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# Study the effect of buckling on aircraft fuselage skin panel with or without airframe

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***Abstract*** **- The fuselage is one of the main components in any aircraft and its function is to hold all parts together and carries passengers. This fuselage part experience a different loads like static, fatigue, dynamic, buckling during landing, flying and take-off conditions. Now a day's aircraft undergo different type of failure modes, due to improper design, pilot error, weather conditions etc. In the present work, study the effect of buckling on aircraft fuselage skin panel with or without airframe. The result shows that fuselage skin panel without frame model undergo failure at low load with maximum deflection due to buckling effect, but fuselage skin panel with frame model can able to sustain high load with minimum deflection before buckling failure. In buckling analysis, buckling loads- critical loads at which a structure becomes unstable and buckled mode shapes, the characteristic shape associated with a structure's buckled response can be calculated.**

***Keywords:*** **Fuselage, buckling, airframe.**

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## I. INTRODUCTION

An aircraft is a machine that is able to fly by gaining support from the air and driven by jet engines or propellers. The main sections of an aircraft, the fuselage, tail and wing, determine its external shape. The load-bearing members of these main sections, those subjected to major forces, are called the airframe. Fuselage is based on French word fuseler, which means "to streamline". The fuselage, or body of the airplane, is a long hollow tube, which holds all the parts of an airplane together. The fuselage is hollow to reduce weight.

In order for an airplane to fly straight and level, the following relationships must be true **[1]**:

- Thrust = Drag
- Lift = Weight

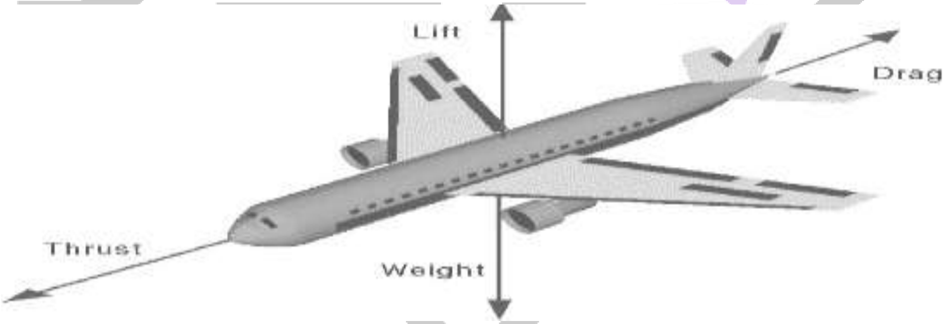


**Fig. 1 The forces acting on aircraft**

For analysis purpose Airbus A321 is used. It is a largest member of A320 family's. The Airbus A321 single-aisle medium range-airliner is the largest aircraft in the A320 range.


**Fig. 2 Airbus A321**

**Airbus A321 Specifications [2]**

| | |
|---|---|
| **Dimensions** | |
| Length | 44.5m |
| Wingspan | 34.1m |
| Height | 11.8m |
| Wing area | 122.4m$^2$ |
| **Weight** | |
| Maximum take-off weight | 83000-93500kg |
| Maximum landing weight | 73500-77800kg |
| Operating empty weight | 48100kg |
| Maximum zero fuel weight | 71500kg |
| Maximum payload | 23400kg |
| Standard fuel capacity | 23700-29680Litres |
| **Performance** | |
| Range with max payload | 5000-5500km |
| Cruise speed | 840km/h |
| Maximum speed | 890km/h |
| Maximum operating altitude | 11900m |
| Take-off field length | 2180m |
| Landing field length | 1580m |
| Engines | CFMI CFM56-5A/5B, 2*30000-33000 lb IAE V2500-A5, 2*30000-33000 lb |
| Fuel efficiency | 18.2g/pass*km |
| Fuel flow rate | 3200kg/h |
| **Cabin Data** | |
| Passengers | 220(1-class) |
| Passengers | 185(2-class) |
| Cabin width | 3.7m |

Many researchers have worked on designing this part through various techniques like finite element method, experimental method and analytical method. The researchers have carried out different analysis related to aircraft fuselage structure such as static, dynamic fracture, fatigue analysis etc., The static analysis can be made by different ways such that different conceptual designs that included as frames spacing was smaller compared to stringers spacing, frames spacing was larger compared to stringers spacing, frames and stringers spacing was approximately equal **[3]** and laminate constructions for stiffened fuselage panels in aircraft design **[4]**. The dynamic fracture analysis can be made by different ways such that dynamic fracture analysis of aircraft fuselage with damage due to two kinds of blast loads **[5]**, blast response of metal composite laminate fuselage structures with two material configurations such as aluminium and GLARE **[6]**. The fatigue analysis can be made by different ways such that damage tolerance analysis of aircraft reinforced panels **[7]**, fatigue cracks at many rivet locations in the skin panel **[8]**, and fatigue analysis for upper and lower folding beams on the rear fuselage **[9]**. The researchers are also made analysis related to predicting the service durability of aerospace components **[10]**, residual strength pressure tests analysis of stringer and frame stiffened aluminium fuselage panel with longitudinal cracks **[11]**, weight comparison analysis between a composite fuselage and an aluminium alloy fuselage **[12]**, impact of engine debris on fuselage skin panel **[13]**, damage analysis of aircraft structure due to bird strike **[14]**, damage prediction in airplane flap structure due to bird strike **[15]**, and analysis of high energy impact on a sheet metal aircraft structures **[16]**. The buckling analysis can be made by different ways such that post buckling response behavior of stiffened panels under compression **[17]** and post buckling response of stiffened panels under shear **[18]**. The researchers have worked on aircraft fuselage analysis, but they took flat riveted panel for analysis but fuselage has a circular arrangement with assembled parts due to this it is very difficult to find a critical area where maximum Von-Mises stress occure in fuselage structure under uniform axial compression or shear. Hence, the scope of this work reported in this paper is to study the effect of buckling on circular assembled fuselage skin panel with or without airframe.

## II. GEOMETRY OF THE MODELS

Buckling analysis is carried out for two sets of models; one is for stringer panel without frame and another is for stringer panel with frame as shown in figures 3 and 4 respectively.

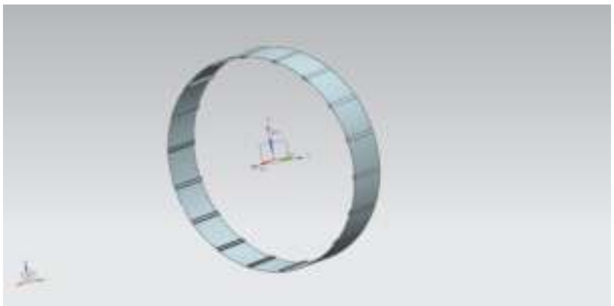

Fig. 3 Stringer panel without frame

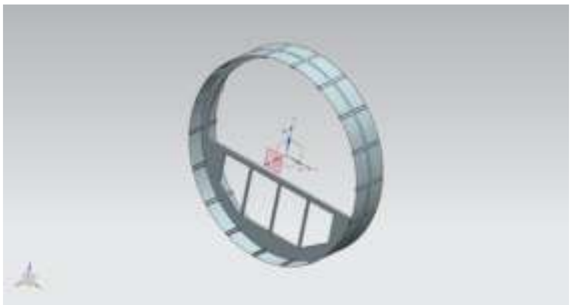

Fig. 4 Stringer panel with frame

## III. MESHING OF THE MODELS

The buckling models are meshed with 8 noded hexahedron elements is as shown in figures 5 and 6.

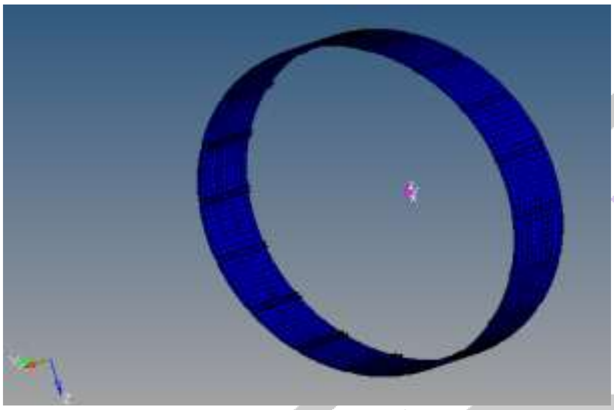

Fig. 5 Meshed model of stringer panel

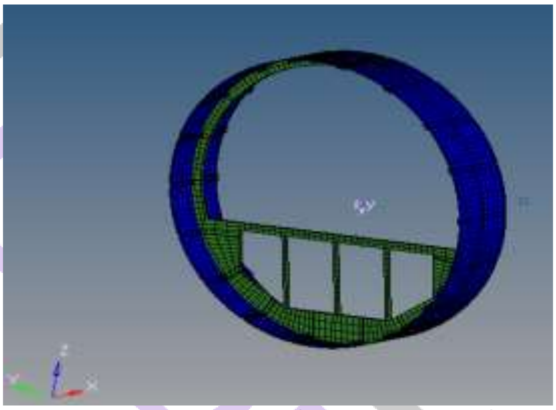

Fig. 6 Meshed model of stringer panel with frame

*3.1.Elements used:*-8 noded hexahedron element

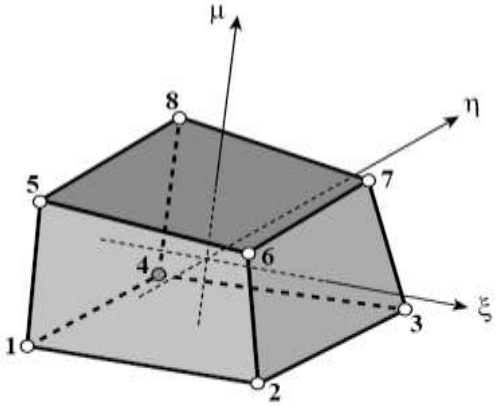


Fig. 7 8 noded hexahedron element

The 8 noded hexahedron element is a three dimensional element with 8 nodes at its corners. The element is defined by 8 nodes having three degrees of freedom at each node: translations in the nodal x, y and z directions (UX, UY, UZ). Hexahedron element is also called Brick element. The figure 7 shows an 8 noded hexahedron element.

## IV. MATERIAL SELECTED

After the meshing process next step is to assign the material properties and its behaviour. Selection of materials in aircraft construction is rather complex and is based on trade off amongst conflicting requirement of high strength, low density and easy of fabrication or processing. The material used in various parts of vehicle structures generally are selected by different criteria. The material used in the fuselage structure is Aluminium alloy 2024-T351 and its composition as shown in table 1.

Table 1 Composition of Aluminium alloy 2024-T351

| Composition | Wt. % |
|---|---|
| Al | 90.7-94.7 |
| Cr | Max 0.1 |
| Cu | 3.8-4.9 |
| Fe | Max 0.5 |
| Mg | 5.2-5.8 |
| Mn | 0.3-0.9 |
| Si | Max 0.5 |
| Ti | Max 0.15 |
| Zn | Max 0.25 |
| Others | Max 0.15 |

## V. LOAD AND BOUNDARY CONDITIONS APPLIED

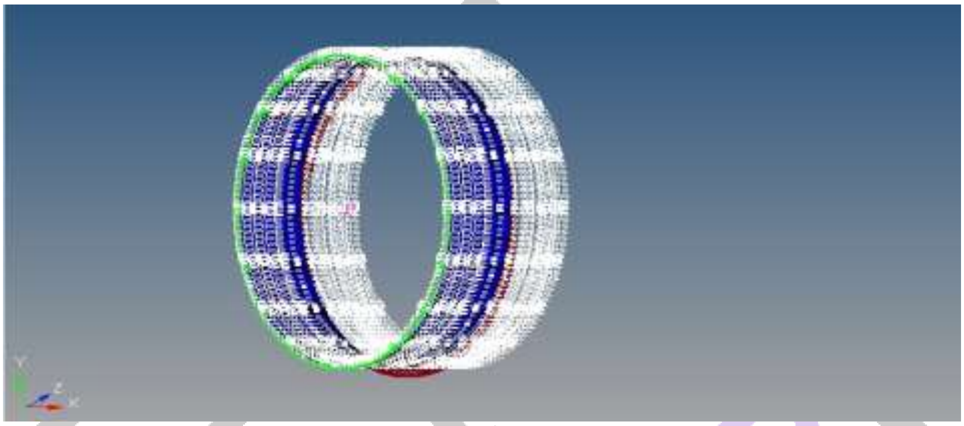

**Fig. 8 Load & BCs for stringer panel**

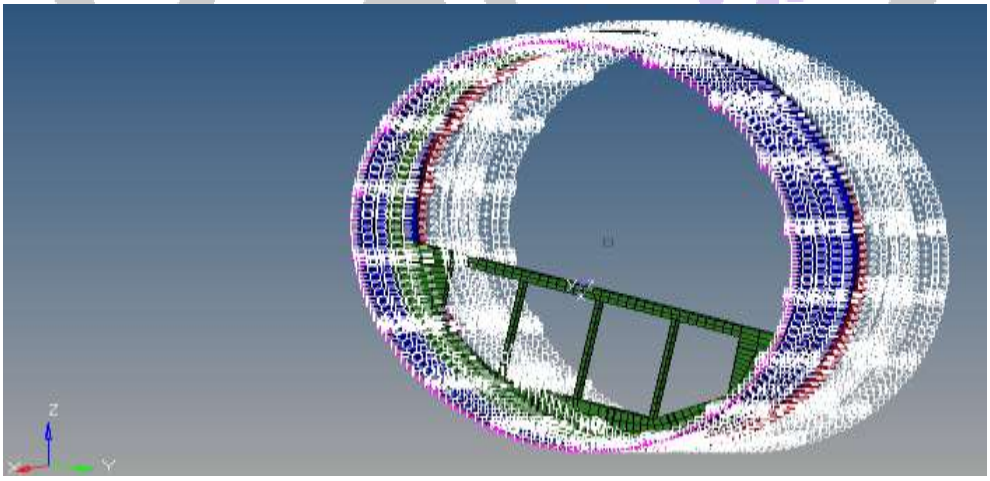

**Fig &. 9 Load BCs for stringer panel with frame**

In Buckling analysis the load is applied on one end of Stringer panel and another end of the stringer panel is constrained for all degrees of freedom. Here the aim is to determine the failure load for buckling for both stringer panel with and without frame. Hence the loads of different magnitudes are applied on one end of stringer panel. Due to space constraints only a load case of 20KN applied for stringer panel without frame and load case of 50KN applied for stringer panel with frame is as shown in figures 8 and 9 respectively.

To apply the load, a load collector of name FORCE is defined and to apply constraint a load collector of name SPC is defined in the software. The next step in the analysis is deck preparation that means preparing the final model for solving. The solver needs static analysis data before determining the failure load for buckling. Hence the control card of name STATICS is defined first and then by defining the control card BUCK, the analysis type is set to buckling analysis. Again in the sub panel by selecting the control card EIGRL the analysis is set to linear buckling analysis (Eigen value buckling analysis). Similarly by using TIME, PARAM and SOL control cards, the required output parameters like displacement, stress and strains are clearly defined. Then this final FEA model which is ready for solving is fed to the solver. Before that the FEA model which is in .hm file format is converted to .bdf or .dat file format because it is the required input file format for the MSC NASTRAN solver.

The solver takes around 10 to 15 minutes of time for solving in a Pentium dual core processor, 2GB RAM equipped PC. A number of output result files were generated after solving, among which a file of .bdf and .op2 format are used to generate the contour plots of stress and deflection in the Hyperview v11.0 software, which is the Post processor used in the analysis.

## VI. RESULTS AND DISCUSSIONS

*6.1. Results of stringer panel without frame*

For Aluminium alloy 2024-T351

We know that Yield strength = 350MPa

Factor of safety considered = 2

$$\text{Hence, Allowable stress} = \frac{\text{yield strength of the material}}{\text{Factor of safety}}$$

$$= 350/2$$

Allowable stress = 175MPa.

Table 2 Results of stringer panel without frame

| Applied Load in KN | End shortening in mm | Induced Von-Mises Stress in MPa |
|---|---|---|
| 0 | 0 | 0 |
| 20 | 1.115 | 25.21 |
| 40 | 2.229 | 50.41 |
| 60 | 3.344 | 75.62 |
| 80 | 4.459 | 100.8 |
| 100 | 5.574 | 126 |
| 120 | 6.688 | 151.2 |
| 130 | 7.246 | 163.8 |
| 132 | 7.357 | 166.4 |
| 134 | 7.469 | 168.9 |
| 136 | 7.580 | 171.4 |
| 138 | 7.692 | 173.9 |
| 139 | 7.747 | 175.2 |
| 140 | 7.803 | 176.4 |

The table 2 shows the Von-Mises stress induced and end shortening for stringer panel without frame at different load cases. In order to design fuselage against the buckling failure the induced Von-Mises stress in the material should be lesser than the allowable stress. Hence failure load under buckling condition for stringer panel without frame is found to be 139KN.

Due to space constraints the von-misses stress plot and deflection plot are shown only for failure load case.

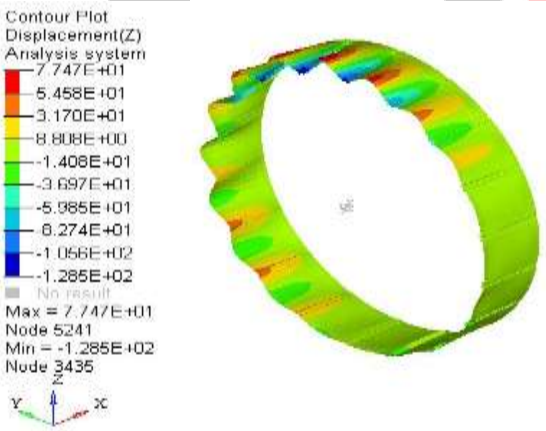


**Fig. 10 Deflection plot**

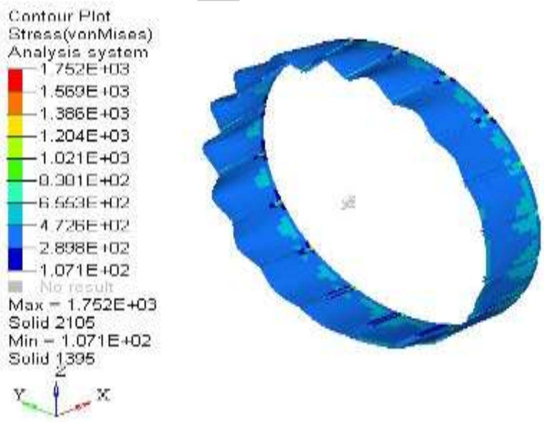


**Fig. 11 Stress plot**

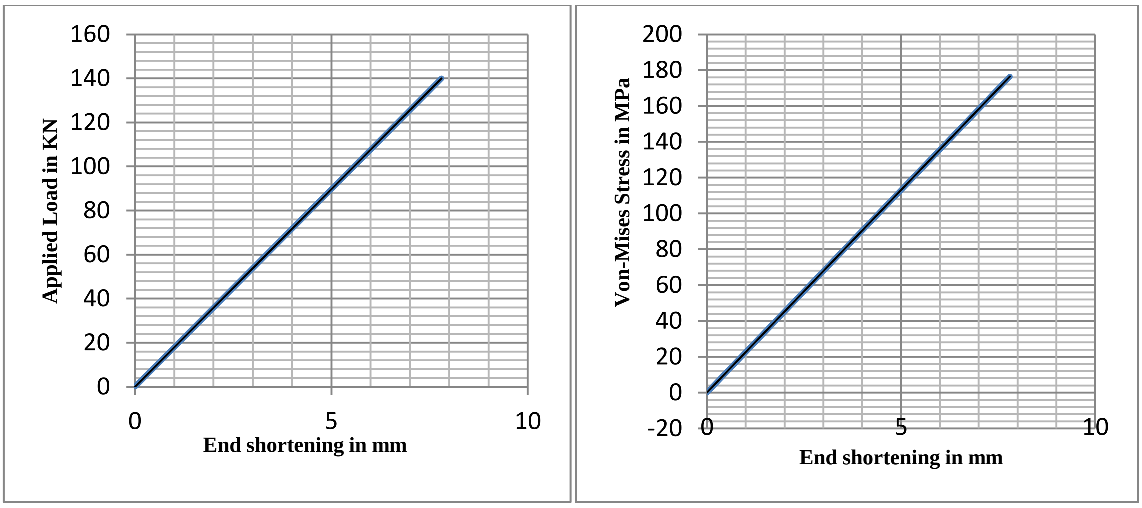


**Fig. 12 Graphs of Load versus End shortening and Stress versus End shortening**

The von misses stress versus end shortening and load versus end shortening graphs are drawn shown above. The end shortening or displacement due to buckling and also induced stress increases with increase in applied load.

*6.2. Results of stringer panel with frame*

For Aluminium alloy 2024-T351

We know that Yield strength = 350MPa

Factor of safety considered = 2

Hence, Allowable stress $= \frac{\text{yield strength of the material}}{\text{Factor of safety}}$

= 350/2

Allowable stress = 175MPa.

Table 3 Results of stringer panel with frame

| Applied Load in KN | End shortening in mm | Induced Von-Mises Stress in MPa |
|---|---|---|
| 0 | 0 | 0 |
| 50 | 0.146 | 14.98 |
| 100 | 0.293 | 29.96 |
| 150 | 0.440 | 44.93 |
| 200 | 0.587 | 59.91 |
| 250 | 0.733 | 74.89 |
| 300 | 0.880 | 89.87 |
| 350 | 1.027 | 104.8 |
| 400 | 1.174 | 119.8 |
| 450 | 1.321 | 134.8 |
| 500 | 1.468 | 149.8 |
| 550 | 1.614 | 164.8 |
| 555 | 1.629 | 166.3 |
| 565 | 1.658 | 169.3 |
| 575 | 1.688 | 172.3 |
| 580 | 1.702 | 173.7 |
| 585 | 1.717 | 175.2 |

The table 3 shows the Von-Mises stress induced and end shortening for stringer panel with frame at different load cases. In order to design fuselage against the buckling failure the induced Von-Mises stress in the material should be lesser than the allowable stress. Hence failure load under buckling condition for stringer panel with frame is found to be 585KN.
Due to space constraints the von-misses stress plot and deflection plot are shown only for failure load case.

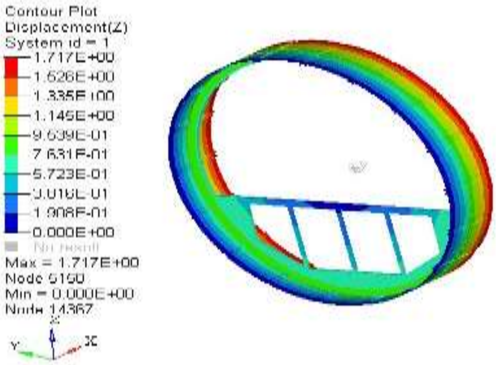


**Fig. 13 Deflection plot**

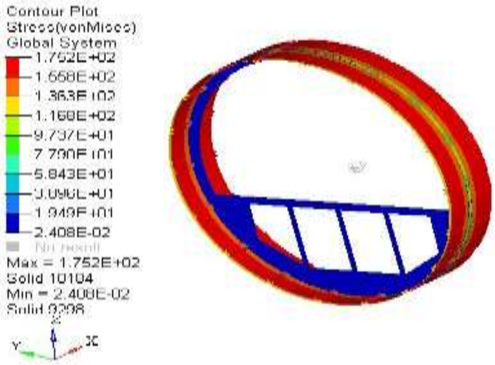


**Fig. 14 Stress plot**

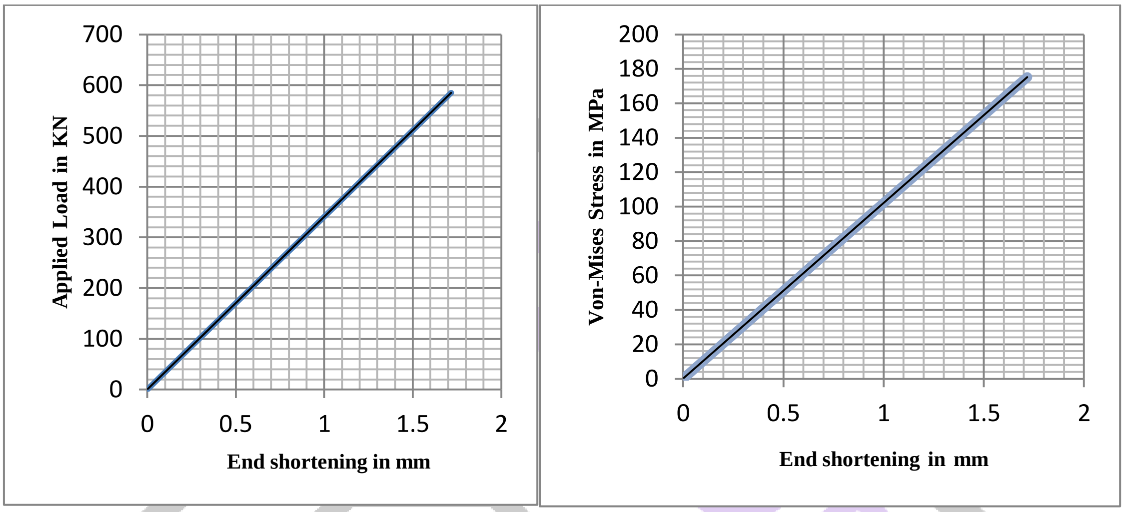


**Fig. 15 Graphs of Load versus End shortening and Stress versus End shortening**

The von misses stress versus end shortening and load versus end shortening graphs are drawn shown above. The end shortening or displacement due to buckling and also induced stress increases with increase in applied load.

## VII. CONCLUSIONS

From results of buckling analysis, stringer panel without frame model undergo failure at low load with maximum deflection due to buckling effect that is a buckling failure load of about 139KN and deflection of about 7.747mm, but stringer panel with frame model can able to sustain high load with minimum deflection before failure that is a buckling failure load of about 585KN and deflection of about 1.717mm, because of frame able to took buckling effect.

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