ISSN (ONLINE): 2321-3051

INTERNATIONAL JOURNAL OF RESEARCH IN AERONAUTICAL AND MECHANICAL ENGINEERING

Transformation of Dynamic Loads on Airplane structures into Equivalent Static Loads by VLM approach

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Abstract

Dynamic loads are complex to analyze and finite element method is utilized for most accurate results which captures considerable CPU time. Static loads are utilized with dynamic influence factors to quantify the structural parameters. But the dynamic factors can be calculated only by codes or by experience. Therefore, static loads assumption cannot offer precise results in analysis and design and typically the results are often unreliable. For large scale problems like commercial aircraft wings Equivalent Static Load (ESL) approach is proposed for structural optimization. Dynamic loads can be well assumed to considerable extend with this approach. Vortex Lattice Method (VLM) is used for the calculation of the dynamic load distribution at equal intervals using flow analysis simulation. ESL from the simulation process is taken to approximate its effects in the structural deformation. Displacement based approach is proposed in this regard and then it will be implemented to assess the net effect on different location of the aircraft. The effect of ESL on performance will be assessed through experiments. CPU time is minimized and efficient results are expected. Airloads fixed at different locations of aircraft can be assessed with maximum accuracy.

Keywords— Equivalent Static Loads, Approximate Methods, Vortex Lattice Method, Dynamic Vibrations

1. Introduction

All the forces exist in the real world act dynamically on the airplane surface. It is difficult to analyze the dynamic loads changing with respect to time but it can be solved by changing it in to static loads. Therefore it is significant to transform a dynamic load into a proper static load for the good evaluation and the definition of the applied load in design procedure. The external forces are given as an input to the Finite Element Analysis (FEA) tools, which is one of the best choices for computational analysis [1], [3]. Transient analysis demonstrates the real and precise phenomena of structures under dynamic loading conditions. But Transient analysis is extremely complex and pricey in nature because of the large computing requirements. Therefore, the equivalent static loads are normally used in the optimization process. Structural optimization seems to be successful for static loads with dynamic design factors [8], [12].

The conversion of dynamic loads into equivalent static loads has been studied in civil engineering for analysis and design under earthquake loads. Researchers have applied optimal criteria and techniques for two-dimensional structures using time history analysis procedures coupled with actual seismic excitations [4]. Structural optimization is subjected to random seismic excitations and reliability theory is still in its development stage. However, these researches are limited to seismic loads and the transformation is based on experimental codes which lacks generalization [2]. Some other researchers are focused on load identification, which is one of the inverse problems. Especially, the so-called load identification problem in structural dynamics is not only an inverse problem but also ill-posed [4], since the response typically is a continuous vector valued function of the spatial coordinates, whereas the response is defined at only a few points [5] and [6].

Thus, no unique solution is guaranteed using the reliability theory and inverse problem approach. Moreover, solutions are frequently found to be unstable in the sense that small perturbations of input data result in large changes in the calculated force magnitudes. To find a unique solution, presumptions are needed, such as the shape or the location of the load, so the presumption tends to determine the quality of the solution. The quality of the solution can be determined by how well it simulates the original response and the robustness of the solution in the case of the perturbation of the input. To meet the quality and to overcome the absence of a unique solution, an equivalent static load (ESL) is used whose inputs are calculated from Vortex Lattice Method (VLM).

2. Problem Description

Aircraft design is the special consideration between the numbers of challenging factors. One of the most important requirements in aircraft conceptual design is the airfoil selection for lifting surfaces. For accessing the aircraft usage in design phase, calculation of aerodynamic coefficients and their derivatives are important parameters. The lift and drag coefficients of lifting surfaces determined by extracting the pressure distribution of flow field around the geometry. For large scale problem like aircraft wings Equivalent Static Load (ESL) approach is proposed for structural optimization. Computational Fluid Dynamics (CFD) analysis is used to simulate the flow field; one of the numerical approaches is Vortex Lattice Method (VLM). VLM is used to compute the force distribution over the aircraft lifting surfaces like wing and tail planes where the influence of thickness and viscous effects are neglected. Flow field around the fairly complex geometry like tapered wing is computed by considering the wing as infinite thin sheet of discrete vortices for required resolution and accuracy. The dynamic air loads acting on the aircraft wing surfaces during flights are fixed at different locations of load bearing parts. Hence, the unnecessary drag force must be balanced with the required thrust in steady level flight. This numerical approach is benefited for the arrival of computers for large amounts of computations that are required in less time.

A) Vortex Lattice Method

The vortex lattice method (VLM) is a potential flow solver based on the lifting line model. It represents the wing as a planar surface on which a grid of horseshoe vortices is superimposed. Although this definition is correct for the classical method, there are several VLM that can modify that statement. In this section only the classical implementation is described (fig 1), so the basic horseshoe vortex model is taken into account.

The following assumptions are made regarding the problem in VLM; the flow field is incompressible, inviscid and irrotational, the influence of thickness on aerodynamic forces is neglected and small angle approximation [14]. From the Kutta-Joukowski theorem, it is known that the vortices represent the lift vector. One popular approach to place the vortex and then satisfy the boundary condition at zero normal velocity on the wing surface is the "1/4 - 3/4 rule". The procedure is:

- Divide the plan form up into a pattern of quadrilateral panels, and then put a horseshoe vortex on every panel.
- Put the bound vortex of the horseshoe vortex on the 1/4 chord element line of each panel.
- Position a collocation point on the 3/4 chord point of every panel at the middle in the spanwise direction
- Assume a flat wake in the common conventional method
- Establish the strengths of each Γ_n necessary to satisfy the boundary conditions by solving system of linear equations.
- Calculation of the forces and moments by using the Kuttta-Joukowski theorem.

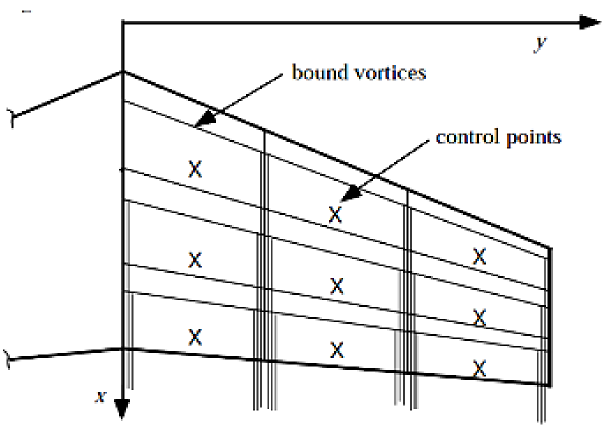

Figure 1. Classical Vortex Lattice Method (VLM)

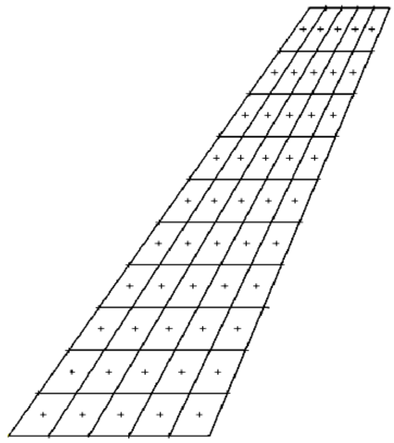

Figure 2. VLM applied on wing planform

B) Governing Equations:

The discretized wing has to satisfy the zero normal flow condition and the boundary condition is defined as an addition of the known free stream speed V_∞ and the unknown velocities induced by the wing bound vortices V_b:

$$(V_\infty + V_b).n = 0 \tag{1}$$

Eq. (1) is solved at every collocation point by following a straightforward process. A calculation of the induced velocities starts with Boundary condition for Γ = 1, in a horseshoe model. The outcome is the influence coefficients, which are defined in the form of Eq. (2)

$$a_{ij} = (u, v.w)_{ij}.n_i \tag{2}$$

Where n is the normal vector to the panel i and $(u, v.w)_{ij}$ are the velocities induced on the collocation point *i* by the vortex *j*. From Eq. (1) and (2) a linear system of equations (3) is set

$$\begin{bmatrix} a_{11} & a_{12} & \dots & a_{1N} \\ a_{21} & a_{22} & \dots & a_{2N} \\ . & & & . \\ . & & & . \\ a_{N1} & a_{N2} & \dots & a_{NN} \end{bmatrix} \begin{bmatrix} \Gamma_1 \\ \Gamma_2 \\ . \\ . \\ \Gamma_N \end{bmatrix} = \begin{bmatrix} -V_\infty.n_1 \\ -V_\infty.n_2 \\ . \\ . \\ -V_\infty.n_N \end{bmatrix} \tag{3}$$

Where the circulations Γ_j are the unknowns. After the unknowns are calculated, the lift generated by each bound vortex is obtained through Kutta-Joukowski theorem. In this case, it is posed in a discretized way as Eq. (4).

$$\Delta L_i = \rho V_\infty \times \Gamma_i \, \Delta b_i \tag{4}$$

Where ΔL_i is the lift of the panel 'i' and Δb_i is the length of the bound vortex of the same panel. To obtain the total lift of the wing, the lifts of all the bound vortices of the wing are summed.

3. Mathematical Model

A) Geometry Selection

Although NACA experimented with many theoretical methods for 2-Series through the 5-Series, none of the methods was found to produce the preferred airfoil behavior. The 6-Series airfoil was derived using an advanced theoretical method that, like the 1-Series, depend on specifying the preferred pressure distribution and used advanced mathematical calculation to derive the required geometry. The purpose of the approach was to design airfoils that maximized the region over which the airflow remains laminar. By doing so, the drag over a small range of lift coefficients can be significantly decreased. One of the more common NACA 6 series airfoil is selected for analysis, NACA 64-009.

B) Wing Model

As stated, wing of a large aircraft is taken for calculating the Equivalent Static Load problem. It is because of the fact that a large wing plan form will give a better understanding of the problem. So it is decided to have the wing of the large aircraft.

C) Application of VLM

Vortex Lattice Method is used Explicit and Implicit route for generating unsteady wakes. Body-fixed reference frame is used for the problem i.e. wing is viewed as moving through air either at rest or in motion. Effects of gust on maneuver can be modeled.

The given wing model is divided into many panels for calculation of forces as shown in Fig.2.Considering the various chord wise and spanwise location the forces are calculated from various theoritical formulaes available. From the Fig.2, it is evident that chord wise the wing is divided to five parts and spanwise it is ten parts.

IV. NUMERICAL SIMULATION

Ansys structural analysis is carried out to study the deformation and stress distribution on the wing surface for the calculated air loads by VLM approach. The selected tapered wing domain have been modeled and meshed using the design software to solve the problem with desired level of accuracy. The element boundary is defined as the cantilever beam of aluminum alloy. The load is applied on the nodal points of descritized panel locations for the free stream velocity of 233 m/s. Table-1 describes the mesh details of the wing model designed as shown in Fig (3).

Table- 1

S.NO	Parameter	Values
1.	Number of Elements	108072
2.	Number of Nodes	175919
3.	Number of panels	50

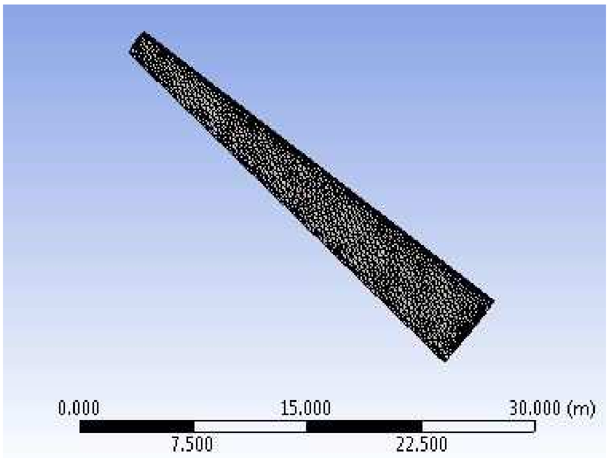

Figure 3. Meshed wing Model using HYPERMESH tool

4. Results and Discussion

The structural behavior of the wing structure against the applied Uniformly Varying Load (UVL) is presented in Fig.4. During the cruising flight, the load distribution over the wing is not uniform due to the unsteady aerodynamic forces and the stress distributed from the root to tip from higher to lower value.

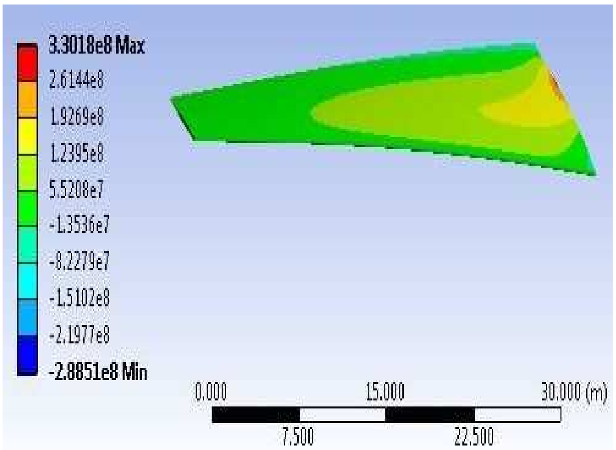

Figure 4. Stress Contour from the root to tip of wing model

Wing structure is a cantilever beam element in which the maximum stress occurred in the fixed end of the wing. Stress value varying linearly towards span wise direction. Force distribution per unit area is least at the free edge of the geometric portion. It hits the peak value at the preset corner. By examining the flow over the wing surface, design coefficients are arrived.

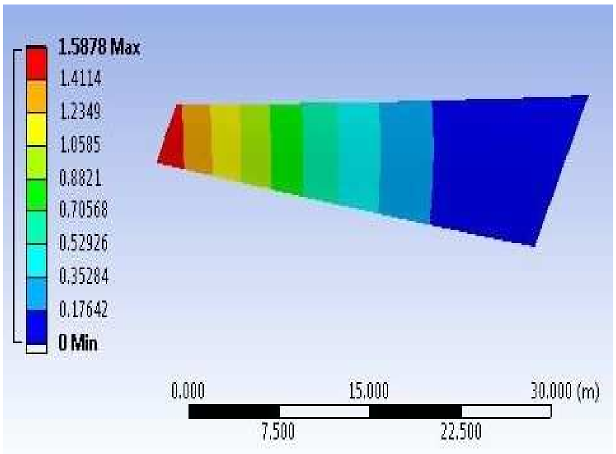

Figure 5. Total Deformation Contour for the wing model

Figure.5 illustrates the total deformation of wing structure, results from a stress field induced by the applied forces due to changes in air loads. The deformation smoothly grows towards the wing span. Deformation hit the peak value at the tip of the leading edge, demonstrates the load carrying capacity of the aircraft wing structure to the gust load.

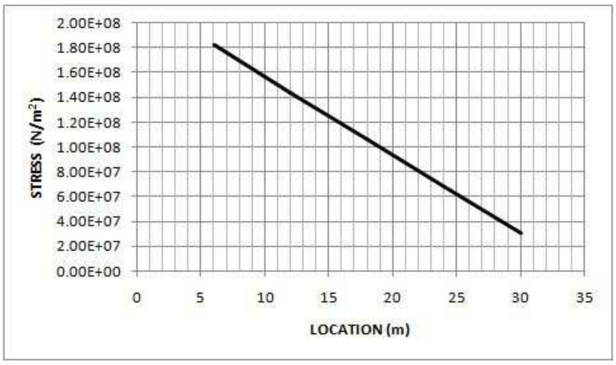

Figure 6. Stress Vs Span wise Location for the Model

Figure 7. Total Deformation Vs Span wise locations

Fig (6) and Fig (7) is the graphical representation for numerical variation of stress and corresponding deformation of wing at different locations for the given operating conditions by VLM.

5. Conclusions

The analytical and numerical simulation results reveal that, by utilizing Vortex Lattice Method the precise allocation of structural behavior on lifting surfaces of an aircraft can be obtained. The results are matching with the theoretical prediction of stresses and deformations by mechanics of materials approach. For different operating conditions, the maneuvering loads acting on an aircraft are dynamically changing. The distribution of equivalent static loads of such dynamic forces on the thin complex structure is derived numerically with less computational time. Aerodynamic coefficients like lift coefficient, drag coefficient for the design structure are simulated well using this approach.

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