Numerical simulation of aerodynamics performance for winglet in the low speed compressible flow

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Abstract - The main purpose of the present study is to develop a code based on surface singularity distribution concept called paneling method, in order to investigate and to estimate the physical effect of a linked winglet to the tip of the wing and to compute the new aerodynamic performances of various wing – winglet combination profiles type depending on the dihedral angle variation that is made by the winglet with respect to wing plane in an inviscid, subsonic flow at small angles of attack. The body surface to be analyzed was divided into a finite number of panels and each one was replaced by a singularity distribution choice of sources, sinks and vortex lattice components of perturbation velocity induced at specified control points of local panels are determined. They make up the aerodynamic matrices which are needed to calculate the pressure distribution and (Cp) consequently the aerodynamic coefficients (Cl, Cd, Cm).

Résumé - Le but essentiel de cette étude est de développer un code basée sur le concept de la distribution de singularités appelé méthode des panneaux, pour étudier et estimer les effets physiques d’une ailette en bout d’aile et de calculer les nouvelles performances aérodynamiques de différents configurations du type aile – ailettes pour une variété d’angles dièdre dans un écoulement subsonique, non visqueux à de faibles angles d’attaque. La surface du corps a analyser a été divisée en un nombre fini de panneaux et chacun de ces derniers a été remplacé par une implantation d’une distribution d’un choix de singularités de sources, puits et de tourbillons. Les composantes de la vitesse perturbée induite aux points de contrôle spécifiques, localisées au niveau de chaque panneau sont déterminées. Elles formeront ainsi les matrices aérodynamiques qui sont essentielles pour le calcul de la distribution locale de la pression (Cp) et par conséquent des coefficients aérodynamiques (Cl, Cd, Cm).

Keywords: Aerodynamics - Low speed - Compressible fluid - Subsonic flow - Winglet.

1. INTRODUCTION

Winglets are small wing-like lifting surfaces, fitted at the tips of some wings, usually with the objective of decreasing the trailing-vortex drag and thereby increasing the aerodynamic efficiency of the basic wing. Winglets may comprise both upper and lower elements or upper elements only. Those comprising upper elements only are the more common.

This Item deals with the aerodynamic effects of winglets at subsonic speeds, mainly so far as they affect the aerodynamic efficiency of a configuration, defined either as L/D or M.L./D. Fitting winglets can provide improvements in aerodynamic efficiency for a range of lift and Mach number conditions by decreasing the trailing-vortex drag by amounts that more than compensate for any increases in the profile and

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wave drag contributions. In addition to their effects on trailing-vortex drag, some other beneficial effects may arise from winglets, for example on the vortex wake characteristics (important for aircraft used for agricultural spraying), see (1) and (2), for the directional stability of canard configurations (3), and, arguably, on the appearance of the aircraft.

Fig. 1: Wing with winglet

2. DEVELOPMENT OF THE THEORY APPROACH

2.1 Basic theory (Exact equation of potential flow)

The two-dimensional strip theory aerodynamics, for convenience, implies a number of major assumptions about the aerodynamic load distribution (e.g. neglects tip effects) and is only moderately accurate for low speed, high aspect ratio and upswept wings. Of particular importance is the assumption that the aerodynamic forces acting on one chord wise strip have no effect on other chord wise strips. In order to perform a more accurate aeroelastic analysis, aerodynamic theories need to be developed that are able to define more accurate pressure distributions over the entire wing.

The so-called three-dimensional panel methods were developed to model the interaction between the aerodynamic forces on different parts of the lifting surfaces (wings, fin and horizontal tail surfaces) more accurately. It will be showing in this study how it is possible to couple fully the panel method aerodynamics with a finite volume model; consequently, panel methods are the primary aerodynamic tool used by industry for aeroelastic analysis. However, it should be noted that panel methods cannot be used to give accurate lift distributions in the transonic flight regime and corrections based upon wind tunnel tests are often employed.

Also only the induced drag is able to be estimated. Consequently, there is an increasing use of higher fidelity computational fluid dynamics (CFD) methods, often solving the full Navier–Stokes equations, coupled with VE models to determine the time response to some initial displacement in the transonic region.

Theoretically, the method of the panels, which takes into account the flows with lift, takes root on the disturbances of the potential by effect of displacement.

Now examine some elementary solutions. One such solution is the so-called source. Let's suppose we have a source with strength $\Gamma(s)$, positioned at some point $P$. The potential $\varphi$ caused by this source at some point $P$ then is

$$\varphi_s(P) = \frac{1}{4\pi} \frac{\Gamma(s)}{r(s,p)}$$

(1)

If the source strength $\Gamma$ is negative, the source is often called a sink.
What can we do with these sources? Well, we can put a lot of them on a curve $S$. We then get a source distribution. To find the velocity potential at some point $P$, caused by this distribution, we use an integral. This velocity potential thus will be:

$$\phi_s (p) = \frac{1}{4\pi} \iint_s \frac{\Gamma(s)}{r(s,p)} \, ds$$  \hspace{1cm} (2)

A problem might occur if the point $P$ lies on the source distribution itself. Such a situation should always be considered separately.

Another elementary solution is the doublet. The potential at some point $P$, caused by a doublet at $S$, is given by

$$\phi_d (p) = \frac{1}{4\pi} \mu(S_i) \frac{\Gamma(s)}{r(S_i,p)}$$  \hspace{1cm} (3)

where $\mu(S_i)$ is the strength of the doublet and $i$ is the direction of the doublet.

Once more, we can put a lot of doublets in a row. We then get a doublet distribution. To find the velocity potential at $P$, we now have to use:

$$\phi_d (p) = \frac{1}{4\pi} \iint_{s_i} \mu(S_i) \frac{\Gamma(s)}{r(S_i,p)} \, ds_i$$  \hspace{1cm} (4)

We see that we need to solve the Laplace equation. However, to solve it, we need boundary conditions.

For example, set the value of $\phi$ along the boundary. This is so-called Dirichlet boundary conditions. If we do this, then there is a unique solution for $\phi$. (We say that the problem is well-posed.)

However, we can also set the value of $\frac{\partial \phi}{\partial n}$ at the boundary. We now have Neumann boundary conditions. This time, there is no unique solution for $\phi$. We can only determine $\phi$ up to a certain unknown constant. Of course we can also use combinations of Dirichlet and Neumann boundary conditions. However, as long as $\phi$ is set somewhere, then there is always a unique solution for $\phi$.

If it represents the intensity of the dipole in a surface that remains to determine, the total potential is the result of the superposition for equations (2) and (4) that is to say:

$$\phi(p) = \phi_s (p) + \phi_d (p)$$  \hspace{1cm} (5)

By using the boundary, condition so below

$$\left(\frac{\partial \phi(p)}{\partial n}\right)_s = -n \ddot{U}_\infty$$  \hspace{1cm} (6)

One can represent the solution of the limiting value of the problem by

$$\frac{1}{4\pi} \left[-\ddot{n} \iint_s \frac{\partial \phi}{\partial r} \, ds + \ddot{\ddot{n}} \iint_{s_i} \mu(S_i) \left(\frac{1}{r}\right) ds_i\right] = -\ddot{n} \ddot{U}_\infty$$  \hspace{1cm} (7)

2.2 Formulation of the numerical method

The method used for the simulation of this configuration:

1. Cover body with 3D constant strength doublet panels of unknown strength (N panels)
2. Place a control point at the center of each panel (N control points)
3. Write down an expression for the normal velocity at each $C_p$ produced by the velocity and $N$ panels
4. Solve resulting $N$ equations for $N$ panel strengths
5. With panel strengths compute tangent velocity at c.p.s and thus surface pressure, and rest of flow.

![Three-dimensional wing with winglet](image1)

Consider a cut through the body surface:

![Body surface](image2)

- Panels appear as a series of discrete vortices with strengths equal to the difference between adjacent panel strengths. At the control points we see almost none of the tangential velocity generated by nearby vortices. This is wrong.
- It is as though the surfaces were a continuous vortex sheet, and we calculated the tangential velocity at the Cp. ignoring the discontinuous jump between the inside and outside. This jump, equal to half the local sheet strength is called the principal value and must be added to the tangent velocity calculated at a control point
- The principal value for a doublet panel can be estimated as…
- This must be added to the tangential velocity calculated at any control point
- **Panel Influence**

Velocity due to panel $m$ at control point $n$:

$$ V = V_{\tan} $$

**Fig. 3-b: $V = V_{\tan}$**

The normal component is:

$$ V(r^{(n)}_n) = \sum_{m} C_{nm} \Gamma^m $$

Or

$$ V(r^{(n)}_n) = \sum_{m} C_{nm} \Gamma^m $$

Summed velocity at control point $n$ is thus:

$$ V(r^{(n)}_n) = V_\infty + \sum_{m} C_{nm} \Gamma^m $$

Normal component is:

$$ V(r^{(n)}_n) \times n^{(n)}_c = V_\infty \times n^{(n)}_c + \sum_{m} C_{nm} \times n^{(n)}_c \times \Gamma^m $$

So, to get the $\Gamma^m$'s we need to solve the simultaneous equations:
We replaced the structure wing - winglet by a system of distribution of sources-sinks on all contour to simulate the thickness, and the distribution of vortex sheet in the field of the cord of the wing to simulate the influence of the camber and angle of attack.

The theory of the image will emphasize the phenomenon of interference in the combination wing – winglet in wing end. Plane flows makes it possible to understand certain practical flows but they represent limitations when one wants to model a three-dimensional flow.

Applied Aerodynamics is three-dimensional but the study is complex. The determination of the field of flow is always of topicality in the field of theoretical research as well as experimental. The basic equations of the method of the singularity are represented by the flow chart, which one notices that the starting point of this one is the equation relating to the incompressible and three-dimensional potential, with like boundary conditions.

The disappearance of the disturbances of the potential with infinite and the condition of the cinematic flow on contour (null orthogonal speed). The application of these conditions allows the resolution of the equation in accordance with the flow chart of figure 1. For the determination of the intensities of the singularity, one notice in case of double integral is identified as being the influence of the $C_{ij}$ coefficient of the panel $J$ on the point of control $I$. This procedure brings back us easily to system linear equations with solution. Once sizes determined, the defined equation, the distribution of pressure can be obtained, after differentiation of the velocity vector and finally the aerodynamic characteristics ($C_l$, $C_d$ and $C_m$) approximated.
We illustrate now method which is simple enough to be presented as example completely, and at same time by which is possible to study three-dimensional configurations.

The key building block for the application of this method is subroutine which calculates induced velocity at arbitrary point from the polygonal and in general case three-dimensional, vortex line. It is expected that the wing is defined by its corner.

The elaborate simulation program allows the determination of the geometry, the choice of the suitable grid and the calculation of the aerodynamic characteristics of the flow around profile (NACA 4 and 5 of the configuration (settable), as well as the properties of the flow (Mach number, viscosity, compressibility, etc…)).

**Table 1**: Table of the data of configuration

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Wing</strong></td>
<td>NACA 4415</td>
</tr>
<tr>
<td><strong>Winglet</strong></td>
<td>NACA 0012</td>
</tr>
<tr>
<td>Angle of attack alpha</td>
<td>3 °</td>
</tr>
<tr>
<td>E = cord</td>
<td>0.5 and 1.0</td>
</tr>
<tr>
<td>Angle of winglet</td>
<td>20° - 50° - 80°</td>
</tr>
<tr>
<td>Length of wing</td>
<td>2.8 m</td>
</tr>
<tr>
<td>Length of winglet</td>
<td>0.2 m</td>
</tr>
<tr>
<td>Numbers panels extrados</td>
<td>20</td>
</tr>
<tr>
<td>Numbers panels has the under-surface</td>
<td>20</td>
</tr>
<tr>
<td>Numbers panels on axis Z</td>
<td>18</td>
</tr>
<tr>
<td>Arrow</td>
<td>0° - 25° - 45°</td>
</tr>
</tbody>
</table>

Knowing that the three-dimensional flow around a wing and aerodynamic characteristics $C_l$, $C_d$ and $C_m$ are according to the various parameters which more particularly define the form in plan of this one, such as lengthening, the arrow and tapering, corresponding to the three types of planes used currently and which differentiate only through the speed of flight.

We adapt in this application our wing in accordance with the arrows represented in figure 4 with a configuration of exhaustive grid of the stations of study. (Fig. 4-b and Fig. 4-c).
3. RESULTANTS AND INTERPRETATIONS

For a tapering of $e = 1.0$, figure 5 which represents the fields of pressure for four stations of wing NACA 4415 [without winglet], shows a very delicate variation of the flow pressure around the wing and which is reduced as one approaches the end of the wing. This variation is all the more better in the distributions either of the pressure or in speed along the length of the combined wing that the arrow of the latter $= 0°$. With a winglet $= (20°, 80°)$ (Fig. 6 and Fig. 7).

Figure 8 represents a description in 3D of the distribution of the aerodynamic characteristics in the area of establishment of the winglet, we notice in particular the widening of the zone of influence in the fields of flow in end of wing in accordance with the variation of the winglet of $(0°, 20°, 50°, 80°)$.

With an $arrow = 0°$. in order to develop this study, we represented for the second tapering ($e = 1.0$) the same aerodynamic characteristics with the same data base but with an angle of inclination of $3°$, we have noticed that the variations in the fields of flows follow the same process as in the case of represented for $e = 0.5$ (Example C1).

Figure 10 nevertheless the increase in the values of lift are according to the size of the airfoil ($e = 1.0$ and $e = 0.5$) on the other hand for the same airfoil the increase in the angle of inclination of the winglet reduces total $Cd$ of the combined wing ($arrow = 25°$).
Fig. 5: Fields of pressure for four stations

Fig. 6-a: Pressure along wing + winglet 20°  Fig. 6-b: Speed along wing + winglet 20°

Fig. 6-c: Pressure along wing + winglet 80°  Fig. 6-d: Speed along wing + winglet 80°
Fig. 7-a: $C_p$ along wing + winglet 20°

Fig. 7-b: $V_{tan}$ along wing + winglet 20°

Fig. 7-c: $C_p$ along wing + winglet 80°

Fig. 7-d: $V_{tan}$ along wing + winglet 80°

Fig. 8-a: Distribution of the fields of $C_l$ along the compound wing-winglet

Fig. 8-b: Distribution of the fields of $C_d$ along the compound wing-winglet
Fig. 8-c: Distribution of the fields of $C_m$ along the compound wing-winglet

Fig. 9-a: Comparison of $C_l$ for $\alpha = 1$ with different winglet

Fig. 9-b: Comparison of $C_d$ for $\alpha = 1$ with different winglet

Fig. 9-c: Comparison of $C_m$ for $\alpha = 1$ with different winglet
4. CONCLUSION AND PERSPECTIVES

In flight at subsonic speed the aerodynamic efficiency of a wing (without winglets) is ultimately constrained by its span. Increasing the span permits decreases in the trailing-vortex drag (at given lift) and may provide an increased aerodynamic efficiency. Winglets, comprising either upper elements, or a combination of upper and lower elements, can similarly lead to decreases in the trailing-vortex drag and increased aerodynamic efficiency. The magnitudes of the increases in aerodynamic efficiency that can be achieved by adding winglets are related principally to the lengths of the winglet elements relative to the basic wing span, to the orientation angles of the elements (cant angles) and to the spanwise loading distribution of the basic wing. For most winglet configurations, the increases in aerodynamic efficiency are of the order of a few percent.

![Fig. 10: Comparisons of Cl for θ=0.5 with various slopes winglet](image)

![Fig. 11: Comparisons of Cd for θ=1 with arrows =45 and various slopes winglet](image)

The results presented above do not represent but a part of the recorded results, in this study devoted to this phenomenon of aerodynamic design of end of wing. The handiness of the program on the possibility of diversification of treated examples really captured the phenomenon of interaction in the area of root of the winglet, this code computer provides in a very appreciable way and technique the solutions of the integral equation. It provides in particular the speed and pressure fields as well as the local and total forces of the studied configuration, it is particularly adapted to various studies of aerodynamics for its following qualities: to have directly the above mentioned results without solving the equation of Laplace fields of flow, overcoming the limits of the theory of the thin wing by using firstly the solutions of orders superior to emphasize the precision of the approximation (multipolar expansion) and secondly the boundary conditions tangential of the exact flow. In this perspective it is very possible to direct research on the insertion of the viscous effects due to the boundary layer and for the resolution of the problem of aeroelasticity (the divergence in torsion of the wings into subsonic).

Finally the validity of the code is always subject to the experimental results which we exploited at the base of cases test worked out before the study itself of this type of configuration.
NOMENCLATURE

\( A_{ij} \): Normal component speed induced
\( A_n \): Coefficients of Fourier
\( a \): Slopes of the coefficient of Fourier
\( B_n \): Coefficients of Fourier
\( C \): Twist profile
\( C_p \): Pressure coefficient
\( C_m \): Coefficient of moment
\( C_d \): Coefficient of drag
\( j^{\text{ième}} \) segment respectively

\( \rho \): Not arbitrary locates on the surface of the profile
\( K \): Index in the direction of axis Z
\( M \): Mach number
\( N \): Numbers total surface elements or cinematic viscosity
\( v_{ij} \): Speed induced with \( j^{\text{ième}} \) not of control by a source of intensity unit placed on the \( j^{\text{ième}} \) element.

\( X, Y, Z \): Cartesian co-ordinates

\( \Gamma \): Circulation of the swirling filament
\( \eta \): Equation of the profile
\( \gamma \): Intensity of swirl
\( \alpha \): Angle of attack
\( T_2 \): Tangential component speed in the case of a transverse flow around an axisymmetric body (tangent \( T_2 \) with the profile is given in plan XY).

\( 0 \): Angle
\( \sigma \): Intensity of source
\( \Phi \): Potential speed
\( \Phi_{ij} \): Induced potential with \( i^{\text{ième}} \) not of control by a unit source placed on the \( j^{\text{ième}} \) element
\( \Psi \): Function of current
\( \infty \): Ad infinitum indicate the quantities associated with the flow upstream

\( \Delta \): Laplacian or length of an element
\( V \): Gradient \( q \): not or one locates the source, and in particular the point which is located on the surface of the profile
\( R \): Radial co-ordinates (in the system of coordonne cylindrical)
\( \mathbf{n} \): Normal unit vector on the surface or the normal vector on each segment
\( r \): Outdistance between two points in the three-dimensional case, in particular the distance between two points or the source is placed and a point or speed and the potential evaluated sounds
\( S \): Surface border of the body (contour wraps).

\( U \): Rate of the nondisturbed flow
\( T \): Tangential speed in the case of two-dimensional bodies or in the case of an axisymmetric flow around an axisymmetric body
\( T_3 \): Circumferential component parallel speed has the axis Y in plan (XY).

\( U, v, w \): the component induced speed

REFERENCES


