VOL. 9, NO. 10, OCTOBER 2014 ISSN 1819-6608
ARPN Journal of Engineering and Applied Sciences

www.arpnjournals.com

STRUCTURAL COMPOSITES FOR AIRCRAFT DESIGN

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ABSTRACT
Composite structures such as CFRP offer significant weight reduction over the conventional aluminum alloys for aircraft. Weight reduction improves fuel efficiency of the aircraft by approximately 20% which results in cost savings and simultaneously reduces the operational environmental footprint. However, the new aluminum-lithium alloys offer significant improvements and are viable alternatives to CFRP. Aluminum lithium alloy 2195 with Friction Stir Welding is introduced as a successful alternative to CFRP primary structures. A "thick skin" monocoque design with integral stringers as crack stoppers is discussed. An old Macchi 205 WWII fighter plane has been redesigned both in CFRP and 2195-FSW for comparison. The final designs are comparable in weight, but 2195-FSW is more competitive based on mass production costs, reparability, and environmental impact. Macchi 205 airplane is used due to in-depth experience with the original aircraft geometry and loads. Knowledge gained here can be directly transferred to larger structures, from corporate jets to large transport category airplanes [1].

Keywords: aircraft structures, composite material, aluminum alloys, CFRP, 2195-FSW.

1. INTRODUCTION

Many natural composites exist in nature such as wood and plants such as bamboo and palms. Some artificial composites were known literary for thousands of years and concrete is the most common one used in construction. Also plywood was known to ancient civilizations. Recently, new generation of composites became popular in aerospace/aviation industry.

Carbon fiber reinforced plastic (CFRP) is apparently an ideal material for an aerospace designer: its strength is comparable to that of steel and its density is half the average aluminum alloy used for airplane manufacturing. Notch sensitivity is very low and fatigue life outstanding. However, the physical properties of composite materials are mostly tailored for UD loads. Many aircraft structures are nevertheless exposed to three-dimensional loads and modern lightweight metal alloys are still the best choice for such parts.

The, so-called, “all-composite” Boeing 787 is really not what many may think a “plastic airplane”. Only roughly half of its structural mass is made of composites. Boeing 787 still has a lot of metal in it. Future Airbus 350 will be about 40% composite, but 60% will still be mostly metal. Existing superjumbo Airbus 380 has fuselage made mostly of GLARE which is a combination of carbon fiber and aluminum. The new Lockheed-Martin’s F-35 fighter jet has a titanium and aluminum internal structure supporting the exterior composite skin. Statements that “metal” airplane is obsolete and will be completely substituted with “plastic” airplanes are thus highly exaggerated in our opinion.

Some good mechanical properties and low density in new aluminum alloys comes from lithium, a lightest and least dense metal of all. Lithium is most commonly found in China, Russia and Australia. Additionally, various aluminum-, titanium-, nickel- and cobalt-based super-alloys have extensive applications in modern turbofan engine designs (e.g., Pratt and Whitney’s geared turbofan) from compressor and turbine blades to air inlet/intake designs.

Despite many wonderful properties the composite CFRP performances are highly dependent on the manufacturing process and damage detection is difficult. While metals usually deform locally upon impact, CFRP internal cracks are usually undetectable visually. Formula 1 (F1) car racing, military, and civil aircraft utilization so far found different solution for these problems. However, a high price in terms of weight has to be paid for such specialized solutions. CFRP automated fabrication is not as easy as for aluminum alloys and repair knowledge and experience is not widespread or readily available. Afterlife disposal is also a problem for many composites. Corrosion, aging and varying thermal expansion in combination with metal structures are also important properties. NASA’s, now retired, space shuttle orbiter (STS) launch vehicle’s oxygen tanks have been manufactured using the 2195-FSW aluminum-lithium alloy. Alcoa’s aluminum alloy family 2099 is currently used in many modern aircraft structures and will most likely be used in foreseeable future as well. FSW type of structure has been thoroughly examined and experimented and it is now commercially available and operationally ready for new aircraft structure designs. Monocoque, "thick-skin” technology has been widely investigated in the automotive field. Historically, it has been also partially experimented with in the Japanese WWII airplane fighters (e.g., Mitsubishi Zero). Very low aspect ratio ribs are easily manufactured in aluminum alloy panels as crack stoppers.

In this article, various CFRP structures are discussed along the technological and safety aspects. Several composite designs to fabricate a full-scale replica of Macchi 205 WWII fighter plane are proposed. The reasons for that are not only in pure convenience of having access to such a model, but also that it covers small-to-medium GA piston-prop airplane types and also addresses important issue of kitplanes. The alternative design using aluminum alloy 2195-FSW instead of composite CFRP is then examined. A comparison between the CFRP and 2195-FSW aircraft designs completes the discussion.

Like everything else in technology, the future space, aerospace, and aviation designs will most likely be using, both, the composite materials and metal alloys utilizing their best features where and when needed.

2. CFRP TECHNOLOGY

Several different CFRP manufacturing technologies are available today. The main issue is how to combine the resin matrix and the reinforcing fibers to achieve multi-parameter performance optimization. In order to achieve the best unidirectional physical and mechanical properties, tapes or fabric should be used. To obtain good mechanical properties, the rule is to increase fiber volume fraction while minimizing the matrix amount. Currently, high-volume CFRPs have 58-60% of weight in fibers alone. The matrix can be made with thermoset or thermoplastic resin systems. The thermoset system is easier to manufacture and it is more frequently used for this same reason. The most common and flexible process to produce high performing CFRPs requires prepreg (preimpregnated) material cured in an autoclave.

Pressure and thermal cycles are applied in an autoclave during curing process. The higher the pressure, the smaller the resin part, the thinner the panel, and the better CFRP properties are achieved. Complex shapes require very flexible textiles. UD fibers are difficult to wrap up around corners. Both carbon and plastic can be heat treated. Fibres are usually heat treated by the original manufacturer, while the resin is heat treated partly during manufacturing process and partly during life time. Matrix resins, usually epoxy based, are optimized with the specific carbon fiber and cure cycle to obtain optimum performance in term of toughness, strength, adhesion, fire resistance, etc. Mechanical properties of CFRP parts are subject to aging due to exposure to light, temperature cycling, corrosion, and stress cycling [2-4].

2.1. Automated lay-up

Laminates are laid up by robots (flexible, but slow) or other specialized machines. Such machines use FW (Filament Winding), AFP (Automated or Advanced Fibre Placement) for components with curvatures, or ATL (Automated Tape Lay-up) for long straight components. Automatization is essential to improve the repeatability of the fabrication process and reduce manufacturing time and cost. Careful process inspection and monitoring is required during manufacturing, since CFRP part quality and final mechanical properties largely depend on it. Care should also be taken about the shelf-life of the resin system. In these cases, the cure of the components does not generally require an autoclave. It is often achieved with vacuum applied and heated moulds.

3. CFRP DESIGN PHILOSOPHY

Two different approaches are used in CFRP design here: the thin protected sandwich (as in F1 racing cars) and the thick unprotected skin. Both design principles will be addressed in more details.

3.1. Thin protected sandwich

The thin protected skin is based on the concept that “thinner is better” for composite panels [5]. So, when possible, the outer and the inner skin are laminated and cured separately. The adhesion of the outer skin to the honeycomb core is achieved by an adhesive film and the inner skin is cured directly on the core. The FIA’s side-panel intrusion homologation is achieved by most of the F1 racing teams using aluminum honeycomb, due to their higher capability to absorb energy over Nomex. Inserts should be as small as possible [6-9] and should be made of titanium, titanium alloy, or carbon-carbon to avoid corrosion problems [10]. Bonded joint is preferred to riveted joints. In bonded joint rivets should be avoided, hence "hybrid" joints are avoided. The rivets should be used only were the joint design is such that peeling stress may occur [11-13].

The CFRP panel or structure is closed in the vacuum-bag(s) with absorbing material, called breather and is shown in Figure-1. The UD laminate is stacked to the mould (bottom plane) that is covered by a release agent. Over the stack another release film, sometimes perforated depending on the resin system, and then a breather are laid. Everything is closed in one or more vacuum bags [14].

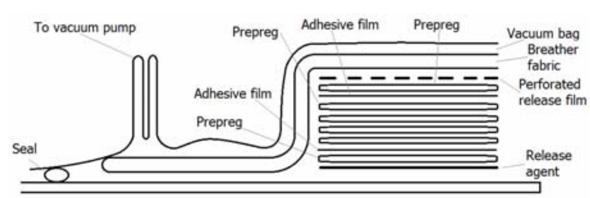

Figure-1. A schematic representation of the thin protected sandwich autoclave manufacturing process.

High pressures and temperatures are applied on this bag in the autoclave. Heat liquefies the matrix resin while high pressure squeezes the resin out of the fiber stacks and into the absorbing material (breather fabric). Pressures of up to 10 bars have been used for optimum results. Usually the first and the last ply are made with T300 woven fabric to obtain about 0.1 mm of the sacrifice material. Additional layers for lightning protection, and insert embedding may be added. The result is a laminate with good tolerances on the side of the mould, and poor geometry and high roughness on the side of the breather fabric.

Void content decreases with high pressure with and improvement in interlaminar shear strength. Influence of voids on interlaminar shear strength of carbon/epoxy fabric laminates has been described in [15]. However, the

 ISSN 1819-6608
ARPN Journal of Engineering and Applied Sciences

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void content is not so critical for CFRP thermoset composites.

F1 racing cars have chassis that is very rigid in torsion and are thus often overdesigned for twisting loads. The chassis is usually made of two halves and then carefully bonded in a single component [16-17]. Extensive crash tests and analysis resulted in extremely strong chassis designs of today [18]. Where possible the chassis is protected by fairing and accessories that protect the chassis outer skin from direct impacts. After the race the chassis is thoroughly inspected for de-lamination with tapping or using US NDE (UltraSonic Non Destructive Testing). Repair techniques widely used are resin injection, laminated double patch, and scarfed patch [19]. Again, rivets are accurately avoided also in repairs. Self drilling/tapping screws are prohibited after the fatal Ratzenberger's F1 accident (San Marino F1 Gran Prix, Imola, 1994).

This approach makes it possible to use high stress and high strength fibers in the same laminate to optimize the part behavior and the possibility of an effective manufacturing of the part. Generally, the properties of composite materials are anisotropic and 2nd-order tensors mathematics is required to describe changes of directional physical properties. The orthogonal isotropy often simplifies the analysis. For FEA use, the following theoretical expressions are used in first approximation [20]:

$$VF_{fiber}=\frac{V_{fiber}}{V_{composite}}=\frac{V_{fiber}}{V_{fiber}+V_{resin}}=\frac{M_{fiber}/\rho_{fiber}}{M_{fiber}/\rho_{fiber}+M_{resin}/\rho_{resin}} \qquad VF_{fiber}+VF_{resin}=1 \tag{1}$$

$$E_{long}=\left[VF_{fiber}\times E_{fiber}+\left(1-VF_{fiber}\right)\times E_{resin}\right] \tag{2}$$

Here, E_{long} is Young's modulus parallel to fiber length. Equation (2) holds quite well as shown in Fig. 2. However, Young's modulus does not take into account void content (porosity). Accordingly, the modulus and the strength are always overestimated. The thicker the material the larger is the overestimation. The influences on material strength are less important, but still, the thicker the worse.

Void content depends on manufacturing process, bends, pressure, and honeycomb core void content increase [21] as shown in Figure-3. A void content (porosity) up to 2% is normal. A fiber content of 56% in volume is typical of a very good part.

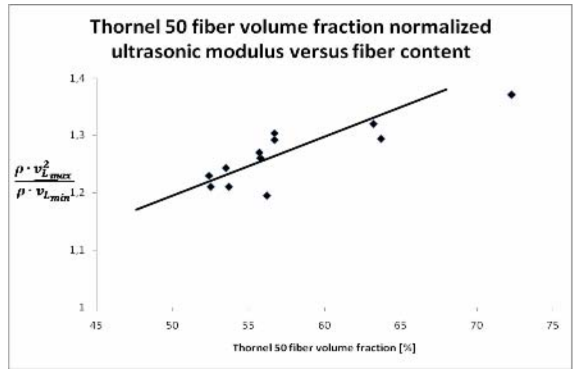

Figure-2. Fiber volume fraction versus Young modulus [21].

3.2. The thick-skin design

In this case, the composite is thick enough to tolerate small impacts during flight [22]. Riveted or bolted joints and repairs are possible due to sufficient strength [23-30]. The thick lay-up makes it possible to obtain quasi isotropic laminates. Large quantities of high stress fibers are used for damage tolerance. Automated composite lay-up (ATL) enables reduction in manufacturing time. The large parts are manufactured at different sites to reduce manufacturing costs and then transported for a full assembly. The thick skin design provides thermal insulation and fire protection [31-32].

3.3. Comparison of the thin protected and the thick skin approach

The protected skin design enables the use the best fiber-resin combination in order to obtain high quality composites as well as easier manufacturing process control. Bonded joints are always critical, but this manufacturing process is well proven. For the "protection" skin tougher and cheaper materials can be used. The structure is more complex and more expensive to assemble.

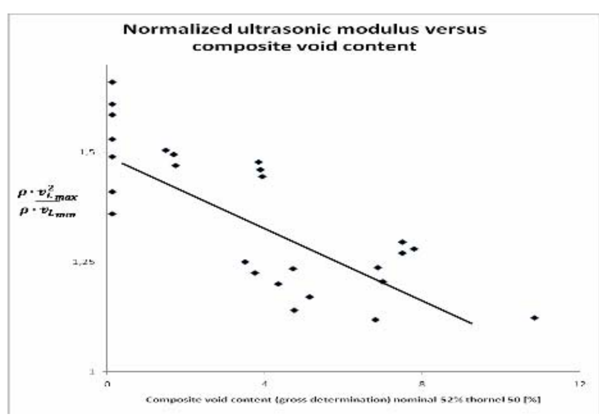

Figure-3. Elastic modulus vs. void content.

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The thick-skin design is more critical from the manufacturing point of view. It is more difficult to obtain a nice looking and a well manufactured component. Large component assembly is critical for joint tolerance. CNC machining may be required. Large quantities of composites are to be used for each single component, with the risk of large scraps. In aircraft fuselage (hull) and other structures, the thick skin provides thermal insulation and fire protection. As the thick-skin technology becomes more advanced and better understood it will be possible to manufacture parts at different physical locations with the clear advantage in costs and time. The composite material weights for the two CFRP design strategies are similar. The CFRP potential cannot be fully exploited, since impact resistance and, more importantly, impact detection limits the material usage. Additionally, the manufacturing process seems to be more critical than for traditional metallic structures.

3.4. The CFRP Macchi 205

Since the authors mostly come from the F1 racing world, the thin protected solution was adopted first for aeronautical application. An internal frame, shown in Figure-4, was designed to meet the USA's FAR 23 aerobatic category airplane class certification. The installed engine is the Orenda OE600.

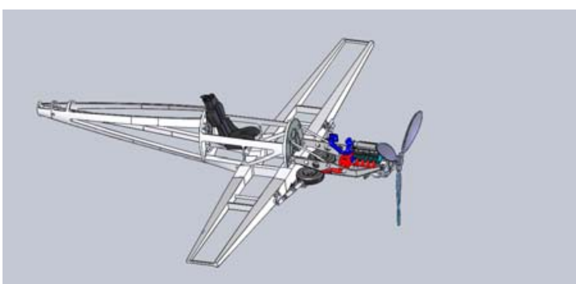

Figure-4. The internal frame of the "black replica" CFRP Macchi 205.

The frame is made of two parts bolted together at the seat bulkhead. The single frame parts can be housed in an existing autoclave. The beams are made with M46J-epoxy on a structural foam core. Titanium alloy inserts are included at several points to allow the connection between the two parts and the installation of the CFRP/foam skin panels (see Figure-5). A bubble top two-piece canopy was installed enabling excellent 360^{o} visibility. Such canopy is also required for the ejection seat installation and operation considered in the design example. The external skin protects the vital structure from the impact damage. This design was finalized to evaluate actual mass (weight) savings of the composite-material airframe which ended up being only 108 kg. The original WWII airframe mass was about 350 kg, or more than three times the new CFRP composite structure. The retractable landing gear assemblies in the case utilizing CFRP can be made lighter as well. This airplane model is of the conventional landing gear design (with tailwheel). The original Macchi 205 was an Italian fighter plane with the Daimler Benz 605A reciprocating engine (1470HP at 780 kg) manufactured by FIAT under license and MTOW of 3408 kg (about 7500 lb). The Orenda engine is lighter and less powerful (600HP at 350kg) and a single-rubber-bladder fuel tank conforming to the specifications FIA/FT5-1999 / MIL-DTL-27422 is installed onto the front bulkhead. The suggested MTOW of the CFRP replica of the Macchi 205 is 1200 kg (2640 lb), which of course must be officially obtained during the certification process.

Figure-5. The "black replica" of the Macchi 205 with transparent skin and fins. The single-rubber-bladder MIL-DTL-27422 fuel tank behind the Orenda engine is not shown. It is bolted directly to the front bulkhead.

3.5. The thick-skin Macchi 205 structural design

The true monocoque design depends almost completely on the strength of the outer skin to carry primary loads. The skin must be stiff enough without any bracing members, formers, or bulkheads. The presence of riveted joints in conjunction with the strength-to-weight problems of pure monocoque construction has given place to the semi-monocoque construction that has the skin reinforced by longitudinal member (longerons) in addition to having formers, frame assemblies (stringers), and bulkheads. The main idea is to design a Macchi 205 fuselage and wing structure into a single part welded using FSW.

4. HYBRID METAL COMPOSITES

Hybrid metal composites can be traced back to British and French aircraft of the 1930s, notably the Morane-Saulnier M.S.406 where Plymax was used for stressed skin. British Halifax model used Plymax panels as floor material. Plymax is (three-ply-Okoumè) plywood with a sheet of duraluminum glued to it [33].

Such structure solution evolved into TiGr (Titanium alloy/graphite fiber-reinforced polymer matrix composite) and GLARE (Aluminum alloy/glass fiber reinforced polymer matrix composite) materials of today. As mentioned earlier much of the Airbus 380 fuselage is fabricated with GLARE. Metallurgically, titanium

VOL. 9, NO. 10, OCTOBER 2014 ISSN 1819-6608
ARPN Journal of Engineering and Applied Sciences

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combines well with composites. An advantage of hybrid metal-composites is the improved fatigue life compared to pure composites. The dominant material failure mechanism is fatigue crack growth in the metal plies accompanied by delamination between the metal and composite plies [34-35].

Another major advantage of hybrid metal composites is the possibility to have an outer metal skin which enables easier to crack detection by visual inspection. Additionally, the fuselage structure is also conductive to electrical currents making it safer in lightning strikes incidents. However, hybrid metal composites have certain fabrication difficulties. In fact, the absorbing material cannot be inserted easily and it is quite difficult to obtain the high fiber content.

4.1. NASA's experience with 2195-FSW

FSW was the most recent upgrade to the recently retired Space Shuttle's External Tank (ET), the single non-reusable largest element of the STS. In 1993, Lockheed-Martin laboratories in Baltimore, Md., developed replacement for the aluminum alloy Al 2219 semi-monocoque structure used on the original ET. The new SLWT (Super Light Weight Tank) was made with Aluminum Lithium Al-Li 2195, which reduced the original mass of 30,000 kg of the LWT (Light Weight Tank) by 3,175 kg (about 10% mass/weight savings). An image of the SLWT structure is shown in Figure-6.

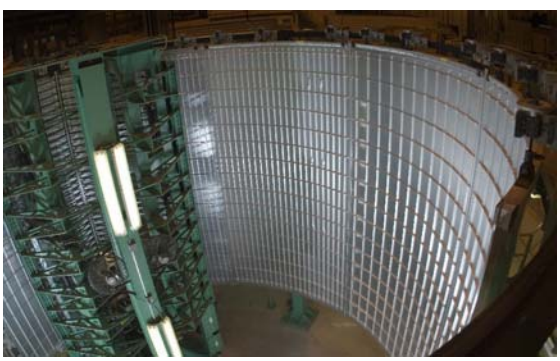

Figure-6. The NASA's space shuttle 2195-FSW SLWT.

4.2. Macchi 205 design with 2195-FWS

The first step in designing a Macchi 205 with the Al-Li alloy 2195-FWS was to decide on the mesh type. Considering the type of analysis encountered, shell mesh is more appropriate since the material thickness does not exceed 10 mm while the whole geometry is on the scale of meters. The aircraft skin was then derived from the original drawings as shown in Figure-7. A second seat was added in our design. It is possible that such modification had been already used on some Veltros during WWII but we have no proof of it so far [36-40].

Macchi's skin has been deformed with the maximum manufacturing tolerances in order to take into account the effect of buckling also for the static FEA computations. For example, wings were curved upwards and panels were buckled to the tolerance limits. For the structure optimization the worst case was assumed to be a sudden pull-up after a deep dive at VNE (Never Exceed Speed for FAR 23 certified airplanes) with accompanied tailslide. In this case the airframe takes the maximum vertical and lateral g-forces. Results of FEA analysis [41] [42] are shown in Figure-8. Other conditions required under FAR 23 certification utilizing high-g flight loads (aerobatic maneuvers) were used as verification of the structure strength. A minimum thickness of skin panels of 0.7 mm was necessary for structure integrity. Simulations also revealed that a critical ground load was single-wheel landing with results shown in Figure-9. For such ground load condition an ad-hoc structure was embedded in the basic monocoque design. FSW design enables stringers and reinforced plates to be installed inside the original structure. The same operation will be made for the other reinforcement needed for the assembly of the aircrafts. In welded aluminum alloy structures it is common to join parts manufactured with different techniques as it can be observed in Figure-10. The total structural mass of the 2195-FWS Macchi 205 is only 110 kg and only 2 kg heavier of the CFRP design.

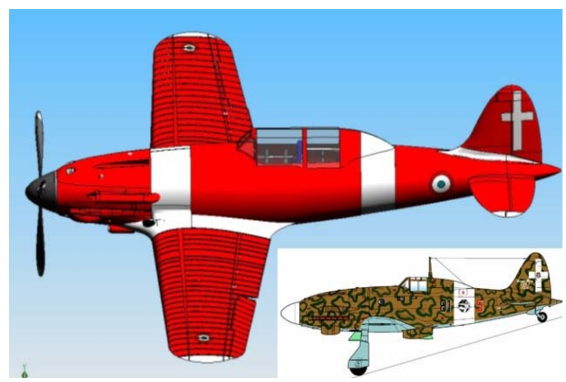

Figure-7. Macchi 205 Veltro skin from the original WWII drawings (courtesy of Aermacchi Aircraft Company).

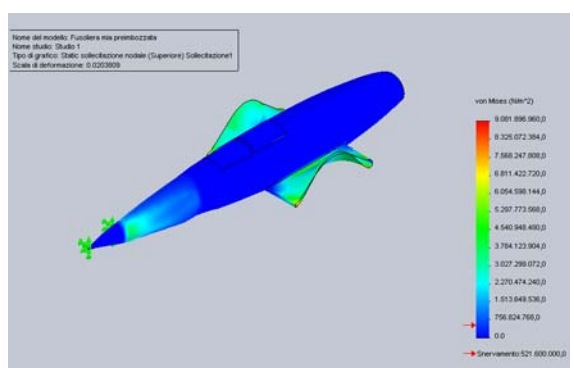

Figure-8. FEA simulation of the fuselage stresses for worst load condition: pull-up at max-g with subsequent tailslide.

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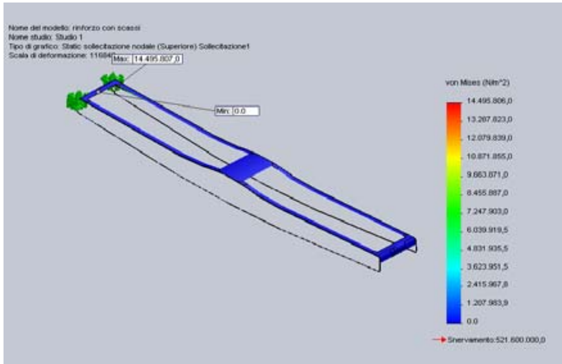

Figure-9. Reinforcement optimization of one-wheel (asymmetric) touchdown at MTOW and +2.5g vertical acceleration.

Figure-10. Rosmoto SR 744R's swing-arm, cast, machined and tubular parts are welded together.

4.3. Integral stringers in the skin of the 2195-FWS monocoque

Further optimization can be made by using low aspect ratio stringers/ribs manufactured in the skin of the monocoque structure. An example of this structural solution can be seen in the cast aluminum alloy head of the FIAT's "Fire" Engine and is shown here in Figure-11. These ribs can be laminated, forged, or coined as a 3D pattern in the aluminum-alloy sheets. Such design gives important improvements in stiffness, strength and crack resistance.

Figure-11. Ribs and stringers in the casting of the head of a FIAT "Fire" engine.

As an example, the 0.7 mm thick skin of the Macchi 205 can be replaced by a thinner skin but with ribs geometry shown in Figure-12.

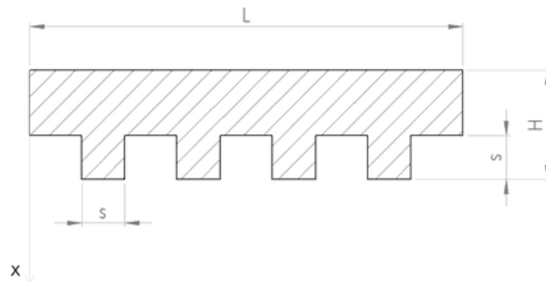

Figure-12. Thin skin design reinforced with integral ribs.

We can optimize the stiffness of such ribbed skin by using the following equations for the center of gravity location and moment of inertia:

$$x_G = \frac{L(H-h)\frac{(H-h)}{2} + \sum_{i=1}^{Nribs}(s\,h)\left(H+\frac{h}{2}\right)}{L(H-h) + \sum_{i=1}^{Nribs}(s\,h)} \tag{3}$$

$$I = \frac{L(H-h)^3}{12} + L(H-h)\left(x_G - \frac{(H-h)}{2}\right)^2 + \frac{sh^3}{12} + (s\,h)\left(x_G - \left(H+\frac{h}{2}\right)\right)^2 \tag{4}$$

For the optimization we can fix L and H. The condition of equal mass between the original and the reinforced skin can be imposed using the following mass balance:

$$L(H-h) + \sum_{i=1}^{Nribs}(s\,h) = L\,H_0 \tag{5}$$

The number of ribs can be calculated in order to maximize plate stiffness [36-38]. The width of the ribs is fixed at 1 mm and the total thickness of the reinforced sheet is also 1 mm. The optimal solution results in reinforced skin of 0.5 mm with 400 of ribs/meter. The ribs are 1 mm wide and 0.5 mm tall. This optimal reinforced skin has a stiffness increment of 76% in comparison with the original 0.7 mm skin of the same mass (no ribs).

5. COMPARISON BEWEEM THE CFRP AND THE 2195-FSW MACCHI 205 DESIGNS

On a pure mass/weight basis the advantage of the all-composite CFRP Macchi 205 is marginal over the hybrid metal composite monocoque 2195-FWS design. Actually, the monocoque structures can be made even lighter with the use of integral low aspect stringers in panels. This is primarily due to the fact that the CFRP structure is composed of a frame that carries the loads with skin then providing smooth aerodynamic surface for good performance. It is possible that with the "thick" skin approach and a truly monocoque structure the CFRP could become favourable again. However, impact resistance is a major problem for this type of structures involving CFRP. It is also possible that our calculation on a monocoque structure has been overly optimistic. However, serious doubts remain over the advantage to use CFRP over a more traditional FSW welded aluminum alloy structure.

6. CONCLUSIONS

The apparent weight advantage of the CFRP aircraft is reduced by the low impact strength. The difficulty of skin damage/crack detection further reduces the advantage of CFRP airplane. Ease of repair is important operational factor that should be taken into account. Two different approaches are currently used; the thick skin and the thin protected bearing structure. Both designs increase the structural mass of the CFRP aircraft. On the other hand, the implementation of the pure monocoque airplane structure utilizing 2195-FSW aluminum-lithium alloy can reduce the overall structural weight of the traditional semi-monocoque solution. A rib pattern can be easily manufactured on the flat panels to improve stiffness and crack resistance. The new manufacturing technology utilizing modern aluminum-lithium alloys may then render the CFRP structure obsolete [43-52].

ACKNOWLEDGEMENTS

The authors would like to thank Alenia Aermacchi for kindly providing the original company drawings of the WWII Macchi 205 fighter plane.

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Nomenclature

AFP Automated (or Advanced) Fibre Placement
ATL Automated Tape Laying
CFRP Carbon Fibre Reinforced Plastic
CNC Computer Numerical Control
E Young's modulus [Pa]
FAR Federal Aviation Regulations (USA)
FEA Finite Element Analysis
FIA Fédération Internationale de l' Automobile
FW Filament Winding
FSW Friction Stir Welding
GLARE Glass Reinforced aluminum alloy
h Rib height [m]
H Total thickness of stiffened skin [m]
H_0 Thickness of the "original unreinforced plane" skin [m]
J Moment of inertia of the section [kg m^2]
L Portion of skin considered for the optimization [-]
M Mass [kg]
MTOW Maximum Takeoff Weight/Mass [kg], [lb]
NDE Non Destructive Testing
N_{ribs} Number of ribs in L [-]
S Rib's width [m]
SLWT Super Light Weight Tank
STS Space Transportation System
UD Uni-Directional
US Ultrasonic (Ultrasound)
V Volume [m^3]
X_G Centre of gravity position along the x axis [m]
ρ Density [kg/m^3]