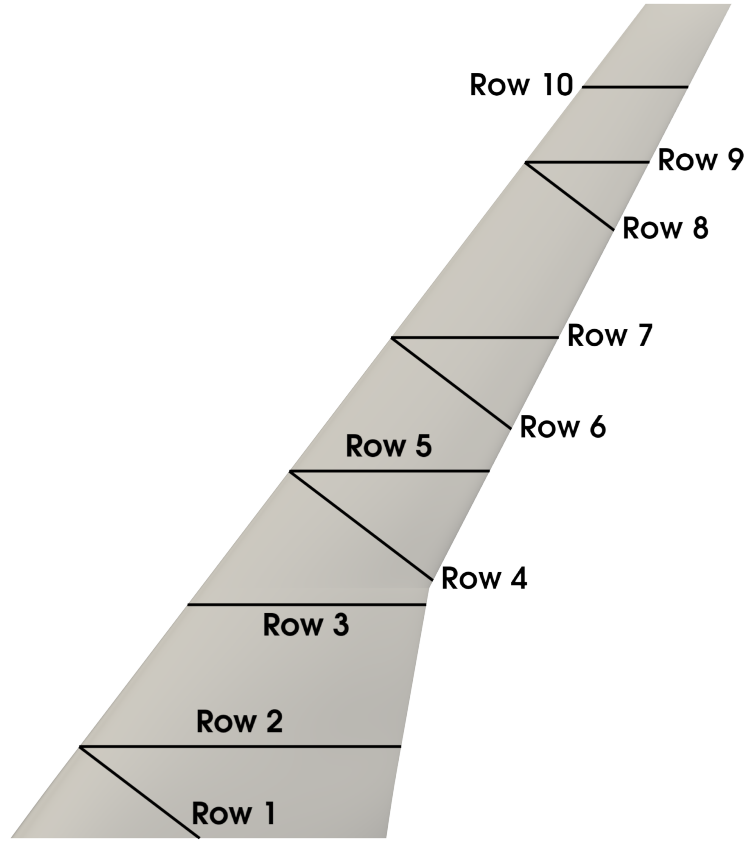


Figure 2: Pressure tap locations for the CRM65 wing.

Table 4: Pressure tap locations for the CRM65 wing [23, 25].

Row Number	Orientation	Normalized Spanwise Location*, y/b	Spanwise Location (m), y
1	Normal	0.11	0.16764
2	Streamwise	0.11	0.16764
3	Streamwise	0.28	0.42672
4	Normal	0.44	0.67056
5	Streamwise	0.44	0.67056
6	Normal	0.60	0.91440
7	Streamwise	0.60	0.91440
8	Normal	0.81	1.23444
9	Streamwise	0.81	1.23444
10	Streamwise	0.90	1.37160

*Spanwise location is provided for the leading-edge taps at rows 1, 4, 6, and 8 that were oriented normal to the leading edge.

down for the CRM65 wing, corresponding to a 60-grit surface. Figure 3 presents the experimental aerodynamic performance of all CRM65 configurations examined in this paper: tripped (no ice), smooth, smooth with grit, and high-fidelity ice. The crude approximation of the high-fidelity ice by the smooth ice leads to significant differences between the experimental aerodynamic loads predicted by these geometries, particularly in lift. Furthermore, up to approximately 15° angle

of attack, the smooth ice cases with and without roughness perform nearly identically. At higher angles, however, the smooth ice with roughness exhibits aerodynamic loads that better reflect the predictions of the high-fidelity ice shape.

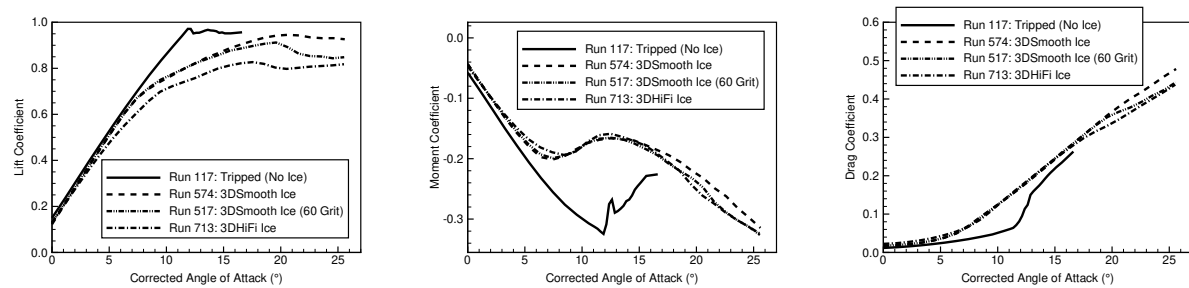


Figure 3: Experimental aerodynamic loads for CRM65 configurations.

3 Computational Methodology

In this work, large-eddy simulation is performed by solving the low-pass filtered Navier-Stokes equations using the Launch, Ascent, and Vehicle Aerodynamics (LAVA) framework [37]. LAVA supports Cartesian, curvilinear, and unstructured meshes through distinct solvers. The present work employs the LAVA Unstructured solver, which is a second-order accurate finite-volume solver that discretizes the governing equations on arbitrary polyhedral meshes. In this section, details regarding the flow solver and numerical discretization are presented followed by a summary of the mesh construction procedure using the LAVA Voronoi mesh generator within the LAVA framework.

3.1 Numerical Discretization

Details regarding the numerical discretization are described in [38, 39]. For the convective flux derivatives, two fluxes are evaluated at each face based on the left and right states. A non-dissipative asymptotically entropy-conservative and kinetic-energy preserving numerical flux [40] is evaluated based on left and right cell-center values, while a dissipative upwind flux is computed using the AUSMPW+ flux function following MUSCL [41] (i.e., monotonic upstream-centered scheme for conservation laws) reconstruction of the left and right states to the cell face. AUSMPW+ [42] is an improved version of the advection upstream splitting method that uses weighting functions to suppress spurious pressure wiggles that can occur, for example, in high-speed flows. Although limiting is generally unnecessary for subsonic flows, a minmod-type limiter [43] is applied to enhance robustness in exceptionally poor-quality cells that can affect gradient evaluation, for example, those near the high-fidelity ice. The non-dissipative and dissipative convective fluxes are then blended using a Ducros-type sensor [44]. The diffusive flux derivatives are computed using corrected face-averaged gradients to prevent even-odd decoupling [45]. The correction replaces the face-normal component of the face-averaged gradient with a directional derivative based on neighboring cell values, which is then used to evaluate the stress tensor at the face and compute the diffusive flux. Time marching is performed using the explicit third-order total variation diminishing Runge-Kutta scheme [46].

WMLES [47–50] is employed to efficiently capture the unsteady separated flows typical of icing applications [17]. The constant-coefficient Vreman model [51] is used as the subgrid-scale turbulence closure with a fixed coefficient of 0.08. This coefficient choice is based on prior studies on canonical turbulence cases [38]. An algebraic equilibrium wall model based on Musker [52] is used that incorporates optimized inverse tangent and logarithmic approximations for numerical efficiency. The wall-model exchange location is the first cell centroid off the wall and the resulting wall stress is directly imposed as the viscous flux at wall faces.

By default, the Musker-type wall model is applied to both the wing and ice surfaces. For the smooth ice, due to the sensitivity of the flow topology near the leading edge, a comparison is performed between the Musker-type wall model and a no-slip boundary condition. In addition, a simple rough-wall model is tested to examine the effect of mimicking the experimental grit roughness. The sensitivity to these boundary condition choices is examined in Sec. 4. The rationale behind the no-slip boundary condition is that for the smooth ice, little resolved turbulence may exist near the leading edge. Therefore, applying a fully turbulent wall stress could be excessive. For the no-slip boundary condition, the wall stress is estimated assuming a linear velocity profile between the wall and the first off-wall cell center. In contrast, when grit roughness is applied to the smooth ice experimentally, it is assumed that resolved turbulence is generated near the leading edge, and the rough-wall boundary condition provides a numerical mechanism to promote a similar onset of turbulence in the WMLES context. For the rough-wall case, the wall shear stress is

obtained from the rough-wall logarithmic law, where the friction velocity is estimated as [53]

$$u_\tau = \kappa u_\parallel \left[\ln \left(\frac{y_{\text{match}} - d}{y_0} \right) \right]^{-1}, \quad (1)$$

where $\kappa \approx 0.41$.

To estimate a suitable equivalent sand-grain roughness, the process begins with the 60-grit roughness used in the experiment [25]. Based on an empirical industry reference table [54], the corresponding root-mean-square (RMS) and average surface roughness (Ra) values are obtained. These values are then converted to equivalent sand-grain roughness values using the following scaling relations from [55]:

$$k_{s,\text{RMS}} = 3.100k_{\text{RMS}}, \quad k_{s,\text{Ra}} = 5.863k_{\text{Ra}}. \quad (2)$$

The results of this process are summarized in Table 5. In the fully rough regime, $y_0 = \exp(-8.0)k_s \approx 0.04k_s$ based on canonical studies [56]. For simplicity, in the present work, $d = 0$ is chosen and three values of y_0 are investigated: $y_0 = k_{s,\text{Ra}}$, $y_0 = k_{s,\text{RMS}}$, and $y_0 = 0.04k_{s,\text{Ra}} \approx 0.5 \mu\text{m}$. Because the roughness model is only applied to the smooth ice region near the leading edge and its purpose is primarily to trigger resolved turbulence as hypothesized for the experimental case with grit roughness, using $y_0 = k_s$ provides a conservative estimate that seeks to ensure a sufficient downward shift of the log-law profile to promote the generation of turbulence.

Table 5: Grit finish and estimated roughness values based on [54] and [55].

Grit Size	k_{RMS} (μm)	k_{Ra} (μm)	$k_{s,\text{RMS}}$ (μm)	$k_{s,\text{Ra}}$ (μm)
60	2.49	2.21	7.72	13.00

Finally, for the tripped wing without ice, numerical tripping is used near the leading edge of the wing to trigger a turbulent boundary layer. The upper-surface trip in the experiment was a 0.0025 in thick by 0.25 in wide continuous strip of flash tape [23]. A slightly different strategy is used in the present work to achieve a similar effect. A trip line is constructed projecting the line between [0.0101 m, 0 m, 0.01168 m] and [1.1626 m, 1.524 m, 0.0023 m] onto the wing surface. Along the trip line, a series of cylinders are defined relative to a minimum cell size, Δx . The width and radius of each cylinder are $8\Delta x$ and $2.5\Delta x$, respectively, and there is a gap of $3\Delta x$ between the cylinders. The effect of the trip line on the flow is implicitly enforced with penalty terms added to the governing equations using a source-term-based immersed-boundary method [57–61] formulated as a Brinkman penalization [62]. Any cells that are within the trip line cylinders have penalty terms applied to locally drive the flow velocity towards zero, promoting the generation of turbulence. The penalty terms include a relaxation factor where a unit relaxation factor corresponds to locally enforcing zero flow velocity and larger relaxation factors allow limited non-zero flow velocity. In the present work, a relaxation factor of five is used to balance numerical stability with triggering the onset of turbulence.

3.2 Voronoi Meshes

Unstructured meshes based on Voronoi diagrams were generated using the LAVA Voronoi mesher [63]. Voronoi meshes are advantageous because they can be generated in an automated manner, feature isotropic cells, and provide an explicit representation of the ice geometry within the mesh. These properties make Voronoi meshes a particularly attractive choice for WMLES. The Voronoi meshing

process requires a watertight surface triangulation defining the geometry of interest within a given domain. In this work, a free-air configuration is used to enable comparison against wind-tunnel corrected data. The rectangular far-field domain extends 50 m from the origin in the positive spanwise direction and 50 m from the origin in both the positive and negative streamwise and vertical directions, just over 100 mean aerodynamic chords (42.342 m) in all directions. A symmetry boundary condition is applied to the plane coinciding with the wing root, while far-field boundary conditions are enforced on the remaining domain extents. The details of the constructed Voronoi meshes are summarized in Table 6. The mesh names follow the convention L#, where each mesh resolution is defined relative to a single length-scale parameter, $L_s = \# \times 10^{-1} \text{mm}$, which enables the creation of progressively refined mesh families. The base cell size distribution is defined using telescoping regions that double in cell size with increasing distance from the surface, each with a thickness corresponding to eight cells. Reducing L_s by a factor of two globally refines the mesh and results in approximately a factor of four increase in the total cell count. This cell count increase is less than the factor of eight increase expected from simply refining each cell because both the seed spacing and layer thicknesses are reduced at the same time. Body centered cubic seeding is used, which generates truncated octahedron cells, because it was found to provide a good balance between the number of face evaluations and accuracy [63]. The minimum Δx values in Table 6 can be compared to the MAC given in Table 3 or a reference scallop length of 0.005 m for the high-fidelity ice [32]. In [5], additional relevant length scales for the smooth ice are discussed including a reference radius of curvature and a reference laminar boundary-layer height [5].

Table 6: Summary of Voronoi meshes.

CRM65 Configuration	Mesh Name	Mesh Descriptor	Minimum Δx (mm)	Cells (million)
3DHiFi Ice	L8	Coarse	0.8	6
	L4	Medium	0.4	22
	L2	Fine	0.2	97
3DSmooth Ice	L8	Coarse	0.8	6
	L4	Medium	0.4	21
	L2	Fine	0.2	90
	L1	Super Fine	0.1	445
	L0.5	Hyper Fine	0.05	470
Tripped (No Ice)	L8	Coarse	0.8	13
	L4	Medium	0.4	47
	L2	Fine	0.2	181

With the exception of the L0.5 mesh for the smooth ice, the relative size of the first layer was set to $4L_s$ on the upper surface of the wing and $8L_s$ on the lower surface of the wing for both the high-fidelity and smooth ice meshes. The cell size near the far-field was set to $10^4 L_s$. Additional targeted refinement regions were added to the base cell distribution. A single 8-cell-high layer with $2L_s$ spacing was added up to 30% of the chord on the upper surface. Moreover, two constant thickness regions with $2L_s$ cell spacing were specified, one 4 mm-thick region near the upper portion of the leading-edge ice and one 7.5 mm-thick region above the wing extending from the leading edge to 10% chord. Finally, a region with variable thickness equivalent to four cells in height with unitary relative length scale was specified near the ice. These cell size distributions are shown for the L4 and L2 high-fidelity ice meshes in Fig. 4. For the L0.5 mesh for the smooth ice, it was decided to

adjust the meshing strategy to concentrate the location of the most refined cells to the leading edge and to reduce the off-body refinement over the upper surface near the leading edge to maintain a reasonable cell count. Ultimately, the L0.5 mesh retained the same general cell size distribution as the L1 mesh with two exceptions: the constant-thickness regions used an absolute cell size of 0.4 mm instead of 0.2 mm and the ice refinement was restricted to the upper portion of the ice with an 8-cell-thick layer of 0.05 mm spacing. The effect of these adjustments are exhibited in Fig. 5. For the tripped configuration, two modifications were made to ensure that instabilities introduced by the tripping mechanism could be resolved downstream. First, the relative size of the first layer was set to $1.5L_s$ on the upper surface of the wing. Second, the variable-thickness region equivalent to four cells in height with a unit relative length scale was extended to 10% chord. These adjustments account for the increased cell count observed for the tripped meshes relative to the high-fidelity and smooth ice meshes.

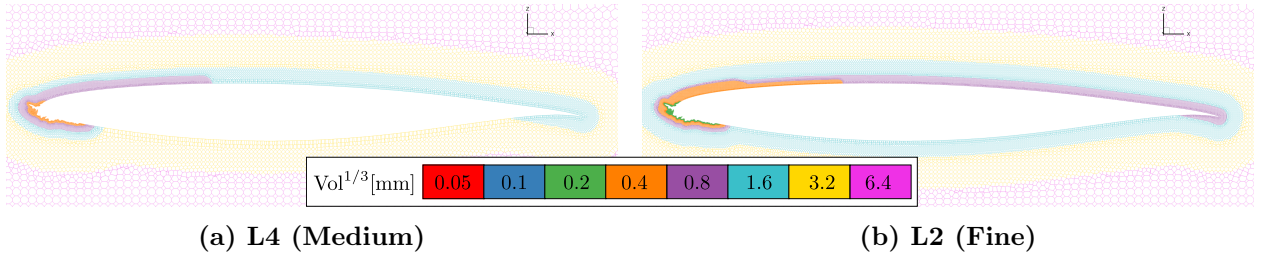


Figure 4: Comparison of L4 (Medium) and L2 (Fine) high-fidelity ice meshes at $y/b = 0.28$ (Row 3).

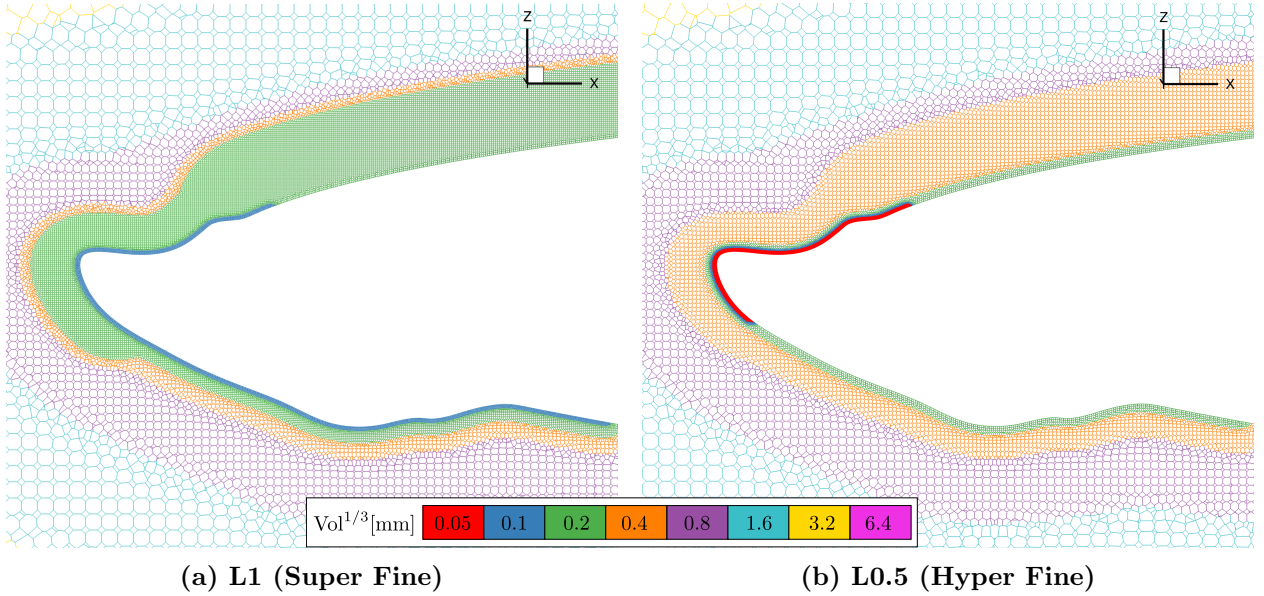


Figure 5: Comparison of meshing strategies for L1 (Super Fine) and L0.5 (Hyper Fine) smooth ice meshes at $y/b = 0.28$ (Row 3). Refinement for the L0.5 (Hyper Fine) mesh is localized to the region near the leading edge.

4 Results

Numerical results are compiled for the CRM65 wing with high-fidelity ice, smooth ice, and without ice. The simulations are run until the integrated loads reach a statistically stationary state, which occurs after approximately 40–100 convective time units (CTUs), where one CTU equals the mean aerodynamic chord divided by the freestream velocity magnitude. Experimental data for comparison is taken from [13, 24, 25] and is based on wind-tunnel corrected values as the computational domain uses a free-air configuration. The numerical simulations use the corresponding wind-tunnel corrected angles of attack. The aerodynamic moments are taken about the MAC (1/4-chord) location.

4.1 High-Fidelity Iced Wing

The high-fidelity ice was simulated across angles of attack ranging from 2.185° to 24.524° . The integrated lift, pitching moment, and drag coefficients are plotted in Fig. 6. The numerical results capture the overall experimental trends observed, including the local minimum in pitching moment around 8.423° , where the lift coefficient begins to depart from the linear regime. This inflection point is significant because it corresponds to the “usable” lift defined by [13] (following the criteria introduced by [64]) as the maximum lift coefficient that can be obtained before significant flow separation develops over the wing, which can lead to a rapid shift in the aerodynamic center and potentially result in serious control design problems or even aircraft uncontrollability. On the L8 mesh, there is a consistent over-prediction of lift, under-prediction of pitching moment, and over-prediction of drag despite the overall experimental trend being reproduced. In contrast, the L4 and L2 meshes give significantly more accurate results and are consistent with each other, suggesting convergence toward the experiment. The errors in the integrated coefficients relative to the experiment on the L2 mesh are given in Table 7. The drag differences are given in counts, where one count equals a drag coefficient of 10^{-4} . The largest percentage errors in lift and drag occur at the lowest angle of attack considered, where the magnitudes of the aerodynamic coefficients are small. For pitching moment, the largest percentage error is near the local minimum at 8.423° , where the moment changes rapidly with angle of attack.

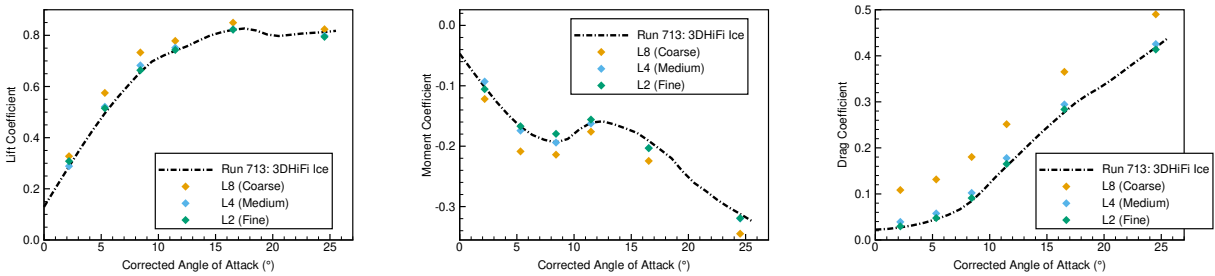


Figure 6: Aerodynamic loads for the CRM65 wing with high-fidelity leading-edge ice compared to the experiment.

Table 7: Errors in aerodynamic loads relative to the experiment for the CRM65 wing with high-fidelity leading-edge ice on the L2 (Fine) mesh.

α ($^\circ$)	ΔC_L ($\times 10^{-2}$)	C_L Error (%)	ΔC_M ($\times 10^{-3}$)	C_M Error (%)	ΔC_D (counts)	C_D Error (%)
2.185	2.06	7.17	-3.36	3.29	23.62	8.66
5.319	1.96	3.96	0.89	-0.53	25.27	5.61
8.423	0.71	1.09	14.37	-7.41	65.39	7.74
11.477	0.32	0.43	5.13	-3.18	47.90	2.99
16.530	0.02	0.03	-12.16	6.36	60.34	2.17
24.524	-1.87	-2.30	-6.97	2.23	-46.25	-1.11

Time-averaged pressure profiles at several spanwise stations are shown for four representative angles of attack in Figs. 7 (2.185 $^\circ$), 8 (8.423 $^\circ$), 9 (16.530 $^\circ$), and 10 (24.524 $^\circ$). The numerical pressure profiles exhibit a convergence trend toward the experimental data with mesh refinement, indicating that the agreement in the integrated loads on the L2 mesh reported in Table 7 is due to accurately reproducing the flow as opposed to error cancellation. Furthermore, when the flow is fully or partially attached, as in Figs. 7 and 8, there are clear differences in upper-surface suction across mesh resolutions. However, in the case of significant separation as in Figs. 9 and 10, there is little difference in upper-surface suction across the meshes. On the L8 (Coarse) mesh, a steep over-prediction of upper-surface suction near the leading edge at $y/b = 0.60$ (Row 7) is observed in Figs. 7–10 that decreases in magnitude with increasing angle of attack. Furthermore, a shallow region of upper-surface suction over-prediction aft of the ice shape is present in Figs. 7 and 8. In Figs. 8–10, a region of pressure over-prediction on the lower surface near the leading edge is also present. These features contribute to the persistent drag offset noted in Fig. 6, which arises primarily from differences in pressure drag.

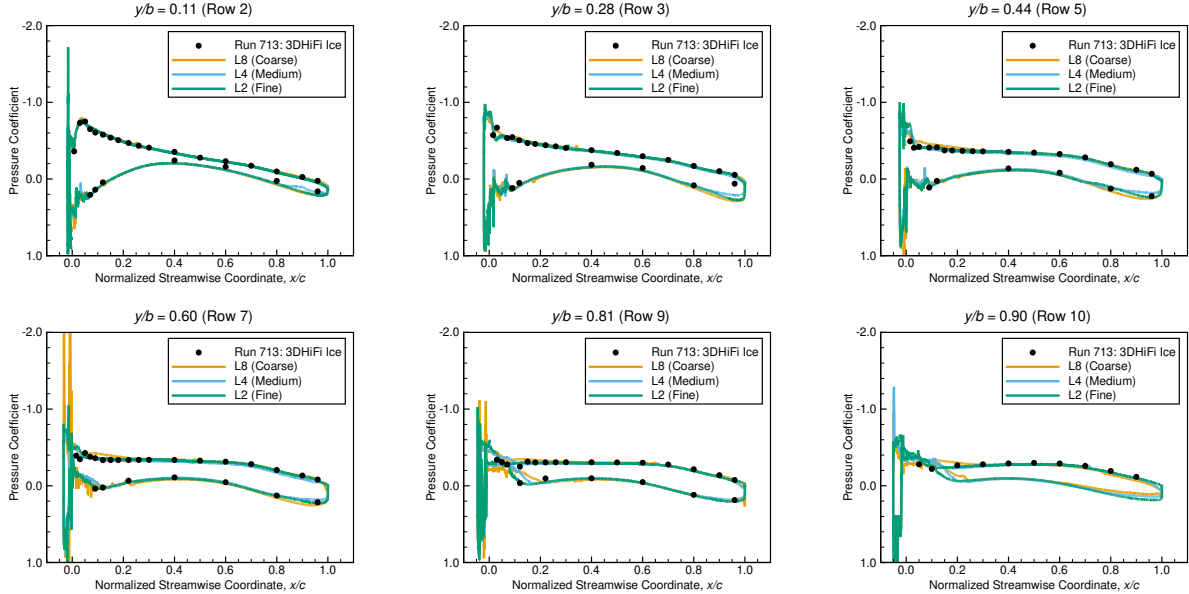


Figure 7: Time-averaged pressure profiles for the CRM65 wing with high-fidelity leading-edge ice at 2.185 $^\circ$ compared to the experiment.

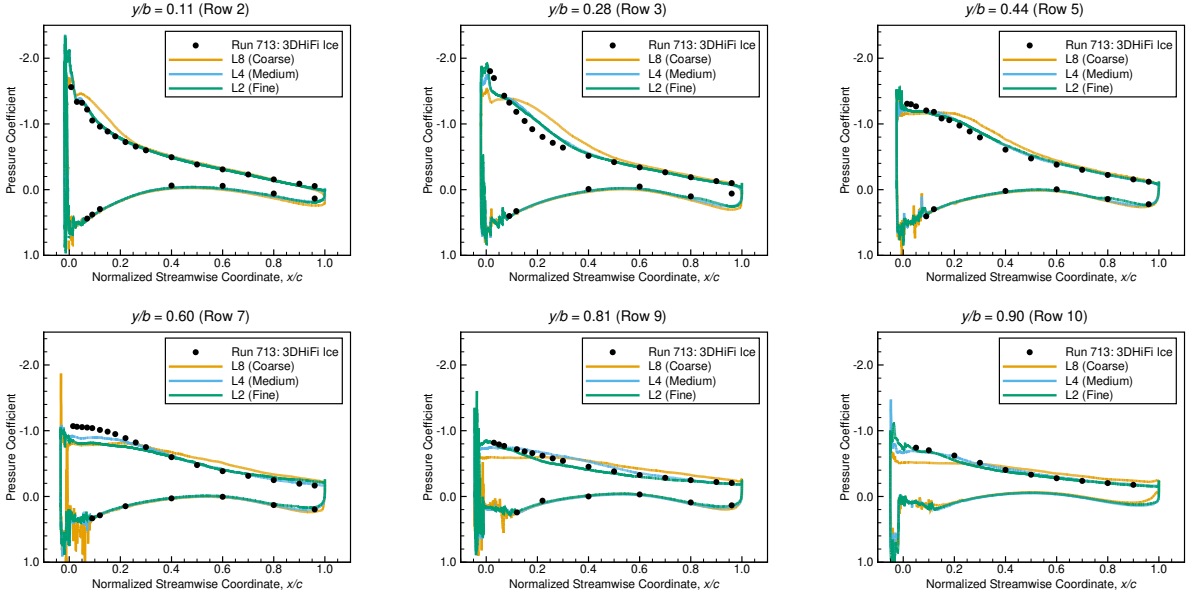


Figure 8: Time-averaged pressure profiles for the CRM65 wing with high-fidelity leading-edge ice at 8.423° compared to the experiment.

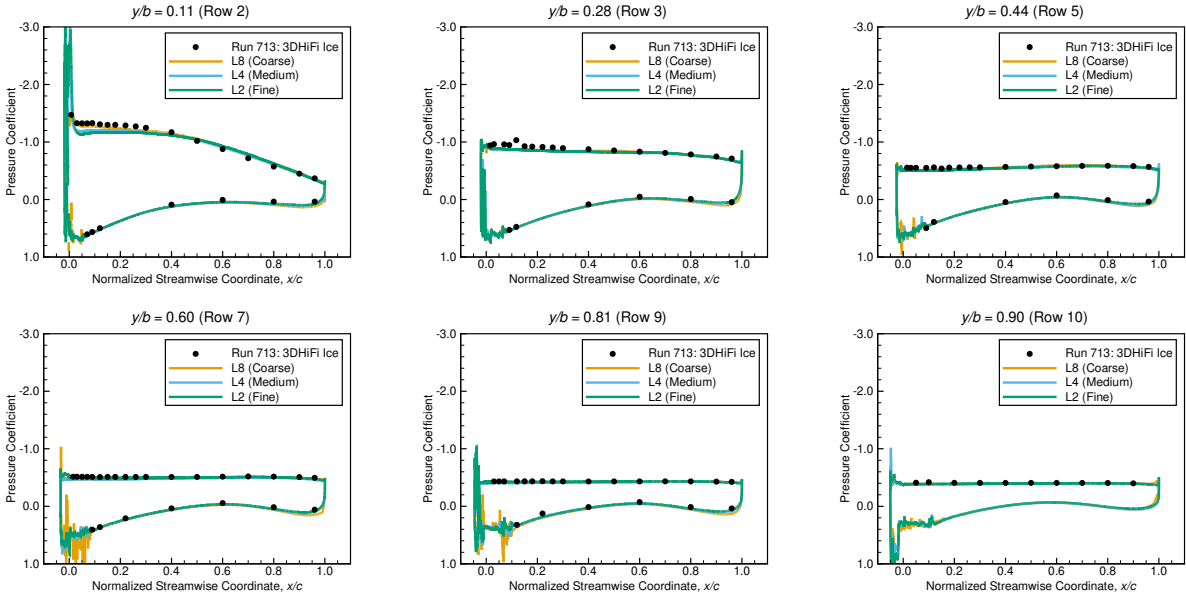


Figure 9: Time-averaged pressure profiles for the CRM65 wing with high-fidelity leading-edge ice at 16.530° compared to the experiment.

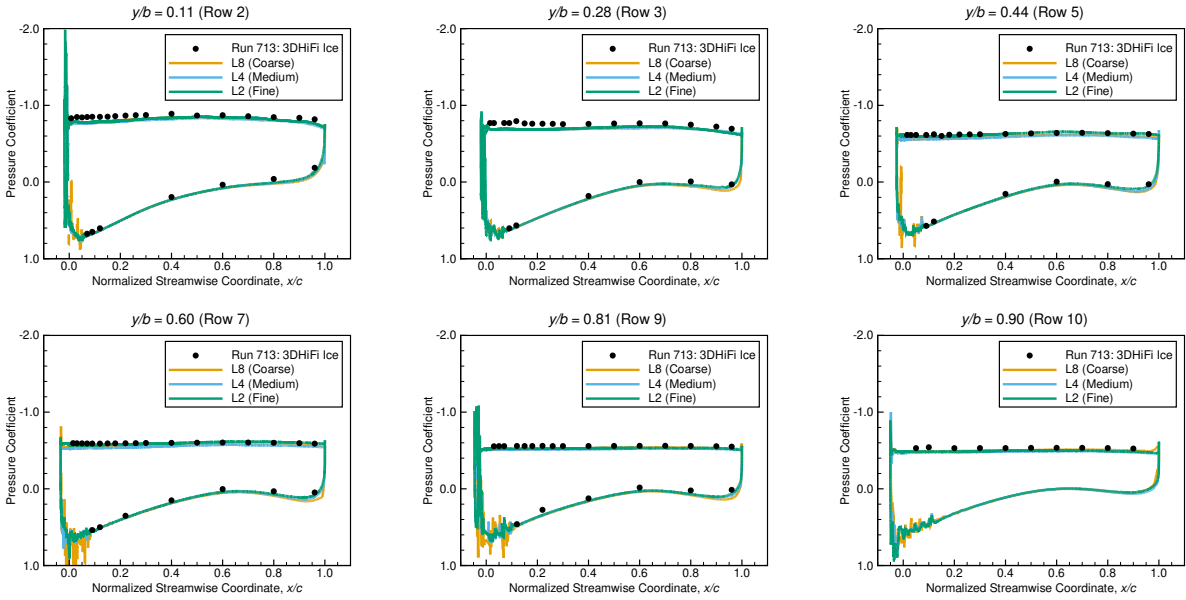


Figure 10: Time-averaged pressure profiles for the CRM65 wing with high-fidelity leading-edge ice at 24.524° compared to the experiment.

Figures 11 and 12 show experimental oil flow visualizations compared to time-averaged and instantaneous skin-friction contours on the L2 mesh at angles of attack of 2.185° and 8.423° , respectively. Examining Fig. 11, at 2.185° streamwise streaks can be observed at quasi-regular intervals along the full span of the wing that break down into less coherent turbulent structures toward the trailing edge. These vortical structures are clearly reproduced by the numerical skin-friction contours and appear to be generated by the leading-edge ice scallops, which align the flow into a distinctive streamwise pattern. At 8.423° , Fig. 12 shows that as flow separation progresses inboard from the wing tip with increasing angle of attack, the streak structures become confined to an inboard region. Furthermore, a localized area of higher time-averaged skin friction is observed near the midboard around the Yehudi break, indicating a region separating a pair of leading-edge spanwise vortices.

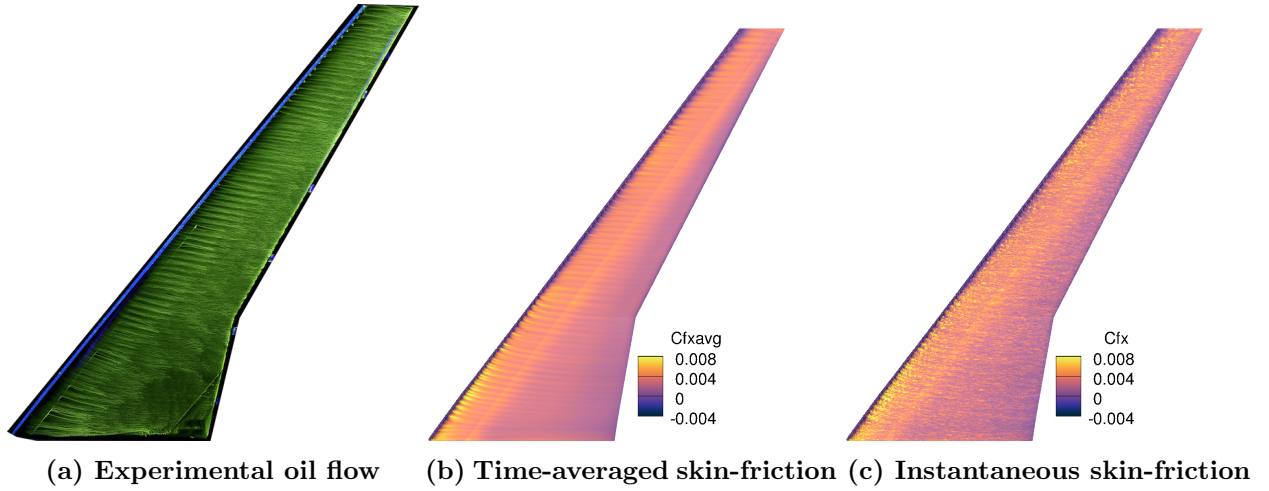


Figure 11: Experimental oil flow compared to contours of time-averaged and instantaneous coefficient of skin-friction for the CRM65 wing with high-fidelity leading-edge ice on the L2 (Fine) mesh at 2.185° .

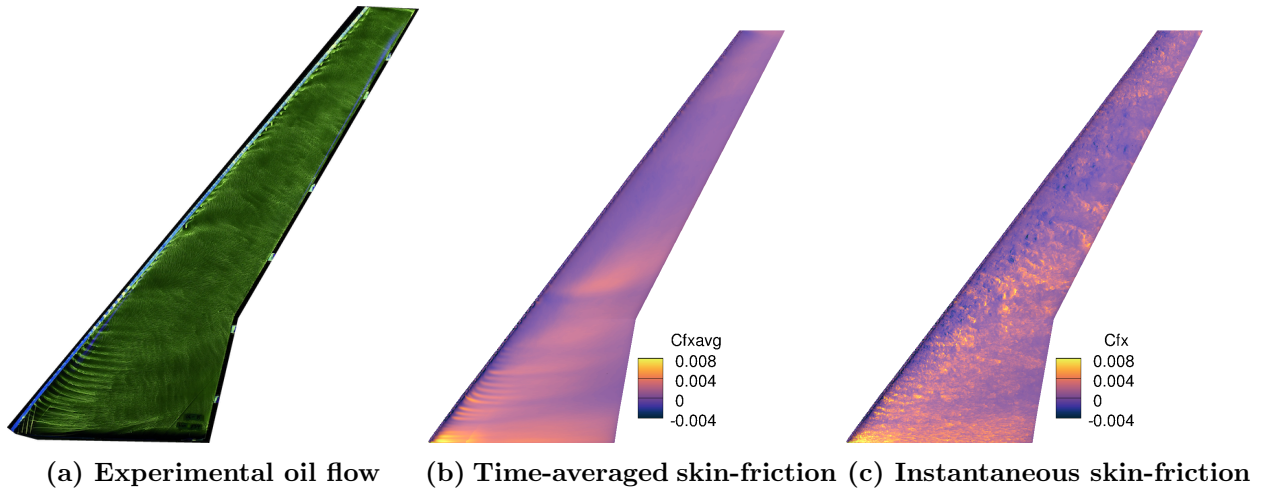


Figure 12: Experimental oil flow compared to contours of time-averaged and instantaneous coefficient of skin-friction for the CRM65 wing with high-fidelity leading-edge ice on the L2 (Fine) mesh at 8.423° .

4.2 Smooth Iced Wing

Similar to the meshes used for the high-fidelity iced wing, the smooth iced wing was simulated at a single angle of attack of 8.458° across the L8 (Coarse), L4 (Medium), and L2 (Fine) meshes. However, the error magnitudes of the integrated coefficients on the L2 mesh were significantly larger than those on the analogous L2 mesh for the high-fidelity iced wing. Therefore, additional simulations were performed across the L1 (Super Fine) and L0.5 (Hyper Fine) meshes. All of these simulations used the default Musker-type wall model applied to both the wing and ice surfaces. The errors in the integrated coefficients relative to the experiment on the L2, L1, and L0.5 meshes are given in Table 8. On the L2 mesh, the errors in lift, pitching moment, and drag are 4.71%, 14.22%, and -27.83% , respectively. In contrast, for the high-fidelity iced wing, the corresponding errors in Table 7 are 1.09%, -7.41% , and 7.74%. On the L1 mesh, the error magnitudes are considerably reduced; however, the drag error remains relatively large. Only on the L0.5 mesh are the error magnitudes of the integrated coefficients reduced below the corresponding maximum error magnitudes observed for the high-fidelity ice case on the L2 mesh across all angles of attack considered.

Table 8: Errors in aerodynamic loads relative to the experiment for the CRM65 wing with smooth leading-edge ice at 8.458° .

Mesh	ΔC_L ($\times 10^{-2}$)	C_L Error (%)	ΔC_M ($\times 10^{-3}$)	C_M Error (%)	ΔC_D (counts)	C_D Error (%)
L2 (Fine)	3.35	4.71	-27.85	14.22	-255.73	-27.83
L1 (Super Fine)	2.87	4.03	-13.95	7.12	-86.13	-9.37
L0.5 (Hyper Fine)	1.72	2.42	-0.02	0.01	-27.11	-2.95

Figure 13 shows time-averaged pressure profiles at several spanwise stations. With the exception of the most inboard station at $y/b = 0.11$ (Row 2), the three coarsest meshes show a large over-prediction in upper-surface suction near the leading edge followed by a rapid decrease in pressure coefficient around 20% chord. At $y/b = 0.44$ (Row 5) and $y/b = 0.60$ (Row 7), this steep decrease transforms into a shallow under-prediction in upper-surface suction that persists until the trailing edge. This under-prediction is also observed toward the trailing edge at $y/b = 0.81$ (Row 9) and $y/b = 0.90$ (Row 10), albeit to a lesser degree. In contrast, the two finest meshes have only a minor over-prediction in upper-surface suction near the leading edge and a much smaller under-prediction toward the trailing edge. As suggested in [5], the high roughness of the high-fidelity ice promotes the generation of resolved turbulence near the leading edge that can be readily captured by WMLES, whereas the smooth ice case lacks such a mechanism, which makes it more difficult to simulate. Furthermore, it appears that two families of solutions are present in Fig. 13: one family formed by the three coarsest mesh solutions and one family formed by the two finest mesh solutions. Since the L1 and L0.5 meshes have y^+ values within the viscous sublayer ($y^+ < 5$), this could represent a critical threshold needed for resolved turbulence to develop sufficiently close to the leading edge in a manner consistent with the experiment.

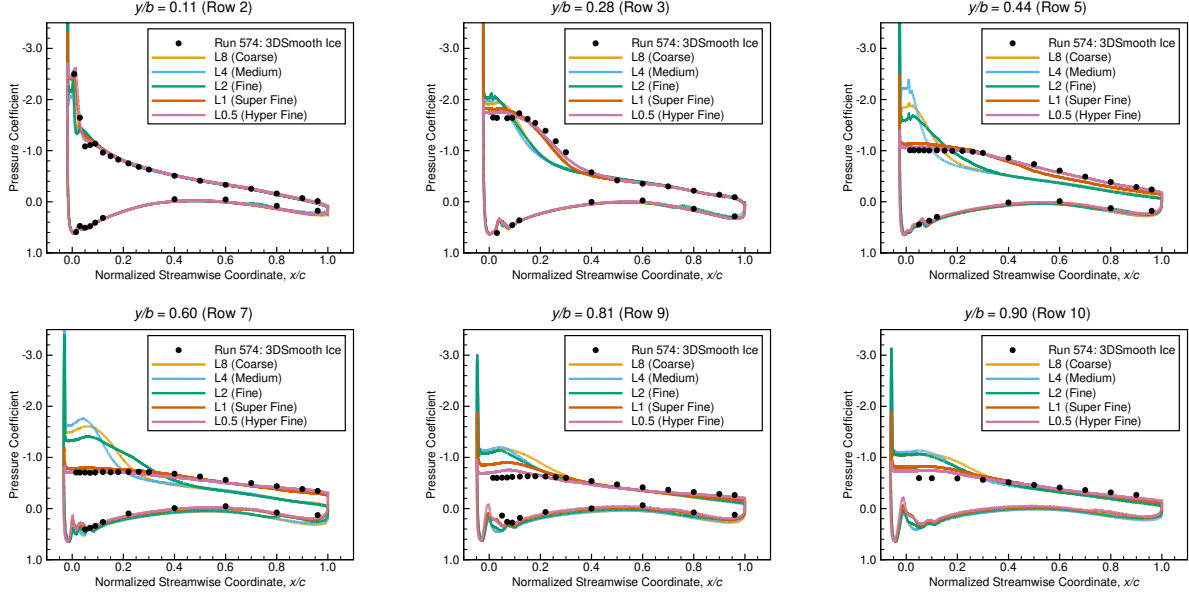


Figure 13: Time-averaged pressure profiles for the CRM65 wing with smooth leading-edge ice at 8.458° compared to the experiment.

Figures 14 and 15 show experimental oil-flow visualizations compared with time-averaged and instantaneous skin-friction contours at 8.458° on the L2 and L0.5 meshes, respectively. The experimental oil flow indicates a main spanwise vortex originating near the inboard leading edge and extending diagonally toward the trailing edge past the Yehudi break. In contrast, the L2 mesh predicts the origin of the vortex closer to the midboard and spurious high and low skin-friction regions along the span. This indicates an incorrect separation pattern and is consistent with the over-prediction of upper-surface suction appearing in the pressure profiles. However, the L0.5 mesh captures the primary vortex much better and shows a higher skin-friction near the wing tip that appears to be consistent with the experiment.

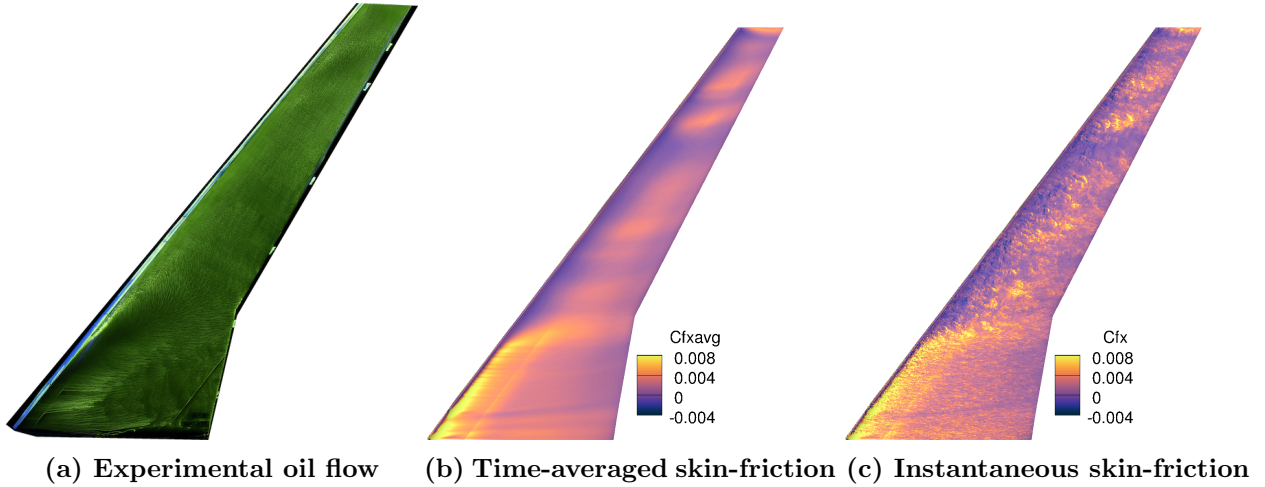


Figure 14: Experimental oil flow compared to contours of time-averaged and instantaneous coefficient of skin-friction for the CRM65 wing with smooth leading-edge ice on the L2 (Fine) mesh at 8.458° .

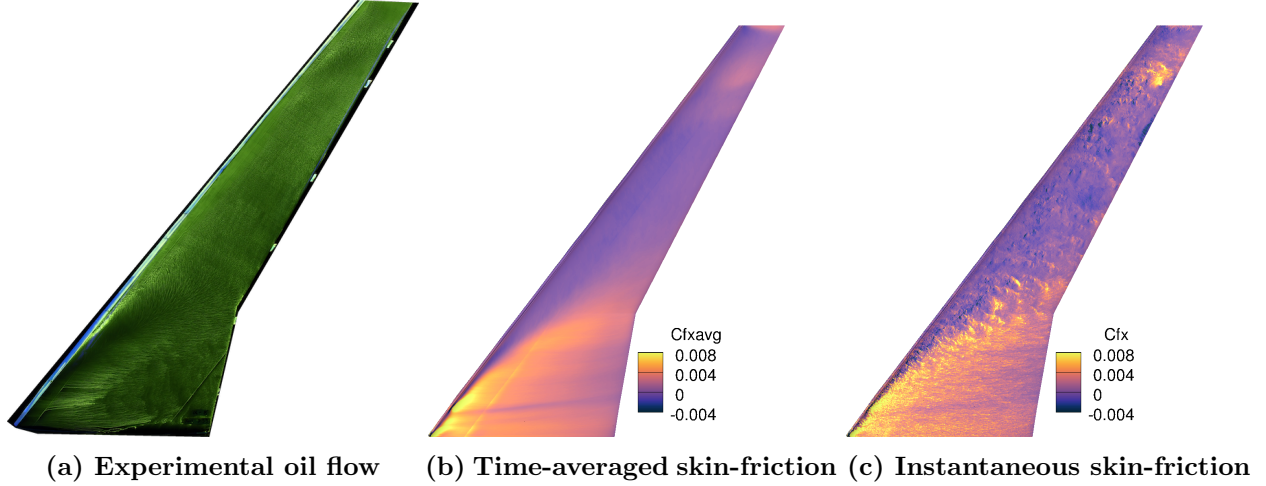


Figure 15: Experimental oil flow compared to contours of time-averaged and instantaneous coefficient of skin-friction for the CRM65 wing with smooth leading-edge ice on the L0.5 (Hyper Fine) mesh at 8.458° .

4.3 Smooth Iced Wing with No-Slip Boundary Condition

In [5], a no-slip boundary condition was applied to the smooth ice along with a region immediately aft of the smooth ice along the upper surface until an approximate flow reattachment location based on the experimental oil flow. This boundary condition was applied to a mesh with 1.2 billion cells and a minimum cell size similar to that of the L0.5 mesh in the present work. The purpose of this procedure was to better resolve the flow topology near the leading edge of the smooth iced wing. Here, a similar strategy is investigated for the L0.5 mesh. However, instead of applying the no-slip boundary condition from the smooth ice up to an approximate flow reattachment location, it is only applied to the smooth ice shape. Table 9 gives the errors in the integrated coefficients relative to the experiment with both the default Musker-type and no-slip boundary conditions. The errors relative to the experiment are slightly higher with the no-slip boundary condition but the difference is quite small. The corresponding pressure profiles in Fig. 16 suggest that this small difference results mostly from slightly greater over-prediction of upper-surface suction near the leading edge and under-prediction near the trailing edge. This is most noticeable at $y/b = 0.81$ (Row 9). The reason for the agreement between boundary conditions might be because the Musker’s law reduces to a linear velocity profile within the viscous sublayer, which is also enforced by the no-slip condition. Therefore, because the L0.5 mesh has y^+ values within the viscous sublayer near the leading edge, this could be the reason that the default Musker-type and the no-slip boundary conditions produce similar results. Finally, it is observed that the error magnitudes on the 1.2 billion-cell mesh in [5] are comparable to those on the 470 million-cell L0.5 mesh in the present work, despite the latter mesh having a factor of 2.55 fewer cells. This suggests that mesh construction is an unclosed problem and motivates the use of mesh adaption to select an appropriate number of cells to achieve a given accuracy level.

Table 9: Errors in aerodynamic loads relative to the experiment for the CRM65 wing with smooth leading-edge ice on the L0.5 (Hyper Fine) mesh at 8.458°.

Boundary Condition	ΔC_L ($\times 10^{-2}$)	C_L Error (%)	ΔC_M ($\times 10^{-3}$)	C_M Error (%)	ΔC_D (counts)	C_D Error (%)
Musker-Type	1.72	2.42	-0.02	0.01	-27.11	-2.95
No-Slip	2.32	3.26	-7.23	3.69	-44.69	-4.86

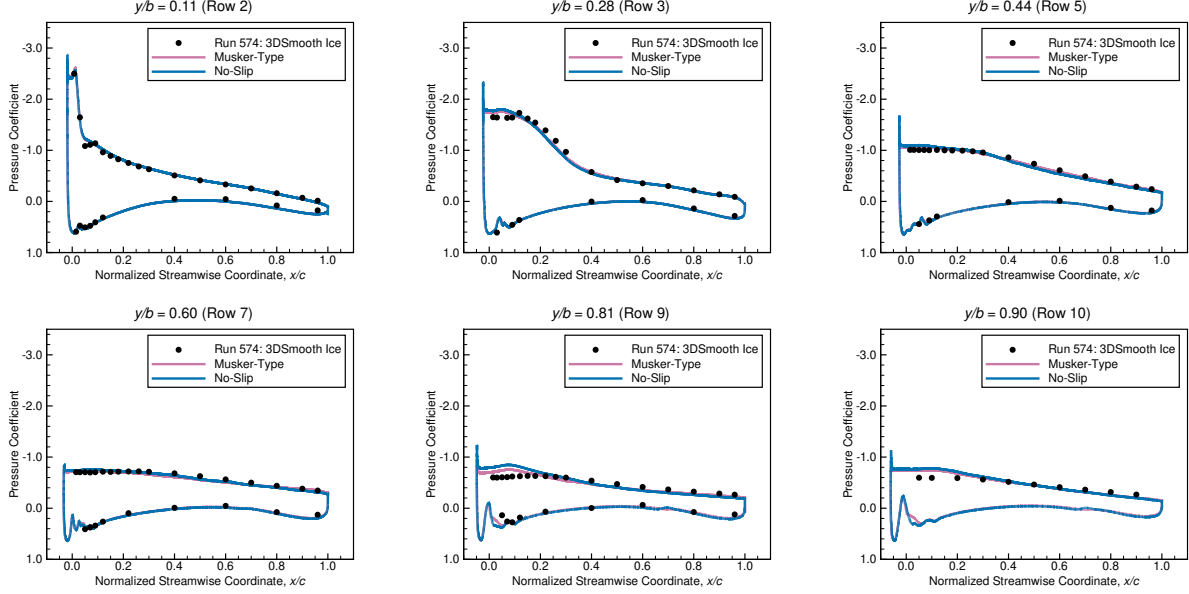


Figure 16: Time-averaged pressure profiles for the CRM65 wing with smooth leading-edge ice on the L0.5 (Hyper Fine) mesh at 8.458° compared to the experiment.

4.4 Smooth Iced Wing with Roughness

The smooth iced wing needed a much finer mesh to achieve similar accuracy compared to the high-fidelity iced wing because the lack of leading-edge roughness delayed the onset of turbulence near the leading edge. To address this, a simple rough-wall model is tested to try and promote resolved turbulence near the leading edge, perhaps circumventing the need for a very fine mesh. To begin, three values of y_0 are assessed on the L2 mesh at 8.458°: $y_0 = 0.04k_{s,Ra} \approx 0.5 \mu\text{m}$, $y_0 = k_{s,RMS} = 7.72 \mu\text{m}$, and $y_0 = k_{s,Ra} = 13.00 \mu\text{m}$. As discussed in Secs. 2 and 3, these values are based on the 60-grit roughness used in the experiment informed by FAA Advisory Circular 25-25A and estimates of equivalent sand-grain roughness. The errors in the integrated coefficients relative to the experiment are given in Table 10. With $y_0 = 0.5 \mu\text{m}$, the error magnitudes are slightly reduced compared to using no roughness, but are still quite large. However, with $y_0 = 7.72 \mu\text{m}$ and $y_0 = 13.00 \mu\text{m}$, the error magnitudes are greatly reduced. Examining the corresponding pressure profiles in Fig. 17 shows that $y_0 = 13.00 \mu\text{m}$ results in slightly less over-prediction of upper-surface suction near the leading edge, especially at $y/b = 0.81$ (Row 9), and is therefore selected for simulating additional angles of attack and mesh resolutions. Note that $y_0 = 50.00 \mu\text{m}$ was also tested, but found to be unstable.

Table 10: Errors in aerodynamic loads relative to the experiment for the CRM65 wing with smooth leading-edge ice with grit roughness on the L2 (Fine) mesh at 8.458°.

y_0 (μm)	ΔC_L ($\times 10^{-2}$)	C_L Error (%)	ΔC_M ($\times 10^{-3}$)	C_M Error (%)	ΔC_D (counts)	C_D Error (%)
0.50	2.66	3.72	-23.62	12.10	-214.76	-23.70
7.72	-0.40	-0.56	10.28	-5.27	-21.13	-2.33
13.00	-0.51	-0.72	11.23	-5.75	-10.12	-1.12

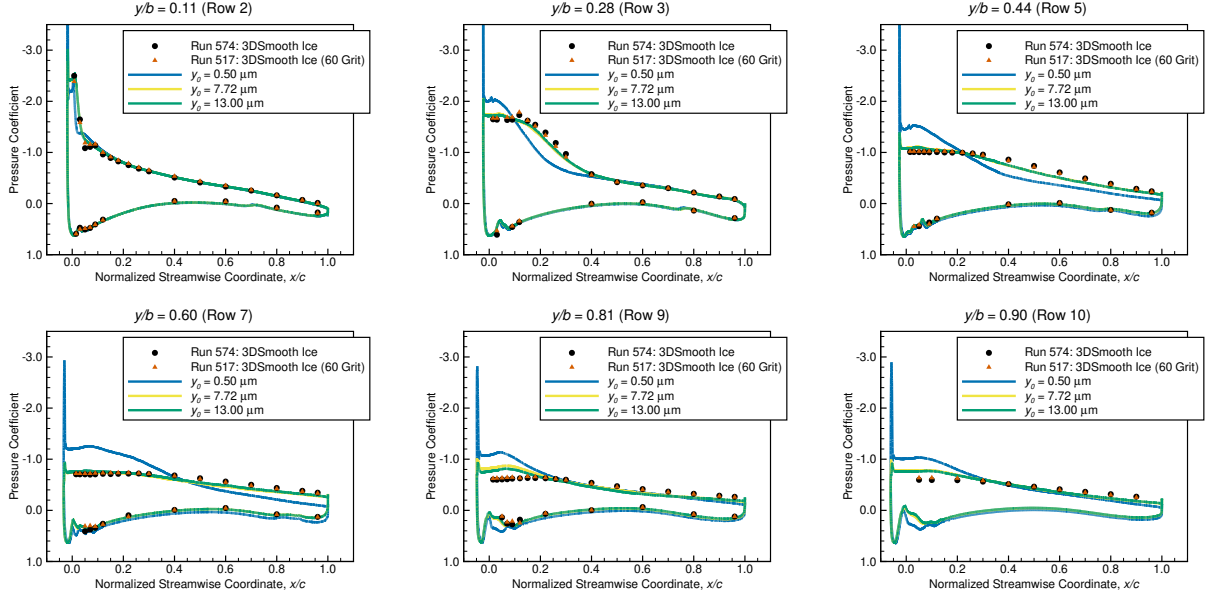


Figure 17: Time-averaged pressure profiles for the CRM65 wing with smooth leading-edge ice with grit roughness on the L2 (Fine) mesh at 8.458° compared to the experiment.

Using $y_0 = 13.00 \mu\text{m}$, the smooth iced wing with roughness was simulated at six angles of attack ranging from 2.189° to 24.600° and three mesh levels: L8 (Coarse), L4 (Medium), and L2 (Fine). The integrated aerodynamic coefficients (lift, pitching moment, and drag) are shown in Fig. 18. The numerical results appear to reproduce the main experimental trends. At the higher angles of attack, where there are clear differences between the experimental values with and without grit roughness, the numerical predictions converge toward those with grit roughness. This is consistent with the introduction of the rough-wall model. The errors in the integrated coefficients relative to the experiment on the L2 mesh are given in Table 11. The maximum error magnitudes for lift, pitching moment, and drag are 6.66%, 5.75%, and 3.16%, respectively, which are all smaller than the corresponding maximum error magnitudes for the high-fidelity iced wing on the L2 mesh across all angles of attack considered. This demonstrates that incorporating roughness effects for the smooth ice can achieve comparable accuracy relative to the high-fidelity iced wing on similar mesh resolutions. Moreover, this indicates that incorporating some mechanism into the numerical set-up that generates resolved turbulence near the leading edge results in more closely representing the physics of the problem.

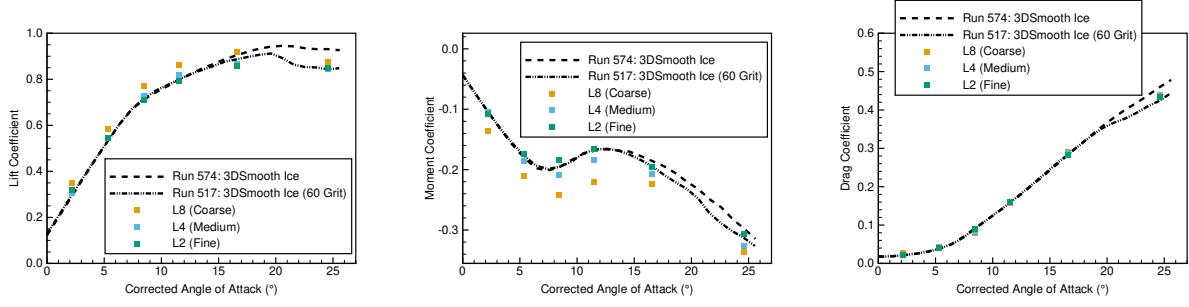


Figure 18: Aerodynamic loads for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) compared to the experiment.

Table 11: Errors in aerodynamic loads relative to the experiment for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) on the L2 (Fine) mesh.

α (°)	ΔC_L ($\times 10^{-2}$)	C_L Error (%)	ΔC_M ($\times 10^{-3}$)	C_M Error (%)	ΔC_D (counts)	C_D Error (%)
2.189	1.99	6.66	-4.07	3.93	6.06	2.83
5.344	1.07	2.00	1.93	-1.09	12.12	3.16
8.458	-0.51	-0.72	11.23	-5.75	-10.12	-1.12
11.514	-0.67	-0.84	1.79	-1.07	15.25	0.96
16.583	-2.72	-3.07	-1.25	0.65	-12.51	-0.44
24.600	0.56	0.67	5.42	-1.74	94.25	2.23

Time-averaged pressure coefficient distributions at several spanwise stations are shown for four representative angles of attack in Figs. 19 (2.189°), 20 (8.458°), 21 (16.583°), and 22 (24.600°). In general, the numerical pressure profiles approach the experimental profiles as the mesh is refined. At 8.458°, there is an over-prediction of upper-surface suction near the leading edge of the wing and an under-prediction toward the trailing edge, which are both reduced with refinement. At 16.583° and 24.600°, the numerical results appear to converge toward the experimental profiles with grit roughness. Furthermore, at 16.583°, there is a slight under-prediction of upper-surface suction near the leading edge that persists on the L4 and L2 mesh levels. Moreover, the mesh sensitivity observed at higher angles is smaller than at lower angles due to the large amount of flow separation present.

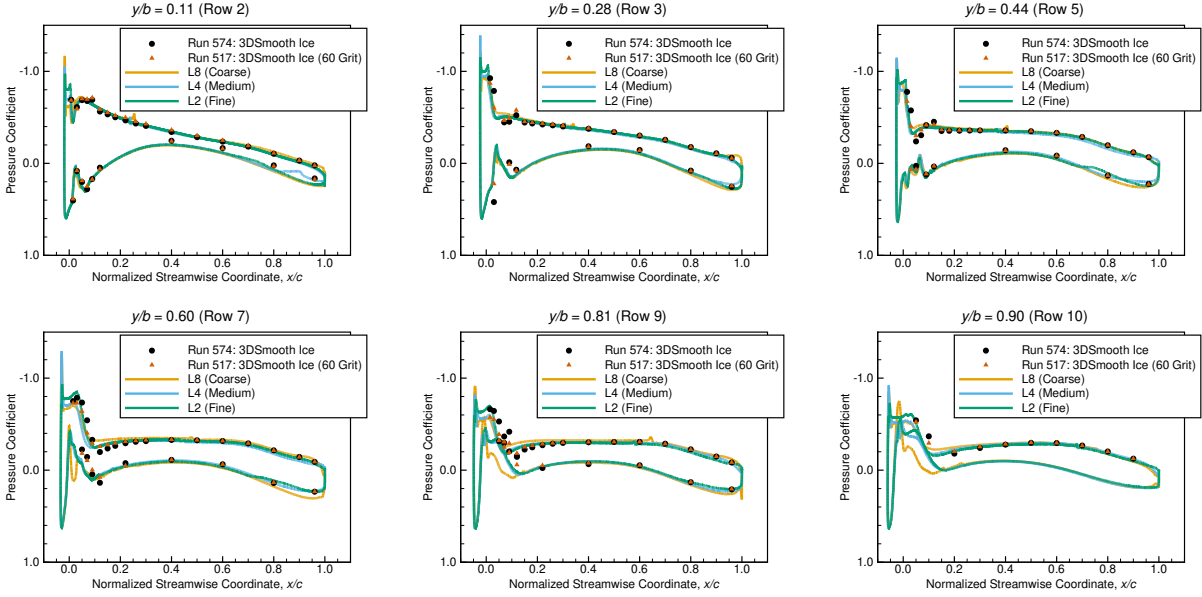


Figure 19: Time-averaged pressure profiles for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) at 2.189° compared to the experiment.

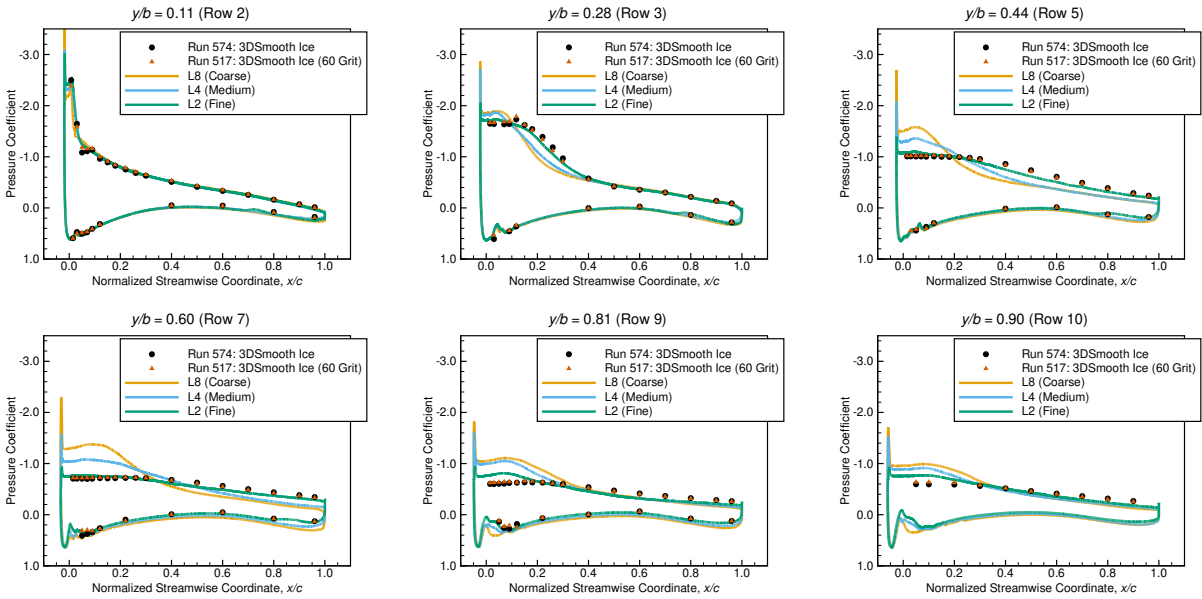


Figure 20: Time-averaged pressure profiles for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) at 8.458° compared to the experiment.

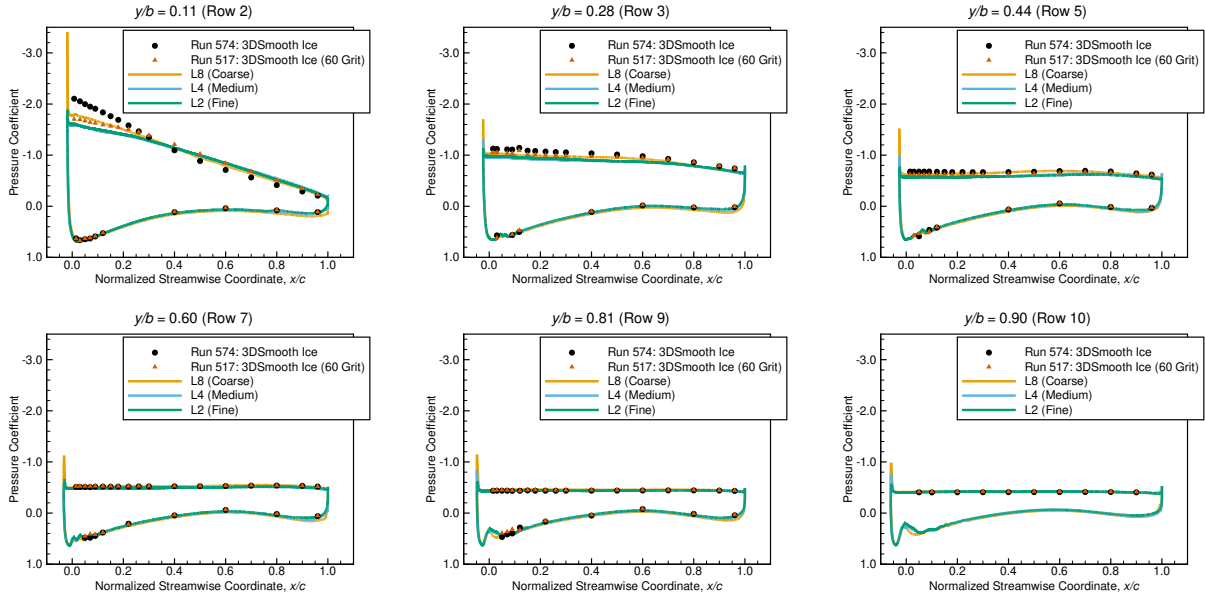


Figure 21: Time-averaged pressure profiles for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) at 16.583° compared to the experiment.

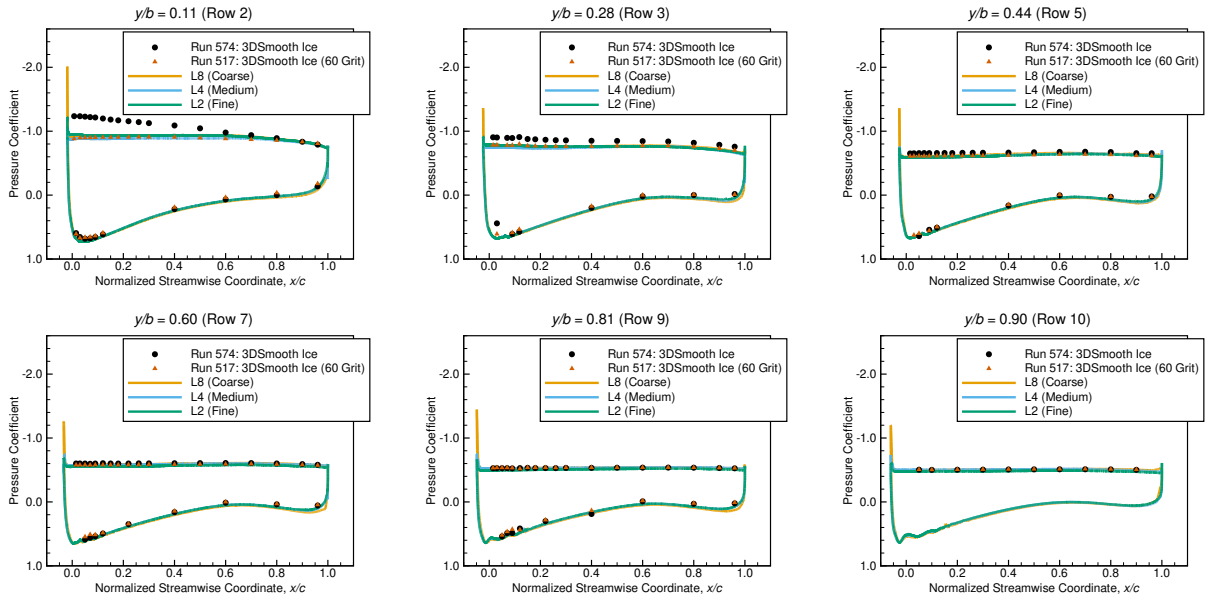


Figure 22: Time-averaged pressure profiles for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) at 24.600° compared to the experiment.

Figures 23 and 24 compare experimental oil-flow visualizations with time-averaged and instantaneous skin-friction contours on the L2 meshes at 2.189° and 8.458° , respectively. At 2.189° , leading-edge spanwise perturbations are observed in the experimental oil flow that appear in the numerical skin-friction contours as regions of lower skin friction. At 8.458° , the numerical contours show a spanwise vortex extending from the inboard leading edge toward the trailing edge past the Yehudi break, which is consistent with the experimental oil flow. A region of higher skin friction near the wing tip is also observed, similar to both the experimental oil flow and the L0.5 mesh result in Fig. 15.

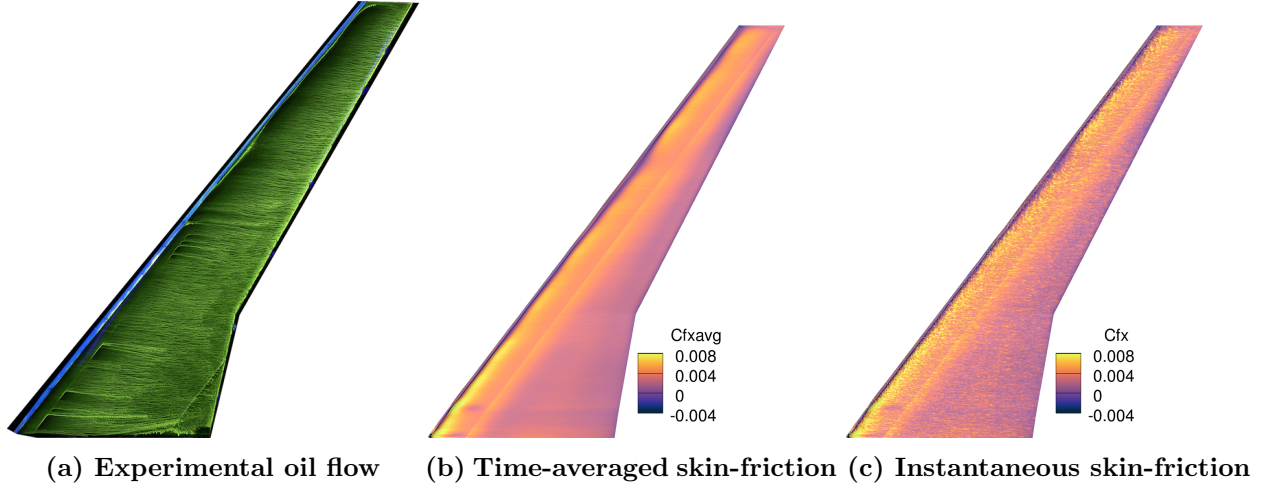


Figure 23: Experimental oil flow compared to contours of time-averaged and instantaneous coefficient of skin-friction for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) on the L2 (Fine) mesh at 2.189° .

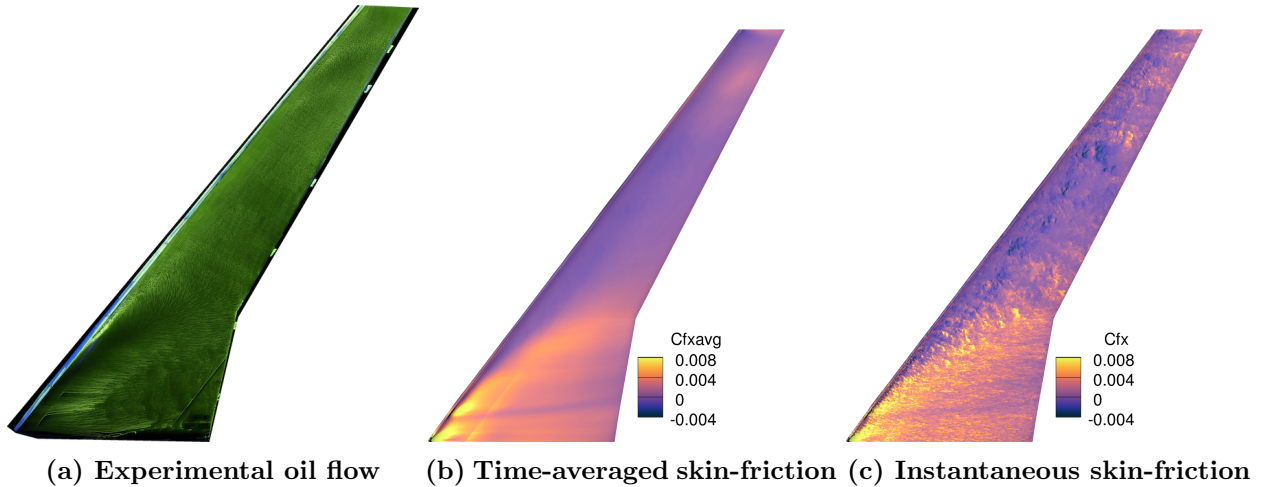


Figure 24: Experimental oil flow compared to contours of time-averaged and instantaneous coefficient of skin-friction for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) on the L2 (Fine) mesh at 8.458° .

4.5 Tripped Wing and Performance Differences

To assess the predicted aerodynamic degradation induced by icing, simulations of the tripped wing without ice were performed at five angles of attack ranging from 2.204° to 16.617° . The resulting aerodynamic coefficients are shown in Fig. 25. The corresponding errors are reported in Table 12. There is minor mesh sensitivity observed at the two lowest angles of attack where the flow is mostly attached. The next two angles of attack immediately preceding stall show the largest mesh sensitivity, where the flow is starting to exhibit outboard separation. Moderate mesh sensitivity is observed at the highest angle of attack. On the L2 mesh, a slight under-prediction in lift occurs at the two highest angles. However, the plateauing of lift at the post-stall angle of 16.617° is captured.

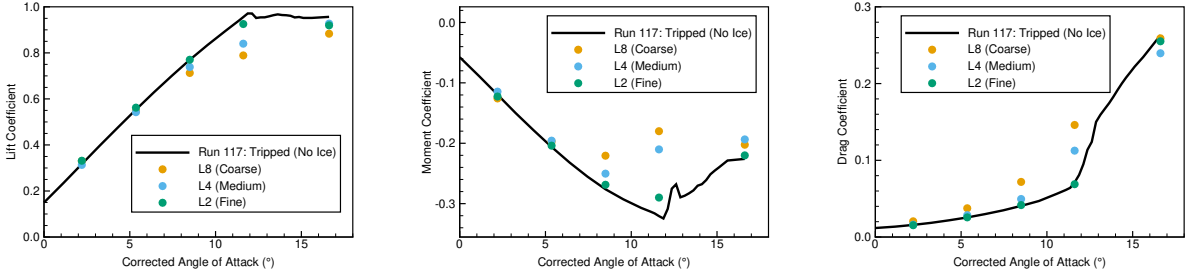


Figure 25: Aerodynamic loads for the tripped CRM65 wing without ice compared to the experiment.

Table 12: Errors in aerodynamic loads relative to the experiment for the tripped CRM65 wing on the L2 (Fine) mesh.

α ($^\circ$)	ΔC_L ($\times 10^{-2}$)	C_L Error (%)	ΔC_M ($\times 10^{-3}$)	C_M Error (%)	ΔC_D (counts)	C_D Error (%)
2.204	1.46	4.59	-3.12	2.61	-3.89	-2.46
5.357	0.86	1.56	2.17	-1.05	0.85	0.33
8.497	0.01	0.02	7.38	-2.67	10.58	2.61
11.616	-3.02	-3.16	30.71	-9.58	-23.31	-3.27
16.617	-3.70	-3.87	5.85	-2.59	-78.58	-2.99

To further examine the observed mesh sensitivity in the aerodynamic loads, Figs. 26 and 27 show pressure coefficient distributions at several spanwise stations for the angles of attack of 8.497° and 11.616° , respectively, compared with the experiment. Steep pressure spikes are observed on the upper surface close to 5% chord that are associated with the numerical tripping procedure. At 8.497° , Fig. 26 shows that both the L8 and L4 meshes predict spurious outboard separation not present in the experiment. The spurious separation is not observed on the L2 mesh. Similarly, at 11.616° , both the L8 and L4 meshes predict separation from the inboard to the outboard portions of the wing, which is inconsistent with the experiment except at $y/b = 0.90$ (Row 10). This discrepancy in numerical and experimental pressure profiles likely contributes to the large differences in the integrated loads observed between L8 and L4 mesh simulations and the experiment shown in Fig. 25. In contrast, the L2 mesh predicts attached flow over most of the wing with the exception of the most outboard spanwise station, which is more consistent with the experiment.

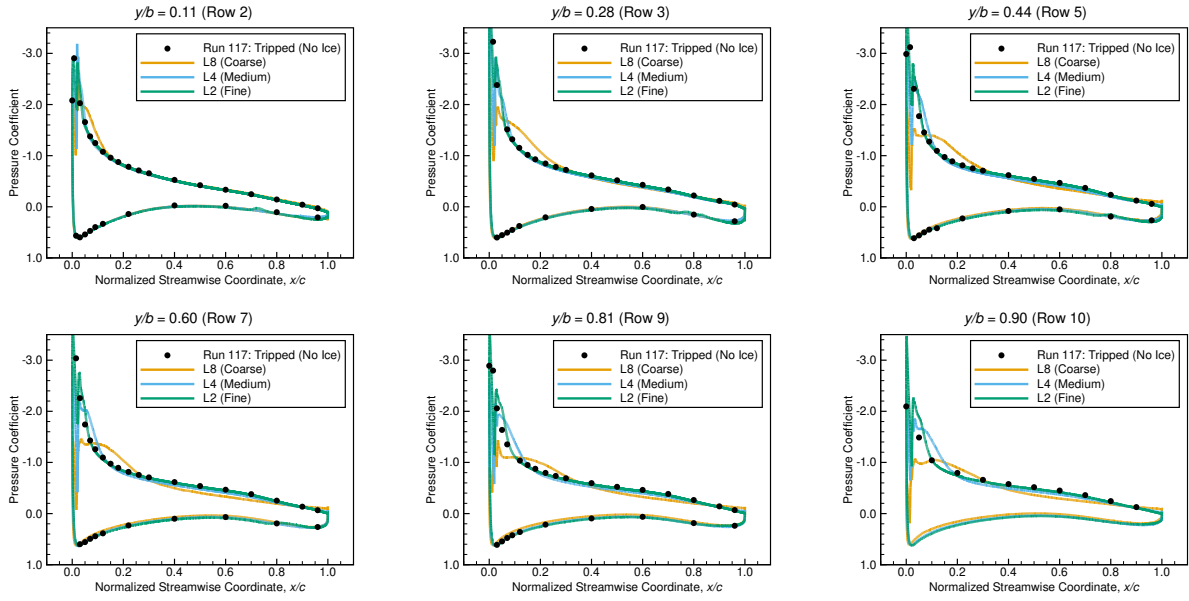


Figure 26: Time-averaged pressure profiles for the tripped CRM65 wing without ice at 8.497° compared to the experiment.

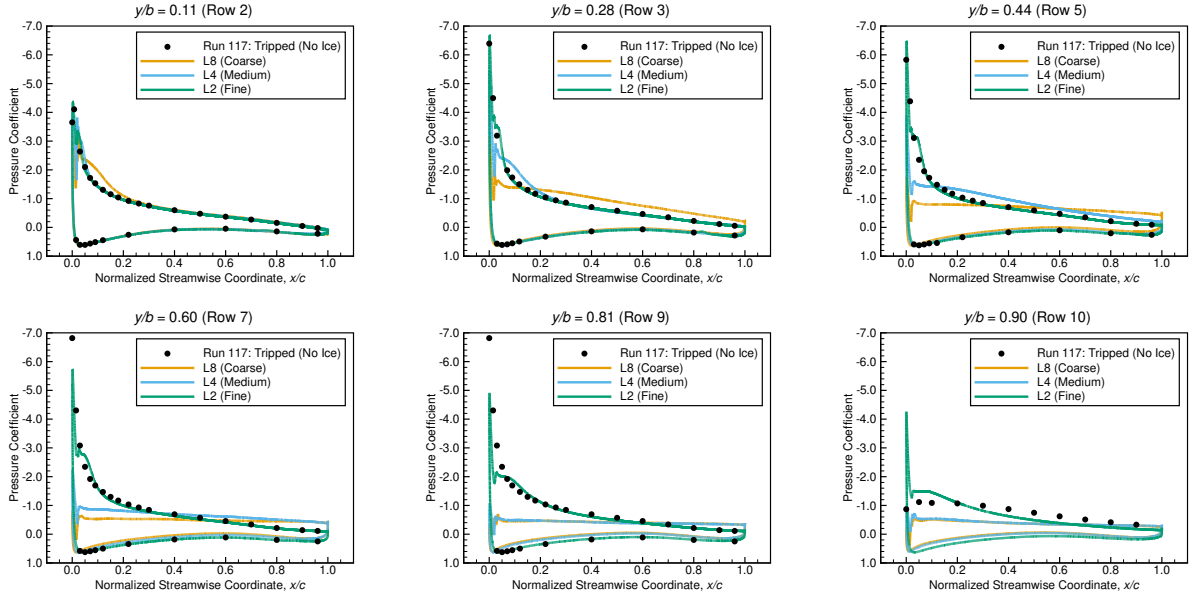


Figure 27: Time-averaged pressure profiles for the tripped CRM65 wing without ice at 11.616° compared to the experiment.

To better understand why the L8 and L4 meshes predict greater midboard and outboard separation compared to the L2 mesh at the two angles of attack preceding stall, it is helpful to examine the skin-friction contours across all three mesh levels, shown in Fig. 28. The two coarser meshes exhibit a region immediately aft of the leading edge having low skin friction that is noticeably larger compared to the finest mesh. This observation may be explained by the fact that the radii of the tripping cylinders is proportional to the local mesh size, such that the tripping on the coarser mesh produces larger turbulent structures that can be resolved by the mesh. Near stall, the larger cylin-

ders may promote premature midboard and outboard separation on the coarser meshes. However, following stall, this effect could be less pronounced because the flow is already nearly separated in the physical flow, which could in turn explain the reduced mesh sensitivity observed at 16.617° .

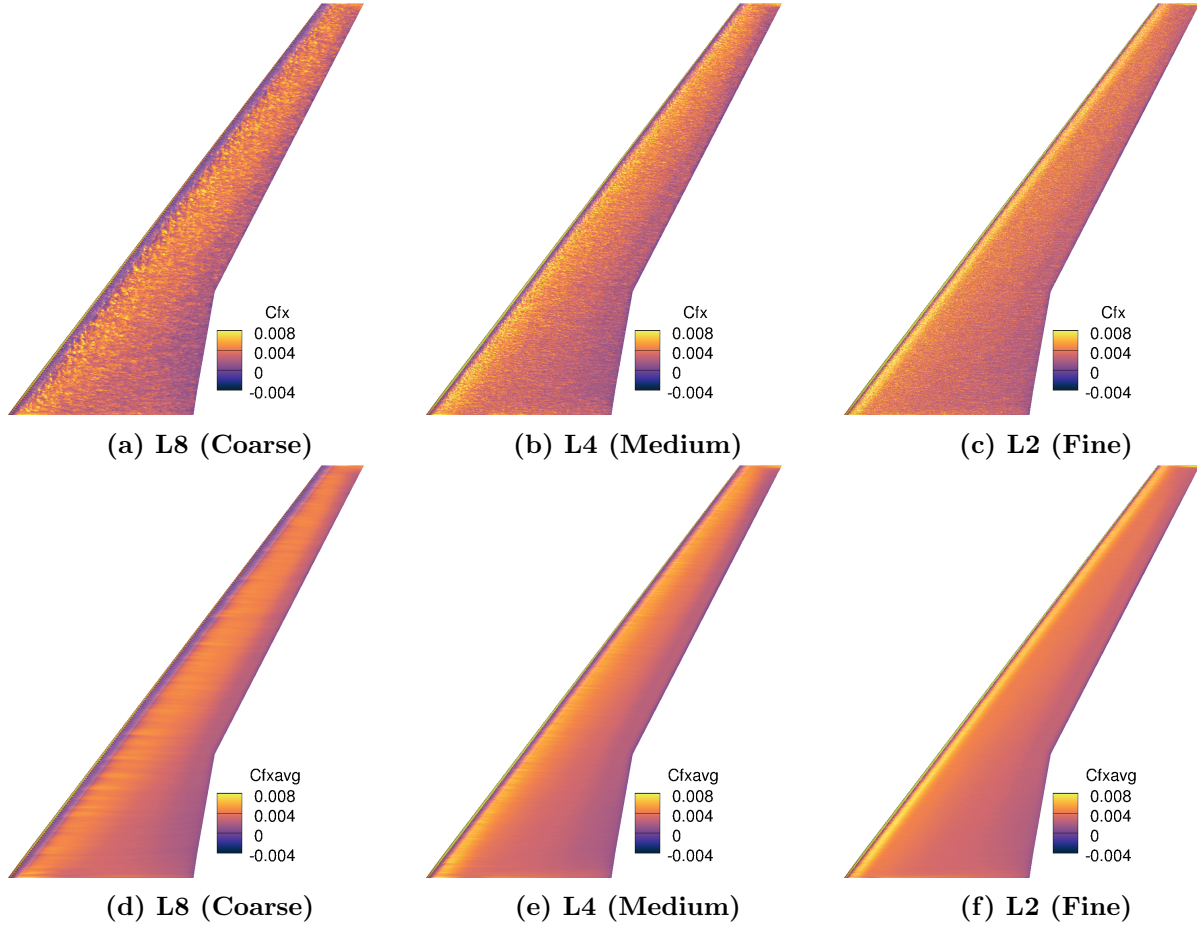


Figure 28: Instantaneous (a–c) and time-averaged (d–f) coefficient of skin-friction for the tripped CRM65 wing without ice at 2.204° .

Finally, Fig. 29 gives the differences in aerodynamic loads for the smooth leading edge ice with roughness ($y_0 = 13.00\mu\text{m}$) and high-fidelity ice on the L2 (Fine) meshes compared to the experiment. The coefficient differences are computed as the coefficient for the tripped wing without ice minus the coefficient for the iced configuration. For the experimental differences, only the values corresponding to the numerical simulation angles of attack are plotted and successive experimental values are connected with straight lines. For the experiment, the smooth ice with roughness mimics the trends of the high-fidelity for the pitching moment and drag, and follows a similar lift pattern. However, the high-fidelity ice shows greater lift degradation, especially at the higher angles of attack. The numerical results capture the main features of the experiment. Specifically, the gradual increase in difference magnitude up to the stall angle of the tripped wing near 11° is captured, followed by a decrease. Compared to the smooth ice with roughness, the high-fidelity ice is predicted to have a greater difference in lift and the relative ordering of pitching moment and drag coefficient differences at low angles of attack is preserved, consistent with the experiment. The largest discrepancies between the numerical and experimental differences are present in lift at the two highest angles of attack, mainly due to the under-prediction of lift for the tripped wing configuration.

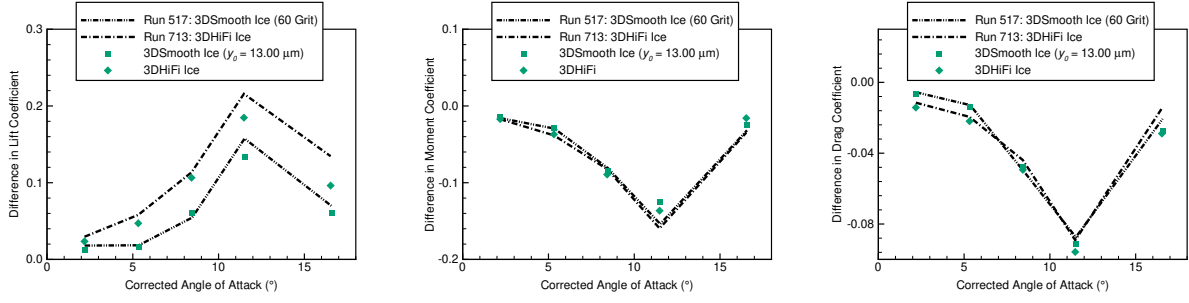


Figure 29: Difference in aerodynamic loads for the CRM65 wing with smooth leading-edge ice with roughness ($y_0 = 13.00 \mu\text{m}$) and high-fidelity ice compared to the experiment.

4.6 Computational Cost

The computational costs of the simulations across CRM65 configurations and mesh resolutions are summarized in Table 13. The reported costs are representative and remain similar across angles of attack. All simulations were performed in double precision leveraging graphics processing unit (GPU) nodes, most of which were based on the NVIDIA GH200 Grace Hopper Superchip architecture. The number of GPUs used for each simulation was determined based on the number of mesh cells and memory requirements. The energy usage per convective time unit is reported as a relative performance metric across simulations. Energy consumption is estimated from peak power recordings of 1000 W per compute node, where each compute node has one GPU.

Table 13: Summary of computational costs.

CRM65 Configuration	Mesh Name	Cells (million)	Averaged Δt ($\times 10^{-7}$ s)	Number of GPUs used	Wall-clock time per CTU (h)	Energy usage per CTU (MJ)
3DHiFi Ice	L8	6	7.09	2	0.24	1.73
	L4	22	2.93	4	0.52	7.48
	L2	97	1.54	20	0.72	51.85
3DSmooth Ice	L8	6	7.36	1	0.16	0.57
	L4	21	3.54	4	0.28	4.08
	L2	90	1.67	20	0.54	38.89
	L1	445	0.87	50	2.24	403.53
	L0.5	470	0.35	50	5.78	1040.25
3DSmooth Ice ($y_0 = 13.00 \mu\text{m}$)	L8	6	7.60	1	0.15	0.54
	L4	21	3.70	4	0.29	4.16
	L2	90	1.85	20	0.49	35.19
Tripped (No Ice)	L8	13	7.50	2	0.16	1.18
	L4	47	3.71	5	0.51	9.14
	L2	181	1.84	25	0.78	70.44

The energy consumption across subsequent mesh levels is expected to increase by about factor of eight since the number of cells increases by approximately a factor of four and the time step size is roughly halved. An even more accurate estimate of the relative performance factor can be obtained by multiplying the ratio of successive cell counts by the inverse ratio of successive averaged time

step sizes. For example, for the smooth ice, going from the L4 (Medium) mesh to the L2 (Fine) mesh, the cell count increases by a factor of 4.29 and the averaged time step size decreases by a factor of 2.12, which results in an estimated factor of 9.08 increase in energy consumption. However, dividing the actual energy usage values results in a factor of 9.53 increase. The L0.5 (Hyper Fine) mesh is an exception to this increase trend because it is not an in-family mesh. Relative to the L1 (Super Fine) mesh, the cell count only increases by a factor of 1.06, while the averaged time step decreases by a factor of 2.49, leading to a factor of 2.58 increase in energy consumption, which is almost entirely due to the reduced time step.

Based on the numerical results in Tables 8 and 11, the smooth ice with roughness ($y_0 = 13.00 \mu\text{m}$) on the L2 (Fine) mesh and the smooth ice on the L0.5 (Hyper Fine) mesh achieve comparable accuracy. However, computing the relative performance factor indicates that the smooth ice with roughness reduces the computational cost by a factor of 29.56. Furthermore, if the L0.5 (Hyper Fine) mesh were constructed as an in-family mesh, the relative performance factor is expected to increase to at least 64, assuming a factor of eight increase in computational cost across consecutive mesh levels.

5 Conclusions

This work investigated WMLES for the purpose of predicting swept iced wing aerodynamics with high-fidelity and smooth leading edge ice shapes. For the high-fidelity ice, the numerical results captured the experimental trends, including demonstrating convergence toward the measured pressure profiles and reproducing key features of the oil flow visualizations. This good agreement was partially attributed to the high roughness of the high-fidelity ice shape, which fostered the development of turbulence near the leading edge, and led to accurate results at reasonable mesh sizes.

In contrast, for the smooth ice, large numerical errors were observed in the integrated coefficients, along with inaccurate pressure profiles particularly near the leading edge and an incorrect separation pattern was identified in the skin-friction contours. However, the much finer L1 and L0.5 meshes showed considerable improvement, potentially because they allow flow instabilities near the leading edge to be resolved and grow, leading to the onset of turbulence consistent with the experiment.

A simple roughness model was used to avoid the need for a very fine mesh with the smooth ice and to enable comparison to experimental results incorporating 60-grit roughness in accordance with FAA guidelines on artificial ice shapes. Notably, simulating the smooth ice with grit roughness on the L2 mesh led to a factor of 29.56 reduction in computational cost compared to simulating the smooth ice on the L0.5 mesh, while achieving similar accuracy. This reduction factor in computational cost is expected to increase to a factor of 64 if the L0.5 mesh were to be constructed as an in-family mesh.

Overall, the results demonstrate that WMLES can reasonably reproduce swept iced wing aerodynamics and performance degradation due to icing at practical computational costs, as long as resolved turbulence is appropriately triggered. Future work may focus on more sophisticated roughness modeling, alternative scale-resolving approaches like hybrid RANS-LES, and advanced iced configurations such as the high-lift CRM and transonic truss-braced wing.

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