ISSN 2349-6193
Impact Factor (PIF): 2.243

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International Journal OF Engineering Sciences & Management Research

STRUCTURAL MODELING AND ANALYSIS OF COMPOSITE WING RIB USING FINITE ELEMENT METHOD

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Keywords: Finite Element Method, Wing-Rib, Composites.

ABSTRACT

In recent years, Composite materials have received considerable attention as alternatives to steel and aluminum as structural materials in the construction, automotive, marine and aerospace industries due to a variety of reasons; these include a high strength-to-weight ratio, a high stiffness-to-weight ratio, corrosion and fatigue resistance, ease of handling, and ease of fabrication. Composite materials have been employed due to their life-cycle cost competitiveness. This research paper deals with the design and analysis of aircraft structural wing-rib using composite materials. The optimum design parameters for an aircraft structural wing-rib are suitably selected based on the classical approach. The three dimensional structural wing-rib is designed based on the design parameters using Computer Aided Design (CAD) software. The procedure of Finite Element Analysis and the detailed description about various Computer Aided Engineering (CAE) tools have been studied and implemented in this work. Designed three dimensional structural wing-ribs are exported to the CAE tool and Finite Element Modeling is prepared based on the design parameters. Composite Material properties and boundary conditions are executed with suitable conditions in CAE tool. Analysis is carried out for structural wing-ribs based on the various loading conditions and various fiber orientations of composite materials. A complete set of finite element analysis were conducted on different fiber oriented composite systems. Critical displacement and Stress tensor were obtained from Finite Element Tool. The results are compared based on the fiber orientation.

INTRODUCTION

Now a day, composite materials are used in large volume in various engineering structures including spacecrafts, airplanes, automobiles, boats, sports' equipment, bridges and buildings. Widespread use of composite materials in industry is due to the good characteristics of its strength to density and hardness to density. The possibility of increase in these characteristics using the latest technology and various manufacturing methods has raised application range of these materials. Application of composite materials was generally begun only at aerospace industry in 1970s, but nowadays after only three decades, it is developed in most industries. Meanwhile, the automotive industry considered as a mother one in each country, has benefited from abilities and characteristics of these advanced materials. Along with progress in technology, metallic automotive parts are replaced by composite ones.

MATERIALS

The term composite could mean almost anything if taken at face value, since all materials are composed of dissimilar subunits if examined at close enough detail. But in modern materials engineering, the term usually refers to a “matrix” material that is reinforced with fibres. For instance, the term “FRP" (Fiber Reinforced Plastic) usually indicates a thermosetting polyester matrix containing glass fibres, and this particular composite has the lion's share of today's commercial market.

Many composites used today are at the leading edge of materials technology, with performance and costs appropriate to ultra-demanding applications such as spacecraft. But heterogeneous materials combining the best aspects of dissimilar constituents have been used by nature for millions of years. Ancient society, imitating nature, used this approach as well: the Book of Exodus speaks of using straw to reinforce mud in brick making, without which the bricks would have almost no strength.

The fibers used in modern composites have strengths and stiffness’s far above those of traditional bulk materials. The high strengths of the glass fibers are due to processing that avoids the internal or surface flaws which normally weaken glass, and the strength and stiffness of the polymeric aramid fiber is a consequence of the nearly perfect alignment of the molecular chains with the fiber axis.

 ISSN 2349-6193
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Of course, these materials are not generally usable as fibers alone, and typically they are impregnated by a matrix material that acts to transfer loads to the fibers, and also to protect the fibers from abrasion and environmental attack. The matrix dilutes the properties to some degree, but even so very high specific (weight-adjusted) properties are available from these materials. Metal and glass are available as matrix materials, but these are currently very expensive and largely restricted to R&D laboratories. Polymers are much more commonly used, with unsaturated styrene-hardened polyesters having the majority of low-to-medium performance applications and epoxy or more sophisticated thermosets having the higher end of the market. Thermoplastic matrix composites are increasingly attractive materials, with processing difficulties being perhaps their principal limitation.

FIBRES

The primary function of the reinforcement in composites reinforced with continuous fibres is to provide strength and stiffness and to support the structural load. The purpose of the matrix is to provide shape and form, to protect the fibres from structural damage and adverse chemical attack, to distribute stress, and to provide toughness. The matrix also stabilizes the composite against buckling in compressive loading situations. Fibres, also known as filaments, have finite lengths of at least 100 times their diameter, and are prepared by drawing from a molten bath, by spinning, or by chemical vapour deposition on a substrate such as tungsten or carbon. They are grouped into bundles or strands of 500 to 12 000 filaments. The bundles or strands may be chopped into short fibres, or twisted into yarns suitable for the weaving of fabrics or three dimensional performs, using a variety of weaving patterns. The strands may be further combined to form tows, with as many as 40 000 to 300 000 filaments each. Woven fabrics and tows can be processed into chopped fabric squares or chopped fibre tows. In the plain-weave pattern, yarns are interlaced in alternating fashion over and under every other yarn to provide maximum fabric stability and firmness, and minimum yarn slippage. At the time of their formation, the fibres or yarns are sized, to protect the surface and aid the process of further handling such as weaving. Before the fibres are finally used in the fabrication of a composite, the size is usually removed by heat cleaning or washing.

Fibres are also now becoming available as mixtures. An advanced producer of composites, Textron Specialty Materials (TSM), has started to market a continuous-fibre epoxy resin prepreg tape that contains a mixture of large boron fibres and smaller-diameter carbon fibres, with a fibre density of 70 to 80 per cent. While carbon is good in tension, it lacks good compressive properties. This deficiency is overcome through the use of boron fibres that exhibit their best properties in compression. The combined material has good flexural properties, which are of importance to the manufacturing of submersible structures, sporting goods, and medical equipment.

COMPOSITES WITH POLYMER MATRICES

Polymer-matrix composites (PMCs) have matrices of thermoplastic or thermosetting polymers-traditionally glass fibre available in the form of woven material embedded in polyester. These materials are utilized at temperatures of not more than 200°C in commercial, industrial, and transportation applications, including chemically resistant piping, valves, pressure vessels, and reactors. The large numbers of resin formulations, curing agents, and fillers provide an extensive range of possible properties. Because they are superior to polyesters in resisting moisture and offer superior mechanical properties, epoxies have been the commonest thermosetting matrix material for more demanding applications. Bismaleimide resins (BMIs), on the other hand, possess many of the same desirable properties as epoxies, such as handlability, relative ease of processing, and good mechanical properties, and have service temperatures of up to 250°C compared with 180°C for epoxies. Thermosets are constantly being upgraded to tougher grades of higher heat resistance. Newly developed polyimide (PI) resins, for instance, can withstand exposure to temperatures of more than 300°C.However, when these polymerscure, volatile matter is released, which produces undesirable voids in the final product. Although this problem has been solved, polyimides are too brittle for very demanding applications. Phenolic resins suffer the same disadvantage but are used for applications that demand relatively high heat resistance .Difficult to ignite, phenolics produce less smoke, and are less toxic when they do burn. They are therefore used for the interior panels of aircraft, where combustion requirements justify lower properties. Also, these materials and the newly developed thermoplastic polyethersulphone contain additives that react to fire by emitting contained water as vapour, which extinguishes the frill. Attempts to improve the hot-wet performance and impact resistance of thermosetting resins like epoxies and BMls are continuing. Thermoplastic matrix materials exhibit high strain to failure, and are ideally suited for use as matrix material combined with high-strength and high-strain carbon fibres. These materials include resins such as polyetheretherketone (PEEK), with a melting point of 334°C, polyphenylene sulphide (PPS), poly etherimide (PEI), polyamideimide (PAl), and polyether sulphone (PES). In general, they have an unlimited shelf life, and offer lower-cost composite processing because they can be potentially remoulded by the application of heat and

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Impact Factor (PIF): 2.243

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pressure. Composite thermoplastics are very different from the general commercial thermoplastics such as polyethylene, polyvinyl chloride and polystyrene. These thermoplastics are tougher and can withstand higher service temperatures.

Experiments with a blowtorch on aluminium and a carbon fibre-PEEK composite showed that the latter withstood the flame much better .Apart from the cost aspect, the choice of a polymer for a specific application must be based on a full knowledge of the material properties required for the intended service temperatures and loads. Excellent accounts are available covering these aspects for epoxies, polyimides, bismaleimides, thermoplastic systems, and high-temperature polymers.

DuPont recently announced its thermoplastic engineered preforms (Tepex), which consists of a consolidated sheet composite that can be formed into complex parts in less than 60 seconds, avoiding labour intensive, time consuming, and costly fabrication techniques. It is reported that end-users can choose from a wide selection of resins-from low performance types to the more exotic polymers' such as PEEK-and also make a selection from different continuous fibre systems (fabrics, and unidirectional or, non-woven systems) of varying types such as glass, Kevlar, carbon, and hybrids. New grades of performance polymers increasingly appear on the market. Among the newcomers is a family of polymers based on polycyclohexylene-dimethylene-terephtalate or PCT, which is a high-temperature, semi-crystalline, thermoplastic polyester that melts at 285°C and is capable of long-term service temperatures of up to 170°C.

ABOUT AEROSPACE COMPOSITES

Composite materials are widely used now days in aerospace industries on account of their high stiffness and strength as well as their low density has created a need for the study of its behaviour at elevated temperatures. These materials present specific properties such as stiffness/weight and strength/weight ratios higher than those of metallic materials. However, mechanical or structural components made of composite materials may suffer large damage extension when subject to impact loads, tensile, bending, compression and shear loads with the corresponding decrease of their residual strength and the subsequent risk of structural failure under service loads. The possibility of using composite materials for load bearing applications in future supersonic transport aircraft , where the service temperatures are likely to fall in the range 120-150°C, has created a need for data on materials at these elevated temperatures. Aircraft structures are mostly made of composites. These materials are extremely strong and light. They comprise long fine glass, Kevlar and carbon fibres immersed in a so-called matrix of epoxy. In its uncured state, the epoxy is sticky, soft, and pliable. In its cured state, the epoxy is hard and rigid. Curing involves heating the epoxy until a chemical reaction begins. This reaction provides additional heat and converts the epoxy into its final physical and chemical form. One of the other important advantages of composites is that they can be assembled into large structures by bonding rather than riveting. Airframe cost forms the significant portion in the overall cost of aircraft. Any reduction in the airframe cost will also reduce the overall acquisition cost of the aircraft.

The Boeing 787 Dreamliner is pushing envelope with a total composites of 50% by weight, including the integration of an all composite fuselage, wings and applications.

The Airbus A380 will be 25% by weight composites including 23% carbon fiber-reinforced polymers and 2% GLARE fibre glass reinforced aluminium.

CHARACTERIZATION OF DAMAGE MODES

Composite laminates used in engineering applications are susceptible to low velocity impact damage within their service lives when the objects such as runway debris, hand tools fall down on composites. Low velocity impact damage causes delamination which can reduce the structural integrity of the composite materials significantly. An understanding the effect of damage and post impact damage due to low velocity impact is an important subject to be investigated in a natural fiber reinforced composites. However, the impact resistance of a composite material is always difficult to determine due to some other factors such as delimitation at the interface, fiber breakage, matrix cracking and fiber pull out .In generally using the acoustic emission techniques used to classify the modes and different ranges obtained for the failure mechanisms in composite materials.
The matrix cracking is overwhelmingly higher than for the other damage modes (although of considerably lower amplitude). This suggests that the geometry and nature of the laminate reinforcement may in some cases lead to

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the detection of a very small number of signals related e:g:, to the fiber failure, with some detriment for their significance.

The frequency range suggested for delamination is much larger than for the other damage modes and is likely to depend on the dimension of the delimitation and therefore on the rubbing of the internal delaminated surfaces during loading.

AIRCRAFT WING RIBS

In an aircraft, ribs are forming elements of the structure of a wing, especially in traditional construction. By analogy with the anatomical definition of "rib", the ribs attach to the main spar, and by being repeated at frequent intervals, form a skeletal shape for the wing. Usually ribs incorporate the airfoil shape of the wing, and the skin adopts this shape when stretched over the ribs.

For aerodynamic reasons the wing contour in the chord direction must be maintained without appreciable distortion. Unless the wing skin is quite thick, span wise stringers must be attached to the skin in order to increase the bending efficiency of the wing. Therefore to hold the skin-stringer wing surface to contour shape and also to limit the length of stringers to an efficient column compressive strength, internal support or brace units are required. These structural units are referred to as wing ribs.

The ribs also have another major purpose, namely, to act as transfer or distribution unit. All the loads applied to the wing are reacted at the wing supporting points, thus these applied loads must be transferred into the wing cellular structure composed of skin, stringers, spars, etc., and then reacted at the wing support points. The applied loads may be only the distributed surface air-loads which require relatively light internal ribs to provide this carry through or transfer requirement, to rather heavy ribs which must absorb and transmit large concentrated applied loads such as those from landing gear reactions, power plant reactions and fuselage reactions. In between these two extremes of applied load magnitudes are such loads as reactions at supporting points for ailerons, flaps, and leading edge high lift units and the many internal dead weight loads such as fuel and military armament and other installations. Thus ribs can vary from a very light structure which serves primarily as a former to a heavy structure which must receive and transfer loads involving thousands of pounds.

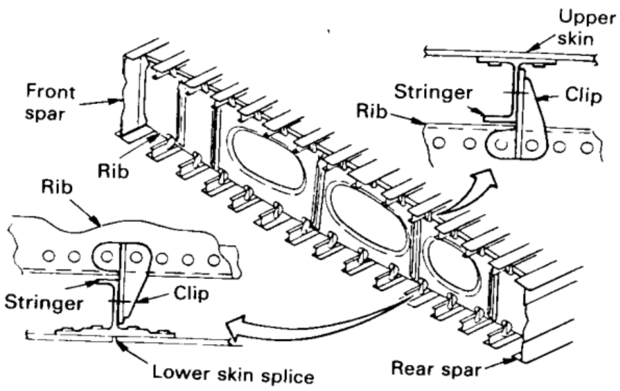

Figure 0-1 Wing Rib Construction

Wing ribs are usually manufactured from either wood or metal. Aircraft with wood wing spars may have wood or metal ribs while most aircraft with metal spars have metal ribs. Wood ribs are usually manufactured from spruce. The three most common types of wooden ribs are the plywood web, the lightened plywood web, and the truss types. Of these three, the truss type is the most efficient because it is strong and lightweight, but it is also the most complex to construct.

TYPES OF WING RIBS

There are several types of ribs. Based on manufacturing ribs are classified as

- Form-ribs

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- Plate-type ribs
- Truss ribs
- Closed ribs
- Forged ribs
- Milled ribs

a) *FORM-RIBS*
Form-ribs are made from a sheet of metal bent into shape, such as a U-profile. This profile is placed on the skin, just like a stringer, but then in the other direction.

b) *PLATE-TYPE RIBS*
Plate-type ribs consist of sheet-metal, which has upturned edges and (often has) weight-saving holes cut into it.

c) *TRUSS RIBS*
Truss ribs are built up out of profiles that are joined together. These joints require great attention during design and manufacture. The ribs may be light or heavy in design which makes them suitable for a wide range of loads.

d) *CLOSED RIBS*
Closed-ribs are constructed from profiles and sheet metal and are suitable for closing off sections of the wing (e.g.: the fuel tank). Here too, particular care must be taken with the joints and this type of rib is also suitable for application in a variety of loading conditions.

e) *FORGED RIBS*
Forged ribs are manufactured using heavy press-machinery. The result is fairly rough; for more refined parts, high-pressure presses are required, which are very expensive. Forged pieces (usually) have to undergo further treatment (for smoother edges and holes). Forged ribs are used for sections where very high loads apply - near the undercarriage.

f) *MILLED RIBS*
Milled ribs are solid structures. They are manufactured by milling away excess material from a solid block of metal (usually using computer-controlled milling machines). The shape of these ribs is always accurately defined. Such ribs are used under similar conditions as those for forged ribs.

Ribs are made out of wood, metal, plastic, composites, foam. The wings of kites, hang gliders, paragliders, powered kites, powered hang gliders, ultra lights, windmills are aircraft that have versions that use ribs to form the wing shape.

For full size and flying model aircraft wing structures that are usually made of wood, ribs can either be in one piece (forming the airfoil at that rib's "station" in the wing), or be in a three-piece format, with the rib web being the part that the one-piece rib consisted of, with capstrips for the upper and lower edging of the rib, running from the leading edge to the trailing edge, being the other two component parts.

RIB SPACING

Rib spacing is one of the major criteria in wing construction. Preliminary rib spacing is arrived based on the structural weight. And the location of control surface and heavy weight locations, ribs are provided to support.

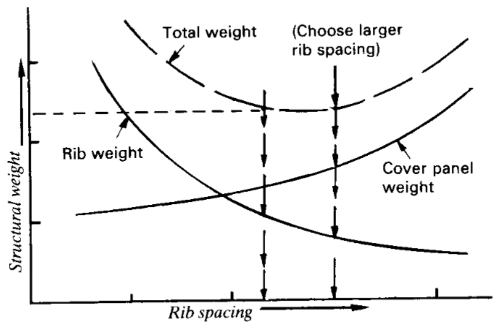

Figure H-1 Wing Rib spacing vs. Structural weight

ISSN 2349-6193
Impact Factor (PIF): 2.243

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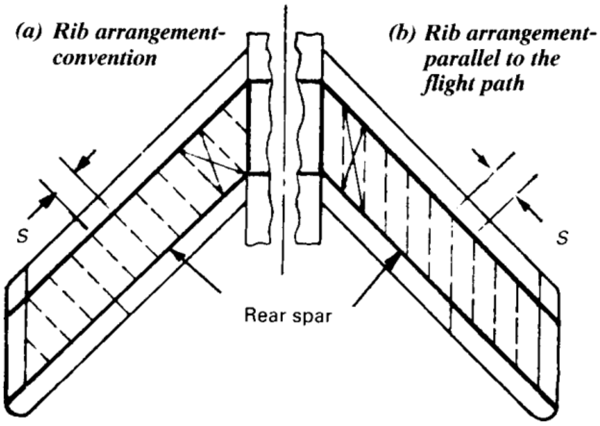
RIB ORIENTATION
There are two types of rib orientation is followed in wing construction.

- Perpendicular to rear spar
- Parallel to flight path



FigureI-1 Wing Rib Orientation

Advantages

- **Perpendicular to rear spar**
  - Rib length is less
  - Connection easy
- **Parallel to flight path**
  - Provides better aerodynamic shape

Disadvantages

- **Parallel to flight path**
  - Rib length is more (nearly 28%)
  - Maintaining 90° at joints
  - Skin gauge is more

DESIGN OVERVIEW
GEOMETRY SPECIFICATION
Airfoil details are mentioned below.

X in m	Y in m	X in m	Y in m
0.004253	0.125173	0.869036	0.392405
0.015594	0.157195	0.91837	0.388912
0.021265	0.163017	0.968271	0.384835
0.02807	0.171168	1.017608	0.381925
0.042247	0.182811	1.070912	0.376685
0.059259	0.196784	1.120813	0.370281
0.077972	0.207846	1.171283	0.363875
0.102923	0.227641	1.218349	0.357472
0.138082	0.250347	1.271087	0.347574
0.171539	0.271305	1.320422	0.337678
0.203862	0.289936	1.370891	0.328361

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0.233916	0.302163	1.426464	0.318465
0.257734	0.314389	1.497914	0.304493
0.283251	0.323705	1.576738	0.285862
0.30877	0.333019	1.625506	0.2748
0.357538	0.346992	1.674274	0.261991
0.418214	0.361549	1.720774	0.249765
0.465849	0.367952	1.768975	0.236956
0.513481	0.374937	1.818876	0.225312
0.566219	0.382506	1.86878	0.211339
0.614421	0.386583	1.916979	0.196202
0.667156	0.388328	1.966882	0.181065
0.717626	0.391239	2.017352	0.16651
0.768096	0.392405	2.065551	0.149626
0.818566	0.392986	2.113186	0.133907

X in m	Y in m	X in m	Y in m
2.16252	0.11644	1.375428	0.002911
2.211288	0.099557	1.278458	0.002329
2.260623	0.079762	1.147465	0.004658
2.307123	0.064624	1.045393	0.006404
2.359294	0.044829	0.972241	0.006986
2.381976	0.035514	0.92234	0.007569
2.401824	0.029692	0.874138	0.008151
2.383112	0.028528	0.768096	0.008733
2.357026	0.024453	0.619524	0.01048
2.323	0.020959	0.42275	0.008733
2.295215	0.020377	0.304234	0.013391
2.254954	0.015719	0.260568	0.015137
2.20108	0.012808	0.23505	0.016302
2.152312	0.01048	0.203862	0.021541
2.101845	0.008733	0.166435	0.026199
2.055345	0.006404	0.137514	0.032021
2.028124	0.002911	0.108027	0.039008
1.989564	0.002329	0.088746	0.045994
1.931723	0.002911	0.05699	0.057638
1.867078	0.002911	0.034308	0.071029
1.804132	0.001747	0.020698	0.083255
1.747992	0.002911	0.012759	0.091406
1.676542	0.002329	0.008223	0.100139
1.580139	0.002911	0.005954	0.109454
1.48147	0.002329	0.004253	0.119352

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Table 0-1 Wing Rib – Airfoil Details

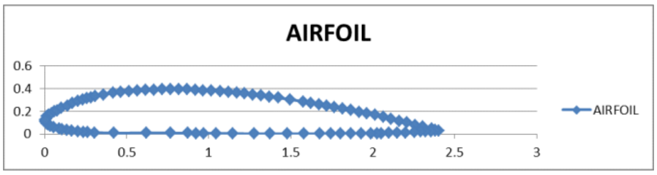

Figure 0-1 Wing Rib Airfoil Specificstion

Length(m)	X	2.401824
Height(m)	Y	0.392986

DESIGN SPECIFICATIONS:

The total rib design consideration is based on the airfoil section which is referred from above section. The total number of ribs, web and Rib panels are employed from various design configurations.

Number of spars = 2

Number of web = 2

Panels = 3

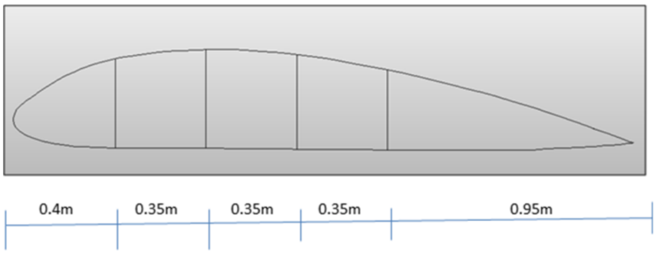

Figure 0-1 Wing Rib Design Specificstion

Thickness of skin = 0.02m

MATERIAL SPECIFICATION:

Properties of a carbon/epoxy lamina are considered for this analysis. Material properties are referred from "DEPARTMENT OF DEFENSE HANDBOOK" in composite materials handbook volume 3. Polymer matrix composites materials usage, design, and analysis. The material properties are mentioned below.

$E1 = 172$ Gpa

$E2 = 12$ Gpa

$G12 = 4.5$ Gpa

$v12 = 0.30$

$\rho = 1550$ kg/m3

$F1 = 760$ Mpa

$F2 = 28$ Mpa

$F12 = 62$ Mpa

$\alpha 1 = 0.54x10\text{-}6$ mm/mm/C°

$\alpha\ 2 = 35.1x10\text{-}6$ mm/mm/C°

LOAD SPECIFICATION:

Literature Survey for Load details:

The various loading condition is gathered from the literature survey.

LOCATION	Fx in N	Fy in N	Fz in N

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1	-3545	2490	1257
2	-2605	2519	1131
3	-1731	1746	1341
4	-1358	871	1597
5	-1289	746	1513
6	1156	-13	1080
7	1328	-546	668
8	-1898	-1428	514
9	-1353	-917	904
10	1568	621	1312
11	308	-98	1788
12	1881	1400	1718
13	1218	3141	747
14	2984	2554	484
15	2051	4943	1128
16	2032	4876	1146
17	1899	1378	1699
18	3229	3151	746

Table 0-1Wing Rib – Load Details

FINITE ELEMENT ANALYSIS

INTRODUCTION

FEA consists of a computer model of a material or design that is stressed and analyzed for specific results. It is used in new product design, and existing product refinement.
A company is able to verify a proposed design will be able to perform to the client's specifications prior to manufacturing or construction. Modifying an existing product or structure is utilized to qualify the product or structure for a new service condition. In case of structural failure, FEA may be used to help determine the design modifications to meet the new condition.

There are generally two types of analysis that are used in industry:

- 2-D modeling
- 3-D modeling

While 2-D modeling conserves simplicity and allows the analysis to be run on a relatively normal computer, it tends to yield less accurate results.

3-D modeling, however, produces more accurate results while sacrificing the ability to run on all but the fastest computers effectively. Within each of these modeling schemes, the programmer can insert numerous algorithms (functions) which may make the system behave linearly or non-linearly. Linear systems are far less complex and generally do not take into account plastic deformation. Non-linear systems do account for plastic deformation, and many also are capable of testing a material all the way to fracture.

HOW DOES FINITE ELEMENT ANALYSIS WORK

FEA uses a complex system of points called nodes which make a grid called a mesh. This mesh is programmed to contain the material and structural properties which define how the structure will react to certain loading conditions.
Nodes are assigned at a certain density throughout the material depending on the anticipated stress levels of a particular area. Regions which will receive large amounts of stress usually have a higher node density than those

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which experience little or no stress. Points of interest may consist of: fracture point of previously tested material, fillets, corners, complex detail, and high stress areas. The mesh acts like a spider web in that from each node, there extends a mesh element to each of the adjacent nodes. This web of vectors is what carries the material properties to the object, creating many elements.

TYPES OF ENGINEERING ANALYSIS

1) STRUCTURAL ANALYSIS
   It consists of linear and non-linear models. Linear models use simple parameters and assume that the material is not plastically deformed. Non-linear models consist of stressing the material past its elastic capabilities. The stresses in the material then vary with the amount of deformation.
2) VIBRATIONAL ANALYSIS
   It used to test a material against random vibrations, shock, and impact. Each of these incidences may act on the natural vibrational frequency of the material which, in turn, may cause resonance and subsequent failure.
3) FATIGUE ANALYSIS
   It helps designers to predict the life of a material or structure by showing the effects of cyclic loading on the specimen. Such analysis can show the areas where crack propagation is most likely to occur. Failure due to fatigue may also show the damage tolerance of the material.
4) HEAT TRANSFER
   This analysis models the conductivity or thermal fluid dynamics of the material or structure. This may consist of a steady-state or transient transfer. Steady-state transfer refers to constant thermo properties in the material that yield linear heat diffusion.

MSC PATRAN

Patran is the world's most widely used pre/post-processing software for Finite Element Analysis (FEA), providing solid modeling, meshing, analysis setup and post-processing for multiple solvers including MSC Nastran, Marc, Abaqus, LS-DYNA, ANSYS, and Pam-Crash.

Patran is the leading pre- and post-processor for CAE simulation. The program's advanced modeling and surfacing tools allow you to create a finite element model from scratch.

You can also take advantage of Patran's advanced CAD access tools to work directly on your existing CAD model. With direct access, Patran imports model geometry without modifications.

No translation takes place, so your CAD geometry remains intact. After geometry is imported, you can use Patran to define loads, boundary conditions, and material properties.

5) PATRAN BENEFITS
   - Reduce cost by replacing physical testing with less expensive digital simulation
   - Standardize on your modeling environment with access to multiple solver technologies
   - Customize your modeling environment to fit your simulation process

6) PATRAN APPLICATIONS
   - Aerospace
   - Automobile
   - Ship building
   - Medical products
   - Consumer products

MSC NASTRAN

MSC Nastran is the world's most widely used Finite Element Analysis (FEA) solver. When it comes to simulating stress, dynamics, or vibration of real-world, complex systems, MSC Nastran is still the best and most trusted software in the world – period.
Today, manufacturers of everything from parts to complex assemblies are choosing the FEA solver that is reliable and accurate enough to be certified by the FAA and other regulatory agencies.

ISSN 2349-6193
Impact Factor (PIF): 2.243
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MSC Nastran is a general purpose finite element analysis program. Nastran is capable of solving wide variety of engineering problems including:

- Linear static analysis
- Static analysis with geometric and material non-linearity
- Transient analysis with geometric and material non-linearity
- Normal mode and buckling analysis
- Linear static and vibration analysis with cyclic symmetry
- Linear and non-linear steady-state heat transfer
- Heat transfer analysis
- Vibration analysis
- Aero elasticity
- Multilevel super elements
- Design sensitivity and optimization
- Acoustics
- P-version elements and adaptivity

7) Advantages of Nastran

- Extensively documented and quality assurance tested
- Continually being enhanced by the addition of new capabilities
- Efficient due to its use of modern database technology and use of modern sparse matrix and numerical analysis techniques
- Used extensively by Aerospace, transportation, bio-medical and general industries

HYPERMESH

Altair HyperMesh is a high-performance finite element pre-processor to prepare even the largest models, starting from import of CAD geometry to exporting an analysis run for various disciplines.

HyperMesh enables engineers to receive high quality meshes with maximum accuracy in the shortest time possible. A complete set of geometry editing tools helps to efficiently prepare CAD models for the meshing process.

Meshing algorithms for shell and solid elements provide full level of control, or can be used in automatic mode. Altair's BatchMeshing technology meshes hundreds of files precisely in the background to match user-defined standards. HyperMesh offers the biggest variety of solid meshing capabilities in the market, including domain specific methods such as SPH, NVH or CFD meshing.

8) HYPERMESH BENEFITS

- A Powerful FEA Modeling Solution for the Enterprise
- High-speed, High-quality Meshing
- Increase End-user Efficiency with Batch Meshing and Automated Model Assembly
- Close the Loop between CAD and FEA
- It has a flexible set of morphing tools allows users to modify legacy meshes without re-meshing to automate the investigation new design proposals.

Problem Solving Process

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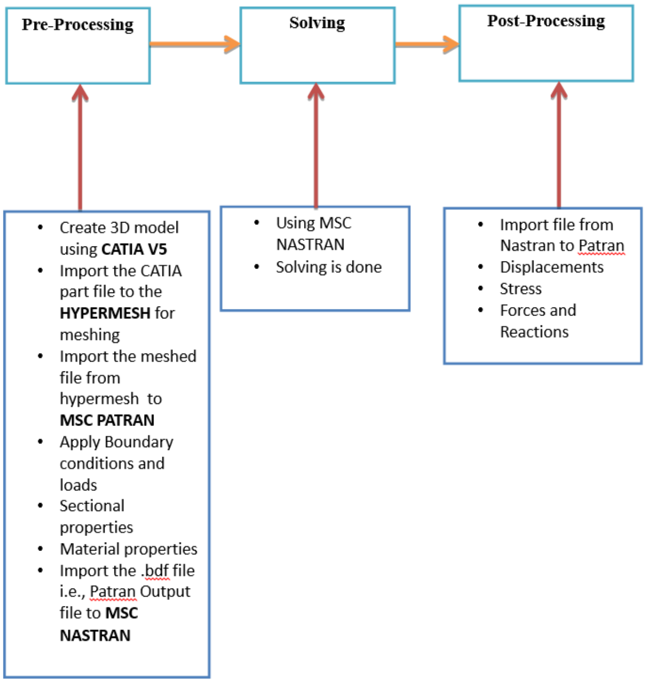

FEM MODELING

FEM OF WING-RIB DESIGN USING HYPERMESH

2D Shell element is considered for FE modeling throughout the Wing-Rib structure.

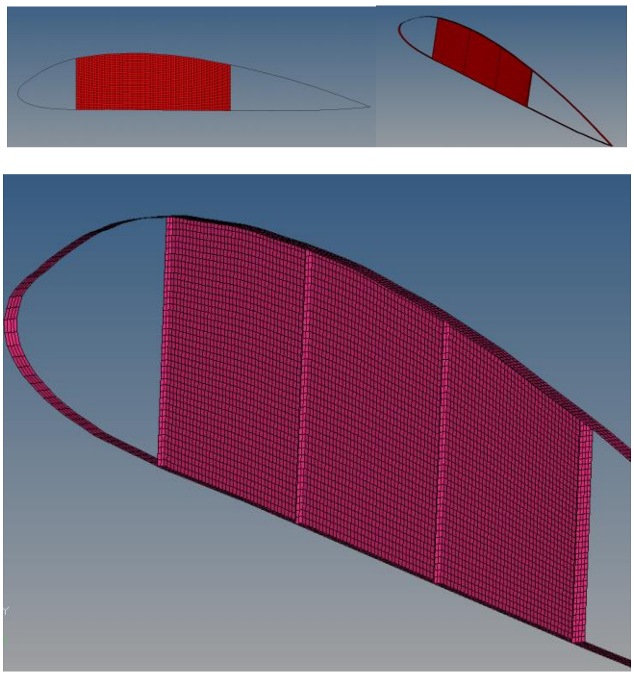

Figure 0-1 Wing Rib Model

CG location
X = 0.9667121
Y= 0.1809171 Total no. of Elem = 7834
Z = 0.009998 Total no. of Node = 7400

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BOUNDARY CONDITIONS:
Since shell element is having 6 Degrees of freedom, all 6 Degrees of freedom is constraint at the spar locations

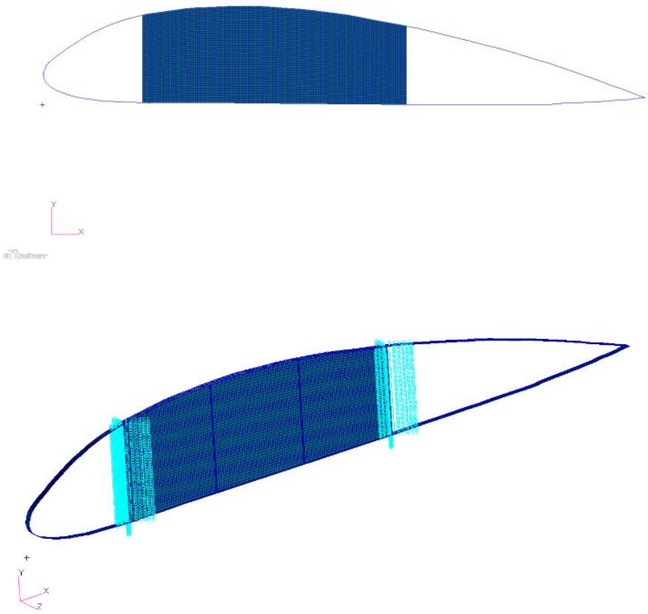

Figure 0-1 Wing Rib Boundary Conditions

RESULTANT LOAD DETAILS IN PATRAN

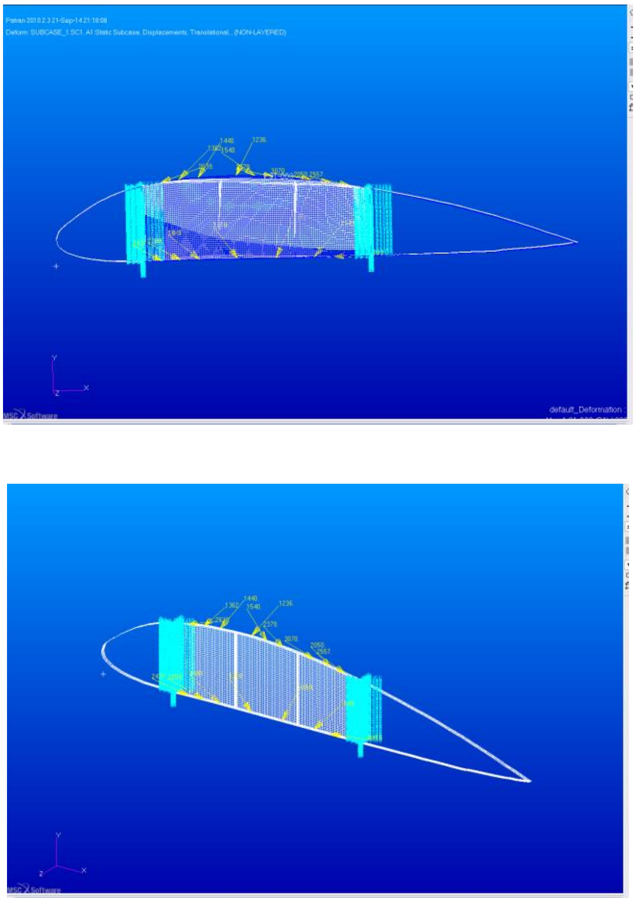

Figure 0-1 Resultant Load Details

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RESULT VALIDATION
ITERATION 1: 0/45/90/90/45/0

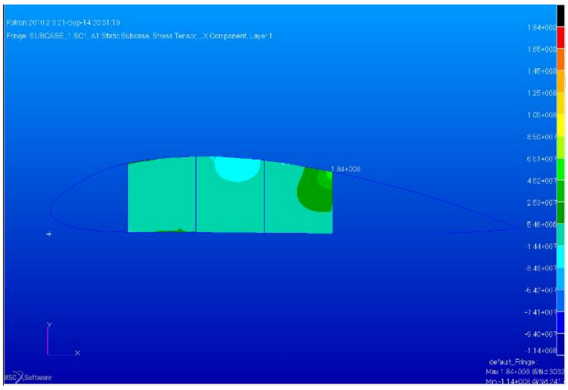
Table 0-1 Iteration 1 – 0/45/90/90/45/0

Orientation Detail		X Direction		Allowable considered	RF in tension	RF in Compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/45/90/90/45/0	Layer 1	204295152	-120383104	760000000	3.72	6.31	3.72
	Layer 2	34252396	-37941276	760000000	22.19	20.03	
	Layer 3	22130704	-36845248	760000000	34.34	20.63	
	Layer 4	65532040	-40580336	760000000	11.60	18.73	
	Layer 5	55600776	-49659020	760000000	13.67	15.30	
	Layer 6	133911024	-174860992	760000000	5.68	4.35	

Orientation Detail		Y Direction		Allowable considered	RF in tension	RF in Compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/45/90/90/45/0	Layer 1	3930982	-4663110.5	28000000	7.12	6.00	2.50
	Layer 2	11004487	-5261050	28000000	2.54	5.32	
	Layer 3	11202616	-6974120.5	28000000	2.50	4.01	
	Layer 4	9815930	-6444062.5	28000000	2.85	4.35	
	Layer 5	9027870	-6150916	28000000	3.10	4.55	
	Layer 6	11115873	-7396589.5	28000000	2.52	3.79	

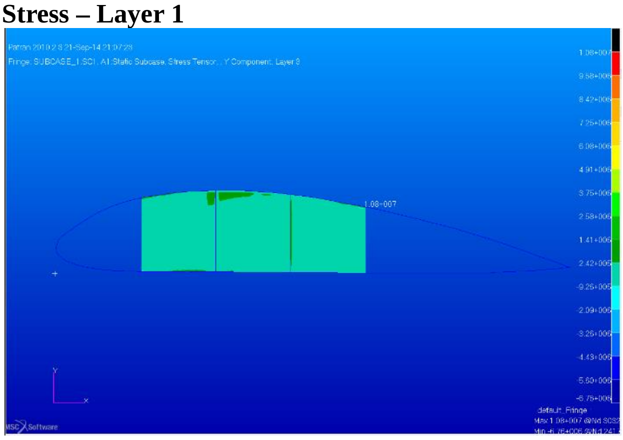
Orientation Detail		XY Direction		Allowable considered	RF in tension	RF in Compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/45/90/90/45/0	Layer 1	2114860.75	-3821076.75	68000000	32.15	17.80	13.50
	Layer 2	3272625.25	-5038519.5	68000000	20.78	13.50	
	Layer 3	3380191.5	- 1443327.125	68000000	20.12	47.11	
	Layer 4	3495811.75	-2087791.75	68000000	19.45	32.57	
	Layer 5	3457449	-4239710	68000000	19.67	16.04	
	Layer 6	3659359.25	-4261850.5	68000000	18.58	15.96	

Table 0-2 Iteration 1 – 0/45/90/90/45/0



Max tension Stress= 1.84E8 Pascal, Max. Compression Stress = -1.14E8 Pascal

Figure 0-1 X Component Stress – Layer 1



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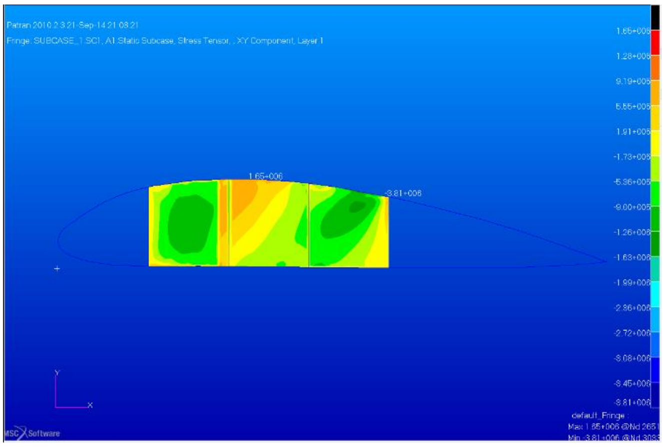
Max. Tension Stress= 1.08E7 Pascal, Max. Compression Stress = -6.76E6 Pascal

Figure 0-2 Y Component Stress – Layer 3



Max. Tension Stress= 1.65E6 Pascal, Max. Compression Stress = -3.81E6 Pascal

Figure 0-3 XY Component Stress – Layer 2

ITERATION 2: 0/30/60/60/30/0

Orientation Detail		X Direction		Allowable considered	RF in Tension	RF in Compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	pa			
0/30/60/60/30/0	Layer 1	192549280	-123007712	760000000	3.95	6.18	3.95
	Layer 2	69458016	-64437180	760000000	10.94	11.79	
	Layer 3	20166458	-48855852	760000000	37.69	15.56	
	Layer 4	43831444	-39433676	760000000	17.34	19.27	
	Layer 5	67212280	-77829688	760000000	11.31	9.76	
	Layer 6	121874760	-174610160	760000000	6.24	4.35	

Orientation Detail		Y Direction		Allowable considered	RF	RF	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	pa			
0/30/60/60/30/0	Layer 1	4563441	-6965412	28000000	6.14	4.02	1.43
	Layer 2	8806789	-5237697.5	28000000	3.18	5.35	
	Layer 3	11700238	-6887546.5	28000000	2.39	4.07	
	Layer 4	10424863	-7000794.5	28000000	2.69	4.00	
	Layer 5	16429572	-7601491.5	28000000	1.70	3.68	
	Layer 6	19588588	-10125427	28000000	1.43	2.77	

Orientation Detail		XY Direction		Allowable considered	RF	RF	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/30/60/60/30/0	Layer 1	2469168.5	-4569447	68000000	27.54	14.88	11.53
	Layer 2	3372395.5	-5707910	68000000	20.16	11.91	
	Layer 3	2840255.75	-2287847.5	68000000	23.94	29.72	
	Layer 4	4735513.5	-2391659.5	68000000	14.36	28.43	
	Layer 5	3474274.25	-4966677	68000000	19.57	13.69	
	Layer 6	4069480	-5899166.5	68000000	16.71	11.53	

Table 0-1 Iteration 2 – 0/30/60/60/30/0

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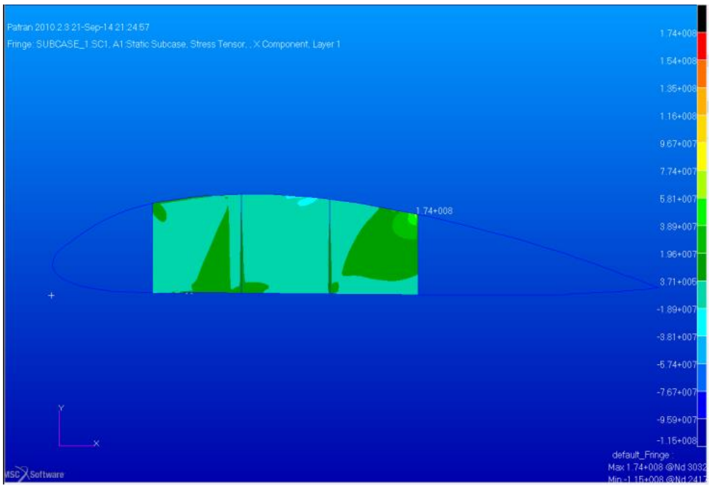
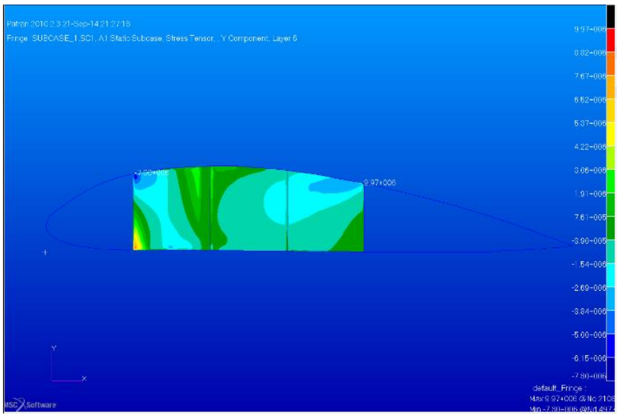
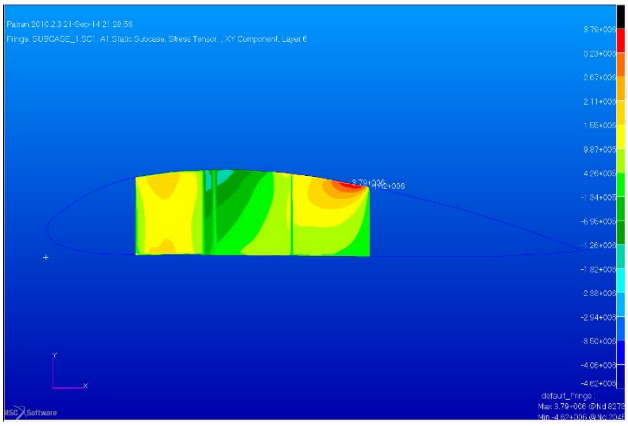


Max. Tension Stress = 1.7E8 Pascal, Max. Compression Stress = -1.15E8 Pascal
Figure 0-1 X Component Stress – Layer 1



Max. Tension Stress = 9.97E6 Pascal, Max. Compression Stress = -7.30E6 Pascal
Figure 0-2 Y Component Stress – Layer 6



Max. Tension Stress = 3.79E6 Pascal, Max. Compression Stress = -4.62E6 Pascal
Figure 0-3 XY Component Stress – Layer 6

ITERATION 3: 0/15/30/30/15/0

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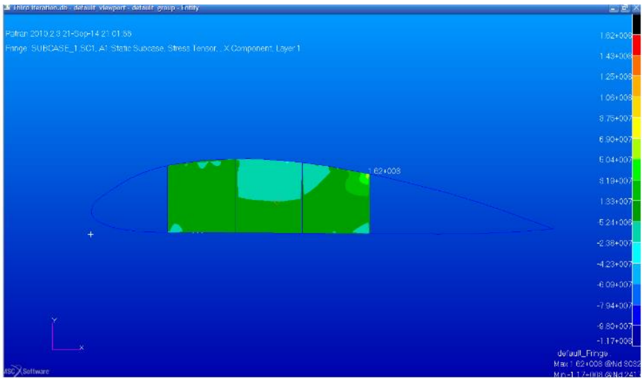
Orientation Detail		X Direction		Allowable considered	RF in Tension	RF in Compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/15/30/30/15/0	Layer 1	179293568	-125930264	760000000	4.24	6.04	4.24
	Layer 2	98628808	-85629480	760000000	7.71	8.88	
	Layer 3	27107882	-21690516	760000000	28.04	35.04	
	Layer 4	43506060	-28955076	760000000	17.47	26.25	
	Layer 5	69719672	-98682888	760000000	10.90	7.70	
	Layer 6	103801240	-163448720	760000000	7.32	4.65	

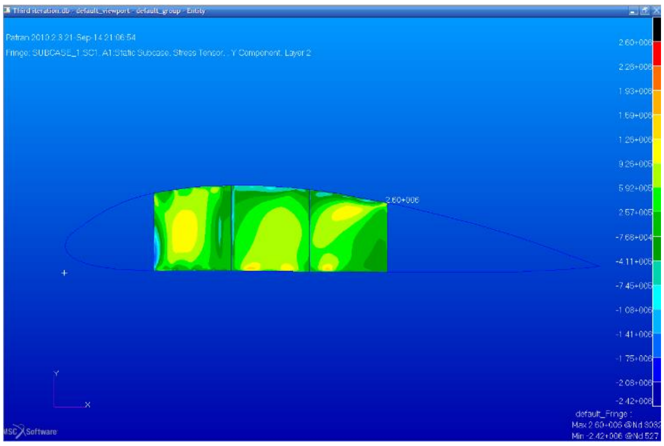
Orientation Detail		Y Direction		Allowable considered	RF in Tension	RF in Compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/15/30/30/15/0	Layer 1	6726740	-11302622	28000000	4.16	2.48	1.00
	Layer 2	4029955.75	-5564220.5	28000000	6.95	5.03	
	Layer 3	8808876	-4189188.5	28000000	3.18	6.68	
	Layer 4	15708963	-7521310.5	28000000	1.78	3.72	
	Layer 5	22994844	-11484183	28000000	1.22	2.44	
	Layer 6	28113520	-14592349	28000000	1.00	1.92	

Orientation Detail		XY Direction		Allowable considered	RF in Tension	RF in Compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/15/30/30/15/0	Layer 1	2800086.5	-6192761	68000000	24.28	10.98	9.20
	Layer 2	3846558.25	-7388821.5	68000000	17.68	9.20	
	Layer 3	4420469.5	-6851313.5	68000000	15.38	9.93	
	Layer 4	4523442.5	-6548990.5	68000000	15.03	10.38	
	Layer 5	5133705	-7323011.5	68000000	13.25	9.29	
	Layer 6	5545407	-7165525.5	68000000	12.26	9.49	

Table 0-1 Iteration 3 – 0/15/30/30/15/0



Max. Tension Stress = 1.62E8 Pascal, Max. Compression Stress = -1.17E8 Pascal
Figure 0-1 X Component Stress – Layer 1



Max. Tension Stress = 2.60E6 Pascal, Max. Compression Stress = -2.42E6 Pascal
Figure 0-2 Y Component Stress – Layer 2

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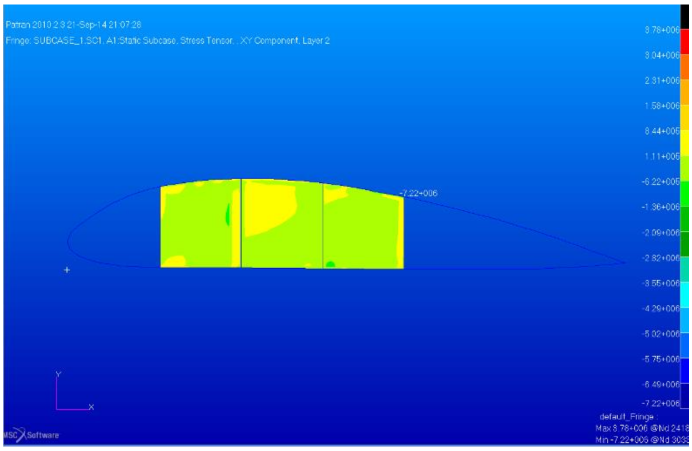

Max. Tension Stress = 3.78E6 Pascal, Max. Compression Stress = -7.22E6 Pascal
Figure 0-3 XY Component Stress – Layer 2

ITERATION 4: 0/10/20/20/10/0

Orientation Detail		X Direction		Allowable considered	RF in Tension	RF in compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/10/20/20/10/0	Layer 1	104393488	-89248568	760000000	7.28	8.52	7.28
	Layer 2	38472724	-37782780	760000000	19.75	20.11	
	Layer 3	42568296	-28734526	760000000	17.85	26.45	
	Layer 4	68067544	-39001244	760000000	11.17	19.49	
	Layer 5	88975800	-52174960	760000000	8.54	14.57	
	Layer 6	88975800	-52174960	760000000	8.54	14.57	

Orientation Detail		Y Direction		Allowable considered	RF in Tension	RF in compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/10/20/20/10/0	Layer 1	4674236	-5815850.5	28000000	5.99	4.81	0.91
	Layer 2	9276381	-3727066.75	28000000	3.02	7.51	
	Layer 3	17279400	-4981698	28000000	1.62	5.62	
	Layer 4	24612872	-8970546	28000000	1.14	3.12	
	Layer 5	30613728	-12451080	28000000	0.91	2.25	
	Layer 6	30613728	-12451080	28000000	0.91	2.25	

Orientation Detail		XY Direction		Allowable considered	RF in Tension	RF in compression	Critical RF
		Max Stress	Min Stress				
		Pa	Pa	Pa			
0/10/20/20/10/0	Layer 1	3696555.75	-7645084.5	68000000	18.40	8.89	7.67
	Layer 2	4580710.5	-7645262	68000000	14.84	8.89	
	Layer 3	4975556.5	-7817925	68000000	13.67	8.70	
	Layer 4	5858518.5	-8304328	68000000	11.61	8.19	
	Layer 5	6583196	-8865250	68000000	10.33	7.67	
	Layer 6	6583196	-8865250	68000000	10.33	7.67	

Table 0-1 Iteration 4 – 0/10/20/20/10/0

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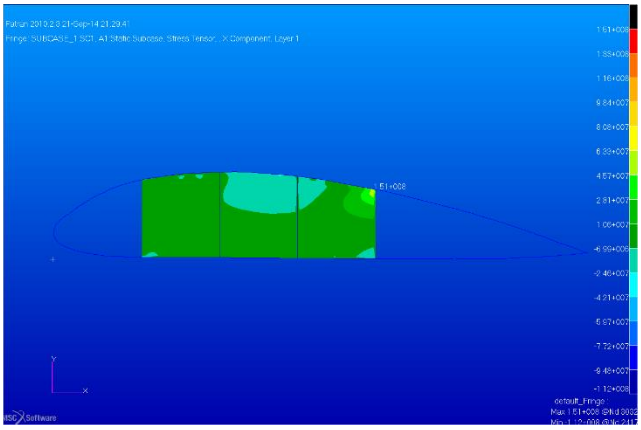
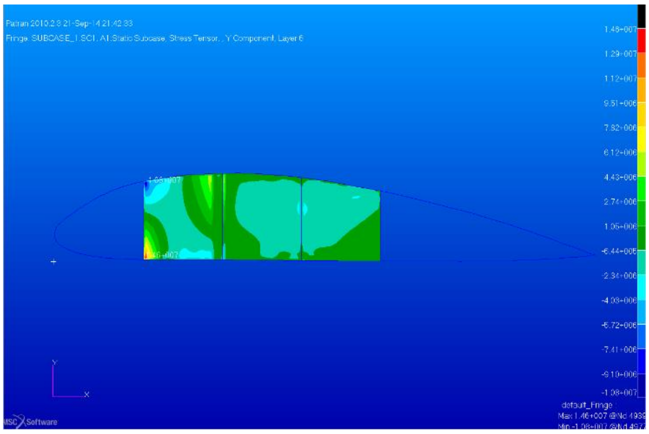


Max. Tension Stress = 1.51E8 Pascal, Max. Compression Stress = -1.12E8 Pascal
Figure 0-1 X Component Stress – Layer 1



Max. Tension Stress = 1.48E7 Pascal, Max. Compression Stress = -1.08E7 Pascal
Figure 0-2 Y Component Stress – Layer 6

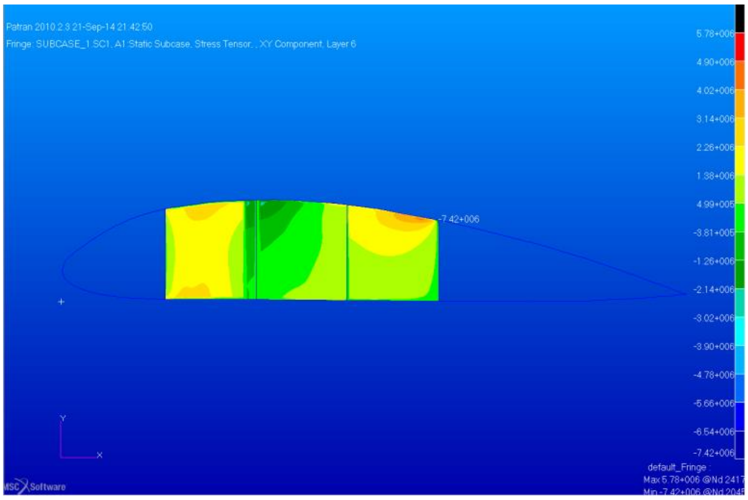


Figure 6.74 Max. Tension Stress = 5.78E6 Pascal, Max. Compression Stress = -7.42E6 Pascal
Figure 0-3 XY Component Stress – Layer 6

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RF COMPARISON FOR DIFFERENT PLY ORIENTATIONS

ITERATIONS	Ply Orientation	Critical RF in X direction	Critical RF in Y direction	Critical RF in XY direction
Iteration 1	**0/45/90/90/45/0**	**3.72**	**2.50**	**13.50**
Iteration 2	0//30/60/60/30/0	3.95	1.43	11.53
Iteration 3	0/15/30/30/15/0	4.24	1.00	9.20
Iteration 4	0/10/20/20/10/0	7.28	0.91	7.67

Most reliable ply orientation based on stresses in X, Y & XY Components is 0/45/90/90/45/0

Table 0-1 RF Comparison for Different ply Orientation

DEFLECTION PLOTS FOR DIFFERENT PLY ORIENTATION

ITERATIONS	*Ply Orientation*	*Resultant deflection in mm*
Iteration 1	*0/45/90/90/45/0*	*1.91*
Iteration 2	*0//30/60/60/30/0*	*1.93*
Iteration 3	*0/15/30/30/15/0*	*1.99*
Iteration 4	*0/10/20/20/10/0*	*1.99*

Minimum Resultant deflection = 1.91 mm
Table 0-1 Deflection Plots for Different Ply orientation

RESULT COMPARISON FROM CHART

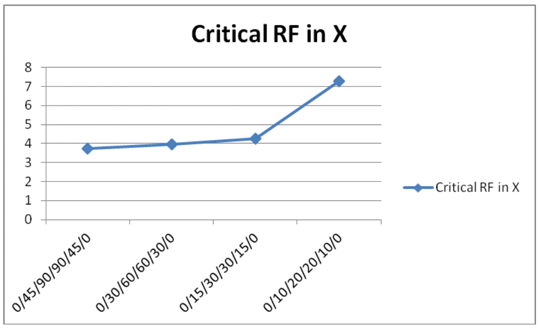

Figure 0-1 Critical RF in X Direction

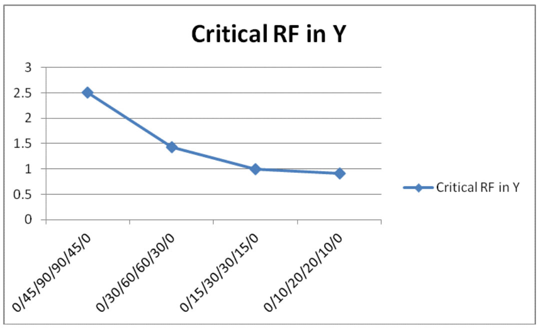

Figure 0-2 Critical RF in Y Direction

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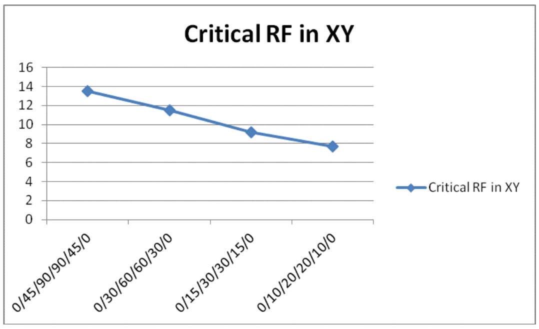

Figure 0-3 Critical RF in XY Direction

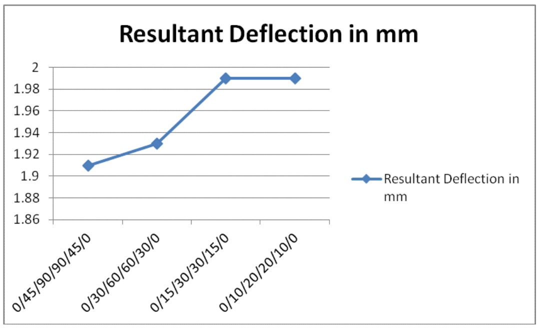

Figure 0-4 Resultant Deflection Chart

CONCLUSION

Based on stresses in X, Y and XY directions, Iteration 4(0/10/20/20/10/0) gives the highest critical RF of 7.28 in X-direction; Iteration 1(0/45/90/90/45/0) gives the highest Critical RF of 2.51 in Y-direction and Iteration 1 (0/45/90/90/45/0) gives the highest Critical RF of 13.5 in XY-direction. From the considered coordinate system to design the wing rib, Y-direction and XY-direction is carrying most of the shear load. Since the wing rib is shear critical, iteration 1 (0/45/90/90/45/0) values are reliable to design the wing rib structures.

Resultant displacement values are compared between four different ply orientations. Iteration 1(0/45/90/90/45/0), Iteration 2(0/30/60/60/30/0), Iteration 3(0/15/30/30/15/0) and Iteration 4(0/10/20/20/10/0) gives the resultant displacement values of 1.91mm, 1.93mm, 1.99mm and 1.99mm respectively. The minimum resultant displacement value lies in Iteration 1(0/45/90/90/45/0) and considered as more withstand ply orientation.

Based on stresses in X, Y and XY directions, iteration 1(0/45/90/90/45/0) gives more reliable than any other ply orientation. Based on Resultant deflection values, iteration 1(0/45/90/90/45/0) gives minimum resultant deflection value of 1.91 mm. Hence it is reasonable to conclude that iteration 1 is more reliable ply orientation for modeling of composite wing-rib. We can conclude that ply orientation of 0/45/90/90/45/0 is safer based on strength values of the composite wing ribs.

SCOPE OF FUTURE WORK

From the above paper, scope of the future work lays in various aspects. Now a day Optimization technology is used to reduce the weight of the aircraft components. This paper leads to various optimization approaches. Optimization of the composite wing rib can be done in the upcoming research. Wing rib morphing also one of the widest topics based on the model and analysis aircraft structural wing using composite materials. De-lamination approach can be carried forward to sustain in composite modeling.

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5. Composite Airframe Structures - Michael C.Y. Niu.
6. Design And Analysis Of A Typical Wing Rib For Passenger Aircraft – ISSN 2319-8753
7. Analysis and Design of Flight Vehicle Structures – E.F. Bruhn.
8. Parameterized Automated Generic Model For Aircraft Wing Structural Design And Mesh Generation For Finite Element Analysis - ISRN: LIU-IEI-TEK-A--11/01202—SE.
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10. Airframe Stress Analysis and Sizing – Michael C.Y. Niu.