International Journal of Latest Engineering and Management Research (IJLEMR)
ISSN: 2455-4847
www.ijlemr.com || Volume 03 - Issue 06 || June 2018 || PP. 65-75

Comparison of Buckling Load-Carrying Capability of Metallic & Composite Panels Used in Wing Structure

Sumanth V Madhav[1], Vinod B[2] & Vinay DL[3]
[1] *Mtech Student, Department of Mechanical Engineering,*
Vidyavardhaka College of Engineering, Mysuru.
[2]*Asst Prof, Department of Mechanical Engineering,*
Vidyavardhaka College of Engineering, Mysuru.
[3]*Asst Prof, Department of Mechanical Engineering,*
Dayananda Sagar Academy of Technology & Management, Bangalore.

Abstract: Wings are aero-foils which produces lift as when moved rapidly through air. Wing structure in most modern aircrafts are aluminum structures. Due to its vital mechanical properties aluminum was used to build wing structures. However, due to advancement in technology composites are proving to be excellent and preferred materials for wing structure as it has better mechanical properties as compared to aluminum. In this study an attempt has been made to understand buckling load carrying capability of aluminum and GFRP composite panels used in wing structure by performing a comparative study which involves analytical and experimental investigations. Finally, based on results it is concluded that for the given geometry, material, loads and boundary conditions which among aluminum and GFRP composite panel has higher buckling load carrying capability.

Keywords: Composite, Buckling, Stiffened panel, Wing structure

1. Introduction

Aircraft wing structure can withstand nearly 82% of lift load in airframe structure of a transport aircraft. Their particular design for a given aircraft depends on a number of factors like size, weight, use of the aircraft, desired speed in flight, landing and desired rate of climb. Wings of a transport aircraft are full cantilever type design supported internally by structural members & covered by skin from outside. In wing design it is required to determine Factor of Safety (FOS) in order to ensure if the structure will withstand expected loadings, it is necessary to know failure mechanism which is a challenging task. Buckling is a one of the major reasons for failure of any aircraft and therefore the possibility of buckling failure should always be considered during wing design. In conventional method of air plane wing design, aluminum was always chosen material type due to its low density, high strength, superior malleability, easy machining, excellent corrosion resistance, good thermal and electrical conductivity. But with the advancement in material science technology, composites are becoming more popular as wing structure material due to ease of fabrication, high strength to weight ratio, fire resistance, chemical & weathering resistance, low thermal conductivity, color and so on. This comparative study provides an understanding on buckling load carrying capability of aluminum and composite panels used in wing structure and also opens wide scope for experimentation of composites for different compositions of fiber and matrix which may offer superior resistance against buckling failure.

2. Literature Survey

The team spent time to know about lift load determination on wing structure. Considerable amount of data was collected and analyzed in order to find out solution for existing problem. A brief explanation on data collected from various research journals are as follows:

1. In an airframe, wing structure carry majority of lift load during flight.
2. In order to determine lift load, common approach is to reconstruct an airplane wing by referring wing parameters of an existing aircraft.
3. CAD model is developed by referring dimensions of an existing airplane.
4. CFD analysis is performed to determine lift load on an airplane wing.
5. Modern aircrafts have a metallic or composite wing. Lift load determination is independent of wing material type.
6. Various stages in wing design includes wing identification, wing modelling and lift load calculation.
7. Methodology of wing design is suitable only for aircrafts operating at low Reynolds number. However, with the complexity of wing structure and pay load on airplane wing design approach may differ.
8. Angle of attack can significantly improve lift load value. But it should be noted that 14 degrees is the maximum critical angle beyond which airfoil will stall.

3. Buckling

In wing design it is required to determine Factor of Safety (FOS) in order to ensure if the structure will withstand expected loadings. Hence it is necessary to know failure mechanism which is a challenging task. Yielding is not the only mode in which any structure fails. During design of any structure it is necessary to ensure that structure do not deform too much. Buckling is one such concept which is not actually a failure, but it leads to failure of any structure. Buckling is a one of the major reasons for failure of any aircraft and therefore the possibility of buckling should always be considered during wing design.

3.1. Critical Buckling Load

Minimum axial compression load at which the column tends to have lateral displacement is called the *critical buckling load*.

Euler's formula for critical buckling load is given by,

$$P_{cr}=\frac{\pi^2 EI}{L_e^2}$$

Where,
P_{cr} = Critical load at which buckling occurs
E = Young's modulus
L_e= Effective length
I= Least moment of inertia

4. Methodology

1. Develop a modified wing model similar to wing specifications of an existing transport category aircraft identified for this study.
2. Identify critical buckling region on wing surface, perform global and local buckling analysis. Compare if approximate buckling factor values are obtained.
3. Fabricate metallic and composite panel for testing.
4. Perform an experimental investigation of metallic and composite panel in a Universal Testing Machine (UTM) to obtain critical buckling load.
5. Conclude based on experimental investigation which among metallic and composite panel has more buckling load-carrying capability.

5. Wing Details

Table 1: BN2B-26 Islander Wing specification

Wing Specification	Existing Wing	Modified Wing
Wing Span	14.940 m	*12.274 m
Fuselage Width	1.210 m	*1.210 m
Chord Length At Root	0.2030 m	*1.437 m
Chord Length At Tip	0.2030 m	*0.761 m
Max Speed of the Aircraft	77.77 m/s	77.80 m/s
Wing Area	30.20 m^2	*13.43 m^2
Dihedral & Sweep Back Angle	NO	NO
Aero foil At The Root	NACA 23012	NACA 23012
Aero foil At The Tip	NACA 23012	NACA 23012
Length Of Each Wing	6.865 m	*6.137 m
Mach Number	0.229	0.229
Density of Air	1.225 kg/m^3	1.225 kg/m^3
Reynolds Number	150000	150000
Maximum Wing Loading	99.2 kg/m^2	99.2 kg/m^2
Maximum Take-off Weight	2993 kg	2993 kg
Load Carried By Two Wings	2544 kg	2588 kg
Load Carried By Each Wing	1272 kg	*1294 kg

* indicates specifications changed in modified wing
Lift load on each wing of modified wing= 1294 kg

6. Material Properties

6.1. Aluminum 2024 T3

The 2000 series of aluminum alloys have copper as major constituent alloying element, and T3 term is associated with the heat treatment. These alloys are heat treatable with ultimate tensile strength varying from 27 to 62 ksi these are aluminum / copper alloys (with copper composition ranging 0.7 to 6.8%) and are high performance and strength alloys, which are often used for aerospace applications. Mechanical properties of aluminum is as shown in table 3 below.

Table 2: Properties of Aluminum 2024 T3

Mechanical Properties	**Values**
Elastic Modulus	70000 n/mm^2
Ultimate Tensile Strength	49.25 kg/mm^2
Yield Stress	34.5 kg/mm^2
Possion's Ratio	0.3
Shear Modulus	26924 n/mm^2
Density	3000 kg/m^3
Reference Temperature	25°C

6.2. Glass Fiber/Epoxy Composite

Glass fiber reinforced epoxy composites results in an effective combination of physical and mechanical properties which cannot be obtained with other materials. Glass fibers are easily available and processing technique is very economic for production of components. Epoxy resin is very commonly used in aerospace structures. Epoxy resin has many advantages like good adherence to metal and glass fibers, curing agents, and modifiers are available, absence of volatile matters during curing, low shrinkage during curing, excellent resistance to chemicals and solvents. Mechanical properties of GFRP/Epoxy Lamina is as shown in table 3 below.

Composite Laminate Details:
Fiber: Woven roving 200 GSM Glass fiber fabric (0.25mm thickness)
Matrix: Epoxy resin
Resin grade: L-12
Lay-up: [(90,0)/(90,0)/(90,0)/(90,0)/(90,0)/(90,0)]

Table 3: Properties of GFRP/Epoxy Lamina

Mechanical Properties	**Values**
Elastic modulus 11	27000 n/mm^2
Elastic modulus 22	2700 n/mm^2
Poisson's ratio 12	0.23
Shear modulus 12	2025 n/mm^2
Shear modulus 23	2160 n/mm^2
Shear modulus 13	2025 n/mm^2
Density	1900 kg/m^3
Thermal Expansion Coefficient 11	$11\times e^{-6} K^{-1}$
Thermal Expansion Coefficient 22	$1.96\times e^{-6}$
Reference Temperature	25°C

7. Finite Element Modeling and Analysis

Finite element modelling is done using hypermesh. Appropriate loads and boundary conditions & post-processing is performed in patran. Nastran is used as solver. To develop FE model, the 3D model of components of wing molded with CATIA is imported into hypermesh software. Mid-surface extraction and geometry correction are preliminary steps prior meshing. FEM model of wing structure is shown in figure 1.

Figure 1 : Finite Element Model of wing

8. Loads & Boundary Condition

1. Wing under lift load acts as cantilever beam bending about an axis. Hence at wing root all degrees of freedom (DOF) are constrained for each node.
2. Lift load during take-off Ptotal=1294kg for modified wing is applied as uniformly distributed load along length of each rib. Refer table 1 for lift load carried by each wing for modified wing design.

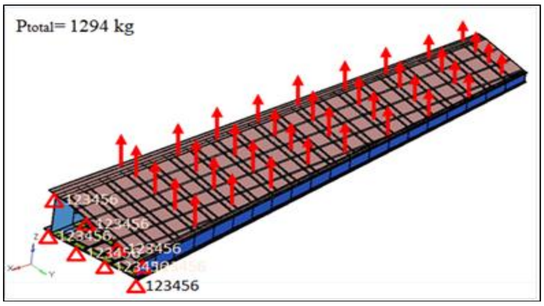

Figure 2: Loads and boundary conditions

9. Finite Element Analysis Results

9.1. Global Analysis

1. Finite element model of an entire structure or a subcomponent of a structure is known as global model.
2. In a global model, region requiring detailed interrogation will be identified. Global response of aircraft wing is obtained with coarse mesh in finite element analysis.
3. Computational time required for Global Analysis is higher.
4. Applied displacement field along the boundary is obtained from global solution, with this local response and critical region can be observed.
5. Results obtained from global analysis will provide an understanding on response of entire structure for the applied loads and boundary conditions.
6. Approximate results in terms of stress, strain, deformation, displacement, Eigen values etc is obtained in comparison with theoretical values.

9.1.1. Linear Static Analysis

a. Resultant stress

Stress developed during local linear static analysis is shown below. The Maximum Stress of 7.59 kg/mm^2 developed at wing root for given loads & boundary conditions. The stress is below yield point 34.5 kg/mm^2, so the modification in wing design is safe. Red color indicates the maximum stress and blue color indicates the minimum stress.

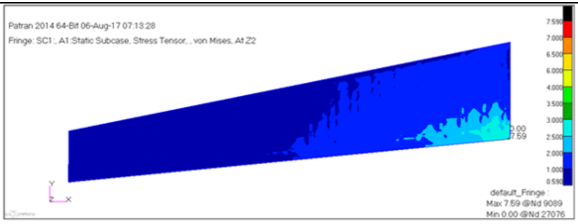

Figure 3: Von-misses stress for linear static analysis

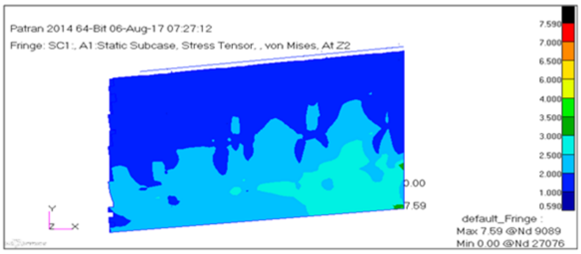

Figure 4: Critical stress region at wing root

b. Total Deformation

Total deformation produced in wing during linear static analysis is shown in figure below. The maximum deformation produced is 20.8 mm for the given load and boundary conditions. In spectrum red color indicates maximum deformation while blue color indicates minimum deformation.

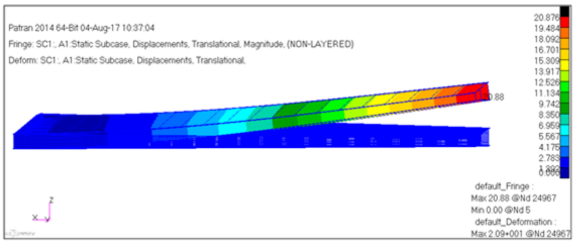

Figure 5: Total deformation of wing for linear static analysis

9.1.2. Buckling Analysis

Buckling analysis is performed to determine buckling factor for wing. In buckling due to axial compression load a structure exhibits sudden, large, lateral deflection due to small increment in applied compressive load. Buckling factor indicates ratio of buckling load to applied load. Buckling load factor obtained for wing is 1.69 for an applied load of 1294kg. Now for buckling to occur, 1.69 x 1294 =2187 kg is the load required and it's called the critical buckling load. Since buckling load factor obtained is greater than 1, buckling will not occur for the applied load.

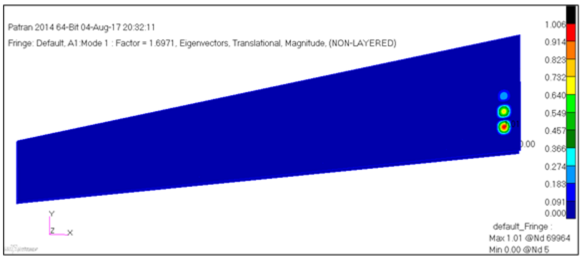

Figure 6: Buckling factor for wing

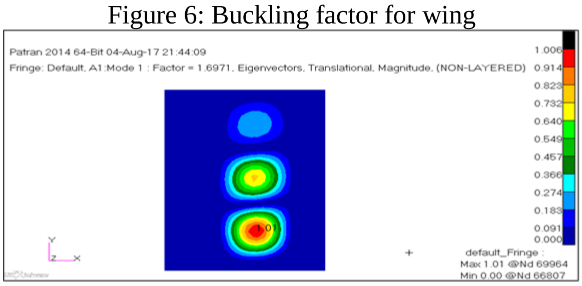

Figure 7: Critical buckling region

9.2. Local Analysis

1. The local finite element model is a specific region in wing structure which has maximum stress obtained during global analysis.
2. Any structural sub-region within the defined global model is called local model. Local analysis has a refined mesh size & computational time required for local analysis in comparison to global is minimum.
3. In Local Analysis specific region of interest such as cut-out in a panel, local buckled region of a curved panel will be modeled with fine mesh.
4. The local model is independent of the global finite element model. The local model accurately represents the geometry of the structure necessary to provide the local behavior and stress state.
5. Results in terms of stress, strain, deformation, displacement, Eigen values and so on will have close convergence with theoretical solution.

9.2.1. Linear Static Analysis

a. Resultant stress

In global analysis maximum stress was observed in the region of intersection of trailing edge and wing root on top surface of skin. A local analysis was performed for this region where maximum stress observed was 7.084 kg/mm^2 for the given loads & boundary conditions. In linear static analysis, stress will be proportional to strain within elastic limit. The maximum von-misses stress is developed at fastener location between top skin and stringer as shown below. The stress is below yield point 34.5 kg/mm^2 for Al2024.

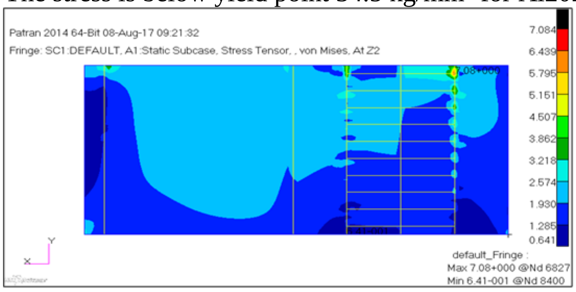

Figure 8: Von-misses stress for linear static analysis

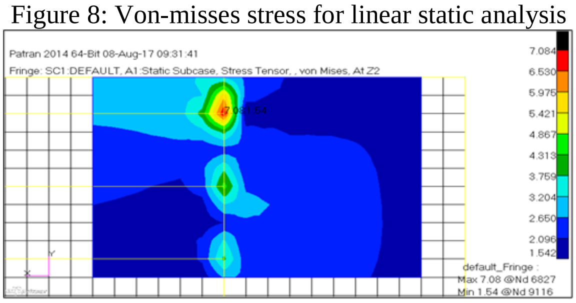

Figure 9: Critical stress region for local analysis

b. Total Deformation

Total deformation produced in wing for linear static analysis is shown below. The maximum deformation produced is 0.110 mm for the given load and boundary conditions. In spectrum red color indicates maximum deformation while blue color indicates minimum.

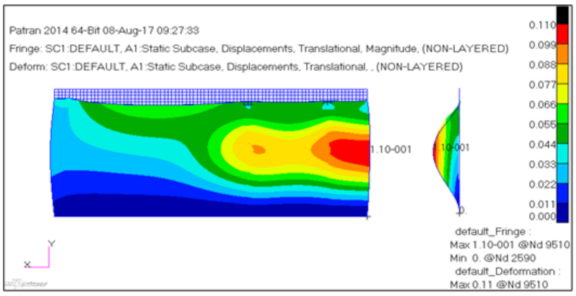

Figure 10: Total deformation in local analysis

9.2.2. Buckling Analysis

Buckling analysis is performed to determine buckling factor for local analysis. In buckling due to axial compression load a structure exhibits sudden, large, lateral deflection due to small increment in applied compressive load. Buckling factor indicates ratio of buckling load to applied load. Buckling load factor obtained for local analysis is 1.71 for an applied load of 2250 kg (430 mm × 5.23 kg/mm^2 = 2250 kg). Now we can say that buckling will occur at 1.71 x 2250= 3850 kg and it's called the critical buckling load. Buckling load factor multiplied by applied load gives buckling load. Since buckling load factor obtained is greater than 1, wing will not buckle for the applied load.

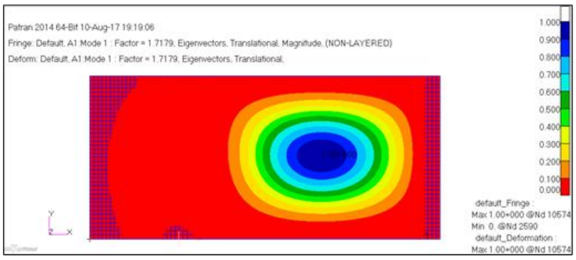

Figure 11: Buckling factor for wing

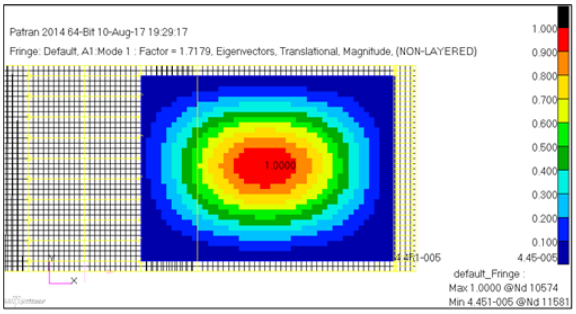

Figure 12: Critical buckling region

9.3. FEM Results

Results for linear static and buckling analysis from global and local finite element models shows a close convergence between resultant stress and buckling factor as shown in table below. Hence modified wing will not fail due to buckling

Table 13: Global and local FEM analysis

Results	Linear Static Analysis		Buckling Analysis
	Resultant stress in kg/mm^2	Total Deformation in mm	Buckling Factor
Global Analysis	7.59	20.8	1.69
Local Analysis	7.084	0.110	1.71

10. Composite and Metal Panel Fabrication

10.1. Composite panel fabrication

Epoxy and glass fiber used as matrix and reinforcement material while Tri Ethylene Tetra Amine (TETA) as hardener. Glass fiber is woven material which is bi-directional of 200 gsm or 0.25mm thickness with density 2.5 gm/cc with lay up of $[(90,0)/(90,0)/(90,0)]_S$. Composite panel is developed using vacuum bag process with glass fiber and epoxy as base materials.

Figure 14: GFRP fabric mats

Figure 15: Epoxy and Hardener

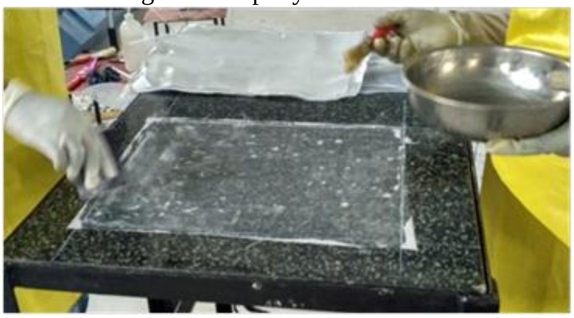

Figure 16: Peel ply on mold surface

Figure 17: Vacuum bag process

10.2. Aluminum Panel Fabrication

Aluminum panel fabrication is simple & easier in comparison with glass fiber composite due to availability of material with required thickness in market. Al 2024 T3 sheet of 2 mm thickness and 300 mm × 300 mm dimension is cut from an aluminum sheet of standard size.

11. Specimen Testing

1. Strain gauge is attached to panel surface with suitable adhesive
2. Location of strain gauge on panel depends on region where maximum deflection occurs due to applied compression load.

3. Between two manually adjustable grips of a 50 KN in UTM machine, specimen is loaded.
4. Test specimens are positioned in fixtures in such a way that no eccentricity occurs when compression load is applied.
5. Aluminum and GFRP composite panels are individually tested in Universal Testing Machine (UTM)
6. Axial compression load is applied on panels till critical buckling load ($P_{critical}$) is attained.
7. As the panel is deformed foil is deformed causing its electrical resistance to change.
8. This resistance change is usually measured using a wheat-stone bridge by quantity know as strain factor.
9. Intermittent change in strain value of specimen due to applied load is measured with a data acquisition system.
10. Strain value is recorded at different load intervals. Test is repeated for same load intervals and an average was taken to calculate final critical buckling load.
11. A graph is plotted for bending strain and slope against applied load and critical buckling load is located.

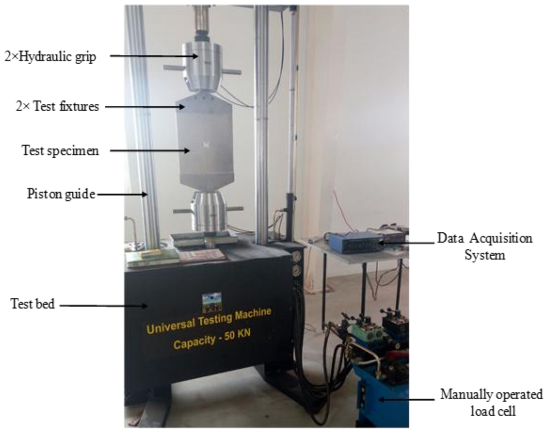

Figure 18: Universal Testing Machine

Critical buckling load (P_{cr}) is determined for both aluminum and GFRP composite panels and a close convergence is found between analytical solution and test conducted on Universal Testing Machine (UTM).

Euler's formula for critical buckling load is given by,

$$P_{cr}=\frac{\pi^2 EI}{L_e^2}$$

Where,
P_{cr} = Critical load at which buckling occurs
E = Young's modulus
L_e= Effective length
I= Least moment of inertia
Boundary Condition: Simply supported column L_e=L

Table 4: Property table of Aluminum 2024 T3 & GFRP/Epoxy Panel

Properties	**Aluminum 2024 T3**	**GFRP/Epoxy Laminate**
Length	300 mm	300 mm
Thickness	2 mm	2 mm
Width	300 mm	300 mm
Young's Modulus (E)	6835.49 kg/mm^2	2700 kg/mm^2
Least moment of Inertia (I)	200 mm^4	106.2882 mm^4
Effective Length (L_e)	300 mm	300 mm

11.1. Critical buckling load (P_{cr}) for aluminum panel

a. Analytical solution

$$P_{cr} = \frac{3.14^2 \times 6835.49 \times 200}{300^2} = 149.96 \text{ kg}$$

b. Test result

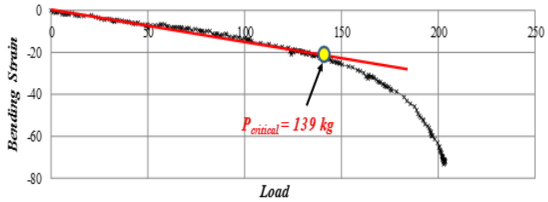

Figure 19: Bending strain vs load curve

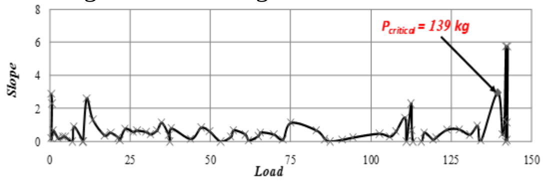

Figure 20: Slope vs load curve

11.2. Critical buckling load (P_{cr}) for GFRP/Epoxy panel

a. Analytical solution

$$P_{cr} = \frac{3.14^2 \times 2700 \times 106.288}{300^2} = 31.48 \text{ kg}$$

b. Test result

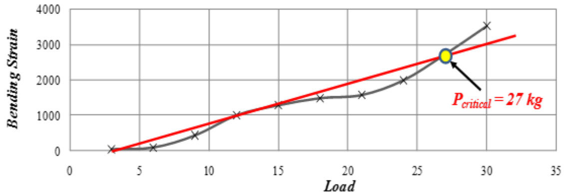

Figure 21: Bending strain vs load curve

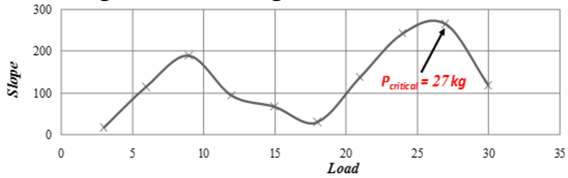

Figure 22: Slope vs load curve

Table 5 : Critical buckling load table

Test specimen	Analytical result	Experimental result
Aluminum Panel	149.96 kg	139 kg
GFRP/Epoxy Panel	31.48 kg	27 kg

12. Conclusion

The results of this study shows stress and buckling factor values determined for global and local finite element models in modified wing structure were within permissible limits. Fabrication and testing of composite and aluminum panels were also performed to understand buckling load carrying capability. Through above analysis for wing structure, following conclusions were drawn.

1. During wing design buckling of upper skin panels of wing is a major factor to be considered.
2. Buckling is an important criterion in wing design and hence buckling factor should be more than one in order to prevent failure of wing due to buckling.
3. Linear static analysis shows maximum stress on wing to be 7.59 kg/mm^2 which is less than the yield strength of aluminum. Hence it's a safe design.
4. Criteria for buckling factor in global and local analysis are in good agreement. It can also be noted that buckling does not occur as buckling factor is greater than one for modified wing design.
5. From analytical and experimental investigations of aluminum and GFRP/Epoxy composite panels, it is clear that for the given geometry, material, loads and boundary conditions aluminum has higher buckling load carrying capability when compared to composite.

Consolidating all the above observations it could be conclude that aluminum has good buckling load carrying capability when compared to GFRP/Epoxy composite for the given geometry. However, by changing ply orientation or by enhancing material properties, buckling load carrying capability of composites can be improved.

13. Future Scope of Work

1. Composite panels can be fabricated with laminate consisting of different ply orientation of GFRP/Epoxy plies followed by testing.
2. Laminate can be fabricated with various combinations of fiber and matrix materials which may yield better results for buckling when compared to aluminum.

14. References

[1]. Abin paul, Subin p George, "Buckling analysis of wing upper skin panels of a transport Aircraft", International Journal of Science, Engineering and Technology Research (IJSETR), Volume 3, Issue 10, October 2014

[2]. Amandeep Sing Sivi, "A Study on The Skin-Stringer Panel of an Aircraft", International Journal of Aerospace and Mechanical Engineering, ISSN (O): 2393-8609, Volume 1 – No.1, September 2014

[3]. K.Mallikarjuna Reddy, B.Sidda Reddy, R.Madhu Kumar," Buckling Analysis of Laminated Composite Plates Using Finite Element Method", INTERNATIONAL JOURNAL OF ENGINEERING SCIENCES & RESEARCH TECHNOLOGY, ISSN: 2277-9655, November, 2013

[4]. F.M. Mazzolani, A. Mandara, G. Di Lauro, "BUCKLING OF ALUMINIUM SHELLS: PROPOSAL FOR EUROPEAN CURVES", Department of Structural Analysis and Design, University of Naples Federico II Naples, Italy, May 2010

[5]. Boris Jerman, "The Local Buckling of the Thin Walled Aluminum Mast", Faculty of Mechanical Engineering, Belgrade, March 2010

[6]. K.Kantha Rao, K. Jayathirtha Rao, A.G.Sarwade3, M.Sarath Chandra, "Strength Analysis on Honeycomb Sandwich Panels of different Materials", International Journal of Engineering Research and Applications (IJERA), ISSN: 2248-9622, Vol. 2, Issue 3, May-Jun 2012, pp. 365-374

[7]. Viviana Meruane, Valentina del Fierro, "IDENTIFICATION OF SKIN/CORE DEBONDING IN HONEYCOMB ALUMINIUM PANELS THROUGH AN INVERSE METHOD BASED ON PARALLEL GENETIC ALGORITHMS", The 21st International Congress on Sound and Vibration, 13-17 July, 2014

[8]. Masaki Hojo, Yasuo Hirose, Takayuki Kusaka, Masaaki Nishikawa, "Recent Advances in Carbon Fiber Reinforced Plastics", Third International Conference on Sustainable Construction Materials and Technologies, 11 March, 2012

[9]. T.John babu, M.Khaja Gulam Hussain, "Characterization of sandwich composites reinforced with glass fibres and Polyurethene Foam", International Journal of Scientific Development and Research (IJSDR), ISSN: 2455-2631, August 2016