Advantages of Composite Materials in Aircraft Structures

Muniyasamy Kalanchiam, and Moorthy Chinnasamy

Abstract**—**In the competitive environment of aircraft industries it becomes absolutely necessary to improve the efficiency, performance of the aircrafts to reduce the development and operating costs considerably, in order to capitalize the market. An important contribution to improve the efficiency and performance can be achieved by decreasing the aircraft weight through considerable usage of composite materials in primary aircraft structures. In this study, a type of composite material called Carbon Fiber Reinforced Plastic (CFRP) is explored for the usage is aircraft skin panels. Even though there were plenty of studies and research has been already carried out, here a practical example of an aircraft skin panel is taken and substantiated the benefits of composites material usage over the metallic skin panel. A crown skin panel of a commercial aircraft is designed using both metal and composite materials. Stress analysis has been carried out for both and margin of safety is estimated for the critical load cases. The skin panels are compared for manufacturing, tooling, assembly and cost parameters. Detail step by step comparison between metal and composite constructions are studied and results are tabulated for better understanding.

Keywords**—**Composites, CFRP, Aircraft Structure, Skin panel.

I. INTRODUCTION

FUSELAGE of a commercial aircraft is usually semi monocoque in construction consists of various structural members such as Skin, Stringers, Frames, bulkheads, Intercostals etc. all fastened together by rivets and bolts. Each structural member in fuselage is designed to perform specific functions. The performance of aircraft mainly depends on the total weight. The design of a fuselage for a commercial transport is impacted by the interaction of its functional requirements and its basic strength, stiffness, and life requirements. New and innovative designs must be explored to accommodate these requirements and to meet the goals of lower weight and more cost effective structure for future airplanes. Even though there plenty of metallic alloy materials are present to choose for design, there are limitations for doing smart design using metals. So, continuous research is going on for replacement of metallic materials with other better performing materials. As a result, composites are considered to be a superior choice to replace metallic structures in order to attain better strength to weight ratio finally resulting in increased performance of aircraft. In this paper, a crown skin panel of a commercial aircraft is designed using both metallic and composite materials to study the feasibility of composite application for an airframe structures. Skin panels are designed in metal as well as composite by considering various functional, strength / stiffness, manufacturability, tooling, assembly and cost requirements. Stress analysis has been carried out for both metallic and composite structures and the margin of safety is evaluated. The obtained results for various parameters are tabulated for comparison.

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II. METALLIC AIRFRAME STRUCTURE-OVERVIEW

In the present scenario of aircraft design, metallic alloys are exploited to use in various load carrying primary structures like Fuselage, Empennage, wing, and etc. Use of light materials in aircrafts has been always considered as an important factor of measure to increase performance. Airframe design demands strong, stiff materials at an acceptable weight and cost. Aluminum, Steel and titanium alloys are predominantly used as principle materials in primary airframe structures. Various researches have been done in improving structural properties of these metals to improve strength to weight ratio.

A. Design of Metallic Skin Panel

A crown skin panel assembly is designed based on existing design data from a commercial aircraft. The assembly consists of Skin, Doubler, Stringers and frame. The skin of the fuselage is stiffened by frames, bulkheads, stringers and longerons. Stringers / longerons carry the major portion of fuselage bending moment by taking axial loads. Fuselage skin carries the shear from the applied external transverse and torsional forces, and cabin pressure. Frames primarily serve to maintain the shape of the fuselage and to reduce the column length of the stringers to prevent general instability of the structure. Frame loads are generally small and often tend to balance each other and as a result, frames are generally of light construction. Frames and stringers are positioned at equal intervals along skin and appropriate thicknesses are calculated to do the initial models. Computer aided design software CATIA V5 is used to model the structure.

2024-T3 material is used because of its high tension strength, fracture toughness and slow crack growth. Roll formed stringers with hat cross section with .05” thick is designed. In total 14 stringers are placed at equal interval of 6.6º. Stringers are attached to skin by countersunk rivets.

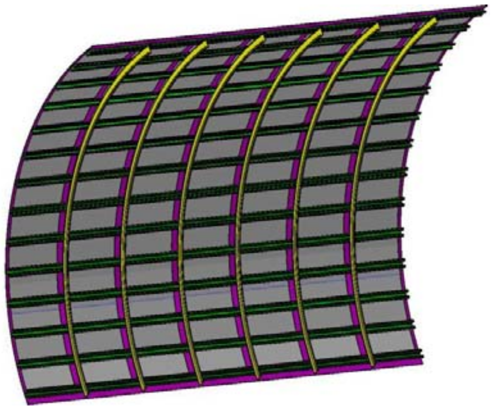

Fig. 1 CAD Model of Metallic skin panel structure

Frames maintain the shape of the fuselage and placed at equal intervals. The Frame main functions of frames are, it acts as panel breaker for skin, distribute concentrated loads, supports fuselage from compression and shear loads. 7075-T6 material is selected because of high strength, low fracture toughness. Roll formed Z-section frames is considered for design. Total of 7 frames are placed at 20” intervals. Stringer and frames fastened together by two rivets at each station.

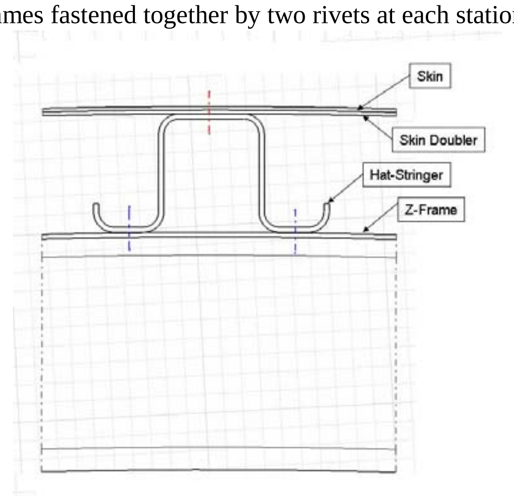

Fig. 2 Section view of metallic skin panel attachments

Skin is the outer cover for fuselage. The main function is of skin is to carry in plane shear loads. The crown skin of radius is 73.96”. Waffle doubler is fabricated separately and bonded together by field fasteners to have a fail safe design. Total length of skin panel is 137.8”.

B. Stress Analysis of Metallic skin panel

Fuselage is the structural backbone of an aircraft that balances the internal and external loads acting upon the aircraft. These loads consist of internal mass inertia forces due to equipment, payload, stores, fuel, flight forces due to propulsion thrust, lift, drag, maneuver, wind gusts, ground forces due to taxi, landing, and decompression loads. The strength capability of the airframe must be predictable to ensure that these applied loads can be withstood with an adequate margin of safety throughout the life of the airplane. In addition to strength, the airframe requires structural stiffness to prevent excessive deformation under load and to provide a satisfactory natural frequency of the structure.

The overall airframe structure is made up of a number of separate components, each of which performs discrete individual functions. The structure consists of primarily frames, stringer and skin. Performance requirements (range, payload, speed, altitude, landing and takeoff distance, and so forth) dictate that the airframe be designed and constructed so as to minimize its weight. All the airframe material must be arranged and sized so that it is utilized as near its capacity as possible, and so that the paths between applied loads and their reactions are as direct and as short as possible.

A fuselage sector of 45 deg has been considered in this study. Even though, there are so many loads that act of this structure, Shear load due to gravity and the Decompression loads are more critical loads to be considered.

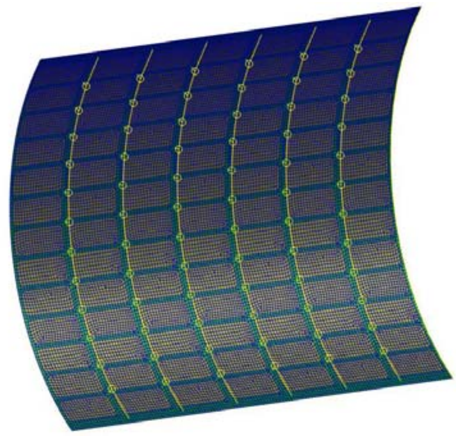

Fig. 3 Finite Element Model of Metallic skin panel structure

Finite Element Analysis has been employed to determine the Margin of safety of this structure under the critical load cases. Bar elements and shell elements have been used to construct the FE model. The number of nodes is 24400 and the number of elements is 22700. Material properties have been applied to the different components like frame, stringer and skin.

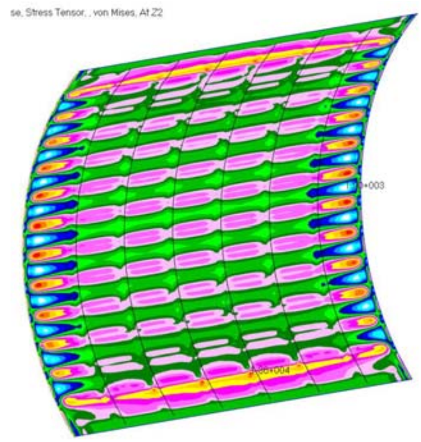

Fig. 4 Stress plot of metallic skin panel under decompression load

Then appropriate boundary conditions have been applied on the model. The above said loads are applied in the model as two load cases.

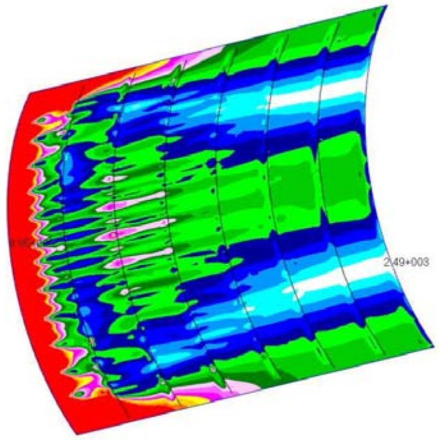

Fig. 5 Stress plot of metallic skin panel under shear load

The results are post processed and the margin of safety is calculated based on the material allowable

C. Assembly and Tooling

Tooling for the skin panel is done at both in the stage of manufacturing and assembly. Based on the manufacturing methods selected to fabricate the frames and stringers, tooling provisions are made. For the assembly purposes jigs are designed to assemble the skin with stringers, and assemble the frame with stringers. Assembly sequence is established to put together all the parts. High quality tolerances are maintained to establish the better structural integrity. Suitable provisions like tooling hole, coordination holes and determinant assembly holes are provided while design to facilitate the manufacturing and assembly. Farming tool is used to bend the rolled frame Z section frames to shape, skins are formed to shape by break forming or stretch forming methods and stringers are bend to suit skin contour. Skin, frame and stringers are attached together by rivets and hi-loks.

D. Manufacturing and Costing

Since the skin chosen is a constant section with out any variations in skin thickness of .036” the manufacturing of this metallic skin is done by stretch forming process. Waffle doubler is machined with cut-out and formed to shape by stretch forming method. Waffle doubler and skin are attached together by bonding method and by field fasteners. This field fasteners helps in keeping the waffle doubler and skin intact under various load conditions. Skins and stingers are fastened together though rivets, pitch of around 4D to 6D is considered to rivet the skin with stringers.

Cost for the conventional method for manufacturing and assembly of skin panels are pretty much standardized. Based on the number of panels needs to be produced the procurement, manufacturing and assembly cost varies. Highly experienced man power is available in market to employ and proven method manufacturing / assembly makes metallic skin panel creeper.

III. Composite airframe structure-Overview

The design of a composite fuselage must provide the necessary strength and rigidity to sustain the loads and environment that it will be subjected during the operational life of the aircraft. The many structural considerations must adhere to the requirements defined in the Federal Aviation Regulation, Part 25 in order to achieve the objectives of 1) unlimited life in operational service and 2) fail-safe characteristics for all reasonable extent of damage. The advisory circulars also sets forth guidance information relating to acceptable means of compliance with the provisions of FAR 25 dealing with composite structures and with damage tolerance and fatigue evaluation certification requirements.

These many requirements impose severe constraints on the design of the fuselage structure. The major structural considerations are presented to indicate the general policy and type of data required to establish criteria for composite fuselage structure design.

A. Design of Composite Skin Panel

Similar to metallic, a crown panel skin is designed with stringers, frames and shear ties. Location of these components is placed in equal intervals.

Skin with integral stringers is designed with thickness of 0.06 inch by considering the feasibility of tooling the inverted hat section configuration is considered for stringer cross sections. Necessary draft angle is considered to easily remove the skin and stringer assembly from tool after curing. Since the inverted cross section stringers are designed, it becomes difficult to attach the frames directly to stringers as in metallic skin panel. So a different configuration of continuous shear tie is designed for attaching skin to frame. It becomes easy to rivet and bond the skin with shear tie in this configuration

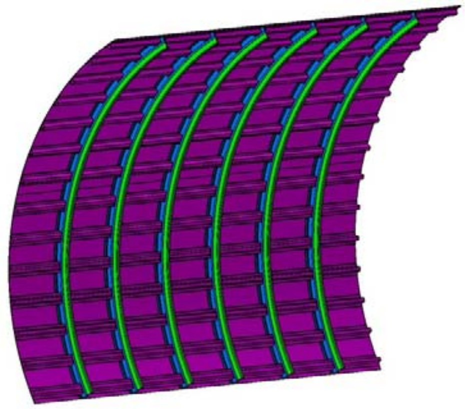

Fig. 6 CAD Model of Composite skin panel

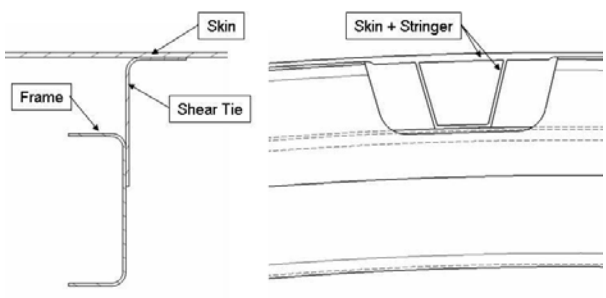

Fig. 7 Section view of composite skin panel attachments

The waffle doubler is integrated to skin design to do the same function; by combining the design of stringer, doubler and skin together the part counts are considerably reduced resulting in fewer numbers of fasteners. Separate "C" section frames are manufactured and attached to stringer with the rivets. "C" section frames and "L" section shear ties results in "Z" cross section resulting in better stress properties.

B. Stress Analysis for Composite Skin Panel

The finite element model has been created for the composite fuselage sector. The methodology is same as the metallic structure. The main difference is material properties and material orientations.

Carbon Fiber Reinforced Plastic (CFRP) is chosen to use for the structure. Carbon fiber reinforced polymer (CFRP), is a very strong, light, and expensive composite material or fiber reinforced polymer.

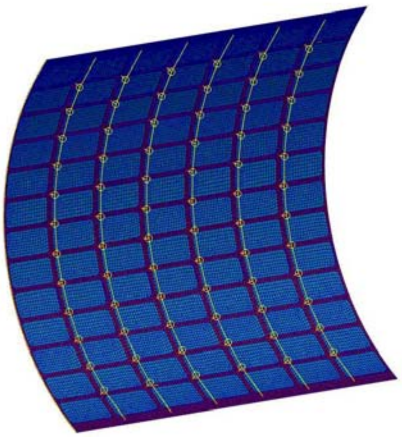

Fig. 8 Finite Element Model of Composite skin panel structure

Similar to fiberglass (glass reinforced polymer), the composite material is commonly referred to by the name of its reinforcing fibers (carbon fiber). The polymer is most often epoxy, but other polymers, such as polyester, vinyl ester or nylon, are also sometimes used.

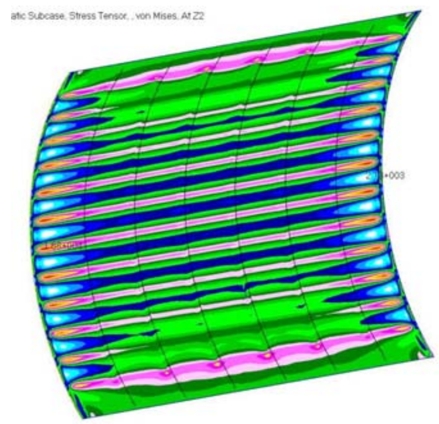

Fig. 9 Stress plot of composite skin panel under decompression load

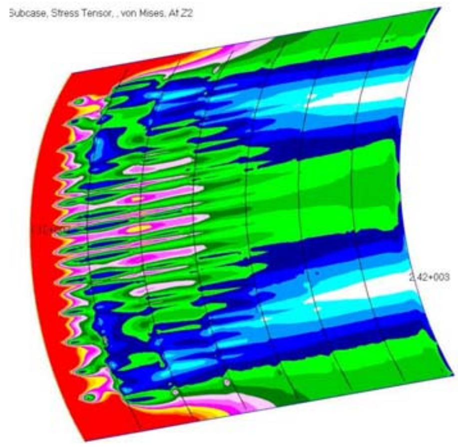

Fig. 10 Stress plot of composite skin panel under shear load

Some composites contain both carbon fiber and other fibers such as kevlar, aluminum and fiberglass reinforcement. It has many applications in aerospace and automotive fields, as well as in sailboats, and notably in modern bicycles and motorcycles, where its high strength to weight ratio is of importance. In the finite element model, the properties of CFRP are applied and the appropriate material orientations are assigned. The load cases are same as in the metallic structure. The margin of safety is estimated by using material allowable values.

C. Assembly and Tooling

The design of the structural elements will reflect on the tooling concepts employed. The tooling must facilitate locating and supporting the prepreg stiffener during the cure cycle. Tooling for frame assemblies should be designed to eliminate contour machining requirements. Elastomeric mandrels can be employed if the design requires the application of transverse pressure on formed parts such as the flange section of frames.

The design of the fuselage components/assemblies should be directed to eliminate and/or minimize the major handwork labor. The utilization of automatic production machines such as: (1) roll-forming machines to form prepreg stiffeners, (2) numerically controlled (N/C) tape laying machines for skins, and (3) N/C water jet and/or Gerber cutter machine to cut out frame patterns from preplied tape and / or cloth material will reduce fabrication costs.

D. Manufacturing and Costing

The design of cost-competitive hardware requires the integration of key manufacturing considerations in the design process. The large components, complex tooling and equipment requirements associated with the manufacture of a producible fuselage structure will have significant cost impacts. Manufacturing considerations must also include quality assurance considerations to ensure the integrity of the fabricated hardware. Manufacturing of composite components are configuration sensitive and must be performed in conjunction with the structural design effort. Composite skin panel was designed by considering the cost factor and made as simple as possible for manufacturing.

IV. Comparison between Metallic and Composite Airframe Structures

The various parameters related to metallic and composite skin panel structures are consolidated and tabulated below:

TABLE I
COMPARISON TABLE OF METALLIC AND COMPOSITE SKIN PANELS

Component Type	Material	Weight in lbs	Cross Section/ Length	Margin of Safety
Metallic Skin Panel				
Skin	2024-T3	56.44		0.92
Stringers	2024-T3	51.43	Hat-Section	
Frames	7075-T6	13.91	Z-Section	
Doubler	2024-T3	17.57		
Assy		139.35		
Composite Skin Panel				
Skin + Stringer	CFRP	107.63		0.78
Frame	CFRP	8.31	C-Section	
Shear Tie	CFRP	4.36	L-Section	
Assy		120.30		

V. Conclusion

Composite structures must be shown to have the crashworthiness capability equivalent to those of conventional aluminum structure: To attain this equivalency, a design data base must be established by conducting both analytical and experimental investigations, exploring the structural response and integrity of composite structure subjected to simulated crash events. Based on above study, it is concluded that metallic and composite skin panel structures have their own advantages in various factors. Metallic structure is superior to composite in strength and cost aspects. Composite structure is superior to metallic structure in weight aspect which will influence in airplane performance. However there are a number of technical issues and potential problems areas which must be resolved before sufficient confidence is established to commit composite materials for application to pressurized fuselage structures.

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