FINITE ELEMENT ANALYSIS OF COMPOSITE
AIRCRAFT FUSELAGE FRAME

by

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THESIS

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Abstract

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Supervising Professor: Andrey Beyle

Composites have been introduced in aircraft industries, for their stronger, stiffer, and lighter properties than their metal-alloys counterparts. The general purpose of an aircraft is to transport commercial or military payload. Aircraft frames primarily maintains the shape of fuselage and prevent instability of the structure. Fuselage is similar as wing in construction which consist of longitudinal elements (longerons and stringers), transverse elements (frames and bulkheads) and its external skin. The fuselage is subjected to forces such as the wing reactions, landing gear reaction, empennage reaction, inertia forces subjected due to size and weight, internal pressure forces due to high altitude. Frames also ensure fail-safe design against skin crack propagation due to hoops stress. Ideal fuselage frames cross section is often circular ring shape with a frame cap of Z section. They are mainly made up of light alloy commonly used is aluminium alloys such as Al-2024, Al-7010, Al-7050, Al-7175. Aluminium alloys have good strength to density ratios in compression and bending of thin plate. A high strength to weight ratio of composite materials can result in a lighter aircraft structure or better safety factor. This research focuses on analysis of fuselage frame under dynamic load condition with change in material. Composites like carbon fibre reinforced plastics [CFRP] and glass fibre reinforced plastics [GFRP] are

compared with traditional aluminium alloy Al-7075. The frame is subjected to impact test by dropping it at a velocity of 30 ft. / secs from a height of 86 inch from its centre of gravity. These parameters are considered in event of failure of landing gear, and an aircraft is subject to belly landing or gear-up landing. The shear flow is calculated due to impact force which acts in radial direction. The frame is analysed under static structural and explicit dynamic load conditions. Geometry is created in ANSYS Design Modeler. Analysis setup is created using ANSYS Explicit Dynamic (AUTODYN) and ANSYS Composite PrepPost (ACP-Pre) modules. Shear flow and Stress Flow equations are solved by generating a MATLAB code.

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April 20, 2017

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Nomenclature

$[\hat{k}]$	Global Stiffness Matrix
A	cross section area of element
E	Young's Modulus
L	Length of element
$[\hat{m}]$	Lumped-Mass Matrix
ρ	Density
$\ddot{x}_i$	components of nodal acceleration (i=1,2,3)
F_i	forces acting on the nodal points
b_i	component of body acceleration
m	mass attributed to the node
Δt	time step
c_x	longitudinal wave velocity
f	stability timestep factor
h	characteristic dimension of the element
σ_{ij}	stress tensor
ε_{ij}	strain tensor
F_{impact}	vertical load
m	mass of the frame
v_{intial}	initial velocity
v_{final}	final velocity of the frame at the impact
s	distance travelled after impact
q	shear flow thru the frame ring
I_y	moment of inertia due to stringer
q_s	shear flow for the open section

Sx	Shear Force in x- direction
Sy	Shear Force in y-direction
Ixx	Area-moment of Inertia in X direction
Iyy	Area-moment of Inertia in Y direction
Ixy	polar moment of inertia
t	thickness of the section
x, y	coordinate location from C.G
s	length of the surface / flange
σ_s	stress flow thru the open section
Mx	Bending moment about x axis
My	Bending moment about y axis
E_{zi}	Laminate Young Modulus in axial-direction

Chapter 1

Introduction

The purpose of an airplane is to transport a commercial or military payload. The portion of the airplane which houses the passengers on payload is referred to as fuselage. Fuselage vary greatly in size and configuration. The fuselage is subjected to large concentrated forces such as the wing reactions, landing gear reaction, empennage reaction etc. in addition to these loads it is also subjects to inertia forces subjected due to size and weight, internal pressures due to high altitude flight. To handle these internal pressures efficiently, a combination or circular cross section is required. The fuselage of a modern aircraft is a stiffened shell commonly referred as semi-monocoque construction. Semi-monocoque structure is very efficient, it has a high strength to weight ratio, and it has design flexibility and can withstand local failure without total failure through load redistribution. The fuselage as a beam contains longitudinal elements (longerons and stringers), transverse elements (frames and bulkheads) and its external skins. [1]

An aircraft frame 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. Fuselage frames are equivalent to that of wing ribs in function, except the frames may be influenced by loads resulting from equipment mounting. Frame loads are generally small and often tend to balance each other, hence they are generally of light construction. Fuselage frames perform functions such as support shell compression / shear, distribute concentrated loads, fail safe. They hold the fuselage cross-section to contour shape and limit the column length. Frames act as a circumferential tear strips to ensure fail-safe design against skin crack propagation. They also distribute external and internal applied loads onto the shell, redistribute shear around structural discontinuities and transfer loads at joints. The frame cap is usually a Z-section which runs around the periphery inside the stringers. The main function of frame is preservation of the circular shape against elastic instability under the compressive loads, have little constraint on the radial expansion of the shell. The frame spacing can have a substantial impact on the compressive skin panel design, the pitch between frames has a conventional value of 20 inches (0.5 meters) for transport airplanes. This value is only 38% effective in helping to reduce the maximum hoop stress in the skin. [2]

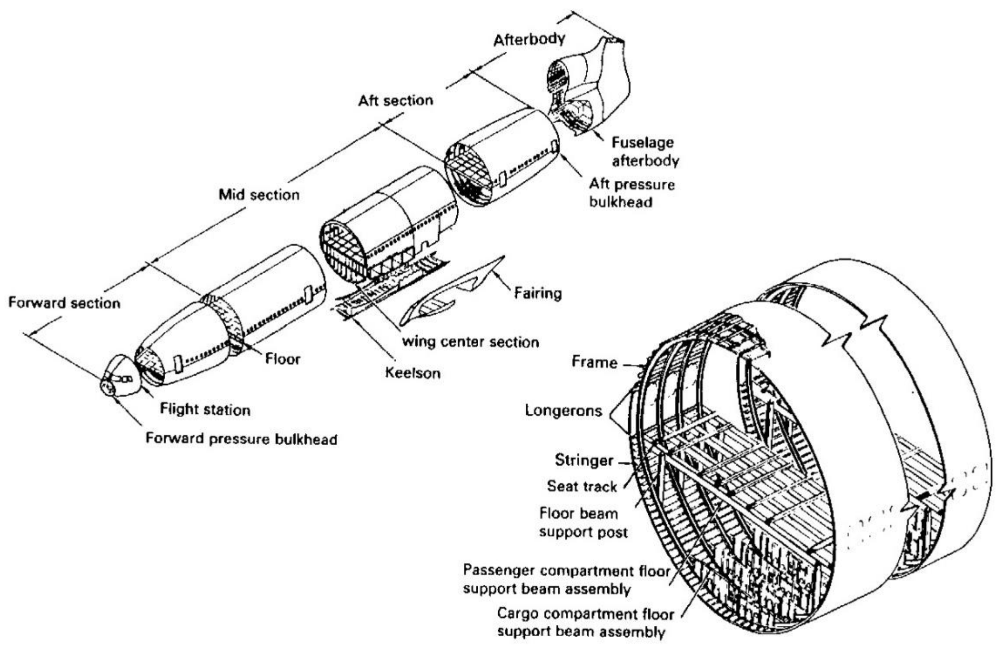


Figure 1.Semi monocoque construction [2]

A closer look at design requirement of frames suggest the shape and sizing parameters. In the crown areas, pressure loads are carried in hoop tension, the frames act as a stabilizing member to maintain the shape and to keep the skin and stiffeners from buckling under bending loads. Pressure loads combine with stabilizing loads to produce large bending moments on the non-radial contours areas of the frames. Due to pressure loads, the skin assumes a radial shape. The shear tie attachment to the skin and stringers is the shear load path between frame and fuselage skin. The shear tie holds the skin in the desired shape, and frame holds the shear ties and stringers to desired shape. The frame therefore experiences a bending load. [2]

An aircraft is basically an assembly of stiffened shell structures ranging from single cell closed section fuselage to multicellular wings and tail surfaces each subjected to bending, shear, torsional and axial loads. The smaller structures consist of thin -walled channel, T-, Z-, ‘top-hat’ or I-sections, which is used to stiffen the thin skins of the cellular components and provide support for internals loads. Structural members are known as open section beams and cellular components are termed as closed section beams. Both sections are subjected to axial, bending, shear and torsional loads. Frequently aircraft components comprise of combination of open and closed section beams. [3]

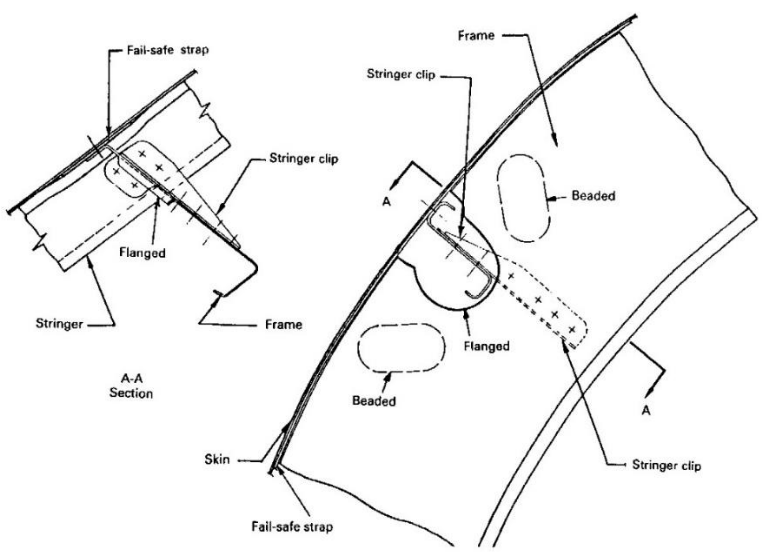


Figure 2. Stringer Frame interaction [2]

The structural layout of an airframe is dependent on the choice of the material of construction. It is desirable to consider different designs before employing alternative materials and compare them. The primary drivers in materials selection are low weight and low cost. These two criteria must be balanced, in order to achieve a desired solution. Both criteria are function of material and manufacturing processes used to make details and assembly of the components. Two fundamental classes of materials applied currently in industries are metal alloys and fiber-reinforced composites. Aluminum alloys are the most commonly used material because of their low densities, excellent range of properties, which can match to any requirements. Recently, composite, or fiber-reinforced plastics [FRP] are been used by the aircraft manufacturing industries. A composite consists of two distinct elements, one of the element is a fibrous material and other is matrix material used to reinforce the other. The most frequently used reinforcing fibers are: carbon, glass, and aramid. [4]

This thesis research deals with the analysis of fuselage frame under shear load due to impact force. The fuselage is dropped with velocity of 30 ft./sec from height of 86 inch from its center of gravity. The impact force creates a reaction force in radial direction, which generates a shear flow. This shear flow gives rise to bending moment, shear force and cross-section stress. An analysis is conducted by comparing the material change from alloy to FRPs which can withstand the generated shear flow.

Chapter 2

Thesis Objective

2.1. Thesis Requirements

The thesis involves calculation and determining the material, require to design a composite aircraft frame. This thesis requires use theoretical as well as computational calculations in determining and validating the material properties and cross section of an aircraft frame. The thesis scope is to suggest a fire-reinforced plastic [FRP] with a frame cross-section which can withstand impact loads and have factor of safety to that of industrial used material. The requirement and scope are the decision-making parameters of the thesis. These parameters consist of the thickness of the cross- section. Young's Modulus, density, strength to weight ratio of composite material.

2.2. Individual Disciplinary Analysis

The main purpose of this report is to suggest or validate a FRP which will give factor of safety approximate to that of currently used aircraft frame material i.e. alloys. The validation requires a comparative study between the materials. The materials considered are aluminum alloy [Al-7075], glass fiber-reinforced plastic [GFRP] and carbon fiber-reinforced plastic [CFRP].

The dimensional and material reference of aircraft frame is taken from a Boeing B737 a commercial aircraft. The frame is subjected to impact by dropping it at a velocity of 30 ft. / secs from a height of 86 inch from its center of gravity. The shear flow is calculated due to impact force which acts in radial direction. The frame is analyzed under static and dynamic load conditions using ANSYS workbench. The theoretical calculations are done by generating a MATLAB code.

Chapter 3

Literary Study

3.1. Aircraft Frame

The fuselage of various kinds of aircraft have significant differences in their layouts, the primary role is similar in all cases. The difference is the pressurization requirement of most passenger aircraft, which affects the fuselage volumes. The structural shape of the fuselage is close to ideal. The depth and width are approximated to match the vertical and lateral bending and the reaction of the torsion. Basically, a rectangular cross-section is advantageous in terms of maximum space utilization. It is not suitable for general commercial aircraft since substantial pressure differential is required. The stresses due to internal pressure are minimized by use of circular arcs. [4]

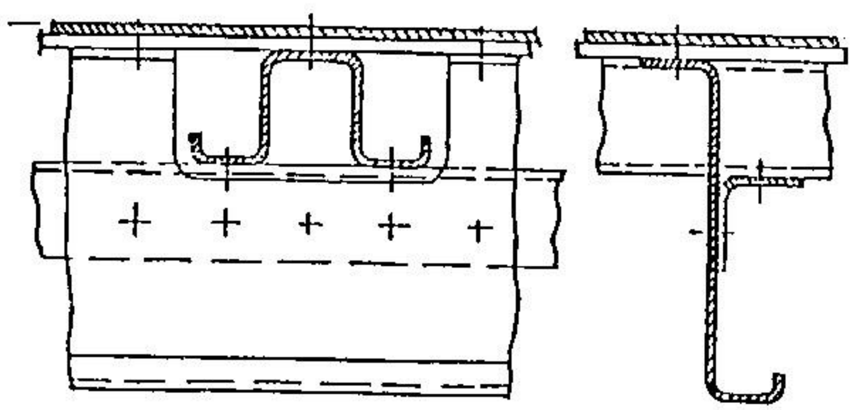


Figure 3. Frame and Stringer Cross section of B737 [2]

Frames are used to stabilize the skin-stringer elements and transmits shear loads into the structure. They also used to react the pressurization loads. Frames act as a crack stopper, it must have a direct contact with skin to do so and stringer passing through it. ‘Floating’ frames are also used in some construction, where they are only attached to stringers. Lightly loaded frames are usually manufactured as pressing with reinforced edges, whereas heavily loaded frames are built up using sheet metal components in conjunction with forged or extruded fittings. Heavily loaded frames are close to pressure cabins or fuel tanks. The heavily loaded frames are curved beams designed to evaluate local loads such as frame shear force and bending moment distribution. [4]

3.2. Loads Acting on Frames

An airplane fuselage consists of number of structural rings or closed frames. Some are light and are required to maintain the shape of the body, to provide stabilizing supports for longitudinal shell stringers. Heavy frames are observed where large concentrated loads are transferred between bodies such as tail, wing power plant, landing gear. The frames are of such shape and the load distribution, that these frames or rings undergo bending forces in transferring the applied loads to the other resisting portions of the airplane body. These bending forces produce frame stresses, such frames are statically indeterminate relative to internal resisting stress. [1]

Frames in fuselage serve the same purpose as ribs in wing structures. Fuselage frames are of the closed ring type of structure and are therefore statically indeterminate relative to internal stresses. The loads on frames are due to discontinuities in the fuselage structures, such as due to doors and windows. When external loads are applied, the frames are in equilibrium by reacting fuselage skin forces, which are usually transferred to the frames boundary by rivets which fasten fuselage skin and frame. [1] [2]

Aircraft fuselage loads on transport aircraft are as follows: [2]

1) Ultimate design conditions
   - Flight loads (acting alone)
   - Flight loads + cabin pressure
   - Cabin pressure only
   - Landing and ground loads
2) Fail-safe design conditions
   - Fail-safe flight loads only
   - Fail-safe flight loads + cabin pressure
   - Cabin pressure only
3) Fatigue
   - Fatigue loads based on flight profiles
   - Fatigue due to design flight hours
4) Special area conditions
   - Depressurization of one compartment
   - Bird strike
   - Hail strike
   - Cargo and passenger loads on floors
   - Crash loads (emergency landing)

3.3. Materials

The original aircrafts where made of mainly wood with fabric covering and steel wire bracing. The main material use today is aluminum alloy, which has improved strength. There has been a steady improvement in the metal properties. This is achieved by improving the manufacturing processes and sometimes by development of new materials. [5]

Aluminum alloys are most widely used airframe materials and have been developed to meet airframe specifications. Aluminum alloys have very good strength to density ratios, especially in the compression and bending of relatively thin plates and sections. A thin outer cladding of pure aluminum can give good corrosion resistance without any adverse effect upon facture toughness. [5] [4]

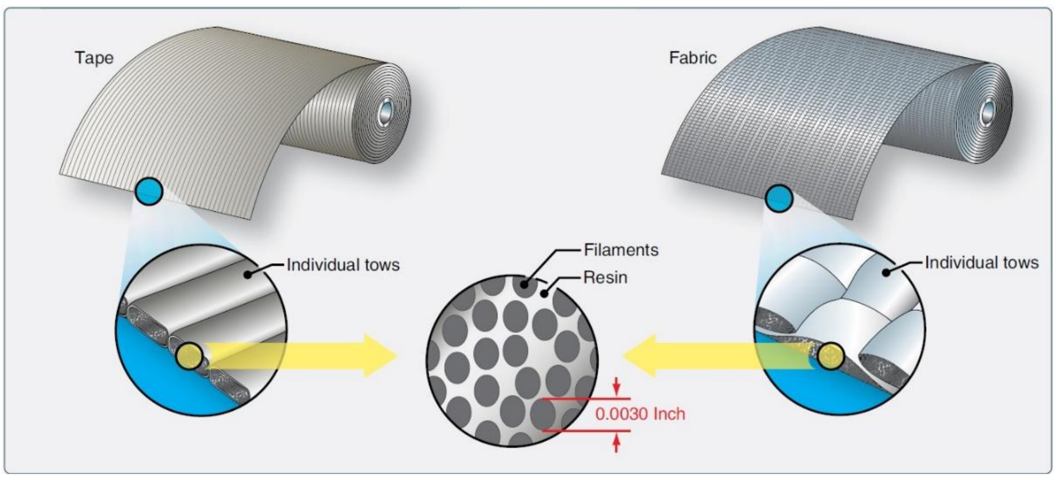


Figure 4. Fiber Reinforced plastic [6]

Airframe are mainly made up of light alloys, introduction of composite has made some exceptions. When two or more substances are used together in a structure it can be termed as a composite. The main material consists of strands of strong fibers stuck together with an adhesive. Composites are normally in a form of thin cloth or flat tapes, and can be easily formed into complex curved shapes of almost any size, giving very clean aerodynamically smooth surfaces. Other than embedding of fibers in a matrix, it is also possible to produce ply-type materials consisting of laminates of reinforced plastics or sometimes sandwich construction of metal and reinforced plastics. Such arrangements have advantages such as stiff materials, which have good acoustic damping properties as well. Careful selection and design of such materials can result in stronger, stiffer, and lighter aircraft structure than metal counterpart. [4] [5]

The composite manufacturing process used in aerospace industries are discussed in the table below.

Method	Process	Advantages	Disadvantages
Filament Winding	A tow of fibers is passed through a bath of resin and wound onto a revolving mandrel by traversing longitudinally along the axis of the rotating mandrel.	The process lends itself to automation such that cycle times and labor costs can be kept low with high reliability and quality.	The mandrel is often enclosed within the winding, if a liner of metal or polymer is used as a mandrel it may form a permanent part of the structure but it is more common that the winding is slit-off at the ends to demold the part.
Pultrusion	Fibers are drawn from a creel board and passed through a resin bath to impregnate the fibers with resin, these are then passed through a pre-die to remove any excess resin and to pre-form the approximate final shape	The operation is automated labor costs are low and the reliability and quality of components is high.	The process is generally limited to constant cross-section components, which greatly restricts applications.
Resin Transfer Molding (RTM)	Unresinated fibers are held within a closed tool cavity with a differential pressure applied to a supply of resin such that the resin permeates into the reinforcement.	The use of low added value materials (dry fibers and low viscosity resins) which do not have to be stored in freezers, thus driving	To guarantee high-quality components, the resin injection and resin flow has to be closely controlled, which requires quite advanced fluid dynamics

		down material and handling costs.	simulations and extensive testing.
Automated Fiber Placement (AFP)	Machines use a robotic fiber placement head that deposits multiple pre-impregnated tows of “slit-tape” allowing cutting, clamping and restarting of every single tow.	High productivity and handle complex geometries, allows engineers to design more efficient structures, such as integrated orthogrid or isogrid composite panels	High capital cost, laps and gaps issues, Distortion in thermoplastic composite processing
Contact Molding	In contact molding resin is manually applied to a dry reinforcement placed onto a tool surface and can be compared to gluing wall paper with a brush.	The process is highly flexible, ideal for one-off-production and requires minimal infrastructure.	The quality is dependent on the skill of the workforce, difficulty in reliably guaranteeing high-quality laminates, due to the limited external pressure void age is difficult to control.
Vac. Bag/Autoclave	These processes use pre-impregnated unidirectional plies or woven cloths, which have been partially cured or beta-staged.	Reduces the bulk factor and helps to prevent delamination between plies and controls the thickness dimension.	The productivity of autoclave molding is generally quite low since the manual lay-up, bagging and demolding cycles consume significant labor and time. The capital expenditure of autoclaves is enormous.

Table 1.Manufacturing process of aircraft composite material [6] [7] [8]

Two main reinforced plastics discussed in this report are (i) glass fiber reinforced plastics [GFRP] and (ii) carbon fiber reinforced plastics [CFRP].

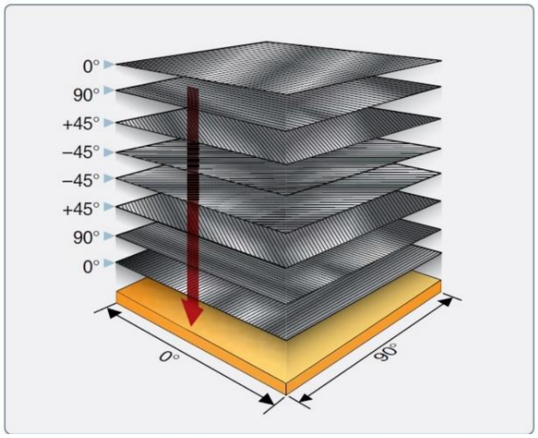


Figure 5. Quasi-Isotropic laminate [6]

Glass-fiber reinforcement has found widespread application in association with thermosetting resins, primarily of its low cost and ease of manufacturing. The two most common grades of fiberglass are ‘E’ and ‘S’. E-glass provides a high strength-to-weight ratio, good fatigue resistance, outstanding dielectric properties, retention of 50% tensile strength and excellent chemical, corrosion, and environmental resistance. S-glass offers 25% higher compressive strength, 40% higher tensile strength, 20% higher modulus and 4% lower density than E-glass. Applications of glass-fiber reinforced plastics can be found as secondary structure, such as fairings and primary structure on relatively lightly loaded aircraft. Glass reinforcement used with thin aluminum sheets to form a ply material is known as GLARE. This material has considerable potential for pressurized fuselage and overcomes the poor compression properties. [9] [4]

Carbon fibers are more expensive than glass but offers a better range of material properties. Carbon fibers are among the strongest and stiffest composite material as they have high strength to weight and stiffness to weight ratios than glass fiber. Proper selection and placement of fibers can make composites stronger and stiffer than metal equivalent at less than half weight. Carbon composite laminates offers fatigue limits more than metal alloys with superior vibration damping. Carbon fiber materials are most suitable for components where the primary load direction is defined, such as aircraft lifting surfaces. Carbon-fiber reinforcement can be sensitive to low levels of impact damage and environmental conditions such as moisture. [9] [4]

3.4. Composite Aircraft Frames

The aircraft fuselage can be constructed using materials of two distinct groups, namely metallic material, and composite materials. Several design reasons for non-construction of a non-metallic aircraft fuselage include (i) dissimilar materials prone to corrosion; (ii) the joint of a metallic member to a composite member is more complex to accomplish; (iii) high interference loads can be generated (iv) technologies involving manufacturing and fabrication of composite are different from that of metallic components. A composite aircraft would be light weight and capable of manufactured economically. An aircraft fuselage is typically constructed as a series of longitudinally spaced circumferential frame members which defines fuselage shape and a series of spaced stringer members running longitudinally with respect to the aircraft fuselage contributing to the stiffness of the skin. Frames and stringers together contribute to the robust internal structure that provide support to external skin. [10]

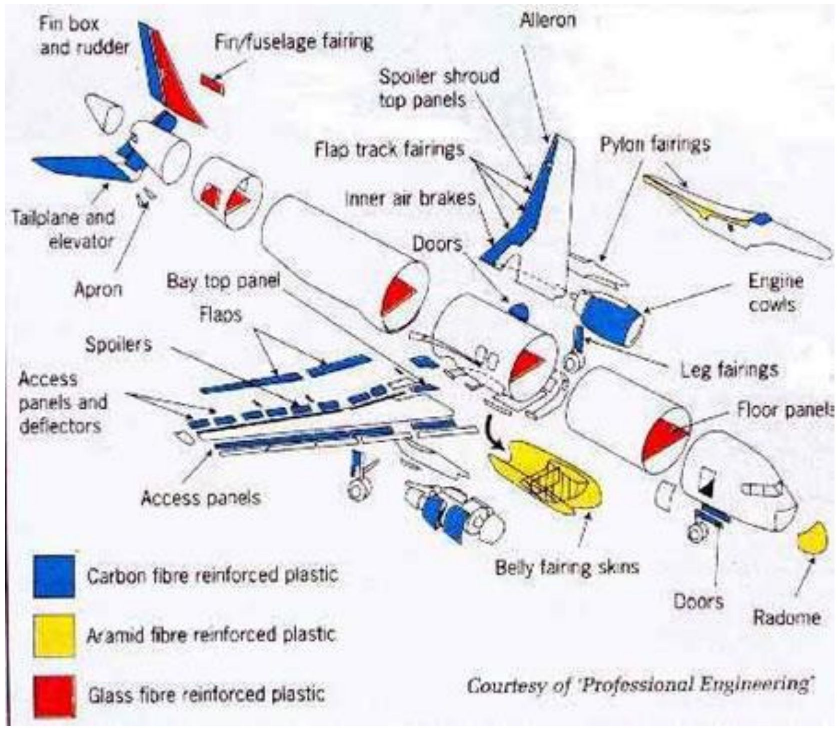


Figure 6. Composite Aircraft Components

Skin panels and frames can be integrated in three different ways. (i) the frame can be directly fastened to the skin panel using rivets or adhesive. (ii) Introduction of an additional member termed shear tie, with separate fasteners for the frame and for the stiffened skin panel. (iii) An adhesive can be used instead of rivets, frames can be attached to the stiffeners of the skin panel without either direct or indirect attachment to the skin of the stiffened panel. Whichever method is adopted it is important that the installation of frames and stringers must be accomplished without structural interference. The stringers are subjected to axial loads whereas, frames are subjected to bending loads. [10].

The requirement to save weight and cost compared to conventional aluminum alloys gave rise to production and research of composite. Composites can save up-to 25 to 35% of the weight over aluminum constructions. The main obstacles to wider use of composite material today are their high acquisition cost compared to aluminum alloy, the labor-intensive construction techniques and substantial capital costs involved in buying production equipment. The use of composite began in early 1970s under USAF funding and in late 1970s NASA instituted various programs for development of composite technologies. These programs succeeded in making primary and secondary aircraft structural designs in commercial services. Aircraft manufactures became more comfortable with the materials and more efficient techniques were developed hence lowering the cost of composite materials. [9]

Chapter 4

Methodology

4.1. General

The NASA Langley Research Center has been involved in crash dynamics research since the early 1970’s. They conducted experimental and analytical data which indicated some general trends in the failure behavior of a class of composite structures which include individual fuselage frames, skeleton subfloors with stringers and floor beams but without skin covering, and subfloors with skin added to the frame stringer arrangement. Although the behavior is complex, a strong similarity in the static/dynamic failure behavior among these structures were illustrated in the research. The analytical results were generated with a non-linear finite element code called DYCAST (Dynamic Crash Analysis of STructures) developed by Grumman Aerospace Corporation with support from NASA and FAA. NASA Langley Research Center conducted three vertical drop tests of metal aircraft sections to support transport aircraft research efforts. Two 12-foot long fuselage sections cut from an out-of-service Boeing 707 transport aircraft were considered for the drop test to measure structural, seat and occupant responses to vertical crash loads, and to provide data for non-linear finite element modelling. The structural damage for the transport aircraft structure were calculated for the 20 ft./sec drop test. [11]

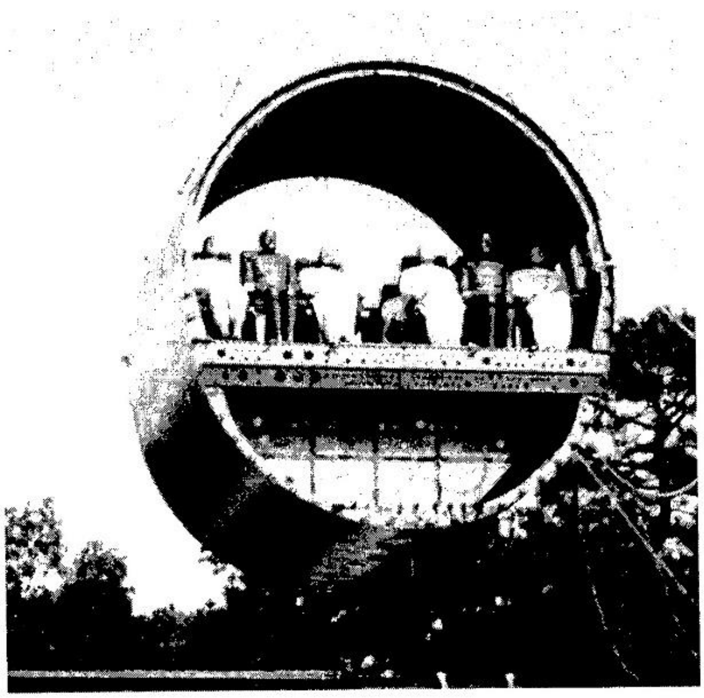


Figure 7. Drop Test conducted by NASA [11]

Similar experiments were again conducted in late 1990's and early 2000's. A 30 ft./sec vertical drop test of a 10-ft-long fuselage section of Boeing 737 (B737) was conducted at Federal Aviation Administration William J. Hughes Technical Center. Two different test were conducted. First test was performed to evaluate the structural integrity of a conformable auxiliary fuel tank mounted beneath the fuselage floor and to determine its effects on the airframe. Whereas, the second was to evaluate the effect of overhead stowage bins. A crash simulation was conducted using explicit non-linear transient dynamic code, MSC.Dytran^{TM}. [12]

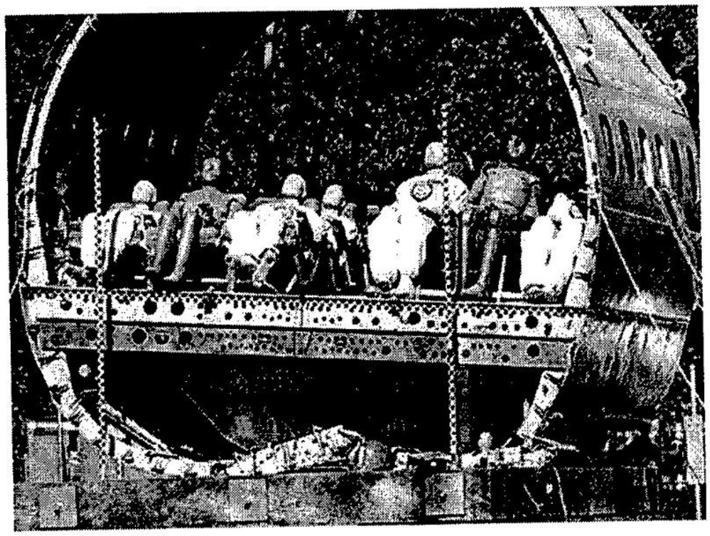


Figure 8. Results of NASA Drop Test [11]

This report consists of initial data from the above-mentioned research works. The drop test is performed only on the aircraft frame structure not on a complete aircraft fuselage section. Interaction of components like stringers, skin, floor beams are not being considered in this analysis. The frame is of circular shape than double lobe structure with a ’Z’ section. The frame is dropped from height of 86 in from its center of gravity with a velocity of 30ft. /sec. Finite element analysis on the frame is done using ANSYS explicit dynamic solver, which has a predefined explicit non-linear dynamic analysis code.

4.2. Drop Test

The basic analysis equations used for drop-test is time dependent analysis or dynamic analysis. Dynamic analysis considers global stiffness matrix, and mass matrix or lumped mass matrix. The global stiffness matrix and mass matrix equations for 1-D element are as shown

Global Stiffness Matrix

$$[\hat{k}] = \frac{AE}{L}\begin{bmatrix} 1 & -1 \\ -1 & 1 \end{bmatrix}$$

Where, A is cross section area of element; E is Young's Modulus; L is Length of element

Lumped-Mass Matrix

$$[\hat{m}] = \frac{\rho AL}{2}\begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix}$$

Where, A is cross section area of element; ρ is density; L is Length of element [13]

The time integration scheme used is central difference method. After forces have been computed at the nodes of the mesh (resulting from internal stress, contact, or boundary conditions), the nodal accelerations are derived by equating acceleration to force divided by mass. Therefore the accelerations are

$$\ddot{x}_i = \frac{F_i}{m} + b_i$$

Where, $\ddot{x}_i$ are the components of nodal acceleration (i=1,2,3); F_i are the forces acting on the nodal points; b_i are the component of body acceleration; m is the mass attributed to the node. If acceleration at time n is determined, then velocity at time n+1/2 can be found by

$$\dot{X}_i^{n+\frac{1}{2}} = \dot{X}_i^{n-\frac{1}{2}} + \ddot{X}_i^{\,n} \Delta t^n$$

The positions at time n+1 can be determined by integrating the velocities [14]

$$X_i^{n+1} = X_i^n + \dot{X}_i^{n+\frac{1}{2}} \Delta t^{n+\frac{1}{2}}$$

The time step is used in the integration process. It has been mathematically proved that the time step must be less than or equal to 2 divided by the highest natural frequency when used central difference method. The time step ensures stability of the integration method. The criterion for selection of a time step demonstrates the usefulness of determining the natural frequencies of vibration, before performing dynamic stress analysis. The approximate time step would be

$$\Delta t = \frac{L}{c_x}$$

Where, L is the element length; c_x is called longitudinal wave velocity = $\sqrt{\frac{E_x}{\rho}}$ [13]

To ensure stability and accuracy of the solution, the size of the time step used in integration is limited by CFL (Courant-Friedrichs-Lewy) condition

$$\Delta t \le f \left[\frac{h}{c}\right]_{min}$$

Where, t is the time increment; f is the stability timestep factor; h is the characteristic dimension of the element; c is the local material sound-speed in the element.

The partial differential equations to be solved in a dynamic analysis is expressed a conservation of mass, momentum, and energy in Lagrangian coordinates. For the Lagrangian formulations, the mesh moves and distorts with the material it models and conservation of mass is automatically satisfied. The density at any time can be determined by the volume of the zone and its mass.

$$\frac{\rho_0 v_0}{v} = \frac{m}{v}$$

The partial differential equations that express the conservation of momentum relate the acceleration to the stress tensor σ_{ij}

$$\rho \ddot{x} = b_x + \frac{\partial \sigma_{xx}}{\partial x} + \frac{\partial \sigma_{xy}}{\partial y} + \frac{\partial \sigma_{xz}}{\partial z}$$

$$\rho\ddot{y} = b_y + \frac{\partial\sigma_{yx}}{\partial x} + \frac{\partial\sigma_{yy}}{\partial y} + \frac{\partial\sigma_{yz}}{\partial z}$$

$$\rho\ddot{z} = b_z + \frac{\partial\sigma_{zx}}{\partial x} + \frac{\partial\sigma_{zy}}{\partial y} + \frac{\partial\sigma_{zz}}{\partial z}$$

Conservation of energy is expressed by

$$\dot{e} = \frac{1}{\rho}\left[\sigma_{xx}\dot{\varepsilon}_{xx} + \sigma_{yy}\dot{\varepsilon}_{yy} + \sigma_{zz}\dot{\varepsilon}_{zz} + \sigma_{xy}\dot{\varepsilon}_{xy} + \sigma_{yz}\dot{\varepsilon}_{yz} + \sigma_{yz}\dot{\varepsilon}_{yz} + \sigma_{zx}\dot{\varepsilon}_{zx}\right]$$

These equations are solved explicitly for each element in the model, based on the input values at the end of the previous time step. Small time increments ensure the stability and accuracy of the solution. [14]

4.3. Shear Flow Through Frame

The structural unit which transfers concentrated loads to the shells of an airplane fuselage is commonly known as a bulkhead or frames. Frames are attached to the fuselage skin continuously around their perimeters. Frames can be solid webs with stiffeners or

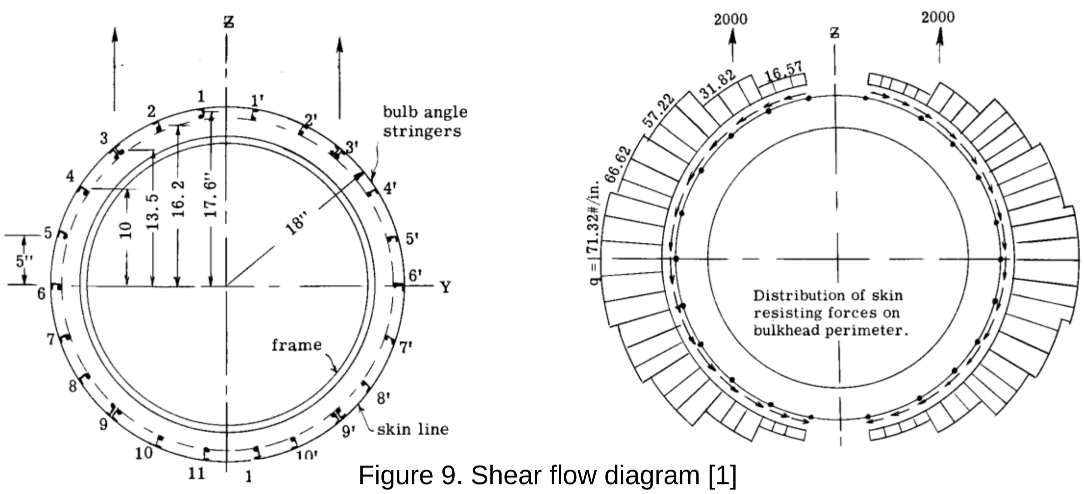


Figure 9. Shear flow diagram [1]

beads, webs with access holes, or truss structures. They transfer the loads to the skin and supply column support to stringers and re-distribute shear flow in skin. When designing a frame, the first step is to obtain the loads which acts on it and holds it in static equilibrium. Due to the shape and load distribution, these frames undergo bending forces in transferring the applied loads to the other resisting portions of the airplane body. Such frames are statically indeterminate relative to internal resisting stress and thus consideration must be given to section and physical properties to obtain solution. [1] [3]

Fuselage shells normally are symmetrical about a vertical centerline and often are loaded symmetrically with respect to the centerline. Consider a fuselage ring is loaded by a vertical load V_z on the centerline, this load is caused due to impact force and can be calculated by

$$F_{impact} = \frac{m{v_{final}}^2 - m{v_{initial}}^2}{s}$$

Where F_{impact} (V_z) is a vertical load; m is the mass of the frame; v_{intial} is the initial velocity (9.144 m/s (30fts/sec)) ; v_{final} is the final velocity of the frame at the impact; s is the distance travelled after impact.

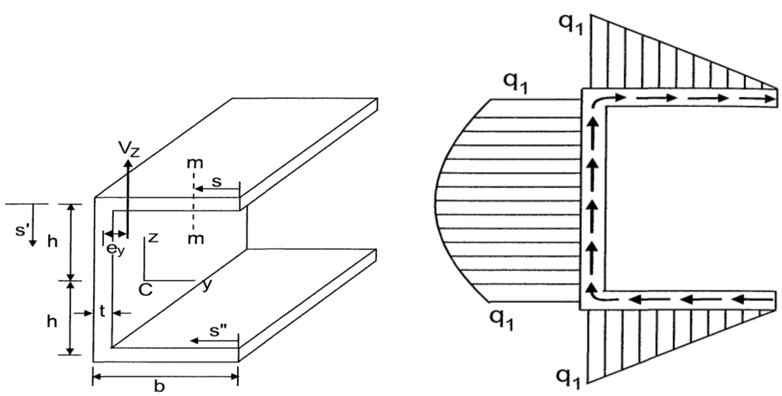
This vertical load must be in an equilibrium with the running loads q, which are applied to the perimeter of the ring.

$$q = -\frac{V_z}{I_y}\sum zA$$

Where q is shear flow thru the frame ring; I_y is the moment of inertia due to stringer on the frame $I_y = A\sum z^2$; z is the location of stringers with respect to center line; A is cross section area of stringer. A MATLAB code is generated to calculate the shear flow for Al-7075 and Fiber reinforced plastics over the frame ring. The comparative results are shown in Appendix – C. [1]

4.4. Cross Section Shear Flow

An aircraft is basically an assembly of stiffened shell structures ranging from the single cell closed section fuselage to multicellular wings. Each surface is subjected to bending, shear, torsional and axial loads. Some structures consist of thin walled channels, T- Z- top hat or I sections, which are used to stiffen the thin skin. Structural members such as these are known as open section beams, while the cellular components are termed as closed section beams.; both types of beams are subjected to axial, bending, shear and torsional loads.



[1] [3]

Figure 10. Cross section Shear

The shear flow for open section is calculated by

$$q_s = -\left(\frac{S_x I_{xx} - S_y I_{xy}}{I_{xx} I_{yy} - I^2{}_{xy}}\right) \int_0^s txds \; - \left(\frac{S_y I_{yy} - S_x I_{xy}}{I_{xx} I_{yy} - I^2{}_{xy}}\right) \int_0^s tyds$$

Where, q_s is the shear flow for the open section; Sx is Shear Force in x- direction; Sy is Shear Force in y-direction; Ixx is Moment of Inertia in X direction; Iyy is Moment of Inertia in Y direction; Ixy is polar moment of inertia.; t is the thickness of the section; x and y are the location from C.G; s is the length of the surface / flange

The stress flow for open section is calculated by

$$\sigma_s = \left(\frac{M_y I_{xx} - M_x I_{xy}}{I_{xx} I_{yy} - I^2{}_{xy}}\right) x + \left(\frac{M_x I_{yy} - M_y I_{xy}}{I_{xx} I_{yy} - I^2{}_{xy}}\right)$$

Where, σ_s is the stress flow thru the open section; Mx is the moment about x axis; My is the moment about y axis.

The Fiber Reinforced Plastics or composites the equation for calculating shear flow and stress flow thru a cross section becomes a function of the Young's Modulus in axial-direction. The second moment of area includes the laminate value of Young's modulus, E_{zi}

$$I'_{XX} = \int_A E_{Z,i} Y^2 dA\ ,\ I'_{YY} = \int_A E_{Z,i} X^2 dA,\ I'_{XY} = \int_A E_{Z,i} XY dA$$

The shear flow for open section is calculated by

$$q_s = -E_{Z,i}\left[\left(\frac{S_x I'_{XX} - S_y I'_{XY}}{I'_{XX} I'_{YY} - {I'_{XY}}^2}\right)\int_0^s txds\ + \left(\frac{S_y I'_{YY} - S_x I'_{XY}}{I_{xx} I'_{YY} - {I'_{XY}}^2}\right)\int_0^s tyds\right]$$

The stress flow for open section is calculated by

$$\sigma_s = E_{Z,i}\left[\left(\frac{M_y I'_{XX} - M_x I_{xy}}{I'_{XX} I'_{YY} - {I'_{XY}}^2}\right) x + \left(\frac{M_x I'_{YY} - M_y I_{xy}}{I'_{XX} I'_{YY} - {I'_{XY}}^2}\right) y\right]$$

The shear and stress flow for cross section is calculated by generating a MATLAB code for both Al-7075 and Fiber Reinforced Plastics. The tables and figures are as shown in Appendix -C. [3] [1]

4.5. Finite element Analysis

4.5.1 Geometry

The frame dimension assumed for the analysis is of Boeing B737 the dimensions are approximated based on the reference dimensions. For analysis, the frame is considered as a circular shape than a double lobe which it usually is. The cross section of ring is of open section ‘Z’ section.

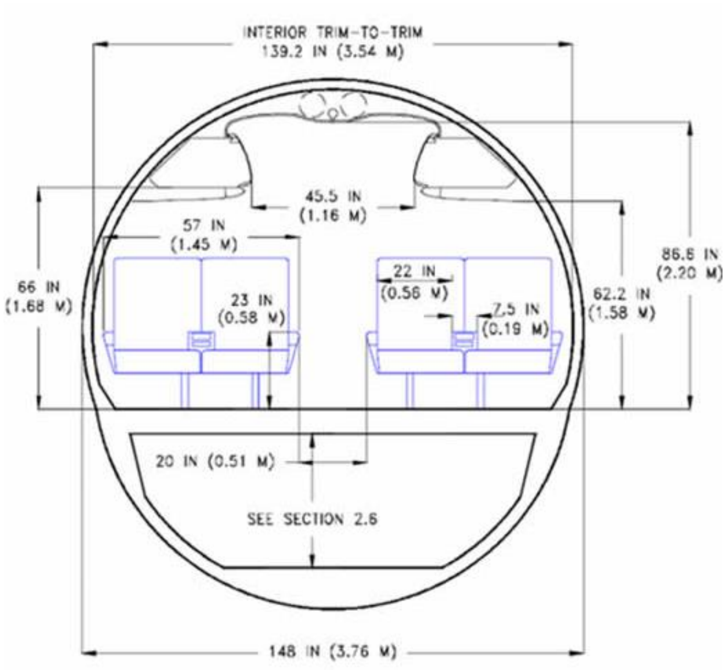


Figure 11. Frame section of B737

Dimensions:	
H1	0.036 in
H2	0.018 in
H3	0.91929 in
H4	1.7087 in
H5	0.036 in
V10	0.036 in
V11	0.2874 in
V6	3.5433 in
V8	0.036 in

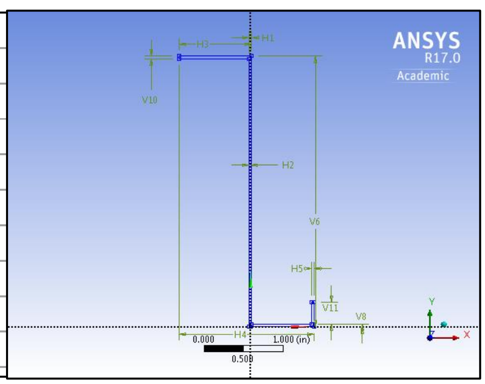


Figure 12. Dimension of Z- Section

4.5.2 Material property

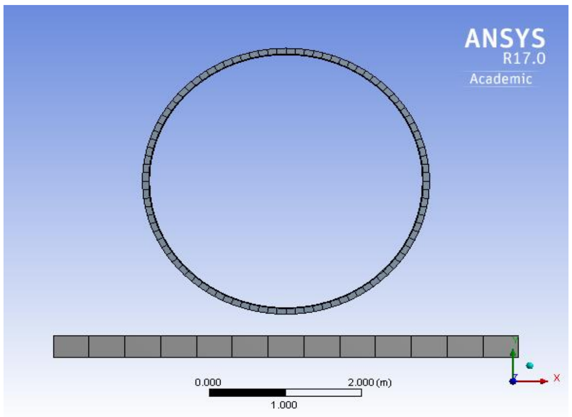
The frame is first considered to be of Aluminum Al7075, properties shown in Appendix -A. For drop test the impact surface is considered as shown in the table. The frame is analyzed with change in material from alloy to fiber reinforced plastics. The test id ran for carbon unidirectional (tape), carbon fabrics, E-glass Unidirectional (Tape) and E-glass fabric. The matrix used in this fiber composite is epoxy. The volume fraction is of 0.6 for tape and 0.5 for fabric. The material properties are shown in appendix A. the composites are placed in a quasi-isotropic laminate structure [0/90/45/-45/-45/45/90/0] with a ply thickness of 0.045 inch each. The laminate structure is modelled using ANSYS composite PrepPost.

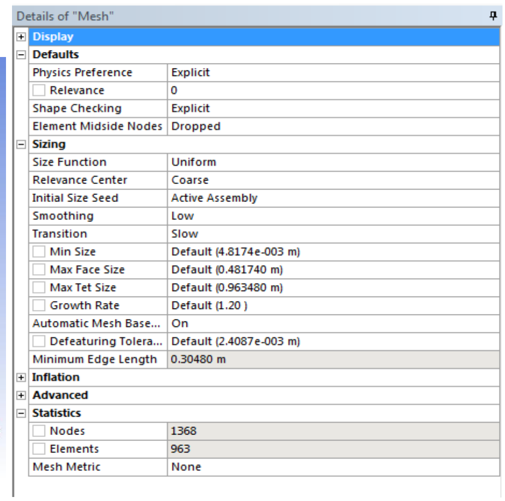
Material Property – Impact Surface		
Physical Properties	Metric	English
Density	7.99 g/cc	0.289 lb./in^3
Young's Modulus	62.05 GPa	90000 ksi
Poisson's Ratio	0.3	0.3

Table 2. Material Property of Impact Surface [12]

4.5.3 Meshing

The solution timing for a dynamic analysis is mainly a function of element size and shape. To reduce the solution time. The frame body was constructed as a shell element by specifying the line element division of 240. The base impact surface is considered as solid element. The total number of element and mesh setting for dynamic analysis is as shown in the figure.





Details of "Mesh"

Display	
Defaults	
Physics Preference	Explicit
Relevance	0
Shape Checking	Explicit
Element Midside Nodes	Dropped
Sizing	
Size Function	Uniform
Relevance Center	Coarse
Initial Size Seed	Active Assembly
Smoothing	Low
Transition	Slow
Min Size	Default (4.8174e-003 m)
Max Face Size	Default (0.481740 m)
Max Tet Size	Default (0.963480 m)
Growth Rate	Default (1.20)
Automatic Mesh Base...	On
Defeaturing Tolera...	Default (2.4087e-003 m)
Minimum Edge Length	0.30480 m
Inflation	
Advanced	
Statistics	
Nodes	1368
Elements	963
Mesh Metric	None

Figure 13. Mesh Setting

During the static analysis to calculate shear flow effect generated due to impact, the frame is considered as beam element. The frame beam is divided in to 48 division. This is based on the location of each stringer.

4.5.4 Boundary Conditions

The boundary conditions for dynamic analysis is as shown in figure. The only load acting on the frame is the velocity load of 30 ft/sec. in negative Y direction. The frame is assumed to be dropped with the velocity of 30 ft./sec from a height of 86 inch from its center of gravity. The impact surface is fixed in all degree of freedom (DOF). The frame geometry consists of 48 points to represent location of stringers. The stringers are fixed in Z-direction to lock the instability caused due to impact. [12] [15]

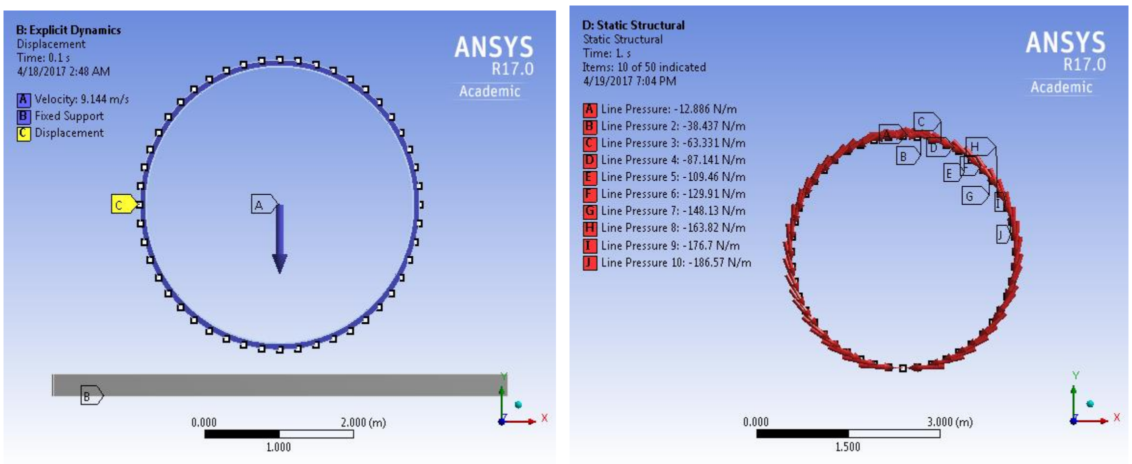
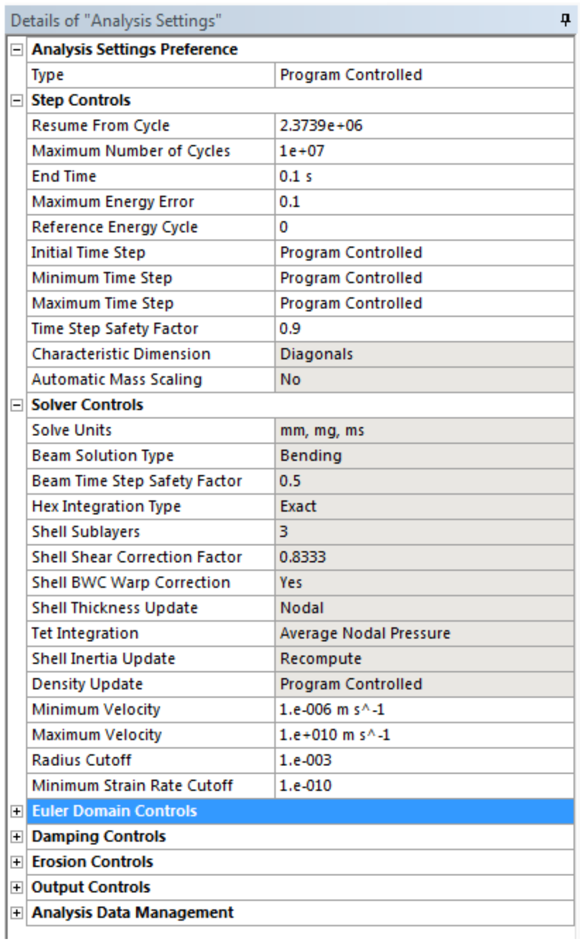


Figure 14. Boundary Conditions

For static analysis, each element is applied with line pressure which is the shear generated due to impact. This calculation is done using analytical formulation. The frame is fixed at a point of impact and other stringers are fixed in Z direction.

4.5.5 Analysis Setting

Dynamic analysis is of nonlinear transient analysis. the total time is 0.1 sec. the settings are as shown. The dynamic analysis is done using AUTODYN settings, which is solver for explicit dynamics module in ANSYS. The static structural analysis is the basic structural solver available in ANSYS Workbench.



Details of "Analysis Settings"

Analysis Settings Preference	
Type	Program Controlled
Step Controls	
Resume From Cycle	2.3739e+06
Maximum Number of Cycles	1e+07
End Time	0.1 s
Maximum Energy Error	0.1
Reference Energy Cycle	0
Initial Time Step	Program Controlled
Minimum Time Step	Program Controlled
Maximum Time Step	Program Controlled
Time Step Safety Factor	0.9
Characteristic Dimension	Diagonals
Automatic Mass Scaling	No
Solver Controls	
Solve Units	mm, mg, ms
Beam Solution Type	Bending
Beam Time Step Safety Factor	0.5
Hex Integration Type	Exact
Shell Sublayers	3
Shell Shear Correction Factor	0.8333
Shell BWC Warp Correction	Yes
Shell Thickness Update	Nodal
Tet Integration	Average Nodal Pressure
Shell Inertia Update	Recompute
Density Update	Program Controlled
Minimum Velocity	1.e-006 m s^-1
Maximum Velocity	1.e+010 m s^-1
Radius Cutoff	1.e-003
Minimum Strain Rate Cutoff	1.e-010
Euler Domain Controls	
Damping Controls	
Erosion Controls	
Output Controls	
Analysis Data Management	

Figure 15. Dynamic Analysis setting

4.5.6 Results

The results are displayed in tables and charts in Appendix B and Appendix C. the time of impact for all the materials is same i.e 0.035 sec. Since the impact velocity if different due to effect of mass. The vertical forces and de-acceleration generated are also different for different materials. The maximum de-acceleration generated is by carbon tape since the Young's modulus is very high.

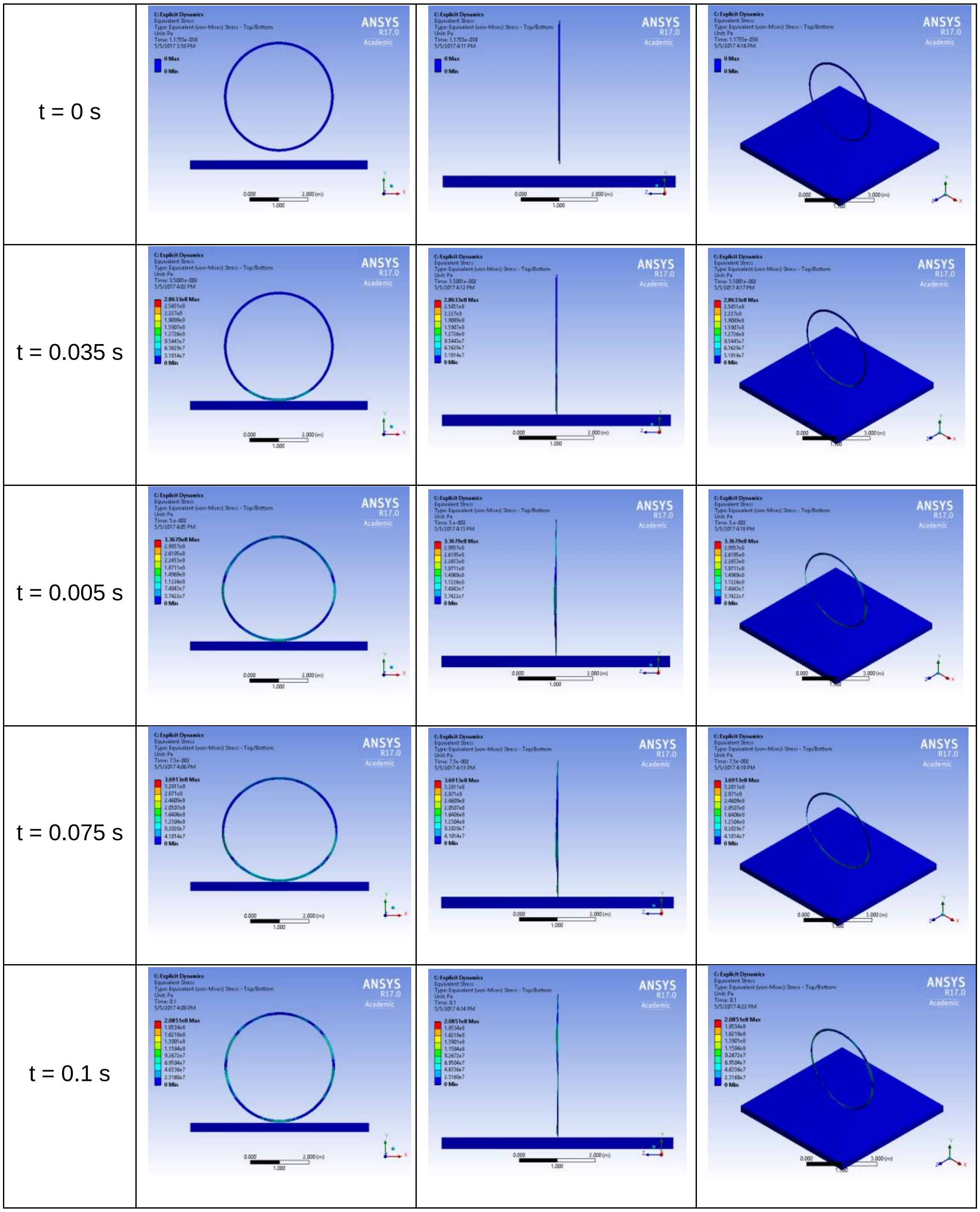


Figure 16. Images showing stress variation wrt time in front, side, and isometric view for Al-7075 T6

Chapter 5

Conclusion

The results suggest a good comparative material with that of current industrial material. E-glass tape and Carbon fabric gives better results or approximate results similar to that of Aluminum Al-7075. Factor of safety for Carbon fiber fabric is higher, but fails in Z direction. Carbon fiber fabric generates less stress for given load condition. Carbon fabric can be used as composite in aircraft structure frame with more analysis in stacking to increase strength. The mass by the application of composite material decreases significantly. Carbon tape can absorb more energy during impact hence, the high de-acceleration.

This conclusion can bring some more future work that can be done on analysis of composite frames. Optimization of cross-section or cross section thickness can be made. The change in cross section can increase the strength of composite in normal direction where it is currently failing. Change in manufacturing process like changing fiber architecture by adding fibers in Z- direction or winding a fiber around a fabric. Variation in fiber volume percentage can bring some significate change in material properties. Use of multiple material stacking like hybrid, or fiber metal laminate can increase the strength/ weight ratio of the composite frame.

Appendix A

Material properties

Table 3.Material Properties of Alloy and FRPs [15]

Physical Properties	Symbol	Units	Al-7075T6	Std CF Fabric	Eglass Fabric	Std CF UD	E glass UD
Young's Modulus 0°	E1	GPa	71.7	70	25	135	40
Young's Modulus 90°	E2	GPa		70	25	10	8
In-plane Shear Modulus	G12	GPa	26.9	5	4	5	4
Major Poisson's	v12		0.33	0.1	0.2	0.3	0.25
Yield Tensile Strength	Y	MPa	503				
Ult. Tensile Strength 0°	Xt	MPa	572	600	440	1500	1000
Ult. Comp. Strength 0°	Xc	MPa		570	425	1200	600
Ult. Tensile Strength 90°	Yt	MPa		600	440	50	30
Ult. Tensile Strength 90°	Yc	MPa		570	425	250	110
Ult. In-plane Shear	S	MPa	331	90	40	70	40
Ult. Tensile Strain 0°	ext	%		0.85	1.75	1.05	2.5
Ult. Comp. Strain 0°	exc	%		0.8	1.7	0.85	1.5
Ult. Tensile Strain 90°	eyt	%		0.85	1.75	0.5	0.35
Ult. Comp. Strain 90°	eyc	%		0.8	1.7	2.5	1.35
Ult. In-plane shear strain	es	%		1.8	1	1.4	1
Density		g/cc	2.81	1.6	1.9	1.6	1.9

Fibres @ 0° (UD), 0/90° (fabric) to loading axis, Dry, Room Temperature, Vf = 60% (UD), 50% (fabric)

Appendix B

Dynamic Analysis Results

Material	Al7075-T6		Eglass UD		Eglass Fabric		Carbon UD		Carbon Fabric	
Type	Total Velocity	Total Acceleration	Total Velocity	Total Acceleration	Total Velocity	Total Acceleration	Total Velocity	Total Acceleration	Total Velocity	Total Acceleration
Minimum	7.979 m/s	0. m/s²	8.6247 m/s	0. m/s²	9.144 m/s	0. m/s²	9.144 m/s	0. m/s²	9.144 m/s	0. m/s²
Maximum	15.635 m/s	1.2564e+005 m/s²	17.935 m/s	2.4855e+005 m/s²	18.933 m/s	2.2535e+005 m/s²	33.166 m/s	1.5641e+007 m/s²	16.957 m/s	1.5661e+005 m/s²

Table 4. Velocity and Acceleration results

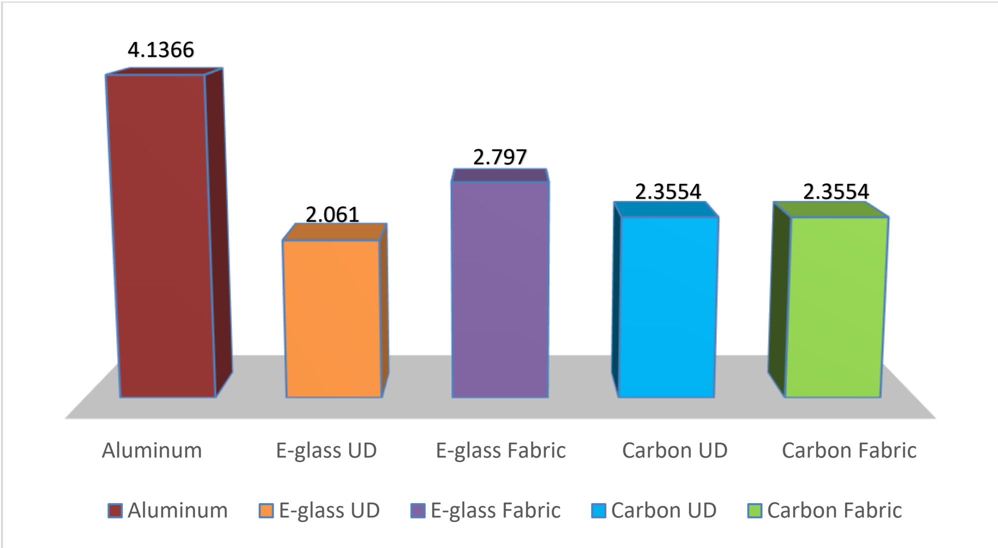


Figure 17. Mass Bar Chart

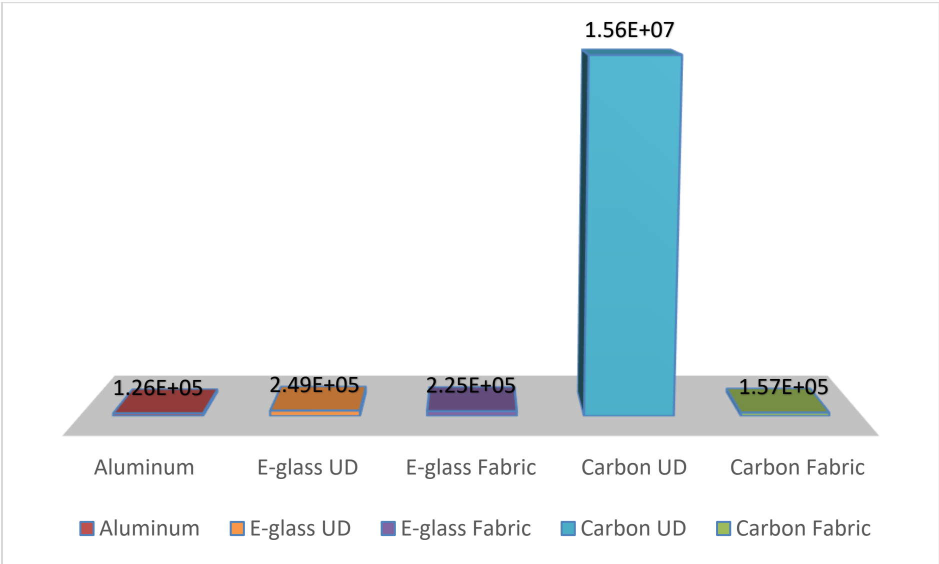


Figure 18. Acceleration Bar Chart

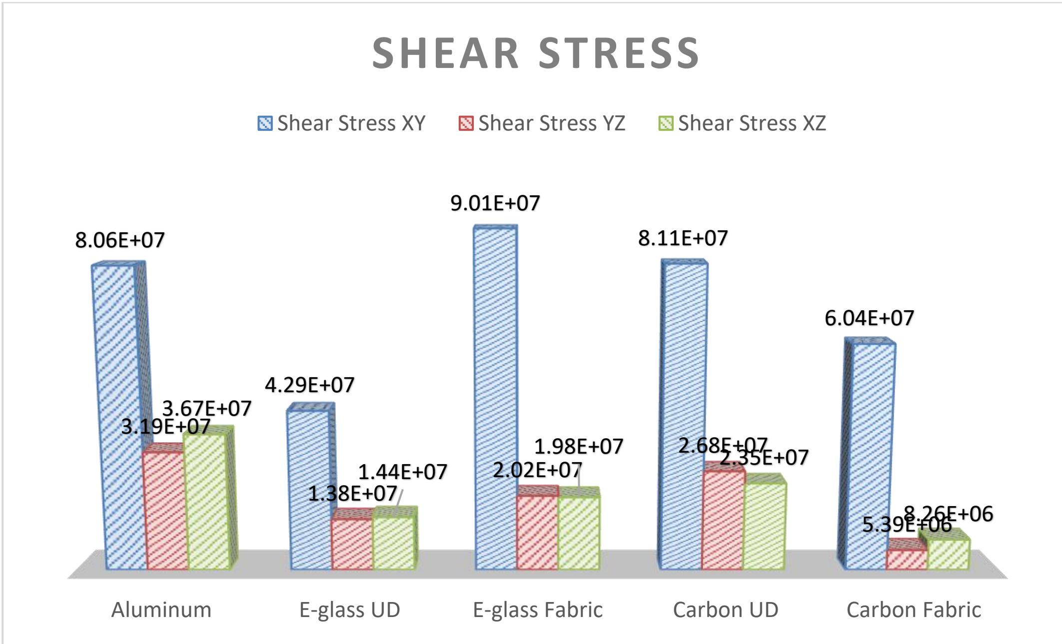


Figure 19.Shear Stress Bar Chart

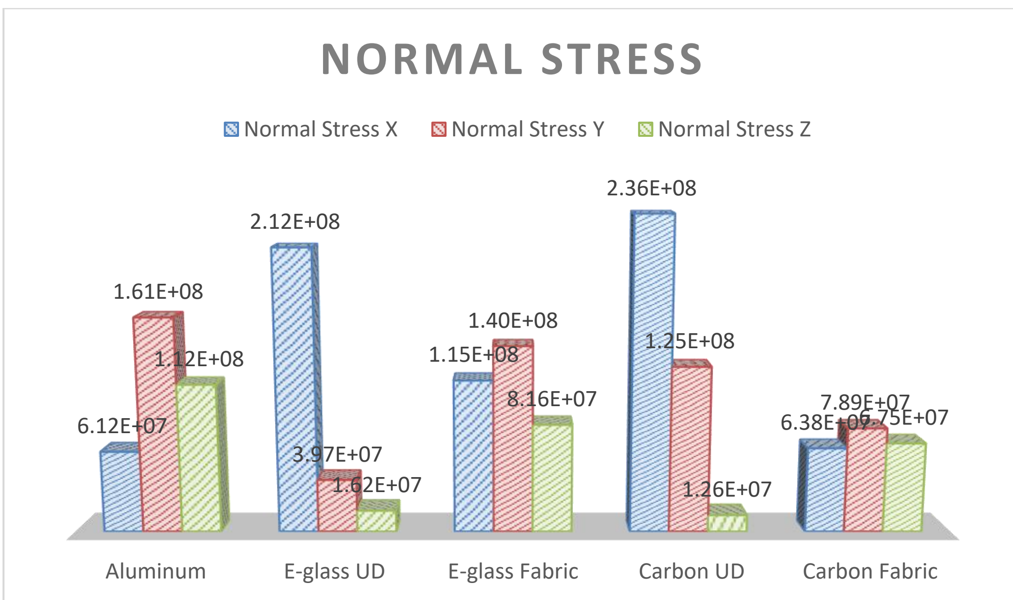


Figure 20. Normal Stress Bar Chart

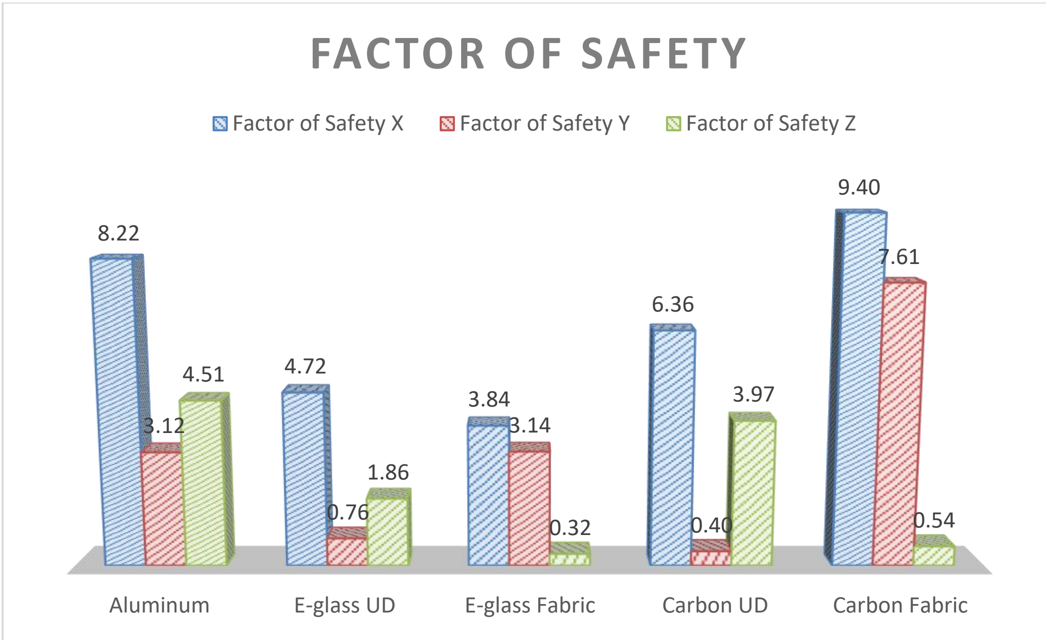


Figure 21. Factor of Safety Bar Chart

Appendix C

Static Analysis Results

Type	Directional Bending Moment		Directional Shear Force		Directional Torsional Moment
Orientation	Y Axis	Z Axis	Y Axis	Z Axis	
Aluminium Al-7075 T6					
Minimum	-527.85 N·m	-130.4 N·m	-440.54 N	-548.75 N	-0.55044 N·m
Maximum	195.87 N·m	0.41948 N·m	440.54 N	548.75 N	0.55044 N·m
E Glass UD (Tape)					
Minimum	-228.03 N·m	-56.494 N·m	-193.95 N	-237.49 N	-4.7775e-002 N·m
Maximum	84.981 N·m	3.4662e-002 N·m	193.93 N	237.48 N	4.7768e-002 N·m
E Glass Fabric					
Minimum	-93.848 N·m	-23.292 N·m	-79.66 N	-97.683 N	-3.6606e-002 N·m
Maximum	34.928 N·m	2.7153e-002 N·m	79.686 N	97.648 N	3.6617e-002 N·m
Carbon UD (Tape)					
Minimum	-1572.9 N·m	-383.22 N·m	-1318.9 N	-1640.5 N	-0.1194 N·m
Maximum	588.25 N·m	8.0025e-002 N·m	1318.9 N	1640.5 N	0.1194 N·m
Carbon Fabric					
Minimum	-433.77 N·m	-107.09 N·m	-368.08 N	-451.96 N	-6.2871e-002 N·m
Maximum	161.83 N·m	4.4721e-002 N·m	368.08 N	451.96 N	6.2871e-002 N·m

Table 5. Shear Flow Analysis Result

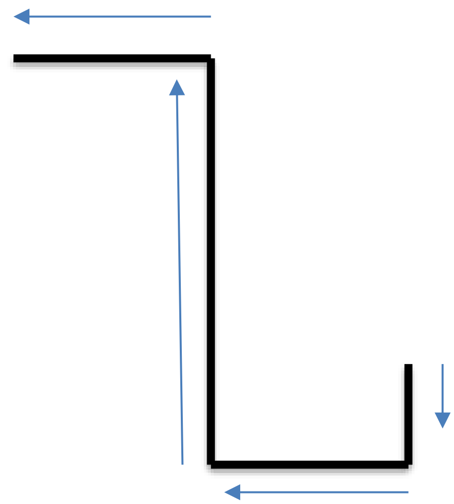


Figure 22. Cross section shear flow

	Shear Flow (N/m)				
	Aluminium	E-glass UD	E-glass Fabric	Carbon UD	Carbon Fabric
1	-9.64E+03	-4.24E+03	-1.74E+03	-2.89E+04	8.05E+03
2	2.01E+04	8.80E+03	3.62E+03	6.01E+04	-1.67E+04
3	-8.88E+04	-3.87E+04	-1.59E+04	-2.65E+05	7.36E+04
4	2.74E+04	1.21E+04	4.95E+03	8.21E+04	-2.29E+04

Table 6. Cross-Section Shear Flow due to Shear force

	Stress Flow (Pa)				
	Aluminium	E-glass UD	E-glass Fabric	Carbon UD	Carbon Fabric
1	-4.19E+08	1.82E+08	7.44E+07	1.23E+09	3.44E+08
2	-5.05E+08	2.18E+08	9.00E+07	1.49E+09	4.15E+08
3	5.10E+08	2.21E+08	9.08E+07	1.51E+09	4.19E+08
4	5.41E+08	2.34E+08	9.64E+07	1.60E+09	4.45E+08
5	4.73E+08	2.05E+08	8.44E+07	1.39E+09	3.89E+08

Table 7.Cross-Section Stress Flow due to Bending Moment

Appendix D

MATLAB Code

Program for shear flow in frame

```
%% Deflection of Fuselage Frame
% R  = Frame Centroid Radius (m)
% As = Area of Stringer (m^2)
% Iy = Moment of Inertia about Y axis (m^4)
% y  = Coordinate location of stringer in Y (m)
% ns = number of stringers
% q  = Shear in the Frame (N/m)
% Vz = Vertical Dynamic Load on the Frame (N)
% Fz = Reaction Force
% v1 = Velocity of frame (m/s)
% v2 = Velocity of frame (m/s)
% m  = Mass of frame (kgs)
% dt = Time at impact (secs)
% g  = Acceralation due to gravity (m/s^2)
% s  = Distance travelled after impact (m)
%% Fuselage Data
v1=9.144; v2=17.935; m=2.061; g=9.807; s=0.5; dt=0.0035;
Vz=(m*((v2^2)-(v1^2)))/(2*s)
%% Frame Data
R=1.833; ns=48; As=1.54079E-4;
%% Y Location in Coordinate
y=zeros(13,1);
i=1;
x=360/ns;
for th = 90:-x:0
    y(i,1)= R*sind(th);
    i=i+1;
end
%% Moment of Inertia
I=ones(1,13);
Y=zeros(13,1);
Y(1,1)=(As/2)*(y(1,1)^2);
for i=2:1:12
    Y(i,1)=As*(y(i,1)^2);
end
Y(13,1)=(As/2)*(y(13,1)^2);
Iy=4*(I*Y)
%% Shear Flow
q=zeros(13,1);
q(1,1)=-(Vz/Iy)*(y(1,1)*(As/2));
for i=2:1:12
    q(i,1)=q(i-1,1)-((Vz/Iy)*(y(i,1)*As));
end
q(13,1)=q(12,1)-((Vz/Iy)*(y(13,1)*(As/2)))
%% Fz
F=zeros(12,1);
for i=1:1:12
    F(i,1)=-q(i,1)*(y(i,1)-y(i+1,1));
end
I=ones(1,12);
Fz=4*I*F
```

Program for shear and stress flow in cross -section

```
%% Deflection of Fuselage Frame
% A= cross section Area (m^2)
% Ixx= Area Moment of Inertia in XX(m^4)
% Iyy= Area Moment of Inertia in YY(m^4)
% E= Youngs Modulus(N/m^2)
% G= Shear Modulus (N/m^2)
% v= Poissons Ratio
% p= Density (kg/m^3)
% t= thickness of crosssection (m)
% n= Number of flanges
% x= X-Coordinates of flanges wrt C.G
% y= Y-Coordinates of flanges wrt C.G
% s= distance (m)
% SFx=Shear Force in X direction (N)
% SFy=Shear Force in Y direction (N)
% Mx= Moment in X direction (N-m)
% My= Moment in Y direction (N-m)
%% Material Data
E=71.7E9;G=26.9E9;v=0.33;p=2810;
%% Cross Section Data
A=0.00021782;Ixx=1.4334E-7;Iyy=8.9166E-9;Ixy=-2.4524e-8;
CGy=0.043649;CGx=-0.00067673;t=0.0009144;
n=4;
x=[0.0217;0.0217;0;0;-0.0217]; y=[-0.05095;-0.043649;-0.043649;0.046351;0.046351];
%% Bending Moments Due to Shear Flow
Mx=-527.85; My=-130.4;
%% Shear Force Due to Shear Flow
SFx=440.54; SFy=548.75;
%% Cross-Section Stress Flow
Sz=zeros(n+1,1);
for i=1:1:n+1
    Sz(i,1)=(((My*Ixx)-(Mx*Ixy))/((Ixx*Iyy)-Ixy^2))*x(i,1)+(((Mx*Iyy)-(My*Ixy))/↙
((Ixx*Iyy)-Ixy^2))*y(i,1);
end
Sz
%% Cross-Section Shear Flow
qz=zeros(n,1)
for i=2:1:n+1
    s=sqrt(((x(i,1)-x(i-1,1))^2)+((y(i,1)-y(i-1,1))^2))
    qz(i-1,1)=-((((SFx*Ixx)-(SFy*Ixy))/((Ixx*Iyy)-Ixy^2))*t*x(i,1)*s)-((((SFy*Iyy)-↙
(SFx*Ixy))/((Ixx*Iyy)-Ixy^2))*t*y(i,1)*s);
end
qz
```

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Aditya Milind Dandekar, a graduate student in Aerospace engineering at University of Texas at Arlington, Texas, started his graduate studies in Fall 2015. A Bachelor of Engineering in Aeronautical Engineering from The Aeronautical Society of India, New Delhi, India. He commenced a professional career with an internationally recognized institute, CADD Centre Training Services, working as a CADD Engineer to train undergraduate and graduate level Mechanical Engineering students and corporate professionals on CAD, CAE, and PPM software in Fall 2011. His field of interest are design and analysis of aircraft component. He has worked on notable coursework projects like **‘***X-43 C+: Geometry Synthesis for Run-way Landing’***,** *‘Potential Flow over an NACA0012 Airfoil Using Finite Difference Method’.* His knowledge and expertise in field of design and analysis of components using CAD and CAE systems helped him during this project work. He plans to use this experience in his future carrier to work on several similar industrial projects.