Flutter Estimation for Low Speed Aircraft Wing Using Fully Coupled Fluid – Structure Interaction

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ABSTRACT

The aero elastic responses and the flutter condition of 3-D flexible aircraft wing were estimated by developed fully coupled fluid-structure interaction approach. The actual wing in this approach was represented by an equivalent plate .Equivalent plate model (structure model) based on assumed mode method was then combined with unsteady panel-discrete vortex method (aerodynamic model) to build relatively simple aeroelastic model. This model could be used for estimation of flutter condition of moderate to high aspect ratio and low sweep wings of aircraft flight at low subsonic speeds. The obtained results from the present model are able to prediction the flutter condition of the actual wing at different angles of attack. The increasing in the angle of attack leads to reduce flutter speed and flutter frequency.

Keywords: Aeroelasticity, Flutter, Fluid-Structure Interaction

تخمين الرفرفة لجناح طائرة منخفضة السرعة باستخدام تداخل التام للمائع –الهيكل

الخلاصة

الاستجابة الديناميكية الهوائية و حالة الرفرفة لنموذج جناح مرن ثلاثي الأبعاد خمنت بواسطة تطوير اسلوب تداخل التام للمائع –الهيكل. الجناح الحقيقي بهذا الاسلوب مثل بشكل صفيحة مكافئة. اعتمد مبدأ التزاوج بين نموذج الصفيحة المكافئة (الجناح) و النموذج الديناميكي الهوائي الذي تم استخدام طريقة الأشرطة غير المستقرة الهجينة لبناء نموذج بسيط نسبيا للمرونة الهوائية .هذا النموذج استطاع التنبئ بحالة الرفرفة لاجنحة ذات نسب باعية متوسطة الى مرتفعة القيمة عند سرع دون الصوتية المنخفضة. اظهرت النتائج مقدرة الطريقة المقترحة بالتنبوء بحالة الرفرفة عند مختلف زوايا الهجوم . كما بينت ان زيادة زاوية الهجوم تقلل من سرعة الرفرفة وتردد الرفرفة.

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NOMENCLACURE

A_n=adjacent area of the node (m^2)
a=semi span of the wing (m)
b=chord of the wing (m)
C_P= pressure coefficient
E=modules of elasticity (Pa)
h= plate thickness (m)
F_n= concentrated force (N)
h_{root}=plate thickness at root (m)
h_{tip}= plate thickness at tip (m)
k_{ij}=stiffness element (N/m)
L=length of panel (m)
m= mass per unit area (kg/m^2)
m_{ij}=mass element (kg)
$\vec{n}$ = Normal unit vector
Q_i=generalized force (N)
q_i= generalized coordinate (m)
u=velocity component in x-direction (m/sec)
u_s=deformation of mid surface in y-direction (m)
v_s= deformation of mid surface in y-direction (m)
v=velocity component in z-direction
V_∞ (m/sec) =air velocity (m/sec)
V_{FR}=induced velocity due to unsteady motion of wing (m/sec)
α_{eff} = effective angle of attack (rad)
Γ =circulation (m^2/sec)
γ=vorticity strength (m/sec)
γ_{xy} =shear strain
$\Delta P =$ pressure difference (Pa)
$\Delta t =$ time step (sec)
δ=node displacement (m)
ε_x =normal strain in x-direction
ε_y = normal strain in y-direction
υ =Poisson's ratio
ϕ = velocity potential (m^2/sec)
ψ_i =coordinate function

INTRODUCTION

The aircrafts wing, flutter is the most important phenomenon in aeroelastic field and is the most difficult to predict. It is an unstable self-excited vibration in which the structure extracts energy from the air stream and often results in catastrophic structural failure [1].

Many theoretical methods were suggested by researchers like [2, 3, 4, 5, 6, and 7] for estimating flutter speed and flutter frequency of aircraft wing. In the closed form models, the estimation of flutter condition was based on identification of the critical condition where one of roots had zero real parts. In the numerical models (like the present suggested approach) the flutter condition may be estimated by observing the aero elastic response of the fully coupled fluid – structure interaction (FSI) problem for the aircraft wing.

All existing 3-D FSI models have many limitations and assumption which are considered sources of difficulties and errors. Some of theses limitations and difficulties are: the representation of the structure model of the wings with all details of their elements, the mathematical formulation of the aerodynamic forces which are transferred to structure boundaries and the fitting surface curve to obtain the pressure distribution as function of coordinates system. However, fitting in three dimensions is inaccurate and needs a huge amount of computer memory and processing time during execution.

The present approach avoids the above limitations and difficulties through:

1- Represent the actual wing with all structural elements like ribs, spurs … by equivalent plate.

2- The aerodynamic pressures are converted to concentrated forces by multiplying the value of pressure at each node by adjacent area of the node. Then these forces will be introduced to the structure equation as discrete forces. This reduces the mathematical processes, resulting data and save time.

In present FSI approach, a hybrid panel-discrete vortex unsteady method combined with the numerical lifting line method is used to describe the aerodynamic model. While the equivalent plate technique [8] which relays on a solved plate equation by an assumed mode method is used to represent the structure wing model. The resulting FSI is then used for estimation of flutter speed and flutter frequency from the behavior of the generalized displacement of the equivalent plate in time and frequency domains.

AERODYNAMIC MODEL

An unsteady panel–discrete vortex method using MATLAB computer program is devised to estimate the unsteady aerodynamic forces acting on vibrating wing-flap configuration, Figure (1).

In this approach the airfoil section surface is divided into a number of panels. Each panel has vortices strength ($\gamma_i(s)$) which change linearly along panel as given in the following equation:

$$\gamma(s_j) = \gamma_j + \frac{s_j}{L_j}(\gamma_{j+1} - \gamma_j) \qquad \dots (1)$$

The wing has unsteady motion (eg. heaving motion), wake vortices will be created behind the wing therefore; the velocity components at each mid point of panel are

$$u_{pi} = V_\infty \cos\alpha + \sum_{j=1}^{N} u_{vij} + \sum_{k=1}^{NV} u_{vik} + V_{FR\,iy}$$

$$v_{pi} = V_\infty \sin\alpha + \sum_{j=1}^{N} v_{vij} + \sum_{k=1}^{NV} v_{vik} + V_{FR\,iz} \qquad \dots (2)$$

In heaving motion the induced velocity $V_{FR\,iy} = 0$ and V_{FRiy} may be approximated by [9]

$$V_{FRiz} = \frac{\delta_{im} - \delta_{im-1}}{\Delta t} \qquad \dots (3)$$

Where δ_i is the displacement of each mid point of panel at each time step comes from vibrated plate (wing).

By using the flow tangency condition (4), Kutta condition Equation(5) and condition of constant circulation around airfoil Equation (6)

$$\vec{V}.\vec{n} = 0 \qquad \dots (4)$$

$$\gamma_1 = \gamma_{N+1} = 0 \qquad \dots (5)$$

$$\Gamma_m + \sum_{k=1}^{NV} \Gamma_{vk} = 0 \qquad \dots (6)$$

Where

$$\Gamma_m = \sum_{J=1}^{N} L_j [(\gamma_j)_m + (\gamma_{j+1})_m] \qquad \dots (7)$$

This leads to the set of linear algebraic equations with unknowns γ_j and Γ_k .These equations are solved by using Gauss elimination with partial pivoting technique to find γ_i and Γ_k [11]. Then the unsteady pressure coefficient at each time step can be obtained from

$$Cp_i = 1 - \frac{\gamma_i^2}{V_\infty^2} - \frac{2}{V_\infty^2}\left(\frac{\phi_m - \phi_{m-1}}{\Delta t}\right)_i \qquad \dots (8)$$

Equation (8) was derived from the unsteady Bernoulli's equation. The unsteady aerodynamic coefficients at different sections along the semi span at time t_m are determined by integrating the pressure distribution.

To extend the 2-D aerodynamics solution to 3-D, the effective angle of attack at each section along the span must be taken into consideration. The numerical solution for the lifting line theory developed by Anderson, et al [10] is used to determine the angle at each section. Then the unsteady effective angle may be calculated from Equation (9) [11]

$$\alpha_{eff} = angle\ of\ attack\ at\ each\ section + \tan^{-1}\frac{heaving\ velocity}{flow\ velocity} \qquad \dots (9)$$

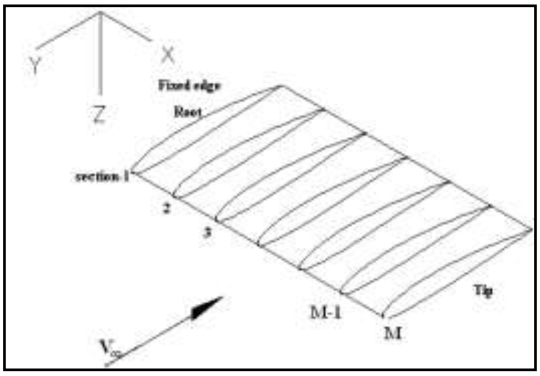

Figure (1) three dimensional wings.

STRUCTURAL MODEL OF A THREE DIMENSIONAL WING

The procedure of Ref.[8] were performed in order to create full scale equivalent plate model for the wing has semispan 4.5 m and chord 1.33m with airfoil NASA GA(W)-1.

The assumed –mode method is used in the derivation of the equations of motion for the plate. This method depends on assuming suitable solution to the displacements of the problem. In plate problem the displacements are assumed to be of the form

$$w(x,y,t)=\psi_1(x,y)q_1(t)+\psi_2(x,y)q_2(t)+\psi_3(x,y)q_3(t)+.........\psi_i(x,y)q_i(t) \quad \ldots (10)$$

Where $q_i(t)$ generalized coordinates, and $\psi_i(x,y)=X_m(x)Y_n(y)$ are the admissible beam functions. $X_m(x)$ satisfy clamped-free boundary conditions and $Y_n(y)$ free-free conditions, which are defined as follows [12]:

$$X_m(x)=\mu_m(\cosh\alpha_m\frac{x}{a}-\cos\alpha_m\frac{x}{a})-\upsilon_m(\sinh\alpha_m\frac{x}{a}-\sin\alpha_m\frac{x}{a})$$

$$\mu_m=\frac{\cosh\alpha_m+\cos\alpha_m}{\sinh\alpha_m\sin\alpha_m},\qquad \upsilon_m=\frac{\sinh\alpha_m-\sin\alpha_m}{\sinh\alpha_m\sin\alpha_m} \quad \ldots (11)$$

Where

$$\alpha_1=1.875,\alpha_2=4.694,\alpha_3=7.854,....\ Y_1(y)=1 \quad , \quad Y_2(y)=\sqrt{3}(2\frac{y}{b}-1)$$

$$Y_n(y)=\kappa_n(\cosh\beta_n\frac{y}{b}+\cos\beta_n\frac{y}{b})-\lambda_n(\sinh\beta_n\frac{y}{b}+\sin\beta_n\frac{y}{b})$$

$$\kappa_n = \frac{\cosh\beta_n - \cos\beta_n}{\sinh\beta_n \sin\beta_n}, \qquad \lambda_n = \frac{\sinh\beta_n + \sin\beta_n}{\sinh\beta_n \sin\beta_n} \qquad \dots (12)$$

Where β_n are the roots of $\cosh\beta_n \cos\beta_n = 1$

The strain energy of plate is

$$U = \frac{1}{2}\frac{Eh^3}{1-\upsilon^2}\iint_A (\nabla^2 w)^2 + 2(1-\upsilon)[(w_{xy})^2 - w_{xx}w_{yy}] \quad dA \qquad \dots (13)$$

The plate kinetic energy is

$$T = \frac{1}{2}\iint_A m\dot{w}^2 \, dxdy \qquad \dots (14)$$

Substituting Equation. (10) into Eqs (13&14) and applying Lagrange's equation the differential equations of plate motion are obtained

$$[m]\left\{\ddot{q}\right\} + [k]\{q\} = \{Q\} \qquad \dots (15)$$

Where

$$m_{ij} = \int_A m\psi_i\psi_j dA, \qquad k_{ij} = \sum_{q=1}^{2}\sum_{p=1}^{2} a_{ijpq} + 2(1-\upsilon)(b_{ij} - a_{ij12})$$

$$a_{ijpq} = \frac{Eh^3}{12(1-\upsilon^2)}\iint_A \frac{\partial^2\psi_i}{\partial x_p^2}\frac{\partial^2\psi_j}{\partial x_q^2} dA \quad , b_{ij} = \frac{Eh^3}{12(1-\upsilon^2)}\iint_A \frac{\partial^2\psi_i}{\partial x_1 \partial x_2}\frac{\partial^2\psi_j}{\partial x_1 \partial x_2} dA$$

The thickness along the semispan was taken as linear function as Equation (16)

$$h(x) = \frac{(h_{tip} - h_{root})}{a}x + h_{root} \qquad \dots (16)$$

By tuning the h_{root} and h_{tip} .The error between the equivalent plate and the actual wing can be minimized.

FULLY COUPLED FLUID-STRUCTURE INTERACTION PROCEDURE

The essential feature of the present aeroleastic model is the innovative way of interaction between the structure model and aerodynamic model. The actual aircraft wing, in structure model, is solved as an equivalent plate while it retains its actual geometry during aerodynamic solution.

The exchange of structural displacements and aerodynamic forcing between the two solutions is illustrated schematically in Figure (2).The wing is divided into N panels and M sections in semispan. Therefore, there are N×M nodes. Each node on the equivalent plate exactly faces the nodes on upper and lower surfaces of the wing section in aerodynamic model. During vibration, the upper and lower nodes (A and B) on the wing section take their displacements from corresponding node P on the equivalent plate.

The solution starts by assuming that the wing is disturbed by initial displacements. Equation (15) is solved by using Runge-Kutta method to obtain generalized displacements. Implementation of Equation (10) the displacement of the plate at each node and at each section is determined. These displacements are fed to the aerodynamic model. Solving this model gives pressure difference (Δp) at each node and at each time step (one iteration). Finally these pressure differences are fed to the structural model (equivalent plate) Equation (15) as a part of the generalized forces Q_i .Equation.(17)

$$Q_i = \iint_A \Delta p(x, y, t)\psi_i(x, y)dxdy \qquad \dots (17)$$

During the development of the present model the pressure differences in equation (17) were found as a mathematical function of the independent variables *x* & *y* of nodes plate at each time step. This can be achieved by using curve fitting between pressure differences and *x,y* coordinates of nodes at each time step . However, it was found that more accurate and economic solution may be obtained by converting the pressure (ΔP_n) to concentrated force (F_n) by multiplying the value of pressure at each node by adjacent area of the node (A_n). Then these forces (F_n) will be introduced into Eq.(17) as shown below:

$$Q_i = -\sum_{n=1}^{NS} F_n\psi(x_n, y_n) + F_o \sin(wt)\psi(x_o, y_o) \qquad \dots (18)$$

Where xn and y_n are the position of the nodes on the equivalent plate; x_o and y_o are the position of excitation force (if exist) on the plate as shown in Figure (3). F_n can be written as

$$F_n = A_n \Delta p_n \qquad \dots (19)$$

Numerical experiments in the present work have shown that the time interval is a very important parameter to obtain physically realistic results.

It was observed that numerical instability can occur depending on the chosen time interval for certain air speed and certain plate .One should be very careful when analyzing the results, to avoid a numerical instability to be interpreted as a physical instability (flutter). This problem has been resolved by taking an initial value for the time interval. This value was then decreased until no noticeable difference between consecutive solutions was observed.

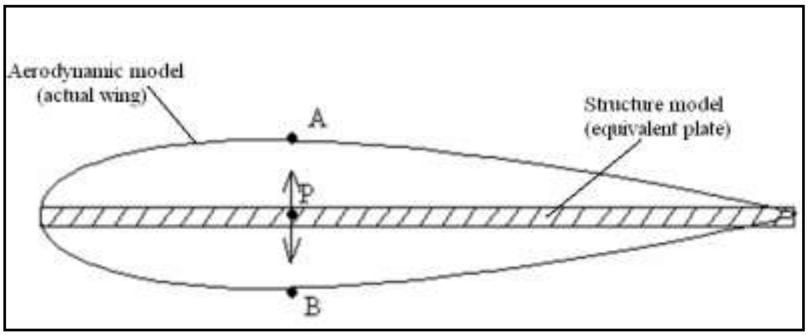

Figure (2) The relation ship between structure and aerodynamic models.

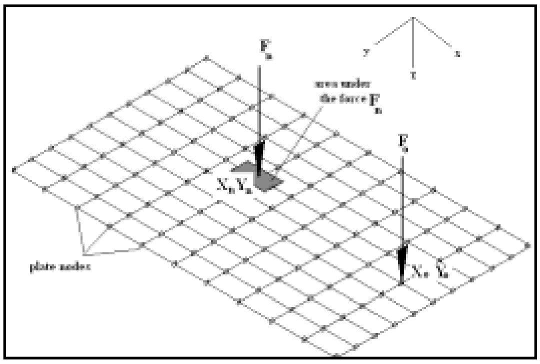

Figure (3) Forces positions on the plate surfaces.

RESULTS AND DISCUSSION

The present FSI technique is applied to compute the dynamic response of the wing from the solution of the equivalent plate. For calculation the numbers of spanwise sections are 10, surface panels are 60 and the physical time step is equal to 0.0005sec

Figures (4 to 8) show the time responses and the fast Fourier transformation (FFT) of the first four generalized modes of the wing model at different air speeds with zero angle of attack and zero flap angles. When the speed is below the maximum design cruising speed of the aircraft (73m/sec) the aerodynamic damping is strongly evidenced for all modes as shown in Figures (4 and 5), and the responses reach finally to the wing static equilibrium position. The wing is vibrating at different damping frequencies for each mode

In Figure (6) the speed is increased to (130 m/sec). This value seems closer to the neutral stability point. The amplitudes of all modes are still decrease in time. In Figure (7), the neutral response appears at V=153 m/sec. The generalized displacements in all modes show a periodic motion, that is a limit cycle oscillation (LCO). All the modes are vibrating at identical frequency, that is the flutter frequency. According to Figure (7) the flutter speed and frequency for the wing model are 153 m/sec and 15.5 Hz respectively.

Beyond the flutter speed, the amplitudes of the generalized displacement grow rapidly; a diverging oscillation is then reached as shown in Figure (8). The frequency is the same for all modes and equal to the flutter frequency. An unstable self-excited vibration will be occurred in which the structure extracts energy from the air stream and often result in structural failure. The general trend of the results in this section seems to be in good agreement with references [4 and 6].

The aeroelastic responses of the wing during a flutter condition in 3-D are shown in Figure (9). The effect of LCO is pronounced on the aeroelasticity responses of the wing.

Figure (10 and 11) show the effect of angle of attack on the flutter speed and flutter frequency of the actual wing respectively. These figures indicate that the aircraft may approach to flutter condition at high angles of attack even if the aircraft speed was less than flutter speed at zero angle of attack. It should be noted that the general trend of the results in Figures (10 and 11) seems to be in good agreement with reference [13]

From a practical point of view, the pilot must reduce the speed of his aircraft when the wing exhibits a (LCO) at high angle of attack to speed less than the critical value. This will prevent catastrophic and will increase its survivability. The aircraft designer must be careful for this situation and try to make the maximum cruising speed below the onset of flutter within the aircraft flight envelope.

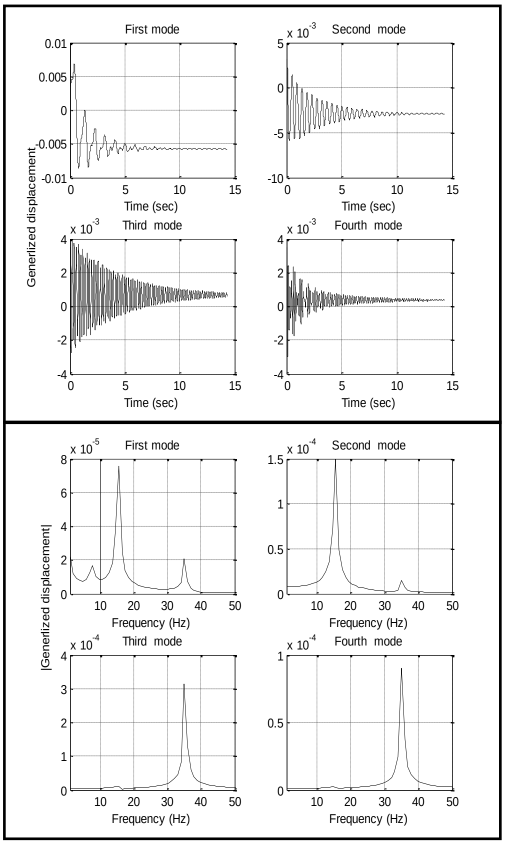

Figure (4) Responses and FFT of the generalized Displacements for the rectangular wing at V=53 (m/sec).

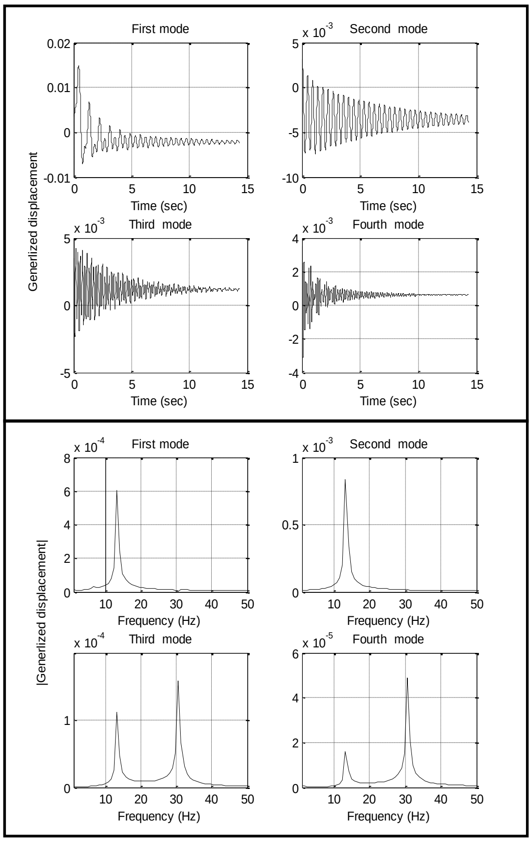

Figure (5) Responses and FFT of the generalized Displacements for the rectangular wing at V=70

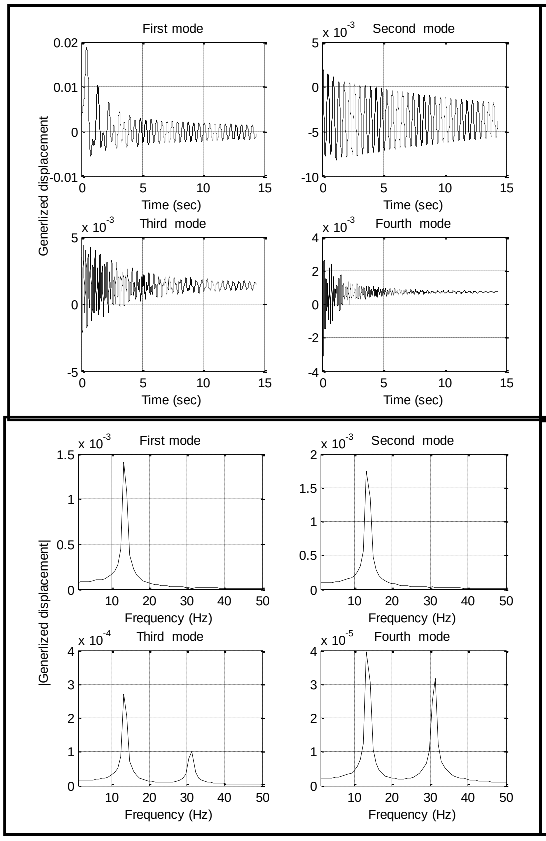

Figure (6) Responses and FFT of the generaliz displacements for the rectangular wing at V=130.

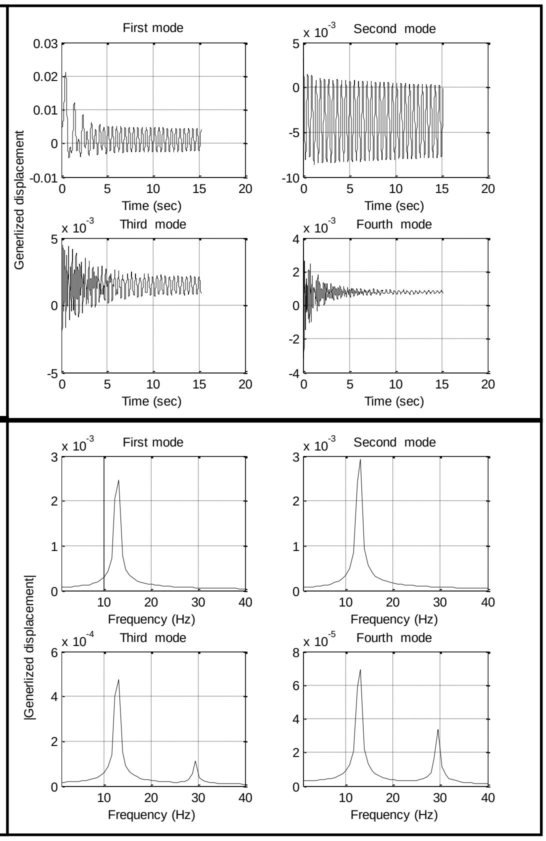

Figure (7) Responses and FFT of the generalized displacements for the rectangular wing at V=153 (m/sec).

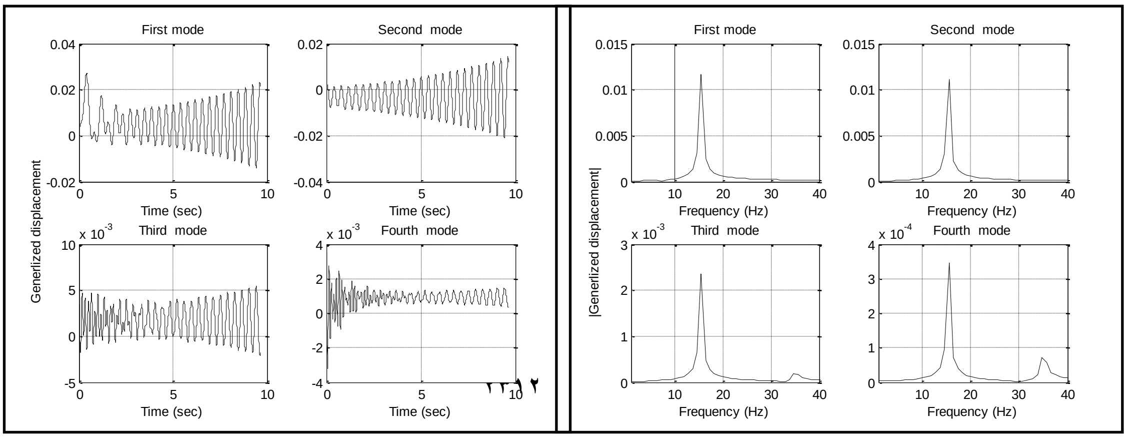

Figure (8) Responses and FFT of the generalized displacements for the rectangular wing at V=155 (m/sec) .

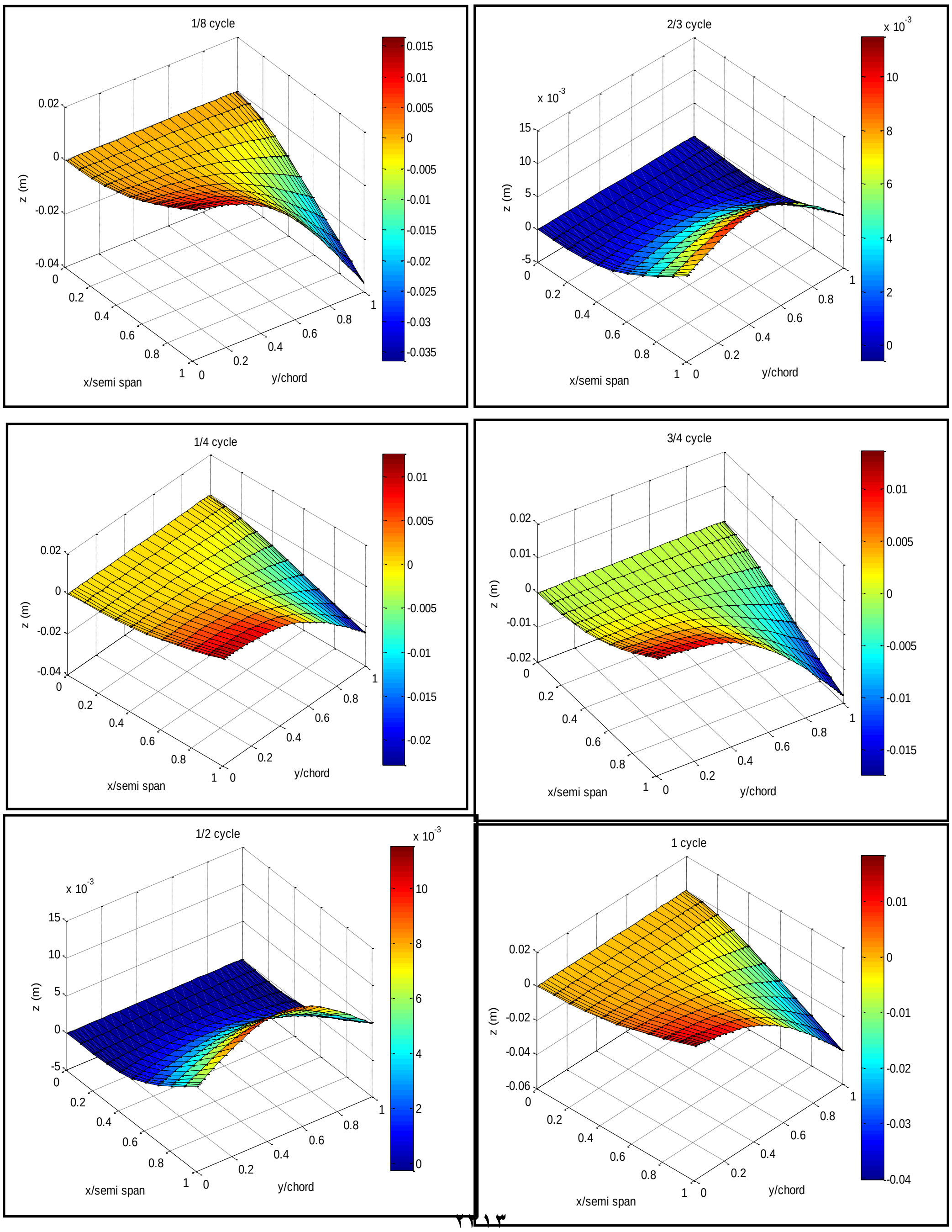

Figure (9) Aeroelastic responses of the wing during flutter condition at V=153(m/sec).

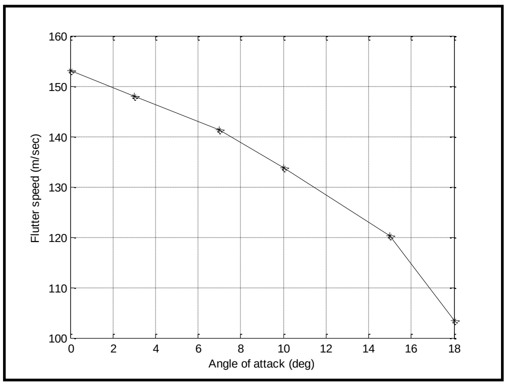

Figure (10) Flutter speed of the actual aircraft wing vs angle of attack.

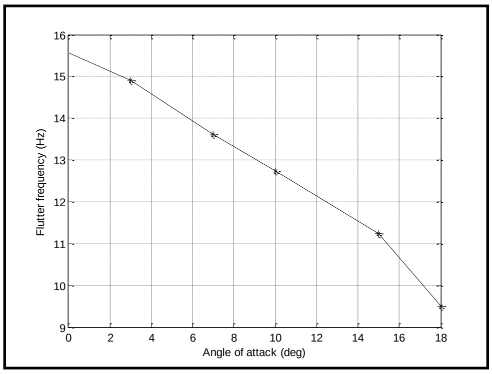

Figure (11) Flutter frequency of the actual aircraft wing vs angle of attack.

CONCLUSIONS

In the preset aeroelastic analysis, equivalent plate model based on Rayleigh-Ritz method were found to be fast and adequately accurate for the analysis of flutter condition for aircraft wing.

According to the obtained results, the following can be concluded:

1- the flutter speed and flutter frequency for aircraft wing may be estimated successfully from the behavior of the generalized displacements of an equivalent plate in time and frequency domains.

2-The first modes are enough to be taken in the analysis of FSI because the first modes are dominate in structure behavior.

3-The aerodynamic forces at speeds below the flutter speed may contribute damping the external vibration until at resonance cases.

4-The increasing in the angle of attack leads to reduce flutter speed and flutter frequency because of increasing in lift force. The aircraft may approach to flutter condition at high angles of attack even if the speed was less than the flutter speed at zero angle of attack. Therefore, the designer must be aware of this situation and try to make the maximum cursing speed below the onset of flutter within the aircraft flight envelope.

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