BUCKLING AND LINEAR STATIC ANALYSIS OF FUSELAGE STRUCTURE SUBJECTED TO AIR LOAD DISTRIBUTION

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ABSTRACT

The most important aspects in the aircraft design are safety and weight of the structure. Fuselage is one of the major components of the aircraft. Depending on the mass distribution of the fuselage structure the inertia forces will vary along the length of the fuselage. The inertia force distribution makes the fuselage to bend downward about wing axis. This bending will create tension in the upper portion of the fuselage; simultaneously the bottom portion of the fuselage will undergo compression. The current investigation addresses the issue of the compression buckling of the panels in the bottom portion of the fuselage. The panels with maximum compression load will be identified as critical panels for buckling analysis. Classical approach will be followed to calculate the critical buckling load on each panel. These calculations will be substantiated by panel buckling analysis through finite element method.

Keywords: Compression Buckling, Fuselage, Liner Static Stress Analysis, Global Buckling Analysis, Critical Buckling Load, Buckling Load Factor.

I. INTRODUCTION

Aircraft design is a complex and multi-disciplinary process that involves a large number of disciplines and expertise in aerodynamics, structures, propulsion, flight controls and systems amongst others. During the initial conceptual phase of an aircraft design process, a large number of alternative aircraft configurations are studied and analysed. Feasibility studies for different concepts and designs are carried out and the goal is to come up with a design concept that is able to best achieve the design objectives [1]. The fuselage is the main structure or body of the fixed-wing aircraft. It provides space for cargo, controls, accessories, passengers, and other equipment. In single-engine aircraft, the fuselage houses the power-plant. In multiengine aircraft, the engines may be either in the fuselage, attached to the fuselage, or suspended from the wing structure. A stiffened fuselage structure is assumed to consist of several stiffened panels which are more widely experimentally tested and numerically verified in the literature [2, 3, and 4]. Depending on the mass distribution of the fuselage structure the inertia forces will vary along the length of the fuselage. The inertia force distribution makes the fuselage to bend downward about wing axis. This bending will create tension in the upper portion of the fuselage; simultaneously the bottom portion of the fuselage will undergo compression. There exist different distinct buckling modes for a stiffened panel. Depending on the spacing and the size of the frames and stringers relative to the shell skin, buckling deflections could be developed within each shell panel or encompass a number of stiffened panels. In the first case, commonly referred to as skin buckling, the line of connection between skin and frames remains virtually straight, with the stringers exhibiting only minor radial movement. The second case, known as overall buckling, involves a variable radial displacement of the stringers [5].

1.1 Fuselage

Fuselage structures carry heavier loads found in the aircraft structure. The particular design of a fuselage depends on many factors, such as the size, weight, speed, rate of climb, and use of the aircraft. The wing must be constructed so that it holds its aerodynamics shape under the extreme stresses of fuselage loading. In its simplest form, the fuselage is a framework made up of bulkheads and longerons and covered with metal sheet. The longerons are the longitudinal structures in the fuselage. A longeron is part of the structure of an aircraft, designed to add rigidity and strength to the frame. It also creates a point of attachment for other structural supports, as well as the skin of the aircraft. They provide lengthwise support and the number of longerons present in an aircraft varies, depending on the size and how it is designed [6].

1.2 Buckling

There are two major categories leading to the sudden failure of a mechanical component: material failure and structural instability, which is often called buckling. For material failures you need to consider the yield stress 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 depends on the stiffness of a component, not upon the strength of its materials. When a structure whose order of magnitude of length is larger than either of its other 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 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. Increasing offset e increases bending, which in turn increases e further which finally causes buckling failure [1,6].

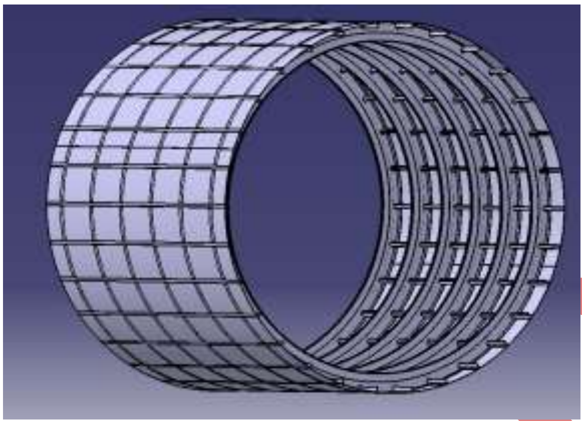
II. PROBLEM DESCRIPTION

When the aircraft is in flight, inertia forces act downward on the fuselage, tending to bend them downward. The fuselage is prevented from folding over the wings by the resisting strength of the fuselage structure. The bending action creates a tension stress on the top of the fuselage and a compression stress on the bottom surface of the fuselage. As the bottom skin of the fuselage is subjected to compression stress it may experience buckling at certain load value, and may lead to failure. Hence I am going to carry out buckling analysis of the fuselage structure to evaluate whether the bottom skin of wing is capable of withstanding the load without buckling.

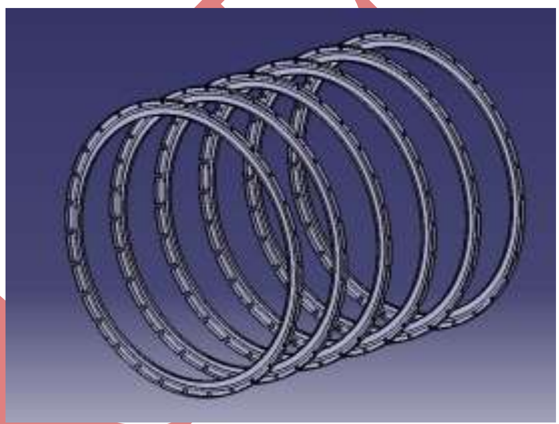
III. PROPOSED WORK

3.1 Modelling

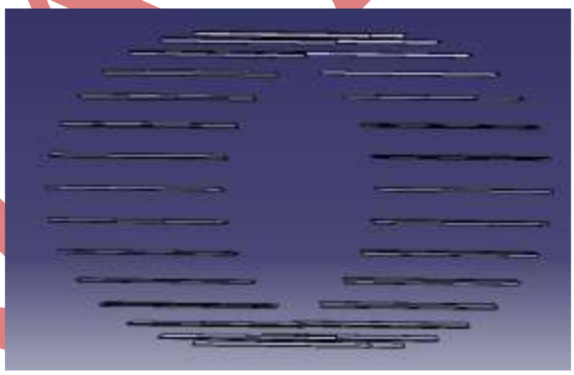
In this paper we have modelled the wing box of a medium transport aircraft using CATIA V5 CAD software. The fuselage model consists of six bulkheads circumferentially and thirty longerons, in longitudinal direction of the fuselage. The model is as shown in figure below



3.1.1 CATIA Model of Fuselage Structure



3.1.2 CATIA Model of L & Z Sec Bulkheads



3.1.3 CATIA Model of L-Section Longerons

3.2 FEA Analysis

The CATIA model is extracted using MSC PATRAN software; it is a pre-processor and a post-processor. This model is then meshed in Patran using suitable elements and the necessary boundary conditions and loads are applied. Boundary condition: The fuselage is fixed at the point of wing axis and load is applied on the upper half skin of the fuselage.

Load Calculation for the fuselage:

The load acting on fuselage by its self weight = 33.73 × 3 = 101.19 kg

Passenger weight +luggage +cargo =780 kg

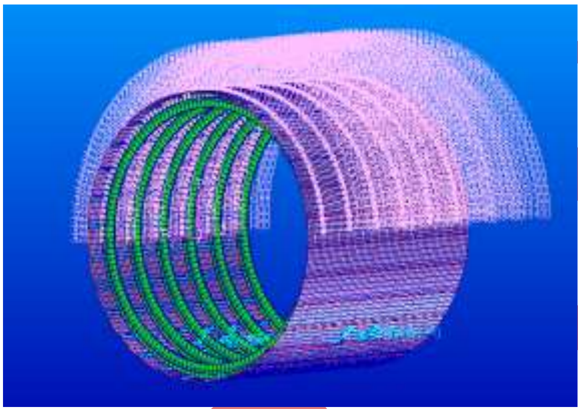
The total weight acting on the structure = 780 + 101.19 = 881.19 kg

The uniformly distributed load is applied on the upper half part of the fuselage.

$$UDL = \frac{Total\ load}{\pi r}$$

$$UDL = \frac{881.19}{\pi \times 1100}$$

UDL = 0.259922 kg/mm

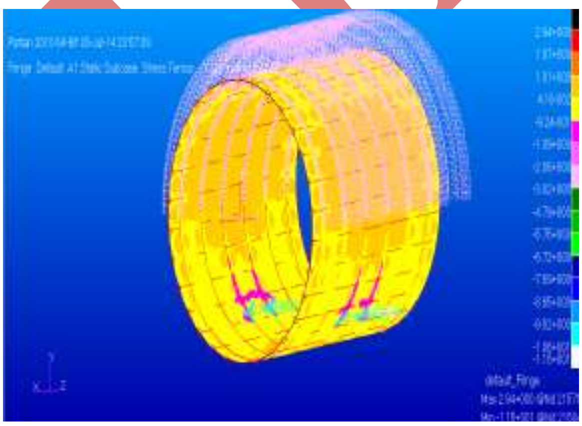


3.2.1 FEA Model of Fuselage with Loading and Boundary Condition

After applying the above mentioned Boundary conditions and load the Static analysis of the wing box is carried out using MSC NASTRAN, it is a solver and the stress contour is obtained. For the same BC's and load the Buckling analysis is carried out.

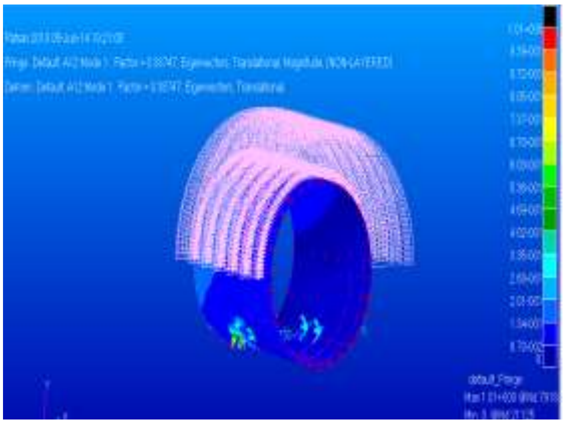
IV. RESULTS

4.1 Static Analysis Stress Contour



4.1.1 Linear Static Analysis Of Fuselage Structure

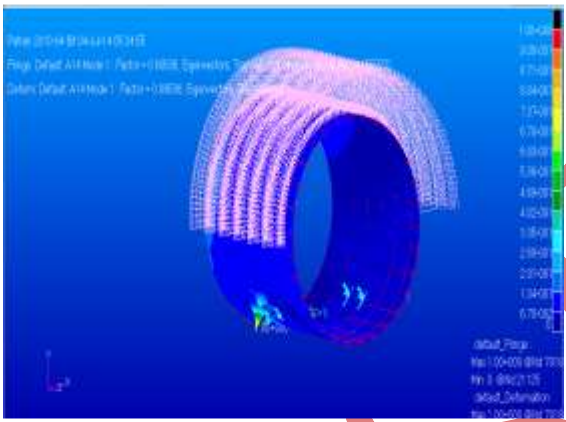
4.2 Buckling Analysis for varying thickness of bulkhead



4.2.1 Buckling Analysis Of Fuselage Structure. (Bulkhead Thickness L=2mm, Z=2mm)

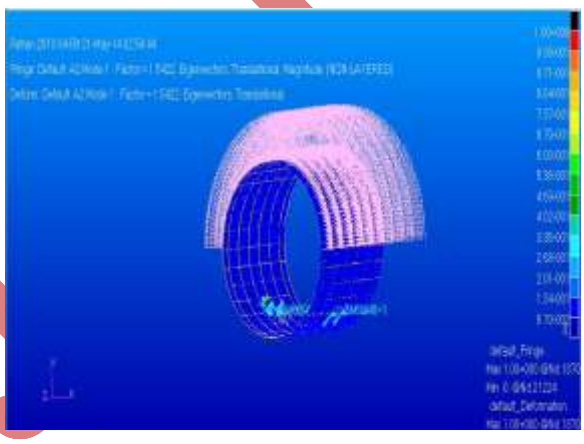
From the linear static analysis it is observed that maximum Stress 15.5kg/mm^2.

From the buckling analysis for the given load the buckling factor is 0.38747, which is less than one. From this we can say that the structure is going to fail. To overcome this thickness of the bulkhead is increased and analysed as shown below.



4.2.2 Buckling Analysis for Modified Fuselage Structure (Bulkhead Thickness L=2.5mm, Z=2mm)

From the above analysis it is observed that the buckling factor is still less than one. The buckling factor for the modified fuselage structure is 0.89536.

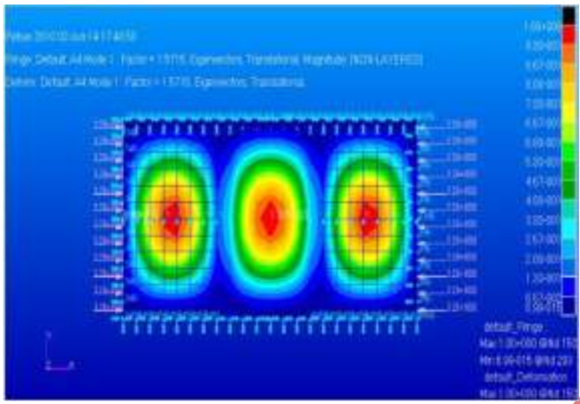


4.2.3 Buckling Analysis for Fuselage Structure (Bulkhead Thickness L=2.5mm, Z=2.5mm)

From the above analysis it is observed that the buckling factor is more than one. The buckling factor obtained from finite element analysis is 1.5422. So the structure will not buckle for given load.

4.3 Local Buckling Analysis

The critical panel of fuselage structure is taken from the linear static analysis results. The critical panel is one where the compressive stresses are maximum. Local buckling analysis is carried out for the critical panel. The loads are calculated for the panel from taking the stress around that panel.



4.3.1 Local Buckling Analysis of the Critical Panel

From the local analysis we got the buckling factor as 1.5715, which is more than one. This value is nearer to the global buckling analysis which is 1.5065.'

$$BF = \frac{p_{cr}}{p_{app}}$$

$$BF = \frac{1198.3}{749.887}$$

BF = 1.598

V. CONCLUSION

The segment of the fuselage is considered for the buckling analysis. Linear static analysis was carried out & it is observed that maximum compressive stress is 15.5kg/mm^2. The stresses are formed to be safe for the given load case. Then from buckling analysis obtained buckling factor is 1.522 which is greater than 1. It indicates that critical buckling load is more than applied load. Hence it is concluded that buckling does not takes, as well fuselage is being considered as capable of taking buckling load. So the design of fuselage is safe. Local buckling analysis was carried out and it is observed that the buckling factor is 1.571. For the validation theoretical calculations were made and the final value of buckling factor from the calculation is 1.598. Local buckling analysis value of buckling factor is in good correlation with theoretical value.

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