

Multifidelity Numerical–Experimental Database for Tailplane Aerodynamics

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The aerospace community is striving to reduce the emissions of new aircraft in accordance with the objectives set by the Clean Sky 2 program. To fulfill these goals, unconventional tail configurations must be explored. The MONNALISA project delivered an open-access multifidelity database of the nonlinear aerodynamic characteristics of unconventional tails. Geometries are parameterized considering the aspect ratio, the taper ratio, and the sweep and dihedral angles. Additional degrees of freedom are the angle of attack and the deflection of a control surface at the trailing edge. The database integrates experimental data with simulations of variable fidelity. Data from different sources are coherent in the linear aerodynamic regime, whereas discrepancies appear at a high angle of attack or for a large deflection of the control surface due to flow separation and nonlinearities related to stall.

I. Introduction

THE Clean Sky 2 European Joint Undertaking aims to reduce the environmental impact of future aircraft by targeting reductions of 20–30% in CO₂ emissions, NO_x emissions, and perceived noise levels relative to year-2000 reference technologies [1]. Achieving such ambitious objectives is increasingly challenging within the framework of conventional aircraft configurations, which are likely approaching the limits of incremental improvement achievable through design optimization alone. Consequently, meeting these targets calls for the investigation of disruptive aircraft concepts and unconventional aerodynamic layouts. Among the aircraft components most affected by these emerging configurations, the rear-end assembly—including the horizontal tail and control surfaces—has been identified as particularly critical. In this context, standard empennage layouts may no longer provide adequate performance or controllability, especially under off-design conditions, large control surface deflections, or highly nonlinear flow regimes. Accurate prediction of the aerodynamic behavior of unconventional tail configurations, therefore, represents a key challenge, both for performance assessment and for flight safety considerations. The analysis of such configurations places significant demands on both experimental and numerical tools. Strong nonlinearities, flow separation, and three-dimensional effects limit the predictive capability of low-order models, while high-fidelity simulations and experiments remain costly and difficult to deploy across large parametric spaces. For these reasons, multifidelity aerodynamic databases have emerged as a valuable approach to support model development, calibration, validation, uncertainty quantification, and reduced-order modeling. By combining experimental measurements with numerical simulations of varying complexity, these databases enable the systematic assessment of modeling assumptions while maintaining computational tractability.

Over the past decades, several multifidelity databases have been developed to address specific aerodynamic challenges. Catalano and

Amato [2] constructed a comprehensive database for the RAE M2155 low-aspect-ratio wing, combining experimental data [3] with RANS simulations using different turbulence closures. This database proved instrumental in evaluating turbulence models but remained focused on wing aerodynamics under relatively canonical conditions. Similarly, extensive datasets have been generated for vortex-dominated flows at high angles of attack, such as the NASA sharp-edged delta wing of Chu and Luckring [4], which served as a benchmark for the Second International Vortex Flow Experiment (VFE-2) [5]. Subsequent numerical studies within the ATAAC [6] and DESider [7] projects employed detached eddy simulation to capture vortex breakdown phenomena [8]. While these efforts significantly advanced the understanding of vortex flows, they were not specifically designed to address control surface effects or tailplane aerodynamics.

More recently, multifidelity approaches have been applied in the context of aerodynamic efficiency improvements for transport aircraft. Within the LuFo VI-1 CATeW project, Mentis et al. [9] investigated the combined effects of hybrid laminar flow control and variable camber on wing performance in transonic conditions. Although these studies demonstrated the value of multifidelity data integration for technology assessment, they remained centered on wing applications and attached-flow regimes. In a complementary direction, Petrilli et al. [10] developed a RANS database for poststall aerodynamics of airfoils and wings, highlighting the limitations of low-order models for swept-wing configurations dominated by three-dimensional flow separation. These results underscore the need for detailed datasets to support model development in highly nonlinear regimes.

Despite these advances, a dedicated, openly accessible multifidelity database focused on unconventional horizontal tail configurations, including large control surface deflections and wide angle-of-attack ranges, has so far been lacking. In response to this gap, the MONNALISA project [11] conducted a comprehensive experimental and numerical campaign to investigate the aerodynamic behavior of unconventional horizontal tail planforms. The study considered variations in aspect ratio, taper ratio, sweep, and dihedral and systematically analyzed their performance over a broad range of angles of attack and elevator deflections. The dataset is specifically tailored to horizontal tail configurations: aspect ratios and taper ratios are low, and full-span control surfaces are included, which are typical features of empennage geometries. While the flow conditions correspond to standard flight regimes, the database captures aerodynamic phenomena particularly relevant to tailplane design, including control-surface effects and poststall behavior.

As a key outcome, the project delivered a multifidelity aerodynamic database integrating wind-tunnel measurements with numerical simulations obtained using computational models of differing

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fidelity. The MONNALISA database is openly available through the Zenodo platform under the MONNALISA Community [12] and is designed to support research and development activities, as well as the calibration and validation of predictive models for non-linear tail aerodynamics. Notably, the database is conceived as a “live” resource and is continuously expanded as new data become available.

This work aims to present the MONNALISA database and to explain its structure. The remainder of this paper is organized as follows. Section II details the full set of tail configurations explored in the MONNALISA project and the test conditions. Section III thoroughly describes the mid-/high-fidelity computational models employed to perform numerical simulations and the experimental setup. In Sec. IV, we perform a comparison of data from all available sources to highlight similarities and discrepancies. The structure of the resulting database is described in Sec. V. Finally, Sec. VI reports the conclusions and final remarks.

II. Parameterization of the Tail Geometry

In the aim of the MONNALISA project, the database provides the characterization of the aerodynamics of a number of possible horizontal tail configurations. In particular, the database includes the aerodynamic coefficients of low-aspect-ratio wings of different planforms across a wide range of positive and negative angles of attack $\alpha \in [-26^\circ, 26^\circ]$ with a 2° increment, thus including deep-stall conditions. With reference to Fig. 1, the planform is described by a set of parameters. Namely, the quarter-chord sweep angle λ , the aspect ratio Λ , the taper ratio τ , the dihedral angle Γ , and the model semispan b . The latter is fixed at 1.6 m for all the models in order to reduce blockage effects in the wind tunnel. In our convention, a positive sweep indicates that the tip is positioned downstream of the root, whereas a positive dihedral indicates upward deflection. Note that the dihedral angle varies between two discrete values $\Gamma = \{0^\circ, 45^\circ\}$. Additionally, the geometry is equipped with a full-span control surface that rotates about the 70% chord line with a discrete angle $\delta = \pm\{0^\circ, 10^\circ, 20^\circ, 30^\circ, 34^\circ\}$.

According to this parameterization, at the current stage, the database includes data concerning three different horizontal tail planforms specified in Table 1. A visual representation of the three in-plane geometries is shown in Fig. 2.

For all of them, the airfoil shape is fixed along the span and consists of a symmetric profile with thickness-to-chord ratio (t), as reported in Table 1. Hereinafter, “Geom. I” will also be referred to as the *reference planform*.

The three configurations have been extensively tested experimentally, namely, for all possible combinations of α , Λ , and δ . Instead, only the reference planform has been fully investigated numerically.

Table 1 Values of the geometric parameters for all the tested configurations

Parameter	Symbol	Unit	Geom. I	Geom. II	Geom. III
Sweep angle	λ	deg	30.0	−15.0	22.8
Dihedral angle	Γ	deg	0, 45	0, 45	0, 45
Aspect ratio	Λ	—	5.0	5.2	3.76
Taper ratio	τ	—	0.4	0.6	0.6
Reference surface	S_{Ref}	m ²	1.024	0.9846	1.3617
Mean aerodynamic chord (reference)	MAC	m	0.679184	0.6282	0.8688
Airfoil thickness	t	%	9	9	10

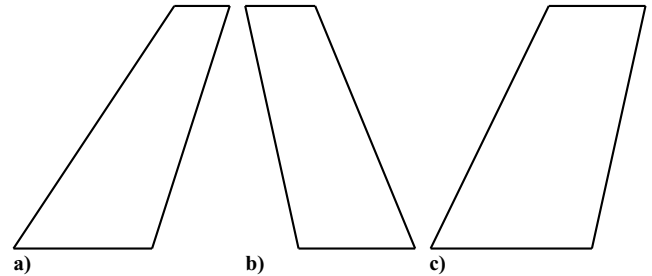


Fig. 2 Visual representation of the investigated geometries: a) first, b) second, and c) third geometry.

III. Numerical and Experimental Setup

This section describes the computational models employed to carry out the numerical investigation and the experimental model implemented in the GVPM facility (a large wind tunnel at Politecnico di Milano) used to perform tests. The test conditions refer to the ones in the experimental wind tunnel, which have then been replicated in the numerical simulations. The Mach number is 0.147, and the Reynolds number (based on the semispan) is 5.5×10^6 . The semispan is chosen as the characteristic length to account for wind-tunnel constraints, particularly to control blockage effects, and it is kept constant across all models to ensure consistent aerodynamic comparisons. The freestream temperature (T_∞) is 300 K. Moreover, the freestream turbulence intensity is $TI_\infty \leq 0.1\%$.

A. Midfidelity CFD Model

DUST [13] is an open-source software package developed from a collaborative research effort between Politecnico di Milano and A³ by Airbus SE. DUST is an open-source computational tool designed to obtain fast and reliable aerodynamic predictions. By virtue of its

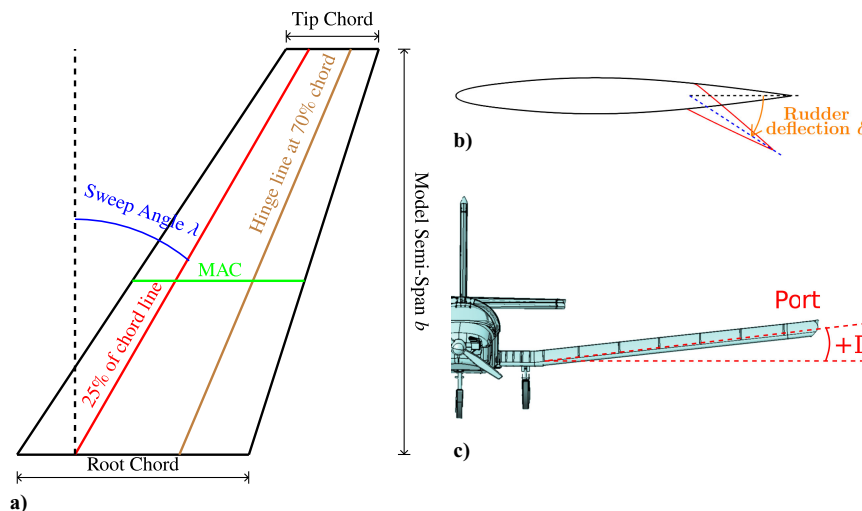


Fig. 1 Planform definition of a trapezoidal lifting surface: a) in-plane parameters, b) rudder deflection definition, and c) dihedral angle.

flexibility, the solver allows the simulation of arbitrarily complex applications, e.g., vertical take-off and landing (VTOL) configurations.

Several aerodynamic models are available to predict loads and to simulate the evolution of the wake. Among these, the lifting-line model was selected.

The circulation of the lifting line is determined through the exploitation of tabulated aerodynamic coefficients. Within DUST, these are provided in the form of C81 lookup tables. In practice, slender lifting bodies are modeled as 1D lines discretized into sections. For each section, the aerodynamic coefficients are tabulated as a function of the local angle of attack, the local Reynolds number, and the local Mach number. In our framework, the C81 tables are evaluated with SU2, using a two-dimensional RANS model of the sectional profile. Moreover, we employ an in-house mesh-adaptation software named Darwin (see [14]) to improve grid quality in domain regions characterized by steep gradients in flow properties. Once the tables are available, we employ the Γ method to solve the resulting nonlinear problem. Namely, the circulation is computed from tabulated data, taking advantage of the analytical expression of the Kutta–Joukowski theorem.

In this work, the lifting-line model has been augmented with a sweep correction proposed by Goitia and Llamas [15]. This correction extends the limits of applicability for the LL approach by improving the modeling of three-dimensional effects. Indeed, the sweep angle produces a spanwise flow from the root to the tip if the sweep angle λ is positive (or vice versa for a negative λ). For positive λ , the boundary layer is thinner at the root, and this delays stall [16]. At the tip, the boundary layer is thicker, anticipating the onset of stall. The classical sweep theory for infinite wings (see, for instance, [17]) claims that purely 2D characteristics may be corrected and transformed into 2.5D through

$$C_L^{2.5D} = C_L^{2D} \cos^2 \lambda \quad (1)$$

$$C_D^{2.5D} = C_D^{2D} \quad (2)$$

$$\alpha^{2.5D} = (\alpha^{2D} - \alpha_0) \cos \lambda + \alpha_0 \quad (3)$$

being C_L and C_D the lift and drag coefficients, respectively, and α_0 the zero-lift angle for the 2D sectional airfoil. In this context, the term 2.5D denotes the augmentation of inherently two-dimensional data by including corrections that account for selected three-dimensional effects, such as sweep-induced flow components, while retaining the underlying 2D formulation. As such, the resulting description does not constitute a fully three-dimensional analysis but rather a corrected framework in which 3D-inspired effects are embedded into a 2D baseline. This terminology is adopted here to clearly distinguish the approach from both purely 2D and fully 3D modeling strategies. The correction reported above is valid for infinite-span wings.

The correction suggested in [15] consists of implementing a further empirical correction function for finite wings. In a finite wing, the aerodynamics of each section vary along the spanwise coordinate y , e.g., the maximum lift coefficient is larger at the root than at the tip [16]. Therefore, an additional correction $\kappa = \kappa(y)$ is introduced to approximate the spanwise variation. The two-dimensional polars are therefore assigned an effective sweep angle:

$$\lambda_{\text{eff}}(y) = \lambda_{\text{loc}}(y) \cdot \kappa(y) \quad (4)$$

The results from high-fidelity CFD simulations [10] suggest implementing $\kappa(y)$ as the superimposition of two different and independent contributions, which become active at the root and at the tip:

$$\kappa(y) = 1 + \kappa_{\text{root}}(\eta'_{\text{root}}) + \kappa_{\text{tip}}(\eta'_{\text{tip}}) \quad (5)$$

Note the coordinate transformation occurring through the definition of η'_{root} and η'_{tip} [15], which represent the nondimensional spanwise position normalized with the mean geometric chord. They are defined as

$$\eta'_{\text{root}}(y) = \frac{y - y_{\text{root}}}{\text{MAC}} = \Lambda(\eta(y) - \eta_{\text{root}}) = \Lambda\eta \quad (6)$$

$$\eta'_{\text{tip}}(y) = \frac{y - y_{\text{tip}}}{\text{MAC}} = \Lambda(\eta(y) - \eta_{\text{tip}}) = \Lambda(\eta - 1) \quad (7)$$

where η is the standard nondimensional spanwise position $2y/b$. The variables η'_{root} and η'_{tip} measure *how many chords away* a spanwise position is from the root or the tip. In the original paper, both the tip and the root functions are modeled using Bézier curves [18] defined as follows:

$$B(y) = C_1 (1 - y)^3 + 3C_2 (1 - y)^2 y + 3C_3 (1 - y)y^2 + C_4 y^3 \quad (8)$$

Therefore, the two curves are parameterized by means of four coefficients each. A variation of these parameters changes the shape of the spanwise correction, accounting for three-dimensional effects associated with a finite swept wing.

B. High-Fidelity CFD Model

SU2 [19] is an open-source suite used for solving partial differential equations (PDEs) on unstructured grids. The finite volume method (FVM) is applied using a standard edge-based data structure on a dual grid with control volumes constructed using a median-dual, vertex-based scheme. Median-dual control volumes are formed by connecting centroids, faces, and edge midpoints of all primal cells sharing a particular vertex.

In this work, we employ SU2 to build a computational model of the tail wing based on the steady Reynolds-averaged Navier–Stokes (RANS) model. In the discretized equations, the convective fluxes are solved using the upwind Roe scheme with the Monotonic Upstream-Centered Scheme for Conservation Laws (MUSCL) reconstruction to achieve second-order accuracy. The Venkatakrishnan flux limiter has been applied to mitigate local spurious oscillations.

The implicit Euler scheme has been used to advance the computation within a fictitious time step. The convergence to a steady solution is monitored considering the mean variation of both the lift (C_L) and drag (C_D) coefficients. Convergence is achieved when such indicators are below a certain threshold, according to the formula

$$\Delta C_i = \frac{\sum_{j=N-M}^N |C_{i,j} - C_{i,j-1}|}{M} < \epsilon_i, i = L, D \quad (9)$$

where N is the current iteration and M is an arbitrary number of previous iterations used to calculate the mean variation. Both the C_L and C_D tolerances have been set to $\epsilon_i = 3 \times 10^{-6}$, $i = L, D$.

The Spalart–Allmaras (SA) [20] turbulence closure is employed in conjunction with the algebraic Bas–Cakmakcioglu laminar-to-turbulent transition model [21].

1. Turbulence Modeling

In agreement with the Boussinesq hypothesis, the effect of turbulence is represented as an increased viscosity, thus leading to the computation of the total viscosity as the sum of a laminar viscosity μ_l and of a turbulent viscosity μ_t . The latter is modeled through the Spalart–Allmaras (SA) [20] turbulence model, which introduces the following transport equation for the variable $\hat{\nu}$:

$$\frac{\partial \hat{\nu}}{\partial t} + u_j \frac{\partial \hat{\nu}}{\partial x_j} = P_{\hat{\nu}} - D_{\hat{\nu}} + \frac{1}{\sigma} \left[\frac{\partial}{\partial x_j} \left((\nu + \hat{\nu}) \frac{\partial \hat{\nu}}{\partial x_j} \right) + c_{b2} \frac{\partial \hat{\nu}}{\partial x_j} \frac{\partial \hat{\nu}}{\partial x_j} \right] \quad (10)$$

where $P_{\hat{\nu}}$ and $D_{\hat{\nu}}$ are the production and destruction terms for $\hat{\nu}$, respectively. The turbulent convective fluxes are obtained through a first-order upwind scheme. The new variable $\hat{\nu}$ is directly related to the turbulence viscosity through the following formulations:

$$\mu_t = \rho \hat{\nu} f_{v1} \cdot f_{v1} = \frac{\chi^3}{\chi^3 + c_{v1}^3} \cdot \chi = \frac{\hat{\nu}}{\nu} \nu = \frac{\mu_t}{\rho} \quad (11)$$

where $c_{v1} = 7.1$ is a constant of the model.

2. Transition Modeling

Laminar-to-turbulent transition can be computed through the algebraic Bas-Cakmakcioglu model [21], which premultiplies the production term (contained within Q) from Eq. (10) by the intermittency distribution function γ_{BC} , defined as

$$\gamma_{BC} = 1 - e^{-(\sqrt{T_1} + \sqrt{T_2})}, \quad T_1 = \frac{\max\left(\frac{Re_v}{2.193} - Re_{\theta c}, 0\right)}{\chi_1 Re_{\theta c}},$$

$$T_2 = \max\left(\frac{1}{\chi_2} \frac{\mu_t}{\mu}, 0\right), \quad Re_v = \frac{\rho d_w^2}{\mu} S,$$

$$Re_{\theta c} = 803.73(Tu_\infty + 0.6067)^{-1.027} \quad (12)$$

where Re_v is the vorticity Reynolds number, S is the magnitude of the mean vorticity tensor, and $Re_{\theta c}$ is the critical momentum thickness Reynolds number, which is obtained through empirical relations applied to the freestream turbulence intensity Tu_∞ .

3. Boundary Conditions

The model surface is treated with no-slip and nonpenetration boundary conditions, along with zero heat flux. On the far-field boundaries, standard freestream conditions with a zero-pressure gradient are applied. The root plane of the model is treated as a symmetry boundary. A more detailed discussion of the implications of this choice is provided in Sec. IV.

4. Grid Description

High-quality grids of the volume surrounding the different planforms were generated using an automated meshing procedure devised ad hoc for the MONNALISA project. The parametric CAD model is first built using CATIA V5 software. Through a collection of wrapping scripts, the CAD model is provided to the Pointwise [22] meshing software. The two key steps in the automated procedure are the creation of the surface and volume meshes. For the surface mesh, highly anisotropic triangular cells are used to increase resolution in high-curvature regions, e.g., the leading edge or the tail-elevator junction. For the volume grid, a hex-dominant, octree-based mesh is built to limit numerical dissipation. Both surface and volume meshes are refined where complex flow patterns are expected, e.g., at the wing tip, where vortices are produced. An α -adaptive wake refinement is performed in the freestream direction to increase the resolution of the volume mesh and better capture the wake vortices. The surface and volume mesh of a random realization of the tail wing are reported in Fig. 3.

A mesh convergence study was conducted by varying the parameters specified in Table 2. Given the necessity to perform mesh convergence on representative cases, the highest deflection of the

Table 2 Values of the meshing parameters for the convergence study

Parameter	Symbol	Value			
		Coarse	Nominal	Fine	Very fine
LE angle	σ_{LE}	5	2.5	1.25	1
Junction angle	σ_J	2	1	0.5	0.25
TE points	n_{TE}	5	7	9	13
Max surface spacing	h_{max}	0.04	0.02	0.01	0.005
Growth rate	r_{BL}	1.2	1.15	1.1	1.075

control surface was chosen ($\delta = 35$ deg). Additionally, two angles of attack, denoted as $\alpha_1 = 0$ deg and $\alpha_2 = 18$ deg, were selected to represent smooth and separated flow conditions, respectively. Both values of the dihedral angle were also considered. In total, four distinct configurations were investigated, each with four levels of refinement ranging from coarse to very fine. Each level nearly halves all the meshing parameters from the previous grid except σ_{LE} (due to CAD tolerances). The resulting number of grid points (N) for each configuration is presented in Table 3. Note that the table also establishes a labeling for identifying the grids.

Figure 4 reports the convergence analysis. The relative error $\Delta c_{F,i}$ with respect to the finest refinement level is plotted as a function of the $1/N^{1/3}$ factor, which describes the change in mesh size. $\Delta c_{F,i}$ is defined as

$$\Delta c_{F,i} = \frac{|c_{F,i} - c_{F,i_{max}}|}{|c_{F,i_{max}}|} \quad (13)$$

where F denotes the targeted quantity of interest (L and D for the lift and drag coefficients), i represents the current mesh, and i_{max} denotes the finest mesh in the refinement sequence. This approach enables the comparison of percentage variations in c_L and c_D relative to the finest-grid values to assess the improvement in prediction across refinement levels. The relative errors remain below 4%, suggesting that the increment in the number of points improves the prediction of aerodynamic coefficients only marginally. This observation holds for all other configurations, although with a non-monotonic convergence pattern visible in a few cases. Overall, the nominal mesh appears to strike the best compromise, demonstrating

Table 3 Number of points for each case of the mesh convergence study

Refinement level	Number of points ($\times 10^6$)			
	$\Gamma = 0$ deg		$\Gamma = 45$ deg	
	$\alpha = 0$ deg	$\alpha = 18$ deg	$\alpha = 0$ deg	$\alpha = 18$ deg
Coarse	4.79	4.85	5.13	5.29
Nominal	9.94	12.74	9.67	9.97
Fine	22.55	26.56	22.33	25.14
Very fine	50.11	65.24	51.80	65.89

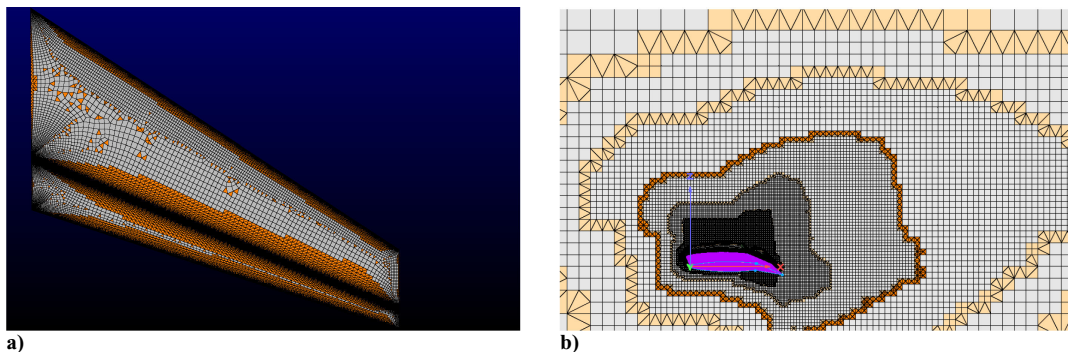


Fig. 3 Visualization of employed meshing strategies: a) surface mesh and b) volume mesh.

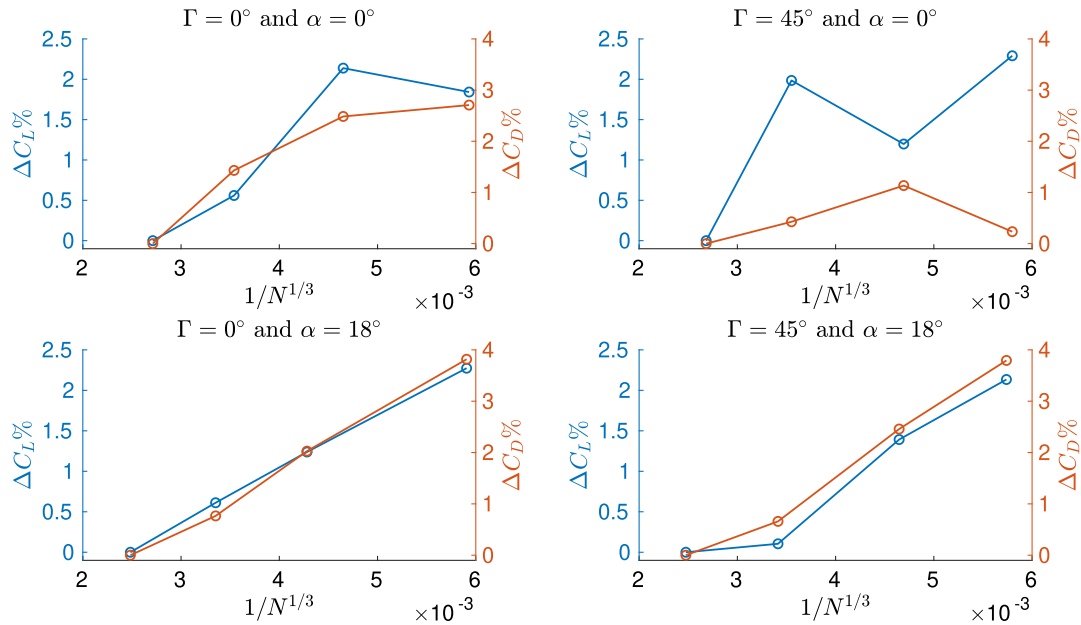


Fig. 4 Mesh convergence expressed with respect to the lift and drag coefficients (respectively, c_L and c_D).

low relative errors while maintaining a reasonable computational cost given the available resources. Consequently, the parameters associated with the *nominal* refinement, as reported in Table 2, are employed to generate all meshes.

C. Wind-Tunnel Testing

The GVPM at Politecnico di Milano is a large closed-circuit wind tunnel. It features a special layout with two test sections located along the closed loop, as shown in Fig. 5a. The Low Turbulence Test Section has been employed to perform the experimental campaign presented in this paper. The size of the test section has been reduced due to the installation of a dummy floor necessary to hide the model support, balance, and the actuation system for the control surface. The actual test section was 4 m wide, 349 m high, and 6 m long. The maximum wind velocity is 55 m/s, and the turbulence level is less than 0.1%. This turbulence level has been measured by means of both the turbulence sphere method and hot-wire anemometry. The total pressure distribution out of the wall boundary layer, measured by means of a Pitot tube at the chamber inlet section, is quite uniform ($\pm 0.5\%$ of dynamic pressure over most of the section and never larger than $\pm 1\%$). The same quantity measured on a horizontal line at half the height of the chamber outlet section confirmed this uniformity. The test chamber is slightly divergent to correct for boundary-layer growth. The small residual static pressure longitudinal gradient has been measured by means of a static-pressure tube, resulting in $\partial C_P / \partial x = -0.004 \text{ m}^{-1}$.

The force measurements were acquired by means of an RUAG 767 strain-gauge balance installed under the floor of the test section, which measured the forces acting on the complete model. The balance signals were acquired by means of an HBM QuantumX MX1615 system. The freestream velocity, absolute pressure, temperature, and humidity were acquired by means of an HBM ClipX system. The signals from the model torque balance were acquired by means of an NI PXI-6289 board installed in an NI PXIe-1078 chassis. The measurement uncertainties reported in the error bars (1σ) were estimated by computing the standard deviation of the measured signals.

The facility is equipped with infrared thermography instrumentation (see Fig. 6a), namely, an FLIR A6751 LWIR camera with a strained-layer superlattice sensor of $640 \times 512 \text{ px}$ resolution (square pixel size of $15 \mu\text{m}$ edge). The thermal sensitivity is below 45 mK. The camera was robustly attached to the wind-tunnel floor using an aluminum structure (red box in Fig. 6a). A set of silver conductive paint markers was depicted on the wing's upper and lower surfaces and exploited to dewarp and calibrate the images (blue box in Fig. 6a). The markers were accurately sanded to avoid any spurious perturbation of the flow that could affect transition. Three different fields of view (FOVs) produce infrared thermography measurements and were established to capture in detail the root, middle, and tip regions (see Fig. 6b). Measurements were made for an angle of attack ranging between $\alpha = -1.1^\circ$ and $\alpha = 2.9^\circ$ at a wind-tunnel freestream velocity (V_∞) equal to 50 m/s. During

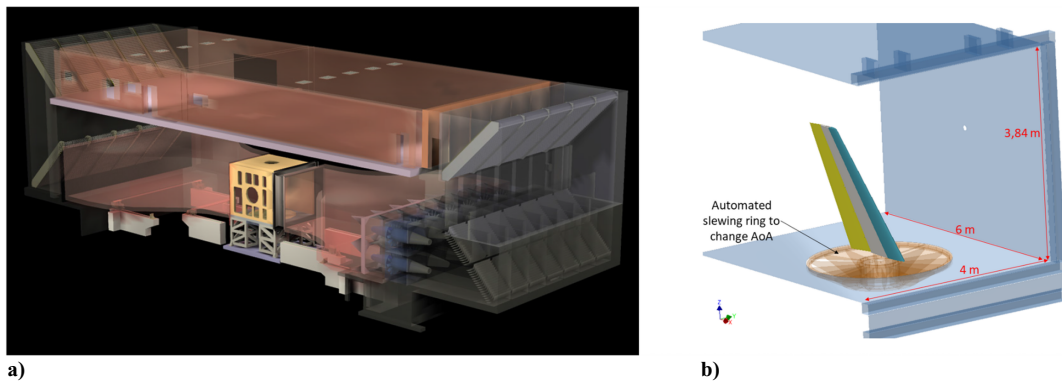


Fig. 5 The large wind tunnel of Politecnico di Milano (GVPM): a) general layout of GVPM and b) tailplane model installed in the wind tunnel.

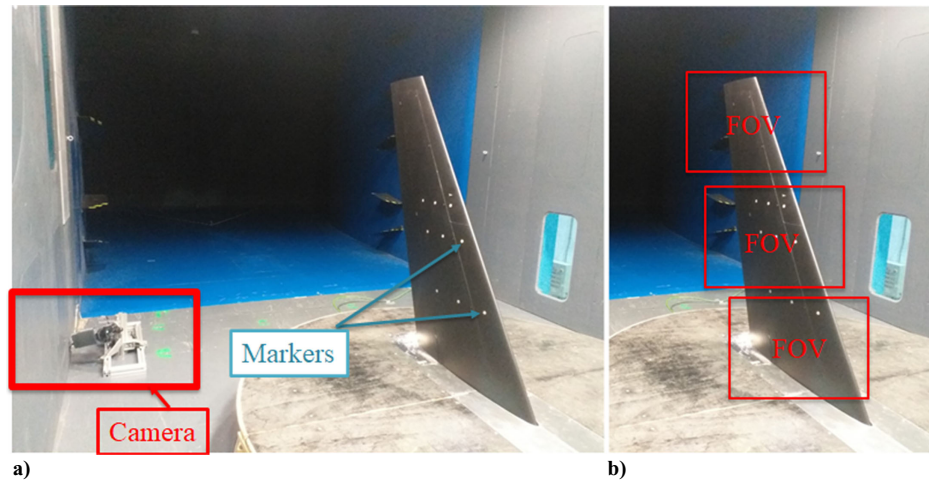


Fig. 6 Setup of the infrared thermography instrumentation: a) camera position and reference markers and b) field of views (FOV) of the infrared thermography measurements.

the tests, the model was colder than the air and thus heated by the freestream. The regions characterized by a turbulent boundary layer were detected as warmer than those with a laminar boundary layer. For each model, angle of attack, and field of view, a series of 500 individual images was acquired at 100 Hz. For each database point, reference images were also acquired without wind.

This operating condition was dictated by the adopted model technology. Since the model was machined from solid aluminum, heating by means of lamps was not feasible due to its thermal capacity and conductivity. Therefore, the infrared images were acquired by exploiting the faster temperature increase of the free-stream with respect to the model. The main advantages of this procedure were its simplicity (no additional equipment was required in the flow), the small temperature difference between the model and the fluid vein (typically on the order of 2 K or less), and the initially uniform temperature distribution within the model. Conversely, the temperature difference was not actively controlled and, as a consequence, was not strictly repeatable; for this reason, the infrared images were acquired in the early phase of each run. The lack of control of the temperature difference did not affect the repeatability of the transition detection, as confirmed by the repeatability checks performed during the test campaign, likely due to the limited magnitude of the temperature jump.

PIV surveys were conducted to investigate the flow behavior near the movable surface during dynamic actuation. Due to the high deflection angles involved, these measurements provide valuable

insight into possible flow separation in the region adjacent to the control surface. In these dynamic tests, the motion of the control surface was imposed through harmonic oscillations with a frequency of up to 2 Hz and an amplitude of $\delta = \pm 30$ deg.

The PIV setup used for the wind-tunnel tests is composed of a Quantel Evergreen Nd:YAG double-pulse laser with an output energy of 200 mJ (wavelength 532 nm) and an ILA.PIV.sCMOS camera with a 16-bit 2560×2160 pixel array. The laser was positioned over one of the lateral walls of the wind-tunnel test section to generate a laser sheet aligned with the chordwise direction of the model by means of a 90 deg optic mirror. The camera was arranged under the floor, which was holed to allow the optical view of the camera over the upper surface of the model airfoil, as depicted in Fig. 7. The laser and the camera were controlled by an ILA5150 Synchronizer unit designed to handle and logically combine up to 6 TTL inputs with up to 12 TTL outputs, with a signal resolution down to nanoseconds and a jitter on the order of picoseconds. While these tests provide valuable insight into unsteady aerodynamic behavior, the corresponding data are not included in the current version of the MONNALISA database. They will be incorporated in future updates, ensuring that the database progressively expands to cover both steady and unsteady flow conditions relevant to tailplane aerodynamics.

In all wind-tunnel runs, the aerodynamic forces were measured with respect to the balance reference system and converted into the wind reference system for coherence with numerical predictions.

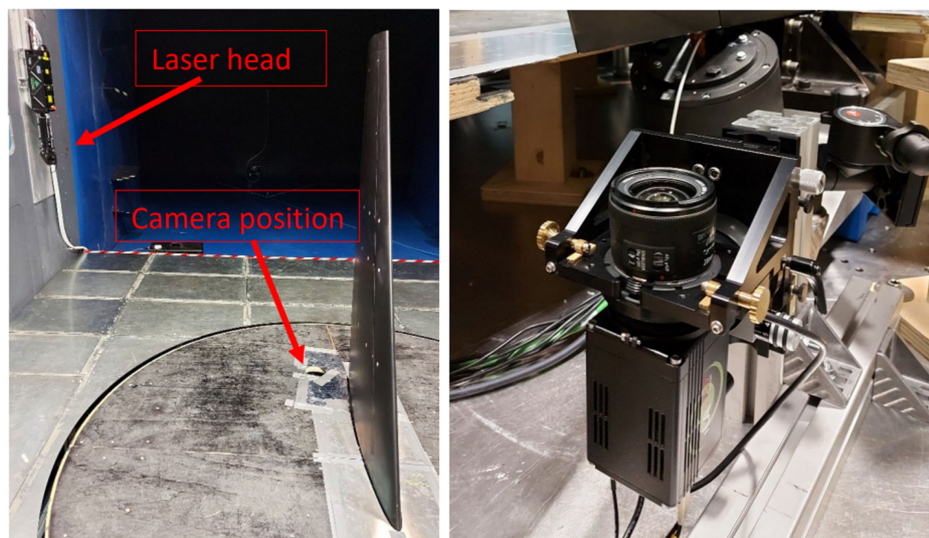


Fig. 7 PIV surveys: laser and camera installation.

In the tunnel, the angle of attack is controlled by rotating the base of the cylindrical frame supporting the model (visible in Fig. 5b). The pitching moment, the only relevant moment in the tests, is considered positive nose up. We assumed the aerodynamic center of the mean aerodynamic chord (MAC) as the pole with respect to which the moment is computed and the MAC itself as the reference length. The moment coefficient is computed directly from the balance force components. Owing to the support geometry, F_{Xb} does not contribute to the pitching moment. Therefore, we have

$$C_{m,CA_{MAC}} = \frac{M_{Zb} - F_{Xb}(X_{0b} - X_{CA_{MAC}})}{1/2\rho U_\infty^2 S_{Ref} MAC} \quad (14)$$

To compute the position of the aerodynamic center, we impose

$$\frac{\partial C_{M,CA}}{\partial \alpha} = \frac{\partial C_{M,Zb}}{\partial \alpha} - \frac{\partial C_{F,Xb}}{\partial \alpha} (X_{0b} - X_{CA}) = 0 \quad (15)$$

So that

$$X_{0b} - X_{CA} = \frac{\frac{\partial C_{M,Zb}}{\partial \alpha}}{\frac{\partial C_{F,Xb}}{\partial \alpha}} \quad (16)$$

and

$$C_{m,CA_{MAC}} = C_{M,b} - \frac{\frac{\partial C_{M,Zb}}{\partial \alpha}}{\frac{\partial C_{F,Xb}}{\partial \alpha}} C_{F,Xb} \quad (17)$$

The position of the aerodynamic center was calculated in the configuration $\delta = 0$ deg and applied to all the deflections.

Several wind-tunnel corrections [23] were applied to the raw data and are summarized hereinafter.

1. Horizontal Buoyancy

The pressure gradient dp/dl in the streamwise direction was first measured in the empty chamber using pressure taps mounted on the wind-tunnel floor. The following correction

$$c_D = \frac{D_m - \Delta D_B}{q_m S_{Ref}}, \quad \text{with} \quad \Delta D_B = -V_{Ref} \frac{dp}{dl} \quad (18)$$

is then applied to obtain a provisional value of the drag coefficient. In the above formula, D_m is the measured drag, whereas D_B is the buoyancy correction to it. The reference volume, V_{Ref} , refers to the volume of the wind-tunnel model, which, for the three different geometries, is equal to 0.051, 0.054, and 0.075 m³, respectively. The reference surface, S_{Ref} , is taken from Table 1, whereas q_m is the measured dynamic pressure.

2. Solid and Wake Blockage

For solid blockage, the dynamic pressure q can be corrected by adding to it the following contribution:

$$(\Delta q)_{SB} = 2q_m \frac{KV_{Ref}}{\sqrt{2}C^{3/2}} \quad (19)$$

where $K = 0.98$ can be considered in the present case, and the subscript SB stands for solid blockage.

For wake blockage, the dynamic pressure q can be corrected by

$$(\Delta q)_{WB} = q_m \left[\frac{S_{Ref}}{2C} C_{D0} + \frac{5S_{Ref}}{2C} (C_D - C_{D,i} - C_{D,0}) \right] \quad (20)$$

where the induced drag coefficient $C_{D,i}$ can be estimated as

$$C_{D,i} = \frac{C_L^2}{\pi\Lambda} \quad (21)$$

Eventually, the corrected value of q is

$$q_C = q_m + (\Delta q)_{SB} + (\Delta q)_{WB} \quad (22)$$

so the corrected lift and moment coefficients can be computed as

$$C_{L,C} = \frac{q_m}{q_C} C_L, \quad \text{with} \quad C_{D,C} = \frac{q_m}{q_C} C_D \quad (23)$$

3. Downwash Correction

Due to the downwash modification, the angle of attack must be corrected as

$$\alpha_C = \alpha_m + (57.3)\Delta \frac{S}{C} C_{L,C} \quad (24)$$

so that the corrected drag coefficient becomes

$$C_{D,C} = \frac{q_m}{q_C} C_D + \Delta \frac{S}{C} C_{L,C}^2 \quad (25)$$

where, for the present case, we can take $\Delta = 0.12$.

IV. Preliminary Comparison of Aerodynamic Characteristics

A preliminary comparison between all the investigated methodologies is given in this section. The geometry selected for comparison is the reference one, i.e., Geom. I in Table 1, with $\Gamma = 0$ deg. Thanks to the symmetry of the geometry in the horizontal plane, both the wind-tunnel and numerical campaigns considered only positive values of δ for $\alpha \in [-26 \text{ deg}, 26 \text{ deg}]$. To maintain good readability of the figures, only three control surface deflection angles are compared, namely, $\delta = \{0 \text{ deg}, 10 \text{ deg}, 35 \text{ deg}\}$. A more in-depth analysis will be presented in a follow-up paper. Figure 8 presents the aerodynamic coefficients computed using the three approaches at identical deflection angles. The experimental data include vertical error bars, which account for uncertainties from the measurement equipment as well as flow unsteadiness at higher angles of attack. Despite these factors, the error margins remain relatively small. The largest uncertainties are observed in the pitching moment coefficient, which is expected, given the greater complexity involved in measuring moments compared to forces. For low values of δ , in the linear portion of $C_L(\alpha)$ curve, and in the corresponding region for C_D and $C_{M_y,AC}$, numerical predictions are in excellent agreement with the experimental campaign. This holds for both the mid- and high-fidelity simulations. However, the higher the values of α and δ , the larger the discrepancies between numerics and experiments. Clearly, this is due to flow separation and nonlinearities possibly triggered by the combination of large α and δ . As expected, when considering large δ , the difference between numerics and experiments is smaller for the RANS simulations than for those employing the lifting-line method.

With reference to the lift coefficient, the midfidelity model successfully captures the linear region for small control surface deflections. Additionally, poststall values exhibit the same discrepancies from experimental results as seen in the RANS simulations. However, at higher angles of attack, a more complex pattern emerges, likely due to instabilities in the model associated with stalled predictions from 2D simulations. The stall angle prediction is accurate and consistent with RANS simulations. Yet, for larger control surface deflections, discrepancies increase in magnitude. Despite this, the overall curve shape and stall angle are correctly captured. In contrast, the slope and stall behavior at negative angles of attack deviate significantly, resulting in c_L values that differ substantially from both RANS and experimental data.

The pitching moment coefficient (C_{M_y}) is shown in Fig. 8c and is computed w.r.t. the aerodynamic center of the reference planform. Its location has been determined using Eqs. (16) and (17), considering $\delta = 0$ deg and kept constant despite the δ variations.

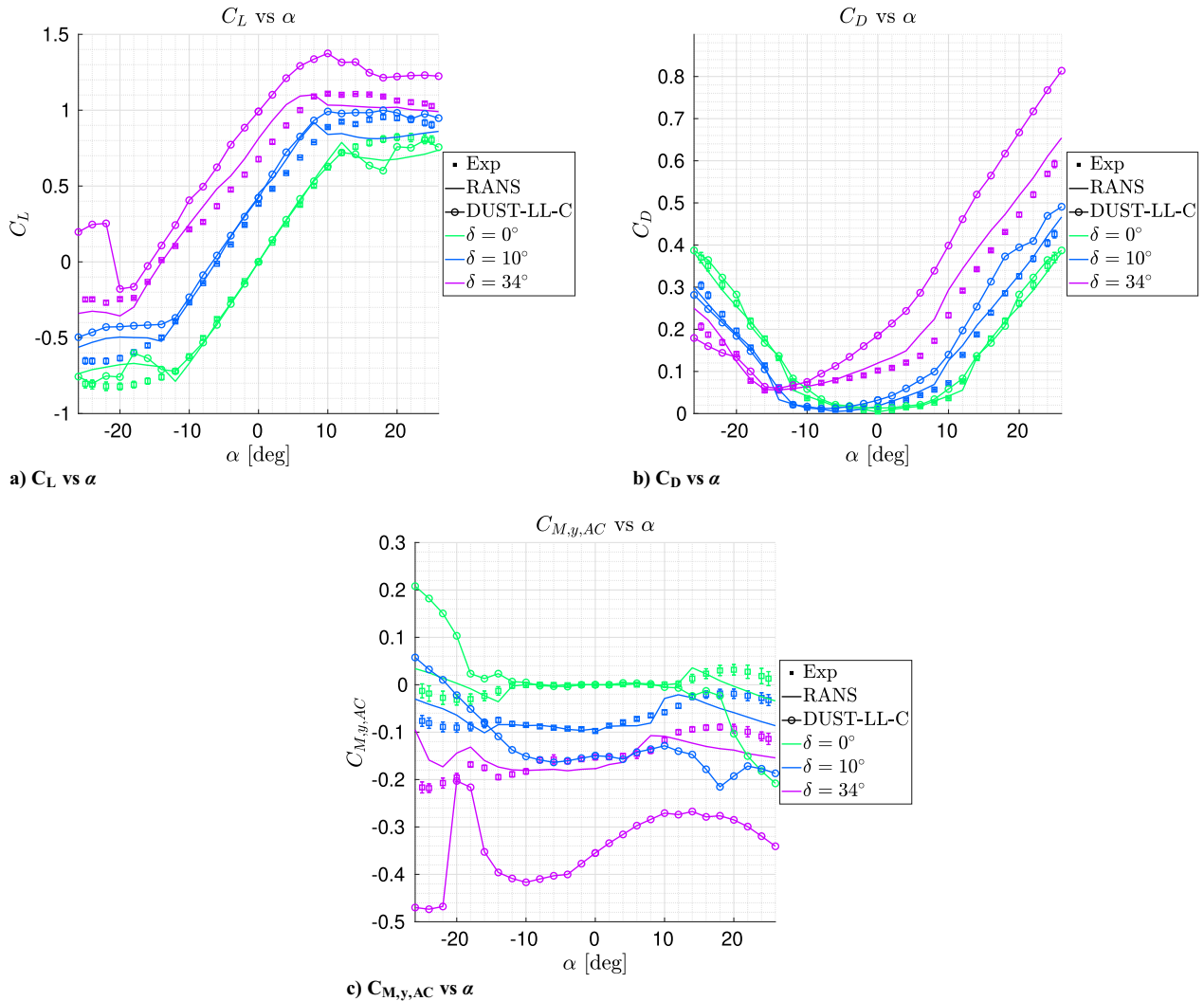


Fig. 8 Comparison between wind tunnel, RANS, and lifting-line method results in terms of the aerodynamic coefficients for $\Gamma = 0$ deg.

The concavity of the experimental results is well captured by the RANS simulations, as well as the increase in the coefficient value associated with the stall at the aerodynamic surface. However, midfidelity predictions are significantly inaccurate and only capture the general trend. This discrepancy worsens as δ increases and is currently under investigation. It is likely attributable to errors in the pitching moment coefficient derived from the 2D simulations.

A clear path to reduce these discrepancies involves improving the underlying 2D simulations through finer grids, alternative turbulence models, and additional validations, as well as enhancing the inclusion of 3D effects via corrections applied to these 2D results. By maintaining a “live” database, these updates will progressively refine midfidelity predictions, improve agreement with experimental data, and provide a reliable reference for the community to benchmark and validate their own simulations.

By inspection of Fig. 9, which shows the flow pattern on the upper surface of the tail geometry with a control surface deflection of $\delta = 0$ deg, it is possible to investigate the spanwise translation of the leading-edge vortex. This phenomenon is characteristic of an aerodynamic surface with positive sweep [24,25]. Here, the zero-control-surface-deflection case is shown near the stall angle. For $\alpha = 12$ deg, the flow is still fully attached over the suction side of the tail. A small component of spanwise flow is visible, especially near the trailing edge. As the incidence is increased to $\alpha = 14$ deg, most of the flow has separated, and a vortex locus is visible near the leading edge. This causes an abrupt decrease in the lift and pitching moment coefficients, which is not visible in the experimental result. Moving to higher values of α , the locus translates toward the root.

Consequently, the separation region increases in size and peaks at $\alpha = 18$ deg. Due to a symmetry boundary condition on the plane at $y = 0$, the vortex cannot proceed any further. Since the vortex remains over the suction side, the lift coefficient stagnates even in deep-stall conditions. This phenomenon is visible both in the numerical and experimental results.

However, the boundary condition applied at the tail root does not faithfully reproduce the flow environment present in the wind-tunnel experiments. In the experimental configuration, this region is immersed in the boundary layer developing along the wind-tunnel floor, which significantly influences the spanwise evolution of the separated flow. This discrepancy is evident in Fig. 10 for an angle of attack of $\alpha = 22$ deg, where differences in the spanwise position of the vortex locus can be observed. The physical constraint imposed by the wind-tunnel boundary layer on the inboard migration of the recirculating region is stronger than that introduced by the symmetry condition employed in the RANS simulations. Consequently, the experimental vortex footprint retains a larger spanwise extent, which explains the higher lift coefficients measured experimentally at the same angle of attack.

With a large deflection of the control surface ($\delta \geq 20$ deg), non-linear phenomena appear in the linear part of the lift coefficient. To investigate this, the case with $\delta = 20$ deg is considered. Figure 11a shows the lift coefficient versus angle of attack for this geometry. The red rectangles help isolate such nonlinearities, which can be associated with stall at the control surface, as visible in Figs. 11b–11d. Here, the surface streamlines over the suction side of the lifting surface are reported, with arrows aligned with the skin-friction

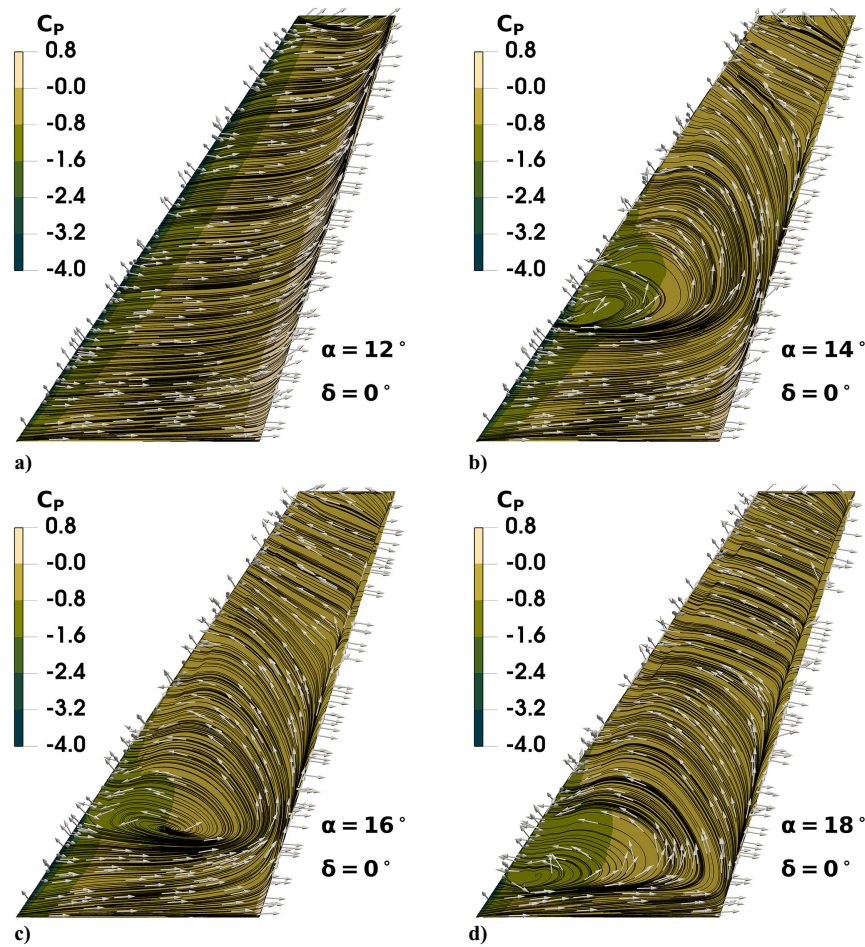


Fig. 9 Tail stall at positive angles of attack for $\Gamma = 0$ deg, $\delta = 0$ deg: pressure coefficient distribution and skin-friction visualization on the lower surface.

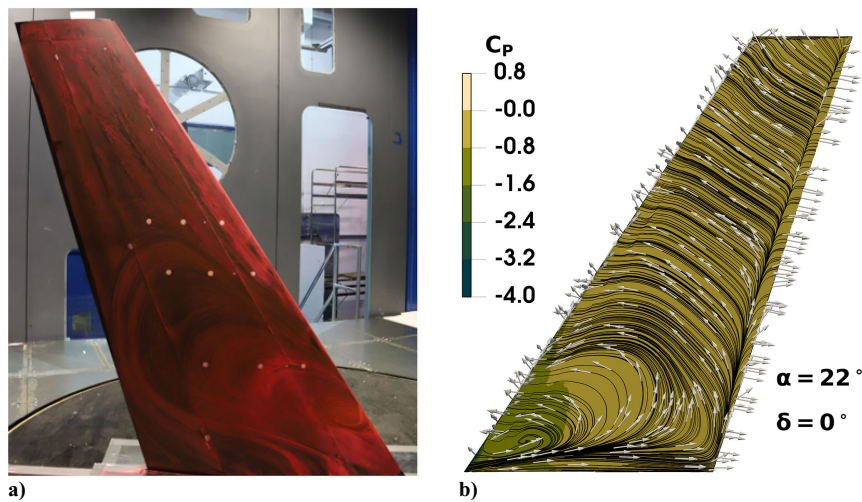


Fig. 10 Comparison of the spanwise location of the vortex locus on the suction side of the model: a) experimental china-clay flow visualization, and b) numerical RANS results with surface streamlines aligned with the skin-friction coefficient and colored by the pressure coefficient.

coefficient vector. The flow direction is from left to right, and the control surface is at the right end of the geometry. Each subsequent figure represents consecutive angles of attack, starting from $\alpha = -2$ deg up to 2 deg. Already at the lowest incidence, there is a small region of separated flow at the tip of the control surface. As the angle of attack is increased, this reverse flow region enlarges toward the root until, for $\alpha = 2$ deg, the control surface is completely stalled. This causes a small decrease in the lift coefficient, leading to the change in trend in the $c_L - \alpha$ curve. Since the stall is related

only to the rudder and the flow remains attached to the main surface, the lift coefficient keeps increasing with the angle of attack. The experimental results generally locate this separation approximately at incidences 2 deg lower than the RANS solutions.

The same qualitative behavior is also captured by the lower-fidelity approach, although the onset of separation is predicted at lower angles of attack. This is consistent with the underlying modeling assumptions, as the method is primarily based on two-dimensional simulations, although corrected to account for some

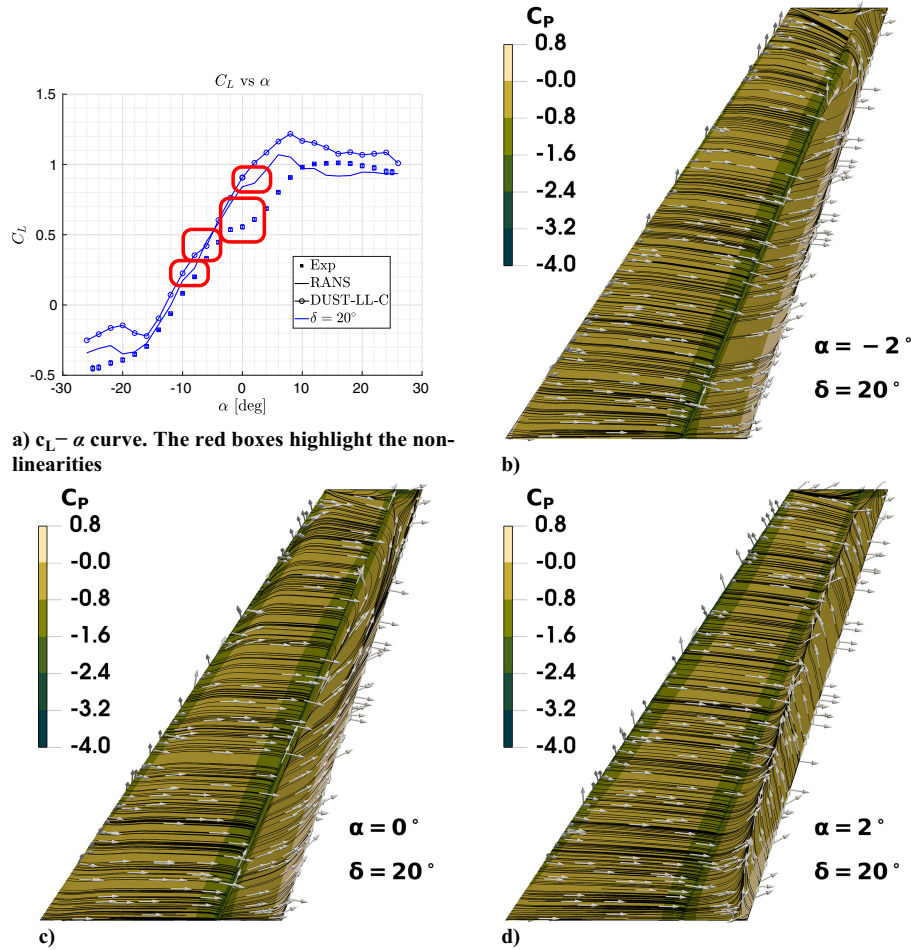


Fig. 11 Stall visualization at the control surface for $\Gamma = 0$ deg and $\delta = 20$ deg: a) lift curve and b–d) pressure distribution and skin-friction streamlines.

three-dimensional effects. As is well known, flow separation in purely two-dimensional analyses generally occurs prematurely compared to fully three-dimensional simulations, leading to a conservative prediction of stall onset.

V. Access to Database

The database was uploaded to the Zenodo Library under the MONNALISA Community [12]. It is openly available and easily accessible. The reference page with the description of the methodologies and the discussion of the results is available at the ERCOF-TAC Wiki [26]. In the following, we describe the structure of the database and explain how to access its content. A tree-like representation of the complete database for a single planform is shown in Fig. 12.

As stated in Sec. I, only Geom. I (the reference planform) has been fully characterized numerically via both mid- and high-fidelity methods. Nevertheless, the database is constantly updated with new simulations, including also the other two geometries.

A. Experimental Database

The experimental database is associated with a single upload due to its small size and contains the following files:

1) ***AerodynamicCoefficients.csv**: It contains the measured aerodynamic coefficients (c_L , c_D , and $c_{M,*}$) (and their standard deviation sigma*) for all the tail configurations tested in the wind tunnel. The geometric nondimensionalization parameters are the reference ones discussed in Sec. II.

2) **LeadingEdge_Geometries/**: This folder contains the database files in IGS format for the sectional airfoil profile with the three leading-edge geometries.

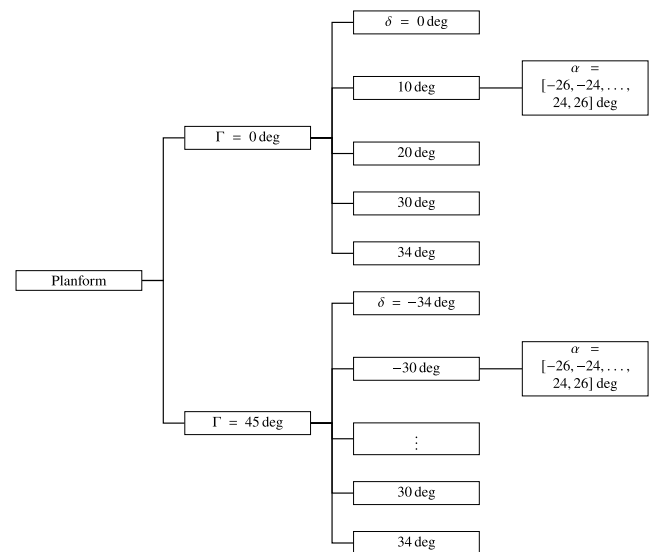


Fig. 12 Tree-structure of the experimental-numerical database encompassing all variables and parameters.

3) **SurfaceVisualization/**: This folder contains the surface images obtained through oil-paint visualization.

4) **IRThermography/**: This folder contains the infrared images from the thermal camera. The images were postprocessed to render the transition line more distinguishable. The final images are uploaded to the respective folder.

5) **PIV**: This folder contains the postprocessed PIV results, provided in Tecplot format (also compatible with ParaView). The dataset includes both static measurements at 0 deg, ± 30 deg, and dynamic measurements at 2 Hz for the same angles.

B. Numerical Database

Due to the large size of the computational flow solution (which is available for the entire volume mesh), the numerical database has been split into different uploads, each associated with a specific tail configuration. Since the numerical results have been computed only on the reference planform, the parameters that distinguish each upload are the dihedral angle (Γ) and the deflection of the control surface (δ). The name of the upload follows the notation *Num_DA_Γ_SA_30_AR_5_TR_0.4_Delta_δ*, where the Greek letters have to be substituted with the desired values. Data are gathered into two separate main folders to distinguish those obtained through high-fidelity (RANS) and through mid-/low-fidelity (DUST) simulations:

DUST: This contains the numerical predictions obtained using DUST v2-2018, and it is further divided into subfolders. These subfolders refer to a specific computational model:

- 1) Panel method (**PANELS**/)
- 2) Vortex lattice method (**VORTEXLATTICE**/)
- 3) Lifting-line theory with and without the crossflow correction (**LIFTINGLINE_CAL_COR**/ or **LIFTINGLINE**/).

Each subfolder contains a set of folders associated with the angle of attack and includes the following files:

- 1) DUST input files (*.in) for mesh generation, computation, and postprocessing
- 2) Geometry files (geo_input.h5, profile.dat) representative of the airfoil with imposed δ deflection
- 3) Solution files (solution.vtu, loads.dat, span_*.dat, in Binary ParaView File Format) and integral loads over the wing (*.dat)

The main folder of each DUST method (**DUST/Method**) contains the polar.dat file, which reports the integral values of the aerodynamic coefficients for each angle of attack.

RANS: This contains the polar for the first geometry, computed through a RANS approach. Each subfolder corresponds to a specific angle of attack and contains the following:

- 1) SU2 configuration files (*.cfg)
- 2) Mesh in CGNS format (mesh.cgns): To replicate the results, depending on the SU2 version, it may be necessary to modify the boundary conditions in the configuration file.
- 3) Solution files (*.vtu, *.dat), i.e., the volume and surface solutions in binary ParaView file format. Note that simulation results are nondimensional, and scaling may be necessary.

Each folder also contains the file Coeffs_Delta_δ.csv, which is a table containing the postprocessed aerodynamic coefficients for different values of δ .

VI. Conclusions

The objective of this work was to present a multifidelity experimental–numerical database for the aerodynamic assessment of unconventional low-aspect-ratio horizontal tails, including a full-span control surface. The database collects a large amount of data corresponding to a wide range of angles of attack and deflection angles of the control surface. The dataset combines numerical and experimental data sources, structured by increasing levels of fidelity. These include midfidelity CFD solutions based on a lifting-line method, high-fidelity RANS simulations with laminar-to-turbulent transition modeling, and wind-tunnel measurements conducted in a low-turbulence environment.

An initial comparison among these methodologies has been conducted for the reference geometry with zero dihedral angle. The lifting-line method demonstrated good agreement with experimental data in the linear regime of lift and drag for small control surface deflections, accurately predicting stall onset and general trends in aerodynamic coefficients. However, at higher angles of attack and deflection, the method showed limitations, particularly in

capturing the complex flow phenomena induced by flow separation and control surface stall.

The RANS simulations, while computationally more intensive, offer improved fidelity and more accurate prediction of nonlinear effects and flow separation patterns, particularly at large control surface deflections. Notably, the stall of the control surface was captured with high detail and showed good correlation with experimental observations, though slight discrepancies in stall onset remain, likely due to modeling limitations or experimental uncertainties.

Overall, this multifidelity database provides a comprehensive foundation for validation and development of reduced-order models and control algorithms for aircraft with low-aspect-ratio empennage configurations. A more exhaustive analysis of the full dataset, including additional geometries and deflection settings, will be the subject of future publications.

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