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GJRE-A Classification :

A STUDY OF THE EFFECT OF SURFACE DEFECT CAUSED BY IMPACT LOAD FROM SHOCK WAVE ON WING DESIGN IN TRANSONIC SPEEDS USING CLASSICAL PLATE THEORY

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I. Introduction

In many modern design of aircraft it was needed for operational efficiency to develop a complex process of determining the local mach number along the wing taking into consideration the unsteady aerodynamic phenomena in the context of nonlinear flow behavior for transonic flow which is terminated by a shock wave allowing the flow to slow down to subsonic speeds .This causes the shock wave to be propagated from leading edge to the trailing edge so a complicated in the analytical analysis specially with free stream mach numbers from M = 0.6 to M = 1.4 which must airplane kept flying fast enough to encounter transonic flow. This required that the viscous effect and incompressibility must be taken into consideration .

The loads are caused by aerodynamic forces and interacted with elastic forces which have been taken as a static or steady state systems knowing that aero elasticity, R.Duvigneau [1] have been solved this problem taking into consideration the aerodynamic shape optimization with uncertain operating conditions is addressed in this paper, a two-level modeling strategy is proposed, that relies on the use of CFD simulations in conjunction with metamodels. Also Andrey M.Shevchenko (etal) [2] compute the transonic flows over a complex 3D aircraft configuration, a viscous/in viscid interaction method is developed by coupling an integral boundary-layer solver with an Euler solver, showing results in good agreement with the wind tunnel data The object of this contribution is to introduce an efficient interactive boundary-layer method. Another object of this paper is to present a far-field drag prediction technique.

While for dynamic aero elasticity Gareth A.Vio (etal) [3] taken the main objective of the work is the characterization of the dynamic response of aeroelastic models resulting from coupled Computational Fluid Dynamic and Finite Element .At higher free stream Mach numbers the shock wave has moved all the way to the trailing edge at steady-state conditions and zero angle of attack and cannot move any further, despite the bending and twisting motion. Consequently, the aero elastic instability disappears and the system becomes stable again.

Atef. Alshabu etal [4] taken into consideration the preliminary experimental and numerical investigation of wave processes taking place in the flow field on a supercritical airfoil in a defined Mach and Reynolds number range. Time-resolved pressure measurements performed, reveal the unsteady behavior of these waves, the experimental and numerical results are in good agreement. some researchers using a numerical study for shock wave oscillations Akira oyama (etal) [5] have been obtained a method to understand Mach number effect on flow field over a delta wing with blunt leading edge in supersonic and high angle of attack region .The present results indicate that a delta wing with blunt leading edge can be mixed flow of two different types of flow structure in supersonic and high angle of attack flow region and the location of the boundary of the two types of flow moves toward the apex of the wing as the free-stream Mach number increases.

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Mylene thiery and Eric coustols[6] have been deals with recent numerical results from ongoing research conducted at ONERA/DMAE regarding the prediction of transonic flows ,simulation was performed to demonstrated the real impact of all lateral wind – tunnel walls on such a flow . Also R.Bur etal [7] investigate and analyses the response of a transonic channel flow when the shock wave is subjected to a periodic motion at a well defined and controlled frequency in the shock oscillation region. The good agreement between the evolutions of the measured and computed fluctuating quantities which is part of a study investigating the prediction of the aero elastic behavior of aircraft subjected to non-linear aerodynamic forces . Antony Jamson [8] examines the use of computational fluid dynamics as a tool for aircraft design . the paper discusses the use of techniques drawn from control theory to determine optimal aerodynamic shapes . Essential elements of algorithm design are discussed in detail with unified approach to the design of shock capturing schemes .

The geometry and the swept angle affect the design of the wing as in P.C Steimle etal [9] Wind tunnel tests with force and pressure measurements have been performed using an oscillating rigid swept wing to investigate unsteady aerodynamic phenomena in the context of non-linear flow behavior and the development of aero elastic instabilities in the transonic flight regime. V.N.Zudov and E.A.Pimonov [10] investigated an external flow around supersonic airplanes with a large swept wing and low aspect ratio . experimental investigation indicate that at some interaction regimes of the stream wise vortex with shock wave , a vortex explosion arises to model with different types of the stream wise vortex with the inclined shock wave using three dimensional non stationary Eulers equations .

In this research two things have been taken to be investigated, the first one is to modify the Mach number which consist the effect of shock wave position on the wing surface for mediums that comes in transonic flow on the stress distribution along the shell of the wing as an low impact velocity load in the mach number and then substituted in classical plate theory for finding the deflection and the pressure coefficients distribution. The second one is using ANSYS 10 as a finite element method for determining the position of the shock wave pressure value which depends on the Mach number for transonic flow.

II. Analytical Solution

In this section the analysis of the wing as a thin elastic , isotropic , homogeneous , clamped rectangular plate of uniform thickness h with sides a and b based on the Levey hypothesis .

The governing equation of the subject plate is derived by utilizing the Hamiltons principle :

$$\delta \int_{t_1}^{t_2} (T - U - V)dt = 0 \cdots\cdots (1)$$

Where T=the kinetic energy , U= the strain energy , V= the potential energy produced by external loads and δ=the first variation operator .

The strain energy U is given by :

$$U = W \cdot \tfrac{1}{2}\int_0^a\int_0^b \left(M_x k_x + M_y k_y + 2M_{xy}k_{xy}\right) d_x d_y \cdots\cdots (2)$$

Where k_x , k_y and k_{xy} = the curvatures and twist of the middle surface respectively . k_x = -w,xx , $k_y =$ -w,yy , k_{xy} =-w,xy and w(x,y,t)= the deflection

M_x = the bending moment per unit width =

$$-Dd(x,y)\left(w_{,xx} + \mu w_{,yy}\right) \cdots\cdots (3)$$

M_y = the bending moment per unit width =

$$-Dd(x,y)\left(w_{,yy} + \mu w_{,xx}\right) \cdots\cdots (4)$$

M_{xy} = the twisting moment =

$$-(1-\mu)Dd(x,y) - w_{xy} \cdots\cdots (5)$$

Where D = the flexural rigidity of the plate = $Eh^3/12(1-\mu^2)$

E= young modulus and μ = poisons ratio

The strain energy becomes :

$$U=\frac{1}{2}\int_0^a\int_0^b D\cdot d(x,y)\cdot\left[(k_x)^2+(k_y)^2+2\mu k_x k_y+2(1-\mu)(k_{xy})^2\right]dxdy\ldots\ldots(6)$$

Now the variation of potential energy can be expressed as :

$$V=\iint_{00}^{ab}(p.w-\rho.g.h.Y)dxdy+\int_{-0}^{b}(M_o)_x.w_{,x}dy+\int_{-0}^{a}(M_o)_y.w_{,y}dx \quad\ldots\ldots\ldots\ldots\ldots(7)$$

Where Y= the variation in the altitude of the airplane
P = external forces prescribed from boundary edges

$(M_o)_{x,y}$ = the aerodynamic pitching moment of the aircraft in the x , y direction respectively

The kinetic energy can be written as :

$$T=\frac{1}{2}\rho\cdot h\int_0^a\int_0^b(v^2+(w^\bullet)^2dxdy\ldots\ldots(8)$$

Where v= the free stream speed of the airplane

From compressibility effect in aerodynamic flow causes change in the density and this will lead to pressure change especially when the Mach number is greater than 1 , various types of waves can form in the flow

So that in equation (8) the value of v can be substituted as :

$$v^2=\frac{p-p_\infty}{\frac{1}{2}\rho c_p}\ldots\ldots\ldots(9)$$

Where $p-p_\infty$ is the gauge pressure which is various along the airfoil , c_p is the pressure coefficient along the airfoil and its value depend on the magnitude of the mach number . the transonic problem is difficult because it is inherently nonlinear , and the steady solution changes math type , being elliptic in the subsonic portion of the flow and hyperbolic in the supersonic part of the flow .

So the value of c_p can found by substitute eq (6) ~ eq(9) into eq (1) and the differential equation of motion can be obtained associated with the following boundary conditions :-

$$(w)_{x=0}=0\ ,\ (\frac{\partial w}{\partial x})_{x=0}=0\ldots\ldots\ldots(10)$$

$$(M_X)_{X=0}=(\frac{\partial^2 w}{\partial x^2}+\mu\frac{\partial^2 w}{\partial y^2})_{x=0}=0\ldots\ldots(11)$$

$$Dd\left(\frac{\partial^2 w}{\partial x^2}+v\frac{\partial^2 w}{\partial y^2}\right)=m_x$$ along the edge x=a ……….. (12)

The similar boundary conditions can be written along the edges y=0 and y=b .

III. The Shock Wave Effect on The Boundary Condition in Transonic Flow

The load applied on the wing consider as a subsonic condition depending on the value of the pressure coefficient c_p that obtained from the solution of the above equation of motion .

Adding to the boundary conditions that the value of c_p in transonic flow have a value at a point on the airfoil a critical value indicate that the region where the flow is locally supersonic or isentropic flow and the good basic formula for compressible flow is :-

$$c_{p\,crit}=-\frac{2}{\gamma M_\infty{}^2\left[1-\left\{\frac{2}{\gamma+1}+\frac{\gamma-1}{\gamma+1}M_\infty{}^2\right\}^{\frac{\gamma}{\gamma-1}}\right]}\ldots\ldots\ldots(13)$$

So that the pressure coefficient obtained from plate theorem compared with the $c_{p\,crit}$ if it reach this value then the load at this region have been taken as shock wave load and the behavior of the load with time as an low velocity impact with the following assumption [11] :-

$$P=P_o(1-\frac{\tau}{t})\exp(-\alpha\frac{\tau}{t})\ldots\ldots\ldots(14)$$

Where p_o = the max load per unit width .

t = the time duration of the shock wave .

τ = the period of time duration of the shock wave .

α= critical damping coefficient ranges between 2 to 20 %

while below the $c_{p_{crit}}$ value the load assumed to be represent as p(x ,y) by a Fourier series as :-

$$p(x,y) = \sum_{m=1}^{\infty}\sum_{n=1}^{\infty} a_{mn} \sin\frac{m\pi x}{a} \sin\frac{n\pi y}{b} \dots\dots\dots (15)$$

And this will effect on the pressure distribution and the value of the locally much number along the wing cord .

IV. The Effect of The Defects on The Pressure Coefficient

In the surface of the wing there will be some defects which come from the metal working of the shell that covers the reinforced beams of the wing or from the cycling aerodynamic pressure under shock waves that causes a surface crack initiation in the region of high local mach number value along the wing . this defects can be detected in this analysis in the region of high shock waves by taking the variation of the thickness of the plate and the depth of the surface crack as a percentage of the plate thickness along the defect. This will cause decrease in the value of the flexural rigidity of the plate D and the pressure coefficient c_p leading to increase the value of the stresses around the crack causes the stress intensity factor increased and the defect will be start to propagate .

It must be noticed that the stress intensity factor have been determined from the max principal stress which cab calculated by the equation :-

$$\sigma_{1,2} = \frac{\sigma_x + \sigma_y}{2} \pm \sqrt{\left(\frac{\sigma_x - \sigma_y}{2}\right)^2 + \tau_{xy}^2} \cdots\cdots (16)$$

And the angle inclined of the crack with the principle axis can be calculated by the equation :

$$\alpha = \frac{1}{2}\tan^{-1}\left(\frac{2\tau_{xy}}{\sigma_x - \sigma_y}\right) \cdots\cdots\cdots (17)$$

So the elastic stress for a point located a distance r and θ from the crack tip is given by :-

$$\sigma_x = \frac{KI}{\sqrt{2\pi r}}\cos\frac{\theta}{2}\left[1 - \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right] - \frac{KII}{\sqrt{2\pi r}}\sin\frac{\theta}{2}\left[2 + \cos\frac{\theta}{2}\cos\frac{3\theta}{2}\right] + (\sigma_1 - \sigma_2)\cos 2\alpha \dots\dots (18)$$

$$\sigma_y = \frac{KI}{\sqrt{2\pi r}}\cos\frac{\theta}{2}\left[1 + \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right] + \frac{KII}{\sqrt{2\pi r}}\sin\frac{\theta}{2}\left[\sin\frac{\theta}{2}\cos\frac{\theta}{2}\cos\frac{3\theta}{2}\right] \dots\dots (19)$$

$$\tau_{xy} = \frac{KI}{\sqrt{2\pi r}}\left[\sin\frac{\theta}{2}\cos\frac{\theta}{2}\cos\frac{3\theta}{2}\right] + \frac{KII}{\sqrt{2\pi r}}\cos\frac{\theta}{2}\left[1 - \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right] \dots\dots (20)$$

$$KI = \frac{\sigma_1\sqrt{\pi a}}{2\left[\left(1 + \frac{\sigma_2}{\sigma_1}\right) + \left(1 - \frac{\sigma_2}{\sigma_1}\right)\cos 2\propto\right]} \dots\dots\dots (21)$$

$$KII = \frac{\sigma_1\sqrt{\pi a}}{2\left[\left(1 - \frac{\sigma_2}{\sigma_1}\right)\sin 2\propto\right]} \dots\dots\dots (22)$$

And these stresses have been calculated for a specific angle of attack and swept angle , also it must be noticed that this solution is valid only for surface crack less than 0.5h because the classical plate theory assumed that the middle line along the thickness of the plate will remain straight and its position in the z direction will changed with the varying thickness .

V. Numerical Analysis

In this research FLUENT software and ANSYS software had been used.

First FLUENT program part:

The X- 29 aircraft wing had been taken with NACA0004

Global Journal of Researches in Engineering (A) Volume XI Issue VII Version I December 2011

The two dimensional model drawn in many sections in GAMBIT program as shown in Fig (1) .

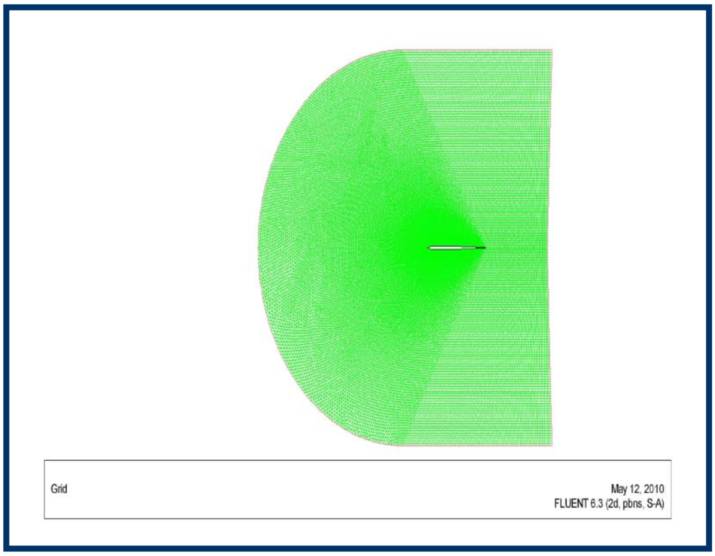


Figure (1) : mesh in Gambit

the mesh have done using quadrilateral element, boundary conditions were , 1) airfoil surface as wall 2) circumference about airfoil are all pressure far field type . Airflow applied on airfoil sections in FLUENT in order to obtain pressure distribution on wing body. Specifications had been used are: Pressure based, two dimensional, implicit formulation, steady time, absolute velocity, Green - Gauss cell - based solver option. Air properties; Ideal-gas, Sutherland viscosity (as shown in Fig 2), solution controls; modified turbulent viscosity, standard pressure, second order density, momentum, viscosity ,For Mach number from (.8, 1, 1.2, and 1.6) iteration has been done and pressure distribution for five sections on wing in order to find average resultant pressure on wing for each Mach number.

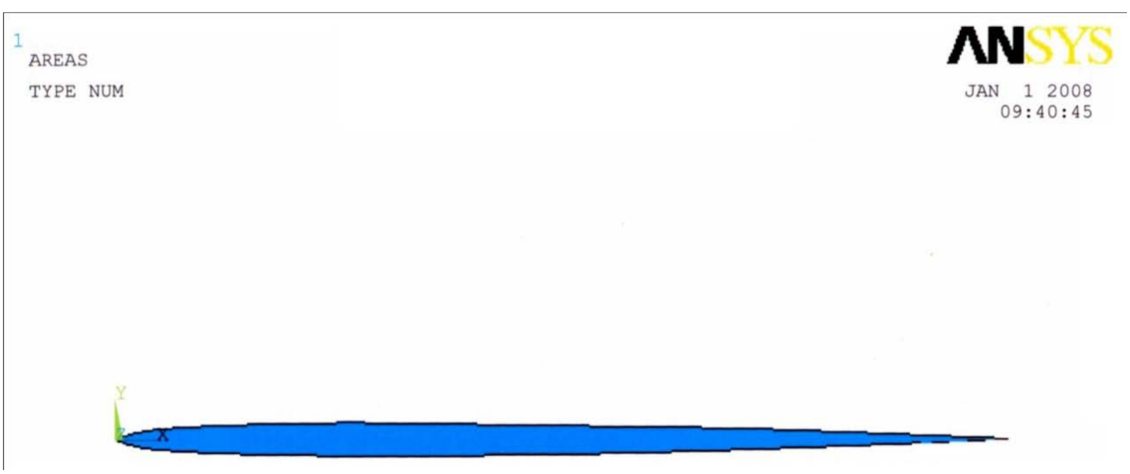


Fig (2) : the profile of airfoil and the origin of the axis.

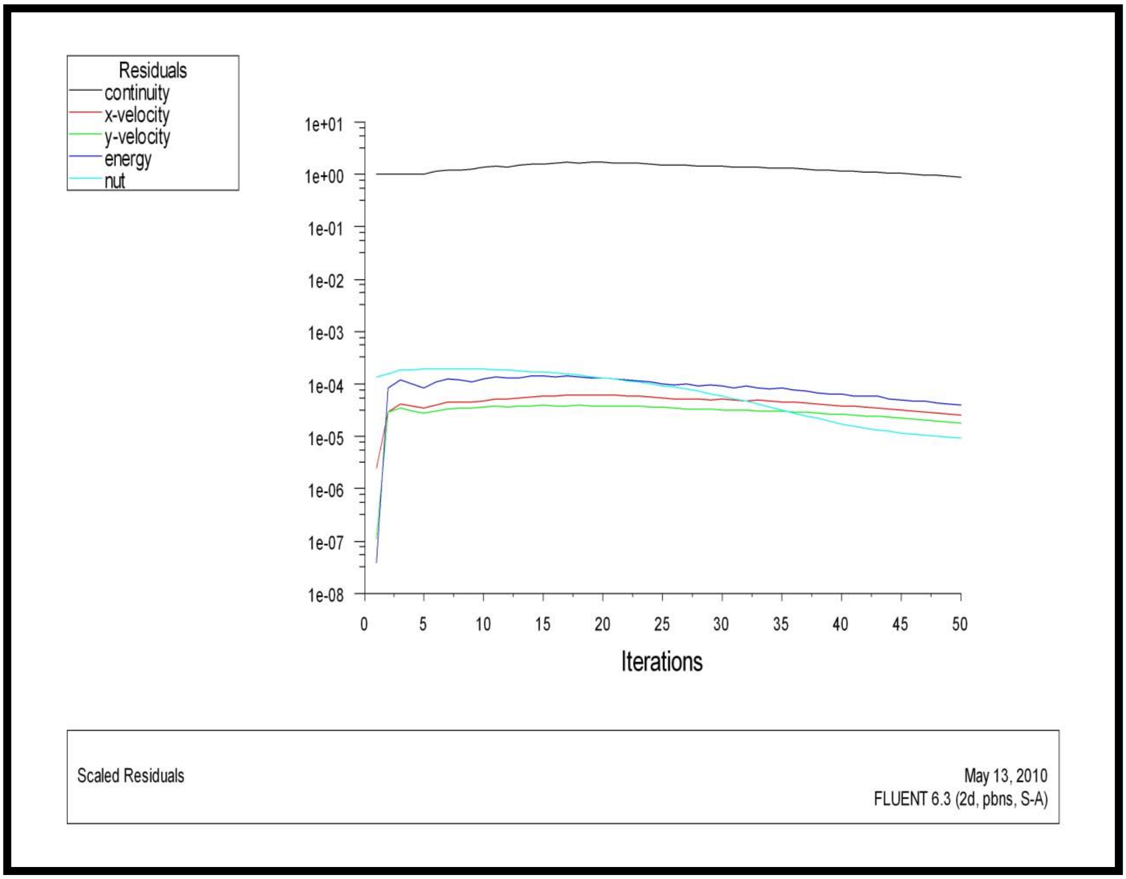


Figure (3) : iterations of solver

ANSYS program first the full X-29 aircraft wing drawn with section NACA0004 but only a skin (shell element) Second part; structural analysis in ANSYS program Shell thickness (0.01m) material used (aluminum) young modulus 70e9 Pascal, Poisson ratio 0.3 density 2800 kg/m3 element type was shell 63 for isotropic type material as shown in Fig (4).

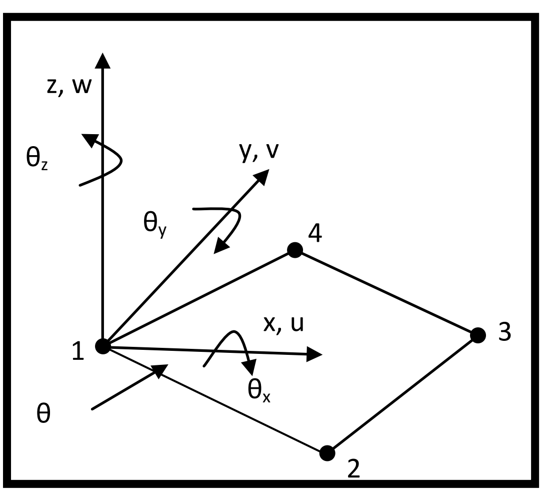


Figure (4) : shell 63 element

The boundary conditions taken are clamped from wing root and free for all other points like cantilever beam as shown in fig (5a,b,c), the pressure distributed on the wing area for example at Mach = 0.8 average pressure then static analysis have been done to obtain stress distribution and deflection distribution on the wing ,each velocity of air generates a different value of pressure on the other hand , the dynamic load function applied (exponential relationship) with time to find pressure change with time , so we take deferent times (0 to 1) and calculate pressure at each time and apply that pressure on wing area in ANSYS also to get stress and deflection distribution at each value to compare stress values and the position of maximum stress for times and velocities of air changes .

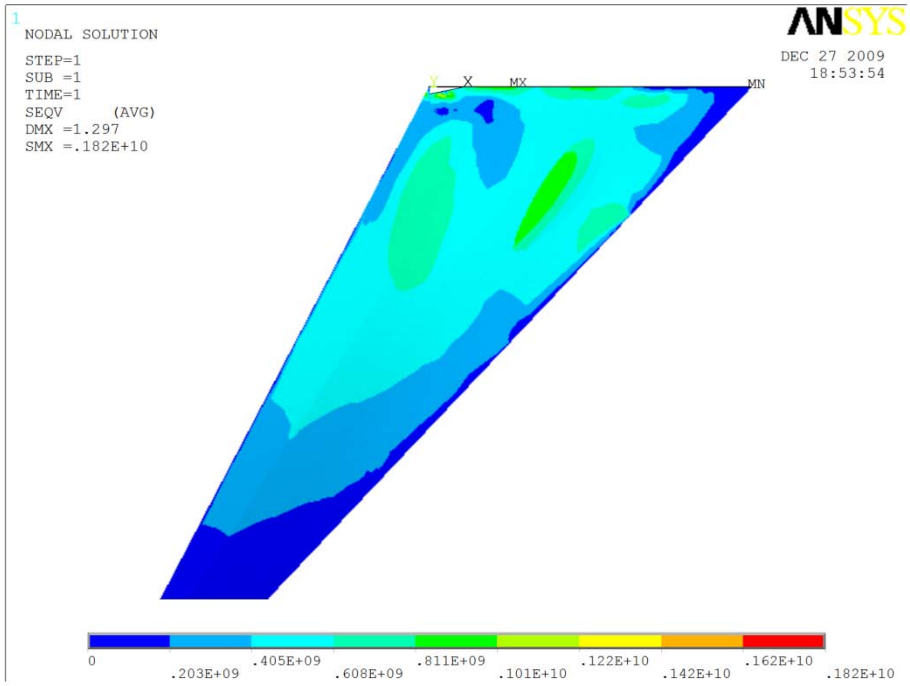


Fig (5-a) : the pressure distribution at mach number 0.6

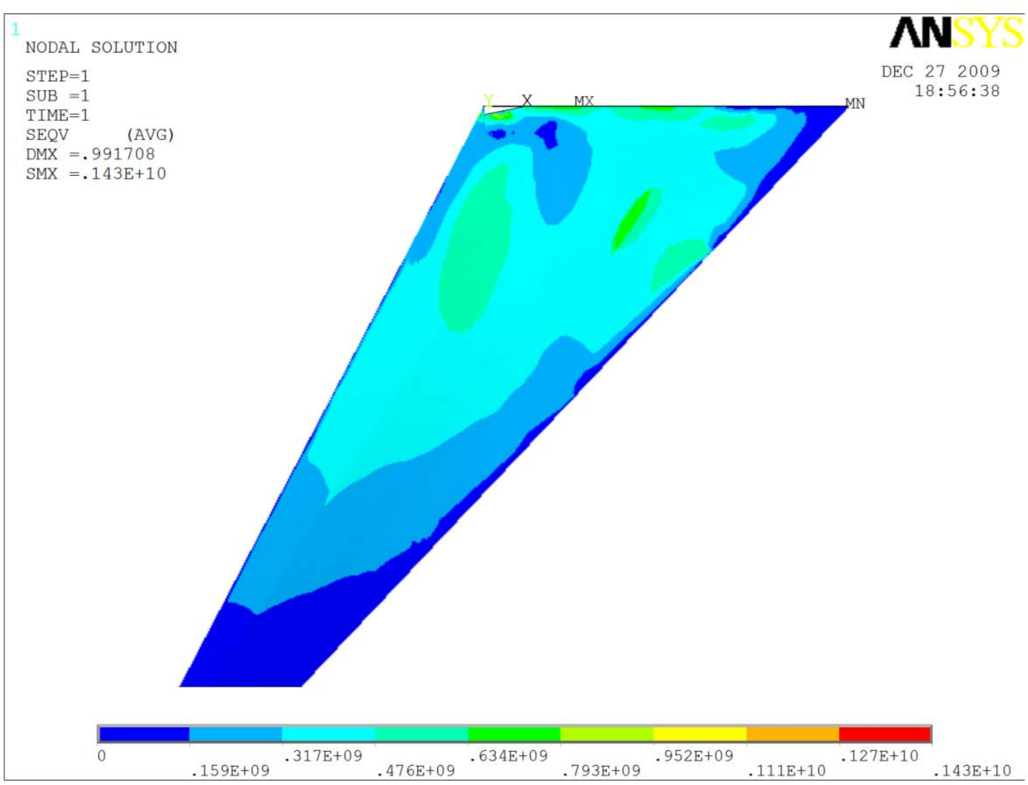


Fig (5-b) : the pressure distribution at mach number 0.8

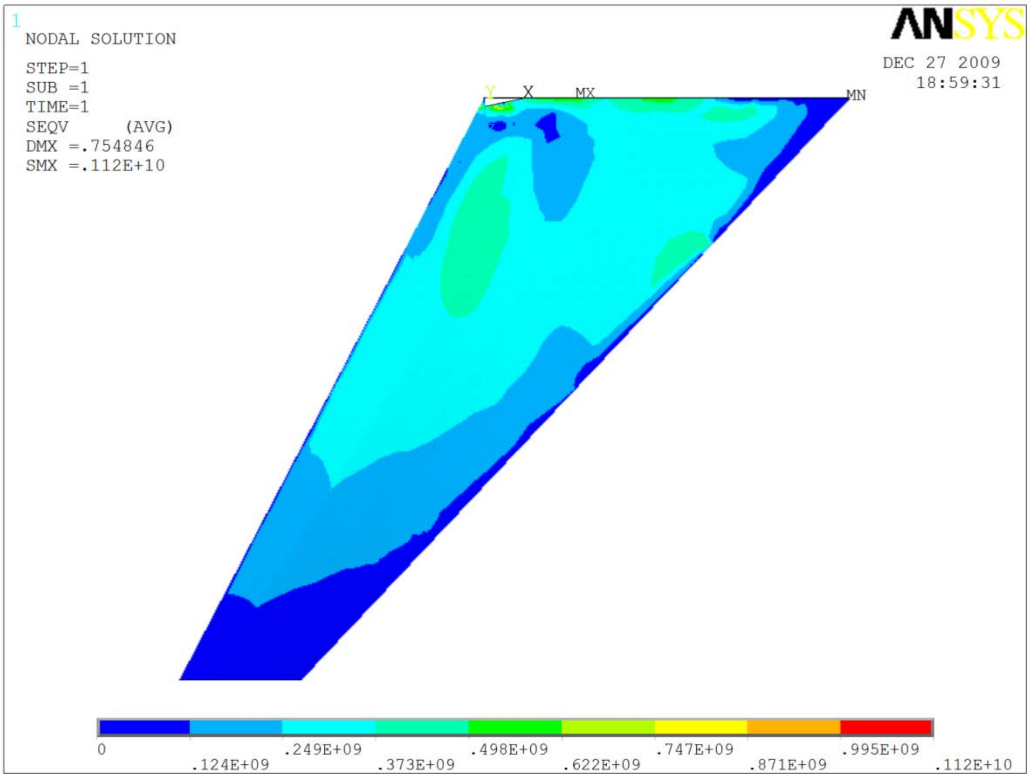


Fig (5-c) : the pressure distribution at mach number 1.0

VI. Model Generation by ANSYS10

The ultimate purpose of a finite element analysis is to re-create mathematically the behavior of an actual engineering system . In other words, the analysis must be an accurate mathematical model of a physical prototype . In the broadest sense, the model comprises all the nodes, elements, material properties, real constants, boundary conditions and the other features that used to represent the physical system. In ANSYS10 terminology, the term model generation usually takes on the narrower meaning of generating the nodes and elements that represent the special volume and connectivity of the actual system. Thus, model generation in this study will mean the process of defining the geometric configuration of the model's nodes and elements. The program offers the following approaches to model generation :(a) Creating a solid model , (b) Using direct generation and (c) Importing a model created in a computer-aided design CAD system. The method used in this research to generate a model is solid model. In solid modeling some one can be described the boundaries of the model, establish controls over the size and desired shape elements automatically, i.e. drawing the three dimensional model and meshing using meshtool. Solid modeling is usually more powerful and versatile than other modeling, and is commonly the preferred method for generation models. The one Dimension model is done by drawing and dragging to get three dimension model then meshing with element shell63 .

Procedure is presented for modeling of wing skin shell with stiffeners by ANSYS10 software by using solid-modeling approach method. Hence the following is the program of modeling the characteristics of wing skin shell in APDL (ANSYS Parametric Design Language). (in the end of this file)

VII. Shell 63

SHELL63 has both bending and membrane capabilities. Both in-plane and normal loads are permitted. The element has six degrees of freedom at each node: translations in the nodal x, y, and z directions and rotations about the nodal x, y , and z-axes. Stress stiffening and large deflection capabilities are included. A consistent tangent stiffness matrix option is available for use in large deflection (finite rotation) analyses. Other elements are SHELL181 (plastic capability) and SHELL281 (midside node capability) .

SHELL63 Assumptions and Restrictions

- Zero area elements are not allowed. This occurs most often whenever the elements are not numbered properly.
- Zero thickness elements or elements tapering down to a zero thickness at any corner are not allowed.
- The applied transverse thermal gradient is assumed to vary linearly through the thickness and vary bilinearly over the shell surface.
- An assemblage of flat shell elements can produce a good approximation of a curved shell surface

provided that each flat element does not extend over more than a 15° arc. If an elastic foundation stiffness is input, one-fourth of the total is applied at each node. Shear deflection is not included in this thin-shell element.

- A triangular element may be formed by defining duplicate K and L node numbers as described in Triangle, Prism, and Tetrahedral Elements. The extra shapes are automatically deleted for triangular elements so that the membrane stiffness reduces to a constant strain formulation. For large deflection analyses, if KEYOPT(1) = 1 (membrane stiffness only), the element must be triangular.
- For KEYOPT(1) = 0 or 2, the four nodes defining the element should lie as close as possible to a flat plane (for maximum accuracy), but a moderate amount of warping is permitted. For KEYOPT(1) = 1, the warping limit is very restrictive. In either case, an excessively warped element may produce a warning or error message. In the case of warping errors, triangular elements should be used (see Triangle, Prism, and Tetrahedral Elements).

VIII. Analytical and Numerical Results

From numerical analysis of distribution the pressure along the wing cord , it can be found that the peak pressure coefficient is not affected by changing mach number from (0.6) to (1.4) as shown in figure (6) while when using these values of the leading edge of the wing in the main equation of the analytical solution of this research and finding the value of C_p along the rectangular plate as shown in figure (7) , it can be seen that for mach number below 1 , the theoretical results is the same when compared with the numerical one and this is because the value of C_p is not reach the critical value .

When using much number greater than 1 the numerical results shows different behavior then the theoretical results and this is because the value of C_p at some regions become greater than the critical value and the load substituted as an impact load and its value taken from eq (14) so that the effect of shock wave have been pronounced and its peak values increased with increasing the mach number when its position become nearer to the trailing edge in the x direction . also in the y direction on the face of the airfoil the effect of aerodynamic pitching , yawing and rolling moments increased the potential energy non linearly causes a fluctuating of the value of C_p higher than the experimental values from previous researches which give more safety in the design and must be avoided in the design of air craft wing as shown in Fig (8).

In fig (9) it can be shown using the analytical solution that the behavior of pressure coefficient C_p will be changed along the plate when there is defect of length 4 mm with different surface depth of (0.1) of thickness h and there will be many asperities of shock waves and are much higher than the plate without defect causes sudden change in the aerodynamic pressure and this is because the variation of the stress intensity factor at the edge of the crack along the time of impact of shock wave in period time τ =(0.25 , 0.5 , 0.75) of the total impact time t so that it must be avoided in the design of airfoil for specified angle of attack and swept angle . It can be seen from Fig (10) the percentage of increasing the pressure coefficient and locally mach number will be max at 0.5 t with sudden drop at the end of 0.75 t .

IX. Conclusion

A new method have been used in this research for calculating the stress distribution on a surface of wing for a specific angle of attack and swept angle by using classical plate theory combined with numerical results by ansys10 .

Also the effect of the defects at the surface of the wing on the pressure coefficient C_p and the values of stress intensity factor at the edge of the crack tip have been studied during the time of shock wave in transonic flow .

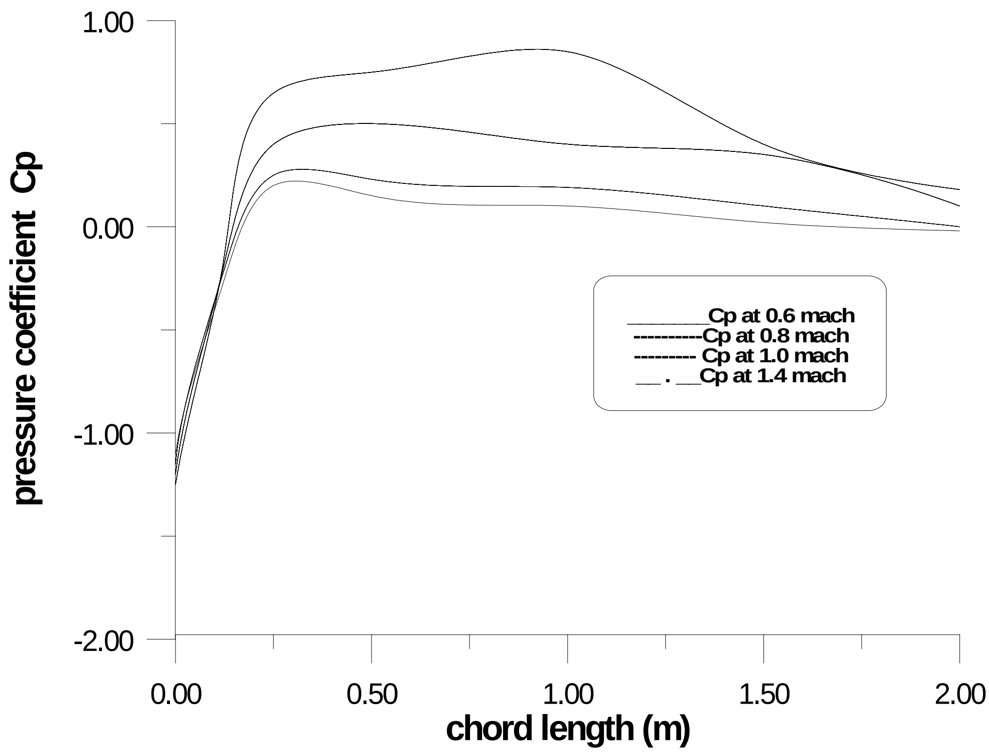


Fig (6) The variation of the pressure coefficient numerically with the cord length .

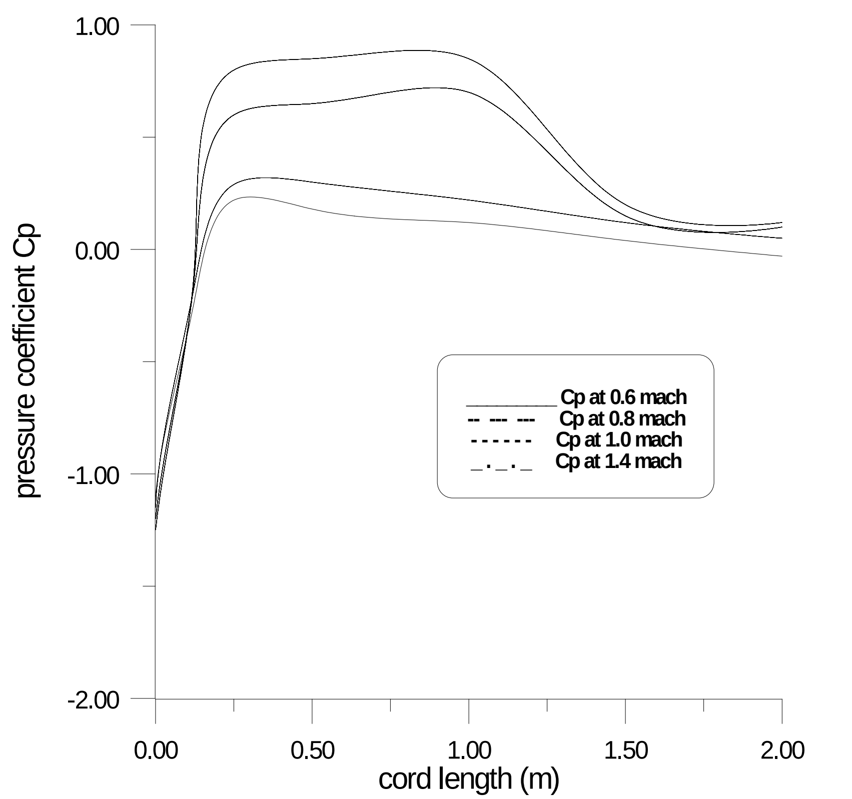


Fig (7) the variation of the pressure coefficient analyticaly with mach number

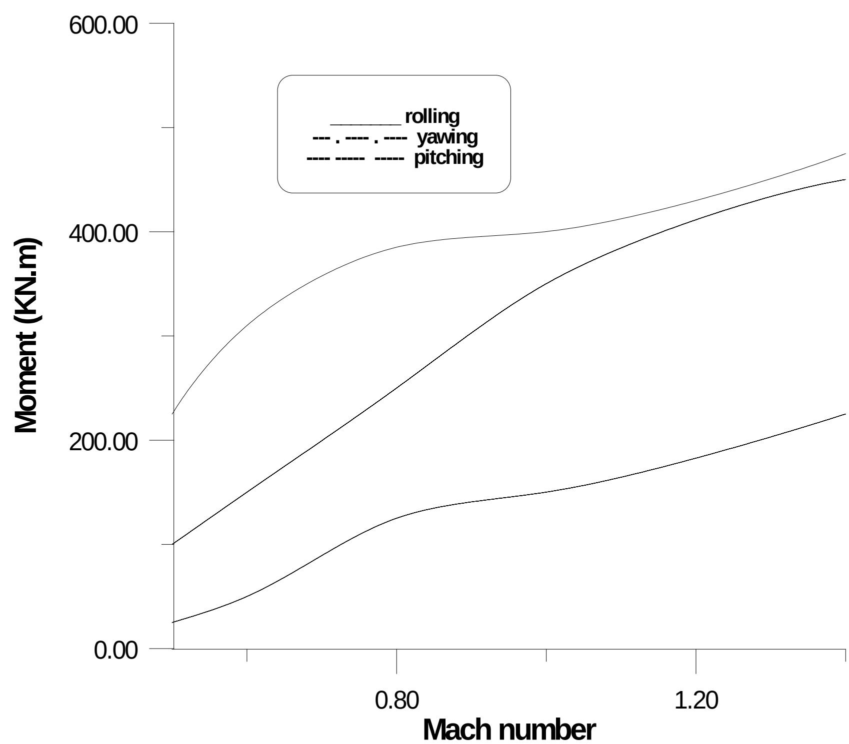


Fig (8) Aerodynamic moments with Mach number

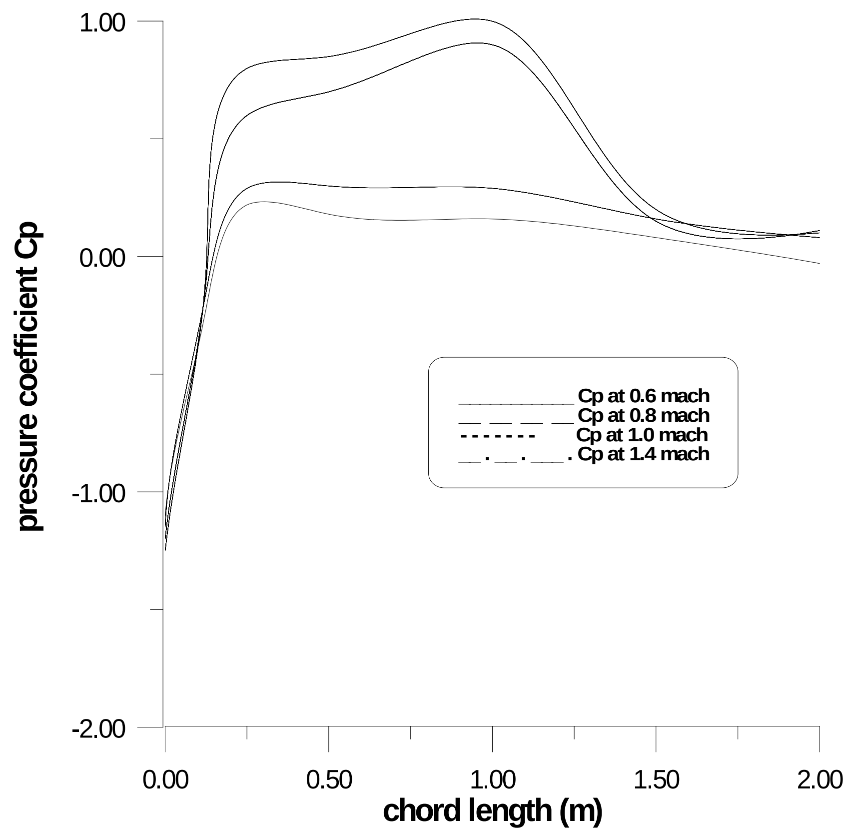
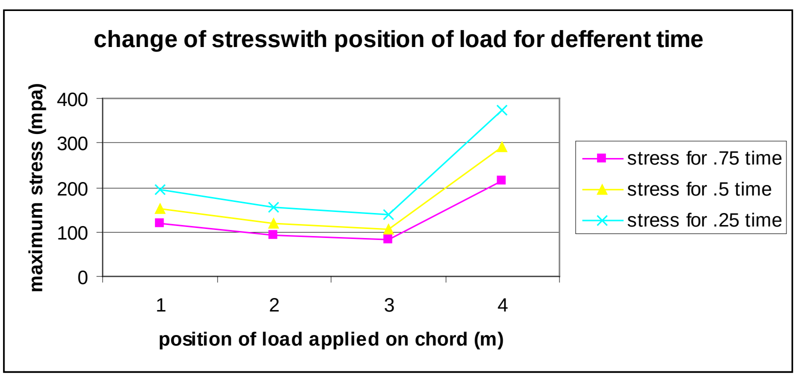


Fig (9) The variation of the pressure coefficient analatically along the cord length with surface crack 4 mm in 1 m cord length



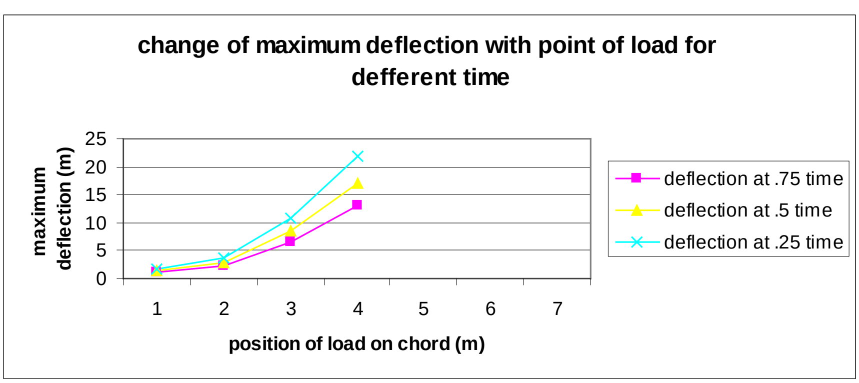


Fig (10) : the variation of the stress and deflection with impact time .

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