Study of Various Methods for Modeling Gust Penetration

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Master Thesis performed at Dassault Aviation, St Cloud, FRANCE
3rd of February - 20th of June 2014

Abstract: Gust load computations are one of the various load cases computations required by the regulation in order to get an aircraft certified. These loads can even be the ones which have to be used for the structural sizing of the vertical and horizontal tail or the main wing of the aircraft. Three numerical methods to model penetration into a cylindrical gust are presented and their results are compared in terms of loads prediction on the structure. The influence of model parameters is studied and some general thoughts about delay and penetration are gathered. It is shown that the spatial discretization of the phenomenon does not need to be fine to yield accurate results and that different treatments of aerodynamic interactions between the subparts of an aircraft might bring rather large discrepancies in the results. In order to evaluate whether these numerical methods predict realistic loads, a comparison between experimental data and numerical computations on a simple wing profile is performed.

Nomenclature

\[ A = \text{Influence coefficients matrix} \]
\[ AC = 'Aircraft' \]
\[ b = \text{Profile span (m)} \]
\[ b_0 = \text{Span of the experimental profile (m)} \]
\[ [C],[c] = \text{Damping matrix (FE model, reduced)} \]
\[ C_p = \text{Pressure coefficient} \]
\[ c = \text{Profile chord (m)} \]
\[ F = \text{External forces (FE model, N)} \]
\[ F_{def} = \text{Reduced aerodynamic forces due to deformations (Ns}^2/kg) \]
\[ F_g = \text{Reduced gust forces (N)} \]
\[ f = \text{Local aerodynamic forces (N)} \]
\[ H = \text{Gust wavelength (ft)} \]
\[ h = \text{Vertical displacement (m)} \]
\[ j = \text{Complex number} \]
\[ [K],[k] = \text{Stiffness matrix (FE model, reduced)} \]
\[ k = \text{Reduced frequency} \]
\[ [L] = \text{Elementary loads matrix (FE model)} \]
\[ M = \text{Mach number} \]
\[ [M],[m] = \text{Mass matrix (FE model, reduced)} \]
\[ N_e = \text{Number of mesh elements} \]
\[ [P] = \text{Modal projection matrix} \]
\[ q = \text{Dynamic pressure (Pa)} \]
\[ U_{ds} = \text{Gust amplitude (m/s)} \]
\[ U_g = \text{Gust velocity profile (m/s)} \]
\[ V = \text{Aircraft velocity (m/s)} \]
\[ [V] = \text{Transformation matrix} \]
\[ X = \text{State vector (FE model)} \]
\[ x = \text{Coordinate from nose of the aircraft (m)} \]
\[ x/c = \text{Profile relative chord position} \]
\[ z = \text{Aircraft altitude (ft)} \]

\[ \alpha = \text{Angle of attack (rad)} \]
\[ \alpha_g = \text{Gust induced angle of attack (rad)} \]
\[ \gamma = \text{Reduced modal stiffness matrix} \]
\[ \delta_f = \text{Flap deflection (rad)} \]
\[ \mu = \text{Reduced modal mass matrix} \]
\[ \nu = \text{Reduced modal damping matrix} \]
\[ \omega = \text{Pulsation (rad/s)} \]

Introduction

Gust penetration is a complex phenomenon to model and to take into account when computing the various loads on an aircraft in order to get it certified. Indeed, the regulation requires the aircraft to be able to sustain a certain level of gust. However, it is also a crucial point for aircraft design since the response of the structure to a gust excitation might be one of the criteria used for structural sizing.

This shows how important the computation of these responses is. Indeed, an error in the model used might lead to either an underestimation of the loads induced by the gust or an overestimation. An underestimation might be a critical mistake in the aircraft design process because the characteristics of the structure would be underestimated and thus the designed aircraft could be too weak to sustain real gust loads. In the contrary, an overestimation of the loads would lead to an aircraft heavier than necessary to sustain gust loads. This kind of mistake does not lead to an aircraft failure in real flight but it means an oversized structure for its purpose and thus higher operating and production costs.

Furthermore, given two theoretical distinct models for
gust load, it is rather difficult to state which one is the best. The main way to estimate the relevance of a model is to compare its predictions with measurements but measurements of loads on a structure generated by gust are really hard to perform. This comes from the fact that during flight tests, one cannot generate a ‘clean’ gust to go through and has to deal with what nature provides and during wind tunnel tests (for which it is possible to get the desired excitation), the structure is not the real aircraft.

The approach used in the project presented in this paper is to derive a theoretical model for cylindrical gust, to use it to get some predictions on a real aircraft and to compare its predictions for a wind tunnel test experiment with the experimental data in order to get a process allowing to improve its accuracy. The real aircraft is a business jet of type Falcon from Dassault Aviation and the experimental data come from an experimental campaign performed at the ONERA within an European program Cleansky-SFWA.

In this paper, some theoretical background related to penetration into a cylindrical gust will be presented before deriving three numerical methods to model it. The results obtained with these methods will be compared and the influence of some parameters defined by the methods will be studied. Finally, some experimental data will be analysed and a comparison with the numerical results in a simple case will be performed.

1 Theory and Background

1.1 Equation of Motion and Modal Analysis

In this section, some general thoughts about Finite Elements modelling and modal analysis are presented. To compute the deformations and loads on a structure, a finite element model is usually built. With this model, the equation of motion can be written as (1), in which the finite element model is usually built. With this model, the equation of motion is expressed by the relation (4). The number of elementary loads is significantly smaller than the number of degrees of freedom. This is a good way to improve the time and space efficiency of the computations. The main drawback of this reduction is the loss of some loads which can be expressed on the finite element model but not by linear combination of the elementary loads. However, this drawback can be diminished by a smart choice of the elementary loads.

In order to get an even more time and space-efficient model, it is possible to perform a modal analysis of the reduced system. Here, one has to be careful with the fact that the modes computed by the modal analysis are not the eigen modes of the finite element model (and thus of the structure if the finite element model has been carefully built) but they are the modes of the reduced system. To determine this new basis, one has to solve the eigenvalues problem shown in (3) and get the transfer matrix $[P]$ from the reduced basis to the modal reduced basis, composed by the eigenvectors.

$$\begin{array}{c}
(\left[\mu\right] - \omega^2\left[m\right])\hat{\nu} = 0 \quad \text{(3)}
\end{array}$$

In the reduced modal basis, the mass ($\mu$) and stiffness ($\gamma$) matrices are diagonal which improves further the efficiency of the computations. For each mode it is possible to define a damping coefficient using some empirical or theoretical laws. This leads to a damping matrix ($\nu$) which is also diagonal in the reduced modal basis. After these two reductions, the equation of motion is expressed by the relation (4).

$$\begin{array}{c}
\mu\ddot{\hat{\nu}}(t) + \nu\dot{\hat{\nu}}(t) + \gamma\hat{\nu}(t) = \left[P]\left[V\right]^TF(\mathbf{\tilde{X}},\mathbf{\tilde{X}},X,t,...) \quad \text{(4)}
\end{array}$$

The last step to get rid of the derivatives of the state vector is to go to the Laplace domain. The damping of the Laplace variable is chosen equal to zero which means that the Fourier transform (denoted by a hat) can be used to perform this transfer. After this transformation, the left-hand side of the equation (4) is quite simple. The right-hand side however needs some more work.

From now on, only aerodynamic forces will be considered as external forces and these forces are split in two. The aerodynamic forces on the structure due to gust $F_g$ and the aerodynamic forces due to the deformations of the structure. For instance, when loaded, a wing bends
and this changes the external shape of the aircraft. This new shape induces a change in the aerodynamic forces which leads to a new load on the structure. To be able to model these deformation forces, a Taylor’s expansion of this quantity is performed and thus they can be written \( q F_{\text{def},f} X \), with \( q \) being the dynamic pressure.

The problem is now to compute this \( F_{\text{def},f} \) term. All the aerodynamic computations presented in this paper come from a DLM computation and some adjustments to fit more advanced CFD results, wind tunnel or flight test measurements. The mesh used for DLM computations is different from the one defined by the finite element model far less complex and with much fewer elements. On this mesh a set of elementary deformations are defined and the aerodynamic forces they create is computed (see [1] for more details). Then, any given deformation of the structure is approximated by a linear combination of the elementary deformations and the associated aerodynamic can be computed thanks to the same linear combination to a linear combination of the deformation of the finite element model \( X \).

After all these steps the equation of motion that is used in this paper is shown in (5).

\[
[-\omega^2 \mu - j \omega \nu + \gamma - q F_{\text{def},f}(\omega)] x(\omega) = [P]^T[V]^T \dot{F}_g(\omega)
\] (5)

### 1.2 Gust Definition and Equation of Penetration

In order to be certified, an aircraft needs to be proven able to sustain the loads caused by a discrete gust. In the regulation [2] §25.341 (a), the discrete gust profile is defined by formula (6), where \( H \) is the wavelength of the gust which has to be between 30 ft and 350 ft, \( V \) is the true airspeed (TAS) of the aircraft, \( z \) is the altitude of the aircraft at which it encounters the gust, \( U_{ds} \) is the amplitude of the gust depending on the wavelength, the altitude and some characteristic masses of the aircraft. Figure 1 shows an example of this profile for an amplitude of 1 m/s, an aircraft speed of 50 m/s and \( U_{ds} = 1 \) m/s.

\[
\left\{
\begin{align*}
U_g(t) &= \frac{U_{ds}(H,z)}{2} \left[ 1 - \cos \left( \frac{\pi V}{H} t \right) \right], \quad t \leq \frac{2H}{V} \\
U_g(t) &= 0, \quad t \geq \frac{2H}{V}
\end{align*}
\right.
\] (6)

According to the regulation, this gust profile has to be used for both vertical and lateral gust. In this paper, only the vertical case is studied but the workflow would be the same for lateral gust. One can show that experiencing a gradient of vertical speed is equivalent to a change in the angle of attack of the aircraft [3]. The induced angle of attack is expressed by relation (7), under the assumption that the gust amplitude is smaller in front of the aircraft velocity.

\[
\alpha_g(t) = \frac{-U_g(t)}{V}
\] (7)

Using these definitions, it is possible to derive an expression for the aerodynamic forces due to gust which is given in equation (8) as an integral of the local aerodynamic forces over all the points describing the aircraft structure. These local forces are expressed by a convolution product of the aerodynamic due to a local angle of attack \( (\partial f/\partial \alpha) \) by the angle of attack induced by the gust profile at this local point of coordinate \( x \) (used for the definition of the delay term).

\[
F_g(t) = \int_{-\infty}^{+\infty} \left( \int_{-\infty}^{+\infty} -\frac{\partial f}{\partial \alpha}(x, \tau) \frac{U_g(t - \frac{x}{V} - \tau)}{V} \ d\tau \right) \ dx
\] (8)

Thanks to some usual properties of the Fourier transform, it is possible to derive the equation of penetration of the aircraft into the gust zone (9).

\[
[-\omega^2 \mu - j \omega \nu + \gamma - q F_{\text{def},f}(\omega)] x(\omega) = -\frac{U_g(\omega)}{V} \int_{-\infty}^{+\infty} e^{-j\omega \tau} \frac{\partial f}{\partial \alpha} \ dx
\] (9)

The same work can be done with an oscillating vertical displacement of the structure instead of an angle of attack. In this case, one needs the relation between the vertical position \( h(t) \) and the gust profile \( U_g(t) \) which is given in (10). Once again, using the properties of the Fourier transform, it is possible to derive the equation of motion in the Laplace domain (11).

\[
h(t) = h(0) + \int_{0}^{t} U_g(\tau) \ d\tau
\] (10)
The computations of the aerodynamic forces due to a local angle of attack or a local vertical displacement is realised on a mesh and thus implies a discretisation of the integral in equation (11). The purpose of this paper is to study several ways to perform this transformation, while everything else remains constant. There are two main choices to make when defining a method to model this integral: the number of zones for the discretisation on which the quantity in the integral will be considered constant and the order between the cut into zones and the computation of the aerodynamics induced by the deformed shape of the mesh.

### 1.3 Three Different Models

#### 1.3.1 First Method

The first method uses 3 zones and computes first the aerodynamics due to an angle of attack (or a vertical displacement) of the entire aircraft. Then the loads created by this deformation are distributed on the 3 zones (see Figure 2). These zones are named the forward zone, the middle zone and the rear zone. The forward zone contains all the finite elements of the fuselage placed before the main wings, the rear zone contains the elements of the tail and the rear of the fuselage and the middle zone contains the elements of the main wings and the middle part of the fuselage.

This produces a coarse discretization of the structure but might be enough given the relative size of the gust and of the aircraft. Practically speaking, the aerodynamics due to an angle of attack (or a vertical displacement) applied to the entire structure is obtained by using a Doublet Lattice Method code, then the loads on the finite elements model are computed. This is done thanks to an operator allowing to transform the pressure field on the aerodynamic mesh into a load field on the finite elements mesh. These loads can be further split into basic loads to be expressed by a linear combination of the elementary loads on the finite elements mesh.

Finally, three partial effects are defined, each of them containing the loads on one of the three zones due to the local angle of attack or vertical displacement. To get the final input needed for the computation of a response to gust, one has to multiply each of the three effects by an adapted delay term. This method can be summed up by equation (12).

\[
\int_{AC} e^{-j\omega t} \frac{\partial F}{\partial h}(x,\omega) dx = \sum_{i=1}^{3} e^{-j\omega t} \int_{Z_i} \frac{\partial F}{\partial h}(x,\omega) dx
\]  

This method shows two major limits. The first one is more an approximation which needs to be validated: it uses only three zones. The second one is a bias in the method: the interaction of one zone over another is not excited at the right time. Indeed, the main wing disturbs the air which flows around the tail and this effect can be seen in the loads on the tail. These loads are "stored" in the rear zone and thus excited at the same time as the tail whereas they are produced by the main wing and should be triggered accordingly. The other methods presented below try to evaluate or correct these limits.

#### 1.3.2 Second Method

The second method studied uses either 3 or 10 zones (to evaluate the influence of this parameter) and split the mesh into these zones before the computation of the aerodynamic forces (see Figure 3). Once the N zones are defined (N being either 3 or 10), each of them is placed at the given angle of attack (or vertical position) while the other zones remain at their zero deformation state and the aerodynamic is computed thanks to a DLM code.

This means that for each partial effect, there are loads over the entire structure and not only on the zone experiencing the deformation. This is a good way to get the interactions between the different parts of the aircraft. This method can be formulated by the equation (13), in which \( h_i \) is there to express the fact that even though the vertical displacement used is the same for all the zone, they do not move at the same time and thus the shape used to compute the aerodynamic in the DLM code is different from one effect to the other.

#### Figure 2: Scheme explaining the main steps to compute the aerodynamic forces in the first method.
For this study the number of zones has been limited to 10 because the delay for each zone has to be computed "manually" and added after the computation of the aerodynamic. This is probably the main drawback of this method if one wants to increase the number of zones. Partial Effects on the FE Mesh

\[ \int_{AC} e^{-j\omega t} \frac{\partial F}{\partial h} \, dx = \sum_{i=1}^{N} e^{-j\omega t} \int_{AC} \frac{\partial F}{\partial h_i}(x, \omega) \, dx \quad (13) \]

For a vertical displacement effect which represents the effect of a vertical wind, the rotation ($\alpha$) is zero and the aim is to take into account the delay term in the vertical displacement $h$. This is done in equation (15). It makes sense to delay the pressure field to model the gust penetration because a delay in the pressure field has the consequence to introduce the same delay in the load field (obtained after the use of an operator). Thus, instead of having the delay term out of the aerodynamic forces due to a local angle of attack or a vertical displacement the third method has the delay term already included within the aerodynamic forces one gets through the DLM computation.

\[ C_p(M, k, shape) = A(M, k)^{-1} ([\alpha] + jk[h]) \quad (14) \]

1.3.3 Third Method

The third method tries to overcome the problems highlighted in the second one. The main idea for this method is to compute the aerodynamic (with the DLM code) due to an angle of attack modulated by the delay term along the $x$-axis of the aircraft. This would allow to get rid of the step during which one has to delay "manually" the various partial effects, to have only one effect instead of $N$ and to get as many "zones" as elements in the mesh, $N_e$. Except for this practical aspect, the results of the second and the third methods are expected to be the same if the second method uses as many zones as the third one. In all the computation realised for this paper, the third method uses a mesh containing about 2500 elements thus the number of zones is about 2500.

To get a better understanding of what that means, one needs to look at the formula used in the DLM code. The vector containing the pressure coefficients of each element for a given computation point (Mach number, reduced frequency and deformation of the mesh) is computed with formula (14). In this relation $A^{-1}$ is the influence matrix ($A_{ij}^{-1}$ contains the pressure coefficient of the element $i$ due to an angle of attack of the element $j$), $[\alpha]$ is the vector of rotations of the elements, $[h]$ the vector of vertical displacements and $k$ the reduced frequency ($k = \omega c/V$).

\[ \int_{AC} e^{-j\omega \tau} \frac{\partial F}{\partial h} \, dx = \sum_{i=1}^{N_e} \frac{\partial F}{\partial h_i}(x, \omega) \quad (16) \]

1.4 Experimental Setup

To compare numerical results with experimental data, wind tunnel experiments have been realised. In these experiments, a 2D profile has been fixed to the walls of the tunnel. This profile is an in-house profile from the ONERA named OAT15A, presenting an actionable flap with the hinge axis at 75% of the chord. Its characteristics are presented in Table 1. The wind tunnel is a closed loop, transonic wind tunnel presenting a cross-section in the test section of 800 mm by 760 mm and the test area is 2.2 m long.

<table>
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<th>Property</th>
<th>Value</th>
</tr>
</thead>
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<tr>
<td>Chord $c$</td>
<td>250 mm</td>
</tr>
<tr>
<td>Span $b$</td>
<td>800 mm</td>
</tr>
<tr>
<td>Maximum thickness</td>
<td>12.2 %</td>
</tr>
<tr>
<td>Mach design point</td>
<td>0.73</td>
</tr>
<tr>
<td>Lift coefficient design point</td>
<td>0.65</td>
</tr>
<tr>
<td>Flap axis $x$-position</td>
<td>0.75c</td>
</tr>
</tbody>
</table>

Table 1: Characteristics of the OAT15A profile used for the wind tunnel tests.

Upstream of the test section of the wind tunnel, two identical profiles are used on top of each other to generate turbulent air velocity field. Used harmonically, they produce a vertical speed component following a sine function...
and modelling a kind of gust. On the OAT15A profile, accelerometers and unsteady pressure sensors are placed in order to get measurements of the pressure field on the mid-span chord (both steady and unsteady) and the motion of the profile due to the perturbations in the velocity field.

2 Influences of some Parameters

2.1 Number of Zones

In this section, the influence of the number of zones used in the numerical computations is checked in order to assess whether or not 3 zones yield the loads with a good accuracy. The second and third methods are used to get the results and a distinction is made between the gusts with long and short wavelengths.

2.1.1 Influence over small Wavelength Gusts

The computations presented in this section have been performed for a zero altitude, a wavelength of 90 ft and a fixed aircraft velocity of 180 m/s (M=0.53). In all this paper two kind of loads are studied: a bending moment on the main wing and a bending moment on the horizontal tail. Figure 4 shows a comparison to evaluate the influence of the number of zones (everything else being the same) used in the model.

![Figure 4: Bending moments on the main wing computed by the second method with 3 zones and 10 zones for short wavelengths.](image)

From Figure 4, one can see that the differences are very small between the two computations, about 1% on the first oscillation. It means that 3 zones might not be enough to get results with an accuracy smaller than 1% but the results are almost converged for 3 zones, thus they should be converged for 10 zones.

2.1.2 Influence over Large Wavelength Gusts

The same study is performed for large wavelengths: H=350 ft, for the same speed and altitude. Figure 6 shows the bending moment on the main wing computed by the second method with 3 zones and 10 zones.

![Figure 6: Bending moments on the main wing computed by the second method with 3 zones and 10 zones for large wavelengths.](image)

In order to validate this assumption, Figure 5 presents the comparison between the second method with 10 zones and the third method with \( N_e \) zones (\( N_e \) is about 2500). It seems clear that for 10 zones the loads are converged since the computation with the third method gives the same loads.

The conclusion is the same as the one for short wavelengths. This means that the wavelength does not influence the number of zones required to model gust penetra-
tion. This might not be true if the wavelength is smaller than the size of one zone (which is not the case here).

Figure 7: Bending moments on the main wing computed by the second method with 10 zones and the third method for large wavelengths.

2.2 Treatment of Interactions

2.2.1 Comparison of the Global Effect

In order to evaluate the influence of the change in the treatment of interactions between two zones, the loads yielded by the first method and the second method with three zones (to make sure the differences do not come from the change in the number of zones) are compared. Figure 8 presents this comparison for the bending moments on the main wing and it appears that there is a quite large difference, around 16% on the first peak.

Figure 8: Bending moments on the main wing computed by the first method and the second method with 3 zones.

The same computations but on the bending moment of the horizontal tail yield a very different conclusion (Figure 9). The two curves are very similar even though not exactly the same. To grasp the differences between the two methods, one needs to break down the contribution of each effect on the loads and not only look at the total resultants.

2.2.2 Comparison of the Partial Effects

To break down the contribution of each partial effect, some computations using only one of them as turbulent input have been run. Figures 10 and 11 show that the effect of the middle zone on the horizontal tail as well as the effect of the rear zone on the horizontal tail are not the same at all for the two methods, even though their sum yields a similar general load (Figure 9).

This comes from the fact that the aerodynamic loads due to the main wing on the horizontal tail are not stored in the same partial effect in the first and the second method. Indeed, in the first method they are stored in the third partial effect (the one linked to the rear zone) whereas in the second method, these loads are stored in the second partial effect (middle zone).

Figure 9: Bending moments on the horizontal tail computed by the first method and the second method with 3 zones.

Figure 10: Bending moments on the horizontal tail computed by the first method and the second method with 3 zones, for an excitation only on the middle zone.
This rises the question of the right time to trigger the loads due to the main wing on the tail. As these loads are induced by the changes in the wake of the main wing due to its deformations, they will appear on the tail after a short time. This short delay is taken into account in the DLM computation through the phase shift induced by the imaginary part of the influence coefficients.

Figure 11: Bending moments on the horizontal tail computed by the first method and the second method with 3 zones, for an excitation only on the rear zone.

3 Work on Experimental data

3.1 Analysis of Experimental data

A common assumption to model the pressure coefficient is to use a linear dependency with the angle of attack $\alpha$ and the flap deflection $\delta_f$ (equation (17)). In order to identify the various slopes and to validate the assumption, the experimental data for the steady case (without generation of gust in the wind tunnel) are analysed.

$$C_p(M, \alpha, \delta_f) = C_{p0}(M) + C_{p,\alpha}(M)\alpha + C_{p,\delta_f}(M)\delta_f \quad (17)$$

From the pressure hole measurements, one can compute the pressure difference between the upper surface and the lower surface of the profile and thus obtain the pressure distribution along the chord. From the various experiments (for different values of $\alpha$ and $\delta_f$) one can identify the slopes and check whether or not the linearity assumption is valid. Figure 12 shows three slope distributions obtained for $\delta_f = 0^\circ$ and four angles of attack at $M=0.3$. Figure 13 presents the same curves for $M=0.73$.

It appears that the linearity is quite good at subsonic Mach number ($M=0.3$) at least for angles of attack up to $2.4^\circ$. For transonic Mach number, this is not the case because of the presence of a shock which appears for $\alpha$ between $1^\circ$ and $2^\circ$ at $M=0.73$. The DLM computations yield linear results with the angle of attack which means that at transonic Mach numbers, it is not the most suited method if one wants to get the pressure distribution on the profile.

Figure 12: Pressure coefficient slopes along the relative chord from three experimental measurements, $\delta_f = 0^\circ$ and $M=0.3$.

Figure 13: Pressure coefficient slopes along the relative chord from three experimental measurements, $\delta_f = 0^\circ$ and $M=0.73$.

The same kind of study has been realised for the flap deflection and the conclusions are the same at $M=0.3$ and for $M=0.73$, on the first 40% of the relative chord the pressure coefficient slopes are not linear at all with $\delta_f$ but they become linear after 40% of the relative chord $x/c$.

3.2 Comparison with DLM Computations

3.2.1 Convergence Study

The model used in the wind tunnel is clamped to the walls. This prevents the recirculation of the flow (air flowing from the lower side of the wing to the upper side of the wing at...
wingtips). In order to render this aspect in the numerical computations, there are two strategies: the use of meshed walls to model the tunnel walls or the use of a much larger span of the wing to reduce the influence of the wingtip recirculation on the pressure field of the mid-span chord.

A convergence study on the span has been performed and Figure 14 shows its results. The span of the experimental model is \(b_0 = 0.8\) m, and the spans used for the study are \(b_0\), \(3b_0\), \(5b_0\) and \(10b_0\). The DLM computations have been run on meshes containing elements of the same size (only their number varies with the span), so that the differences are only due to the change of the span and not to a finer or a coarser mesh.

![Figure 14: Pressure coefficient distribution along the relative mid-span chord from DLM computations for various spans of the model.](image)

It appears that for \(b = b_0\) the results have not converged. It gets better for \(b = 10b_0\) which yields almost the same predictions than the case with \(b = 5b_0\), so that one might consider that the convergence is reached for \(b = 10b_0\). In the following computations, the pressure field will be computed over a flat plate with a span of \(10b_0\) and only the coefficients on a subdivision of the plate of span \(b_0\) in the middle of the large one will be considered.

Thanks to this method, one gets the pressure coefficients on a flat plate of span \(b_0\) as if there were vertical walls at wingtips, which is basically the experimental setup. For a finer modelisation, one should look into the influence of the upper and lower walls on the flow around the profile. However, this level of details would not really make sense for a DLM computation and a RANS or Euler CFD case should be set-up.

### 3.2.2 Experimental-DLM Comparison

From the previous DLM computations, it is possible to get the pressure coefficient distribution along the relative mid-span chord and to compare it with the experimental measurements. This comparison makes sense only for subsonic Mach number (\(M=0.3\)) for which there is no nonlinear phenomenon such as a shock. Figure 15 presents this comparison for an angle of attack effect and Figure 16 for a flap deflection.

![Figure 15: Pressure coefficient distribution along the relative mid-span chord from DLM computations and experimental measurements, due to an angle of attack effect (\(M=0.3\)).](image)

For the angle of attack effect, the DLM predictions are in good agreement with the experimental results and a simple fitting would suffice to get an almost perfect match. Nevertheless, there is a big discrepancy between the numeric and experimental results at the leading edge but that is a well-known feature of DLM computations.

![Figure 16: Pressure coefficient distribution along the relative mid-span chord from DLM computations and experimental measurements for a flap deflection effect (\(M=0.3\)).](image)
Concerning the flap deflection effect, the agreement is not as good as for the angle of attack. There is still the divergence at the leading edge but another divergence appears at the hinge axis of the flap, where numeric results predict a pressure peak whereas the experiments show nothing special at this $x$-coordinate. This might come from a lack of pressure holes close to the hinge axis and thus a lack of experimental points at this location.

On the main body of the wing (before 75% of the relative chord), the numerical predictions match quite well the experimental measurements.

Conclusion

Three methods to model gust have been compared. It appeared that the predicted loads are somehow similar from one method to another even though there are some major differences from a physical point of view. It has been observed that the influence of the number of zones used to discretize the gust effects on the structure plays a small role since the loads obtained with 3 zones and some thousands of zones are very close.

What really changes from one method to another is the way to treat the interaction between two zones. This plays an important role on the loads induced by one zone on the entire aircraft but when combining all the contributions, the differences partially compensate each other and the total difference does not appear as big as assumed.

The fact that even if the models are different, their results are close tends to show that there is no major mistake or wrong approximation in these models. This validates to some extent the approximations made in the first method as well as the theoretical models developed for the three methods. The next step is to get their results as accurate as possible according to the wind tunnel tests.

This means that a method to fit the DLM results on the experimental measurements has to be developed and validated for the cases without gust in the tunnel and once this fitting has been determined, the numerical and experimental results can be compared. From this comparison, three conclusions might arise: the gust model yields predictions that are too far from the experimental results and should not be trusted, the gust model catches the trends and levels with a certain accuracy and thanks to an adapted fitting, it is possible to get reliable results or the gust model is good enough without fitting and its results are already reliable.

References

