EFFICIENT EVALUATION OF PRELIMINARY AERODYNAMIC CHARACTERISTICS OF LIGHT TRAINER AIRCRAFT

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Abstract: *In the preliminary aerodynamic aircraft design, optimum calculation methods should be used, which enable engineers to perform a vast number of test runs in a reasonably short time. Such tools should not only be fairly simple but also reliable. In the first part, this paper briefly presents aerodynamic analyses performed by a 3D vortex lattice method (VLM), with an aim to verify its capabilities to give results that coincide well with the experimental data of an existing airplane. Since this computational model is based on inviscid flow concept, effectiveness of control surfaces and flaps are inherently overestimated. In order to compensate for the omitted boundary layer influence, a set of calibration diagrams for effectiveness and circulation influence has been successfully derived and good agreements with wind tunnel test data have been achieved. After several necessary adjustments, calibration functions have been applied to VLM analysis within a new light aircraft conceptual study. Those results have been compared with results obtained by well known Datcom method and verygood agreements have been achieved, proving that VLM computations with properly defined calibration functions can be both efficient and reliable tools in preliminary aerodynamic design.*

Keywords:
– trainer aircraft
– aerodynamic design
– preliminary CFD analysis
– calibration factors

1. INTRODUCTION

Within the complex multidisciplinary airplane design process, proper early estimates of aerodynamic characteristics are essential for the fulfillment of the assigned flying and technical requirements. In the initial design stages, it is necessary to investigate at least several possible configuration options, and to converge them efficiently to the most promising one. Thus, the preliminary aerodynamic analyses should be quick, efficient and reliable. The use of available analytical and semiempirical calculation methods for such analyses can give good results, but those calculations are often quite time consuming and not flexible enough for quick design changes. The application of wind tunnel tests is almost never considered in the initial design stages since they are very time consuming and expensive. Also, experience has shown that in spite of the use of very complex modern CFD (computational fluid dynamics) computer packages, there is no total guarantee of the quality, and they can be prone to errors. Within such methods, design of the detailed airplane model, optimization of the grid, optimum turbulence model verification, numerous test runs etc, turn out to be also very time consuming.

In search for an optimum choice, this paper considers an "intermediate" solution. In this test, the authors have compared a fairly simple CFD model based on the 3D vortex lattice method with a simplified 3D airplane model. For software reliability tests, the popular Serbian light trainer aircraft, Utva-75, shown in Figure 1, has been selected, for which a wide scope of wind tunnel test results exist. The applied inviscid computational model inherently neglects boundary layer influence and separation effects, so effectiveness of the flaps and control surfaces are overestimated at moderate and higher deflection angles. The problem was solved by deriving non-linear calibration diagrams for theireffectiveness and circulation influence.

Figure 1. Utva 75 light trainer aircraft

Figure 2. Transparent CAD model of the NLA

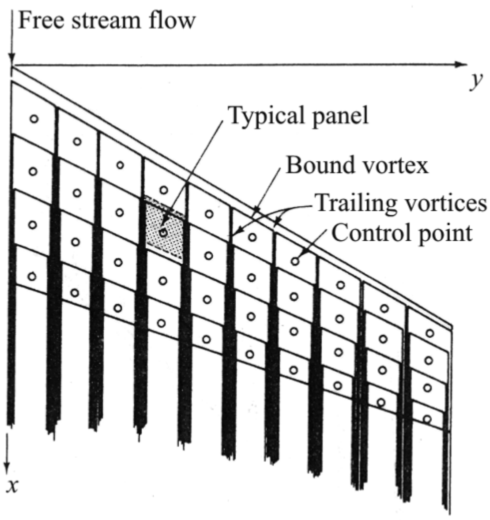

Figure 3. Horseshoe vortices on wing panels

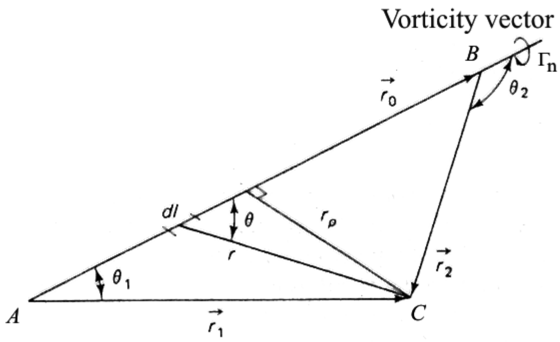

Figure 4. Nomenclature for the calculation of velocity induced by finite vortex segment

In this way, very good agreements with wind tunnel tests have been achieved, enabling an efficient prediction and optimization of aerodynamic characteristics for a new light aircraft (NLA – see Figure 2), during the first stages of its conceptual analyses.

2. CALCULATION PROCEDURE

The vortex lattice method VLM represents the wing (or horizontal tail, vertical tail, etc.) as a planar surface on which a system of horseshoe vortices is superimposed (Figure 3). The velocities induced by each horseshoe vortex at a specific control point are calculated using the law of Biot-Savart [1, 2]. The velocity induced by a vortex filament of strength Γ_n and length dl (Figure 4) is:

$$\overrightarrow{dV} = \frac{\Gamma_n (\overrightarrow{dl} \times \vec{r})}{4\pi r^3} \qquad (1)$$

If we consider a vortex segment of finite length AB, and if C is a point in space at a normal distance r_p from line AB, then the magnitude of total velocity V induced at C by this segment will be according to the nomenclature shown in Figure 4:

$$V=\frac{\Gamma_n}{4\pi r_p}\int_{\theta_1}^{\theta_2}\sin\theta\, d\theta=\frac{\Gamma_n}{4\pi r_p}(\cos\theta_1-\cos\theta_2) \quad (2)$$

Each panel is represented by a horseshoe vortex (consisting of the bound vortex *BC*, positioned at the quarter panel chord position, and two semi-infinite trailing vortices *BA* and *CD*), and by the control point at 3/4 of the panel chord at its mid section, as shown in Figure 5.

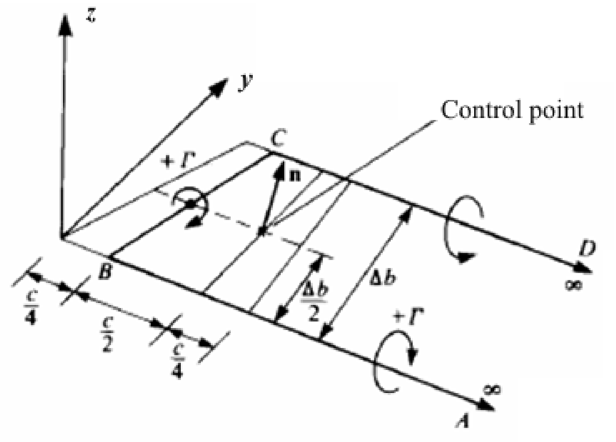

Figure 5. A horseshoe vortex represents each panel

The summation is performed for all control points to produce a set of linear algebraic equations for the horseshoe vortex strengths that satisfy the boundary condition of "no flow through the surface" which this panel represents, i.e. the velocity tangency condition. The general form of the velocity induced in a control point is given by:

$$\vec{V}=\frac{\Gamma_n}{4\pi}\frac{\vec{r}_1\times\vec{r}_2}{\left|\vec{r}_1\times\vec{r}_2\right|^2}\left[\vec{r}_0\cdot\left(\frac{\vec{r}_1}{r_1}-\frac{\vec{r}_2}{r_2}\right)\right] \quad (3)$$

Suppose that the airplane VLM configuration is in a free stream of the velocity V_0, at an angle of attack α. At the *m*-th panel, the induced velocity is $\vec{V}_m(u_m,v_m,w_m)$. If the side slope of the panel is denoted as φ, and if its longitudinal slope is δ, the velocity tangency condition (free stream velocity plus induced) is given by:

$$\begin{aligned}-u_m\sin\delta\cos\varphi-v_m\cos\delta\sin\varphi+w_m\cos\varphi\cos\delta\\ +V_0\sin(\alpha-\delta)\cos\varphi=0\end{aligned} \quad (4)$$

Although the vortex lattice method is based on the planar *presentation* of the airplane configuration, the influence of actual mean surface cambers, incidences, dihedral and twist angles, deflections of control surfaces and flaps must be taken into account. For example, for Utva's wing, coordinates of NACA 65_2-415 airfoil [3] had to be assigned, while for fuselage, its side shape had to be defined.

The vortex strengths are related to the lifting surface circulation and the pressure differential dC_P between the upper and lower surface sides $dC_P=C_{PU}-C_{PL}$ (according to here applied convention). Pressure coefficients for the upper side C_{PU} and the lower side C_{PL} of the panel are calculated as $C_P=(p-p_0)/q_0$ where p_0 and q_0 are free stream static and dynamic pressures, while p is static pressure on the given panel side. The pressure differentials are integrated to yield the total forces and moments. For the comparisons with the wind tunnel data, this paper considers the lifting force *L* and the pitching moment *M* about the center of gravity expressed in terms of their coefficients (*S* is the wing's aerodynamic area, and c_{MAC} is mean aerodynamic chord):

$$C_L=L/(1/2\cdot\rho_0\cdot V_0^2\cdot S);\quad C_M=M/(1/2\cdot\rho_0\cdot V_0^2\cdot S\cdot c_{MAC}) \quad (5)$$

It should be mentioned that the Prandtl-Glauert equation [1] lies in the background of here applied VLM, meaning that the flow is treated as compressible, inviscid and irrotational (for the explanation of existence of the vorticity Γ in irrotational flow, also see [1]). Particularly, for here presented analyses, the flight Mach number M_0 (ratio between the flight speed and the speed of sound) is small, of the order of $M_0\approx 0.14$, and thus the compressibility corrections become negligible. On the other hand, this calculation model confines the global C_L and C_M analyses only to the angles of attack which correspond to their linear domains where boundary layer and flow separation effects are not so immanent.

This paper considers only symmetrical flow cases, meaning that only influences of flaps and elevator deflections have been analyzed (i.e. it is assumed that aileron or ruder deflections, as well as the sideslip angle, roll or yaw rates are equal to zero). The overall task of here presented analyses can be summarized in the following global steps:

1) Design two compatible full 3D models for vortex lattice calculations of Utva 75 and NLA (see Figure 6).
2) Using the vortex lattice method, perform the Utva 75 analyses for cases from wind tunnel tests.
3) Compare the "raw" VLM results with results obtained from wind tunnel tests. Considering the simplifications introduced by vortex lattice method, lift and moment coefficients should be analyzed and comparisons made only for linear domains of these functions.
4) Try to select and determine values of proper calibration factors (considering the physical consistency of the problem which should be solved) that will bring the CFD results as close to the experimental values as possible.
5) Perform VLM calculations for the new light aircraft, applying Utva's calibration factors, recalculated for different airplane geometry where necessary.
6) Compare VLM results for the new light aircraft with results obtained by some well recognized and reliable semiempirical method, such as Datcom [4].
7) If agreements of the results are satisfactory, perform additional VLM analyses for several optional wing planform shapes with different taper values, etc.

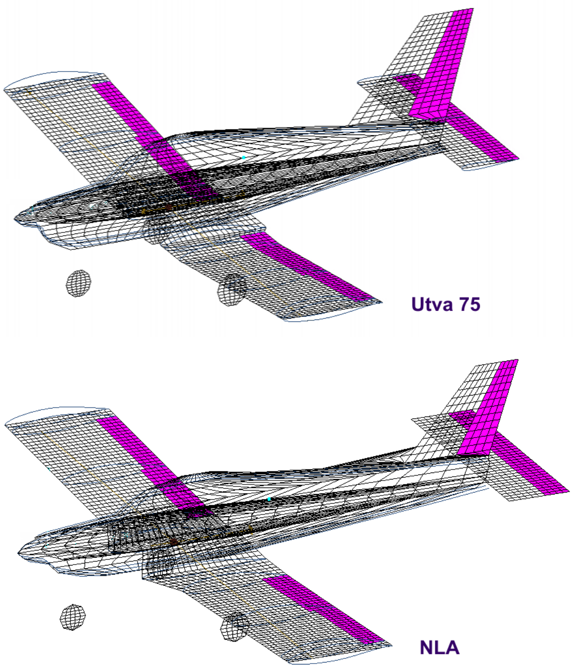

Figure 6. Paneling of full 3D models for vortex lattice aerodynamic analyses

3. DETERMINATION OF CALIBRATION FUNCTIONS

In all VLM analyses, the angle of attack α_f, defined with respect to the fuselage reference axis, varied in the range $\alpha_f = -12^o \div +10^o$, whereas angles $\alpha_f > +10^o$ have not been considered.

3.1. Flaps and Elevator at Zero Deflection

Initial comparisons of the wind tunnel data and VLM results have been done for Utva with flaps retracted and with zero elevator deflection. The results obtained by VLM, without any calibrations applied, have shown good agreements for the gradients (slopes) of C_L and C_M curves (dashed lines in Figure 7), but both curves were shifted with respect to the wind tunnel points.

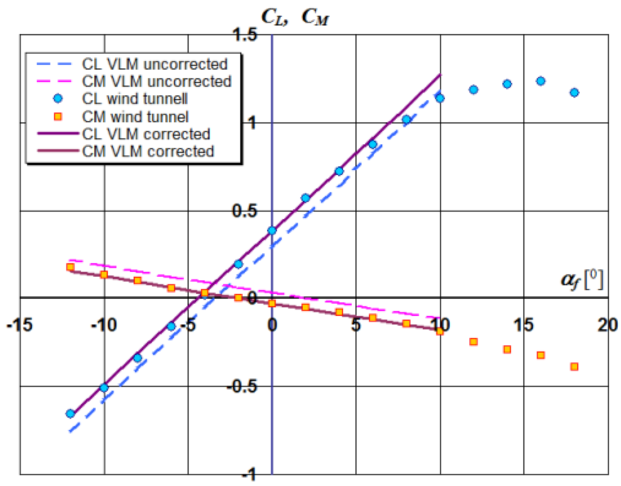

Figure 7. Wind tunnel data and VLM results, flaps and elevator deflection 0^o (incidence calibrations)

In reference [5], it was not stated whether the incidence angles of the wing and horizontal tail on wind tunnel model during tests in 1976 were the same as on Utva 75 production airplanes ($+2^o$ for the wing, and -2^o for the tail, which were initially applied to the VLM model). Thus, the first calibration parameters to be considered for VLM were the incidences. Incidence corrections of $+0.9^o$ for the wing and $+2.0^o$ for the tail have been determined. With these values, very good agreement between wind tunnel and VLM results has been obtained, and they have been preserved and applied to all other VLM Utva 75 analyses.

3.2. Elevator Deflections

The next set of analyzed cases was Utva 75 configuration with elevator deflections in range $\delta_e = -30^o$ (up) ÷ $+20^o$ (down), with 10^o steps, without the deflection of flaps. Neglecting the boundary layer effects, VLM inherently overestimates control surface effectiveness. In this case, gradients were also good, but shifts, especially of moment curves, were quite large (see examples in Figures 8 and 9). The use of some generalized corrections, as suggested in reference [3], failed to give any satisfactory matches with wind tunnel data.

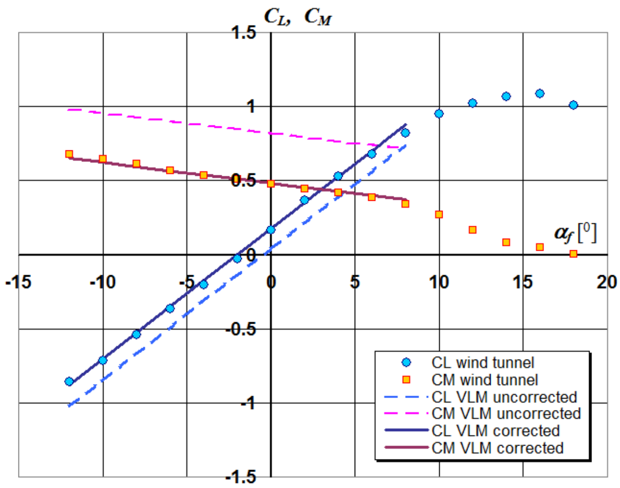

Figure 8. Wind tunnel data and VLM results, elevator deflection $\delta_e = -30^o$ (η_e calibration)

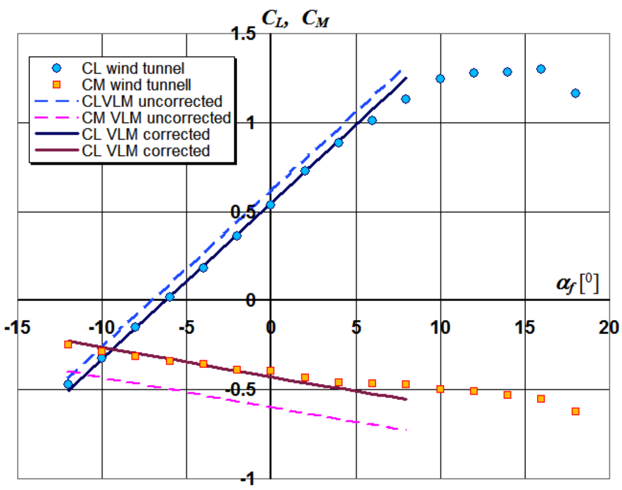

Figure 9. Wind tunnel data and VLM results, elevator deflection $\delta_e = +20^o$ (η_e calibration)

Through repeated tests, finally the elevator calibration factor η_e values giving good match were η_e = 0.6; 0.75; 0.9; 1.0; 0.82 and 0.7 for the previously mentioned δ_e range, being a non-linear and asymmetrical function with respect to $\delta_e = 0^o$.

3.3. Flaps Deflections

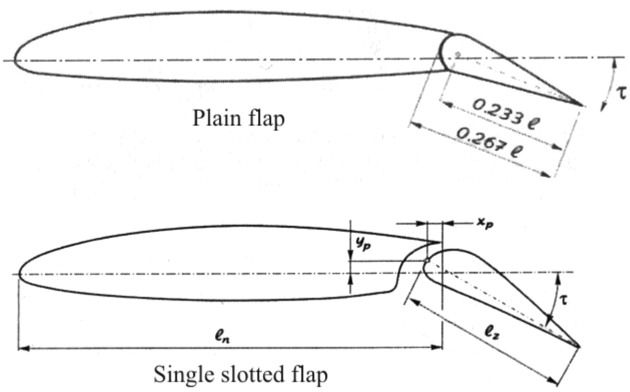

Figure 10. Examples of plain and slotted flaps

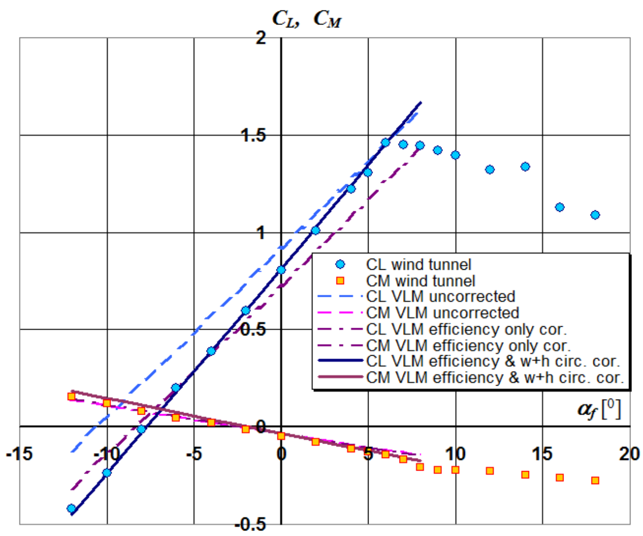

Figure 11. Wind tunnel data and VLM results, $\tau = 25^o$, $\delta_e = 0^o$ (calibration: η_f, $Cirw_f$ $Cirh_f$)

Utva 75 has single slotted flaps, and the same type of flaps is considered for the NLA. On the other hand, the VLM inherently treats flaps in the same way as the control surfaces - as simple plain flaps (Figure 10). The principal difference between the two types is that slotted flaps rotate about the axis which is below the wing structure. In this way, a convergent gap between the slotted flap and the wing structure appears. The airflow through the gap "energizes" the boundary layer and keeps it attached to the upper flap surface longer so that this type of flap generates higher lift (or "circulation" in the mathematical sense) than the plain flap [6]. Utva's wind tunnel data exist for flap deflections of $\tau = 25^{\circ}$ and 45° (also combined with the same elevator deflections). Uncorrected VLM results for C_M were quite good, but C_L showed both gradient and shift discrepancies (dashed line in Figure 11).

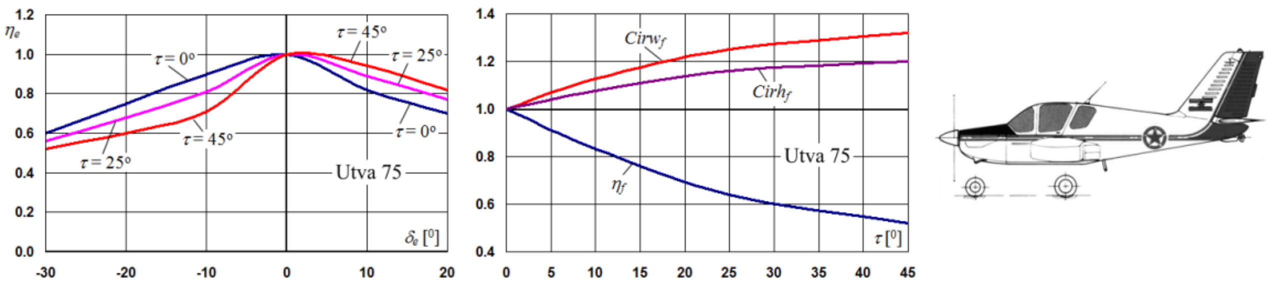

Figure 12. Calibration diagrams determined for Utva 75, from the existing wind tunnel tests

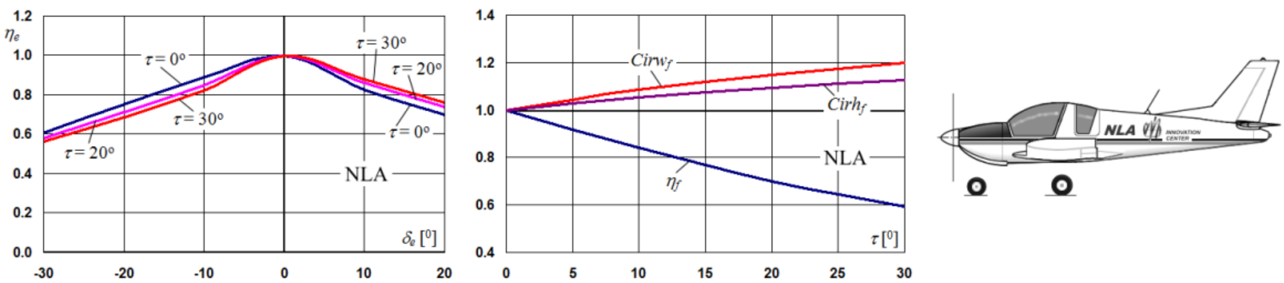

Figure 13. Calibrations for NLA, recalculated taking into account different geometry and flap deflections

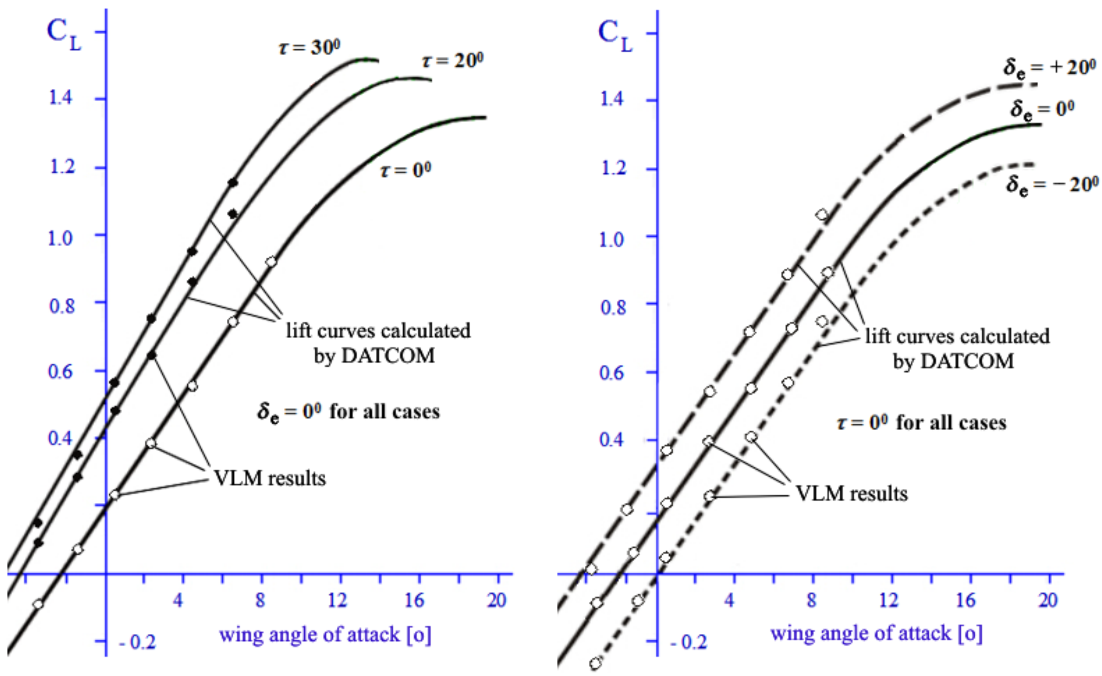

Figure 14. Lift coefficients of the NLA, obtained by calibrated VLM and by Datcom

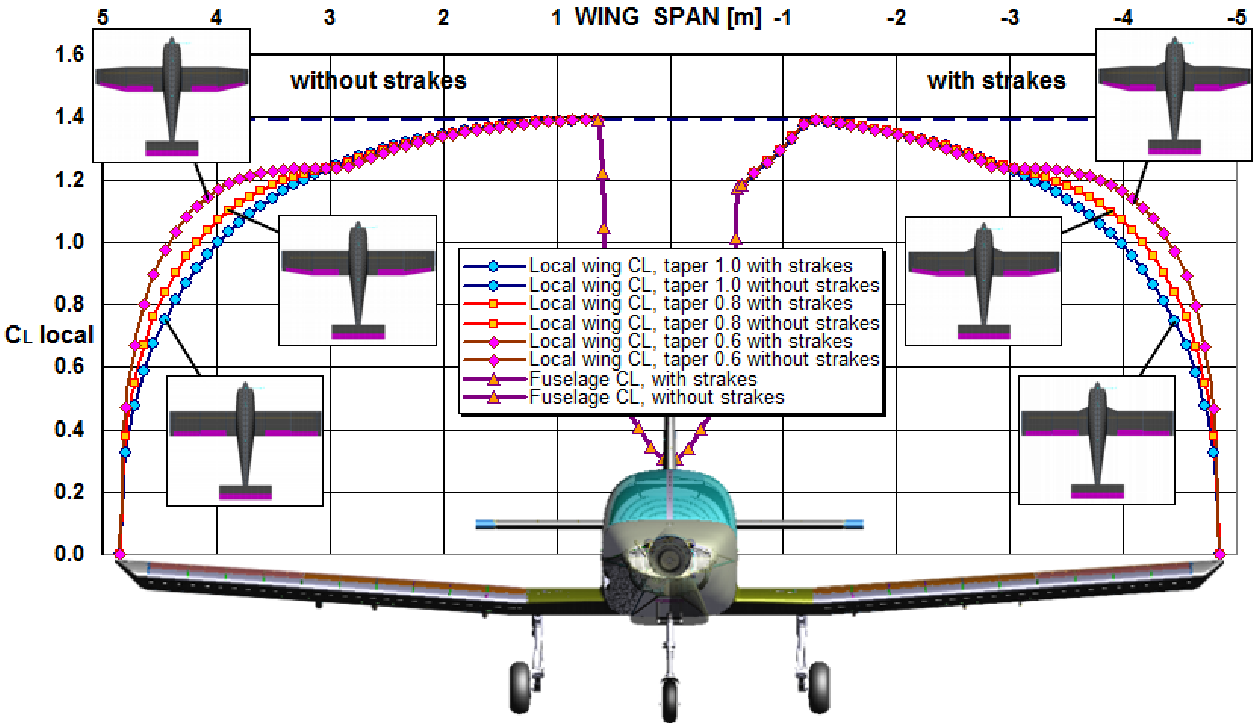

Figure 15. Different possible NLA wing planform shapes without and with strakes, analyzed by VLM

After a number of tests, initially for the case $\delta_e = 0^o$, simultaneous combinations of flaps efficiency η_f factor (affecting shift) and circulation calibration $Cirw_f$ factor (affecting gradient) have been defined. These calibrations gave good agreements with the experiment.

On the other hand, the final "fine tuning" of C_M also required the introduction of the third parameter, the horizontal tail circulation calibration $Cirh_f$ due to flaps deflection which, combined with the other two, gave precise agreements considering the longitudinal stability derivatives.

Then, tests have also been repeated for all elevator deflections, and η_e for $\tau = 25^o$ and $\tau = 45^o$ have been successfully determined.

The set of VLM calibrations for Utva 75, derived from the existing wind tunnel test results, is given in the form of summary diagrams and shown in Figure 12.

4. EXAMPLES OF THE NEW LIGHT AIRCRAFT CALCULATIONS

Using interpolation methods, appropriate calibration factors have been recalculated for NLA, taking into account differences of wing areas in flaps domains, and different reference flap deflections. Additionally, similar aerodynamic analyses have been repeated (Figure 13). Parallel calculations for NLA have also been done by well recognized Datcom method, based on completely different calculation approach [4]. When placed on the same diagrams, VLM calculations have overlapped very well (naturally, in linear domains) with Datcom results, verifying the accuracy of predicted calibrations (Figure 14).

The advantages of a properly calibrated VLM model have proved to be numerous. In addition to the efficient aerodynamic calculations, obtained results have provided quick and reliable estimates of stability and control derivatives as well as detailed chordwise and spanwise load distributions necessary for preliminary structural analyses. Figure 15 shows the example of local - spanwise lift coefficient distributions, calculated by VLM on six considered NLA's wing planform shapes (here without aileron deflections), obtained by reasonably simple and easy modifications of geometry input parameters.

5. CONCLUSION

The paper presents the methodology of calibration functions derivation, which has been applied to VLM calculations of two light aircrafts. The unique and physically consistent calibration functions have been obtained through comparative analyses using experimental results of the existing airplane. After necessary recalculations by interpolation procedures, they were applied to VLM analysis of a new light aircraft at the conceptual design stage. These results have shown good agreement with results obtained by Datcom. Taken as a whole, it has been verified that fairly simple VLM methods with proper calibrations applied can be flexible, efficient and reliable tools in determination of most important aerodynamic characteristics in the initial aircraft design stages.

6. LIST OF SYMBOLS

circulation	Γ,	m^2/s
free stream static pressure	p_0,	Pa
free stream dynamic pressure	q_0,	Pa
free stream density	ρ_0,	kg/m^3
free stream speed	V_0,	m/s
wing area	S,	m^2
mean aerodynamic chord	c_{MAC},	m
local static pressure	p,	Pa
pressure coefficient	C_P	
lift force	L,	N
pitching moment	M,	Nm
moment coefficient	C_M	
elevator deflection angle	δ_e,	deg
flap deflection angle	τ,	deg
elevator efficiency calibration factor	η_e	
flaps efficiency calibration factor	η_f	
wing circulation calibration factor	$Cirw_f$	
tail circulation calibration factor	$Cirh_f$	

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