AERODYNAMIC ANALYSIS OF A SUSTAINABLE-ENERGY SAILPLANE USING CEASIOM

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March 2013

Master’s Thesis in Sustainable Energy Engineering
Computers are extremely fast, accurate and stupid. Human beings are notoriously slow, sloppy and highly intelligent. When the intelligence of humans is coupled with the speed and accuracy of computers, data processing that far outreaches anything in the past becomes possible.

Robert B. O’Connor
in *Journal of Occupational Medicine*, April 1968
Acknowledgments

This page is devoted to greatly thanks to those who have provided me with invaluable support during the development of this Master program.

To Professor Arthur Rizzi, my supervisor, for offering me such an amazing topic and his continuous support throughout the entire work. Special thanks to Mengmeng Zhang, who always supported me with advice on the work, her comments were always important to achieve advances in the thesis. I learned a lot from this whole process, which would not be possible without their support.

To our facilitator in the distance program, Professor Fabio Sierra, and the entire Colombian group, the union we had through the program helped me to come up to this point.

To my beloved family: parents and wife. Without their support none of this would have ever happen.

To God, He is always there.
Abstract

This master thesis explains in detailed the process followed to make an aerodynamic analysis of the Warsaw University of Technology sailplane, The PW-5 Smyk, using a computerized environment developed as part of the SimSac project: CEASIOM. Throughout the document it is possible to find a step-by-step description on how the geometry should be defined in the CEASIOM’s stand alone module for Vortex Lattice Method solutions, Tornado. Besides, it provides a full description of the grid generation and the criteria used to determine if a mesh could be adequate for the specific purpose. At this point a mesh validation is carried by comparing aerodynamic predictions with experimental measurements taken during the sailplane’s flight. Moreover, once the domain discretization and calculation parameters have proven to be effective, a series of flight states are defined in Tornado with the aim of evaluate the airplane aerodynamics during different operating conditions. In fact, those computations were made following a special organization, which is required by SDSA in order to produce graphics as a solution found by the VLM code. At the end of the document the reader will find the presentation of the main plots achieved with CEASIOM and a short discussion on the implications that each of them has on the aircraft’s aerodynamics.
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<td>$Q$</td>
<td>Pitch rate</td>
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<td>$Q_{\text{int}}$</td>
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<td>Reynolds number</td>
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<td>$RS$</td>
<td>Rate of sink</td>
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<td>$S$</td>
<td>Reference wing area</td>
<td>$\text{m}^2$</td>
</tr>
<tr>
<td>$S_{\text{ref}}$</td>
<td>Reference area</td>
<td>$\text{m}^2$</td>
</tr>
<tr>
<td>$S_{\text{wet}}$</td>
<td>Wetted area</td>
<td>$\text{m}^2$</td>
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<td>$TAS$</td>
<td>True airspeed</td>
<td>$\frac{\text{m}}{\text{s}}$</td>
</tr>
<tr>
<td>$V$</td>
<td>Speed</td>
<td>$\frac{\text{m}}{\text{s}}$</td>
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<tr>
<td>$V_c$</td>
<td>Compressibility correction to the speed</td>
<td>$\frac{\text{m}}{\text{s}}$</td>
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<tr>
<td>$W$</td>
<td>Airplane weight</td>
<td>$\text{N}$</td>
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<tr>
<td>$x_{J-o}$</td>
<td>Distance from the leading edge to the panel edge</td>
<td>$\text{m}$</td>
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**Greek letters**

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<thead>
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<td>$\alpha$</td>
<td>Angle of attack</td>
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<tr>
<td>$\beta$</td>
<td>Sideslip</td>
<td>$^\circ$</td>
</tr>
<tr>
<td>$\delta_a$</td>
<td>Ailerons deflection angle</td>
<td>$^\circ$</td>
</tr>
<tr>
<td>$\delta_c$</td>
<td>Canard deflection angle</td>
<td>$^\circ$</td>
</tr>
<tr>
<td>$\delta_e$</td>
<td>Elevators deflection angle</td>
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</tr>
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<td>$\delta_r$</td>
<td>Rudder deflection angle</td>
<td>$^\circ$</td>
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<td>$\varepsilon$</td>
<td>Wing twist</td>
<td>$^\circ$</td>
</tr>
<tr>
<td>$\gamma$</td>
<td>Angle of descent</td>
<td>$^\circ$</td>
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<td>$\Gamma$</td>
<td>Dihedral angle</td>
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<tr>
<td>$\lambda$</td>
<td>Taper ratio</td>
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<tr>
<td>$\Lambda_{c/4}$</td>
<td>Sweep angle of the quarter chord line</td>
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<tr>
<td>$\rho$</td>
<td>Density</td>
<td>$\frac{kg}{m^3}$</td>
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**Subscripts**

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<td>Parameter closer to the wing root</td>
</tr>
<tr>
<td>outer</td>
<td>Parameter closer to the wing tip</td>
</tr>
<tr>
<td>r</td>
<td>Value measured at the wing root</td>
</tr>
<tr>
<td>SL</td>
<td>Condition found at sea level</td>
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**Abbreviations**

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<td>CAD</td>
<td>Computer Aided Design</td>
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<tr>
<td>CEASiOM</td>
<td>Computerized Environment for Aircraft Synthesis and Integrated Optimization Methods</td>
</tr>
<tr>
<td>CFD</td>
<td>Computational Fluid Dynamics</td>
</tr>
<tr>
<td>CG</td>
<td>Center of Gravity</td>
</tr>
<tr>
<td>FAA</td>
<td>Federal Aviation Administration</td>
</tr>
<tr>
<td>KTH</td>
<td>Royal Institute of Technology</td>
</tr>
<tr>
<td>MAC</td>
<td>Mean Aerodynamic Chord</td>
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<tr>
<td>PreSTo</td>
<td>Preliminary Sizing Tool</td>
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<tr>
<td>RANS</td>
<td>Reynolds-Averaged Navier–Stokes</td>
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<tr>
<td>SDSA</td>
<td>Simulation and Dynamic Stability Analysis</td>
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<td>TCR</td>
<td>Transonic Cruiser</td>
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<td>VLM</td>
<td>Vortex Lattice Method</td>
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1 Introduction

Emissions of carbon dioxide coming from aviation accounted for about 2.5% of the global average production in 2007 and, with a projected growth of 4.8% per year in the industry, it is expected that the aviation participation in CO$_2$ generation increases even more [32]. This is one of the main reasons why European Union adopted the proposal of integrate international and domestic aviation in the European Union’s Emissions Trading Scheme in 2008 [32]. Indeed, even though over the past years there has been good improvements in aviation emission intensity (44% decrease of fuel intensity per revenue passenger kilometer [1]), carbon dioxide production has increased in this industry since the early 1990’s [21]. Consequently, there are some initiatives to use non motorized transportation, which is crucial for sustainable development [23], because of air pollution caused by emissions and the decrease in petroleum reserves.

The Computerized Environment for Aircraft Synthesis and Integrated Optimization Methods Software (CEASIOM) was developed as part of the Simulating Aircraft Stability and Control Characteristics for Use in Conceptual Design (SimSac Project) as a tool for improve the effectiveness of the initial stage in aircraft design: conceptual design. Indeed, is in this phase where decisions taken worth around 80% of the life-cycle cost of an airplane [37]. Moreover, while decreasing cost at this level, the tool increases fidelity in initial stages of a project. In fact, it permits a designer to achieve better aerodynamic characteristics in the actual airplane when the manufacture is finished, since fidelity in early stages of a project is better for a the virtual model than that achievable by the classical approach [37]. Therefore, attaining higher aerodynamic qualities would help airplanes to get better performance during its operation, which could potentiate sustainability in aviation. For instance, a decrease of 1 drag count ($C_D \approx 0.0001$) in supersonic cruise drag could reduce airplane’s gross weight by 10,000 pounds and save up to 7,500 pounds of fuel for high speed civil transport vehicles [18, 19]. From this point of view, it is possible to assume CEASIOM as a tool that could help aviation industry to improve its sustainability.

Sailplanes are a sustainable type of transport, since they do not burn fossil fuels to fly. They rather use potential energy to descent unpowered or rising hot air streams for soaring. The more aerodynamically efficient a sailplane is, more horizontal distance can cover during its flight. Adaptations in CEASIOM’s code for analysis of gliders could improve their
aerodynamic performance through improvements in fidelity of early design phases. As a consequence, having more efficient gliders could enhance the utilization of this environmentally friendly kind of transport.

The environment used to make the aerodynamic analysis of the sailplane, CEASIOM, has proven to be very effective when analyzing aircrafts. Indeed, it has been used to make analysis of different types of aircraft ranging from conventional big civil transport [35] to small jet trainer [10, 11] or even asymmetric aircraft configurations [36] whose results are available in the literature.

Firstly, it is required to know the detailed geometrical configuration of the selected sailplane. This is important to create the virtual model of such an airplane in the VLM module for aerodynamic computations (Tornado), where all the main geometric features are defined and located according to the sailplane configuration. In addition, it is required to establish a proper mesh, which will be used by the code to perform discretized calculations over the entire domain. This whole process is explained in detail during Chapter 3.

Secondly, once the geometry and mesh have been properly defined in the VLM CEASIOM’s module, it is important to determine if these are able to provide a close description of the aerodynamic phenomena around the actual airplane. Consequently, it is required to have experimental data about the glider’s flight in order to compare it with VLM results and validate the geometry, mesh and calculation parameters introduced to the code. Throughout Chapter 4 is possible to find a detailed description on how the mesh and inputs to Tornado were validated and how close they are to on-flight measurements.

Thirdly, results coming from Tornado may be translated into the CEASIOM’s module for Simulation and Dynamic Stability Analysis (SDSA). Therefore, it is important to arrange the aerodynamic outputs in an acceptable format for SDSA. Chapter 5 provides a description of how this process was done. Furthermore, it is possible to find a short description on how the main angles and rotations are taken as positive by the code.

Finally, Chapter 6 presents an analysis of the aerodynamic results obtained with CEASIOM. It is possible to find some comments about the aerodynamic implications of predictions calculated by the code. Results are presented in graphics, where each of them has its respective discussion. In addition, at the end of the chapter, a summary of the aerodynamic analysis carried on the PW-5 Smyk sailplane is performed.
2 Description of the Computational Environment

The tools used for the analyses presented in this document were all developed as a part of the Computerized Environment for Aircraft Synthesis and Integrated Optimization Methods (CEASIOM), which is aimed to support engineers during early stages of an aircraft design. In fact, CEASIOM is able to predict, through one application the aerodynamics, structure, flight dynamics and performance characteristics a design will likely have during its operation. Therefore, it permits a general evaluation-feedback of an early virtual model, which increases the fidelity of a given design from its conceptual design.

The Software, which according to von Kaenel et. al. [44] was started from the Isikveren’s doctoral thesis [15], requires an initial quite simple definition of the model and refines it through the user’s introduction of additional parameters as long as the design phase advances toward the preliminary design. Once a virtual model has been defined it is possible to make different analyses over the design, based on the models and modules selected by the user. Figure 2-1 shows a summary of the CEASIOM’s functionality and the connectivity between its modules.

First of all, there are the modules for airplane’s geometry generation based on its parametrization: AcBuilder and SUMO. The virtual model created by former module is suitable to be used in calculations through both, Vortex Lattice Method and DATCOM, which is a compendium of experimental computations [9]. However, the geometry created on it needs to be transferred to SUMO if higher fidelity tools are to be used. In fact, geometry from this last module can be directly used for panels methods analyses, for Euler solutions though Tet-Gen and for computations of Reynolds-Averaged Navier–Stokes (RANS) equations through ICEM. All of these computing methods are available in CEASIOM through its Aerodynamic module AMB-CFD.

After an aerodynamic evaluation of the airplane’s geometry, it is possible to evaluate the model in several ways. One of the possibilities that arises at this point in the Environment is the evaluation of the model’s statical and dynamical stability, along with performance predictions and test flight simulations, with the Stability and Control module S&C. Moreover, it is also possible to use the commercially available J2 Universal Tool-Kit, which according
to its developers is “a powerful analysis, modeling, performance prediction and simulation framework for fixed wing aircraft design and evaluation” [16]. Furthermore, CEASIOM has additional modules available, whose inputs are taken from the outputs obtained with AMB-CFD module, these are analyses of aeroelasticity (NeoCASS), flight control system design (FCSDT) and decision support system (DSS).

As mentioned before, there are different levels of fidelity when CFD tools are used in an airplane design process. According to Rizzi [37], this term is associated “to the representation of the aircraft geometry and of the physical modeling that determines the aircraft behavior and performance”. Consequently, for higher levels of fidelity more detailed geometrical characteristics are required to be introduced for more complex models in the Software, which generates greater accuracy in predictions and takes longer to achieve good results. Furthermore, low-fidelity tools should be used during the conceptual design in order to be able of evaluate different possibilities, while high-fidelity are supposed to be used in more advanced phases in the airplane’s design [37]. Figure 2-2 schematizes the levels of fidelity as a function of airplane’s geometrical representation quality and CFD tools.

The purpose of this work is to present an aerodynamic analysis in a CFD tool suitable to be used from the initial stages of the conceptual design. In fact, sustainability in aviation may be achieved by taking care of the airplane’s aerodynamics from its earliest conception. Therefore, the study in following chapters will be done using the Vortex Lattice Method.

Figure 2-1: Overview of CEASIOM’s process and dataflow [37].
approximation offered by CEASIOM. Even though VLM is on the next level of fidelity when compared with DATCOM [6], this last method is not considered as a CFD tool (See Figure 2-2), and as a consequence, the study here does not use this kind of methodology to analyze the aircraft.

In addition, VLM calculations, which are done in the CEASIOM’s module Tornado, are suitable to achieve reasonable accurate results requiring low time of computation when compared to that taken by higher fidelity tools. Moreover, it is expected that a sailplane, the sustainable energy type of airplane chosen for the analysis, develops a low-speed flight with small angles of attack. Therefore, mathematical models assuming linear aerodynamics are suitable for the study of a glider.

As the aerodynamic predictions based on the geometry were mainly focused in the VLM method, they were done in the Tornado stand-alone code (CEASIOM’s module for VLM computations). Therefore, it was required to translate its outputs into a file readable by SDS (CEASIOM’s module for Stability and Control S&C). In fact, such procedure was necessary because the AMB module is by-passed when the stand-alone modules are used for computations.

Figure 2-2: Levels of complexity and fidelity of CFD tools [39].
3 Geometry and Mesh for Aerodynamic Computations

The Software used to develop the aerodynamic analysis of the sailplane, Tornado, is a code written in MATLAB as a Master Thesis in the Royal Institute of Technology [25] and further developed with the collaboration between this Institute and the Universities of Bristol and Linköping [24]. Tornado was implemented as a Vortex Lattice Method (VLM) solver to calculate linear aerodynamics on fixed wing aircrafts, and what makes it different from other VLM codes is its extension for the theoretical solution an its geometry definition [24]. Its primary purpose was to find aerodynamic forces on airplanes flying at low subsonic speeds, which is the case of the PW-5 glider. Furthermore, Tornado can be used as an integrated part contained within the environment or as stand-alone application. All the study presented in this section is done with the code as a stand-alone application.

European researches have made analysis using the code in order to compare its accuracy among others, empirical methods and experimental information [20, 35, 47]. In general terms, they found that Tornado produces favorable results when compared against other codes and with experimental measurements. Moreover, they suggest it for designs different from conventional configurations (which is the case of the studied glider) and cases where time plays an important role. Consequently Tornado has been widely used to produce aerodynamic computations at low aerodynamic speeds [5, 6, 38, 43, 44].

Present chapter describes the methodology applied to produce good aerodynamic results in the Tornado code. As it is a computational tool to numerically solve the Vortex Lattice Method (VLM) it requires a grid, where equations are to be solved for the prediction of some aerodynamic characteristics of an airplane. Therefore, as a mesh must be created over an appropriate geometry definition, it is also important to introduce in the code the dimensions of the glider for accurately resemble the actual shape of the aircraft. Furthermore, once the domain has been correctly discretized it is necessary to define the flight conditions in which the predictions will be made.

This part of the document describes the criteria followed to obtain a good aerodynamic description of the PW-5 sailplane. Some processes were selected according to successful
results reported in literature. In Chapter 4 a validation of results is made in order to evaluate if the virtual model defined in the code agrees with the actual sailplane.

### 3.1 Geometry

As mentioned in the preamble, Vortex Lattice Method, which is the procedure followed by Tornado, is a numerical method based on the discretization of the domain where the calculation is going to be applied. Therefore, it is of critical importance the definition of an adequate mesh that helps to capture as well as possible all the critical aspects of the airflow around the lifting surfaces. Consequently, it is necessary to devote a section to describe the process followed to determine a proper mesh for VLM calculation, which is aimed to predict the aerodynamic characteristics of the PW-5 sailplane. The criteria used to determine the final mesh is exposed in the following subsections.

It is only necessary to introduce the main characteristics of wings in Tornado since the code does not require a fuselage to be defined, if the calculation does not need a parasite drag estimation [20]. Therefore, it is important to keep this limitation in mind, since the absence of fuselage neglects the contribution of it to important aerodynamic characteristics, as moment coefficients, lift or drag related to interference between components. However, Tornado is capable of produce an estimation of the body drag, which is explained and developed in Section 4.2. In addition to a detailed description of the main wing, tail unit (horizontal and vertical) must also be defined as a couple of wings. In general terms the geometry of the wings was defined as follows:

- **Wing 1**: This is the main wing of the airplane; it was divided in three partitions, one containing the section from the root until the aileron, other containing its control surface and the last one from the end of the aileron until the tip of the wing. The airfoil used in this part of the aircraft was *NN 18-17* (Fig. 3.1(a)) without aerodynamic twist. In addition, ailerons on it were assigned to have unsymmetrical deflection.

- **Wing 2**: This is the horizontal stabilizer which only has one partition since it is a rectangular wing and the elevator spans entirely on it. The airfoil used in the tail unit (vertical and horizontal) was a *Wortman FX-71-L-150/30* (Fig. 3.1(b)). Elevators, which are located in this “wing” move symmetrically, and as a consequence their movement was defined in this way on Tornado.

- **Wing 3**: This is the vertical stabilizer which was divided in two partitions: the former, from the root until its control surface (the rudder) and one containing it. Due to the characteristics of Tornado, this wing was introduced as non symmetric in the $x$-$z$ plane and with a dihedral angle of 90°.
(a) Main wing: NN 18-17

(b) Tail unit: Wortman FX-71-L-150/30

**Figure 3-1:** Airfoils used by the PW-5 sailplane

**Figure 3-2:** Top view of the digital model compared with the actual geometry
Table 3-1 summarizes the main geometrical characteristics introduced in Tornado. This was obtained by approximating high resolution images of the PW-5 glider with a CAD Software. This methodology was required, since it was not possible to get access to detailed drawings and information of the airplane design. The geometry generated with this code is shown in Figures 3-2 and 3-3.

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3.2 Mesh design

In Tornado it is possible to generate 5 different types of meshes for VLM calculation: (1) Linear, (2) Spanwise half-cosine, (3) Spanwise half-cosine plus chordwise cosine, (4) Spanwise cosine and (5) Chordwise cosine. In order to decide which of these kinds would be selected a literature review was done for an evaluation of previous outcomes. On one hand, Woodcock [45] found that coarser meshes with smaller cells close to the leading edge could reach the same accuracy as that reached by a finer mesh with constant spacing chordwise, this implies that chordwise cosine is a good option when trying to efficiently use the computational resources. This is also supported by Jackson and Fiddes and Fiddes and Gaydon who probed full cosine tendency to produce better results than other types of meshes [8]. Furthermore, Burger [4] found that full cosine distribution in the chordwise direction and half cosine in the spanwise direction produce a faster convergence than full cosine chordwise and even spacing spanwise.

With regard to the number of panels, different authors have found that around 20 to 30 cells spanwise generates very accurate results with errors of about 1% to 2% [7, 13, 28] and between 4 and 10 panels chordwise have reached good accuracy when using full cosine
In addition, as the control surface requires a different discretization, it is possible to find that Mendenhall and Spangler successfully used 4 panels chordwise in a flap test for a NASA Contractor Report [26].

![Figure 3-4: Mesh with 140 panels used for chordwise convergence analysis](image)

The discussion developed above gives some insight on how the mesh should be done for this specific case if aiming to generate as good results as possible. As a direct consequence of the literature review, the type of grid selected for the PW-5 sailplane contains finer cells to the edges of wings: tips, leading and trailing edges. In fact, as mentioned before, the main wing was divided in three partitions, where the two inner portions were assigned to have a chordwise cosine and linear spanwise distribution (type 5), while spanwise half-cosine and chordwise cosine (type 3) was selected for the outer division. The reason for only using half-cosine in partition number three is because it contains the tip of the wing (Figure 3-4 shows the type of mesh used in the main wing). Similar methodology was followed for the vertical tail, since it has two partitions, where the last one was selected to be a cosine grid in both directions. The horizontal tail was only one partition which entirely had a Spanwise half-cosine and Chordwise cosine mesh (type 3).
3.3 Convergence analysis

Even though facts extracted from previous researches, discussed in Section 3.2, offer a general view on how the mesh should be constructed, there is still the need to make a convergence analysis which helps to determine the final configuration of the grid. In fact, consulted authors reached appropriate results with their calculations, but their scales were different from that required for the present document, which reinforces the need of a convergence analysis.

At an earliest approach, increasing simultaneously the number of panels in the main wing and tail unit in both directions was attempted. However, this showed different results without a clear tendency line. From these results, it was concluded that a process with two independent convergence analyses should be done: one for chordwise increments and other for changes in the spanwise direction. Moreover, it was decided that the convergence analysis would be done only on the main wing and the chosen mesh would be applied to the tail unit.

In this sort of ideas, it was first selected to make a convergence analysis changing number of panels in chordwise direction and keeping a constant number of panels spanwise; results and full description of this study are presented further in this Chapter. After the chordwise analysis, as outcome, a number of panels in chord direction was chosen to start the spanwise convergence study, varying grid density in this direction, while maintaining constant the amount found to be adequate in the previous analysis. This whole process is fully documented in following paragraphs.

Another important aspect of the convergence analysis is that the glider’s wing was analyzed at two different angles of attack ($\alpha = 0^\circ$ and $\alpha = 8^\circ$) aiming to check the independence of the mesh convergence to the operating condition. The airspeed at sea level for both flight conditions was $25 \, m/s$ ($90 \, km/h$) which, according to the sailplane flight manual [46], is contained within the range of the aircraft normal operating range, since its indicated air-speed during an average operation commonly goes from $75 \, km/h$ to $150 \, km/h$ [46].

3.4 Chordwise convergence

As mentioned before, this analysis was made by taking a constant number of spanwise panels on the wing while increasing the amount of panels chordwise. In wingspan direction a 28 panels mesh was set, based on the results and conclusions found by different authors (see discussion on Section 3.2). Another factor taken into account when defining the number of panels spanwise was that the divisions of partitions 1 and 2 of the main wing, and the inner portion of partition 3, were aimed to have similar length.
The analysis in the chord direction was made on eleven different meshes, starting from 4 panels with increments of one division up to 12, plus two additional grids with 14 an 16 divisions chordwise. Figure 3.4 shows values and tendency achieved by the simulations with the different grids for lift (Fig. 3.5(a)) and drag (Fig. 3.5(b)) coefficients as a function of the number of panels in the main wing. From these figures it is possible to observe that results from meshes with approximately more than 300 panels started to show little change with increases in the density of the discretization. Nevertheless, it is not only important to achieve almost invariant results between increments of divisions, but also attain an efficient use of computational results. Therefore, a proper selection of chordwise number of panels would be 11, which produced a mesh of 308 panels, because nearly constant results were generated from this point. As can be noted from previous discussions, this amount is closely related with that found by previous researchers (see Section 3.2).

![Graphs](image)

**Figure 3-5:** Convergence analysis varying number of panels chordwise

### 3.5 Spanwise convergence

With the selection of 11 panels chordwise it is now necessary to evaluate the convergence for increments in the number of panels spanwise, while maintaining the number in the chord direction as a constant. The analysis was started by testing a discretization with 24 panels spanwise (a total of 264 divisions), which was gradually increased by two elements in the first division of the wing for each new grid. Augmentation in remaining partitions was established by trying to keep panels of similar length throughout the wing (except for the finner tip elements, due to the cosine distribution).
Figures 3.6(a) and 3.6(b) show results and tendencies obtained from the spanwise convergence analysis. It is possible to note that meshes with nearly 1000 panels are achieving convergence, since variations in lift (Fig. 3.6(a)) and drag (Fig. 3.6(b)) coefficients are small while the number of divisions in the grid is augmented. It was necessary to generate a 96 panels spanwise for this behavior started to appear, which is indeed much higher than the values found adequate by other authors (see Section 3.2).

![Figure 3.6](image)

**Figure 3-6**: Convergence analysis varying number of panels spanwise

López [20] used an additional criteria to evaluate grid convergence: when the variation among single results was lower than 1% the mesh was considered to be converged. Consequently, a similar criterion was chosen for the present document to be analyzed. Results obtained for such an evaluation over the sailplane’s main wing for $\alpha = 0^\circ$ are presented in Figure 3-7. As figure shows, difference between most results is really low, lower than 1%, which could probably be because a previous convergence analysis was carried before for the chordwise direction. For the analysis developed in present research a difference lower than 0.5% is considered to be adequate.

From Figure 3-7 it is possible to note that meshes with higher resolutions than 350 panels fulfill the less than 0.5% criteria. Therefore, the grid with 462 panels, generated with 42 divisions spanwise, was selected because it accomplishes the convergence criteria and is less computationally expensive than finer grids. Furthermore, it permits the generation of meshes over the two unstudied wings without entirely consume the computer resources. Indeed, meshes with much more panels spanwise than chordwise are expected to reach good results. For instance, Professor Mason from the Virginia Polytechnic Institute found, when studying panel convergence over a F/A-18 [22], that using a large number of divisions in the span direction is important, while low amounts of chordwise panels were enough.
3.6 Application of the selected mesh size to the tail unit

As can be noted from preceding sections, the procedure developed to establish a proper mesh type and size were only focused on the main wing of the glider. Consequently, there is still the need to carry a discussion about how to discretize remaining parts of the aircraft: tail unit. The method followed to apply a proper grid on the vertical and horizontal tail is showed in this section.

\[
x_{J-o} = \frac{c}{2} \left( 1 - \cos \left( \frac{\pi J}{N} \right) \right)
\]

(3-1)

The most critical parts on the mesh to emulate good results attained in the main wing are those panels close to the boundaries of wings. Therefore, sizes for former panels from edges (tips, leading and trailing edges) are maintained within similar values than those found to be adequate in the previous convergence analysis. In this sense, it is possible to find in literature, that according to Miranda [27], the cosine distribution for a grid is build using Equation 3-1. Consequently, 3-1 is used to compute the proper panel size from the tip, leading and tail unit.
trailing edges of the tail unit. The mesh selected from this convergence analysis is shown in Figure 3-8, which is very much alike to those obtained in previous analogous works [24, 25].

Figure 3-8: Mesh selected for the VLM analysis with Tornado
4 Validation of Grid with Experimental Information

Before proceed further, it is necessary to evaluate if the discretization of the domain was done in a proper way, or if, instead the grid or geometry must be reconsidered. As mentioned in Section 3.1, there were a lack of an exact geometrical description of the sailplane PW-5 Smyk. As a consequence, this document presents rather an approximation to the measures of the wing and vertical and horizontal tails. Moreover, it is important to do an evaluation of the amount of panels assigned to the domain, since VLM is a numerical method highly dependent on the mesh quality in order to produce proper results. Therefore, the analysis presented in this section gives insight on the how good the meshing process was done.

This section provides the reader with a description on how the validation of the mesh and geometry was done. It offers an explanation of the setup established in Tornado, and how its results were operated in order to produce comparable results with available experimental measurements. Furthermore, it gives a short discussion about the experimental information found in literature.

4.1 Experimental information

The experimental data used to compare Tornado’s outputs is that published by Richard H. Johnson [17], who wrote performance results obtained through various flight tests carried on gliders, among them the PW-5. For this specific aircraft the outcomes from the test, where he mainly evaluated sink rates as a function of airspeed, were corrected for sea level conditions. The polar curve published by Johnson is shown in Figure 4-1, and as can be noted it contains information about the airplane’s weight and computes sink rates against calibrated airspeed (CAS) corrected at sea level conditions.

With the information showed in Figure 4-1 it is possible to compare how good the code’s prediction fit with measurements taken on the actual glider. As will be explained in Section 4.3, the polar curve of the sailplane can be easily related with the aerodynamic coefficients
obtained from Tornado ($C_L$ and $C_D$) through performance equations.

**Figure 4-1:** PW-5’s experimental polar curve [17].

### 4.2 Calculation setup

In order to do aerodynamic calculations on the glider, Tornado requires the definition of the grid and the type of calculation aimed to be solved. The former was that defined through a discussion developed during previous sections and whose final appearance is shown in Section 3.6. For the last, it is necessary to select a type of calculation among several options, for this specific case the option chosen was sequential alpha sweep solution in order to evaluate changes in lift and drag coefficients as a function of the angle of attack. In fact, those non-dimensional parameters can be used to compare results with the available experimental information.

It is important to keep in mind that the option selected to be solved produce a drag polar of the aircraft, which does not take into account the parasite drag. In other words, results coming from a default sequential alpha sweep solution only calculate the induced drag coefficient generated on the airplane [39]. Consequently, as the total drag coefficient is the sum of the parasite and induced terms, it is necessary to make an evaluation of the drag produced
4.2 Calculation setup

when the main wing is not generating lift and that produced by the fuselage (which has not been defined in the code yet). This whole process is explained in the following lines.

4.2.1 Induced drag coefficient

The portion of drag that arises due to the generation of lift is calculated by the code with the alpha sweep option, which is the type of computation selected for this validation. As its name suggests, it is necessary for this operation to determine a range of angles of attack to solve the VLM equations. Therefore, the calculations for the sailplane were made for the range $-7^\circ < \alpha < 10^\circ$ with increments of $0.5^\circ$ between predictions. Results obtained for such a process are showed in Figure 4-2.

![Induced Drag Polar](image)

**Figure 4-2:** Predicted induced drag polar

It is possible to observe from Figure 4-2 that the graph makes sense because it approaches to a quadratic shape, which is expected for this kind of airplanes. This result permits to obtain couples of coefficients $C_L$ and $C_D$ corresponding to each other and which can be used to be compared with the experimental measurements. However, from Figure 4-2 is noticeable that the drag coefficient when no lift is generated is too low. In fact, it is important to keep in mind that the computation made only accounts for induced drag, and parasite
drag still needs to be calculated (the operation in the code is explained below). After an estimation of the total drag coefficient (induced plus parasite), it is now possible to relate Tornado results with experimental information through performance equations (Section 4.3).

4.2.2 Parasite drag coefficient definition

The viscous drag of the digital model may be calculated for: (1) zero lift drag though a flat plate approximation and (2) the drag caused by the presence of a fuselage with the Eckerts equation (Equation 4-2). The former is obtained from the previously established mesh. For the blunt body drag Tornado uses Equation 4-1, which requires to define the diameter and the length of fuselage in order to compute the fineness ratio \((f = l/d)\) and the form factor \((FF)\) (Equation 4-3) [34]. The rest of parameters in Equation 4-1 are calculated by the code, except for the body interference factor \((Q_{int})\) which must be entered and, according to Raymer [34], it is negligible for most fuselages and can be taken as 1.

\[
C_{D_{body}} = \frac{C_f FF Q_{int} S_{wet}}{S_{ref}} \\
C_f = \frac{0.455}{(\log Re)^{2.58} (1 + 0.144 M^2)^{0.65}} \\
FF = 1 + \frac{60}{f^3} + \frac{f}{400}
\]

During some previous calculations it was found that if the diameter of the fuselage, for fineness ratio calculation, was taken as the maximum value in it \((d \approx 0.94 mts)\), the operation tends to generate over prediction of the parasite drag coefficient. Instead, if the diameter was taken as an average throughout the fuselage’s length \((d \approx 0.45 mts)\) the code produces an under prediction of the same coefficient. Consequently, Tornado was changed for this project and the ratio used in the calculation was that suggested by Hoerner [12] and showed in Equation 4-4. Therefore, based on fuselage design, this last equation gives a value of \(f\) equal to approximately 7.668. Moreover, the code requires fuselage’s diameter to compute its wetted surface \((S_{wet})\) which is found as the product between \(\pi\), and fuselage’s diameter and length. Therefore, this equation was also modified to introduce the fineness ratio on it, Equation 4-5 shows the modification written in the code for \(S_{wet}\).
4.3 Performance analysis for aerodynamic computations

It is important to use aircraft performance equations in order to generate comparable results with experimental information. Calculations from Tornado need to be computed using the performance model for unpowered descent in order to make the validation and find the total parasite drag coefficient of the sailplane. In fact, sections below explains the methodology followed to find this drag coefficient and the validation outcome.

4.3.1 Parasite drag coefficient estimation

Total parasite drag coefficient depends on the flight condition as can be noted from $C_f$ (Equation 4-2) which is one of the terms required for $C_{D_{body}}$ computation (Equation 4-1). Therefore, it must be defined in order to proceed further with computations. In fact, Tornado needs the definition of an angle of attack and the speed of flight at sea level for its calculations. With this sort of ideas, and having in mind that induced drag polar is already known (see Section 4.2.1), it is possible to estimate the speed for a given value of angle of attack from a performance equation for unpowered flight (Equation 4-11).

If a close look is taken to Equation 4-11, it is possible to realize that it needs the definition of an angle of glide ($\gamma$), which is function of the total drag coefficient (Equation 4-9) and which is not know yet. However, as glide angles are small (commonly lower than 15°) the cosine could be taken as close to unity and airspeed is easily solve. Table 4-1 shows values for parasite drag calculations at several lift coefficients, those were related with angles of attack through the results obtained from the alpha sweep computation.

Table 4-1 shows an increment on the drag coefficient while speed increases. This is be explained from the Eckert’s equation (Equation 4-2), which is used by the code for compute the skin friction coefficient. In fact, it is possible to note that $C_f$ is mostly dependent on the Reynolds number: the higher $Re$, the lower it will be. It is also noticeable that Mach number has an effect on the skin friction drag coefficient, but, due to the low values taken by this parameter, all the parenthesis in Equation 4-2 related to it takes a value close to one.
Table 4-1: Predicted parasite drag for the glider.

| Flight Condition | Outputs | |
|------------------|---------| |
|                  | $C_L$  | $V$ | $\alpha$ | $Re$ | $C_{D_{\text{zero, lift}}}$ | $C_{D_{\text{body}}}$ | $C_{D_0}$ |
| 0.20             | 0.00819 | 0.00408 | 0.01227 |
| 0.25             | 0.00820 | 0.00415 | 0.01235 |
| 0.30             | 0.00820 | 0.00421 | 0.01241 |
| 0.35             | 0.00821 | 0.00426 | 0.01247 |
| 0.40             | 0.00821 | 0.00431 | 0.01242 |
| 0.50             | 0.00823 | 0.00438 | 0.01261 |
| 0.60             | 0.00824 | 0.00445 | 0.01269 |
| 0.80             | 0.00825 | 0.00455 | 0.01281 |
| 1.00             | 0.00827 | 0.00464 | 0.01290 |
| 1.20             | 0.00828 | 0.00471 | 0.01299 |
| 1.40             | 0.00829 | 0.00476 | 0.01306 |

4.3.2 Relations for validation purposes

When trying to make a comparison between Figure 4-1 and results found in this Chapter it is possible to note that the aerodynamic information predicted by Tornado must be operated in some way in order to produce commensurable information. In other words, aerodynamic coefficients must be related with sink rates and airspeed, and the calibrated airspeed ($CAS$) to the true airspeed ($TAS$). As a consequence, a set of performance equations should be used to develop such operations.

Firstly, it is necessary to evaluate the relationship between $CAS$ and $TAS$. This operation is required since measurements on the airplane were made as a function of $CAS$ and Tornado needs the true airspeed for its computations. According to Asselin [3], $CAS$ can be first related with the equivalent airspeed ($EAS$) by adding a compressibility correction (Equation 4-6). However, $V_c$ can be underestimated for this specific case, as suggested for Mach numbers below 0.3 [3], because the sailplane maximum speed during normal operation is 150 km/h [46], which at sea level implies a Mach of about 0.12. Consequently, $CAS$ and $EAS$ could be taken as having equal values. Furthermore, it is necessary to find $TAS$ from
values obtained for \( EAS \) though the relationship showed in Equation 4-7. Nevertheless, experimental information was corrected to sea level conditions [17], resulting in equal values of \( EAS \) and \( TAS \) (because \( \rho_{SL} = \rho \)). The discussion presented above demonstrates that experimental information can be taken as presented as a function of the true airspeed.

\[
EAS = CAS + V_c \quad (4-6)
\]
\[
TAS = EAS \sqrt{\frac{\rho_{SL}}{\rho}} \quad (4-7)
\]

Secondly, aerodynamic coefficients found by the code must be computed in order to generate airspeed (\( V \), which is the same as true airspeed) and rate of sink (\( RS \)). The process of information computation, which is also suggested by Asselin [3], consisted in the sequential solution to Equations 4-8 to 4-11. In fact, sink rates and airspeeds were obtained using predicted couples of aerodynamic coefficients (\( C_L \) and \( C_D \)), aerodynamic efficiency (\( E \) in Equation 4-8), the angle of descent (\( \gamma \) in Equation 4-9), glider’s weight (\( W \)) taken as 262.18 kg (experiments were made with a weight of 578 lb) and the density (\( \rho \)) at its standard sea level value.

\[
E = \frac{C_L}{C_D} \quad (4-8)
\]
\[
\tan \gamma = \frac{1}{E} \quad (4-9)
\]
\[
RS = \sqrt{\frac{2W \cos \gamma}{\rho C_L S}} \sin \gamma \quad (4-10)
\]
\[
V = \sqrt{\frac{2W \cos \gamma}{\rho C_L S}} \quad (4-11)
\]

### 4.4 Validation of mesh and calculation parameters

This section presents a comparison between aerodynamic predictions made by Tornado and experimental measurements obtained by Johnson (see Section 4.1). As can be observed from Figure 4-3, code’s outcomes obtain good results at low airspeeds, while it is also evident its tendency to under-predict the sink rate at higher speeds. However, it is clear that VLM results fit with reasonable accuracy to flight information taken on the actual sailplane. As a consequence, and having in mind that the code produced good predictions with the mesh obtained in Chapter 3, it is now possible to generate the information required for exportation.
to the code where the dynamic stability analysis will be performed: the CEASIOM’s module for Simulation and Dynamic Stability Analysis (SDSA).

![Graph showing the relationship between airspeed and sink rate. The graph includes two curves, labeled "Experimental" and "VLM".]  

**Figure 4-3**: Validation of Tornado predictions with experimental information.
5 Aerodynamic Calculation for Stability Analysis

Present chapter devotes some lines to explain the configuration used to set the aerodynamic computations of the glider. Even though calculations carried on Tornado have a strong background on aerodynamics, it is out of the scope of this work to fully explain them, they are rather addressed by the Melin’s work [25]. Therefore, present section offers an insight on the inputs given to the Software and how they were organized in order to be exported to SDSA.

It is also important to keep in mind that information coming from calculations in Tornado need to be modified in order to provide SDSA with an acceptable file for it. This section also offers a guide about what was done in order to make the aerodynamic information array “understandable” for the application where the stability analysis is to be done.

5.1 Format for aerodynamic data

After a successful determination of a grid for aerodynamic computations, it is now possible to establish the required flight conditions in which the glider will be evaluated. Therefore, it is necessary to generate an arrangement containing all the coefficients estimated by Tornado. According to Rizzi [37], Vallespin et. al. [43] and Zhang [48] the organization of the necessary information to compute the aircraft’s aerodynamic state in the CEASIOM’s module for Simulation and Dynamic Stability Analysis (SDSA) may be done in a matrix as showed in Table 5-1.

According to Zhang [48], the input table to SDSA must contain at least 60 cases because each parameter should take a minimum of two different values. However, it is clearly stated that bigger arrays of results would produce better predictions. The information contained in the table could come from different sources as CFD or experimental measurements. The data exported in this case is that obtained by computations in Tornado.
Table 5-1: Aerodynamic information organization for SDSA [48].

<table>
<thead>
<tr>
<th>$\alpha$</th>
<th>$M$</th>
<th>$\beta$</th>
<th>$Q$</th>
<th>$P$</th>
<th>$R$</th>
<th>$\delta_e$</th>
<th>$\delta_r$</th>
<th>$\delta_a$</th>
<th>$C_L$</th>
<th>$C_D$</th>
<th>$C_m$</th>
<th>$C_Y$</th>
<th>$C_l$</th>
<th>$C_n$</th>
</tr>
</thead>
<tbody>
<tr>
<td>x</td>
<td>x</td>
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<td>x</td>
<td>x</td>
</tr>
</tbody>
</table>

Figures 5-1 and 5-2, which were adapted from SDSA manuals for Coordinates, signs and units definition [41] and the theoretical basics [42] respectively, show the positive orientation of control surfaces deflections, sideslip, rotations and angle of attack. This clarification is important since it helps to analyze the aerodynamic results obtained with CEASIOM.

Figure 5-1: Positive deflection of control surfaces [41].
5.2 Flight state definition

Computations of the PW-5’s aerodynamics were made using the sequential state parameter sweep method, which is available at Tornado. Additionally, parasite drag was added to the drag coefficient, since this type of calculation only predicts induced drag coefficient (see Section 4.2). As can be noted from Table 5-1, the flight state may be defined through three variables related to the airspeed ($\alpha$, $\beta$ and $M$), plus three rotations around the airplane’s axes ($P$, $Q$ and $R$) and the position of the control surfaces ($\delta_a$, $\delta_e$ and $\delta_r$).

The two variables describing how airspeed is reaching the glider (angle of attack and sideslip) were defined from typical values found in airplanes operating under normal conditions ($-6^\circ < \alpha < 10^\circ$ and $-4^\circ < \beta < 4^\circ$ with increments of $2^\circ$ and $1^\circ$, respectively). Three different values of Mach number were defined at sea level within the glider’s normal operating range (41 to 121 knots) according to the airplane’s flight manual [46]. In addition, roll, pitch and yaw rates were also defined to be contained within usual ranges for airplanes with fixed wings ($-15 \frac{\text{deg}}{\text{sec}} < P$, $Q$, $R < 15 \frac{\text{deg}}{\text{sec}}$).

The center of gravity of the airplane was located at the 31% of the MAC, since it is at the mid point of the CG travel according to its type certificates emitted by the Federal Aviation Authority [33] and the Spanish civil aviation authority [31]. Moreover, the deflections of
control surfaces were also introduced according to the FAA type certificate [33]. Table 5-2 summarizes all the parameters defined as inputs in Tornado for the aerodynamic computations.

Table 5-2: Conditions used to define the aerodynamic analysis.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Min. Value</th>
<th>Max. Value</th>
<th>Increment</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\alpha$</td>
<td>$-6^\circ$</td>
<td>$10^\circ$</td>
<td>$2^\circ$</td>
</tr>
<tr>
<td>Mach</td>
<td>0.06</td>
<td>0.18</td>
<td>0.06</td>
</tr>
<tr>
<td>$\beta$</td>
<td>$-4^\circ$</td>
<td>$4^\circ$</td>
<td>$1^\circ$</td>
</tr>
<tr>
<td>$P$</td>
<td>$-15^\circ$</td>
<td>$15^\circ$</td>
<td>$5^\circ$</td>
</tr>
<tr>
<td>$Q$</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>$R$</td>
<td>$-25^\circ$</td>
<td>$15^\circ$</td>
<td>$5^\circ$</td>
</tr>
<tr>
<td>$\delta_e$</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>$\delta_r$</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>$\delta_a$</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

As mentioned before, Tornado was used to make aerodynamic predictions using the mesh obtained in Chapter 3 and inputs showed in Table 5-2. Furthermore, results reached from them were organized in a matrix having the format exposed in Table 5-1. This whole process resulted in the evaluation of 1431 single flight cases, which produced a matrix having the same amount of rows and 15 columns. The time required to obtain such amount of information did not exceed a couple of weeks.

5.3 Setting the input for stability analysis

It was previously stated (see Section 5.1) that results obtained from Tornado computations must be organized in a determined way in order to make them useful for stability analysis in SDSA. However, such an array is not enough since the file may me be modified in order to put it in an acceptable format for the Software. Therefore, for this specific case aerodynamic information have the order shown in Table 5-1 and is contained in a proper file for SDSA.
Solutions for aerodynamic computations were made with the sequential state parameter sweep method in Tornado. Consequently, predictions obtained by this code were arranged in a big table containing the entire solution for the whole flight state. The file in which the array was stored had a plain text format (.txt), which must be modified in order to be taken by SDSA as an adequate input.

The translation from a plain text format to .xml, which is a proper extension for SDSA, was done with a Matlab code written in a previous work at the Division of Aerodynamics at KTH. There was also necessary to provide such a code with those values taken by the parameters defining the flight state (Table 5-2 as well as the CG location (as the reference point), wingspan, wing area, mean aerodynamic chord (the las two were calculated by Tornado) and power status, which was set to zero since the glider doe not have power unit.
6 Aerodynamic Predictions and Analysis

The stability analysis of the PW-5 Smyk was done on the CEASIOM’s module capable of calculate airplane’s responses to control surfaces and disturbances, fixed-stick stability modes of motion and low-speed limitations based on aerodynamic derivatives, moments of inertia and center of gravity location: Simulation and Dynamic Stability Analysis (SDSA) [48]. This Software accepts various input formats and can be used together within CEASIOM or as stand-alone application [10, 11]. Pester [30] stated that it “is a very complex tool, it gives the user the possibility to cover many fields of the aircraft design process in a comfortable way”, while evaluating an Airbus A-320.

SDSA is a widely used tool to evaluate the dynamic behavior of airplanes based on a sort of aerodynamic inputs that could come from different sources as CFD software results or wind tunnel measurements, among others. For instance, Seeckt [40] used it while coupling CEASIOM with the Software PreSTo (Preliminary Sizing Tool, a project from the Hamburg University of Applied Sciences) to re-design the regional transport aircraft ATR-42. Furthermore, researchers from the Warsaw University of Technology [10, 11] analyzed the accuracy of SDSA with results from aerodynamic computations of the PW-6 glider and experimental measurements taken on the Ranger 2000. They found SDSA to be a great tool for stability analysis, which is highly dependent on the quality of the aerodynamic data used as input.

Von Kaenel et. al. [44] used SDSA to predict and evaluate the dynamic modes of a Transonic Cruiser (TCR) design as phugoid, short-period and Dutch roll. Moreover, researchers from the Royal Institute of Technology, Swedish Defense Research Institute, Warsaw University of Technology, CFS Engineering and University of Bristol [38] re-designed the TCR in order to accomplish a request from SAAB which required a reduction in the longitudinal control deflection needed to bring the aircraft to the trim point. Such variation was made based on studies carried on CEASIOM, where flying qualities were studied using SDSA from a set of results obtained with CFD tools. They found good agreement between computational predictions and wind tunnel measurements when comparing results for trim conditions (\(\alpha\) and \(\delta_c\)). Additionally they analyzed short period and phugoid dynamic modes of the aircraft and made a couple of simulations of the airplane into SDSA.


6.1 Static stability analysis

This section presents and analyses the predictions made by CEASIOM concerning to the static aerodynamic behavior of the glider. The study is made based on the dimensionless representation of forces and moments: aerodynamic coefficients. Results presented below intend to show the behavior of the airplane when exposed to different flying conditions, which implies evaluations at different angles of attack, sideslip, control surfaces deflections and Mach numbers.

6.1.1 Forces acting on the airplane

Figures 6-1 and 6-2 show the results obtained by CEASIOM when predicting the coefficients of the most important forces acting on the airplane: lift and drag, respectively. As expected, lift force coefficient has a positive slope and its values does not vary as a function of the Mach number. The independence to the Mach number can be explained because the speeds ranged for the sailplane are very low and compressibility effects are not expected to appear in the glider’s flight [2]. Even though there are not compressibility effects on the airplane, there is possible to note that parasite drag force coefficient does change as a function of the Mach number, while induced drag is kept constant (Figure 6-2). This is explained from the prediction of the friction coefficient made by Tornado, which is an inverse function of the Mach and Reynolds numbers (See Section 4.2). As can be noted from Equation 4-1 the parasite drag coefficient will decrease if the friction coefficient does.

<table>
<thead>
<tr>
<th>Derivative</th>
<th>Predicted Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{L_{\alpha}}$</td>
<td>5.730</td>
</tr>
<tr>
<td>$C_{D_{\alpha}}$</td>
<td>0.191</td>
</tr>
<tr>
<td>$C_{m_{\alpha}}$</td>
<td>-2.149</td>
</tr>
</tbody>
</table>

Table 6-1: Aerodynamic coefficients derivative with respect to the angle of attack.

From Figures 6-1, 6-2 and 6-4 it is possible to obtain the glider’s aerodynamic coefficients derivative with respect to the angle of attack predicted by CEASIOM. Those variations, which are important values when the airplane is to be analyzed aerodynamically, are shown in Table 6-1. It is noticeable that Table 6-1 shows only one value for the drag coefficient slope, despite the fact that Figure 6-2 has several lines. The reason for that behavior
is that all the plots contained in the figure are parallel due to the invariability in induced drag coefficient, and as a consequence all of them have the same derivative with respect to $\alpha$. 

**Figure 6-1**: Lift coefficient vs. angle of attack

**Figure 6-2**: Drag coefficient vs. angle of attack.
Figure 6-3 shows $C_{Y_β}$ as a function of the Mach number. As expected, the side force coefficient is independent of the Mach number due to the low speeds at which the sailplane flies. Moreover, it is preferred to plot $\frac{\partial C_{Y_β}}{\partial \beta}$ (or $C_{Y_β}$, which is the same) as a function of the Mach number rather than as a sideslip function since the sailplane is symmetric and side force acting on it will be zero when no sideslip is presented.

![Figure 6-3: Side force derivative with respect to the sideslip vs. Mach number](image)

### 6.1.2 Moments acting on the airplane

This section presents moment coefficients or their derivatives obtained as a result from the aerodynamic computations made on the PW-5 Smyk, and their respective discussion. Results contained here account for pitching, rolling and yawing moments, which are shown in Figures 6-4, 6-5 and 6-6, respectively. These graphics offer an idea on how the glider will respond, in its longitudinal, directional and lateral axis, immediately after an aerodynamic perturbation is encountered in operation.

Aerodynamic results contained in Figures 6-4 to 6-6 provide light about the static stability tendencies the glider will likely have during flight. From them it is noticeable that the sailplane is stable, because it accomplishes the the static criteria: pitching moment coefficient when $\alpha = 0^\circ$ is positive ($C_{m_0} > 0$), pitching moment coefficient derivative with respect...
to $\alpha$ is negative ($C_{m\alpha} < 0$), rolling moment coefficient and yawing moment coefficient derivatives with respect to $\beta$ are negative and positive, respectively ($C_{l\beta} < 0$ and $C_{n\beta} > 0$) [14, 29].

![Figure 6-4: Pitch moment coefficient vs. angle of attack](image)

![Figure 6-5: Roll moment coefficient derivative with respect to the sideslip vs. angle of attack](image)
Once again, it is important to clarify that directional and lateral analyses are made based on moment derivatives rather than on coefficients, this occurs due to the symmetric geometry found in the airplane. Therefore, it is expected that rolling and yawing moments are absent when no sideslip occurs during flight.

![Diagram](image-url)

**Figure 6-6**: Yaw moment coefficient derivative with respect to the sideslip vs. Mach number

### 6.2 Dynamic stability analysis

This section offers an analysis on how the airplane is expected to dynamically behave under different circumstances on flight. The study presented here shows the variation of main aerodynamic moments and forces acting on the airplane as a function of different operating conditions, among them control surfaces deflections and rotations. All the graphs presented in this section appeared as result from the aerodynamic computations made with CEASIOM.

#### 6.2.1 Longitudinal stability derivatives

Figures presented here give a computational estimation on how a steady pitch rate affects lift force and pitch moment on the sailplane. For instance, Figure 6-7 shows the glider’s lift derivative with respect to the pitch rate as a function of the angle of attack. In other words,
it gives an idea of the effect caused by a steady pitch rate on the lift coefficient. This effect is mainly caused by the change in $\alpha$ generated by the pitching motion.

![Graph of $C_{L_Q}$ vs. $\alpha$](image)

**Figure 6-7**: Lift coefficient derivative with respect to the pitch rate vs. angle of attack

From Figure 6-7 is possible to note that the PW-5’s $C_{L_Q}$ is negative, which makes sense due to the decrement in angle of attack caused by a positive pitching motion. Moreover, 6-7 shows a stronger decrease when the airplane is flying at higher angles of attack. This very same tendency is also noticeable, for angles of attack above 2°, with Mach number: the higher the speed of flight, the higher will the decrement in the lift force suffered by the aircraft be. However, for $\alpha$ below 2° such behavior is inverted and lower Mach numbers produce less effect on the lift coefficient.

On the other hand, the pitching moment coefficient derivative with respect to the pitch rate, which is also known as the damping in pitch derivative, shows a similar tendency to that reached by the lift derivative with respect to the pitch rate. As can be observed in Figure 6-8, for angles of attack up to 2° the strength of the decrease in pitching moment coefficient generated by the pitch rate will be higher while increasing angles of attack and decreasing Mach number. Nevertheless, tendency with the Mach number is inverted for $\alpha$’s greater than approximately 2°, just as found with the previous lift derivative analyzed.

To sum up, both coefficients, lift force and pitching moment, are diminished by the presence
of a positive pitch rate. In other words, they vary inversely with $Q$, which can be concluded from the negative sign reached by both derivatives within a typical operative range. In addition, a negative derivative, in the case of the pitching motion, implies a damping response from the airplane, since the higher the pitch rate, the lower the pitching moment will be.

![Figure 6-8](image)

**Figure 6-8**: Pitch moment coefficient derivative with respect to the pitch rate vs. angle of attack

### 6.2.2 Lateral-directional stability derivatives

The damping in roll derivative, as the derivative plotted in Figure 6-9 is also known, is an estimation of the rolling moment caused by a roll rate on the airplane. It is possible to note that CEASIOM predicts a negative value of the derivative for the analyzed range. Such result implies that a restoring aerodynamic moment will be expected to appear for the PW-5 sailplane when making a rotation about its longitudinal axis. Consequently, an augmentation in the roll rate will generate a decrease in the rolling moment, which is a damping response. Moreover, it is possible to note that the damping effect is diminished by the angle of attack and that Mach number is only important at high values of $\alpha$.

On the other hand, the estimation of the rolling moment experienced by the glider when its flight is under a yaw rate, which is also named as the rolling moment coefficient derivative with respect to the yaw rate, is showed in Figure 6-10. Aerodynamic predictions estimate
that the moment about the longitudinal axis, caused by the difference in speed between both sides of the wing when a yaw rate appears is damped. This conclusion is supported in the fact that $C_{l_R}$ is negative for angles of attack above $-6^\circ$, which is the most common condition found during operation. Moreover, 6-10 shows that the effect is augmented by both, increases in Mach number and angle of attack. Even though wings commonly tend to produce a destabilizing effect [29], the negative value of $\frac{\partial C_l}{\partial R}$, which implies a damping effect, presents the vertical tail in the sailplane as the one accomplishing the damping task.

![Figure 6-9: Roll moment coefficient derivative with respect to the roll rate vs. angle of attack](image)

A derivative that may be taken into account in an aerodynamic analysis is the yaw moment coefficient derivative with respect to the roll rate, which is an estimation on how a rolling rate influences the appearance of a yaw moment on an airplane. From Figure 6-11 is possible to note that $\frac{\partial C_n}{\partial P}$ is negative for angles of attack greater than approximately $-3^\circ$ and its magnitude increases as long as the angle of attack is augmented from this $\alpha$. Consequently, it is expected that the glider will experience a tendency to move to the left wing when it is under a positive roll rate (left wing up). This behavior makes sense because, as long as the left wing is raising, it will experience a higher lift force. As a direct consequence of it such part of the wing will undergo a greater drag than that generated in the right part of the wing.
6.2 Dynamic stability analysis

Figure 6-10: Roll moment coefficient derivative with respect to the yaw rate vs. angle of attack

Figure 6-11: Yaw moment coefficient derivative with respect to the roll rate vs. angle of attack
One important lateral-directional derivative is the damping in yaw derivative, which is an estimation of the yaw moment generated on an airplane by its own yaw rate. As can be seen in Figure 6-12 it is expected to be negative throughout the analyzed range of angles of attack. In fact, if $\frac{\partial C_n}{\partial P}$ is less than zero, it implies that increments in aircraft’s yaw rate will likely to produce a decrease in the yaw moment, which weakens the moment rate of the airplane. Furthermore, it is noticeable that the damping effect is expected to increase if the sailplane angle of attack is augmented, which is the same tendency achieved with the Mach number.

As can be noted from present section, according to the aerodynamic computations made on CEASIOM’s modules, the most important dynamic stability derivatives ($C_{mQ}$, $C_{lP}$, and $C_{nR}$) of the PW-5 Smyk sailplane produce damping effect when it is under a steady movement around its axes. This offers an idea of a suitable airplane, from the aerodynamic point of view, to accomplish its mission, because it is expected that the glider tends to have a soft flight, creating corrections to perturbations only by its own aerodynamic.

![Figure 6-12: Yaw moment coefficient derivative with respect to the yaw rate vs. angle of attack](image)

Figure 6-12: Yaw moment coefficient derivative with respect to the yaw rate vs. angle of attack
6.3 Airplane’s response to control surface deflections

One important aerodynamic result that can be evaluated through SDSA is the effect caused on the aerodynamic moments by the control surfaces deflection. Therefore, this section presents and analyzes predictions calculated with CEASIOM about how good are the control surfaces to manage moments on the airplane according to its geometry and position on the sailplane.

6.3.1 Longitudinal control

The longitudinal control surface, named as elevator, influences the lift force and, mainly, the pitch moment exerted on the airplane. Figure 6-13 shows how the lift coefficient derivative with respect to the elevator deflection varies as a function of the angle of attack. As expected, $\left(\frac{\partial C_L}{\partial \delta_e}\right)$ is positive throughout the analyzed range of $\alpha$’s. This is explained because a positive elevator deflection (downward) produces an increment in the lift force coefficient of the horizontal tail, which generates augmentation in the glider’s total lift coefficient. In addition, it is possible to note that the change in lift force due to the longitudinal control deflection is almost independent from the Mach number for the operating speeds of the glider.

![Figure 6-13: Lift coefficient derivative with respect to the elevator deflection vs. angle of attack](image)

Figure 6-13: Lift coefficient derivative with respect to the elevator deflection vs. angle of attack
The pitch moment coefficient variation with respect to the elevator deflection as a function of the Mach number and angle of attack is showed in Figure 6-14. As can be noted from it, a positive deflection of the elevator produces a decrease in the pitch moment on the glider, due to its value lower than zero. The reason for this behavior is that a downward deflection on the longitudinal control surface (which is positive) generates an increase in the lift coefficient of the horizontal tail, which causes a negative pitching moment.

In order to make an analysis on how good elevators on the glider work it is necessary to compare Figures 6-4 and 6-14. In other words, it is important to evaluate if those control surfaces are capable of bring the sailplane to trim in any condition. According to 6-4 the most extreme conditions for pitching moment would be at the higher and lower angles of attack studied: $-6^\circ$ and $10^\circ$. In both cases absolute values of pitching moment coefficient were about 0.2, positive for the negative $\alpha$ and negative for $10^\circ$.

![Figure 6-14: Pitch moment coefficient derivative with respect to the elevator deflection vs. angle of attack](image)

It is now required to check the value of $C_{m_{\delta_e}}$ at the angles of attack mentioned above, which, according to Figure 6-14, are $-1.584 \text{ [rad}^{-1}]$ for $-6^\circ$ and $-1.6 \text{ [rad}^{-1}]$ for $10^\circ$. With these derivatives in mind and the pitch moment coefficient found before it is possible now to determine that a PW-5 flying at those $\alpha$’s would require approximately an elevator deflection of $7.2^\circ$ (positive for $-6^\circ$ and negative for $10^\circ$) to bring it to trim. If it is kept in
mind that the elevator movement range is between $-25^\circ$ and $15^\circ$ (see Section 5.2) is possible to conclude that the elevator works perfectly in the analyzed range and even further than that. Additional possible elevator deflection could be used to make the sailplane controllable under more critical atmospheric conditions or maneuvers and account for a safety margin.

### 6.3.2 Lateral-directional control

The roll moment coefficient derivative with respect to the rudder deflection, which is an estimation of the roll moment induced by the rudder deflection, predicted for the PW-5 Smyk is shown in Figure 6-15. As can be seen, for angles of attack below $8^\circ$ a positive deflection in the directional control surface causes a positive roll moment on the airplane. The effect can be explained because the center of pressure in the vertical tail has a distance over the aircraft’s center of gravity, which generates a rolling moment when lift is created by this part of the empennage. However, the induced rolling moment due to a deflection of the control surface on the vertical tail is very small for $\alpha$’s lower than that mentioned before. Furthermore, it is possible to note that the effect is dramatically increased (in magnitude) for angles greater than $8^\circ$.

![Figure 6-15: Roll moment coefficient derivative with respect to the rudder deflection vs. angle of attack](image)

The roll moment coefficient derivative with respect to the ailerons deflection, which is of
critical importance for evaluation of the lateral controllability of the airplane, is shown in Figure 6-16. As expected, from deflection and moment definition (see Section 5.1) its value is lower than zero through the entire range of angles of attack analyzed. However, its magnitude starts to decrease at higher $\alpha$’s, which implies that the control surface will loose its efficiency as long as the angle of attack is increased. Moreover, it is possible to note that the effect of ailerons over the airplane moment does not depend upon the Mach number at which the glider is flying.

![Figure 6-16: Roll moment coefficient derivative with respect to the ailerons deflection vs. angle of attack](image)

When Figures 6-5 and 6-16 are compared, it is possible to determine what would be the required deflection from ailerons to correct an atmospheric disturbance causing roll. From Figure 6-5 it is noticeable that the higher magnitude of $\frac{\partial C_l}{\partial \beta}$ is reached when the airplane is flying at an angle of attack of $-6^\circ$, which is $-0.023$ rad$^{-1}$. Consequently, a disturbance causing a sideslip of $-4^\circ$ (which is the highest magnitude of $\beta$ evaluated) will generate a rolling moment coefficient of approximately $0.0016$. Furthermore, if $\frac{\partial C_l}{\partial \delta_a}$ is taken from Figure 6-16, at the same angle of attack, it gives a value of $-0.331$ rad$^{-1}$. As a consequence, the airplane would require a deflection lower than $1^\circ$ to correct the atmospheric condition. In fact, this analysis demonstrates that the sideslip does not cause a great effect over the lateral motion of the sailplane and that ailerons are well designed in order to maintain the aircraft controllable. Therefore, it is possible to note that ailerons in the sailplane are suitable to
6.3 Airplane’s response to control surface deflections

control it during pronounced maneuvers, because its aileron deflections ranges from $-25^\circ$ to $15^\circ$ (see Table 5-2).

It is also convenient to make the controllability analysis to the rudder deflection. According to Figure 6-6, the value of $\frac{\partial C_n}{\partial \beta}$ is approximately 0.052 \(\text{[rad}^{-1}\text{]}\) throughout the range of angles of attack. This implies a yaw moment coefficient of 0.0036 when a sideslip of $4^\circ$ appears. In addition, when taking a look at the Figure 6-17 it noticeable that the lower efficiency of the rudder is achieved between $-6^\circ$ and $8^\circ$, where it has an approximate constant value of $-0.03$. In consequence, the sailplane would require a rudder deflection of $6.9^\circ$ in order to counteract the effect of the sideslip. When reviewing Section 5.2 it is possible to note that the airplane’s type certificate establishes a a maximum deflection for the directional control surface of $25^\circ$ in both directions, which implies that the sailplane could easily be balanced in an atmospheric condition causing a sideslip of $4^\circ$. As the discussion developed for previous control surfaces, the rudder could impulse to the airplane to withstand worse flight conditions than those evaluated here and additionally account for a safety factor.

![Figure 6-17: Yaw moment coefficient derivative with respect to the rudder deflection vs. angle of attack](image)

As noted from present section, none of the control derivatives is affected by changes in the Mach number. This phenomena is expected and can be explained due to the low speeds achieved by the sailplane during its operation. Therefore, there will not be appearance of
compressibility effects over the aerodynamic surfaces of the airplane.

6.4 Stability summary

Table 6-2 offers a summary of the aerodynamic coefficients predicted by CEASIOM on a geometry similar to that found in a PW-5 Smyk. In fact, its information is more a short description of the calculated coefficients implication on the aerodynamic behavior of the sailplane. 6-2 also contains the criteria used to make the concluding comments about a specific parameter. The table presented below briefly resumes the outcomes of the aerodynamic computations developed over the sailplane.

As found during the analysis carried in past section it was possible to note that the sailplane requires low control surfaces deflection to counteract atmospheric perturbations or, in general, sudden changes in the vector velocity direction. One possible reason for this behavior is that the center of gravity was located at the mid point in the CG travel (see Chapter 5) and not in one of its extremes, which would require higher deflections in order to make the airplane controllable. Furthermore, not extreme flight conditions were analyzed, which will probably require a higher effort from the airplane’s control to make it maneuverable.

<table>
<thead>
<tr>
<th>Concluding remark</th>
<th>Criteria</th>
</tr>
</thead>
<tbody>
<tr>
<td>The sailplane is statically stable in all directions</td>
<td>Predicted $C_{ma}$ and $C_{1\beta}$ were negative while $C_{mo}$ and $C_{n\beta}$ were positive</td>
</tr>
<tr>
<td>The sailplane damps its rotations by itself in all directions</td>
<td>The damping derivatives $(C_{mQ}, C_{1\nu}, C_{nR})$ were all negative</td>
</tr>
<tr>
<td>The sailplane’s control surfaces are capable of counteract atmospheric conditions</td>
<td>The deflections required according to aerodynamic computations were contained in the sailplane’s operating range</td>
</tr>
</tbody>
</table>
7 Conclusions and Perspectives

The study presented here was focused in two main areas: the utilization of a Computerized Environment for Aircraft Synthesis and Integrated Optimization Methods (CEASIOM), which was developed as a part of the SimSAC project, funded by the European Commission 6th Framework Programme, and the aerodynamic analysis of the PW-5 Smyk sailplane, which is a single seat airplane designed by the Warsaw University of Technology. Therefore, conclusions and perspectives obtained from this work are divided into these two main fields.

7.1 Software utilization

The Computerized Environment for Aircraft Synthesis and Integrated Optimization Methods and its modules comprises a powerful tool to provide support in the early stages of an airplane design. As found in present document, CEASIOM only requires a good description of the geometry an aircraft will likely have during operation and information about the atmospheric conditions in which it would fly. Moreover, the Environment is not only good for design purposes, as showed in precedent chapters, it is also very reliable to analyze built models in order to modify or have a better understanding of their aerodynamics (only aerodynamics are mentioned because it was the focus of this project, however it also has modules for other different aspects related in an aircraft design).

The Vortex Lattice Method implementation proved to be very reliable when aerodynamic predictions are to be performed over a specific model. Despite the fact that it is considered a low reliability computational tool, its results are in close agreement to experimental measurements. This makes VLM an efficient support for aerodynamic modifications and improvements that may be applied over a defined airplane. Furthermore, the analysis through VLM offers a series of aerodynamic coefficients that can be used to predict the airplane’s stability and performance when it is in flight.

What is of critical importance in a code using Vortex Lattice Method for aerodynamic computations is the mesh definition. Even though the code proved to be very effective for the case analyzed during the present work, it was necessary to devote some time in choosing an adequate discretization of the domain over the sailplane. An effective selection of the type
of grid and the number of divisions on it should be done carefully, which should include a convergence analysis, in order to capture the main aerodynamic features properly. Otherwise, as supported by the literature review, predictions coming from the code would not be as accurate as they were in the present work.

As demonstrated in this document, CEASIOM does not even require an exact description of the aircraft. In the absence of exact plans of the sailplane, some geometrical approximations were made during its definition as an input for Tornado. However, it was found that the code predicted a sailplane flight polar very similar to that found during experimental flights. In fact, this reflects the robustness of the code, because it reached good accuracy even though calculation had predetermined numerical errors and in geometrical definition.

Another important aspect found in the Environment is the low time it requires to achieve a series of solutions. Indeed, the fulfillment of the matrix of results would have need more than a month if more advanced CFD tools were used. Instead, as noted in the document, CEASIOM produced reasonable results in just a couple of weeks. Consequently, the accuracy showed by it and its low time-consuming solution makes it suitable for industrial purposes as design or model development. Furthermore, it is also applicable to academy, because it could be used as supporting material for lectures and academic research.

The way the code was developed let the user customize it according to the requirement. The theory used to write some aspects of the code is easily accessible and the user can modify the code as long as the task needs it. For instance, for the work presented in this document some changes were made in order to improve the definition of the fuselage for parasite drag prediction. The modifications resulted in a good prediction of the sailplane flight polar.

Due to all the advantages exposed here, CEASIOM is widely used for research purposes. Indeed, lot of publications can be found in scientific literature, carried mainly by academic institutions based in Europe. In addition, vast majority of them agrees in that they reached good results when making analysis with this computational Environment.

On the other hand, the Software would improve its quality if a more user-friendly code for transferring of outputs between its stand-alone modules were developed. In fact, the translation of Tornado results to become in SDSA inputs is quite complicated and time-consuming. Furthermore, when analyzing the methods used to predict drag, it would also be pertinent to suggest an improvement in the drag prediction made by the code. However, if the state of the art in computational drag prediction is reviewed, it is possible to conclude that failures in calculation are rather a generalized problem of computational fluid mechanics than a specific issue of CEASIOM.
An aspect that should be introduced in Tornado is the estimation of the fuselage effect on the analysis. Despite CEASIOM Software includes it in all the analyses, Tornado stand-alone only takes it into account for parasite drag coefficient estimation. Even though fuselage's contribution to the generation of aerodynamic forces and moments over airplanes is often negligible (not true for drag purposes), it does produce an effect over them. Consequently, including this part of an airplane into the calculation would improve overall accuracy of prediction. There are several sources which could be used in order to introduce fuselage in the mathematical model as, for example, the USAF Stability And Control DATCOM [9], which is already implemented in CEASIOM.

From the Software utilization it was possible to infer that a tool like this could help improve aerodynamic fidelity in early stages of an airplane design. Indeed, predictions coming from CEASIOM could be obtained from the earliest drafts of the airplane’s geometry and, as a consequence, it could permit to take accurate engineering decisions about a specific configuration. As demonstrated, the computation of a sailplane design on the Environment helps to achieve good aerodynamic conditions in the final model. Therefore, this sustainable-energy kind of transportation could achieve a higher penetration worldwide because tools like this are helpful in improving its performance.

### 7.2 Aerodynamic analysis

The aerodynamic analysis of the PW-5 sailplane proved it to be statically stable. This kind of stability implies that the immediate response, coming from the airplane, to an external change in the direction of the velocity vector will be bringing itself to the position it was before the perturbation. This conclusion came when comparing aerodynamic coefficients with the stability criteria: all of them were fulfilled by the airplane. Despite the fact that the aircraft was found to be statically stable, it does not necessarily mean that it will have a proper dynamic behavior. Therefore, a dynamic analysis was also performed.

Results coming from SDSA also permit a dynamic evaluation of the sailplane during flight. From them, it was found that all the rotations about the main axes of the sailplane are self-damped. In fact, the criteria used was that a moment about an specific axes was decreased by the rotation rate. In other words, it was find that all the moment coefficient derivatives with respect to the moment rate were negative, which implies a reduction in the moment while the rate increased.

CEASIOM is also capable to predict how good a control surface is designed. With the plots
produced by their modules is possible to predict if the moment caused by a specific control surface deflection could counteract the moment caused by a sudden change in the direction of the velocity vector. When the PW-5 Smyk was analyzed in this sense it was noticeable that all the control surfaces make the airplane controllable within the operational range. This conclusion was obtained from a numerical estimation of the required deflections to control the glider during the most critical circumstances simulated. Nevertheless, the conditions analyzed in the present document were average conditions that the glider would likely have during flight. In order to produce a better conclusion about its controllability, it would be necessary to analyze extreme conditions that could appear during the aircraft’s operation.

As expected previously, aerodynamic computations gave results almost independent from the Mach number. The main reason for this behavior is that speeds at which the airplane operates are much lower than those at which compressibility effects are expected to appear. However, there is possible to note that moment derivatives with respect to the moment rate are susceptible to changes as a function of the Mach number. This phenomena is understandable from the point of view that rotating rates will vary depending upon the speed of flight of the sailplane.

Despite the fact that geometry, mesh and calculation parameters introduced in Tornado were successfully validated with experimental measurements taken during the glider’s flights, final results were not compared with any kind of measurements (not real flight, nor wind tunnel tests). Indeed, such an information is not available for general public. Therefore, it is recommended that aerodynamic predictions made in the present document are validated with more detailed experimental information.

As can be noted from the whole work presented in this document, it is possible to note that transportation in sailplanes is feasible because these vehicles are designed to be perfectly stable and controllable. The utilization of gliders in order to move from one place to another helps to achieve sustainability in aviation since its operation is more environmentally friendly than that in other types of flying vehicles. In fact, gliders do not produce carbon dioxide or any other kind of harmful gases for the environment, because they do not burn fossil fuels to fly. Instead, they use potential energy and warm ascendant air streams to developed their operation through the skies. Furthermore, sometimes they are equipped with electric engines in order to improve their performance.
Bibliography


[41] SimSAC. *Coordinates, signs and units definition*, 2008.


