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Predictions of stability and control for a flying wing

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A B S T R A C T

The numerical simulation of a generic reduced radar signature tailless aircraft is considered. Investigation compares simulated data to low-speed wind tunnel experiments. Focus is on numerical predictions of steady longitudinal and lateral aerodynamics and influence of control surfaces on aerodynamic forces. Fully turbulent and transitional Reynolds Averaged Navier–Stokes (RANS) simulations predicted in agreement with experiment unstable pitch characteristics for low angles of attack (α), this was not the case for inviscid or laminar simulations. However, all simulations captured a sudden rapid increase in nose up pitch moment at higher angles of attack compared to experiments. Time accurate computations (URANS) captured non-linearity and unsteadiness in yaw moment with respect to differential split flap deflections for the studied angles of attack.

1. Introduction

In the pursuit to design aircraft with low radar signature for military application, the traditional tail has been removed and all-flying wing concepts considered. The operational B2 Spirit and the demonstrator X-45C are two examples and research is currently being performed on other similar concepts [7]. The purpose to reduce radar signature is relatively new, however, the tailless concept is not. Already in the 1930's the German Horten brothers developed tailless gliders with the aim to achieve better performance, i.e. less drag [9]. In the 1940's and 1950's, military powered tailless aircraft such as the Ho 5 and the N1-M designed by Jack Northrop were developed with the aim to fly long distances, particularly crossing the Atlantic ocean [3]. In recent years, research for civil applications with the goal to reduce fuel burn by improving the performance has become of interest. Here, the blended wing body concept is currently being studied in several research groups [8,21,22], and in a joint project by NASA and Boeing [11] flight tests with remotely controlled scale models, the X-48B and the X-48C, have been successfully performed.

For any new aircraft concept, it is essential to be able to predict aerodynamics and its influence on stability and control prior to flight tests. Wind tunnel experiments are fundamental, however, quite expensive and time consuming. Hence, Computational Fluid Dynamics (CFD) plays an increasingly important role in design and validation. For the tailless design it is of particular interest to be able to predict the pitch and yaw characteristics, due to the lack of a conventional stabilizer and rudder. To introduce better pitch authority, the solution has commonly been to sweep the wings backwards and thereby increasing the lever of longitudinal control. In addition, the sweep leads to a beneficial radar cross section. However, sweep introduces vortex flow for higher angles of attack, which in turn is expected to influence the pitch moment. Yaw control can be achieved by producing unsymmetrical drag using for instance split ailerons. In contrast to the response in yaw moment for a conventional rudder, the yaw moment with respect to differential drag is likely to be different. In order to rely on computational stability and control predictions it is essential that the flow simulation methods are validated and evaluated to state-of-the-art wind tunnel tests and/or in-flight data.

In this work CFD methods are assessed by comparing CFD results to low-speed wind tunnel test results for a generic tailless aircraft configuration called *Swing*, Fig. 1. In earlier work on a similar configuration termed the Stability and Control Configuration (SACCON), it was shown that adequate predictions are hard to achieve on the SACCON flying wing concept, Tomac et al. [20]. Wind tunnel tests were performed both before and after the installation of four control surfaces on the Swing model. These are mounted next to each other, two on each trailing edge of the wings. For yaw control, differential drag is applied by deflecting flaps on one wing only, where one flap is deflected up and the other down. This will be referred to as a split flap deflection in this study. In previous work, experimental studies have shown, that the pitch moment with respect to angle of attack is strongly influenced by the vortex flow on the wing [13]. A following study revealed that the yaw response created by split flap deflections was highly nonlinear and dependent on which flap was deflected up or down,

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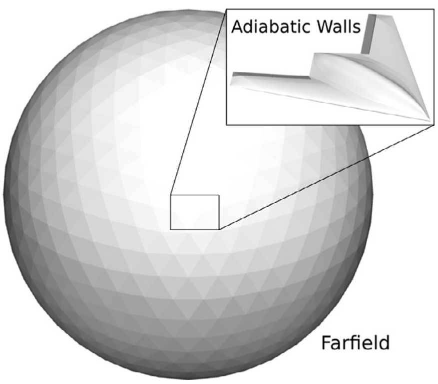
1270-9638/

(a) The Swing model in the low-speed wind tunnel.



(b) Boundary conditions outline of computational model.

Fig. 1. The Swing model (a) with rear mounted sting in low-speed wind tunnel. Outline of boundaries of the computational model (b).

respectively [15]. In the following work, several fidelity levels of CFD methods are used to assess the aerodynamics of the *Swing* model.

2. Experimental and numerical setup

Fig. 2 and Table 1 outline the *Swing* model planform and geometric data respectively. The model has a NACA-66009 airfoil over

Table 1
Dimensions of the wind tunnel model.

Entity	Notation	Value
Span	b	1.0 m
Mean chord	c_{ref}	0.3373 m
Wing area	S_{ref}	0.3373 m
Moment reference point	MRP	[0.439 0.0 0.0] m
Leading edge sweep	Λ	56°
Trailing edge flap chord	c_{TE}	0.04 m
Trailing edge flap deflection interval	δ	±25°
Wing airfoil profile	1	NACA-66009

the span of one meter with a leading edge sweep of 56 degrees and an outer twist-up of 3 degrees. Note that the NACA-66009 profile used is symmetric, thus flow breakdown at fairly low alpha is expected. Due to the requirements to resemble a UAV with low radar signature, the angles in the geometry are few and many edges are parallel. This resulting in a pointy zero chord wingtip, where conventional aircraft have a defined wingtip chord parallel to the flow direction. Further details of the model and its structure are found in technical report [13]. The experimental tests were performed in the low-speed wind tunnel L2000 at the Royal Institute of Technology (KTH). Further details of experimental data sets are presented by Stenfelt in [13–15]. In particular the data sets at 30 m/s, Mach = 0.125 ($Re = 6.9e^5$) were used in this study for comparison with numerical results.

The spatial discretization for the computational model is done by first producing surface grids using `icemcfd` [1]. When the surface grids are of acceptable quality, hybrid prismatic-tetrahedral grids are produced using the newly implemented RANS meshing capability in `sumo` [19]. Trailing edges are modeled as sharp and gaps between control surfaces and wing are included. However, cavity geometry inside control surfaces is not considered and is filled and replaced by flat walls. Wind tunnel walls and sting mounting are as well not included in the numerical simulation, Fig. 1(b). The final typical grid consisted of around 35 million grid points and the surface is typically resolved by approximately 1 million surface nodes. The viscous boundary layer is resolved by 35 prismatic layers with an initial height of $1e^{-6}$ m corresponding to a y^+ value of less than one. In Fig. 3 the surface and volumetric resolution are shown.

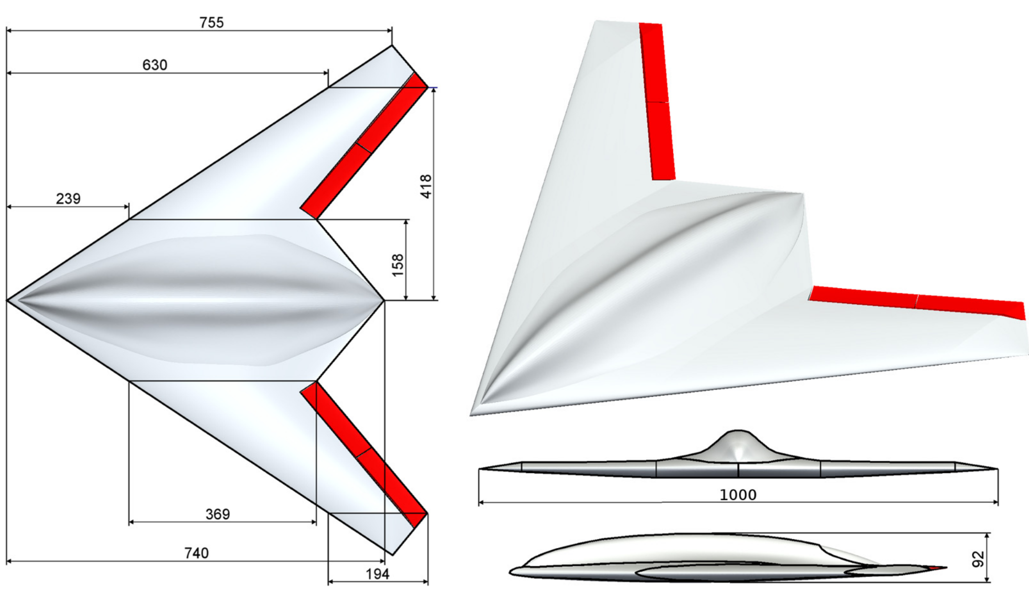


Fig. 2. Definition of the Swing planform, control surfaces colored in red, sizes are given in [mm]. (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)