Optimization of Light Weight Aircraft Wing Structure

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Abstract

In this paper, the static analysis for finding the optimum design of wing structure has been performed. The two types of wings for unmanned airplanes were chosen; the first wing was the taper designed at (M=0.71) and the other wing was rectangular designed at (M=0.21).

The research covers both the aerodynamic and structural design. The aerodynamic study was achieved by using the vortex lattice method and the structural analysis was achieved by using the Ansys 5.4 package where isotropic and composite materials were used.

From results the mass saving of (32%) when composite material is used instead of isotropic material for the taper wing and (23%) for the rectangle wing. The optimum design for each wing was obtained according to the mass, stress and displacement.

الخلاصـــــــة

يتضمن البحث ايجاد التصميم الامثل لهيكل الجناح من خلال التحليل السكوني للهيكل. لانجاز العمل تم اختيار نوعين من الاجنحة لطائرات مسيرة حيث ان الجناح الاول هو من النوع المستدق وتم تصميمه عند سرعة (M = 0.71) والجناح الاخر مستطيل وتم تصميمه عند سرعة (M = 0.21).

شمل البحث الجانب الديناهوائي والتصميم الهيكلي، الدراسة الديناهوائية تم انجازها من خلال استخدام طريقة الدوامات الشبكية. تم تنفيذ التحليل الهيكلي باستخدام برنامج (Ansys 5.4) لانجاز توزيع الاجهاد والازاحة على الجناح باستخدام مواد مركبة واخرى متماثلة الخواص الميكانيكية في تصميم الجناح.

وجد من خلال النتائج ان النقصان الواضح في قيمة الكتلة عند استخدام المواد المركبة هو بمقدار (32%) للجناح المستدق في حالة استخدام مواد مركبة بدلا من مواد متماثلة الخواص في تصميم الغلاف واختزال بنسبة (23%) في حالة الجناح المستطيل. تم انجاز التصاميم المثلى للاجنحة طبقا الى الكتلة، الاجهاد، الازاحة.

1. Introduction

The optimum structural design of an A/C wing is an important factor in the performance of the airplanes i.e. obtaining a wing with a high stiffness/weight ratio and sustaining the unexpected loading such as gust and maneuvering situations. This is accomplished by studying the different design parameters required to specify the wing geometry. the idea of the structural optimization in the classical sense, has been considered to be the minimization of structural mass by varying member sizes or shell thickness of a model in which the geometry remains unchanged therefore many studies have been made during the last years to find the structural optimization.

In 1986 [1] Vinson presented closed-form analytical solution for the analysis and design of minimum weight , composite material hex-cell and square cell honey comb core sandwich and panels subjected to in-plane uniaxial compressive loads. These methods account for overstressing, overall buckling, core shear instability, face wrinkling, and mono cell buckling. The methods insure minimum weight, as well as provide methods to compare various material systems, compare honey comb sandwich construction with other panel architectures.

In 1986 [2] Iyengar and Joshi found the minimum weight design of a laminated fiber-rein forced composite plate subjected to in plane and transverse loading. Restrictions are imposed on the buckling load and transverse deflection. The fiber orientation and thickness of each ply are treated as design variable.

Optimization studies are carried out by using an unconstrained minimization technique some of the observations are that, with preassigned fiber orientation, the optimum weight design results in a unique thickness distribution of the plies, and that the stability constraint is active at low aspect ratios, while the deflection constraint is active at large aspect ratios.

In 1988 [3] Canfield, Grandhi and Venkayya determined what techniques are reliable and efficient for optimization of a complex design problem. The study examined the relative numerical performance of various optimization methods as candidates for a hybrid algorithm using optimality criteria and mathematical programming methods, several optimization programs were used to design truss-and wing type structures.

In 1995 [4] Rohl, Marris and Schrage presented a combined procedure for the aerodynamic and structural optimization of a high-speed civil transport wing. Primary goal of the procedure is the determination of the jig shape of the wing necessary so that it deforms into its optimum shape in cursive flight. The wing structure is sized subject to strength, buckling, and aero elastic constraints. Various analysis have been performed with different material configurations and structural concepts.

In 2000 [5] Cherkaev and Kueuk found the new technique for optimization problem of structures at various loading, they found the optimal pice-wise constant design of an elastic tube which is made out of a non-homogeneous anisotropic material.

In 2001 [6] Liu, used a two level optimization procedure for wing design subject to strength and buckling constraints. The design variables are the orientation of ply and the number of plies of each orientation. The genetic algorithms and response surface method were used to continue the optimization.

In 2001[7] Ubaid, investigated the static stress analysis of wings using the Ansys 5.4 package. He showed that the root region is the most critical stress zone where the maximum

Von-Mises stresses are found and the maximum stress concentration for rectangular wing occurs at mid-chord, and for the swept at third chord. He also showed that the stress component have greatest concentration on the Von-Mises.

In 2003 [8] Jabur used two types of wing section (thick and thin section) for unmanned A/C to find the distribution of the displacements and the stresses by using the Ansys 5.4 package.

The objective of this paper is to develop a more accurate model for such optimal design studies through design the structure of wing that combine the composite and isotropic materials and compare this wing with the same wing that design only from isotropic material in order to obtain high strength/weight ratio for finding optimum design.

2. Aerodynamic Part

The aircraft motion through the atmosphere produces forces called the aerodynamic forces mainly the lift and drug forces and the wing is the major source of this aerodynamic force. The wing is that surface of aircraft which supports the aircraft by means of the dynamic reaction on the air producing pressure distribution. This pressure distribution on the wing changes with different wing angles of attack and flight condition. In this work the aerodynamic load (Lift) was calculated by using vortex lattice-method.

2-1 Vortex Lattice Method

A solution for three-dimensional wings of any general form can be obtained by using a vortex lattice model. For incompressible, inviscid flow, the wing is modeled as asset of lifting panels. Each panel contains a single horse-shoe vortex. Abound vortex is located at the panel 1/4 chord position with two trailing vortex lines shed from each end [9], as shown in **Fig.(1)**.

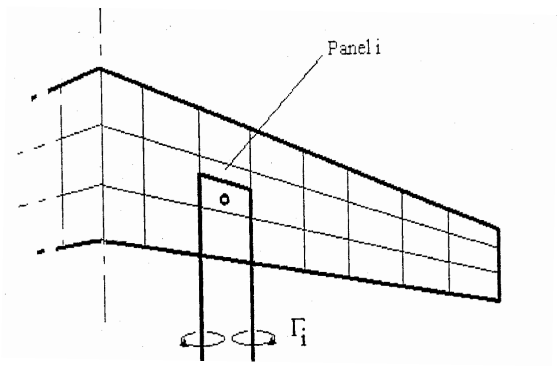


Figure (1) Vortex distribution on the panel

Both span-wise and chord-wise variation in lift can be modeled as a set of step changes from one panel to the next.

The required strength of the bound vortex on each panel will need to be calculated by applying a surface flow boundary condition. The equation used is the usual condition of zero flow normal to the surface. For each panel the condition is applied at the 3/4 chord position along the center line of the panel. The normal velocity is made up of a free stream component and an induced flow component, as shown in **Fig.(2)**.

$$\mathbf{V_n = 0 = \sin(\theta) + W_i} \quad \text{.......... (1)}$$

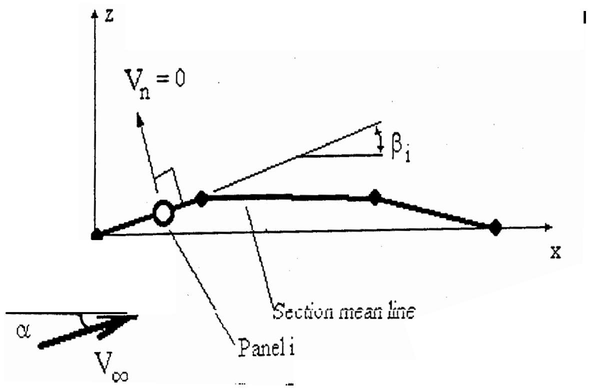


Figure (2) Flow on the wing section

The induced component is a function of strength of all vortex panels on the wing. Assuming small angles

$$\mathbf{\sin(\theta) = \sin(\alpha - \beta_i) = (\alpha - \beta_i) = \alpha - (dz/dx)_i} \quad \text{.......... (2)}$$

$$\mathbf{Wi = \sum_{j=1}^{n} A_{ij} \cdot \Gamma_j} \quad \text{.......... (3)}$$

From 1 and 3 we obtain:

$$\mathbf{\sum_{j=1}^{n} A_{ij} \cdot \Gamma_j = -V_\infty \sin(\theta)} \quad \text{.......... (4)}$$

where:

θ: is the flight path angle and

Aij: is the influence coefficient that will represent the induced flow on panel i due to the vortex on panel j.

A solution for the strength of the vortex lines on each panel is found by solving the matrix of equations the lift coefficient for the wing at a given angle of attack will be obtained by integrating the panel lift distribution. The lift on a particular panel can be found using the Kutta law.

$$\mathbf{Li = \rho V_\infty \Gamma_i \, 2\, k} \quad \text{lift of panel i} \quad \ldots\ldots (5)$$

$$\mathbf{L = \sum_{i=1}^{n} L_i} \quad \text{lift of wing} \quad \ldots\ldots (6)$$

The down wash velocity induced on a panel can be calculated once the strength of the wing loading is known. The variation between local flow angles for the panel and the free stream velocity can be found. A consequence of this down wash flow is that the direction of action of each panels lift vector is rotated relative to the free stream direction. The local lift vectors are rotated backward and hence give rise to a lift induced drag. By integrating the component of panel lift coefficient that acts parallel to the free stream across the span then the induced drag coefficient can be found.

$$\mathbf{D_i = \rho V_\infty \Gamma_i \, 2\, K \, \sin(\alpha i)} \quad \text{drag of panel i} \quad \ldots\ldots (7)$$

$$\mathbf{D = \sum_{i=1}^{n} Di} \quad \text{induced drag of wing} \quad \ldots\ldots (8)$$

The induced flow angle (alpha i) represents the amount of rotation of the lift vector backward and must be calculated from the velocities induced on the bound vortex of the panel by other panels and the free stream.

Pitching moment about the wing root loading edge can be calculated by summing the panel lift multiplied by a moment arm which extends in the x-direction from the loading edge of the wing to the centre of the bound vortex for the panel.

2-2 Vortex Lattice Method Program

The computer program allows defining wing plan-forms and geometries. The program assumes a linear variation of section properties between wing root and tip and that the loading will be symmetric about the wing root. The aerofoil section data for the wing root and tip is input from a file containing section coordinate data points. Files for all the standard 4 and 5 digit NACA sections can be created using the program. Using prandtl factor for taking the effect of compressibility for taper wing [10].

$$\mathbf{CL_{COM} = \frac{CL_{INCOM}}{\sqrt{1-M^2}}} \quad \ldots\ldots (9)$$

The aerodynamic results by this method are used as input load data in the structural analysis, as shown in **Fig.(3)**.

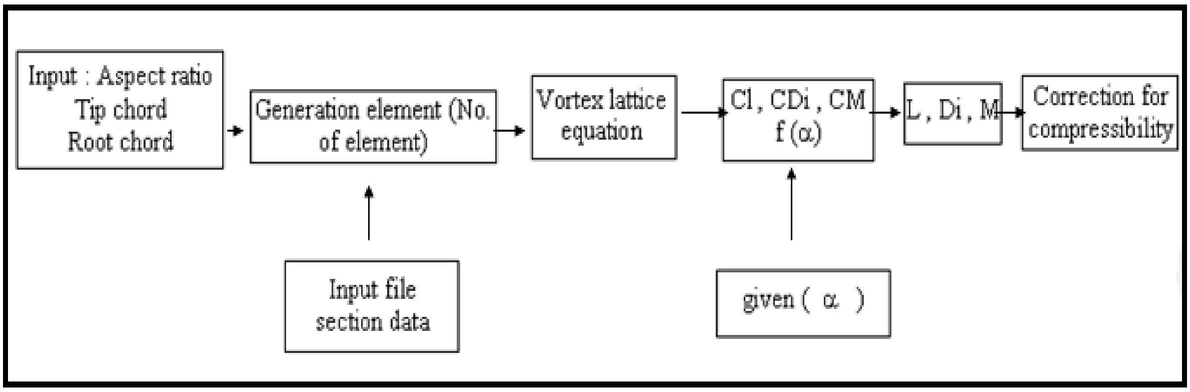


Figure (3) Flowchart for the aerodynamic determination

3. Structural static analysis

Structural analysis is probably the most common application of the finite element method. The static analysis is used to determine displacements, stresses, etc. under static loading conditions. In this work the structural static analysis was achieved by using the Ansys 5.4 package in order to obtain stress and displacement distribution in the wings by using isotropic and composite material which are used commonly these days especially for unmanned A/C. The Ansys program has many finite element analysis capabilities, ranging from a simple, linear, static analysis to a complex, non linear, transient dynamic analysis. A typical Ansys analysis has three distinct steps:

- Build the model.
- Apply loads and obtain the solution.
- Review the results.

3-1 Element Types

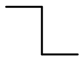
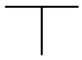
Having built the model geometry (wing), the structural wing components can be represented by using the suitable element. The Ansys element library consists of more than 100 different element formulations or types, as shown in **Table (1)**.

Table (1) Structural components

Material	Element	Representation	D.O.F	No. of node
Isotropic	Shell 63	Skin, web spar	6	4
	Beam 3D	Flange, stringers, ribs	6	2
Composite	Shell 91	Skin	6	8

The stiffeners cross section properties shown below in **Table (2)**.

Table (2) Stiffeners properties

Beam section type	Representation	Cross sectional (mm^2)
	Stringers	44
	Flange	44
	Ribs	44

3-2 Materials

Historically, aluminum materials have been the primary material for aircraft and space craft construction. Today, structural weight and stiffness requirements have exceeded the capability of conventional aluminum, and high-performance payloads have demanded extreme thermo-elastic stability in the aircraft design environment. During the past decades, advanced composite materials have been increasingly accepted for aircraft and aerospace structural materials by numerous developments and flight applications.

Composite materials are those containing more than one bonded material, each with different structural properties.

For the sake of simplicity two types of materials have been used for the optimization procedure one is isotropic material that wing configuration is completely made up of Aluminum or titanium (7075-T6, adv. Aluminum, Ti6A 14V). The material data are shown in **Table (3)**.

Table (3) Material data [4,11]

Material	7075-T6	Adv. aluminum	Ti6A14V
E (N/m^2)	71E+9	82.7E+9	110.32E+9
ν	0.33	0.318	0.29
Yield stress (N/m^2)	470E+6	424E+6	598E+6
ρ (Kg/m2)	2800	2906.4	4429

Other one has composite material where substructure (spars, flanges, Ribs, and stringers) is made up of isotropic material whereas the skin panels of wing are made up of composite material such as graphite/epoxy. The composite material data are shown in **Table (4)**.

Table (4) Composite material data [12,13]

Composite Material	H.M Gr/Ep	Gr/Ep	Sg/Ep
E_1(Gpa)	137.9	145	55
E_2=E_3 Gpa	14.48	10	16
$\nu_{12}=\nu_{23}=\nu_{13}$	0.21	0.25	0.28
G_{12}=G_{23}= G_{13}(Gpa)	5.86	4.8	7.6
ρ (Kg/m^3)	1743.84	1580	1593.42

4. Optimization Procedure

The wing structural design problem is composed into two levels in a hierarchical structure at the first level, the wing configuration is completely made up of isotropic material and the following design parameters were investigated such as number of stiffeners, skin thickness, cross sectional area of stiffener and the type of material. The second level wing substructure (spars, stiffener) is made of isotropic material and the skin panel is made of composite material. For composite material, the work covers the investigation of the effect of changing fiber angle, number of layers and the types of composite material. Then from the results we take the optimum design

5. Results

Results of the theoretical aerodynamic analysis and structural analysis by Ansys package are presented in this paper. **Table (5)** shows the main characteristics of the wings. The aerodynamic results (lift) are used to simulate the wing loading on the wings during the static stress analysis.

Table (5) Main characteristics of the wings [14,15]

Characteristic	Taper wing	Rectangle wing
Wing span (m)	3.05	3.35
Gross area (m^2)	1.4	1.53
Aspect ratio	6.64	7.33
Taper ratio	0.53	1
Tip chord (m)	0.32	0.45
Root chord (m)	0.6	0.45
M. A. C. (m)	0.46	0.45
Sweep angle of leading edge (deg.)	5.24	0
Section profile	NACA 0012	NACA 2412

The aerodynamic results are taken for a range of wing incidences **Figs.(4)** and **(5)** and only the maximum wing loadings are applied to the wings for serve wing stress conditions.

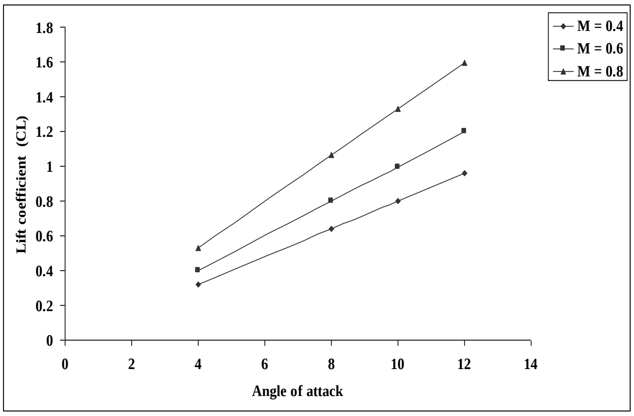


Figure (4) Distribution of the lift coefficient with α and M for taper wing

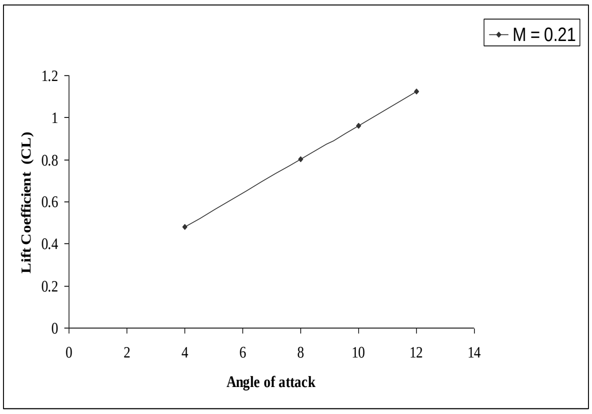


Figure (5) Distribution of the lift coefficient with α for rectangle wing

Figures (6) and (7) show the stability of results at (15744) D.O.F for taper wing and (13440) D.O.F for rectangle wing.

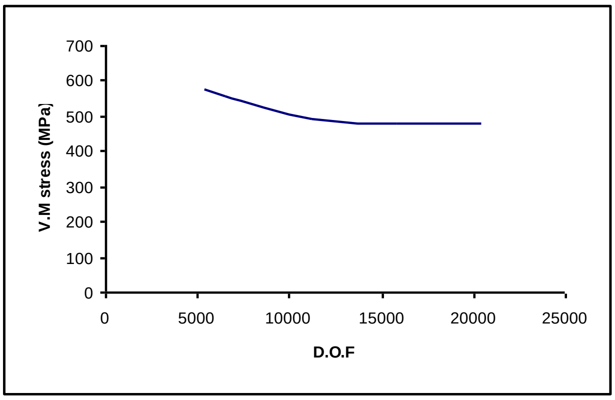


Figure (6) Stability of results for taper wing

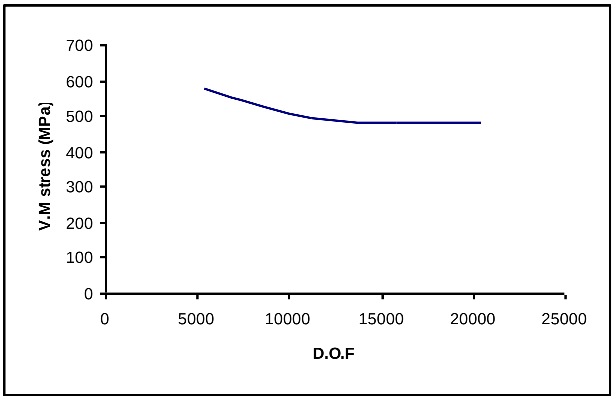


Figure (7) Stability of results for rectangle wing

The effect of compressibility on the stress and displacement are shown in **Figs.(8)** and **(9)** for taper.

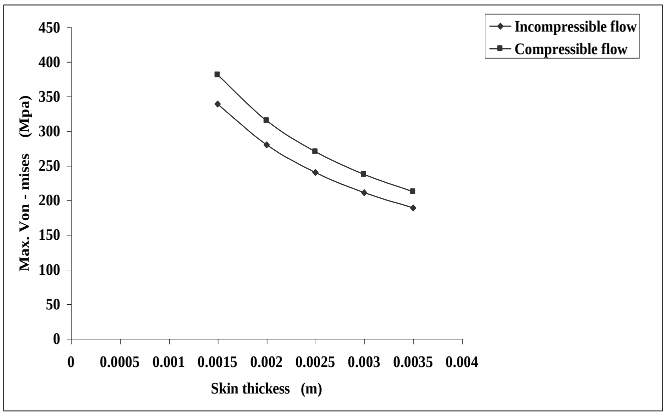


Figure (8) V.M. stress distribution due to compressibility for taper wing

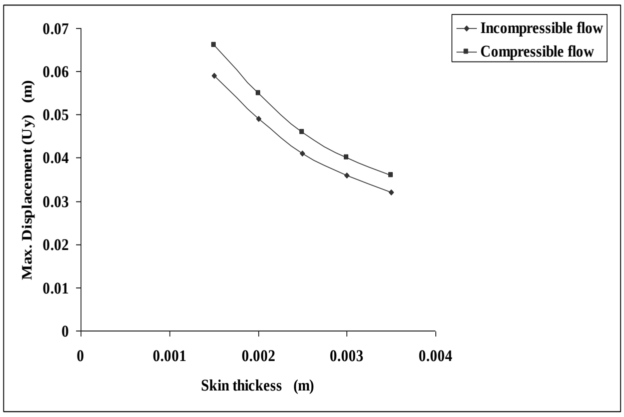


Figure (9) Vertical displacement distribution due to compressibility for taper wing

The structural analysis was achieved by using the Ansys 5.4 package in order to obtain stress and displacement distributions in the wings by using isotropic and composite material for find the optimum design. The work involves the investigation effects of changing the skin shell thickness (0.001-0.003m), the stiffener beam area ($44mm^2$-$81mm^2$), adding ribs span wisely (3-5) adding stringers chord wisely (0-5), fiber angle (0, 90, ±45), type of isotropic material (**Table (3)**), type of composite material (**Table (4)**), number of layers (4-6 layers) and changing ply sequence. The wings are modeled as the skin and spar web are meshed with shell rectangle elements, the spar flange and stiffener are modeled by beam element (**Table (1)**). Next to this, the optimum design parameter was achieved, where the skin thickness (0.003m) for taper wing and (0.001m) for rectangle wing and the stiffener (3×5) for both taper and rectangle wing. The optimum isotropic material is (7075-T6) for taper and rectangle wing and the optimum fiber orientation is [45/90/90/45] for taper wing and [45/90/90/0] for rectangle wing. Also the optimum cross section area of stiffener equal to (44 mm^2) for both wings (**Figs.(10)**, **(11)**).

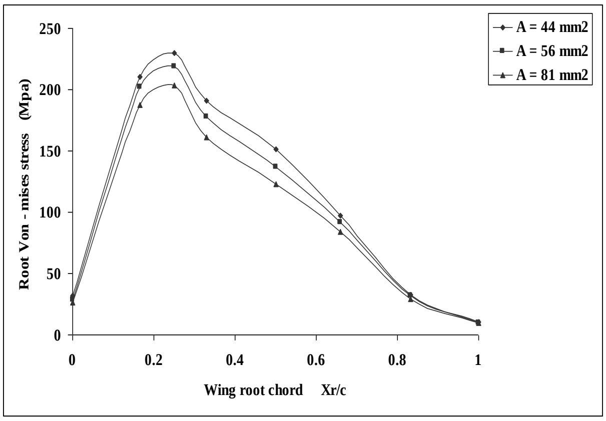


Figure (10) Effect of changing cross section area of beam on root V.M. stress distribution for taper wing

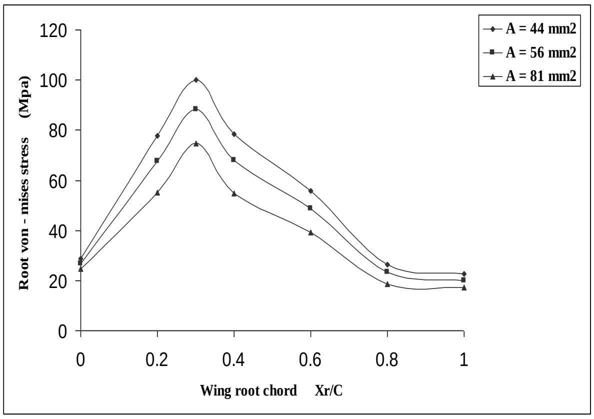


Figure (11) Effect of changing cross section area of beam on root V.M. stress distribution for rectangle wing

The results of this analysis for rectangle wing that have NACA (0012), aspect ratio (5) and wing span (7.5 m) at (M = 0.4) and using isotropic materials were compared with the results of the reference [7], this comparison was presented in **Table (6)**. We show good agreement of results between this analysis and reference [7].

Table (6) Comparison of results

Results of Present analysis for Max. Von-Mises stress (Mpa)	Results of Ref. [7] for Max. Von-Mises stress (Mpa)
239	220

Table (7) shows the Von-Mises stress and (Uy) results for taper and rectangle wing when adding the ribs span wisely, increasing the number of ribs span wisely gives small reduction effect on the Von-Mises stress and displacement distribution.

Table (7) Effect of changing No. of Ribs

	Taper wing			Rectangle wing		
No. of ribs / properties	3	4	5	3	4	5
V.M. stress (M Pa)	261	260	255	139	139	141
UY(m)	0.047	0.047	0.045	0.038	0.038	0.038
Stress ratio %	55.53	55.31	54.25	29.57	29.57	30.00
Mass (Kg)	14.17	14.28	14.40	5.97	6.08	6.20
Yield/weight $*10^6$ ($1/m^2$)	3.38	3.35	3.32	8.02	7.86	7.72
Weight/Area (N/m^2)	99.29	100.06	100.90	38.27	38.98	39.75

By using the stiffener (3x5) and changing the skin shell thickness, will affect the displacement and stress distributions for both wings as shown in **Figs.(12)**, **(13)**, **(14)**, and **(15)**.

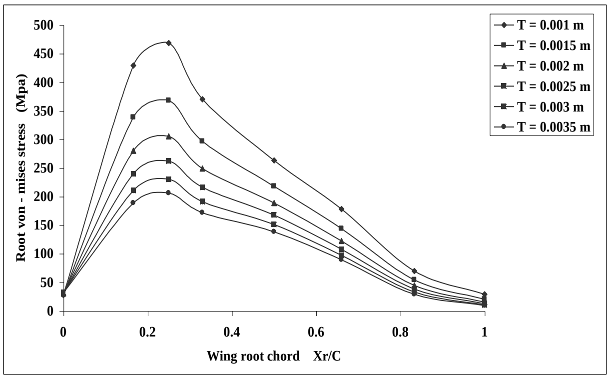


Figure (12) Effect of changing skin thickness on root V.M. stress distribution for taper wing

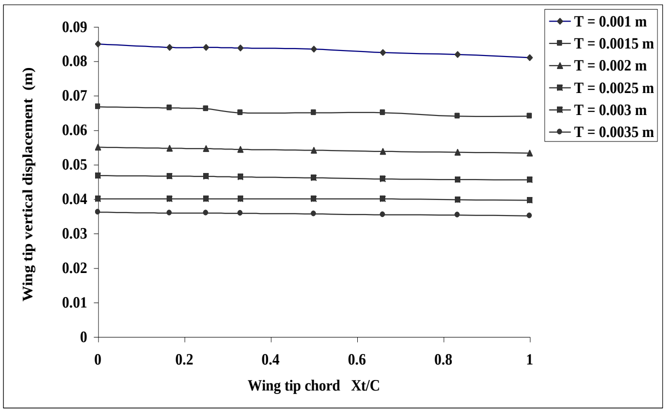


Figure (13) Effect of changing skin thickness on tip Vertical displacement distribution for taper wing

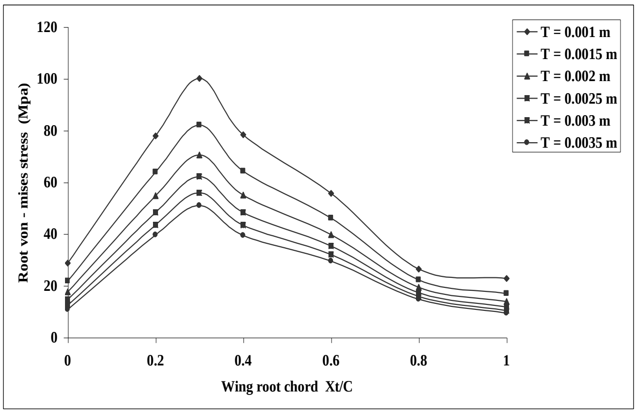


Figure (14) Effect of changing skin thickness on root V.M. stress distribution for rectangle wing

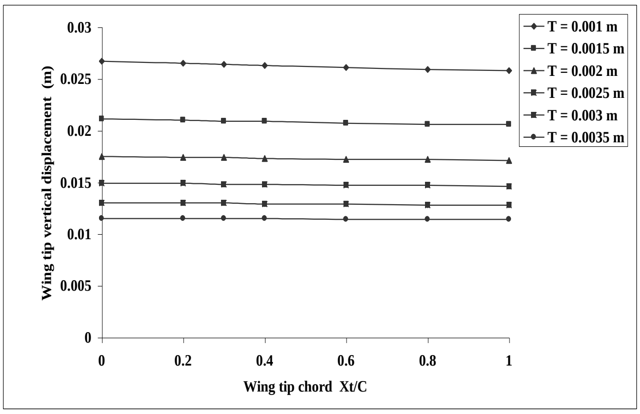


Figure (15) Effect of changing skin thickness on tip Vertical displacement distribution for rectangle wing

To find the effect of design the same laminate with different composite materials, **Figs.(16)**, and **(17)** show the value of the root Von-Mises stress distribution for both wings.

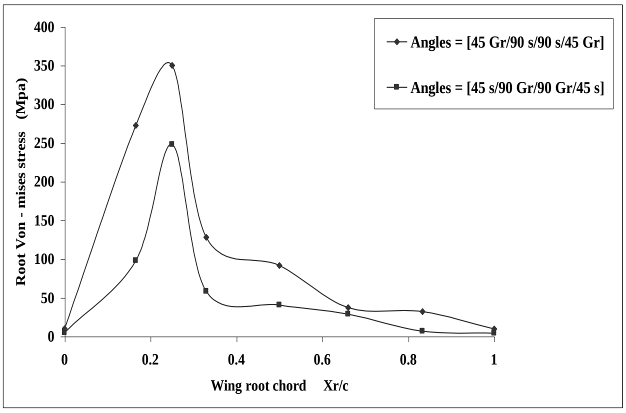


Figure (16) Effect of using different composite materials on root V.M. stress distribution for taper wing

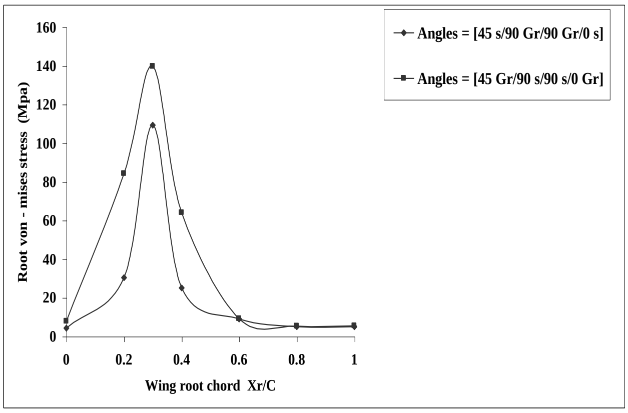


Figure (17) Effect of using different composite materials on root V.M. stress distribution for rectangle wing

Table (8) shows comparison of the results between the isotropic and composite material. The goal comparison is to emphasize that the wing total mass is reduced and Von-Mises stress increases when composite material is used instead of isotropic material for taper and rectangle wings.

Figures (18), and **(19)** show the optimum composite material and ply angles for wings.

Table (8) Result of isotropic and composite material

Material / properties	Taper wing		Rectangle wing	
	Aluminum allay 7075-T6	Graphite/Epoxy 45/90/90/45	Aluminum allay 7075 -T6	Graphite/Epoxy 45/90/90/0
V.M. stress (MPa)	237	249	103	109
UY(m)	0.04	0.038	0.026	0.027
Stress ratio %	50.4	52.97	21.91	23.19
Mass (Kg)	16.05	10.92	8.03	6.17
Yield/weight*10^6 (1/m^2)	2.98	4.38	5.96	7.76
Weight/Area (N/m^2)	112.46	76.51	51.48	39.56

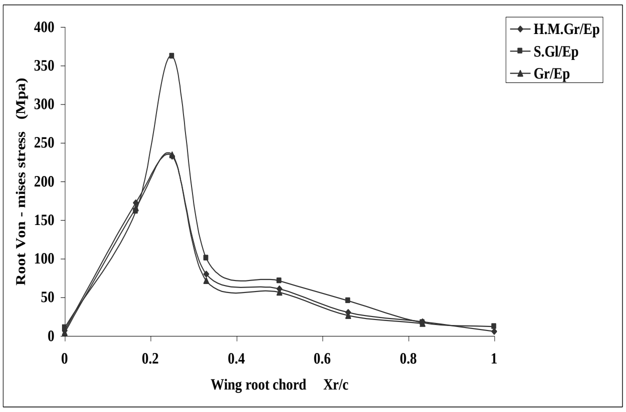


Figure (18) Effect of changing composite materials at ply [45/90/90/45] on root V.M. stress distribution for taper wing

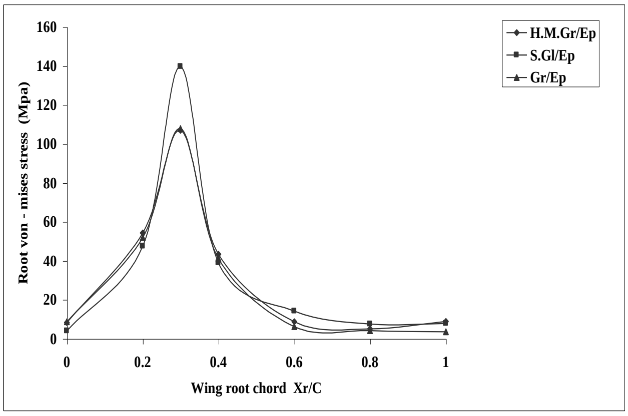


Figure (19) Effect of changing composite materials at ply [45/90/90/0] on root V.M. stress distribution for Rect. wing

Also **Table (9)** shows the optimum designs obtain from this analysis.

Table (9) Optimum design parameters

case	Skin material	Skin thickness (m)	Beam section area (mm^2)
1	Gr/Ep [45/90/90/ 45]	0.003	44
2	$[45_S/90_{Gr}/90_{Gr}/45_S]$	0.003	44
3	7075 – T6	0.003	44
4	$[45\ _S/90_{Gr}/90_{Gr}/0\ _S]$	0.001	44
5	Gr/Ep [45/90/90/ 0]	0.001	44
6	Sg/Ep [45/90/90/45]	0.001	44
7	7075 – T6	0.001	44

	Taper wing			Rectangle wing			
Case / properties	1	2	3	4	5	6	7
V.M. stress (M pa)	249	254	237	109	110	135	103
UY(m)	0.038	0.037	0.040	0.027	0.029	0.037	0.026
Stress ratio%	52.97	54.04	50.42	23.19	23.40	28.72	21.91
Mass (Kg)	10.92	10.95	16.05	6.17	6.18	6.19	8.03
Yield/weight*10^6 ($1/m^2$)	4.38	4.37	2.98	7.76	7.75	7.73	5.96
Weight/Area (N/m^2)	76.51	76.72	112.46	39.56	39.62	39.68	51.48

Figures(20), **(21)** show the Von-Mises stress distribution in case of using isotropic and composite materials for taper wing, in both cases the root Von-Mises variation increases through out the (0.2-0.3) chord zone and is less toward the wing leading and trailing edges span-wisely also the Von-Mises stress decrease toward the wing tip.

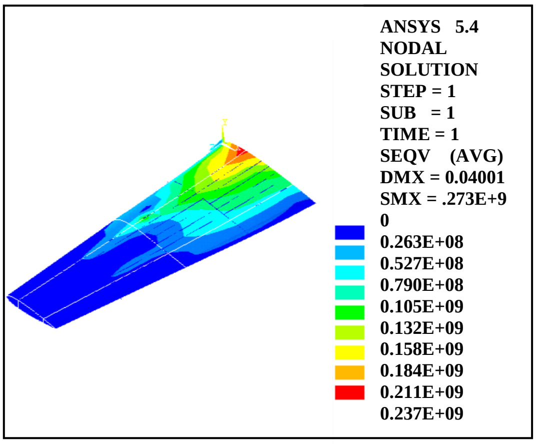


Figure (20) Contours of V.M. stress (Pa) of isotropic material for taper wing

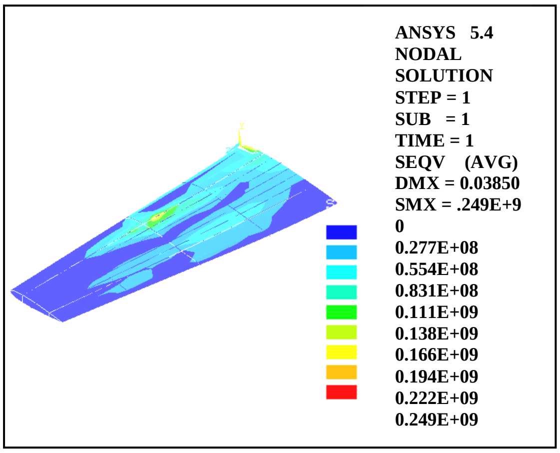


Figure (21) Contours of V.M. stress (pa) of composite material for taper wing

The effects of using isotropic and composite materials on the vertical displacement of the wing are shown in **Figs.(22)**, **(23)** it is notice that the vertical tip displacement (UY) increases towards the tip wing where the displacement at the tip is maximum.

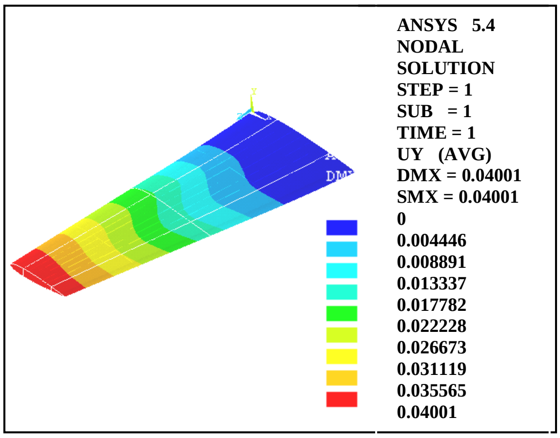


Figure (22) Contours of Vertical displacement (m) of isotropic material for taper wing

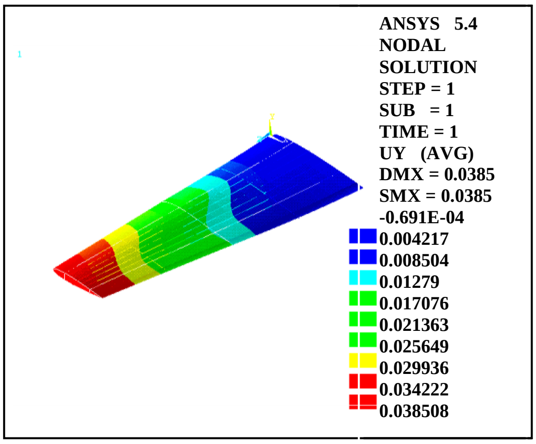


Figure (23) Contours of Vertical displacement (m) of composite material

for taper wing

Also for rectangle wing **Figs.(24)**, **(25)** show the Von-Misses distribution in case of using isotropic and composite materials where at the (0.2-0.35) chord zone the Von-Mises stress is a maximum.

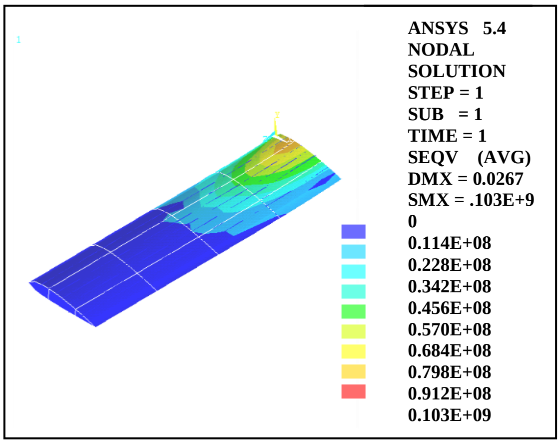


Figure (24) Contours of V.M. stress (Pa) of isotropic material for rectangle wing

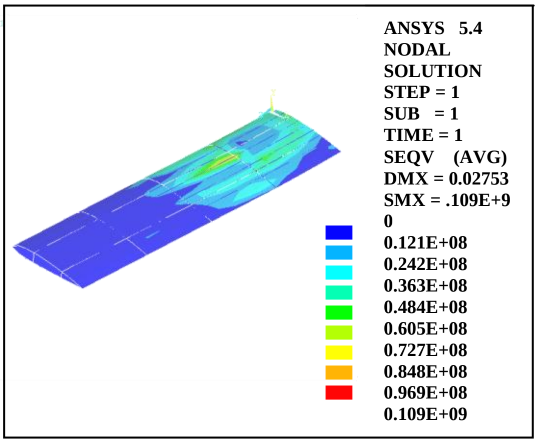


Figure (25) Contours of V.M. stress (Pa) of composite material for rectangle wing

Figures (26), **(27)** show the displacement distribution of the wing in case of using isotropic and composite materials.

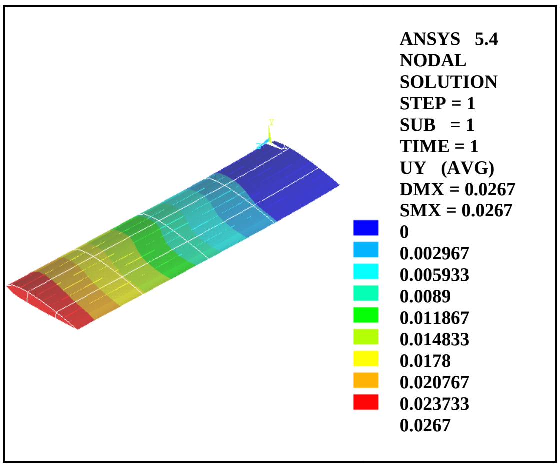


Figure (26) Contours of Vertical displacement (m) of isotropic material for rectangle wing

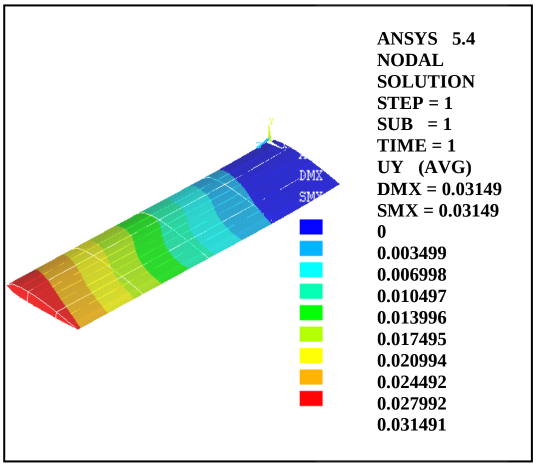


Figure (27) Contours of Vertical displacement (m) of composite material for rectangle wing

6. Conclusions

The static analysis of the wing by Ansys package has been presented and used to determine the optimum design for the wing. This operation involve using the vortex lattice method to obtain the aerodynamic result at (M=0.7) for taper wing and (M=0.2) for rectangle wing. This aerodynamic result (lift) is used to simulate the wing loading on the wings during the static analysis also using isotropic and composite materials in design the parts of wing, therefore the major and general observations and conclusions for this work can be listed below:

1. From the aerodynamic results it can be noticed that the lift coefficient increase with the increase of Mach number from (M=0.4) at the same angle of attacks due to the compressibility.

2. From the structural results it is noticed that increasing the skin thickness is considered as an important factor in reducing the stress and displacement levels compared to the increase of, cross section area of beam and number of layers.
3. Increasing the ribs span wisely does not change noticeable the Von-Mises stress distribution for all wings.
4. In case of using composite material in design the skin of wing it can be noticed that the changing in fiber orientation and the number of layers at the same skin thickness will produce the variation in the Von-Mises stress (increase or decrease).
5. Also we are found that the mass saves (32%) when composite material is used instead of isotropic material for the taper wing and (23%) for the rectangle wing.
6. We are found that it is desirable to use composite and isotropic materials in design of the skin of taper wing but undesirable to use isotropic materials in design the skin of a rectangle wing.

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Nomenclature

V_n:	Normal velocity, (m/s)
V_∞:	Free stream velocity, (m/s)
θ:	Flight path angle, (deg.)
W_i:	Induced flow component, (m/s)
α:	Angle of attack, (deg.)
A_{ij}:	Influence coefficient
L_i:	Lift of panel i, (N)
Γ:	Strength of vortex (m^2/s)
L:	Lift of wing, (N)
D_i:	Drag of panel i, (N)
D:	Drag of wing, (N)
CL_{com}:	Lift coefficient in compressible flow
CL_{incom}:	Lift coefficient in incompressible flow
M:	Mach number
E:	Young s modulus, (pa)
G:	Shear modulus, (pa)
ν:	Poisson s ratio
ρ:	Density, (kg/m3)
D.O.F.:	Degree of freedom
M.A.C.:	Mean aerodynamic chord (m)
A/C:	Aircraft
V.M.:	Von-Mises (Mpa)
Gr/Ep:	Graphite/epoxy
H.M Gr/Ep:	High modulus graphite/epoxy
Sg/Ep:	Glass/epoxy

Subscripts

1, 2, 3:	Longitudinal, transverse, and thickness directions of a unidirectional ply