DEVELOPMENT OF GEODESIC COMPOSITE AIRCRAFT STRUCTURES

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Abstract

The paper is concerned with development of geodesic composite aircraft structures whose stiffness and strength are provided by a system of ribs made of unidirectional carbon-epoxy composite materials by continuous wet filament winding. Design, analysis, fabrication and weight efficiency of geodesic composite structures are discussed along with the efficiency of traditional metal and composite stringer aircraft structures.

1 Stringer and Sandwich Design Concepts

Composite materials are widely used in airframe structures of modern commercial airplanes mainly within the framework of two basic structural design concepts - stringer stiffened and sandwich structures (Fig.1(a),(b)).

(a) (b) (c)

Fig. 1. Stringer (a), sandwich (b) and lattice (c) composite structures

Both concepts are based on the idea of load-carrying skin, whereas the ribs in stiffened structures and the core in sandwich structures provide the proper bending stiffness and the resistance to buckling under compression and shear, whereas the skin takes the main load. It is important that in composite stiffened and sandwich structures the skin and the ribs are not unidirectional and have laminated structures consisting of unidirectional plies with various orientation angles.

Experimental stress-strain diagram for traditional quasi-isotropic (0/90/+45/-45) symmetric composite material with HTS carbon fibers and epoxy resin is shown in Fig.2. Material is made by pressing of plies manufactured by wet filament winding on a flat mandrel and is characterized with specific gravity 1.51, fiber volume fraction 0.55 and porosity 1.65%. As follows from Fig.1, the effective material modulus is 44GPa which is less than the modulus of aluminum.

Tensile strength (experimental ultimate stress (439MPa) is lower than the corresponding theoretical value – 601MPa due to the edge effects) looks rather high, but the diagram has two knees at stresses at 171MPa and 203MPa (see the red circles in Fig.2). Theoretical description of the diagram [1] shows that the first knee corresponds to the fracture of the matrix between the fibers in the 90^o plies (the corresponding theoretical stress is 179MPa), whereas the second knee corresponds to the matrix failure in 45^o plies (the corresponding theoretical value is 211MPa).

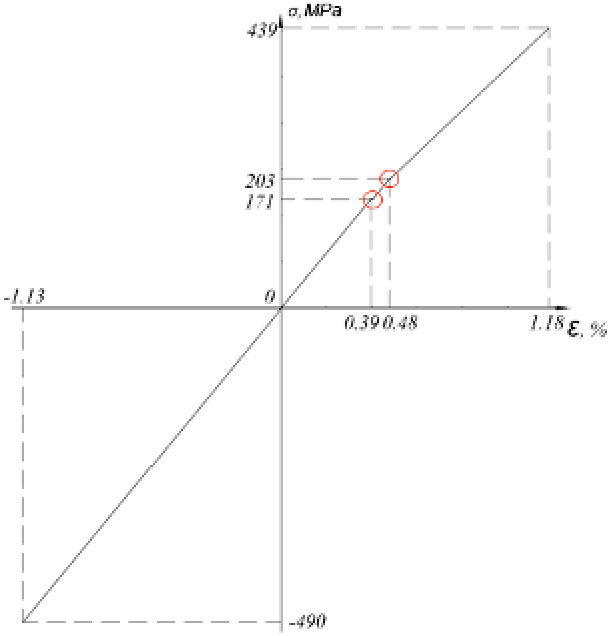


Fig. 2. Experimental stress-strain diagram for quasi-isotropic carbon-epoxy laminate

Naturally, the cracks in the matrix cannot be allowed in commercial airplanes under flight loads and we can conclude that the allowable strain of the material cannot exceed 0.4% (Fig.2), whereas the fiber ultimate strain is 1.6%. Thus, the fibers are underloaded by the factor of 4. Note that for modern toughened composites the allowable strain is higher, but it is still much lower than the fiber ultimate strain. It should be also taken into account that low temperature, moisture, aging and cyclic loading can significantly reduce it.

Under compression, the matrix in 45^o fails at stress making 231MPa (this is the theoretical value because the knee is not visible on the diagram in Fig.2). The experimental ultimatestress (490MPa) is close to the corresponding theoretical result (489MPa). The allowable compressive stress is further significantly reduced by impact delamination of the material. Fig.3 demonstrates the most dangerous barely visible delamination induced by impact with the energy 35J in 8mm thick quasi-isotropic laminate [2]. Delamination area is bounded by the white line and the depth of a barely visible dent is 0.25mm. Such delamination reduces compressive strength of the material under study by 76%

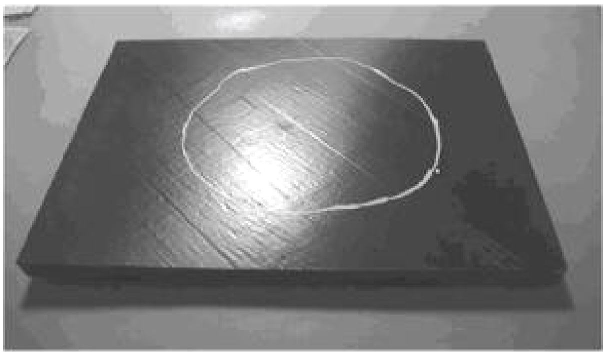


Fig. 3. Delamination area in quasi-isotropic 8 mm thick carbon-epoxy laminate after impact with energy 35 J

One of the most serious problems in composite stringer structures is associated with bolted joints. First, the holes which cut the fibers considerably reduce material strength. For example, the standard [3,4] 6.3mm diameter hole reduces the strength of the quasi-isotropic composite under study approximately by 25% under tension and by 45% under compression. Second, relatively low bearing strength of composite materials significantly reduces the force that can be transferred through the bolt. Fig.4 demonstrates the dependence of the bearing stress $\sigma = P/dh$ (in which P is the force acting on the bolt with diameter d and h is the composite plate thickness) on the strain $\varepsilon = \delta / d$ (where δ is the bolt displacement) for the quasi-isotropic composite under study. As can be seen, the bearing strength (830MPa in Fig.4(a)) looks rather high, but the unloading diagrams (dashed lines in Fig.4(b)) show that if the stress exceeds 160MPa (which is close to the compression strength of the matrix) the residual deformation of the hole caused by the matrix microcracking is observed. The shape of the hole after loading up to the stress 320MPa and unloading is shown in Fig.5. Naturally, such residual strains cannot be allowed in composite parts of commercial airplanes and the allowable bearing stress should be restricted by the level close to 160MPa which significantly reduces the weight efficiency of stringer composite structures.

The other problem is associated with the safety factors. Traditional approach [5] looks rather

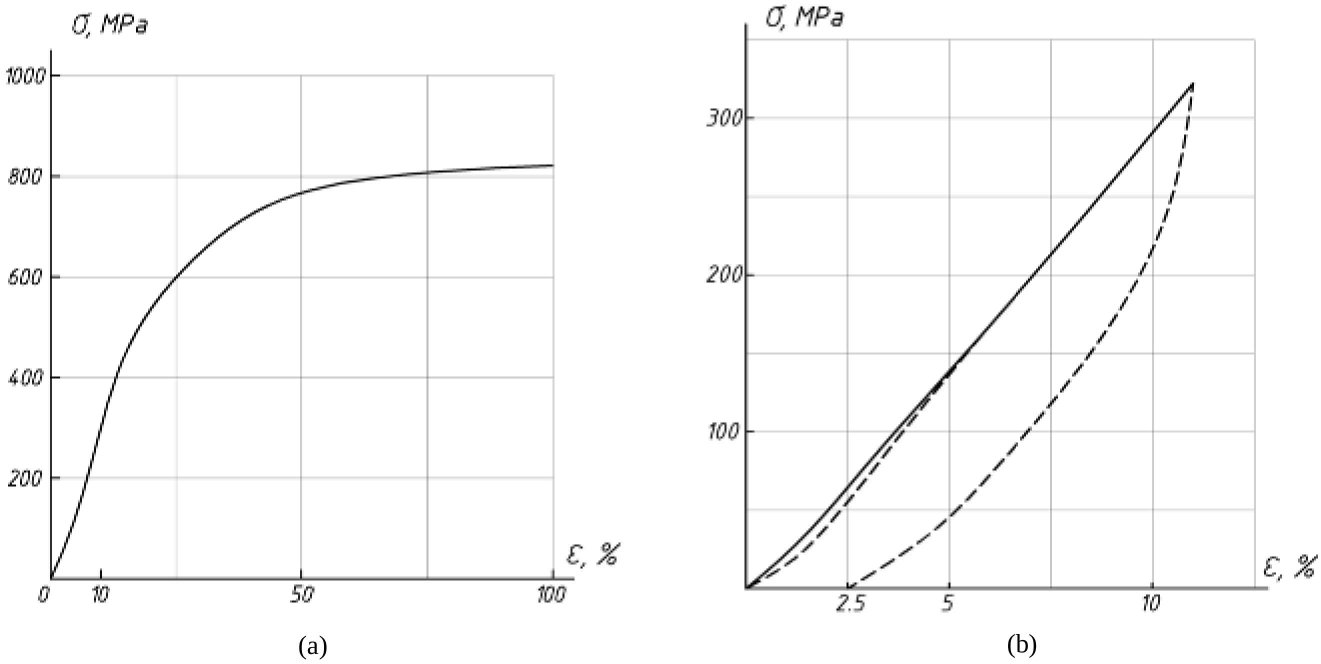
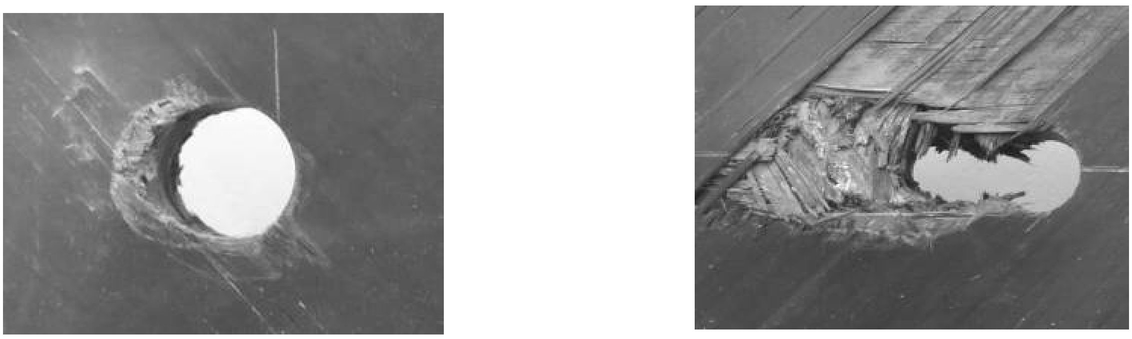


Fig. 4. Bearing stress-strain diagram (a) for quasi-isotropic carbon-epoxy laminate and the initial part of the diagram (b) with unloading curves

conservative. First the flight loads are multiplied by the safety factor 1.5 typical for aluminum aircraft structures. But, in contrast to aluminum, the ultimate stresses or strains of composite laminates are further reduced by some additional factors. For example [5], for the case of compression, material scatter reduces the initial ultimate strain by 0.8, temperature and moisture- by 0.8 and possible damage and delamination – by 0.65. As a result, the total effective safety factor becomes so high that significant weight saving of the composite stringer structure cannot be expected. The same is true for sandwich composite structures which also have load-carrying laminated skin. That is why for the existing composite airframe structures experiencing the action of relatively low loads (control surface, doors, horizontal and vertical tail) the weight savings is on the level of (10-15)% and is caused mainly by relatively low density of composite materials, whereas for heavily loaded structures (wing and fuselage) no weight savings have been demonstrated by now [6].



(a) (b)

Fig. 5. Residual shape of the hole in the plate loaded up to 320 MPa (a) and 830 MPa (b)

2. Geodesic Composite Lattice Structures

In contrast to stringer and sandwich structures, in geodesic structures (Fig.1(c)), the main load-carrying elements are the ribs. Geodesic design concept was invented in 1896 by V.G. Shukhov and was widely used in Russia to build metal towers and masts [7]. In application to aircraft structures, geodesic design concept was used in small wooden airplanes [8] and in the metal WWII English bomber Wellington [9]. The plane frame had a system of helical aluminum ribs covered with fabric skin. It was about 30% lighter then stringer aluminum prototypes and had the outstanding survivability. After WWII, the program under which about 11 thousand of planes had been built was terminated because of manufacturing problems associated with the necessity to join helical ribs with metal skin required for pressurized fuselages.

Composite materials in conjunction with geodesic design concept open new possibilities in development of light-weight and cost-efficient structures. Such structures referred to as Anisogrid (Anisotropic Grid) structures consist of a system of load-carrying ribs made of unidirectional advanced composite materials by continuous automatic filament winding.

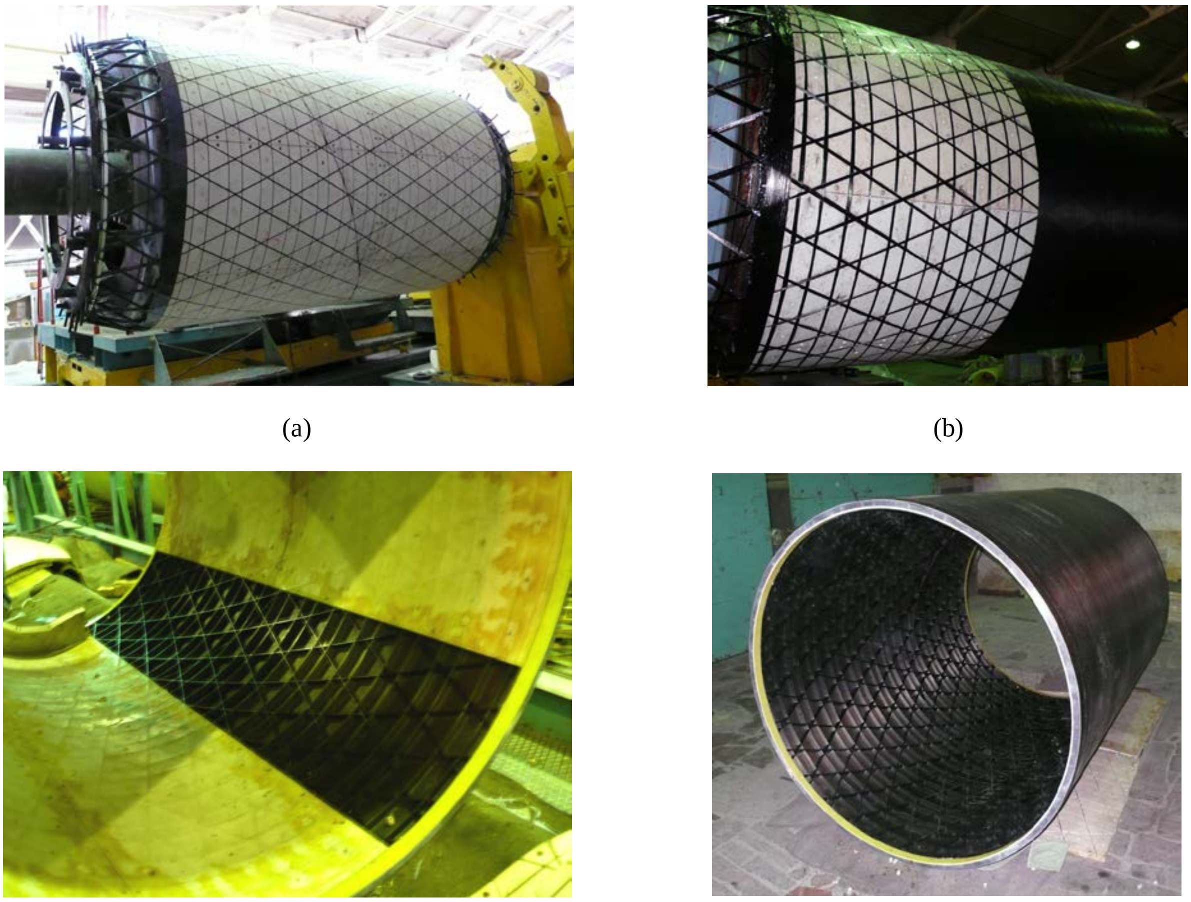
The manufacturing process includes the following main steps illustrated by Fig.6.

- The mandrel is covered with elastic coating formed of silicon rubber and having the groves for the ribs (Fig.6(a)).
- Unidirectional carbon tows impregnated with resin are wound into the groves forming a system of helical, hoop and in some cases axial ribs which are covered with thin composite skin, also made by winding (Fig.6(b)).
- After curing the mandrel is removed and the elastic coating is pulled out as shown in Fig. 6(c) resulting in an integral composite structure (Fig.6(d)).

Anisogrid structures are widely used now as interstages, payload adapters and elements of space platforms [10,11].

Experimental Anisogrid fuselage structure is shown in Fig.7. Lattice structures can be also used to fabricate beam type elements like spars, wing ribs and floor beams shown in Fig.8.

The ribs, being the basic load-carrying elements



(a) (b) (c) (d)

Fig. 6. Fabrication of a composite lattice structure: (a) winding of the ribs, (b) winding of the skin, (c) removal of elastic coating, (d) fabricated structure

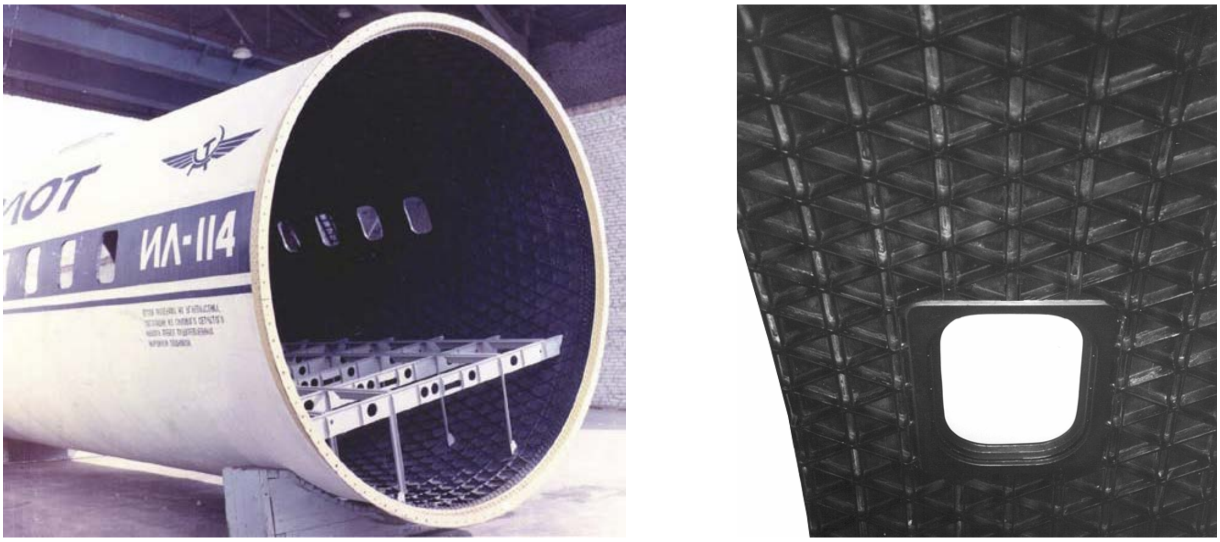


Fig. 7. External and internal views of a composite geodesic fuselage section

Fig. 8. Beam-type anisogrid composite lattice structures

of geodesic composite structures, demonstrate rather high specific strength and stiffness. For traditional carbon-epoxy composites based on AS4, T-300 or HTS type carbon fibers, the rib modulus is close to 90GPa, strength under tension is 1350MPa, strength under tension is 650MPa and density is 1450kg/m^3. The lower mechanical characteristics and density of the ribs in comparison with the corresponding properties of the unidirectional composite materials made by traditional methods (e.g., fiber placement) are associated with the lower fiber volume fraction in the ribs. Because the structure thickness is the same at the points of the rib intersection and between these points, the fiber volume fraction, being about 75% at the points of rib intersection, reduces to about 40% between these points. Nevertheless, the rib modulus is 30% higher and the rib density is about two times less than the corresponding characteristics of aluminum alloys. Application of high-modulus carbon fibers (M46J or M fibers) allows us to increase the rib modulus up to 185GPa or 250GPa, respectively. Fig. 9 shows the fatigue strength of the ribs under compression which is the critical type of loading for geodesic composite structures (tensile strength of the ribs is about two times higher than compressive strength). Under the stress which makes less that 55% of the static strength, no fatigue failure has been observed.

Design of Anisogrid structures is based on the minimum mass criterion under constraints imposed on the rib strength in compression, local buckling of the rib segment between the points of ribs intersection, global buckling and stiffness of the structure. For analysis, two types of Finite Element models are used, i.e., continuum model in which the ribs are smeared over the structure surface and discrete models in which each rib element between the points of ribs intersection is simulated with a beam-type element (Fig.10).

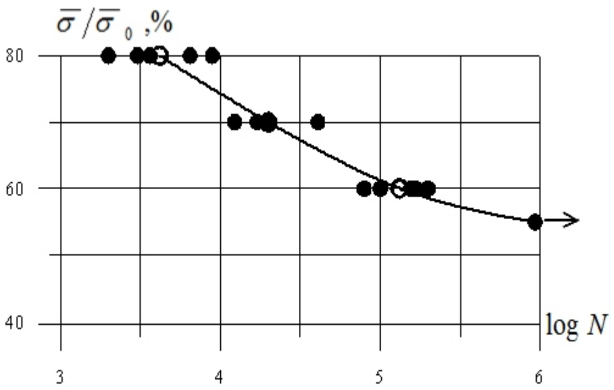


Fig. 9. Fatigue strength of the ribs under compression ($\bar{\sigma}_0 = \bar{\sigma}(N=0)$)

Relatively high thickness of geodesic composite structures (20-30mm for fuselage and wing structures) allows us to join these structures with the adjacent composite or metal structures without high weight penalty. The strength of the joint in Fig. 11 which takes about 100KN is controlled by the strength of the bolt.

Finally, consider the problem of the appropriate safety factors for geodesic composite aircraft structures. Traditionally, for aluminum structures, the limit loads are multiplied by the safety factor f = 1.5, thus giving the ultimate loads for which the structure is designed. The safety factor covers the shortage of information concerning the flight loads, scatter of material properties, the effect of aging, environmental factors, etc. It is quite evident that for the structures consisting of metal and composite parts the unique safety factor cannot exist. For example, the new Russian jet MC-21 which is

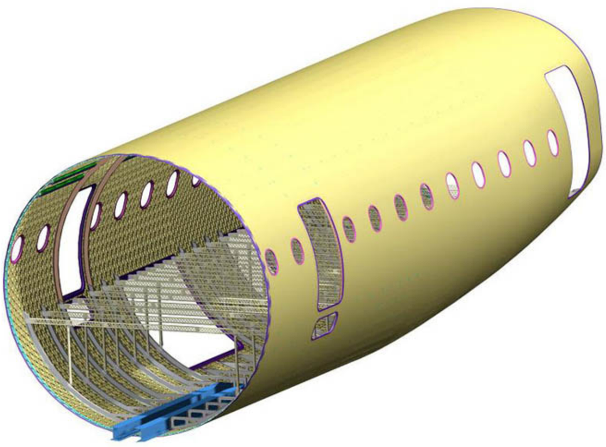


Fig. 10. Finite-element model for geodesic composite fuselage section

under development now should have aluminum fuselage and composite wings. The safety factor 1.5, being appropriate for the metal fuselage, is evidently not enough for composite wings, because the properties scatter of composite materials are much higher than for aluminum. Moreover, this scatter is different for the strength under tension and compression and for the modulus which controls buckling. Thus, for composite structures, the safety factor must depend also on the structure stress state.


Fig. 10. Typical joint of the geodesic composite structure

Geodesic composite structures in which the load-carrying capacity of a thin skin is ignored are actually the truss systems whose stress state and buckling loads can be reliably and rather accurately determined using modern finite-element methods so that there exists a direct correlation between the limit loads on one side and the stresses in the ribs and the buckling loads of the geodesic structures on the other side (limit values). Thus, it is proposed to take into account the scatter of material properties introducing different safety factors as the ratios of the actual ultimate strength and critical stresses to the corresponding limit values. The safety factors are found with the aid of the theory of reliability [12] and depend on the material mean characteristics $\bar{\sigma}$ and $\bar{E}$, the corresponding variation coefficients v, the variation coefficient for the flight (limit) loads v_f and the probability of failure β. For v_f = 0.08 and β = 10^{-5}, the safety factors f for geodesic composite structures are presented in the Table.

Property	Mean Value	Variation Coefficient	Safety Factor
Strength			
-compression	670MPa	8.5%	1.7
-tension	1390MPa	4.6%	1.5
Modulus	94GPa	2.3%	1.4

The safety factors listed in the Table allow for the properties scatter only. To take into account the other factors reducing material strength and stiffness they should be multiplied by the corresponding reduction coefficients. For example, for the material aging the corresponding experimental coefficient is 1.06. Then, the allowable stress for compression becomes 374MPa which makes 55.7% of material static strength. Under such stress, the fatigue failure does not occur (Fig.9) .In flight, the ribs are protected against temperature, moisture and impact by the skin, and the reduction of the skin properties does not affect the structure strength.

In conclusion note that in contrast to composite stringer structures, geodesic composite structures have the following advantages.

- The ribs which are the basic load carrying elements have unidirectional structure and, being reinforced with modern carbon fibers, are characterized with extremely high specific (with respect to density) strength and stiffness allowing us to reduce the structure mass by (30-40)% in comparison with aluminum prototypes.
- Thin (about 1.5mm) fabric skin takes only internal pressure. It does not crack under tension (see Fig.1) and does not experience invisible damage. Skin damage does not cause the failure of the structure whose strength is controlled by the ribs.
- The structure is completely integral. The ribs the skin, the end rings, the window and door frames are made within one and the same manufacturing process (filament winding). Winding machines are less expensive than fiber placement machines and the materials for wet winding are less expensive than prepregs used for tape placement. No autoclaves are necessary, fiber tension is used to provide the proper material density.

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