A NEW VERTICAL TAILPLANE DESIGN
PROCEDURE THROUGH CFD

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Nobody will fly for a thousand years!
— Wilbur Wright, 1901, in a fit of despair

Dedicated to all those who support (and stand) me.
Dedicato a tutti quelli che mi supportano (e che mi sopportano).
ABSTRACT

The objective of this work is to define a new preliminary design procedure for the airplane’s vertical stabilizer, through the Computational Fluid Dynamics. A reliable tailplane design needs an accurate determination of the stability derivatives, usually calculated with semi-empirical methods as USAF DATCOM and ESDU, which derive from NACA reports of the first half of the ‘900, based on obsolete geometries, and give quite different results for certain configurations, e.g. in the case of horizontal stabilizer mounted in fuselage. In this master thesis these methods are compared by a user-defined MATLAB program and a new procedure is built on a series of CFD analyses, simulated by the commercial software Star-CCM+. Test cases, based on the cited NACA reports, are performed to verify the compliance of CFD results with available experimental data before developing the new procedure. A large number of simulations were run in the SCoPE grid infrastructure of the University of Naples ‘Federico II’, that gave the possibility to simulate complex 3D geometries in a small amount of time. The new procedure, developed to be accurate for modern props airliners, gives promising results, close to that provided by other established approaches.

SOMMARIO

L’obiettivo di questo lavoro di tesi è la definizione di una nuova procedura di avanprogetto dello stabilizzatore verticale di un generico aeroplano, attraverso la fluidodinamica computazionale. Un progetto affidabile di tale componente richiede la determinazione accurata delle derivate di stabilità, solitamente calcolate con i metodi semi-empirici USAF DATCOM ed ESDU, che derivano da report NACA della prima metà del ‘900, basati su geometrie obsolete, e restituiscono risultati alquanto differenti per certe configurazioni, come nel caso di stabilizzatore orizzontale posizionato in fusoliera. Nel presente lavoro queste metodologie sono messe a confronto in un programma MATLAB scritto ad hoc e la nuova procedura è costruita su una serie di analisi CFD eseguite con il software commerciale Star-CCM+. Sono effettuati dei casi test, basati sugli stessi report NACA di cui sopra, per verificare la rispondenza dei risultati CFD con i dati sperimentali, prima di sviluppare la nuova metodologia. Un gran numero di simulazioni è stato elaborato nella griglia di calcolo SCoPE dell’Università di Napoli ‘Federico II’, con lo scopo di analizzare complesse geometrie 3D in un tempo relativamente breve. La nuova procedura, sviluppata per essere accurata su moderni velivoli propulsori ad elica, dà dei risultati promettenti, molto vicini a quelli ottenuti con i metodi semi-empirici.
Aeroplanes are not designed by science, but by art [...] it stands on scientific foundation, but there is a big gap between the scientific research and the engineering product which has to be bridged by the art of the engineer.

— a British engineer to the Royal Aeronautical Society, 1922

E, no: le righe [di questa tesi] non sono troppo corte.


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I wish to thank Prof. Fabrizio Nicolosi for the opportunity of this thesis, Eng. Pierluigi Della Vecchia for the many hours dedicated to this work, Prof. Agostino De Marco and CD-adapco for the academic license of Star-CCM+ and the possibility to learn a great tool.

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I cannot forget my friends, who ultimately tolerated my absence and always make me happy. Special thanks to the new friends of the Model Club Airone, for funny and some adrenaline moments.

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¹ Members of \guIt (Gruppo Italiano Utilizzatori di \TeX e \LaTeX)
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ACRONYMS

ORGANIZATIONS

dias Dipartimento di Ingegneria Aerospaziale
easa European Aviation Safety Agency
esdu Engineering Science Data Unit
far Federal Aviation Regulations
naca National Advisory Committee for Aeronautics
nasa National Aeronautics and Space Administration
scope Sistema Cooperativo per Elaborazioni Scientifiche Multidisciplinari
usaf datcom United States Air Force Data Compendium

COMMON ABBREVIATIONS

cad Computer Aided Design
catia Computer Aided Three-dimensional Interactive Application (a Dassault proprietary Product Lifetime Management system)
cfd Computational Fluid Dynamics
cli Command Line Interface
cpu Central Processing Unit
fem Finite Element Method
lfc Local File Catalogue
matlab Matrix Laboratory (a MathWorks proprietary software)
oei One Engine Inoperative
rans Reynolds Averaged Navier-Stokes equations
sftp Secure File Transfer Protocol
ssh Secure Shell
SHORT LIST OF SYMBOLS

FREQUENTLY USED SYMBOLS

A  wing aspect ratio
A_h  horizontal tailplane aspect ratio
A_v  vertical tailplane aspect ratio
B  compressibility parameter
C_L  lift force coefficient
C_{L,a}  lift curve slope
C_{L,a,v}  vertical tailplane lift curve slope
C_L  rolling moment coefficient
C_{L,\beta}  rolling moment due to sideslip derivative
C_N  yawing moment coefficient
C_{N,\beta}  yawing moment due to sideslip derivative
C_Y  sideforce coefficient
C_{Y,\beta}  sideforce due to sideslip derivative
F  aerodynamic force
M  Mach number
Re  Reynolds number
S  wing planform area
S_c  control surface area
S_h  horizontal tailplane area
S_v  vertical tailplane area
V  velocity
\bar{V}_h  horizontal tailplane volume coefficient

\textit{stl}  Stereolithography (file format)
\( \bar{V}_h \)  vertical tailplane volume coefficient

2r  fuselage diameter at vertical tail aerodynamic center
b  wing span
b_h  horizontal tailplane span
b_v  vertical tailplane span
b_v1  vertical tailplane span extended on fuselage centerline
c  wing chord
\( \bar{c}_c \)  chord of control surface
c_{mac}  mean aerodynamic chord
\( c_h \)  hinge moment
\( c_v \)  vertical tailplane chord
d_f  fuselage diameter
\( l_h \)  longitudinal distance between airplane’s c.g. and horizontal tailplane aerodynamic center
\( l_v \)  longitudinal distance between airplane’s c.g. and vertical tailplane aerodynamic center
\( r_f \)  fuselage half diameter or dis
\( z_w \)  wing position

\( \alpha \)  angle of attack
\( \beta \)  angle of sideslip
\( \lambda \)  taper ratio
\( \vartheta \)  fuselage upsweep angle
\( \rho \)  air density

\( \Delta \)  difference
\( \Lambda \)  angle of sweep
SUBSCRIPTS

LE     leading edge
TE     trailing edge
b      body (synonym of fuselage)
f      fuselage
h      horizontal tailplane
v      vertical tailplane
w      wing
INTRODUCTION

1.1 PHENOMENOLOGY

Tail surfaces are little wings that keep the airplane in its intended flight path. A typical arrangement is sketched in Figure 3a. Tail surfaces perform three functions:

1. they provide a state of equilibrium (trim) in each flight condition;
2. they provide static and dynamic stability;
3. they enable aircraft control.

Tail surfaces sizing and shaping is almost exclusively determined by stability and control considerations. Normally they operate at only a fraction of their lift capability since, for the reasons stated above, they must be far away from stall condition [29, 35]. Usually, in preliminary design, only volume coefficients are estimated [30]. Tail volume coefficients are non-dimensional numbers defined as follows

\[
\bar{V}_h = \frac{l_h S_h}{c_{mac} S},
\]

\[
\bar{V}_v = \frac{l_v S_v}{b S}.
\]

Aircraft having the same volume coefficients tend to have similar static stability characteristics [17]. Usually these coefficients are assigned by simply looking up a table or a chart for aircraft with similar configurations. Observations can be made by looking at historical or other general trends, as reported in Figure 1. An example of these coefficients for turboprop aircrafts is shown in Table 1. A plot of these data vs the wing surface is in Figure 2. General aviation airplanes have smaller coefficients, half or less than the values shown, as reported by Hall [17].

This thesis deals with vertical tail planes, that provide directional stability and control. A vertical plane is usually made up of two parts: a fixed wing, called fin, and a plain flap, the rudder (Figure 3). The fin provides directional stability while the rudder is the directional control surface.

There is a large variety of tail shapes, often denoted by the letters whose shapes they resemble in front view [21], for instance T, V, H, Y (see Figure 3b):
Figure 1: Correlation of aircraft vertical tail volume as a function of fuselage maximum height and length [21].

Table 1: Volume coefficients and several dimensions for several turboprop aircrafts [8]. Lengths are in ft and areas are in sq. ft.

<table>
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<tr>
<th>Aircraft</th>
<th>b</th>
<th>$c_{mac}$</th>
<th>$l_{v,h}$</th>
<th>$S$</th>
<th>$S_v$</th>
<th>$S_h$</th>
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<td>0.07</td>
<td>0.72</td>
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Figure 2: Plot for volume coefficients reported in Table 1.
• the standard configuration with roots of both horizontal and vertical surfaces attached directly to the fuselage is ‘structurally convenient’. Aerodynamic interference with the fuselage and horizontal tail increase the effectiveness of the vertical tail. However large areas of the tails are affected by the converging fuselage flow, which can reduce the local dynamic pressure;

• a T-tail is often chosen to move the horizontal tail away from engine exhaust and to reduce aerodynamic interference. The vertical tail is in its most effective position, being ‘end-plated’ on one side by the fuselage and on the other by the horizontal tail. The disadvantages of this arrangement include higher vertical fin loads, potential flutter problems and deep-stall;

• V-tails combine functions of horizontal and vertical tails. They are sometimes chosen because of their increased ground clearance, reduced number of surface intersections, or novel look, but require mixing of rudder and elevator controls and often exhibit reduced control authority in combined yaw and pitch maneuvers;

• H-tails or twin tails use the vertical surfaces as endplates for the horizontal tail, increasing its effectiveness and thus saving vertical tail span. Sometimes are used on propeller aircraft to reduce the yawing moment associated with propeller slipstream impingement on the vertical tail but more complex control linkages and reduced ground clearance discourage their more widespread use;

• Y-shaped tails have been used when the downward projecting vertical surface can serve to protect a pusher propeller from ground strikes.

The problem of directional stability and control is first to ensure that the airplane will tend to remain in equilibrium at zero sideslip and second to provide a control to maintain zero sideslip during maneuvers that introduce moments tending to produce sideslip. Although a tailless airplane is realizable, like the flying wing, whose directional stability is given by the swept wing and pushing propellers or an active control of the lateral control surfaces, the vertical plane is the very main component of directional stability.

From the dynamic point of view, the role of the vertical tail is to provide yaw damping. If the vertical tail volume coefficient is too small, for a given dihedral effect or lateral stability, the aircraft tends to oscillate in yaw as the pilot gives rudder or aileron inputs. This tendency is called dutch roll (see Figure 4) and makes precise directional control difficult.

1 Angle between the relative wind and the longitudinal plane of symmetry of the airplane.
Figure 3: Geometry of various tail arrangements [13, 41].

Figure 4: Dutch roll oscillation tendency from insufficient vertical tail volume [22].
Extreme flight conditions often set design requirements for tail surfaces, like minimum control speed with One Engine Inoperative (OEI) or maximum cross-wind capability: stability and control must be ensured even in very large angles of sideslip, up to 25° [29]. Federal Aviation Regulations (FAR) related to lateral-directional stability are reported in Appendix A. Similar requirements are dictated by the European Aviation Safety Agency (EASA) in Europe with Certification Specification part 25.

Design of vertical planes depends on the type of airplane (and so the flow regime), engine numbers and position, wing-fuselage and horizontal tail position [40]. These factors affect the stability derivatives, that is the variation of aerodynamic coefficients with the independent variable, the angle of sideslip $\beta$. It is somewhat complicated since it involves asymmetrical flow behind the wing-fuselage combination and lateral cross-control¹.

The following design requirements can be formulated for vertical tail planes [29]:

1. they shall provide a sufficiently large contribution to static and dynamic stability, that is the sideforce derivative of the isolated vertical tail

   \[ C_{Y_{\beta v}} = C_{t-a v} \frac{S_v}{S} \]

   has to be determined. The vertical tail directional stability derivative is $C_{N_{\beta v}}$, that is the yawing moment coefficient due to sideslip, however it can be shown that it depends from the coefficient just defined. If a high lift gradient is desirable the aspect ratio should be the largest possible with the minimum sweep.

2. The same can be stated for sufficient control capability. Moreover control should be possible with acceptable control force

   \[ F = C_h \frac{\rho V^2}{2} S_c \bar{c}_c \]

3. High angles of sideslip (up to 25°) can be reached and this condition is more serious when flying in possible icing conditions. In this case a low aspect ratio is required and sweep is beneficial (they delay the stall at higher angle of sideslip, but reduce the lift gradient).

4. Equilibrium has to be achieved in all flight conditions. This gives specific requirements on tail surface areas and on the maximum lift coefficient with various amount of control surface deflection and should include the effect of ice roughness.

¹ Lateral control is provided by ailerons, but side forces on vertical planes cause also a rolling moment.
Table 2: Tail aspect ratio and taper ratio guideline, by Raymer [35].

<table>
<thead>
<tr>
<th></th>
<th>Horizontal tail</th>
<th>Vertical tail</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>$\lambda$</td>
<td>$\lambda$</td>
</tr>
<tr>
<td>Fighter</td>
<td>3 - 4</td>
<td>0.2 - 0.4</td>
</tr>
<tr>
<td>Sail plane</td>
<td>6 - 10</td>
<td>0.3 - 0.5</td>
</tr>
<tr>
<td>Others</td>
<td>3 - 5</td>
<td>0.3 - 0.6</td>
</tr>
<tr>
<td>T-tail</td>
<td>—</td>
<td>—</td>
</tr>
</tbody>
</table>

Table 3: Sizing of the vertical stabilizer in function of the wing parameters, by Raymer [35].

<table>
<thead>
<tr>
<th>Wing</th>
<th>Vertical</th>
<th>Comment</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\Lambda_{LE}$</td>
<td>35° - 55°</td>
<td>Tail stall later than wing and has a higher critical Mach number.</td>
</tr>
<tr>
<td>$\Lambda$</td>
<td>1.3 - 2</td>
<td>Must be lighter than wing.</td>
</tr>
<tr>
<td>$\lambda$</td>
<td>0.3 - 0.6</td>
<td>Close to an elliptical load and easy to manufacture.</td>
</tr>
<tr>
<td>t/c</td>
<td>9% - 12%</td>
<td>Usually similar to wing section’s relative thickness.</td>
</tr>
<tr>
<td>$\bar{c}_c/c_v$</td>
<td>25% - 50%</td>
<td>Typical plain flaps with same taper ratio of tailplane.</td>
</tr>
</tbody>
</table>

5. A high aspect ratio has an adverse effect on weight. Also, for T-tails the flutter analysis requires extra care.

6. Excessive taper ratio may lead to premature tip stall. On the other hand, tapering leads to lower height.

So a compromise in high lift gradient and low aspect ratio and taper ratio must be considered. Some indications come from Raymer [35] and are reported in a general fashion in Table 2 and in function of the wing parameters in Table 3.

1.2 A HISTORICAL PERSPECTIVE

From the ‘30s to the ‘50s, in the USA, the National Advisory Committee for Aeronautics (NACA) provided some results on the directional stability on isolated vertical tailplanes, partial and complete aircraft configurations through many hours of wind tunnel tests, results that were summed up in a method of analysis completely reported and described in the USAF DATCOM by Finck [12]. The investigations strived

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3 A collection, correlation, codification, and recording of best knowledge, opinion, and judgment in the area of aerodynamic stability and control prediction methods, with
to separate the effects of fuselage, wing and horizontal tail from the isolated vertical tail. Lots of geometries were tested, from the early years to the ’50s, i.e. rectangular, elliptical and swept wings, symmetrical and unsymmetrical airfoils, slender bodies with rounded or sharp edges, tails of different aspect ratio and size [9, 18, 5, 34, 6].

However the accuracy of these results were limited by the technology of the past years and, perhaps most important, they dealt with geometries quite different from the actual transport airplanes. In fact most of the work of the NACA was pushed by war and if the aim of the early tests was to gain a certain knowledge on the physics of the problem of directional stability and control [18] and on the mutual interference among aircraft components [9], later tests aimed to improve stability and maneuverability of high speed combat aircrafts [34].

A first effect studied (1939) by Bamber and House [5] was the aerodynamic interference of the wing-fuselage relative position on the vertical tail aerodynamic coefficients. The general trend revealed an increase in sideforce due to sideslip coefficient $C_{Y\beta v}$ and yawing moment derivative $C_{N\beta v}$ when moving the wing from high to low position in fuselage (Figure 6, respectively right and middle charts), excepts for the fin-off combination where there was a minimum for the mid wing position (without the fin the model considered has double symmetry, see Figure 5). This trend held when considering wing dihedral, flaps and angle of attack, though different values were achieved for each combination. The rolling moment derivative $C_{L\beta v}$ decreased and changed sign as the wing was moved downward (Figure 6, left chart). The effect of the angle of attack on $C_{N\beta v}$ was very small and so for that geometry can be neglected (Figure 7). The work of Bamber and House [5] does not consider the horizontal tail, so the interference effect on lateral stability was due to that particular wing-fuselage-fin combination.

An interesting approach to evaluate the fuselage effect on the vertical tail was taken by Queijo and Wolhart [34]: an effective aspect ratio $\lambda_{c_v}$ was defined and it was plotted against the ratio of vertical tail span $b_v$ to the fuselage diameter $d_f$ at the longitudinal location of the vertical tail aerodynamic center (shortly: tail span per body depth) and parametrized for various tail-fuselage combinations (Figure 8). The tail effectiveness increased as the ratio $b_v/d_f$ decreased, that is as the tail became small compared to the fuselage. Theory (based on the assumption that the body acts as an endless plate at the base of the vertical tail) suggested a non linear increase of the effective aspect ratio with the ratio mentioned above. The scatter in experimental data became bigger with low $b_v/d_f$ ratios and two different trend lines were drawn for two aspect ratios.
Figure 5: Geometries for the report of Bamber and House [5].

Figure 6: Some results of the work of Bamber and House [5].
Figure 7: Insensitivity of the stability derivative $C_{n_{\beta}}$ to angle of attack, by Bamber and House [5].

Figure 8: Effective aspect ratio of vertical tails as influenced by the fuselage ($\alpha = 0^\circ$). The dashed line is referred to the equation derived from theory. Taken from the work of Queijo and Wolhart [34].
Before 1950 tests on straight wings and horizontal tails were made, but it was found by Brewer and Lichtenstein [6] that results could not be extended to swept planforms. The model used for the investigations [34, 6] was designed to permit tests of the wing alone, fuselage alone, or the fuselage in combination with any of several tail configurations, with or without the wing. Details will follow in Chapter 4. In the work done by Queijo and Wolhart [34] the horizontal tail was not mounted (Figure 9).

The effect of size and position of horizontal tail was studied by Brewer and Lichtenstein [6] (Figure 10). Here the fin-fuselage combination was ‘frozen’ and the effective lift curve slope of the isolated vertical tail was calculated, as shown in Figure 11. Here, the experimental lift-curve slope is about 13% higher than that predicted by theory. Indeed, according to the lifting surface theory, the gradient of the lift coefficient vs angle of attack curve is linear with aspect ratio for planforms with low aspect ratio (up to about 1.5, see Figure 12). So it results

\[ C_{L_{\alpha v}} = 0.0274 A \text{ deg}^{-1} \]

where

\[ A = \frac{b^2}{S} \]  \hspace{1cm} (5)

so that

\[ L = C_{L_{\alpha v}} \alpha \frac{\rho V^2}{2} b_v^2 \]  \hspace{1cm} (6)

and this leads to the conclusion that, for most vertical tail surfaces, at a given fin angle of attack, the side force is only dependent on the fin height (tail span \( b_v \)) and planform is of secondary importance [29].

Results of the work by Brewer and Lichtenstein [6] show that the effect of the horizontal tail can be positive for stability if moved upward and rearward on the vertical tail or if positioned on the body, for low angle of attack (see configurations of Figure 13). At high angle of attack flow separation and the wing induced downwash can highly affect the stability, though positive effects can be obtained moving the horizontal tail forward and upward.

The ‘final’ effective aspect ratio was found to be function of the ratio of the effective aspect ratio in presence of fuselage (wing off) with horizontal tail on and horizontal tail off (Figure 14), corrected for the horizontal surface vs vertical surface ratio \( S_h/S_v \) (Figure 15).

All this matters were included in the United States Air Force Data Compendium (USAF DATCOM), as will be clearly described in Chapter 2. It is curious that an entire method was built on a dozen-point
Figure 9: Geometries for the report of Queijo and Wolhart [34]. All units are in feet.
Figure 10: Geometries for the report of Brewer and Lichtenstein [6]. All units are in inches.

Figure 11: Comparison with theory of experimental lift curve shapes. Wing and vertical, $\alpha = 0^\circ$. Taken from the work of Brewer and Lichtenstein [6].
A historical perspective

Figure 32.1 - Vertical tail plane lift curve slope. Source: NACA TN 2010, Rep. 1098

This is controlled local flow separation which stabilises the flow further outboard postponing complete flow separation to a higher angle-of-sideslip. Thus a higher maximum lift and a higher stall angle are achieved.

On the full-scale F-27 dorsal fin no.1 was selected. The reason is evident although fin no.6 could also have been a candidate.

Figure 32.2 - Vertical tail lift curve slope (I). Source: Forschungsbericht FB 1519 / 3

Figure 32.3 - Vertical tail lift curve slope (II). Source: NACA TN 2010

A = 1.0
λ = 0.6
Λ = 45 deg
CLαv = 0.027 /deg

Figure 12: Vertical tail plane lift curve slope [29].

Figure 13: Vertical tails analyzed in the report of Brewer and Lichtenstein [6].
Figure 14: Variation of effective vertical tail aspect ratio with horizontal tail position, $\alpha = 0^\circ$, wing off. Taken from the work of Brewer and Lichtenstein [6].

Figure 15: Variation of $K_{SH}$ with horizontal tail area. Taken from the work of Brewer and Lichtenstein [6].
scattered chart proposed by Queijo and Wolhart [34], related to a highly swept wing and tails, military-like aircraft. As a matter of fact, during the bibliographic searches for this work, it was a surprise to discover that the section of the USAF DATCOM method to estimate stability derivatives for subsonic airplanes is mainly based on a couple of reports [6, 34] that dealt with swept wings and stabilizers and mainly sharp elliptical bodies. Important charts were derived from these reports, though they can be applied (and are still applied today) to conventional airplanes.

Apart from the NACA, in the UK, the Engineering Science Data Unit (ESDU) proposed an alternative method to compute the vertical tailplane contribution to directional stability in presence of body, wing and horizontal tailplane, described by Gilbey et al. [14]. This method contemplates conventional geometries, a (almost) circular fuselage and a constant sidewash. It is a synthesis of experimental analyses done from NACA, BAe, SAAB and others, from the ’40s to the ’70s, linked together with potential flow theory where the data were highly scattered. The theory at the base is found in the work of Weber and Hawk [42], who suppose that a fin-body-tailplane combination at incidence (or sideslip) develops a vortex system that induces a constant velocity along the wing span. Here the term ‘wing’ is used as generic lifting surface, since the vertical fin in fuselage is considered as a wing with a cylinder on a tip. Load distribution is computed in the Trefftz-plane (located far downwind for high aspect ratio wings) once the induced velocity is known. The latter is calculated from the wing characteristics as planform, sweep angle and wing section’s lift curve slope.

From the ’50s to the ’90s in the USA researches were concentrated mainly on high subsonic and supersonic flow field and until the ’70s only wind tunnel tests could fit for the objective [1]. Usually the directional stability of the models tested deteriorated with increasing angle of attack and increasing Mach number. Interference effects had a strong influence upon the vertical-tail effectiveness and, consequently, upon the directional stability. These effects are, for the most part, associated with complex flow involving vorticity or shock waves and therefore were difficult to analyze at that time [38]. In the ’70s first computer programs began to appear on the scene, with the application of linear, potential, subsonic flow theory. Initially the quasi-vortex-lattice method was applied and modified to account for wing-body effect in sideslip [24], then more and more sophisticated panel methods provided solution (in potential flow) for almost any ge-

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Panel methods are numerical schemes for solving the Prandtl-Glauert equation for linear, inviscid, irrotational flow about aircraft flying at subsonic or supersonic speeds. There are fundamental analytic solutions to the Prandtl-Glauert equation known as source, doublet and vorticity singularities. Panel methods are based on the principle of superimposing surface distributions of these singularities over small quadrilateral portions, called panels, of the (approximate) aircraft surface. The re-
ometry, developed for rapid accurate estimates of the aerodynamic characteristics of aircraft and missile configurations at supersonic speeds [23].

Since panel methods are restricted to inviscid, irrotational and linear flow it is common practice to include the presence of the boundary layer with a viscous code coupled with the panel method, so that the pressure distribution from the latter is the input for the former to compute the displacement thickness. This incremental thickness is the new input geometry for the panel code and so on, until convergence is reached [10].

Most recent panel codes are used to determine the stability and control derivatives of new aircraft configurations early in the design process, since these parameters are important also to most control law design methods and their early estimate may permit significant improvement in configuration weight, cost, performance and even stealth, through multidisciplinary design [32].

These panel methods are part of the Computational Fluid Dynamics (CFD), though the approach with the Prandtl-Glauert equation has little physics inside\(^5\).

Other and more recent (last 15 years) CFD methods makes use of finite differences [3], Finite Element Method (FEM) [31] and finite volume (see Chapter 3) methods. Any further step in stability and control analysis techniques saw a return to the study of the low subsonic flow field [31, 32].

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\(^5\) The Prandtl-Glauert equation involves only a scalar potential and the free stream Mach number. It’s just the wave equation in a steady compressible flow regime.
Viscosity is responsible of momentum loss in boundary layer because of skin-friction and pressure drag, increasing boundary layer thickness, decreasing lift gradient and flow separation [30]. Neglecting viscosity gives a good approximation only in attached flow regime. This limit can be overcome by modern CFD. Nowadays powerful computational fluid dynamics (Navier-Stokes) tools offer significant benefits as a companion to the experimental methods to predict aerodynamic parameters. In fact, the wind tunnel is traditionally the primary tool to provide aerodynamic inputs for simulation data bases and to predict stability and control characteristics, but shortcomings exist in wind tunnel testing methods (for example operation at a lower than free flight Reynolds number causes large discrepancies on boundary layer separation in certain configurations) that can result in serious errors in the predicted stability and control characteristics [16]. Moreover wind tunnel tests require both the construction of a model and an adequate test facility. Additionally, the lag time between the paper design and the wind tunnel results can be considerable. Furthermore, any configuration change requires a change of the test model [31]. This greatly increases the cost of the product and the time to market.

On the other hand, the simple theory at the base of the USAF DATCOM or other semi-empirical method is only accurate for preliminary relationships between the overall aircraft geometry and stability. CFD offers a more direct approach in finding stability coefficients, computing aerodynamic forces and moments by integration of surface pressures along the aircraft boundaries. In general, the most complicated geometry can be solved with the proper selection of a CFD method (and right boundary conditions) [31], so it is believed that CFD tools can completely cope with stability and control issues and complement the traditional wind tunnel data sources [16]. The decreasing costs and the increasing power of the computers makes the CFD more and more attractive.

As stated by Johnson, Tinoco, and Jong Yu [19], the strength of CFD is its ability to inexpensively produce a small number of simulations leading to the necessary understanding for design. It can be used in an ‘inverse design’ or optimization mode, predicting the necessary geometry shape changes to optimize certain flow characteristics or a payoff function (e.g. drag). Beyond this, CFD is heavily used to provide corrections for the extrapolation of data acquired experimentally. Examples of inverse design and optimization are the work of Nicolosi and Della Vecchia [27] and Kasparov [20].

1.3 A NEW APPROACH

In looking for references [1] it appeared that all the analyses done in the past dealt with specific aircrafts and no one, except for the meth-
ods cited above, approached the stability and control problem from a methodological point of view. Discrepancies between these methods, along their obsolete geometries, made the choice to attempt a different approach through the CFD. The new procedure is explained in Section 5.4 and it is based on a simpler formulation respect to the USAF DATCOM method with an approach as the ESDU. For the purpose of the introduction, here it is stated that in this new procedure there is no longer an effective aspect ratio, while the wing-body, body-fin and horizontal tail effects are taken in consideration by multiplicative coefficients. It is immediate to use, just looking at four charts without defining any other parameter, once chosen the airplane’s configuration and main dimensions and its results are close to that provided by the other methods. This new procedure was built upon lots of (about 200) CFD analyses on a typical regional turboprop configuration. Experimental tests are to be conducted soon to further validate the method.
This chapter deals with the analysis of the (single) vertical tail of a conventional airplane as the regional turboprop. It is valid for low speed (subsonic) in cruise configuration (low angle of attack, low angle of sideslip) and does not account for flaps or engine effects.

Methods involved here, proposed by Roskam [36] and by ESDU [14], have a common feature: the sideforce derivative of the isolated vertical tail is corrected by the effects of wing, body and horizontal tail. However the interference factors are computed in different ways.

The basic equation is the definition of the lift curve slope for tapered wings

$$ C_{L_{\alpha}} = \frac{2\pi A}{2 + \left[ \frac{B^2 A^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{c/2}}{B^2} \right) + 4 \right]^{\frac{1}{2}}} $$

which is the Helmbold-Diederich formula [7], where

- $A$ is the wing (or tail) aspect ratio, $b^2/S$
- $B$ is the compressibility parameter, $\sqrt{1-M^2}$
- $\kappa$ is the ratio of section lift-curve slope to theoretical thin-section value\(^1\), $c_{L_{\alpha}}/(2\pi/B)$, and for thin airfoil ($c_{L_{\alpha}} \approx 2\pi$) it is equal to $B$
- $\Lambda_{c/2}$ is the sweep angle at half chord.

For a variation of the lift curve slope with aspect ratio see Figures 11 and 12.

2.1 USAF DATCOM METHOD

Here follows the description of the method proposed initially by Finck [12] and later adopted for airplane design by Roskam [36]. The sideslip derivative $C_{Y_{\beta}}$ is made up of three components: body cross-flow, horizontal surface, wing-body wake and sidewash effect (the latter two lumped into a single effectiveness parameter).

A body in sideslip exhibits a flow characteristic similar to a cylinder in cross flow. For potential flow the peak local velocity occurs at

---

\(^1\) Roskam [36] defines $\kappa = (c_{L_{\alpha}}/B) / 2\pi$ while ESDU [7] defines $\kappa = c_{L_{\alpha}} / (2\pi/B)$. However, for small Mach numbers and for the typical aspect ratios of tailplanes, the difference is negligible (zero for incompressible flow).
the top at the cylinder and is equal to twice the free-stream cross-flow velocity. Actually, separation exists on the leeward side, reducing the peak velocity from the potential-flow value. In either case, the velocity decays to the free-stream cross-flow value with distance from the body surface. Thus, tail-body combinations with large bodies and small tails have a greater effectiveness per unit area than combinations with large tails and small bodies and this trend is exhibited by test data. The vertical panel itself causes a load carry-over from the panel onto the body. This carry-over increases the effectiveness of the vertical panel and is included in the method presented\textsuperscript{2} [12].

![Streamlines for a body-fin combination at $\beta = 10^\circ$.](image)

The presence of a horizontal panel in the vicinity of a vertical panel causes a change in the pressure loading of the latter if the horizontal panel is at a height where the vertical panel has an appreciable gradient, i.e. at a relatively high or low position. Test data substantiate the greater effectiveness of horizontal panels in these positions and the

\textsuperscript{2} Thus the fuselage directly alters the vertical tail incidence because of the cross-flow around the body.
relative ineffectiveness of a horizontal panel at the midspan position on the vertical panel [12]. This is known as the end-plate effect.

For a wing-body combination there are two contributions to the sidewash present at a vertical tail: that due to the body and that due to the wing. The sidewash due to a body arises from the side force developed by a body in yaw. As a result of this side force, a vortex system is produced, which in turn induces lateral-velocity components at the vertical tail. This sidewash from the body causes a destabilizing flow in the airstream beside the body. Above and below the fuselage, however, the flow is stabilizing. The sidewash arising from a wing in yaw is small compared to that of a body. The flow above the wake center line moves inboard and the flow below, the wake center line moves outboard. For conventional aircraft the combination of the wing-body flow fields is such as to cause almost no sidewash effect below the wake center line [12].

The lift curve slope of the isolated vertical panel is calculated with eq. (7). When coupling the vertical tail with wing, fuselage and horizontal tailplane references [12, 36] define an effective aspect ratio $A_{v_{eff}}$. This accounts for fuselage depth in the region of the vertical panels, horizontal tailplane position and surface tails ratio $S_h/S_v$, according to the formula

$$A_{v_{eff}} = \frac{A_{v(f)}}{A_v} A_v \left[ 1 + K_vh \left( \frac{A_{v(hf)}}{A_{v(f)}} - 1 \right) \right]$$

where

3 This effect, analogous to the downwash in the longitudinal plane, indirectly affects the incidence of the vertical tail because of the generation of a vortex system.

4 At subsonic speeds the vehicle body and horizontal tail affect the flow on the vertical tail in such a way as to increase the effectiveness of the vertical tail. This phenomenon, known as the end-plate effect is represented by an effective change in panel aspect ratio required to give the same lift effectiveness as the actual panel in the presence of the other vehicle components. Interferences also exist between the vertical tail, the body and any forward lifting surface [12].
(a) Pressure coefficient with a body-mounted horizontal tail.

(b) Pressure coefficient with a low fin-mounted horizontal tail.

(c) Pressure coefficient with a mid fin-mounted horizontal tail.
Figure 19: Effect of the horizontal tail on fin pressure distribution.
$A_v$ is the vertical tail geometric aspect ratio, $b_v^2/S_v$

$A_{v(f)}/A_v$ is the ratio of the vertical tail aspect ratio in the presence of the fuselage to that of an isolated vertical tail, defined in Figure 22

$A_{v(hf)}/A_{v(f)}$ is the ratio of the vertical tail aspect ratio in the presence of the horizontal tail and the fuselage to that of the fuselage alone, defined in Figure 23

$K_{vh}$ is a factor which accounts for the relative size of the horizontal and the vertical tail, defined in Figure 24.

Several doubts arise when consulting Roskam [36] for the first time, since some definitions are not clear. Considering the effective vertical tail geometries in Figure 20, it is apparent that tail span $b_v$ changes with the fuselage stern shape. Comparing the same stabilizer with different bodies it could be difficult to define $b_v$ and when it should be extended to some fuselage reference line.

Gilbey et al. [14] (ESDU) define the geometry in a different way (see Sec. 2.2), but both methods define the same planform when using the configuration highlighted in Figure 20, this one was used in numerical analysis. It’s the most common configuration since a slight taper of the fuselage’s stern can be often neglected.

Another question is the meaning of the statement ‘fuselage depth in the region of vertical tail’ (Figures 22 and 23). Consulting the references [5, 6, 34] it was clear that this parameter refers to the quarter point of the mean aerodynamic chord of the vertical tail. That is the depth of the fuselage where is the projection of the quarter point of vertical tail’s mean aerodynamic chord on the fuselage centerline, shown in Figure 21.

Eventually, the former doubts were so solved:

- the geometric definition of the isolated tail plane is that of Figure 20;
- when computing the correction factors (and only in this case) in eq. (8) the vertical tail is ‘extended’ to the centerline, and so its span $b_v$ and taper ratio $\lambda_v$;
- the ‘region of the vertical tail’ is the projection of the quarter point of vertical tail’s mean aerodynamic chord (m.a.c.) on the fuselage centerline (Figure 21);
- however, the vertical tailplane’s m.a.c. is that of the isolated tailplane. In fact, it is impossible to define the m.a.c. of the extended tailplane a priori, since it is necessary to know the extended tail span, which in turn depends on the projection of the quarter point of the m.a.c. on the fuselage centerline$^5$.

$^5$ A possibility could be to ‘submerge’ the isolated vertical tail into the fuselage, so that there’s no need to extend the geometric tail span $b_v$, because the stabilizer’s root
Figure 20: Examples of effective vertical tail geometry, by Roskam [36].
All of these matters have accounted in writing a computer code for the automation of the semi-empirical analysis (see Sec. 2.3).

Returning to the main subject, the vertical tail contribution to the sideforce derivative is

$$C_{Y\beta} = -k_v C_{L_{av}} \left(1 + \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S}$$

where

- $\beta$ is the sideslip angle
- $k_v$ is given in Figure 25
- $C_{L_{av}}$ is the lift curve slope corrected by $A_{v\text{eff}}$ defined in eq. (8)

chord would intercept the fuselage centerline. However the definition of planform area $S_v$, a major factor in the determination of the stability derivatives, would be different from that highlighted in Figure 20.
2.1 USAF DATCOM Method

Figure 23: Fuselage-horizontal tail-vertical tail interference factor, by Roskam [36]. Note that now the span $b_v$ extends to the fuselage centerline.

Figure 24: Factor which accounts for relative size of the horizontal and vertical tails, by Roskam [36].

Figure 25: Empirical factor for estimating sideforce-due-to-sideslip of a single vertical tail, by Roskam [36].
\((1 + d\sigma/d\beta) \eta_v\) is the sidewash effect

\(S_v/S\) is the ratio of the vertical tail area (defined in Figure 20) to the wing area.

So the corrected \(C_{L_{\alpha_v}}\) is corrected again by another factor that accounts for body depth in the region of vertical tail, by the sidewash effect and it is scaled by the surface ratio. The sidewash effect can be amplifying or reductive\(^6\), being the wing position the key parameter

\[
\left(1 + \frac{d\sigma}{d\beta}\right) \eta_v = 0.724 + 3.06 \frac{S_v/S}{1 + \cos \Lambda_{c/4}} + 0.4 \frac{z_w}{z_f} + 0.009A \tag{10}
\]

where

\(\Lambda_{c/4}\) is the wing quarter chord sweep angle

\(z_w/z_f\) is the ratio of the wing position, computed from centerline and negative for a high wing, to the fuselage height in the wing region, that is \(-0.5\) for a high wing, \(+0.5\) for a low wing, 0 for middle wing

\(A\) is the wing aspect ratio.

Sidewash parameters are additive, while effective aspect ratio parameters are multiplicative. The final correction factors multiply the \(C_{L_{\alpha_v}}\).

There are some additional effects that are not accounted for by the method. For instance, dorsal fins may cause a considerable error in the values obtained, although the effect of dorsal fins is more pronounced at the higher angles of sideslip. Dihedral in the horizontal surfaces is known to change the pressure loading on the vertical panel and hence its effectiveness. For rapidly converging bodies, flow separation frequently exists at the juncture of the vertical panel with the body. This effect generally decreases the effectiveness of the vertical tail and is not accounted for by the methods included herein. Similar effects can result when the maximum thickness of two orthogonal panels are made to coincide\(^{12}\).

### 2.2 ESDU Method

The method reported by Gilbey et al.\(^{14}\) is quite simple. The lift curve slope of the isolated vertical tailplane (see eq. (7)) is corrected by multiplying three empirical factors, \(J_B\), \(J_T\), \(J_W\), respectively body-fin, tailplane and wing correction factor, and scaled by the surface ratio \(S_f/S\) (see Figure 26 for the nomenclature). Factors \(J_T\) and \(J_W\) are

\(^6\) At subsonic speeds the empirical formula (9) gives the total sidewash effect directly, i.e. the combined sidewash angle and dynamic-pressure loss.
located on different curves, depending on the horizontal tail position (fuselage or fin)

\[ Y_{V_F} = -J_B J_T J_W C_{L_{\alpha F}} \frac{S_F}{S} \]  

(11)

where \( C_{L_{\alpha F}} \) is the \( C_{L_{\alpha V}} \) defined in eq. (7) when the fin aspect ratio

\[ A_F = 2 \frac{b_F^2}{S_F} \]  

(12)

is substituted to \( A_v \), so that this procedure is initially quite different from the USAF DATCOM method, but as will be shown in Section 2.4, results conform for conventional geometries.

This method contemplates conventional geometries, a (almost) circular fuselage and a constant sidewash. It is a synthesis of experimental analyses done from NACA, BAE, SAAB and others, from the ‘40s to the ‘70s, linked together with potential flow theory [42] where the data were highly scattered.

The fin is considered a trapezoidal panel, any extension like dorsal fairing or a curved fin tip is ignored and the leading edge is extended linearly in the body. The fin panel tip chord is the chordwise distance between the leading and trailing edges of the fin at the maximum height. The fin panel root chord is the chordwise distance between the (extrapolated) leading and trailing edges of the fin at the height where the fin quarter-chord sweep line intersects the top of the body [14] (Figure 26).

Values of \( J_T \) for body-mounted tailplanes are referred to near body centerline’s position, otherwise some caution is necessary for tailplanes mounted high on the body, close to the fin-body junction.

The method is applicable to conventional airplanes in cruise (clean) configuration at small angles of attack and sideslip, where there is an essentially linear variation of the sideforce, yawing moment and rolling moment with the angle of sideslip. In practice, because of departures from a linear variation, static lateral stability derivatives are defined from experimental data over a small range of sideslip angles, typically between \( \pm 2^\circ \) and \( \pm 5^\circ \). Almost all of the data studied come from wind-tunnel tests carried out at low speeds and the method introduces compressibility effects only through the basic fin lift-curve slope estimated from ref. [7].
2. INTRODUCTION
This Item gives a semi-empirical method for calculating $Y_{v}$, $N_{v}$, and $L_{v}$, the contributions of the vertical stabilising fin of an aircraft to the sideforce, yawing moment and rolling moment derivatives due to sideslip, at subsonic speeds. The aircraft geometries covered by the method are those where a single fin is located on top of the aircraft rear-body, and in the plane of symmetry, with the tailplane mounted either on the fin itself or on the rear-body. The shape of the fin is assumed to approximate to a trapezium. The method was developed for bodies with circular or nearly-circular cross-sections. It may be used, with caution, for bodies with elliptical or near-rectangular cross-sections by using a mean body diameter provided that the body height to width ratio in the region of the fin is close to unity. Otherwise the method of Item No. 93007 (Reference 39) that covers a wide range of body height to width ratios and both single and twin fins should be used.

NOTES
(i) Area is shown shaded.
(ii) $BB'$ indicates the fin root quarter-chord station, defined by the plane normal to the longitudinal body axis which passes through the point at which the fin quarter-chord line intersects the body. The body height and width, $h_{BF}$ and $d_{BF}$, are defined at this station.
(iii) Fin chords are defined parallel to the longitudinal body axis and fin, tailplane and body heights are defined perpendicular to it.

Figure 26: ESDU tailplane definition and geometries, taken from ref. [14].
Table 4: Range of geometries for the ESDU method [14]

<table>
<thead>
<tr>
<th>Body</th>
<th>Wing</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\frac{h_{BF}}{h_{BF} + h_F}$</td>
<td>$A_W$</td>
</tr>
<tr>
<td>0.1 to 0.5</td>
<td>2 to 11</td>
</tr>
<tr>
<td>$\frac{h_{BF}}{d_{BF}}$</td>
<td>$\Lambda_{1/4_W}$</td>
</tr>
<tr>
<td>1 to 1.15</td>
<td>$0^\circ$ to $60^\circ$</td>
</tr>
<tr>
<td>$\frac{z_W}{h_{BW}}$</td>
<td>+0.5 to −0.5</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Vertical fin</th>
<th>Horizontal tail</th>
</tr>
</thead>
<tbody>
<tr>
<td>$A_F$</td>
<td>$A_T$</td>
</tr>
<tr>
<td>1.0 to 5.0</td>
<td>0.5 to 5.5</td>
</tr>
<tr>
<td>$\Lambda_{1/4_F}$</td>
<td>$\Lambda_{1/4_T}$</td>
</tr>
<tr>
<td>$0^\circ$ to $60^\circ$</td>
<td>$0^\circ$ to $60^\circ$</td>
</tr>
<tr>
<td>$\lambda_F$</td>
<td>$\lambda_T$</td>
</tr>
<tr>
<td>0 to 1</td>
<td>0 to 1</td>
</tr>
<tr>
<td>$\frac{S_F}{S}$</td>
<td>$\frac{D_T}{h_F}$</td>
</tr>
<tr>
<td>0.05 to 0.27</td>
<td>0.5 to 4</td>
</tr>
</tbody>
</table>

Figure 27: Body-fin correction factor, taken from ref. [14].
Figure 28: Tailplane correction factor, taken from ref. [14].
Figure 3: Wing correction factor, taken from ref. [14].
2.3 SEMI-EMPirical METHODS IN MATLAB

It is interesting to compare the results of the USAF DATCOM [12] and ESDU [14] methods. This has been accomplished by writing a MATLAB script to account for all the effects described in Chapter 1. It is possible to study single complete or partial aircraft configurations as well as parametric studies, to evaluate the effect of a single parameter on the whole airplane or a partial configuration. The code was written thinking about the experimental model that soon will be available at the Department of Aerospace Engineering (DIAS) of the University of Naples ‘Federico II’: it is designed to have a modular vertical tail plane, that is a combination of several parts with a fixed sweep angle and root chord, to increment the aspect ratio, as shown in Figure 31. Actually it is possible to investigate the subsequent parameters:

- geometric tail aspect ratio,
- wing fuselage relative position,
- wing aspect ratio,
- wing span,
- horizontal tailplane relative position,
- horizontal tailplane size,
- vertical tail span vs fuselage depth (thickness),
- fuselage upsweep angle.

The MATLAB script is named VeTaRE.m that is ‘Vertical Tail analysis with the methods proposed by Roskam and ESDU’. It calls an input script and user-defined MATLAB functions, everything written from scratch, except for a standard atmosphere and a figure export utilities. Once defined the airplane’s main dimensions in the inputdata.m file, the main script can be called in the MATLAB Command Window. Giving an external input file is useful, since it is possible to define different airplanes in separate files and call the one desired, without rewriting all the input data.

The script can run in two different ways. In the ‘single shot’ mode the output is shown formatted in the Command Window and optionally written in a .mat file. If the ‘parametric’ mode is on there’s no output in the Command Window, but, since the user asked for a parametric study, the stability derivatives are plotted against the chosen parameter. It calculates the sideforce, yawing moment and rolling moment derivatives, showing also the percentage difference between the methods. In this thesis only the sideforce due to sideslip derivatives are shown, since the yawing and rolling moment derivatives depend on the former. An example application is described in the next section.
Figure 30: The ATR-42 configuration, generated by the MATLAB script. Dimensions are in m.

Figure 31: A modular vertical tail geometry, hypothesized for later wind tunnel testing. Dimensions are in m.
2.4 ANALYSIS FOR A REGIONAL TURBOPROP AIRPLANE

The sideforce coefficient’s derivatives of an ATR-42 (Figure 32) have been evaluated with the methods described in previous sections. Data is written in Table 5.

Results are reported in tables. A side-by-side result cannot be made since the two methods described above define different approaches. Here it is remarked that even a common starting point is difficult to achieve, since Roskam [36] defines the vertical tailplane aspect ratio as

\[ \Lambda_v = \frac{b_v^2}{S_v} \]

while ESDU [14] defines it as

\[ \Lambda_F = 2 \frac{b_F^2}{S_F} \]

where \( v \) stands for ‘vertical tail’ and \( F \) stands for ‘fin’ (with the same meaning), so even for the same planform (Figure 20) aspect ratios (and lift curve slopes) are differently defined.

---

Table 5: ATR-42 main dimensions

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Symbol</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight</td>
<td>W</td>
<td>16700 Kg</td>
</tr>
<tr>
<td>Wing surface</td>
<td>S</td>
<td>50 m²</td>
</tr>
<tr>
<td>Wing aspect ratio</td>
<td>( \lambda )</td>
<td>12</td>
</tr>
<tr>
<td>Wing taper ratio</td>
<td>( \lambda )</td>
<td>0.54</td>
</tr>
<tr>
<td>Wing span</td>
<td>( b )</td>
<td>24.5 m</td>
</tr>
<tr>
<td>Wing chord</td>
<td>( c )</td>
<td>2.04 m</td>
</tr>
<tr>
<td>Fuselage length</td>
<td>( l_f )</td>
<td>22.7 m</td>
</tr>
<tr>
<td>Fuselage width</td>
<td>( d_f )</td>
<td>2.6 m</td>
</tr>
<tr>
<td>Upsweep angle</td>
<td>( \vartheta )</td>
<td>12.5°</td>
</tr>
<tr>
<td>Horizontal tail surface</td>
<td>( S_h )</td>
<td>11.5 m²</td>
</tr>
<tr>
<td>Tailplanes rel. position</td>
<td>( z_h/b_v )</td>
<td>0.82</td>
</tr>
<tr>
<td>Vertical tail root chord</td>
<td>( c_{v\text{root}} )</td>
<td>3.4 m</td>
</tr>
<tr>
<td>Vertical tail span</td>
<td>( b_v )</td>
<td>4.5 m</td>
</tr>
<tr>
<td>Vertical tail aspect ratio</td>
<td>( \Lambda_v )</td>
<td>1.6</td>
</tr>
<tr>
<td>Vertical tail l.e. sweep</td>
<td>( \Lambda_{v\text{LE}} )</td>
<td>32°</td>
</tr>
<tr>
<td>Vertical tail t.e. sweep</td>
<td>( \Lambda_{v\text{TE}} )</td>
<td>20°</td>
</tr>
</tbody>
</table>
2.4 Analysis for a Regional Turboprop Airplane

Figure 32: ATR-42 three view.

Table 6: Correction factors for the ATR-42 stabilizer (USAF DATCOM method).

<table>
<thead>
<tr>
<th>Effect</th>
<th>Lookup</th>
<th>Input</th>
<th>Output</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aft fuselage depth</td>
<td>Figure 22</td>
<td>5.440</td>
<td>1.050</td>
</tr>
<tr>
<td>Position of the horizontal tail</td>
<td>Figure 23</td>
<td>-0.820</td>
<td>1.121</td>
</tr>
<tr>
<td>Relative size of the tailplanes</td>
<td>Figure 24</td>
<td>0.908</td>
<td>0.875</td>
</tr>
<tr>
<td>Empirical factor $k_v$</td>
<td>Figure 25</td>
<td>4.956</td>
<td>1.000</td>
</tr>
<tr>
<td>Sidewash effect</td>
<td>Equation 10</td>
<td>—</td>
<td>1.046</td>
</tr>
</tbody>
</table>

Table 7: Effect of the correction factors (USAF DATCOM method).

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Isolated tail</th>
<th>Coupled bodies</th>
<th>% $\Delta$</th>
</tr>
</thead>
<tbody>
<tr>
<td>$A_v$</td>
<td>1.6</td>
<td>1.86</td>
<td>17</td>
</tr>
<tr>
<td>$C_{L_{av}}$ rad$^{-1}$</td>
<td>2.243</td>
<td>2.528</td>
<td>13</td>
</tr>
</tbody>
</table>

Table 8: Correction factors for the ATR-42 vertical tail (ESDU method).

<table>
<thead>
<tr>
<th>Effect</th>
<th>Lookup</th>
<th>Input</th>
<th>Output</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lift curve slope (rad$^{-1}$)</td>
<td>Equation 7</td>
<td>—</td>
<td>3.619</td>
</tr>
<tr>
<td>Body-fin correction factor</td>
<td>Figure 27</td>
<td>0.206</td>
<td>0.890</td>
</tr>
<tr>
<td>Tailplane correction factor</td>
<td>Figure 27</td>
<td>0.820</td>
<td>1.083</td>
</tr>
<tr>
<td>Wing correction factor</td>
<td>Figure 29</td>
<td>-0.500</td>
<td>0.728</td>
</tr>
<tr>
<td>Corrected lift curve slope (rad$^{-1}$)</td>
<td>Product of previous output</td>
<td>2.537</td>
<td></td>
</tr>
</tbody>
</table>
Table 9: Result comparison (rad$^{-1}$) and deviation from USAF DATCOM.

<table>
<thead>
<tr>
<th>Derivative</th>
<th>Symbol</th>
<th>USAF DATCOM</th>
<th>ESDU</th>
<th>% Δ</th>
</tr>
</thead>
<tbody>
<tr>
<td>Corrected Lift Slope</td>
<td>$C_{L\alpha V}$</td>
<td>2.528</td>
<td>2.537</td>
<td>0.350</td>
</tr>
<tr>
<td>Sideforce</td>
<td>$C_{Y\beta V}$</td>
<td>−0.669</td>
<td>−0.642</td>
<td>4.020</td>
</tr>
<tr>
<td>Yawing Moment</td>
<td>$C_{N\beta V}$</td>
<td>0.375</td>
<td>0.355</td>
<td>5.430</td>
</tr>
<tr>
<td>Rolling Moment</td>
<td>$C_{L\beta V}$</td>
<td>−0.090</td>
<td>−0.086</td>
<td>3.570</td>
</tr>
</tbody>
</table>

2.5 PARAMETRIC STUDIES

Once defined the airplane and its stability derivatives in Section 2.4, here only one parameter is changed (e.g. the aspect ratio of the vertical tail or the position of the horizontal tailplane or something else), the others being kept constant, and it is observed how stability derivatives vary with that parameter. The whole airplane has been investigated, no partial configurations were considered. This is a test for the MATLAB script, based on a real and actual airplane.

Only the sideforce due to sideslip derivative of the vertical tail will be shown, since the others coefficients are dependent from it. Below every plot it is shown the percentage difference (deviation from the USAF DATCOM method). Normalization of the coefficients is realized with the input data, that is with the ATR-42 wing planform surface. All units are per rad. The effects of the following parameters were studied:

- vertical tailplane aspect ratio;
- wing-fuselage relative position;
- wing aspect ratio;
- horizontal tailplane position;
- tailplanes surface ratio;
- fuselage depth in the region of vertical panels;

described in the following sections.
2.5.1 Vertical tailplane aspect ratio

It is apparent the linearity of the methods involved. Differences in absolute value are small, while they increase for low aspect ratio tails.

Figure 33: Sideforce due to sideslip coefficient as function of vertical tail aspect ratio.
2.5.2 Wing-fuselage relative position

Low wing has the maximum effect and this effect is bigger with the ESDU method. The maximum difference is around 20%.

\[
\Delta \% = 20 \%
\]

Figure 34: Sideforce due to sideslip coefficient as function of wing position.
2.5.3 Wing aspect ratio

Here the wing chord $c$ is kept constant, while wing span $b$ and the wing area $S$ increase according to the formulas (for straight wings)

$$A = \frac{b}{c}$$

$$S = bc$$

and it is apparent that the ESDU method considers a constant side-wash. However, differences are small, with a maximum of 4% for the range considered.

Figure 35: Sideforce due to sideslip coefficient as function of wing aspect ratio.
2.5.4 Horizontal tailplane position

End-plate effect is apparent at extreme positions. Body-mounted tailplanes should be out of chart range, since zero abscissa indicates tailplanes mounted at the base of the fin. They are mounted on fuselage centerline instead and are plotted on the $y$ axis only for comparison. The smoother transition from the fin-mounted to the body-mounted tailplane on the USAF DATCOM curve is due to the continuity of the curves of Figure 22 and 23, while ESDU parameters are quite different for body-mounted and fin-mounted horizontal tailplanes (Figure 28 and 29).

![Graph showing sideforce due to sideslip coefficient as function of horizontal tailplane position.](image)

Figure 36: Sideforce due to sideslip coefficient as function of horizontal tailplane position.
2.5.5 Tailplanes surface ratio

Here the horizontal plane has a constant aspect ratio of 4.5, planform area and span increase at the same time. It seems that the size of the stabilizer affects very little the directional stability of the airplane.

Figure 37: Sideforce due to sideslip coefficient as function of tailplanes’ relative size.
2.5.6 Fuselage depth in the region of vertical panels

It is expected an increase of sideforce derivative with the fuselage depth (thickness). Both methods provide it, but USAF DATCOM contemplates a particular effect at low ratios as seen in Figure 22. Figure 38 is obtained varying $b_v/2r$, that is the ratio of the vertical tail span on the fuselage thickness about the location of vertical tail’s aerodynamic center. This can be practically obtained in several ways, changing the fuselage thickness $2r$ (perhaps the whole fuselage diameter $d_f$ or the upsweep angle $\vartheta$), the tail span $b_v$, both of the previous parameters or longitudinally translating the vertical tail on the fuselage. In this example, the change of the tail span $b_v$ is possible, but not correct, since the tail surface $S_v$ would also change and hence the $C_{Y_{\beta_v}}$ would be function of $b_v^2$, that is eq. (9) here repeated

$$C_{Y_{\beta_v}} = -k_v C_{L_{\alpha_v}} \left( 1 + \frac{d\sigma}{d\beta} \right) \eta_v \frac{S_v}{S}$$

would be a parabola. So, Figure 38 just represent the body effect. Additional information can be found in Appendix B.

![Figure 38: Sideforce due to sideslip coefficient as function of fuselage depth.](image-url)
2.6 CONCLUSIVE REMARKS

Both methods give similar results for a regional turboprop airplane like the ATR-42 (see Table 9) and same trends in parametric analyses, except for wing span, where ESDU consider a constant sidewash, and fuselage thickness, where ESDU provides a decreasing curve with the ratio $b_v/2r$, while the curve generated with the USAF DATCOM has a behavior like that of Figure 22, that is the methods have big differences for thick bodies and small tailplanes. So the methods main differences reside in:

- defining the aspect ratio of the vertical tail (see formula (12) for the ESDU method);
- considering the sidewash effect;
- calculating the body effect.

Both refer to the same NACA reports [5, 6, 18, 34], yet ESDU [14] used also some unpublished data from the aviation industry. It can be stated that there is no right method, since both are based on empirical data obtained in the past.

Neither have considered the typical regional turboprop aircraft geometry (a slender body with a straight wing) and this, along the fact that for certain configurations, e.g. the body-mounted horizontal stabilizer, the methods give different results, was the factor that stimulated the actual work: there is no method specifically developed for that kind of aircraft though it occupies a relevant position in the regional air transport market [8]. This new method, described in Chapter 5 is realized with the help of the CFD. The tools used are described in Chapter 3.
THE CFD APPROACH

In Chapter 1 modern CFD was described as a fascinating tool, because it is possible to visualize the flow behavior and the parameters of interest directly on the geometry object of study. The more accurate the physics models involved, the more realistic will be the results. This chapter is a description of the main tool that was used to fulfill the objective of this work, the commercial CFD software Star-CCM+.

Most of the Section 3.1, unless otherwise stated, comes from ref. [39]. Section 3.2 will describe the SCoPE computational grid, necessary to deal with lots of simulations of millions of cells.

3.1 THE SOFTWARE STAR-CCM+

Solving the Navier-Stokes equations, even in their simplest form, for a three-dimensional complex geometry, it’s not a trivial task. It is necessary a (bundle) software that provides Computer Aided Design (CAD) geometry import, mesh generation, solver and post-process analysis. It has to be reliable and possibly easy to use, the latter to better concentrate on the physics of the problem (about this work: external aerodynamics and directional stability). Star-CCM+ is more than a CFD solver. It is an entire engineering process for solving problems involving flow (of fluids or solids), heat transfer and stress, based on object-oriented programming technology. It can handle large models with parallel solver both in local (desktop computer) and on hundreds of CPUs on a cluster grid by Command Line Interface (CLI). In the local client everything run in a single environment, from the geometry creation to the results visualization (Figure 39).

Another interesting feature is the possibility to automate tasks with Java macros. They can be recorded and played inside the software environment, though they can be edited manually and called from CLI. This resulted convenient in the present work, since lot’s of runs were similar, changing only a parameter per run, e.g. the flow angle of attack or a component position on the airplane. Java macros are mandatory when executing Star-CCM+ on a cluster grid like SCoPE.

Star-CCM+ solver is based on the finite volume method. The solution domain is subdivided into a finite number of small control volumes, called cells, formed by a collection of faces, that in turn are a collection of vertices (points in space defined by a position vector, see Figure 40). The faces of a cell should not intersect each other, except where they touch along the common edges, that is the control volumes must not overlap. The volume mesh obtained is the mathematical
representation of the space where the problem is being solved, i.e. the computational domain. To preserve sharp edges, feature curves can be defined. Last, but not least, the software generates a single simulation (.sim) file, containing everything necessary to run locally.

3.1.1 Simulation workflow

The most general workflow is represented in Figure 41 and briefly described here.

**Geometry** can be imported from other CAD software or created directly in Star-CCM+, though in version 6 (used in this work) the CAD environment can handle only very simple shapes. Whatever the method, geometry is a collection of surfaces and curves.

**Simulation topology** is the computational model defined as regions and boundaries to which physics can be applied. For external aerodynamics, a volume (e.g. a block shape) representing the fluid domain to be simulated must contain the entire geometry inside.

**Mesh** is the numerical domain. Star-CCM+ can easily and automatically generate surface and volume mesh, once defined several parameters, including size and refinement quality.

**Physics models** can be easily enabled. Star-CCM+ can handle single and multi-phase fluid flow, heat transfer, turbulence, solid stress, dynamic fluid-body interaction, aeroacoustics and related phenomena.

**Reports, monitors and plots** should be defined and activated to check for convergence, since Star-CCM+ uses an iterative pro-
procedure to reach the solution to the transport equations that satisfies the boundary conditions for a chosen scenario.

**RUN THE SIMULATION** will automatically initialize the solution and launch the solver. For an interactive session, residuals will be plotted in the client workspace and reported in the output window. For batch sessions, residuals will be echoed to the command console. The simulation can be stopped and resumed anytime.

**RESULTS** can be visualized with *scenes* as contours, vectors and streamlines. It is possible to create animated scenes. Scatter plots are also possible. In an interactive session, graphical results can be visualized as the simulation run, step by step.

Examples are shown in Figure 41.

### 3.1.2 Mesh generation

Mesh in Star-CCM+ can be imported from an external source or generated within the software **CAD** environment. The starting point for generating a mesh is a surface description of some kind. These surfaces usually come from **CAD** or similar packages. Ideally, the imported or translated surfaces should be closed, manifold and triangulated with near equal sized triangles. If these criteria are not satisfied
Figure 41: General sequence of operations in a Star-CCM+ analysis.
3.1 THE SOFTWARE STAR-CCM+

(a) A cyclone separator designed with the Star-CCM+ CAD module.

(b) Surface mesh on the cyclone separator.

(c) Pressure contour on the longitudinal section of the cyclone separator.
Figure 41: Example of scenes from Star-CCM+. 

(d) Streamlines in the cyclone separator.

(e) Velocity vectors in the cyclone separator.

(f) Velocity vectors on the longitudinal section of the cyclone separator.
it is always possible to \textit{remesh} the surface or \textit{wrap} the important features with the tools included in Star-CCM+. The starting surface can also contain internal features such as baffles.

The \textit{core volume mesh} can contain either \textit{trimmed}, \textit{polyhedral} or \textit{tetrahedral} type cells (Figure 42). \textit{Prismatic cell layers} can be included next to wall boundaries to account for boundary layers. \textit{Volumetric controls} using shapes as rectangles (bricks) and spheres can also be included to increase or decrease the mesh density of both the core mesh and/or prism layer mesh. Alternatively, the \textit{thin mesher} can be used to produce either a tetrahedral or polyhedral volume mesh for thin geometries.

The mesh is automatically built on the regions and boundaries defined earlier by the user (see figure workflow), once defined (at least) a \textit{mesh continuum} and its parameters. A mesh continuum is the collection of meshing models that are used to generate the surface and volume mesh for the input geometry representing the individual regions to be used for the simulation. It is possible to enable more models at once. For example, starting from a poor CAD data like a STL file, the surface remesher model should be activated, then a volume mesh model must be chosen (if interested in three-dimensional analysis).

An interesting volume mesh model is the polyhedral mesher. It provides a balanced solution for complex mesh generation problems, containing approximately five times fewer cells than a tetrahedral mesh for a given starting surface. The polyhedral meshing model utilizes an arbitrary polyhedral cell shape in order to build the core mesh. In Star-CCM+, a special dualization scheme is used to create the polyhedral mesh based on an underlying tetrahedral mesh, which is automatically created as part of the process. The polyhedral cells created typically have an average of 14 cell faces (Figure 43). All of the simulations ran for this work used this model.

There are lot’s of properties that can be set, the most important being the cell size parameters. Basically the mesh size can be defined in two ways: relative and absolute. The relative size consists in choosing the \textit{base size} of the mesh, that is the reference length of the problem to study, and all other sizes will be a percentage of this base size. On the contrary, an absolute size will be fixed and unrelated to the reference value. Of course it is possible to have a part defined with relative sizes and another part defined in an absolute way. For example, it is possible to define the volume mesh size of the fluid region around an airplane relative to the wing chord (in this case the base size will be the length of the chord) and define the \textit{prism layer}, to account for boundary layer, in an absolute way, so that when changing the base size the entire core volume mesh will change, except for the prism layer. Defining the mesh size relative to a reference value is convenient, especially if a mesh size study has to be done, for a obvious reason: only the base size has to be changed.
Figure 42: Mesh types in Star-CCM+.

(a) Tetrahedral mesh.

(b) Hexahedral (trimmed) mesh.

(c) Polyhedral mesh.
Volume mesh is always created on a surface mesh. That is volume mesh size propagates from the surface mesh size. Three different parameters are used to control the surface size, namely:

- target size;
- minimum size;
- maximum size.

The target size is the desired edge cell length on the surface while the minimum and maximum sizes control the lower and upper bounds of the cell edge lengths (when refinements from curvature and/or proximity effects are included). Three combinations of the target, minimum and maximum surface sizes are allowed:

- minimum and target size;
- minimum and maximum sizes;
- only minimum size.

For example the first choice imposes to the software to not generate cells smaller than the minimum size and with an assigned target size.

### 3.1.3 Defining the physics

The mechanism is the same of the mesh generation: one or more physics continua models must be enabled. Model selection and deletion can only be done through an interactive dialog box. Figure 44 shows some possibilities. In general, Star-CCM+ models and solvers rely on the following areas:

- space, time and motion;
- materials;
• flow and energy;
• species;
• turbulence and transition;
• radiation;
• aeroacoustics;
• combustion;
• multiphase flow;
• solid stress;
• electromagnetism.

The analysis carried on for this work required only a steady state solver and an incompressible viscous flow model. It is known [4] that a viscous flow can be laminar or turbulent and that viscosity can be accounted only in a small region of the fluid adjacent to the body surface – the boundary layer. The flow regime imposed in the simulation files of this work is turbulent, since a laminar flow around an airplane is unrealistic. The turbulence model chosen is Spalart-Allmaras.

The original Spalart-Allmaras model was developed primarily for the aerospace industry and has the advantage of being readily implemented in an unstructured CFD solver, unlike more traditional aerospace models. It solves a single transport equation that determines the turbulent viscosity. This is in contrast to many of the early one-equation models that solve an equation for the transport of turbulent kinetic energy and required an algebraic prescription of a length scale [39]. It gives good results with attached boundary layers and flows with mild separation [43, 39, 16]. In short, it’s simple, fast and suitable for the scope of this work.

3.1.4 Convergence

The stopping criterium chosen is a prescribed number of iterative steps order of magnitude as thousand. Convergence is judged by looking at the residual plot and the wall $y^+$. The residual $r$ in each cell represents the degree to which the discretized equation is not completely satisfied. In a perfectly converged solution, the residual for each cell would be equal to machine roundoff [39].

Given the linear algebraic system of equations

$$Ax = f$$

let $x_d$ the ideal (true, exact) solution of the system. The (vector) residual $r$ is defined as

$$r = Ax - f$$
3.1 THE SOFTWARE STAR-CCM+

Figure 44: Models continua dialog boxes in Star-CCM+.
where \( x_i \) is an approximate solution (initially obtained by a guess and then corrected step by step). So the residual is the error generated by the approximate solution when it is substituted into the linear system (13) [25]. This is the concept of the residual and it is always valid, no matter how complicated its mathematical expression.

Star-CCM+ defines a residual plot averaging on all the cells of the domain, as follows [39]

\[
RM = \sqrt{\frac{\sum_{n\text{ cells}} r^2}{n}}. \tag{15}
\]

Residual monitors are created automatically after model selection, in this case: continuity, the components of the momentum and the turbulent viscosity.

Sometimes it is not sufficient to have low residuals for an acceptable solution. Having chosen Spalart-Allmaras as turbulence model, it is important to check the value of the dimensionless wall distance

\[
y^+ = \frac{u^* y}{\nu} \tag{16}
\]

where \( y \) is the normal distance from the wall to the wall-cell centroid, \( u^* \) is a reference velocity and \( \nu \) is the kinematic viscosity. According to the model’s formulation, the entire turbulent boundary layer, including the viscous sublayer, ought to be accurately resolved and the model can be applied on fine meshes, that is small values, order of magnitude as unity, are required [39].

### 3.2 THE SCoPE GRID INFRASTRUCTURE

At time of writing, no desktop computer could handle CFD 3D simulations of millions of cells in a reasonable amount of time. This work saw the light also thanks to the availability of the University’s cluster grid, since lots of configurations, from wing alone to the whole airplane, at several angles of incidence and Reynolds numbers had to be
analyzed. Runs with 16, 32 or 64 CPUs per simulation were commons to get results within a day.

Sistema Cooperativo per Elaborazioni Scientifiche Multidisciplinari (SCoPE) is a scientific data center, based on a grid computing infrastructure, and it is a collaborative system for scientific applications in many areas of research. It is a project started in 2006 by the University of Naples ‘Federico II’.

The data center hosts about 300 eight-core blade servers, 220 terabyte of storage, and is already able to accommodate 500 more servers. Actually it has over 2400 CPUs. The scientific applications are of the areas of Astrophysics, Chemistry, Mathematics, Medicine, Engineering and Physics. The data center is located in the Monte S. Angelo Campus, which already hosts the Faculty of Sciences and it is close to the Faculty of Engineering, with kilometers of preexisting optical fibers. The network infrastructure is shown in Figure 46.

Here follows some interesting data:

- localization in a building of about 150 m²;
- power plant capable of delivering 1 MW of electric power in a continuous mode;
- efficient cooling system, capable of dissipating 2000 W/m³ and 30 000 W per rack;
- standard (Gigabit Ethernet) networking infrastructure, with a high capacity switching fabric;
- low latency (Infiniband) networking infrastructure, with a single switching fabric for each group of 256 servers;
Figure 47: Some images of the SCoPE data center [2].

- large storage capacity, both NAS (Network Attached Storage) working with the iSCSI protocol, and SAN (Storage Area Network), working with a Fibre Channel Infrastructure;
- open source (Scientific Linux) for the operating system;
- integrated monitoring system for all the devices of the data center, able to monitor the most relevant parameters of server, storage, networking, as well as all the environmental parameters (as temperature, humidity and power consumption) [26].

Figure 47 is a glance of the data center. Running a Star-CCM+ simulation on SCoPE requires three external files, described in Appendix C.
4 TEST CASES

Test cases are necessary to assess the compliance of the CFD results to the available experimental data, before attempting to investigate anything else. Main references for this chapter are [9, 5, 6].

4.1 LONGITUDINAL TEST CASE – WING-BODY INTERFERENCE

In the work of Eastmen and Kenneth [9] 209 wing-fuselage combinations were tested in the NACA variable-density wind tunnel, to provide information about the effects of aerodynamic interference between wings and fuselage at a large value of the Reynolds number (3 100 000). The wing section is a NACA 0012 airfoil. Three of these combinations (respectively mid, high and low wing, marked no. 7, 22 and 67 in ref. [9]) plus a wing-alone configuration were chosen for the test case. In short, for each combination, a round fuselage with a rectangular wing has been analyzed at various angles of incidence, in symmetric flow. No discussion is made on the results since the primary interest of this chapter is the check of the previously stated compliance.

The CAD model was generated in CATIA (Figure 48). Configurations are shown in figures 49. The reference system for wing translation in fuselage is shown in Figure 50.

Mesh data is available in Table 10 and shown in Figure 51. CFD analyses show good agreement with the result of ref. [9], except at high flow separation, as expected (see Chapter 3). The velocity distribution in the boundary layer is shown in Figure 52.

Results in terms of the aerodynamic lift, effective profile drag and moment coefficient $C_L$, $C_{De}$ and $C_M$ are shown from Figure 53 to 56. Effective profile drag coefficient is defined as the difference between the total drag coefficient $C_D$ and the minimum induced drag coefficient $C_{Di}$, that is

$$C_{De} = C_D - \frac{C_I^2}{\pi A}$$

and this to account only for induced drag due to the interference of the wing-body combination.

From the charts it is apparent the validity of the mesh and of the physics model involved, as the configurations at small angle of attack are well simulated by the CFD.
Figure 48: CAD drafting from CATIA – NACA report 540. Dimensions are in inches.
4.1 LONGITUDINAL TEST CASE – WING-BODY INTERFERENCE

(a) Mid wing combination.

(b) High wing combination.

(c) Low wing combination.

Figure 49: Wing-fuselage CAD models for NACA report 540.
4.1 LONGITUDINAL TEST CASE – WING-BODY INTERFERENCE

Figure 50: Model reference system – NACA report 540. For mid wing combination the quarter chord points of fuselage and wing coincide. Figure taken from ref. [9].

Table 10: Mesh and physics data of NACA 540 test case.

<table>
<thead>
<tr>
<th>Mesh type</th>
<th>polyhedral cells (Figure 43)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Base size</td>
<td>50.0 m</td>
</tr>
<tr>
<td>Number of prism layers</td>
<td>20</td>
</tr>
<tr>
<td>Prism layer stretching</td>
<td>1.3</td>
</tr>
<tr>
<td>Numbers of cells</td>
<td>( \approx 1000000 ) (wing-body combination)</td>
</tr>
<tr>
<td>Mesh size (in percentage of the base size)</td>
<td></td>
</tr>
<tr>
<td>Wing</td>
<td>Minimum: 0.02, Target: 0.2, Prism layer: 0.03</td>
</tr>
<tr>
<td>Fuselage</td>
<td>Minimum: 0.02, Target: 0.2, Prism layer: 0.03</td>
</tr>
<tr>
<td>Angle of attack ( \alpha )</td>
<td>from (-4^\circ) to (16^\circ)</td>
</tr>
<tr>
<td>Reynolds number Re</td>
<td>3 100 000 (based on wing chord)</td>
</tr>
<tr>
<td>Mach number M</td>
<td>0</td>
</tr>
<tr>
<td>Flow regime</td>
<td>fully turbulent (Spalart-Allmaras model)</td>
</tr>
</tbody>
</table>
4.1 Longitudinal Test Case – Wing-Body Interference

Figure 51: Mesh of the NACA 540 test case.

Figure 52: Velocity distribution in the prism layer. Negative values are due to the difference in reference systems between the CAD model and Star-CCM+.
Figure 53: Results comparison for the wing alone – NACA report 540.
Figure 54: Results comparison for the mid wing combination – NACA report 540.
Figure 55: Results comparison for the high wing combination – NACA report 540.
4.1 Longitudinal Test Case – Wing-Body Interference

Figure 56: Results comparison for the low wing combination – NACA report 540.
4.2 Directional Test Cases

4.2.1 NACA Report 730

This is the aforementioned report of Bamber and House [5]. A NACA 23012 rectangular wing with rounded tips was tested with a round fuselage at several angles of sideslip, in a high, mid and low wing combination (see Figures 58 and 59 for drafting, Figure 60 for 3D views). Moreover each combination was tested with and without fin. The fin was made to the NACA 0009 section with an area of 45 sq. in. No dihedral angle and no flap device were considered. Reynolds number is 609,000 based on wing chord. Results are evaluated in terms of the rolling moment, yawing moment and sideforce due to sideslip coefficients $C_{L\beta}$, $C_{N\beta}$ and $C_{Y\beta}$. All derivatives are per deg and evaluated assuming linearity with $\beta$ between $0^\circ$ and $5^\circ$. Mesh data is reported in Table 11 and shown in Figure 61.

In Figure 62 it is apparent that the effect of the cross flow for the wing-body configurations without fin has the same effect on the $C_{Y\beta}$ for the high and low wing combinations and a minimum for that coefficient at mid wing position, because of the double symmetry of the model. The wing-fuselage combination without fin is directionally unstable ($C_{N\beta} > 0$). Stability is introduced by the fin (Figure 63). Directional stability, measured by $C_{N\beta}$, is maximum for the low wing combination, as stated in Chapter 2. The rolling moment derivative $C_{L\beta}$ changes sign because of the cross flow over the wing-fuselage system, resulting in an antisymmetric distribution of the normal velocities along the span that is equivalent to an antisymmetric angle of attack distribution [37] (Figure 57). CFD comparison between fin-off and fin-on configuration is shown in Figure 64.
Figure 58: CAD drafting from CATIA – NACA report 730. High wing is translated up by 2.66 in and low wing down by 2.98 in with respect to the mid wing position. Dimensions are in inches. See Figure 5 for the original sketch.
4.2 Directional Test Cases

(a) Reference system and dimensions for wing tips.

(b) CAD views of wing tips.

Figure 59: Wing tips CAD construction for NACA report 730. Screenshots from CATIA.
(a) Mid wing combination.

(b) High wing combination.

(c) Low wing combination.

Figure 60: Wing-fuselage-fin CAD models for NACA report 730.
Table 11: Mesh and physics data of NACA 730 test case.

<table>
<thead>
<tr>
<th>Mesh type</th>
<th>polyhedral cells (Figure 43)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Base size</td>
<td>1.0 m</td>
</tr>
<tr>
<td>Number of prism layers</td>
<td>20</td>
</tr>
<tr>
<td>Prism layer stretching</td>
<td>1.2</td>
</tr>
<tr>
<td>Numbers of cells</td>
<td>≈ 5,000,000 (wing-body-fin combination)</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Mesh size (in percentage of the base size)</th>
<th>Wing</th>
<th>Fuselage</th>
<th>Vertical</th>
</tr>
</thead>
<tbody>
<tr>
<td>Minimum</td>
<td>0.08</td>
<td>0.1</td>
<td>0.05</td>
</tr>
<tr>
<td>Target</td>
<td>2.0</td>
<td>10.0</td>
<td>0.5</td>
</tr>
<tr>
<td>Prism layer</td>
<td>0.2</td>
<td>0.2</td>
<td>0.1</td>
</tr>
</tbody>
</table>

- Angle of attack $\alpha$ \(0^\circ\)
- Angle of sideslip $\beta$ \(5^\circ\)
- Reynolds number Re \(609\,000\) (based on wing chord)
- Mach number M \(0\)
- Flow regime fully turbulent (Spalart-Allmaras model)

Figure 61: Mesh of NACA 730 test case.
Figure 62: Results comparison for the wing-body combination, fin off – NACA report 730.
Figure 63: Results comparison for the wing-body-fin combination – NACA report 730.
Figure 64: CFD results comparison between the fin-off and fin-on configurations – NACA report 730.
The aim of the work of Queijo and Wolhart [34] was to investigate the effects of vertical tail size and span and of fuselage shape and length on the static lateral stability characteristics of a model with 45° swept back (quarter chord line) wing and vertical tail, NACA 65A008 airfoil. As stated in Chapter 1, this report found an interesting relation between the effective aspect ratio of the vertical tail and the fuselage thickness where the tail is located (see Figure 8). This was resumed by the USAF DATCOM method, as seen in Chapter 2 in Figure 22. The original draft is shown in Figure 9. Different fuselages and tails were tested. For the purpose of the test case, only the combination with fuselage no. 4 and tail no. 2 was considered. See Figures 65 and 66 for details. The mesh data is reported in Table 12 and shown in Figure 67. Results are shown in Figures 68-69-70 in terms of force and moment coefficients.

In conclusion, the results obtained with the CFD are close enough to the experimental results, so it appeared reasonable to use Star-CCM+ for the investigation on vertical tailplane design discussed in the next chapter.
Figure 66: CAD drafting from CATIA – NACA report 1049. Dimensions are in inches. See Figure 9 for the original sketch.
Table 12: Mesh and physics data of NACA 1049 test case.

<table>
<thead>
<tr>
<th>Mesh type</th>
<th>polyhedral cells (Figure 43)</th>
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</thead>
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<tr>
<td>Base size</td>
<td>10.0 m</td>
</tr>
<tr>
<td>Number of prism layers</td>
<td>20</td>
</tr>
<tr>
<td>Prism layer stretching</td>
<td>1.3</td>
</tr>
<tr>
<td>Numbers of cells</td>
<td>$\approx 4,000,000$ (wing-body-fin combination)</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Mesh size (in percentage of the base size)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
</tr>
<tr>
<td></td>
</tr>
<tr>
<td>Fuselage</td>
</tr>
<tr>
<td>Vertical</td>
</tr>
</tbody>
</table>

| Angle of attack $\alpha$                  | 0°      |
| Angle of sideslip $\beta$                 | from $-10^\circ$ to $10^\circ$ |
| Reynolds number $Re$                       | 710 000 (based on wing chord) |
| Mach number $M$                            | 0       |
| Flow regime                                | fully turbulent (Spalart-Allmaras model) |

Figure 67: Mesh of NACA 1049 test case.
Figure 68: Results comparison for the fuselage-vertical combination – NACA report 1049.
Figure 69: Results comparison for the fuselage-wing combination – NACA report 1049.
Figure 70: Results comparison for the fuselage-wing-vertical combination – NACA report 1049.
The objective of this chapter is to define a new procedure to calculate the sideforce derivative $C_{Y_\beta v}$ for a vertical stabilizer of general aviation commuter airplanes and civil props airliners. Given the aircraft components (tailplanes, fuselage and wing), the method starts from the lift curve slope $C_{L_{\alpha v}}$ of the isolated vertical tailplane and defines some corrective coefficients, to take into account the effects explained in Chapter 1. Before discussing the method, some introductory remarks on the model and on the numerical setup are necessary, then the CFD analyses will be presented. The new procedure is discussed in Section 5.4.

5.1 CAD MODEL

The aircraft model has the typical characteristics (e.g. fuselage slenderness and wing aspect ratio) of the regional turboprop airplanes, as the ATR-72 and NGTP-5. These parameters are shown in Tables 13-14-15. The wing airfoil is the NACA 23015, while both tailplanes (vertical and horizontal) use the NACA 0012 airfoil.

The model’s dimensions are chosen in order to fit into the wind tunnel (Figure 71) of the Faculty of Engineering of the University of Naples ‘Federico II’. As a matter of fact, to confirm the results of the CFD analyses, future wind tunnel test are necessary. The dimensions of the model are shown in Figures 72 and 73, while the configurations involved are reported in Figure 74. The CAD model with mid-wing and body-mounted horizontal tailplane is shown in Figure 75.

From now to on, the expression ‘CFD model’ indicates the CAD / CFD geometry of the future model to be realized. Additional geometries will be presented in the appropriate sections.

5.2 PRELIMINARY CFD ANALYSES

Some CFD simulations are necessary prior to investigate the vertical tailplane analysis process of Chapter 5. They are about base size, number of CPUs and Reynolds number.

5.2.1 Base size

As stated in Chapter 3, the base size is the reference length of the model to which all cells parameters are related. Changing the base
Figure 71: The wind tunnel of the Faculty of Engineering of the University of Naples ‘Federico II’. All units are in m.

### Table 13: Fuselage parameters

<table>
<thead>
<tr>
<th></th>
<th>fuselage slenderness</th>
<th>nose ratio</th>
<th>cone ratio</th>
<th>wing position</th>
</tr>
</thead>
<tbody>
<tr>
<td>ATR-72</td>
<td>10.3</td>
<td>1.3</td>
<td>3.2</td>
<td>0.41</td>
</tr>
<tr>
<td>NGTP-5</td>
<td>9</td>
<td>1.3</td>
<td>3.3</td>
<td>0.47</td>
</tr>
<tr>
<td>CFD model</td>
<td>9</td>
<td>1.3</td>
<td>3.3</td>
<td>0.45</td>
</tr>
</tbody>
</table>

### Table 14: Vertical tail parameters

<table>
<thead>
<tr>
<th></th>
<th>$A_v$</th>
<th>$\lambda_v$</th>
<th>$\Lambda_{VLE}$</th>
<th>$\Lambda_{VTE}$</th>
<th>$S_v/S$</th>
<th>$x_{VLE}/l_f$</th>
<th>$V_v$</th>
</tr>
</thead>
<tbody>
<tr>
<td>ATR-72</td>
<td>1.56</td>
<td>0.61</td>
<td>32°</td>
<td>17°</td>
<td>0.20</td>
<td>0.83</td>
<td>0.098</td>
</tr>
<tr>
<td>NGTP-5</td>
<td>1.43</td>
<td>0.63</td>
<td>29°</td>
<td>15°</td>
<td>0.24</td>
<td>0.85</td>
<td>0.110</td>
</tr>
<tr>
<td>CFD model</td>
<td>variable</td>
<td>variable</td>
<td>variable</td>
<td>variable</td>
<td>variable</td>
<td>variable</td>
<td>variable</td>
</tr>
</tbody>
</table>

### Table 15: Horizontal tail parameters

<table>
<thead>
<tr>
<th></th>
<th>$A_h$</th>
<th>$\lambda_h$</th>
<th>$\Lambda_{HLE}$</th>
<th>$\Lambda_{HRE}$</th>
<th>$S_h/S$</th>
<th>$x_{HLE}/l_f$</th>
<th>$V_h$</th>
</tr>
</thead>
<tbody>
<tr>
<td>ATR-72</td>
<td>4.1</td>
<td>n.a.</td>
<td>n.a.</td>
<td>n.a.</td>
<td>0.18</td>
<td>n.a.</td>
<td>0.19</td>
</tr>
<tr>
<td>NGTP-5</td>
<td>4.1</td>
<td>n.a.</td>
<td>n.a.</td>
<td>n.a.</td>
<td>0.25</td>
<td>n.a.</td>
<td>0.19</td>
</tr>
<tr>
<td>CFD model</td>
<td>4.1</td>
<td>0</td>
<td>0°</td>
<td>0°</td>
<td>0.20</td>
<td>variable</td>
<td>variable</td>
</tr>
</tbody>
</table>
Figure 72: CFD model draft, generated by CATIA. All units are in m.

Figure 73: The modular vertical tail of the CFD model. Dimensions are in m.
Figure 74: The configurations of the CFD model. For the sake of clarity, here it is only shown the vertical tailplane with $A_v = 2$. Units are in m.
size affects the mesh and hence the solution, as shown in Figure 76. There, it is apparent that the base size of 1.65 m, equal to the fuselage length of the CFD model, is an optimum value and then 10 millions of polyhedral cells are sufficient\(^1\) to obtain a converged solution. The configuration chosen is the complete airplane with high wing and body-mounted tailplane, since this is the most complicated in terms of computational grid (due to the intersections among components), see Figure 76.

5.2.2 CPUs number

The computational time is a non-linear decreasing function with CPUs number, on a linear plot. On a logarithmic plot it is a linear function instead. Figure 77 represents the time necessary to obtain a converged solution vs CPUs number, in the two different scales. These data were obtained running a body-vertical configuration on SCoPE (see Section 3.2). For a complete airplane, it is convenient to operate between 32 and 64 CPUs, to optimize time and computing power. In fact, ref. [39] recommends to employ a CPU every 250 000 cells approx.

5.2.3 Effect of the Reynolds number

The Reynolds number is a non-dimensional ratio between the inertia forces and the viscous forces in a flow [4], defined as

\[
Re = \frac{\rho VL}{\mu}.
\]  

(18)

At high Reynolds number the flow tends to be turbulent, though it does not exist a ‘magic’ number below which the flow is laminar and

\(^1\) As order of magnitude: the use (or the lack) of fine meshed volume shapes in certain configurations alters this value by 1 or 2 millions of cells.
5.2 Preliminary CFD Analyses

(a) The model used for the study on the base size.

(b) Detail of the mesh on the fuselage stern.

(c) Detail of the mesh on the wing-fuselage intersection.

Figure 76: Mesh for the investigation on the optimum base size.
5.2 Preliminary CFD Analyses

(a) Lift coefficient.

(b) Drag coefficient.
Figure 76: Aerodynamic coefficients as function of base size and number of cells for the configuration of Figure 76. It is shown the contribution of each component as well as their sum. \( \alpha = 0^\circ, \beta = 0^\circ, \) \( \text{Re} = 1000000. \)
Figure 77: CPUs scalability for a body-vertical configuration with 1,800,000 polyhedral cells.
Figure 78: Influence of Reynolds number on $C_{Y_v}$. Complete airplane, configuration of Figure 75.

above which the flow is turbulent. In general, a high Reynolds number means high skin-friction drag and low pressure drag (i.e. delayed flow separation). Commercial airplanes have Reynolds number of the order of magnitude as tenth of millions.

In this work, to account for later wind tunnel test, the Reynolds number chosen (based on wing chord) has the same order of magnitude of that of the faculty’s wind tunnel, that is

$$Re \approx 1\,000\,000$$

moreover, the actual investigation only involves cruise conditions (i.e. small angles of sideslip $\beta$) and the difference among Reynolds number up to 10 millions is negligible, as shown in Figure 78.
5.3 CFD ANALYSES

As stated in Chapter 1, accurate determination of static stability derivatives is mandatory to tailplane design. Semi-empirical methods described in Chapter 2 are well established, but based on geometries of the past. These methods have been defined by collecting huge amounts of wind tunnel and flight test data and according them, when possible, to theory. It took years and lots of people to get to the actual results. Computational Fluid Dynamics is the modern approach to achieve such objective, it’s a bridge between theory and experiment. What was an industrial problem in the past can be an ‘academic homework’ today [3]: in fact, few months and few persons were necessary to develop this work. Reynolds Averaged Navier-Stokes equations (RANS) are solved by CFD software like Star-CCM+, described in Chapter 3 and assessed in Chapter 4.

In order to achieve the objective of this work, the CFD analyses were so organized:

1. the lift curve slope of the isolated vertical tailplane has been evaluated and compared to that provided by theory (Sec. 5.3.1);

2. to estimate the effect of the body on the vertical tail, the body-vertical tail configuration has been analyzed and each result of the vertical tail’s sideforce coefficient was scaled by the homologous result of the isolated vertical tail, so as to evaluate the interference effect on the $C_{Y_v}$ (Sec. 5.3.2);

3. similarly it has been done for the wing, by adding it to the previous body-vertical combination and measuring the effects in terms of $C_{Y_v}$ ratios. This has been done for wings of various aspect ratio and position in fuselage (Sec. 5.4.1.3);

4. finally, the effect of the horizontal tailplane position and size was measured as done with the previous effects (Sec. 5.4.1.4).

Almost all analyses, except for the smaller .sim files, were solved on the SCoPE grid infrastructure. The polyhedral mesh and the Spalart-Allmaras turbulence models (see Chapter 3) were chosen to solve the asymmetrical flow field. All of the runs were solved in incompressible flow, $Re = 1000000$, at $\alpha = 0^\circ$ and $\beta = 5^\circ$, since the angle of attack is not relevant for directional stability (see Figure 7) and with a sideslip of $5^\circ$ the effects are linear, that is the aircraft is in cruise condition.

Results are published as ratios of $C_{Y_{...}}$, that is, given a configuration, all the effects of adding a component are scaled by the results of the configuration without that component, e.g. the body effect is evaluated by dividing the $C_{Y_v}$ of the body-vertical configuration by the $C_{Y_v}$ of the isolated vertical tailplane. In this way the effect (increase or decrease of the coefficients) of the aerodynamic interference is high-
Table 16: Mesh and physics data common to all CFD analyses.

<table>
<thead>
<tr>
<th>Mesh type</th>
<th>polyhedral cells (Figure 43)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Base size</td>
<td>1.65 m</td>
</tr>
<tr>
<td>Number of prism layers</td>
<td>20</td>
</tr>
<tr>
<td>Prism layer stretching</td>
<td>1.1</td>
</tr>
<tr>
<td>Block shape size</td>
<td>$60 \times 30 \times 20$ m$^3$ (Figure 79)</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Mesh size (in percentage of the base size)</th>
<th>Minimum</th>
<th>Target</th>
<th>Prism layer</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>0.04</td>
<td>0.5</td>
<td>0.02</td>
</tr>
<tr>
<td>Fuselage</td>
<td>0.1</td>
<td>1.0</td>
<td>0.02</td>
</tr>
<tr>
<td>Horizontal</td>
<td>0.04</td>
<td>0.5</td>
<td>0.02</td>
</tr>
<tr>
<td>Vertical</td>
<td>0.05</td>
<td>0.5</td>
<td>0.01</td>
</tr>
</tbody>
</table>

| Angle of attack $\alpha$ | 0°       |
| Angle of sideslip $\beta$ | 5°       |
| Reynolds number $Re$      | 1,000,000 (based on wing chord) |
| Mach number $M$           | 0        |
| Flow regime               | fully turbulent (Spalart-Allmaras model) |

lighted. In fact, as stated above, the objective of the new procedure is to define corrective coefficients for the $C_{L_{\alpha}}$ of the vertical tailplane.

Here, CFD data are reported on charts as marks, as they were test (indeed they are numerical tests, while experimental tests – wind tunnel – are not available yet), while solid lines represent trends obtained by curve fitting. The best fit chosen is the simplest curve (linear, polynomial, power) that better approximate the numerical results. When possible, the curve equation is directly written in the legend of the chart, otherwise it is reported in the appropriate section.

Derivatives are calculated as an approximation of the incremental ratio

$$C_{Y_{\beta}} \approx \frac{C_{Y_{\beta}} - 0}{5^\circ - 0^\circ} = \frac{C_{Y_{\beta}}}{5} \text{ deg}^{-1}$$

since the linearity is a matter of fact at low sideslip angles (see subsequent sections). Star-CCM+ returns force and moment coefficients, not directly their derivatives, but, for the reason just stated, it is perfectly equivalent. CFD (mesh and physics) data common to all analyses are resumed in Table 16. The mesh is similar of that of Figure 76 and it is shown in Figure 81. A check of the results was done, once residuals were oscillating around $10^{-6}$, by looking at the wall $y^+$ distribution (example in Figure 80), as explained in Chapter 3.
Figure 79: Block shape that defines the fluid domain around the model. Dimensions $60 \times 30 \times 20 \text{ m}^3$.

Figure 80: Example of wall $y^+$ distribution.
The final mesh used on the complete aircraft. 

Transversal section on the vertical tailplane. 

Horizontal and transversal sections.

Figure 81: The final mesh used in CFD analyses. It can be noted the cells fitting around the model and the coarse mesh on the external domain boundaries.
5.3 CFD ANALYSES

5.3.1 Isolated vertical tail

The vertical tail alone has to be analyzed for three reasons: CFD results must verify linearity (Figure 83), be compatible with theory (Figure 84) and provide the lift gradient $C_{L_{a,v}}$. Beyond the vertical tailplane of the CFD model, represented in Figure 73, a rectangular and a constant taper ratio tailplanes were analyzed (Figure 82). Here it is noted that the CFD model component has a constant sweep angle, but a variable taper ratio, and the behavior predicted by CFD is the same of that predicted by theory, as shown in Figure 84. The rectangular tailplane has only a slightly higher trend respect to the swept one and this is in contrast with theory that provides a much higher lift gradient. The constant taper vertical tailplane does not agree with theory simply because it has variable sweep with aspect ratio and hence it should not be compared at all.
Figure 83: Linearity of sideforce coefficient $C_{Y_v}$ vs the angle of sideslip $\beta$. Isolated vertical tailplanes (CFD model), $Re = 1000000$. Here it is 'natural' to show derivatives in deg$^{-1}$.

Figure 84: Lift gradient vs aspect ratio for the isolated vertical tailplanes, computed by Star-CCM+ and compared with theory. Marks represent CFD data while solid lines represent theory. Here, results are in rad$^{-1}$ as often found in literature.
5.3.2 Body effect

The first coupling effect studied was that of fuselage. This effect is measured by the ratio between the sideforce coefficients of the body-vertical configurations of Figure 85 and those of the isolated vertical tails previously analyzed.

Theory [42] and experiment [12, 14] suggest that the fuselage acts as a cylinder at the wing-tip, accelerating the flow and increasing the sideforce on the vertical tail (see also Figures 18 and 19).

Results, in terms of the $C_{Y_v}$ ratios vs $b_v/2r$, are shown in Figure 86. Here $b_v$ is the geometric tail span of the isolated vertical tail, not the one defined by Roskam [36] (represented in Figure 21 as $b_{v1}$), that is there’s no tail span extension to the fuselage centerline. The parameter $2r$ is again the fuselage thickness on the $x$ location of the aerodynamic center of the vertical tailplane. It can be seen a little scatter among different tailplanes of different aspect ratios, sweep angles and taper ratios, even at low $b_v/2r$ ratios (i.e. little tailplanes on a big fuselage), instead of the results of Queijo and Wolhart [34] reported in Figure 8, where there’s a bigger data scatter, especially among tails of different aspect ratio.
Figure 85: Configuration involved in the analysis of the effect of the fuselage: the $C_{Yv}$ of the body-vertical combination is divided by the $C_{Yv}$ of the isolated vertical tail, four vertical tailplanes ($\lambda_v = 0.5, 1.0, 1.5, 2.0$) with one fuselage. $\alpha = 0^\circ$, $\beta = 5^\circ$, $Re = 1000000$. 

Figure 86: Body (fuselage) effect. See Figure 85 for definitions of $b_v$ and $2r$. 

\begin{align*}
C_{Y_{v_{BV}}} / C_{Y_{vV}} &= 1.4685 \left( \frac{b_v}{2r} \right)^{-0.143}
\end{align*}
5.3.3 Wing effect

The objective of these analyses is the evaluation of the effect of wing position and aspect ratio on the sideforce of the vertical tail. In order to achieve this, several rectangular wings, of different aspect ratio, were simulated at high, mid and low position in fuselage. Combinations involved are depicted in Figure 87, while results are shown – in terms $C_{Y_v}$ ratios between configurations – in Figures 88 and 89. The choice of the vertical tailplanes aspect ratios’ $A_v = 1$ and $A_v = 2$ is due to consider two extreme values of this parameter: since the scatter between these values is negligible, it can be argued that the effect of the wing on the $C_{Y_v}$ is independent from the choice of the vertical tailplane. A resume of the results is reported in Figure 89, where the best fit curve has equation

$$
\frac{C_{Y_v,wbv}}{C_{Y_v,bv}} = -0.0131 \left(\frac{z_w}{r_f}\right)^2 - 0.0459 \frac{z_w}{r_f} + 1.0026.
$$

5.3.3.1 High wing

In Figure 88a it is shown that $C_{Y_v}$ decreases with wing aspect ratio $A$ of about 5%, with respect to the body-vertical configuration.

5.3.3.2 Mid wing

Mid wing has no effect, it is neutral from a directional point of view, as reported by Perkins and Hage [33]. Data spread (Figure 88b) is negligible, around 1% of unity.

5.3.3.3 Low wing

Again, $C_{Y_v}$ decreases with wing aspect ratio $A$, probably because of the decreasing magnitude of tip vortices with aspect ratio. In this configuration, however, there’s an increase of about 5% of the sideforce coefficient (Figure 88c), with respect to the body-vertical configuration.
Figure 87: Configurations involved in the analysis of the effect of the wing. The $C_{Y_w}$ of the wing-body-vertical combination is divided by the $C_{Y_v}$ of the body-vertical combination. Three wing position are tested and for each position the aspect ratio $A$ is varied from 6 to 16, with a step of 2. Two vertical tailplanes with $A_v = 1$ and $A_v = 2$ are considered. $\alpha = 0^\circ$, $\beta = 5^\circ$, Re = 1 000 000.
5.3 CFD Analyses

Figure 88: Effect of the wing at various aspect ratio and position, with two vertical tailplanes, for the configurations depicted in Figure 87.
Figure 89: Resume of wing effect. For low aspect ratios add 2%, while for high aspect ratios subtract 2%, both on high and low wing position. For mid wing position all values should be considered as unity, since the scatter is less than 1%.

Figure 90: Pressure coefficient contours, on a transversal section at vertical tail leading edge root chord, seen from behind, for the wing-body-vertical combinations. $\alpha = 0^\circ$, $\beta = 5^\circ$, $Re = 1000000$. Side wind from left to right. Data range constrained for better visualization.
5.3.4 *Horizontal tailplane effect*

The horizontal tailplane influences the vertical tailplane with its position and size. The former is a major effect (as seen in Chapter 1, Figure 19), while the latter has a minor influence, especially for tailplanes of the same surface.

5.3.4.1 *Effect of the tailplane position*

A straight horizontal tailplane, of constant size and aspect ratio (Figure 74, Table 15), is mounted in five positions on the fin and one on the fuselage. Then, the same horizontal tailplane, initially centered in chord, is moved forward and rearward by 25% of the local vertical tailplane chord. This can be seen in Figure 92.

Results are shown in Figure 94, in terms of $C_{Y_v}$ ratios between the wing-body-fin-horizontal configuration (i.e. the complete airplane) and the wing-body-fin configuration, as shown in Figure 91. The trend is much similar to that of Figure 14 by Brewer and Lichtenstein [6] instead of Figure 23 by Roskam [36] (reproduced from Finck [12]), that is the effect of positioning the horizontal tailplane on the fuselage and on the free tip of the vertical tailplane is almost the same. The best fit curves in Figure 94 are second-order polynomials

$$
\frac{C_{Y_{v, WBH}}}{C_{Y_{v, WBV}}} = \begin{cases} 
0.6891 \left( \frac{z_h}{b_{v1}} \right)^2 - 0.6703 \frac{z_h}{b_{v1}} + 1.1296 & x/c_v = 0 \\
0.6502 \left( \frac{z_h}{b_{v1}} \right)^2 - 0.5687 \frac{z_h}{b_{v1}} + 1.0714 & x/c_v = -0.25 \\
0.6864 \left( \frac{z_h}{b_{v1}} \right)^2 - 0.6796 \frac{z_h}{b_{v1}} + 1.139 & x/c_c = +0.25 
\end{cases}
$$

(21)

It is apparent that the screening effect of the horizontal tailplane is maximum at extreme positions and decreases moving it forward.

5.3.4.2 *Effect of the tailplane size*

The effect of tailplanes’ relative size is shown in Figure 95, where it is represented the ratio $C_{Y_v}$ between the complete airplane with the horizontal tail varying in size and the previous configuration with a ratio $S_h/S_v$ close to unity (the original horizontal tailplane for later wind tunnel test). Two extreme positions were analyzed, that is the body-mounted and tip-mounted.
Figure 91: Horizontal tailplanes’ configuration. The $C_{Y_{v}}$ of the complete airplane (wing-body-vertical-horizontal) is divided by the $C_{Y_{v}}$ of the wing-body-vertical combination. Only mid wing with $A = 10$ and the vertical tailplane with $A_{v} = 2$ are considered. Mid points of tailplanes’ chords are coincident. $\alpha = 0^\circ$, $\beta = 5^\circ$, $Re = 1\times10^{6}$.

Figure 92: Translation of the horizontal tailplane in chord. For each position on the vertical tailplane span depicted in Figure 91, the horizontal tailplane is moved forward and rearward by 25% of the local vertical tailplane chord from the middle point.
Figure 93: Configurations involved in the analysis of the relative size of the tailplanes. The reference model has $S_h/S_v = 0.9644$. For the sake of clarity, only the body-mounted tailplanes are shown.
Figure 94: Horizontal tailplane position effect. The length $b_{v1}$ is the vertical tail span extended to the horizontal tailplane position in fuselage, represented, with tail shift in chord, in Figure 92. This ‘span stretch’ is mandatory if a unique chart with all possible tailplane’s positions is desired. $\alpha = 0^\circ$, $\beta = 5^\circ$, $Re = 1\,000\,000$.

Figure 95: Effect of the relative size of tailplanes. It is negligible at a first approximation, since most airplanes have $S_h/S_v$ ratios close to unity.
5.4 DEFINING A NEW PROCEDURE

The most known procedure to calculate the stability derivatives of an aircraft and its components is perhaps the **USAF DATCOM** method, largely discussed in Chapter 2. Here it is stated again that it is based on geometries quite different from today airplanes and it is remarked that it requires the computation of several parameters (e.g. the effective aspect ratio and other semi-empirical factors) before the calculation of the stability derivatives; moreover it is not always clear what is the vertical tail span once it has been mounted on the fuselage (Figure 20) and it is an onerous procedure in the case of hand calculations.

The **ESDU** method instead is quite simpler to remember and execute, but it starts from different definitions, so that it is impossible to compare partial configurations; moreover it considers a constant sidewash, that is it does not provide differences as the wing aspect ratio changes (see Chapter 2).

The method proposed in this section has the following qualities:

1. it has been developed from actual airplanes’ geometries (turbo-props);
2. it is easy to remember and to apply, even in hand calculations, since only few factors are used and no other parameters than the aircraft’s dimensions have to be considered;
3. it ‘was born’ from the **CFD** and so it can be easily extended to other configurations.

So, it is a native method for regional transport airliners, that employs the modern techniques of numerical investigation to place alongside a small number of wind tunnel tests, easy to remember and to apply.

5.4.1 The **DIAS** method

On the basis of the previous investigation, the following procedure, named **DIAS** method as the department where it was developed, is proposed. The vertical tailplane sideforce derivative in sideslip is calculated as

\[
C_{Y_{\beta_v}} = K_F K_W K_H C_{L_{a_v}} \frac{S_v}{S} \tag{22}
\]

where

- \(C_{L_{a_v}}\) is the lift curve slope of the isolated vertical tailplane (Sec. 5.4.1.1)
- \(K_F\) is the effect of the fuselage (Sec. 5.4.1.2)
- \(K_W\) is the effect of the wing (Sec. 5.4.1.3)
The meanings of these corrective factors is the same of the $C_{\gamma_v}$ ratios discussed in the previous section, that is $K_F$ is the effect on $C_{\gamma_{\beta_v}}$ of adding the fuselage to the isolated vertical tail, $K_W$ is the effect of adding the wing to the body-vertical combination, and $K_H$ is the effect of adding the horizontal tailplane to the wing-body-vertical combination.

### 5.4.1.1 Isolated vertical tailplane lift curve slope

The vertical stabilizer alone is a wing, whose lift gradient can be expressed by the Helmbold-Diederich formula (7), here rewritten

$$C_{L_{\alpha_v}} = \frac{2\pi A_v}{2 + \left[ \frac{B^2 A^2_v}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{\gamma_{\beta_v}/2}}{B^2} \right) + 4 \right]^2}$$

where

- $A_v$ is the aspect ratio, $b_v^2/S_v$
- $B$ is a compressibility parameter, $\sqrt{1-M^2}$
- $\kappa$ is the ratio of section lift-curve slope to theoretical thin-section value $c_{1_{\alpha}}/(2\pi/B)$, and for thin airfoil ($c_{1_{\alpha}} \approx 2\pi$) it is equal to $B$
- $\Lambda_{\gamma_{\beta_v}/2}$ is the sweep angle at half chord (Figure 96).

The product $C_{L_{\alpha_v}} S_v/S$ is the $C_{\gamma_{\beta_v}}$ of the isolated vertical tail.

![Figure 96: Definition of the isolated vertical tail.](image_url)

---

2 It has been verified that there’s no difference in $K_H$ when not considering the wing, that is the $K_H$ calculated on the wing-body-vertical-horizontal combination is the same of that calculated on the body-vertical-horizontal combination.
5.4.1.2 *Fuselage correction factor*

The factor $K_F$ is function of the ratio between the vertical tailplane span $b_v$ and the fuselage thickness $2r$. These parameters are defined in Figure 97. The curve has equation

$$K_F = 1.4685 \left( \frac{b_v}{2r} \right)^{-0.143}$$

and it is plotted in Figure 98. In the event of vertical tailplane submerged in fuselage, tail span $b_v$ is the longest vertical distance on wetted surface from tail root to tail tip.

![Figure 97: Definitions of $b_v$ and $2r$ for the DIAS method.](image)

![Figure 98: Fuselage correction factor.](image)
5.4.1.3 Wing correction factor

The factor $K_W$ is function of wing aspect ratio $A$ and position in fuselage $z_w/r_f$. This parameter is defined in Figure 99. The curve has equation (for $A$ close to 10)

$$K_W = -0.0131 \left( \frac{z_w}{r_f} \right)^2 - 0.0459 \frac{z_w}{r_f} + 1.0026$$

(24)

and it is plotted in Figure 100.

Figure 99: Definition of $z_w/r_f$ for the DIAS method. If the fuselage has a non-circular section $r_f$ is half fuselage height.

Figure 100: Wing correction factor. Between the high and low aspect ratios it is supposed a linear variation of the curve of eq. (20) with a maximum difference of 2% (see also Figure 89).
5.4 Defining a New Procedure

5.4.1.4 Horizontal tailplane correction factor

The factor $K_H$ is function of other coefficients that account for of the horizontal tailplane position $K_{Hp}$ and size $K_{Hs}$, related in the following formula

$$K_H = 1 + K_{Hs} (K_{Hp} - 1). \quad (25)$$

The factor $K_{Hp}$ is function of the relative position between the horizontal and the vertical tailplanes $z_h/b_{v1}$, where this non-dimensional parameter is computed from the position of the tailplane in fuselage, as shown in Figure 101. The curves have the following expressions

$$K_{Hp} = 0.6891 \left( \frac{z_h}{b_{v1}} \right)^2 - 0.6703 \frac{z_h}{b_{v1}} + 1.1296 \quad x/c_v = 0 \quad (26a)$$

$$K_{Hp} = 0.6502 \left( \frac{z_h}{b_{v1}} \right)^2 - 0.5687 \frac{z_h}{b_{v1}} + 1.0714 \quad x/c_v = -0.25 \quad (26b)$$

$$K_{Hp} = 0.6864 \left( \frac{z_h}{b_{v1}} \right)^2 - 0.6796 \frac{z_h}{b_{v1}} + 1.139 \quad x/c_v = +0.25 \quad (26c)$$

and are plotted in Figure 102.

The factor $K_{Hs}$ is function of the relative size of the tailplanes $S_h/S_v$. It has equation

$$K_{Hs} = 0.9987 \left( \frac{S_h}{S_v} \right)^{0.0357} \quad (27)$$

and it is plotted in Figure 104. If the horizontal tailplane has a planform area very close to that of the vertical tailplane, the size contribution can be neglected ($K_{Hs} \approx 1$ as shown in Figure 104), thus $K_H \approx K_{Hp}$. 

Figure 101: Definition of $z_h/b_v1$ and $x/c_v$ for the DIAS method. The body-mounted horizontal tailplane has the mid-chord point on the fuselage centerline, coincident with the projection of the mid-chord point of the vertical tail root on the same centerline.

Figure 102: Horizontal tailplane position correction factor.
(a) Variation of $S_h$. 

(b) $S_v$ for $A_v = 2$.

Figure 103: Definition of $S_h/S_v$ for the DIAS method.

$$K_S = 0.9987 \left( \frac{S_h}{S_v} \right)^{0.0357}$$

Figure 104: Horizontal tailplane size correction factor. For surface ratios close to unity it can be neglected and hence, from eq. (25), $K_{Hp} \approx K_H$. 
5.4.2 Examples

5.4.2.1 Application to a regional turboprop airplane

In this section, the DIAS method is applied to the ATR-42. Aircraft data and results of USAF DATCOM and ESDU methods are available in Section 2.4. The $C_{L_{\alpha \nu}}$ is given in eq. (7)

$$C_{L_{\alpha \nu}} = \frac{2\pi A_{\nu}}{2 + \left[ \frac{B^2 A_{\nu}^2}{k^2} \left( 1 + \frac{\tan^2 \Lambda_{\nu/2}}{B^2} \right) + 4 \right]^{1/2}} = 2.240 \text{ rad}^{-1}.$$

The fuselage corrective factor is taken from Figure 98 or calculated with the curve expression (23)

$$K_F = 1.4685 \left( \frac{b_{\nu}}{2r} \right)^{-0.143} = 1.168$$

where $b_{\nu}$ is given in Table 5 and $2r$ is estimated as 0.91 m, so that $b_{\nu}/2r = 4.95$.

The wing corrective factor is taken from Figure 100 or calculated with the curve expression (24)

$$K_W = -0.0131 \left( \frac{z_{W}}{\tau_f} \right)^2 - 0.0459 \frac{z_{W}}{\tau_f} + 1.0026 = 0.944$$

where $z_{W}/\tau_f = 1$ because the ATR-42 is a high wing airplane. It has a wing aspect ratio $\Lambda = 12$, yet the above formula is the expression for $\Lambda = 10$. However, the curve of the former is close to the curve of the latter that it is sufficient to approximate $K_W = 0.940$ without interpolating the curves.
The horizontal tail position corrective factor is taken from Figure 102 or calculated with the curve expression (26a) (the tail is centered on vertical chord)

\[ K_{Hp} = 0.6891 \left( \frac{z_h}{b_{v1}} \right)^2 - 0.6703 \frac{z_h}{b_{v1}} + 1.1296 = 1.051 \]

where \( b_{v1} \) is estimated as 4.96 m. The conversion of \( z_h/b_v \) of Table 5 to the parameter \( z_h/b_{v1} \) of Figure 101 is given by the following expression

\[ \frac{z_h}{b_{v1}} = \left( \frac{z_h}{b_v} - 1 \right) \frac{b_v}{b_{v1}} + 1 = 0.837. \]

From Table 5 the ratio \( S_h/S_v = 0.906 \) so that the corrective factor for the horizontal tail size can be evaluated from Figure 104 or calculated with the curve expression (27)

\[ K_{Hs} = 0.9987 \left( \frac{S_h}{S_v} \right)^{0.0357} = 0.995 \]

that is, unity.
The factor $K_H$ is calculated with eq. (25)

\[ K_H = 1 + K_{Hs} (K_{Hp} - 1) = 1.045 = K_{Hp} \]

being $K_{Hs} \approx 1$.

Finally, the corrected $C_{Y_{\beta_v}}$ is given by eq. (22), that is

\[ C_{Y_{\beta_v}} = K_F K_W K_H C_{L_{\alpha v}} \frac{S_v}{S} \]

where the $C_{Y_{\beta_v}}$ of the isolated tail is

\[ C_{L_{\alpha v}} \frac{S_v}{S} = 2.240 \cdot \frac{12.7}{50} = 0.569 \text{ rad}^{-1} \]

and the global effect of the coupling of fuselage, wing and horizontal tail is given by

\[ K_F K_W K_H = 1.168 \cdot 0.944 \cdot 1.051 = 1.159 \]

that is an increase of about 16%, so that

\[ C_{Y_{\beta_v}} = 0.569 \cdot 1.159 = 0.659 \text{ rad}^{-1}. \]

<table>
<thead>
<tr>
<th>Results comparison and deviation from USAF DATCOM</th>
<th>USAF DATCOM</th>
<th>ESDU</th>
<th>DIAS</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{Y_{\beta_v}}$ (rad$^{-1}$)</td>
<td>0.669</td>
<td>0.642</td>
<td>0.659</td>
</tr>
<tr>
<td>$% \Delta$</td>
<td>—</td>
<td>4.0</td>
<td>2.0</td>
</tr>
</tbody>
</table>

The difference between USAF DATCOM and the new DIAS procedure is only 2%, a remarkable result.
5.4.2.2 Application to a CFD case

A further comparison can be made among the three methods and the results on the CFD model. The configuration chosen is that of Figure 75, whose data is resumed in the following table.

<table>
<thead>
<tr>
<th>Geometric data</th>
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<tbody>
<tr>
<td>Wing position</td>
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<tr>
<td>Wing aspect ratio</td>
</tr>
<tr>
<td>Wing area</td>
</tr>
<tr>
<td>Vertical tail aspect ratio</td>
</tr>
<tr>
<td>Vertical tail span</td>
</tr>
<tr>
<td>Vertical tail area</td>
</tr>
<tr>
<td>Horizontal tail position</td>
</tr>
<tr>
<td>Horizontal tail area</td>
</tr>
<tr>
<td>Fuselage thickness on tail</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Derived data</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tail to wing surface ratio</td>
</tr>
<tr>
<td>Tail to body ratio</td>
</tr>
<tr>
<td>Tail surfaces ratio</td>
</tr>
</tbody>
</table>

The Helmbold-Diederich formula eq. (7) yields to

\[ C_{\alpha, v} = 2.600 \text{ rad}^{-1}. \]

The DIAS method gives

\[ K_F = 1.4685 \cdot 3.83^{-0.143} = 1.212 \]
\[ K_W = 1.0026 \approx 1 \]
\[ K_{H_p} = 1.130 \quad K_{H_s} = 0.9987 \cdot 0.9644^{0.0357} = 0.997 \approx 1 \]
\[ K_H \approx K_{H_p} = 1.116 \]

hence

\[ C_{\alpha, v} \frac{S_v}{S} = 2.600 \cdot 0.207 = 0.538 \text{ rad}^{-1} \]
\[ K_F \cdot K_W \cdot K_H = 1.212 \cdot 1 \cdot 1.130 = 1.370 \]
\[ C_{\beta, v} = 0.538 \cdot 1.370 = 0.737 \text{ rad}^{-1}. \]

<table>
<thead>
<tr>
<th>Results comparison and deviation from CFD result</th>
</tr>
</thead>
<tbody>
<tr>
<td>CFD</td>
</tr>
<tr>
<td>C_{\beta, v} (rad(^{-1}))</td>
</tr>
<tr>
<td>% Δ</td>
</tr>
</tbody>
</table>
5.4.2.3 Application to a general aviation airplane

The aircraft is a Tecnam P2012, a twin piston, eleven seat, multirole.

All units are in m.
### Geometric data

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing position $z_w/r_f$</td>
<td>1 (high)</td>
</tr>
<tr>
<td>Wing aspect ratio $A$</td>
<td>7.870</td>
</tr>
<tr>
<td>Wing area $S$</td>
<td>24.92 m²</td>
</tr>
<tr>
<td>Vertical tail aspect ratio $A_v$</td>
<td>1.565</td>
</tr>
<tr>
<td>Vertical tail span $b_v$</td>
<td>2.176 m</td>
</tr>
<tr>
<td>Vertical tail area $S_v$</td>
<td>3.026 m²</td>
</tr>
<tr>
<td>Horizontal tail position $z_h/b_v$</td>
<td>0 (body-mounted)</td>
</tr>
<tr>
<td>Horizontal tail area $S_h$</td>
<td>5.676 m²</td>
</tr>
<tr>
<td>Fuselage thickness on tail $2r$</td>
<td>0.7481 m</td>
</tr>
</tbody>
</table>

### Derived data

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tail to wing surface ratio $S_v/S$</td>
<td>0.121</td>
</tr>
<tr>
<td>Tail to body ratio $b_v/2r$</td>
<td>2.909</td>
</tr>
<tr>
<td>Tail surfaces ratio $S_h/S_v$</td>
<td>1.880</td>
</tr>
</tbody>
</table>

The Helmbold-Diederich formula eq. (7) yields to

$$C_{L_{\alpha v}} = 2.160 \text{ rad}^{-1}.$$  

The DIAS method gives

$$K_F = 1.4685 \cdot 2.909^{-0.143} = 1.260$$
$$K_W = 0.953$$
$$K_{Hp} = 1.130$$
$$K_{Hs} = 0.9987 \cdot 1.88^{0.0357} = 1.022$$
$$K_H = 1 + 1.022 \cdot (1.130 - 1) = 1.133$$

hence

$$C_{L_{\alpha v}} \frac{S_v}{S} = 2.160 \cdot 0.121 = 0.261$$
$$K_F \cdot K_W \cdot K_H = 1.260 \cdot 0.953 \cdot 1.133 = 1.361$$
$$C_{Y_{\beta v}} = 1.361 \cdot 0.261 = 0.355 \text{ rad}^{-1}.$$  

### Results comparison

<table>
<thead>
<tr>
<th></th>
<th>DIAS</th>
<th>USAF DATCOM</th>
<th>ESDU</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{Y_{\beta v}}$ (rad$^{-1}$)</td>
<td>0.355</td>
<td>0.249</td>
<td>0.410</td>
</tr>
</tbody>
</table>

In this case no experimental data is available and the scatter is bigger. However, this is expected for horizontal stabilizers mounted on the fuselage, as shown in Figure 36. The DIAS method is placed among the other two, a sort of average.
The objective of this work was to implement a new preliminary design procedure for the airplane’s vertical stabilizer, through the Computational Fluid Dynamics. As a matter of fact, tailplane’s design necessarily passes through the determination of the static stability derivatives and the more accurate the latter, the better the former. At time of writing, two semi-empirical methods exist for the determination of the stability derivatives. Both of them correct the lift curve slope of the isolated vertical tailplane by means of semi-empirical factors to account for the effect of the fuselage (body), wing and horizontal stabilizer. They are different in how these corrective coefficients act on the lift gradient.

The first of these methods, perhaps the most known, is provided by USAF DATCOM, that defines an effective aspect ratio to substitute to the geometric aspect ratio of the vertical tailplane in the lift gradient formula. This quantity is function of the ratio of the vertical tail span to the fuselage thickness and horizontal tailplane position and size. This corrected lift curve slope is then modified again by the sidewash effect, due to wing aspect ratio and position in fuselage. In this method, it is not always clear what is the vertical tailplane geometry in fuselage and hence which value of the abscissa consider in some charts.

The other method is proposed in Europe by ESDU, more recent and easier to apply, since it defines three coefficients (body, wing and tailplane correction factors) that multiply the vertical tail lift gradient. There’s no effective aspect ratio, but the definition of the geometric aspect ratio is different from the american literature. Moreover it is valid for almost circular fuselages and does not account for sidewash variations (i.e. wing aspect ratio). Both methods are based on NACA reports of the first half of the ‘900, though ESDU makes also use of some unpublished data by aircraft industries. So both of them refer to the airplane’s shape of the past and tries to link potential flow theory to wind tunnel test, even for high scattered data, as largely shown in the introduction of this work.

These methods have been compared with a MATLAB script that provides stand-alone analyses as well as parametric studies. It appeared that semi-empirical methods give the same results for certain configurations, as the high-winged, T-tail turboprop, but they are less accurate for those configurations (both airliners and commuters) with the horizontal stabilizer mounted in fuselage.

Nowadays computing power permits complex CFD analyses both on desktop computers and cluster grids. For the purpose of this work,
the commercial software Star-CCM+ was used, both in local and on
the SCoPE University’s grid. Several test cases, excerpted from the
same NACA reports to which semi-empirical methods are based, were
simulated to verify the compliance of CFD results to the experimental
data and the results were comforting.

After this compliance, the analyses for the definition of the new
procedure, objective of this work, were planned. The geometry was
chosen with the characteristics (e.g. fuselage slenderness, wing as-
pect ratio) derived from two turboprop airliners, while the dimen-
sions of the CAD model were dictated by the size of the wind tunnel
certainty, to account for future wind tunnel tests. The
simulations files were organized to provide the mutual interference
among the airplane components. For example, the effects of the fuse-
lage (body) on the vertical tailplane’s sideforce coefficient was mea-
sured by dividing the sideforce coefficient of the vertical tailplane
of the body-vertical configuration by the sideforce coefficient of the
isolated tailplane, in the same flow conditions and mesh parameters.
Results of these analyses are published as sideforce coefficients ratios
among the configurations and best fit curves are provided to mathe-
atically describe the trend exhibited by numerical data.

The new procedure, named DIAS method, was developed on these
results (110 runs, not to mention other unpublished data). The ap-
proach is like that of the ESDU method, simple and easy to remember,
but it has the same definitions found in literature, it is based on ac-
tual geometries, it has low scattered data than those found in the
USAF DATCOM method, it provides effects for body, wing aspect ratio
and position, horizontal tailplane and size. Once defined the airplane
configuration and main dimensions, the user has to calculate four
coefficients (three if both horizontal and vertical tailplanes have close
planform areas) by looking at charts or using the curve equations and
multiply them by the lift curve slope of the vertical tailplane.

Three examples show how accurate is the DIAS method compared
with USAF DATCOM and ESDU: the regional turboprop ATR-42, a CFD
run and the Tecnam P2012 commuter. For the ATR-42 this new pro-
cedure differs only by 2% from the other methods, it is exact for the
CFD configuration from which it was derived (though the scatter with
the other methods is higher) and all the methods provide highly dif-
ferent results for the P2012. However this was expected, since it was
known, from the parametric studies done with the MATLAB script,
that the methods differ a lot for body-mounted tailplanes. Even in
this case, the DIAS method constitutes a sort of average among the
other two. So, it can be stated that, even lacking the experimental
data, this method, easier to apply even in hand calculations, provide
good results for props airliners and can be easily extended to other
aircraft types, through the feasibility of modern CFD.
A.1 SEC. 25.171 GENERAL

The airplane must be longitudinally, directionally, and laterally stable in accordance with the provisions of Secs. 25.173 through 25.177. In addition, suitable stability and control feel (static stability) is required in any condition normally encountered in service, if flight tests show it is necessary for safe operation.


A.2 SEC. 25.177 STATIC LATERAL-DIRECTIONAL STABILITY

A. Reserved.

B. Reserved.

c. In straight, steady sideslips, the aileron and rudder control movements and forces must be substantially proportional to the angle of sideslip in a stable sense; and the factor of proportionality must lie between limits found necessary for safe operation throughout the range of sideslip angles appropriate to the operation of the airplane. At greater angles, up to the angle at which full rudder is used or a rudder force of 180 pounds is obtained, the rudder pedal forces may not reverse; and increased rudder deflection must be needed for increased angles of sideslip. Compliance with this paragraph must be demonstrated for all landing gear and flap positions and symmetrical power conditions at speeds from 1.2 VS1 to VFE, VLE, or VFC/MFC, as appropriate.

d. The rudder gradients must meet the requirements of paragraph c at speeds between VMO/MMO and VFC/MFC except that the dihedral effect (aileron deflection opposite the corresponding rudder input) may be negative provided the divergence is gradual, easily recognized, and easily controlled by the pilot.

Summary: These amendments to the Federal Aviation Regulations update the standards for type certification of transport category airplanes for clarity and accuracy, and ensure that the standards are appropriate and practicable for the smaller transport category airplanes common to regional air carrier operation.

Effective date: August 20, 1990.

**A.3 sec. 25.181 dynamic stability**

A. Any short period oscillation, not including combined lateral-directional oscillations, occurring between 1.2 vs and maximum allowable speed appropriate to the configuration of the airplane must be heavily damped with the primary controls –

a) Free; and

b) In a fixed position.

B. Any combined lateral-directional oscillations (‘Dutch roll’) occurring between 1.2 vs and maximum allowable speed appropriate to the configuration of the airplane must be positively damped with controls free, and must be controllable with normal use of the primary controls without requiring exceptional pilot skill.
In Section 2.5.6 it was stated that the effect of the body in the MATLAB script VeTARE.m is treated by changing the ratio \(b_v/2r\).

A change in \(b_v\) alone would result in a parabolic trend of \(C_{Y\beta_v}\) because of the change of the ratio \(S_v/S\). This is shown in Figures 105 and 106, analytically expressed by eq. (9) neglecting the sidewash effect

\[
C_{Y\beta_v} = -k_v C_{L_{\alpha_v}} \left( \frac{A_{v(f)}}{A_v} \right) \frac{S_v}{S} \tag{28}
\]

(the parentheses in the above equation remark that \(C_{L_{\alpha_v}}\) is function of \(A_{v(f)}/A_v\), it’s not a product). To obtain a curve similar to that of Figure 38 it is sufficient to divide the previous equation by \(S_v/S\), resulting in Figure 107. The variation of the functions with which eq. (28) is made up are shown in Figures 108 and 109. This has been done for the USAF DATCOM method only.

Figure 105: Vertical tail \(b_v\) change. The root chord trailing edge is fixed, while the aspect ratio is held constant (\(A_v = 1\)) and the surface \(S_v\) changes with \(b_v^2\). The vertical segments on the fuselage indicate the fuselage thickness \(2r\). At each color of the tails corresponds the same color on the fuselage.
Figure 106: Effect of the change of $b_\nu$. The term $b_\nu^2$ dominates in eq. 28 and hence the curve becomes a parabola.

Figure 107: Effect of the change of $b_\nu$, scaled by $S_\nu/S$. Here the trend is the same of Figure 38.
Figure 108: Variation of $C_{L_{\alpha_{V}}}$ with $b_{V}/2r$. This is the lift gradient of the vertical tail according to USAF DATCOM.

Figure 109: Functions components’ variation of eq. (28) with $b_{V}/2r$. 
In this appendix it is described how to run the software Star-CCM+ on the SCoPE grid infrastructure:

1. writing the files necessary to the job;
2. transferring these files on SCoPE;
3. access to the Local File Catalogue (LFC);
4. running, monitoring and retrieving the job.

It is necessary to have an internet connection, a software to transfer files by Secure File Transfer Protocol (SFTP) (e.g. WinSCP or Filezilla) and a Secure Shell (SSH) client (e.g. Putty or simply a unix terminal). The user guide is an excerpt of the manual written by Eng. Elia Daniele of the DIAS.

C.1 Files necessary to the job

In order to use the Star-CCM+ library on SCoPE, four files are necessary:

- an input file, filename.sim, the same that runs locally;
- a Java macro, macro_filename.java, containing the instructions for Star-CCM+ and at least an autosave statement at the end of the iterations;
- an executable unix file, filename.sh, containing the instructions for SCoPE;
- a job file, filename.jdl, calling the input .sim, naming the output, assigning the CPUs number, calling the executable .sh file and, if any, some utilities.

C.1.1 Java macro example

```
// STAR-CCM+ macro: macro_wbvh_av2_h1_5beta.java
package macro;

import java.util.*;

import star.common.*;
import star.base.neo.*;
```
C.1 files necessary to the job

```java
import star.meshing.*;

public class macro_wbvh_av2_h1_5beta extends StarMacro {

    public void execute() {
        execute0();
    }

    private void execute0() {

        Simulation simulation_0 =
            getActiveSimulation();

        MeshContinuum meshContinuum_0 =
            ((MeshContinuum) simulation_0.getContinuumManager().
                    getContinuum("Mesh 1"));

        MeshPipelineController meshPipelineController_0 =
            simulation_0.get(MeshPipelineController.class);

        meshPipelineController_0.generateVolumeMesh();

        simulation_0.getSimulationIterator().run();

        simulation_0.saveState(resolvePath("wbvh_av2_h1_5beta.sim"));
    }
}
```

C.1.2 Executable file example

```bash
#!/bin/sh
#
# this parameter is the number of CPU's to be reserved for
# parallel
# execution
CPU_NEEDED=$1
SIMFILE=$2

echo "Copy SIMFILE on Local File Catalogue for size in excess of
100 MB
> lcg-cp lfn:/grid/unina.it/ademarcoDIR/Cucco/run/
tsr2/b3/$SIMFILE file:$PWD/$SIMFILE"

HOST_NODEFILE=$PBS_NODEFILE

# copy files on the nodes different from the first one
NPROCTOCOPY='expr $CPU_NEEDED - 1'
FIRSTPROC='head -1 $HOST_NODEFILE'
j=0
for i in 'tail -n $NPROCTOCOPY $HOST_NODEFILE | sort -u' ; do
    echo "Copy SIMFILE on Local File Catalogue for size in excess of
    100 MB
    > lcg-cp lfn:/grid/unina.it/pdellavecchiaDIR/Pierluigi/wt_test/wbvh/
    $SIMFILE file:$PWD/$SIMFILE"
    HOST_NODEFILE=$PBS_NODEFILE
    # creates the working directories on all the nodes allocated for
```
# parallel execution
if [ "$i" != "$FIRSTPROC" ]
then
    WORKING_DIR='pwd'
    DIR_EXISTS[$j]='ssh $i "if [ -f "WORKING_DIR/starccm.sh" ]; then echo SI; else echo NO;fi"';
    if [ ${DIR_EXISTS[$j]} = "NO" ]
    then
        echo "Working directory on node $i doesn’t exist ... Creating"
        # copies the needed files on all the nodes allocated for parallel execution
        /usr/bin/scp -rp $WORKING_DIR/$SIMFILE $i :$WORKING_DIR
    else
        echo "Working directory exists on node $i"
        fi
    fi
fi
j='expr $j + 1'
done

# this is the demo molecule
export STARCCMEXE=/opt/exp_soft/unina.it/STAR-CCM+/STAR-CCM
+6.06.011/star/bin/starccm+

if [ -f /home/$LOGNAME/.flexlmrc ]
then
cp /home/$LOGNAME/.flexlmrc $HOME
fi

echo "starting parallel starccm on nodes ..."
cat $HOST_NODEFILE

echo "Indication of date for calculation start ..."
date
echo "Executing $STARCCMEXE -machinefile $HOST_NODEFILE -rsh ssh -np $CPU_NEEDED -batch macro_tsr2_b3t3.java $PWD/$SIMFILE"
$STARCCMEXE -machinefile $HOST_NODEFILE -rsh ssh -np $CPU_NEEDED -batch macro_wbvh_av2_h1_5beta.java $PWD/$SIMFILE

echo "Indication of date for calculation end ..."
date

j=0
for i in `tail -n $NPROCTOCOPY $HOST_NODEFILE | sort -u`; do
    # remove created directory on all the nodes allocated for parallel execution
    if [ "$i" != "$FIRSTPROC" ]
    then
        WORKING_DIR='pwd'
        DIR_EXISTS[$j]='ssh $i "if [ -f "WORKING_DIR/starccm.sh" ]; then echo SI; else echo NO;fi"';
        if [ ${DIR_EXISTS[$j]} = "NO" ]
        then
            echo "Working directory on node $i doesn’t exist ... Creating"
            # copies the needed files on all the nodes allocated for parallel execution
            /usr/bin/scp -rp $WORKING_DIR/$SIMFILE $i :$WORKING_DIR
        else
            echo "Working directory exists on node $i"
            fi
        fi
    fi
    j='expr $j + 1'
done
C.2 transferring files on scope

The previously mentioned files must be copied on the user’s personal directory on SCoPE. To do this, it is necessary a SFTP software as WinSCP, Filezilla or others, connecting to the address

ui01.scope.unina.it

with the credentials provided after the registration. More info are available on www.scope.unina.it.

```bash
if [ $(DIR_EXISTS[$j]) = 'NO' ]
then
  echo "Deleting working directory on node $i"
  /usr/bin/ssh $i rm -rf $WORKING_DIR
else
  echo "Not needed to delete working directory on node $i"
fi
j='expr $j + 1'
done
```

### C.1.3 Job file example

Type = "Job";
JobType = "MPICH";

Executable = "wbvh_av2_h1_sbeta.sh";
Arguments = "16 wbvh_av2_h1_sbeta.sim";
CpuNumber = 16;

StdOutput = "starccm.log";
StdError = "starccm.err";

Requirements = RegExp("ce0[1-2].scope.unina.it:8443/cream-pbs-uninahcp",other.GlueCEUniqueID);
InputSandbox = {"wbvh_av2_h1_sbeta.sh","macro_wbvh_av2_h1_sbeta.java"};
OutputSandbox = {"starccm.log","starccm.err","wbvh_av2_h1_sbeta.sim"};

PerusalFileEnable = true;
PerusalTimeInterval = 15000;
RetryCount = 0;

C.2 TRANSFERRING FILES ON SCOPE

The previously mentioned files must be copied on the user’s personal directory on SCoPE. To do this, it is necessary a SFTP software as WinSCP, Filezilla or others, connecting to the address

ui01.scope.unina.it

with the credentials provided after the registration. More info are available on www.scope.unina.it.
C.3 copying the simulation file on the LFC

First of all, a SSH tunnel must be opened on ui01.scope.unina.it. This can be done with a unix terminal with the command

```
ssh -l username ui01.scope.unina.it
```

or by means of an emulator like Putty. Then it is mandatory to create a proxy before giving any command to the jobs, if more than 12 hours have passed, typing

```
voms-proxy-init --voms unina.it
```

The Local File Catalogue is a space dedicated to simulation files bigger than 100 MB. Every user has his own personal directory on the LFC (to not be confused with the local personal directory). If not existent, it must be created with the command

```
lfc-mkdir /grid/unina.it/usernameDIR/folder
```

and then the filename.sim file can be copied on LFC with the statement

```
lcg-cr -v --vo unina.it file:/home/username/folders/filename.sim
   -l lfn:/grid/unina.it/usernameDIR/folders/filename.sim
```

C.4 running, monitoring and retrieving the job

From the user’s local folder where there are the previously discussed files, a job can be launched from the shell with the command

```
glite-wms-job-submit -a -o job_ID_filename.txt filename.jdl
```

that includes the registration of the job. Now any further command for that job will refer to its job identifier. For example, the job monitoring is obtained giving

```
watch "glite-wms-job-status -i job_ID_filename.txt"
```

and the job status is updated every two seconds. However this tells the user only if the run is scheduled, running, completed, aborted or cleared. To get a log file, the same that will be obtained when the job ends, but incomplete for obvious reasons, the perusal option can be enable in the filename.jdl and giving the following commands

```
glite-wms-job-perusal --set -f starccm.log https://scopedma-ce.scope.unina.it:9000/U51oobmOGTvXzMKP_pWGZw
```

to set the perusal and

```
glite-wms-job-perusal --get -f starccm.log https://scopedma-ce.scope.unina.it:9000/U51oobmOGTvXzMKP_pWGZw
```

to get the partial log file. The last string of the command is an internal identifier displayed on the shell once launched the job.
Once the job is completed, it can be get by typing

```
$ glite-wms-job-output -i job_ID_filename.txt
```

and retrieved from the temporary folder

```
/root/tmp/jobOutput
```

with the SFTP client.
TABLE OF CONFIGURATIONS

Unless otherwise stated, the data refers to the following conditions
\[ \alpha = 0^\circ \]
\[ \beta = 5^\circ \]
\[ M = 0 \]
\[ Re = 1000000 \]
\[ A = 10, A_v = 2 \) (for complete airplane).

Only the configurations presented in Chapter 5 are shown. Counting the other simulations files, the DIAS method is built on a synthesis of more than 180 CFD successful runs, not to mention the test cases discussed in Chapter 4 and the unsuccessful runs.

<table>
<thead>
<tr>
<th>ID</th>
<th>abbr.</th>
<th>parameters</th>
<th>notes</th>
<th>( C_{V_v} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>V</td>
<td>( A_v = 0.5 )</td>
<td>CFD model</td>
<td>0.0073</td>
</tr>
<tr>
<td>2</td>
<td>V</td>
<td>( A_v = 1.0 )</td>
<td>CFD model</td>
<td>0.0194</td>
</tr>
<tr>
<td>3</td>
<td>V</td>
<td>( A_v = 1.5 )</td>
<td>CFD model</td>
<td>0.0322</td>
</tr>
<tr>
<td>4</td>
<td>V</td>
<td>( A_v = 2.0 )</td>
<td>CFD model</td>
<td>0.0451</td>
</tr>
<tr>
<td>5</td>
<td>V</td>
<td>( A_v = 0.5 ); ( \beta = 10^\circ )</td>
<td>CFD model</td>
<td>0.0156</td>
</tr>
<tr>
<td>6</td>
<td>V</td>
<td>( A_v = 1.0 ); ( \beta = 10^\circ )</td>
<td>CFD model</td>
<td>0.0404</td>
</tr>
<tr>
<td>7</td>
<td>V</td>
<td>( A_v = 1.5 ); ( \beta = 10^\circ )</td>
<td>CFD model</td>
<td>0.0667</td>
</tr>
<tr>
<td>8</td>
<td>V</td>
<td>( A_v = 2.0 )</td>
<td>CFD model</td>
<td>0.0918</td>
</tr>
<tr>
<td>9</td>
<td>V</td>
<td>( A_v = 0.25 )</td>
<td>Straight vertical tail</td>
<td>0.0030</td>
</tr>
<tr>
<td>10</td>
<td>V</td>
<td>( A_v = 0.5 )</td>
<td>Straight vertical tail</td>
<td>0.0085</td>
</tr>
<tr>
<td>11</td>
<td>V</td>
<td>( A_v = 1.0 )</td>
<td>Straight vertical tail</td>
<td>0.0263</td>
</tr>
<tr>
<td>12</td>
<td>V</td>
<td>( A_v = 1.5 )</td>
<td>Straight vertical tail</td>
<td>0.0507</td>
</tr>
<tr>
<td>13</td>
<td>V</td>
<td>( A_v = 2.0 )</td>
<td>Straight vertical tail</td>
<td>0.0796</td>
</tr>
<tr>
<td>14</td>
<td>V</td>
<td>( A_v = 2.5 )</td>
<td>Straight vertical tail</td>
<td>0.1111</td>
</tr>
<tr>
<td>15</td>
<td>V</td>
<td>( A_v = 0.25 ); ( \beta = 10^\circ )</td>
<td>Straight vertical tail</td>
<td>0.0065</td>
</tr>
<tr>
<td>16</td>
<td>V</td>
<td>( A_v = 0.5 ); ( \beta = 10^\circ )</td>
<td>Straight vertical tail</td>
<td>0.0181</td>
</tr>
<tr>
<td>17</td>
<td>V</td>
<td>( A_v = 1.0 ); ( \beta = 10^\circ )</td>
<td>Straight vertical tail</td>
<td>0.0549</td>
</tr>
<tr>
<td>18</td>
<td>V</td>
<td>( A_v = 1.5 ); ( \beta = 10^\circ )</td>
<td>Straight vertical tail</td>
<td>0.1045</td>
</tr>
<tr>
<td>19</td>
<td>V</td>
<td>( A_v = 2.0 ); ( \beta = 10^\circ )</td>
<td>Straight vertical tail</td>
<td>0.1625</td>
</tr>
<tr>
<td>20</td>
<td>V</td>
<td>( A_v = 2.5 ); ( \beta = 10^\circ )</td>
<td>Straight vertical tail</td>
<td>0.2256</td>
</tr>
<tr>
<td>21</td>
<td>V</td>
<td>( b_v = 0.05 ); ( \lambda_v = 0.62 )</td>
<td>Constant taper ratio</td>
<td>0.0026</td>
</tr>
<tr>
<td>22</td>
<td>V</td>
<td>( b_v = 0.10 ); ( \lambda_v = 0.62 )</td>
<td>Constant taper ratio</td>
<td>0.0077</td>
</tr>
<tr>
<td>23</td>
<td>V</td>
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<td>49</td>
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[41] *Virtual skies*. 2012. URL: [http://quest.arc.nasa.gov/aero/virtual/demo/aeronautics/tutorial/structure.html](http://quest.arc.nasa.gov/aero/virtual/demo/aeronautics/tutorial/structure.html).


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