Buckling analysis of wing upper skin panels of a transport Aircraft

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Abstract--- **Aircraft wings are the major lift-load carrying members in the airframe structure. 80% of the lift load will be carried by wing structure in a transport category aircraft. Because of the lift load, wings will bend upward during flight. This action makes the upper skin of the wing to experience compression and the lower skin tension. This will cause the upper skin to buckle. Buckling will reduce the load carrying capability and the component may fail below the design limit load. For a structure to be safe against buckling buckling factor should be greater than one. In this work a standard transport aircraft wing is considered and buckling analysis is carried out. The initial design was found to buckle. So several design modifications were made to make the design safe in buckling. Linear static analysis is also carried out.**

Keywords – **Wing, Buckling, Lift load, Aircraft**

1 INTRODUCTION

Wing of an aircraft consists of basic structural members like stringers running along the wing span, ribs positioned at different directions along the span wise direction. Front and rear spars and upper and lower skins covering these internal components. Each of these components acts like a beam and torsion member as a whole. Aircraft loads mainly consist of air pressure on the skin, concentrated loads from the landing gear, power plants, passenger seats etc... These loads are to be collected locally and transferred to the major load carrying members. The skin is thin and has a little bending stiffness to resist air pressure. To avoid incurring large deflections in the skin, longitudinal stiffeners are provided to pick up the air loads. Materials with moderate amount of bending stiffness are used as stiffeners. So these transverse loads taken by the stiffeners must be transferred quickly to more rigid ribs or frames. The ribs collect all the transverse loads and transfer them to two wide-flange beams (spars) that are designed to take the transverse shear loads. Thus the load is transferred from skin to the spars. Stiffeners also contribute to the total bending capability of the box beam.Wing consists of axial members in stringers, bending members in spars, and shear panels in the cover skin and webs of spars. The wing is a complex structure and its analysis will be tedious, this complex structure reduced to a less complicated shape so that analysis can be made easily. So the wing is considered as a box like section for ease of analysis.

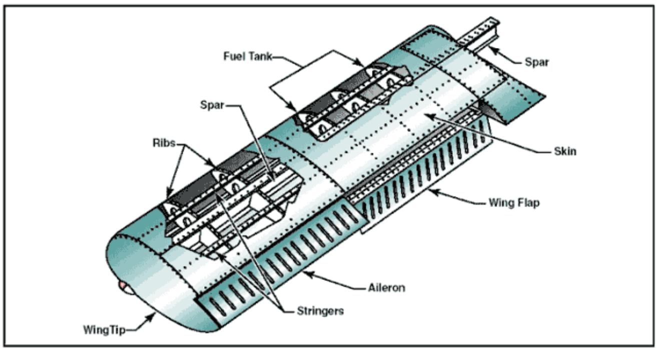

Fig: 1 Parts of a wing

2 BUCKLING

Two major causes which cause the sudden failure of a mechanical component are: material failure and structural instability, which is often called buckling. For material failures yield stress is considered as the design criteria for ductile materials and the ultimate stress for brittle materials. Buckling refers to the loss of stability of a component and is usually independent of material strength. The load at which buckling occurs will depend upon the stiffness of a component, and is independent of strength of the material. When a structure whose length is larger than either of its other two dimensions, is subjected to axial compressive stress, due to its size its axial displacement is going to be very small compared to its lateral deflection this phenomenon is called *Buckling*. Buckling is a tendency of slender compression members to bow out, which causes bending. When the combined bending stress and compressive stress exceeds the buckling capacity failure occurs. Buckling effects all compression members, such as columns, truss bars, bracing, etc. Buckling bends a column progressively.

3 LITERATURE REVIEW

Modern air craft wings are thin walled structures composed of ribs, spars and stiffened panels, where the top skin is subjected to compressive forces in flight that can cause buckling instability. If the skin panels are machined from a single billet of metal then the initial buckling performance can be significantly improved. This can be done by increasing the fillet radius along the line junction between the stiffener webs and the skin. The study shows a substantial increase in skin initiated buckling if the fillet is taken in to account of (1). Mr.pritishchitte, Mr. P k jadhav, Mr. ss bansode (2) in their study has done the preliminary sizing analysis of a wing box with the objective to fix a appropriate structure with in the given envelope. Different sections like rectangular sections, Z-sections and L-sections were analyzed and structure has been optimally designed. Poonamharakare, v k heblikar (3) in their work has estimated the static load carrying capacity of the wing box through linear static stress analysis. Top skin panels between the ribs and stringers were evaluated for their buckling critical loads. The analysis showed that the selected design is safe against buckling. In a thesis work done by Sridhar chitapalli (4) on preliminary structural design optimization of an aircraft wing box has done minimum mass design of compression skin panel. Thickness and width of the skin are identified as design variables. Using these design variables different dimensions of the stringer are found out.

Study about the plate buckling started in the early 1800. Navier derived the stability equation for a rectangular plate based on Kirchhoff assumptions in 1822. In 1891 Bryan formulated the critical buckling stress equation for a simply supported rectangular plate under uniaxial compression. He used the energy method to obtain the values of critical load. In 1925 Timoshenko also solved the same problem using another method. He assumed the plate to be buckled into several sinusoidal half waves in the direction of compression. He also explored the buckling of uniformly compressed rectangular plates that are simply supported along the edges perpendicular to the direction of applied load and other two edges subjected to various end conditions. Results have been reported in standard texts (Timoshenko and Gere 1961, Bulson 1970) (5). Chen Yu (2003) has studied the buckling behavior of rectangular plates subjected to intermediate and end loads. He considered both elastic buckling and plastic buckling behavior of these problems. Plate considered is simply supported along two opposite edges that are parallel to the direction of applied loads. The two edges may take any other combination of clamped, simply supported and free. Study also investigates the effect of various plate aspect ratios, intermediate load positions, boundary conditions on buckling factors(5).

. In airframe structural design data book (6) various wing design loads are given as shears, bending moments and torsion which results from air pressures and inertia loadings. For thin plates buckling equation is given by

$\sigma_{cr} = \frac{\pi^2 k_c E t^2}{12(1-\mu^2)b^2}$(Bryans plate buckling equation)

Where k= buckling coefficient which depends upon boundary conditions, type of loading and geometry. For cylinder the buckling equation is given by $P_{cri} = \pi^2 EI/L^2$

Eduard risks (7) in his study have applied finite strip method for the calculation of the buckling load of stiffened panels in wing box structures. This article describes the implementation of finite strip method that extends the scope of the analysis of the determination of the post buckling stiffness of the panels. Finite strip model (one dimensional) is the simplification of finite element model (two dimensional). Some of the computer implementations of finite strip method are BUCLASP (Vishwanathan and tamakuni, 1973) and VIPASA (wittrick and Williams,1974; plank and wittrick,1974). The method used for analysis of finite strip model here is PANBUCK which has the ability to analyze the initial post buckling behavior also. After obtaining the results it was compared with results of analytical solution.

Brian G Falzon and Grant P Steven (8) in their study have done combined experimental and analytical study of a Hat stiffened carbon fiber composite panel loaded in uniaxial compression. Good correlation between experimental and numerical strain and displacement results was achieved in the prebuckling and initial post buckling region of the loading history. The mode transition phenomenon was explained using Rayleigh-Ritz energy method. Knight&Starnes(9) investigated the post buckling strength of curved stiffened panels with various stiffener spacing, and concluded that combination of curvature and large stiffener spacing resulted in a snap through buckling . A similar observation in the skin bay of a stiffened flat panel was made by Dugundji et al (10). This phenomenon is referred to as a buckling mode shape change, or buckling mode transition. Starnes et al(11) was the first to demonstrate the post buckling strength of I stiffened composite panels loaded in uniaxial compression. Failure occurred due to skin stiffener disbonding followed by global skin buckling across the panel. After skin buckling additional load was distributed to different stiffener regions of the panels namely stiffeners and supporting edges which lead to failure near the edges.

In (2008) Leszek wittenbeck et al has done the analysis of a simply supported rectangular plate subjected to two types of compressive edge loads. First is uniformly applied along a part of the two opposite edges and the second one has a non uniform distribution. Critical value of the plate lies between values for uniformly distributed and concentrated load. Problem is solved by analytical method and results are compared with numerical solution (12).In 1993 J Loughlan and J.M Delaunoy in their study used finite strip method for determining the buckling performance of composite stiffened plates. Load conditions considered are of in plane shear and that of partial edge loads. They studied the effects of stiffener depths and orientation of fibres in the stiffeners for buckling. They concluded that for partial edge loading case use of symmetric cross-ply stiffener would be both efficient and practical (13). In 1994 Loughlan has done the buckling behaviour of composite panels subjected to combined in plane compression and shear loads. He used finite strip method to analyze the problem. A basic strip formulation is developed for the composite material construction, which is able to predict complex buckling modes associated with in plane compression and in plane shear loading (14).

J Rhodes examined some of the research on the postbuckling elastic and plastic behaviour of plates and plate structures. Post buckling behaviour of individual thin plates is governed by non-linear differential equations set up by von karman. In 1930 compression tests on plates with various materials having width variation was carried out by schuman and Back. The effective width expression for a plate simply supported along all edges was derived by von karman in 1932. Winter later modified this equation for plates having critical stress and yield stresses are almost same (15).

4 STATEMENT OF THE PROBLEM

When the aircraft is in flight, top portion will be under low pressure compared to bottom portion. This causes the aircraft to lift upward. As a result of this wing will bend upward causing the bottom skin under tension and top skin under compression. The wings are prevented from folding over the fuselage by the resisting strength of the wing structure. The bending action creates a tensile stresses on the bottom of the wings and a compressive stresses on the top of the wings. As the top skin of the wing is subjected to compression stress it may experience buckling at certain load value, and may lead to failure. In this project buckling analysis of the wing is carried out. The main objective of the work is to carry out the buckling analysis of the wing upper skin subjected to compression with FEM approach and verification through analytical approach.

5 PROPOSED WORK

5.1 Modelling

In this paper the modelling of a medium transport aircraft using CATIA V5 CAD software has been done. The wing box consists of seven ribs and two spars which are enclosed between upper and lower skin. The model is shown in fig below.

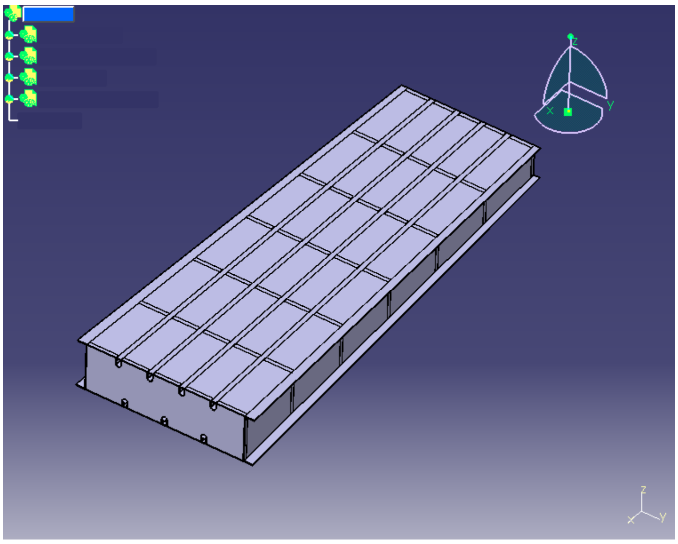

Fig: 2 CAD model of the wing box

5.2 Material used

Material used for the wing is aluminium alloy 2024-T351 whose properties are given below.

Young's modulus	73.1Gpa
Poisons ratio	.33
Density	27.271 KN/m^3
Yield strength	280 Mpa
Ultimate strength	470 Mpa

Table: 1 Properties of aluminium alloy

5.3 FEA Analysis

FEA Analysis of the wing box is done using MSC NASTRAN PATRAN Software. CATIA Model is extracted using MSC PATRAN Software; it acts as a pre processor and post processor. This model is meshed in PATRAN using suitable elements. 1 Dimensional beam and 2 Dimensional QUAD and TRIA elements are used for meshing. After meshing loads and boundary conditions are applied. Problem is solved in MSC NASTRAN software and the results are viewed in MSC PATRAN.

Boundary Condition: one end (larger) is constrained and at the other end a distributed load is applied on the seventh rib.

Load calculation of the wing box

Wing span-13.41m
Maximum take-off weight-2857kg
Capacity- pilot and five passengers
Design load factor -3g condition = 3*2857= 8571 kg
Factor of safety- 1.5
Ultimate design load=1.5*8571= 12859.5 kg
This total lift load is distributed 80% on the wing and 20% on the fuselage. Therefore load acting on wings = 12856.5*.8 = 10285.2 kg
Load on each wing = 10285.2/2 = 5142.6 kg
Length of one wing = wing span/2 = 13.41/2 = 6.705m
Total bending moment acting at the root section = 12188.128 kg-m
The length of the wing box model = 2.8012m
Load to be applied at the tip of the wing box in order to get the same bending moment as in original case is given by,
F*2.8012 = 12188.128
F = 4351.03 kg. This load is converted into UDL and applied at the tip of the win box.
After applying the above mentioned loads and boundary conditions linear static analysis of the wing box is carried out using MSC NASTRAN solver and the stress contour is obtained. For the same loads and boundary conditions buckling analysis is carried out and buckling factor is obtained.

6 RESULTS

6.1 Linear static analysis

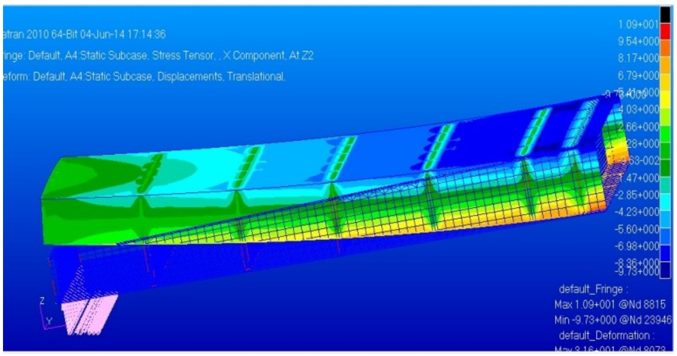

Fig: 3 Stress contour

Maximum stress is obtained as 15.9 kg/mm^2.

6.2 Buckling analysis

Buckling analysis was performed on the default model. But buckling factor obtained was more than one. So different design modifications were tried to get a buckling factor more than one. Finally the buckling factor was obtained as 1.0477.

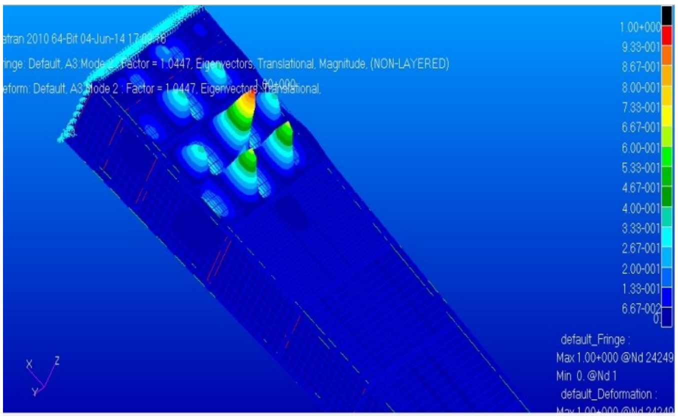

Fig: 4 Buckled shape of critical panel

Table below gives the buckling factor obtained for different design modifications.

Serial No	Design Modification	Buckling factor
1	Default Model	.236
2	Thickness increased	.404
3	Additional stringer at the root	.920
4	Additional rib at the root section	1.014

Table: 2 Buckling factor for different design modifications

6.3 Local buckling analysis of the top skin panels

Local buckling analysis of the critical panel is done in order to verify the results and to understand the manner in which critical panel behaves in reality. The compressive stresses acting on critical panel of top skin is found out and converted it to compressive force. This compressive force is then applied on the critical panel and the buckling analysis is done and the buckling factor is obtained. Theoretically the critical stress is calculated by using the below equation for elastic buckling strength of flat sheet in compression.

$$\sigma_{cr} = \frac{\pi^2 k_c E}{12(1-r^2)} * \left(\frac{t}{b}\right)^2$$

k_c= Buckling coefficient which is obtained from following graph.

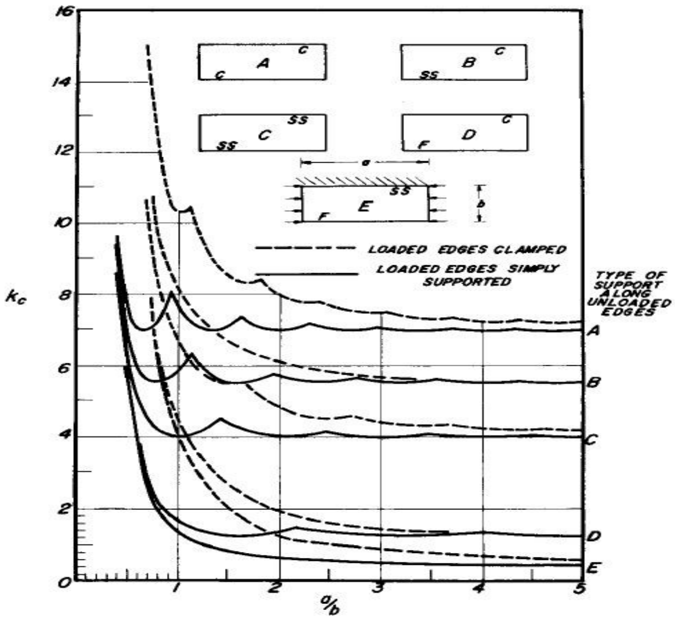

Fig: 5 k_c values for different edge support conditions

This critical stress is multiplied with area of cross section to obtain the critical load. The buckling factor is obtained as the ratio of critical load and applied load.

BF= Critical load/Applied load

6.3.1 Buckling results of critical panel

All edges simply supported

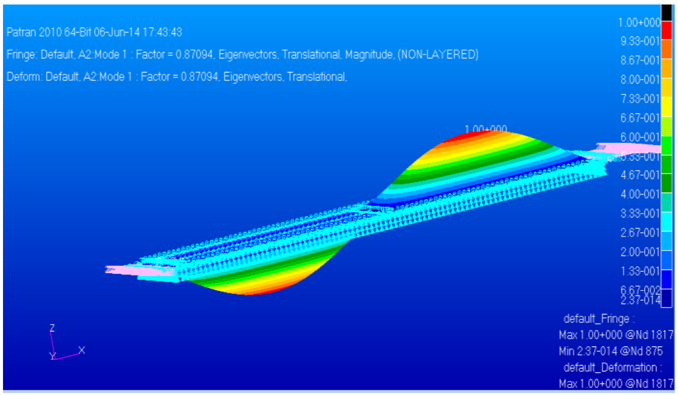

Fig: 6 Buckled shape

All edges clamped

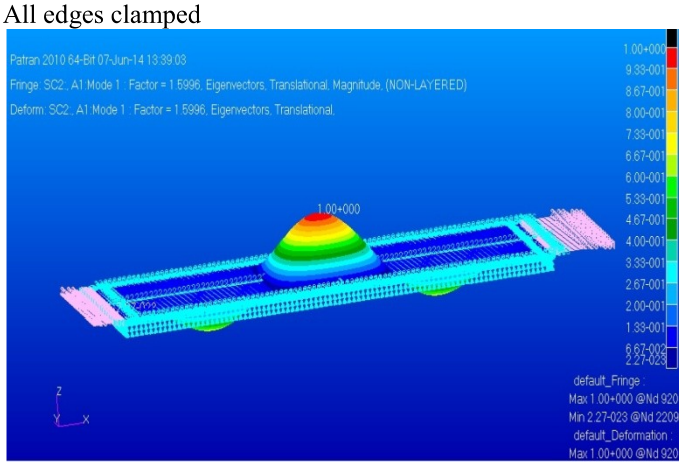

Fig: 7 Buckled shape

6.3.2 Buckling factor obtained for different boundary conditions

Serial No	Edge condition	Patran Buckling factor	Theoretical Buckling factor
1	All edges simply supported	.8706	.8579
2	All edges clamped	1.5996	1.631
3	Long edges clamped	1.4933	1.46
4	Short edges clamped	.9662	.9853

Table: 3 Comparison between theoretical and Patran bucking factor

The buckling factor obtained in global analysis is 1.0477. None of the buckling factor obtained in local analysis matches with global analysis result. This means that in reality the critical panel behaves neither like any of the assumed edge support conditions. The behaviour will be somewhat in between above mentioned support conditions.

7 CONCLUSIONS

Buckling of the upper skin panels of the wing is a major factor that must be considered during the wing design. Different methods to improve buckling load and post buckling strength are studied by scientists. For safety against buckling, buckling factor should be greater than one. Wing should be designed ensuring these criteria. Several methods have been adopted for the calculation of buckling load of stiffened panels in wing box structures. Post buckling strength of stiffened panels is also studied by scientists. From the static analysis maximum stress on the wing is found to be 15.9 kg/mm^2 which is less than the yield strength of aluminium. Hence the design is safe. Buckling analysis is carried out for a number of design modifications. Finally the buckling factor is obtained more than one and hence it is concluded that buckling does not takes place. Local analysis is carried out for critical panel and the results are verified through analytical approach. The FE results are in good agreement with theoretical values.

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