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CASE STUDIES IN COMPOSITE MATERIAL STRUCTURAL DESIGN, MANUFACTURE AND TESTING

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ABSTRACT

Advances in composite materials technology have brought with them a great deal of practical advantages for aircraft structures. The Division of Aeronautical Systems Technology (Aerotek) of the CSIR has designed and built a number of prototype structures using these materials.

Two of the projects utilising carbon fibre are summarised in this paper, namely a technology demonstrator aircraft and a drop tank. The design, manufacturing and testing phases of these projects are discussed.

1 TECHNOLOGY DEMONSTRATOR AIRCRAFT

1.1 Introduction

The aircraft designated as OVID, as shown in Figure 1, was developed as a composite materials technology demonstrator, where the entire airframe was built from carbon fibre reinforced epoxy. The layout of the aircraft is a conventional low wing type with a tandem seating arrangement and a turboprop engine. This aircraft was designed and built to explore the concepts and feasibility of carbon fibre aircraft structures.

1.2 Material Selection

The main aim of project OVID was to develop the technology to design and build this type of aircraft using the expertise available in South Africa. Composites were an attractive choice due to their low tooling cost, low parts count and speed of prototype work when compared to a similar aluminium aircraft. During the initial design phase prepregs showed the greatest promise for the production of composite structures due to their repeatability from a material property and production point of view. The main problem with this type of system was that the availability of prepregs, as well as the technology base for this type of material was minimal in South Africa at the time. Fortunately there was a highly competent team well versed in composite wet lay-up techniques. The cost and speed of prototyping is also less for a wet lay-up system than for prepregs. This made the wet lay-up route the logical one to follow.

Two different carbon fibre fabrics were selected, namely a balanced weave 2 x 2 twill carbon fibre of 240 g/m^2 and a T300 type uni-directional tape of 320 g/m^2. The epoxy resin used in the lay-up was a Ciba-Geigy product, LY 5052. Nomex honeycomb was selected as a core material for skin stabilisation. A thin layer of glass/epoxy fabric was employed on all exterior surfaces to improve damage tolerance.

1.3 Structural Philosophy

The structure consists almost entirely of carbon fibre reinforced epoxy with the assemblies bonded as far as possible.

The cross section through the horizontal stabiliser, as shown in Figure 3, gives a good idea of the structural philosophy of the aircraft. The skins are all Nomex stabilised, which eliminates the requirements for integral stiffeners, thus reducing complexity, parts count and manufacturing costs while improving the damage tolerance. Where a component such as a rib or spar is required to be bonded to the skins, the Nomex is chamfered out at a 30° angle and the facing sheets are joined over the space of the bond. This procedure was also carried out at the leading edge joint. The leading edge joints of the horizontal and vertical tailplanes and the wings have protective glass/epoxy caps bonded to them to improve the impact resistance of these areas.

Internal members such as frames, ribs and spar webs are Nomex stabilised and the facing sheets are continued into bonding lips which form the component into a C shape, as can been seen for the spar webs. These lips are then bonded to the skin or spar caps, as required. For mechanical attachment points, such as the bracket, solid glass/epoxy blocks are substituted for the Nomex in the area of attachment, and the bolts pass through these blocks.

An overview of the wing structure can be seen in Figure 4. The wing is a twin spar layout and is of one piece construction, with the spars continuous across both semi-spans. The spar webs are layed up from

$\pm$ 45°material in order to take the wing shear loads. The spar caps consist of unidirectional rovings bonded between the webs and the skin. In addition to the main spars there are the undercarriage bay spars, which join the main spars over the centre section, forming a box shape spar. The main wing attachments are in this centre section, where two pins pass through glass fibre insert blocks in each of the spars.

The skins, as stated previously, are Nomex stabilised. The facing sheets are of $\pm$ 45° material for carrying the torsional loads on the wings.

The fuel tanks are situated immediately outboard of the undercarriage bays and are integral with the wing. These tanks are demarcated by two ribs, the front spar and a false spar.

The fuselage and vertical fin are integral, where the skins are again Nomex stabilised and laid up at $\pm$ 45° in order to carry the shear and torsional loads.

There are four longerons running the length of the fuselage. These longerons consist of unidirectional carbon/epoxy rovings to take the bending loads. Stabilisation of the longerons is provided by the skins which have a high lateral stiffness.

As can be seen in Figure 5 there are two pairs of frames situated in the cockpit region. These frames serve as the wing and ejector seat attachment points. The skin is discontinued below the floor line between each pair of frames and the wing spars are slotted in, where the wing pins interface directly with the frames. The fuselage torsional loads in these dicontinuous areas are carried by differential bending of the longerons.

The two fin spars continue into the fuselage where they are bonded to a pair of frames. At the rear of the fuselage structure, just behind the fin attachment, there is an attachment frame for the horizontal tailplane. The tailplane is attached to this frame by means of two brackets at the top of the frame that bolt to the main tailplane spar and a rod that connects the hinge line of the tailplane to the bottom of the frame.

1.4 Loads

The aircraft was designed for the following conditions:

Cruise speed	:	240 Knots
Dive speed	:	300 Knots
Aerobatic mass	:	2000 Kg
Maximum all up mass	:	2750 Kg
Positive load factor	:	7g (at 2000 kg)
Negative load factor	:	-3.5g (at 2000 kg)
Horizontal crash	:	30g
Vertical crash	:	20g

The loads calculations on the aircraft were carried out in accordance with FAR 23 except where these loads were exceeded by the specifications.

The fuselage pressure loading, and particularly the loading on the canopy, were determined by means of an aerodynamic panel method program. A plot of the model used is shown in Figure 2.

1.5 Design

The initial structural sizing was done by means of conventional hand stressing techniques. This was simplified by the lay-up directions used, which gave stress states that were relatively easy to determine.

The skins were designed primarily for stresses due to flight loads, buckling criteria and damage tolerance. Over virtually the entire aircraft the damage tolerance case was dominant, with the typical facing thickness being 0.8 mm, which was further reinforced in critical regions.

The critical cases for the fuselage longerons were static strength and buckling, while for the spar caps they were the bending stresses and stiffness to resist flutter.

The frames were sized and positioned in order to introduce wing, tailplane and internal inertial loads into the fuselage and to help resist buckling of the fuselage skins and longerons.

Once the initial structural sizing was complete, a finite element model of the entire airframe was built. This model was used to determine the stresses due to flight loads, buckling and vibration modes and reaction loads at attachment points. An example of this model is shown in Figure 6, which shows the wing root area. In conjunction with this finite element model more refined hand stressing methods were implemented.

No additional reinforcement was required for fatigue resistance in the composite airframe due to the predicted strain values being relatively low from a fatigue point of view. These low values were largely due to the damage tolerance and stiffness requirements.

1.6 Manufacture

Due to the aircraft being a technology demonstrator it was decided to use primarily short-run production moulds and master patterns. A one-step master pattern method was used for all the aircraft components. The larger master patterns were constructed from wood, while room temperature cure wet lay-up techniques were used to manufacture the moulds.

1.6.1 Master Patterns

All the main aircraft components, such as the fuselage, wing, fin, horizontal stabiliser, flaps and control surfaces, had master patterns fabricated from wood. This was an ideal material, because it carved readily and required little labour to eliminate the defects which could adversely affect a pattern. Furthermore, Aerotek possessed a team of experienced pattern makers, making the material an attractive choice.

Jelutong and marine ply were extensively used. The construction approach for all components was essentially the same, with either a marine ply central box spine or jelutong leading and trailing edges used to support the profile frames (see Figure 7). Marine ply board was used to cover the structure. The fuselage pattern, which had complicated curvatures, was splined with 10 mm wood strips over marine ply frames.

The desired surface finish on the patterns was obtained by sanding the rough surface, spraying it with filler and then sanding it smooth again. Once the desired surface finish was attained the patterns were sprayed with a black polyurethane paint, which was then burnished to a near mirror finish.

1.6.2 Mould Design

Due to the large size of the moulds, the criteria applied to their design were stiffness and weight. They were required to be light enough to facilitate ease of handling, but sufficiently rigid to reproduce the desired geometrical tolerance of the design.

The mould laminate consisted of a thin surface layer of gel coat, followed by two layers of twill weave Interglas 92110, and then 10 layers of twill weave Interglas 92140. The glass fibre mould surfaces were supported by egg-crate type structures manufactured from Aerolam F-board. Steel frameworks were manufactured on which the base moulds could rest to achieve a comfortable working height.

1.6.3 Airframe Manufacture

All major and sub-assemblies were formed in their moulds, using wet lay-up techniques, and were then cured at room temperature for 2 to 5 days, followed by a post-cure of 10 to 15 hours at 60°C in an oven to obtain a glass transition temperature of approximately 95 °C.

The bonding of all sub-assemblies into assemblies was done with the epoxy resin LY 5052 and a cotton filler mixture where required, the latter serving as a thickening agent. All Nomex reinforced skins, webs and ribs were bonded with the facing sheets wet onto Redux 410, which improved the peel strength between the carbon fibre and the Nomex.

1.7 Structural Testing

Static structural tests were carried out on the wings, tailplane, fuselage, control system, engine mount and various sub components. Due to the fact that the prototype airframe was required for test flights the structural tests were carried out to limit loads and then all components were inspected for damage.

One exception to the loading philosophy was the wing which was tested for ultimate gust loads which slightly exceeded the manoeuvre limit loads. The critical loads were determined as +7.76g at an aircraft mass of 2000 kg and -4.34g at 1500 kg.

These loads were introduced by means of a conventional whiffle tree arrangement as can be seen in Figure 8. A maximum tensile strain of 2700 microstrain was recorded for the front spar cap at the wing root. At all stages during the loading the flaps and ailerons were checked for jamming, which did not occur. The final test on the wings was to load the undercarriage, in the retracted position, for the +7.76g load to ensure that the locks would function.

The other exception to the test loading was the horizontal tailplane. Two tailplanes were manufactured, with the first being used for destructive testing in order to test the structural philosophy. The loading was again by means of a whiffle tree and the failure occurred at approximately twice the ultimate load. The high strength of the tailplane can be attributed to the same reasons as the predictions on good fatigue life, namely damage tolerance and stiffness requirements.

The fuselage was tested for overall torsion, bending and shear, with the various load cases selected covering the entire design load spectrum. In addition the load introduction points and their supporting structure were tested, where possible as part of the overall fuselage tests. These load introduction points included the interfaces of the wing to fuselage, the engine mount, nose wheel and horizontal tailplane. The engine mount itself was subjected to a separate test for gyroscopic and inertial loads.

The control system was tested for limit pilot loads and simulated vertical inertial loads. Measurements were made of the deflections, friction, hysteresis and overall stiffness.

Following the static tests, ground vibration tests were carried out in order to facilitate the flutter speed predictions.

The final step was to instrument the aircraft for static and dynamic readings and to proceed with a test flight program, of which approximately 80 hours has been completed at the time of writing. In the course of this test flight program the flutter envelope has been completely cleared, the performance tests carried out and limited aerobatics have been performed to date.

2 COMPOSITE MATERIAL DROP TANK

2.1 Introduction

The South African Air Force has used the RP35 type drop tanks extensively on the wing and centreline stations of the Mirage F1 aircraft. These tanks contain approximately 1 200 litres of fuel and are required for long range operations.

In order to investigate the feasibility of local manufacture, Aerotek has constructed three prototype drop tanks from carbon fibre. The first tank was purely for structural tests and contained no fuel or electrical systems. The following two tanks were fully equipped. Figure 9 shows the first prototype.

2.2 Material selection

The original RP35 drop tanks were manufactured in aluminium. Due to the advances in composite material technology it was decided to carry out a feasibility study to determine whether the new drop tanks should be manufactured from composite materials or aluminium. Composite materials were selected for this application for the following reasons:

- Relatively low tooling requirements and cost for complex shapes.
- Good corrosion behaviour with moisture and fuel.
- Lower likelihood to suffer a catastrophic structural failure due to a hostile strike.
- Good retention of strength with time after an impact.
- There is greater scope for dynamic structural tailoring of the new drop tank design to the original so that flutter clearance work does not need to be repeated.

When deciding between glass fibre and carbon fibre for the production of drop tanks, carbon was selected because of its higher strength to mass ratio, implying a lighter structure, and its higher stiffness to mass ratio that would better allow the flutter requirements to be met. The disadvantage of a higher material cost for carbon fibre was partially offset by the laminates requiring fewer layers than an equivalent glass lay-up thereby reducing total material usage and manufacturing time.

Prepregs showed greater promise than a wet lay-up system for production due to repeatability, both from a material property and manufacturing point of view. Due to the experience gained in prepregs at the CSIR by this stage it was decided that this type of system would be feasible for manufacture. After doing a survey of the existing prepregs that could be manufactured in South Africa a plain weave fabric and a unidirectional tape of T300 type fibres were selected.

2.3 Structural Philosophy

The concept of the drop tank that was selected from various options was that of a monocoque structure. With this concept the skin takes the loads without the aid of longerons, local stiffeners or core stabilisation. Where possible the lay-up was made orthotropic in order to avoid problems during manufacture. The skins were made in two halves, with a horizontal split line running through the tank on the horizontal axis. The two halves were bonded together with a double lap joint, where the two skin halves were butted together and lips are bonded to the inside and outside along the joint. Since the drop tank could not be dismantled like the previous aluminium tank, provision had to be made for servicing of the internal components. Two rectangular cutouts were provided, one below each of the float valves. These cutouts provided access to the float valves, the electrical connecter and the transfer valves. The final tank layout is shown in Figure 13.

Frames were situated at all the attachment points, ie for the main attachment lugs, sway brace pads, ball joints and fin attachment points. Where necessary further frames were positioned to resist buckling. The frames were ring shaped, and the flange for attachment to the skin was T shaped in cross section (see Figure 12). These bonding flanges were tapered down towards their ends in order to avoid abrupt changes in stiffness which would adversely affect the bond stresses. For the frames that were used for the main attachment lugs unidirectional material was run from the top to the bottom of the frame in order to aid load transfer into the skin.

The main lugs and their corresponding sway brace attachment points are situated close together. Due to their proximity and the high load transfer between them, the metal fixture for accommodating the main lug and swaybrace points was made as one piece, with the load transfer occurring mainly within this common fixture. This fixture, as with those for the ball joints and tail attachments, is bolted directly to frames.

2.4 Loads

The loads on the drop tank are derived from the inertial loads due to the 1170 litres (931 kg) of fuel and the corresponding aerodynamic loads up to a speed of Mach 0.95 at sea level. The inertial loads were calculated as by MIL-A-8591G. This reference gives an envelope of inertial manoeuvre loads in the vertical and horizontal planes for the store which is general for carriage on any aircraft. These were calculated for both the wing and fuselage mounted positions.

The angles of attack and sideslip were also calculated from MIL-A-8591G for the points corresponding to the inertial load envelope. These angles were then used as input for the USTORE panel method program which calculated the aerodynamic loading corresponding to the particular inertial load cases. A plot of the aerodynamic model used is shown in Figure 10.

The drop tank is held at two main lugs with a 30 inch spacing. Located close to these lugs are the sway brace pads which prevent lateral movement of the tank. Ball joints engage with the front and rear of the pylon. These various attachment points can be seen in Figure 11. The statically indeterminate reactions at these attachment points were calculated by means of the computer program listed in MIL-A-8591E for the initial structural sizing, and once the preliminary design had been done a finite element model was used to calculate these loads more accurately. The two sets of results agreed closely.

Once the inertial, aerodynamic and reaction loads had been calculated, the overall load spectra on the drop tank were drawn up.

In addition to these loads a fuel transfer pressure of 0.8 bar and a proof pressure of 3.0 bar were taken.

2.5 Design

The tank lay-up was sized by means of hand stressing techniques. The initial step was to determine the skin lay-up required to take the shear forces and bending moments from the load spectra. Frames were positioned where required for the attachment points. The lay-up was then checked with these frame positions for buckling using an in-house program for analysing the interactive buckling in shear, bending and torsion using theoretical figures corrected with test results. The areas that proved critical for buckling were then modified by a combination of increasing the skin thickness and addition of frames. The trade-offs between increasing the skin lay-up and adding frames were governed by manufacturing cost, mass and the required vibration modes of the tank.

The detail design of the frames was then carried out. The main criterion for the frames at the attachment points was the stress levels induced by the load transfer. The lay-up of the other frames was determined by the frame stiffness requirements for overall tank stability and the stresses due to Brazier and pressure loads.

The cutouts required extra reinforcement in the regions of the attachment points and the lower access panels. In the case of the cutouts for the attachment points the best reinforcement turned out to be a thickening of the skin, where the overall lay-up of the skin was extended. In the area of the access panels the reinforcement was with angle ply material around the periphery and unidirectional strips on either side to carry the longitudinal loads.

2.6 Manufacture

When designing the tooling for the carbon fibre drop tanks, three main components were identified, namely the shell, internal frames and the stabilizing fins. For all three components the manufacturing approach was similar, namely a one-step master pattern method, (i.e. the master mould is taken off the master pattern without any intermediate steps). This method had the advantage of producing highly accurate components.

2.6.1 Master Patterns

Different techniques were used to construct the master patterns for the tank shell, internal frames and stabilising fins. The reasons for this were the requirements for cost, ease of manufacture and geometrical accuracy.

The first step in constructing the tank shell master pattern was to form a rough profile of the tank around a central spine by bonding on circular polystyrene sheets and marine plyboards, the latter having a stabilising effect. (See Figure 14). A suitable working surface was then formed by splining the profile with a syntactic foam (an epoxy/microballoon mixture). A wet lay-up carbon fibre syntactic foam core laminate was formed around the structure to provide strength and the required stiffness. An N.C. machined lofting template was used to spline the pattern with a gel coat to an accuracy of 0,2 mm. Curing was performed at room temperature at every stage of the manufacturing process, followed by a postcure at 130 °C in an oven. This was advantageous in that all minor distortions, shape changes and warpage were minimised before final splining.

For the internal frame master patterns, circular wooden rings were cut slightly undersize and imprints with syntactic foam applied to the ring edges were taken from the drop tank structural shell. These were sanded and sealed with furane resin.

The fin master patterns were N.C. machined directly from a high temperature tooling block. The patterns were sealed with furane resin, and polished to a high gloss finish.

2.6.2 Drop Tank Moulds

All the moulds were required to operate at temperatures up to 125°C. In order to achieve a good geometrical tolerance on the final structural component, carbon fibre was used for the moulds. This would mean that both the structure and the moulds would heat up and cool down at the same rate and with the same thermal expansion coefficient, meaning negligible warpage and distortion. Carbon fibre tooling prepregs were employed in the manufacture of the shell and fin moulds, while a wet lay-up system was selected for the internal frame moulds.
For the construction of the drop tank skins a total of three master moulds were required. These consisted of two main semi-circular half shells for the skins with approximate dimensions of 5 m x 0.8 m x 0.8 m and a separate bonding lip. (See Figure 16). The laminates for the mould half-shells were laid up symmetrically about the mid-plane to form quasi-isotropic laminates 6 mm thick. Three different types of carbon fibre prepregs of varying thickness and resin content were employed to minimise lay-up time, and also to obtain an excellent mould surface without the use of a gel coat. Avoiding a gelcoat resulted in a virtually maintenance free tool face that would not suffer from the pinholes and thermal cracks sometimes associated with a gelcoat layer. The curing was performed in an autoclave at 130°C and at 7 bar pressure.

Wet lay-up techniques were used for the internal frame moulds. A symmetrical quasi-isotropic laminate was built to a 5 mm thickness using a balanced weave carbon fibre fabric. The moulds were allowed to cure at room temperature, after which the master pattern was removed, and the moulds postcured in an autoclave.

Egg-crate type structures, constructed from carbon fibre honeycomb panels, were used to support the master moulds. The frame moulds required no support structure as a result of their built-in stiffness and compact nature.

2.6.3 Manufacture of Structure

All the solid carbon epoxy laminate components were formed in their appropriate tooling moulds and cured in an autoclave at 120°C and 7 bar pressure. In order to save time the internal bonding lip was co-cured with the top shell.

The internal structure, which included all the circular frames, was bonded to the top semi circular skin shell with a 120 °C curing adhesive. (See Figure 15). Galvanic corrosion between the aluminium fittings and the carbon fibre structure was minimised by means of adding localised layers of glass fibre prepreg in order to isolate the fittings from the structure.

The sealing of the internal structure from the fuel was done with a synthetic rubber coating. Before bonding the two shell halves together all internal plumbing, electrics, metal attachment points, static discharge wiring and anti-slosh foam were fitted to the top tank shell/frame assembly. To achieve the double lap bond on the tank shells, bonding was performed in two stages. Firstly, the top shell assembly was bonded to the bottom shell and cured in an autoclave. Secondly, the outer carbon fibre strap was formed longitudinally around the tank to complete the double lap bond, and then cured again in an autoclave.

2.7 Drop Tank Testing

The first tests to be carried out on the drop tank were for static structural strength. The South African Air Force requirements were used for these tests. These requirements fall within the scope of the design loads.

The specified test loads were as follows:

- A 4 g limit vertical load combined with a 3.7 g limit horizontal load to simulate a rolling manoeuvre with wing mounted tanks.

- A 6 g limit vertical load for the vertical pullout case for all carriage positions.

These loads where factored by 1.5 to obtain the ultimate test loads.

A whiffle tree was used to introduce the combined inertial and aerodynamic loads for these load cases. The tank withstood the ultimate load without any permanent deformation and is now scheduled for flight tests.

The tank tightness test was performed by pressuring up to 2 bar and then checking for the absence of leakage. Structural tests were performed up to 3 bar without failure. Maximum strains of 2500 microstrain were recorded at 3 bar.

The interface and system tests were done by fitting the drop tank on the wing pylon of a Mirage F1 and carrying out the fuel transfer.

3 CONCLUSIONS

At the time of writing the technology demonstrator aircraft has completed approximately 80 hours of test flights. During these tests the strain and vibration levels have been monitored continuously. In addition the aircraft has been subjected to a schedule of extensive ground inspections. During the course of these tests no major problems have been encountered with the structure while all of the performance predictions have been met or exceeded. In light of these results a second prototype is being planned that will make use of prepreg materials.

The drop tank has passed all the required structural, system and interface tests at the time of writing and is being readied for test flying on a Mirage F1 aircraft.

The success of both of the projects outlined in this paper has demonstrated the effectiveness of composite structures for aircraft applications.

FIGURE 1: OVID TECHNOLOGY DEMONSTRATOR AIRCRAFT

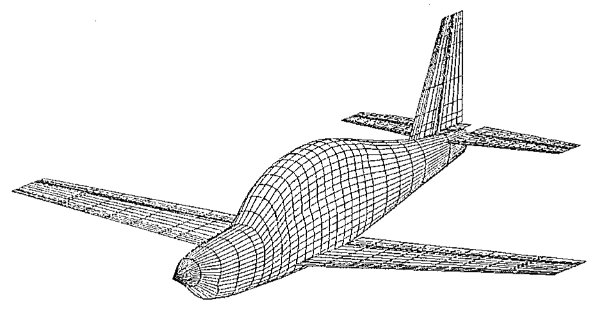

FIGURE 2: AERODYNAMIC MODEL OF OVID

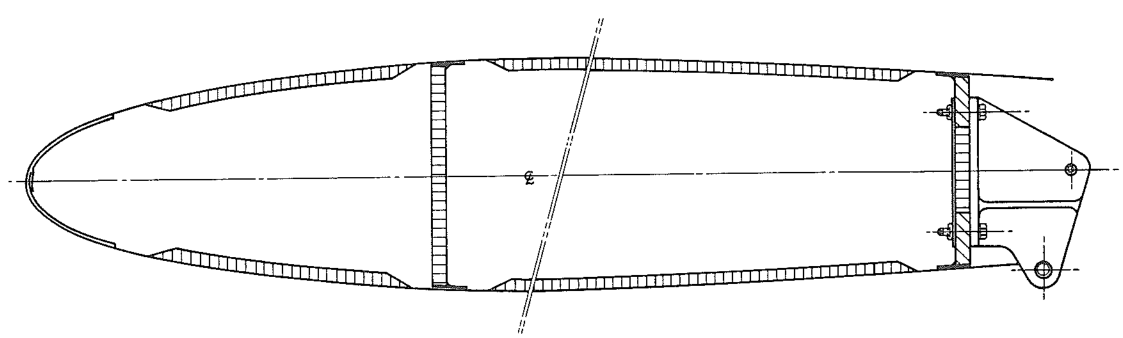

FIGURE 3: CROSS SECTION THROUGH THE HORIZONTAL TAILPLANE

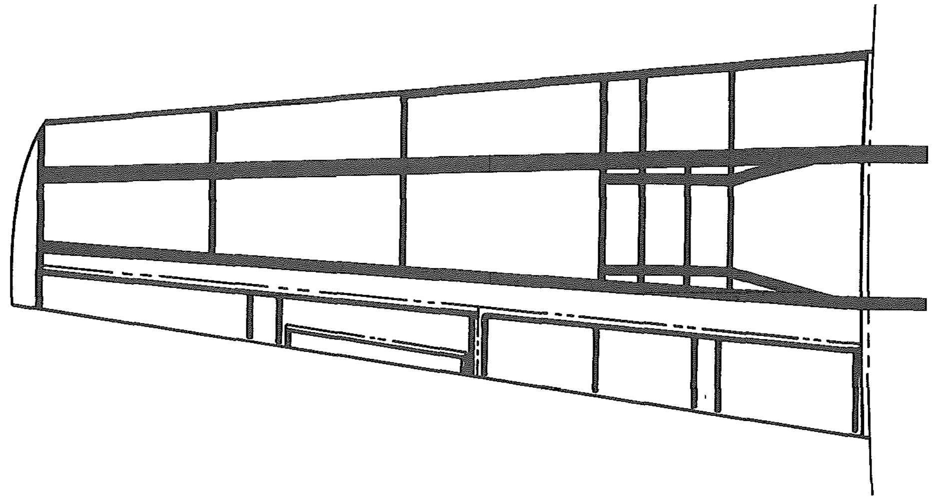

FIGURE 4: PLAN VIEW OF WING STRUCTURE

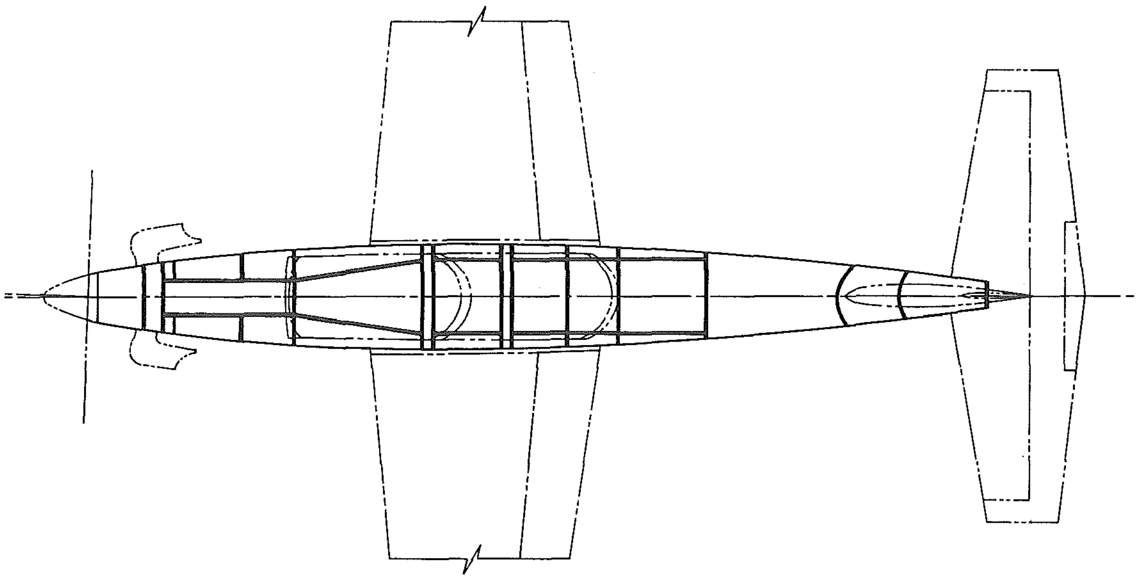

FIGURE 5a: PLAN VIEW OF FUSELAGE STRUCTURE

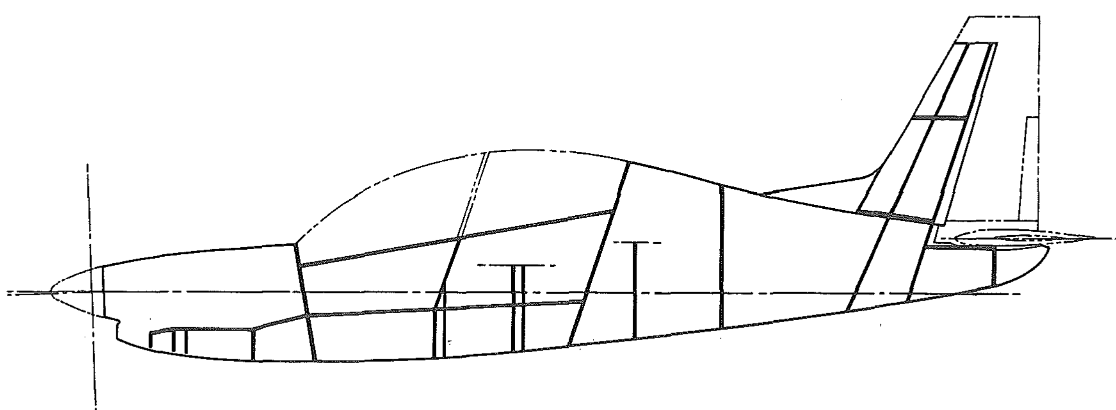

FIGURE 5b: SIDE VIEW OF FUSELAGE STRUCTURE

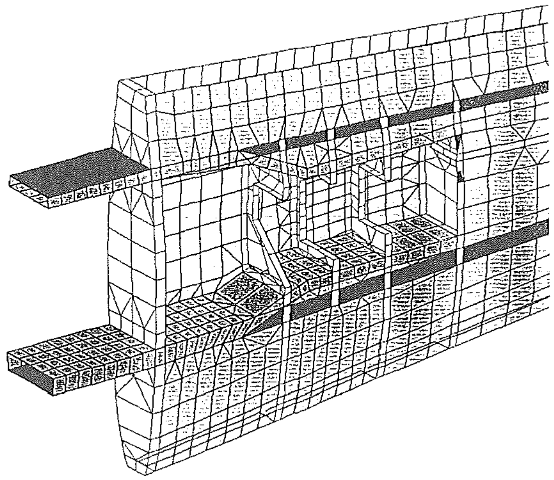

FIGURE 6: FINITE ELEMENT MODEL OF WING ROOT

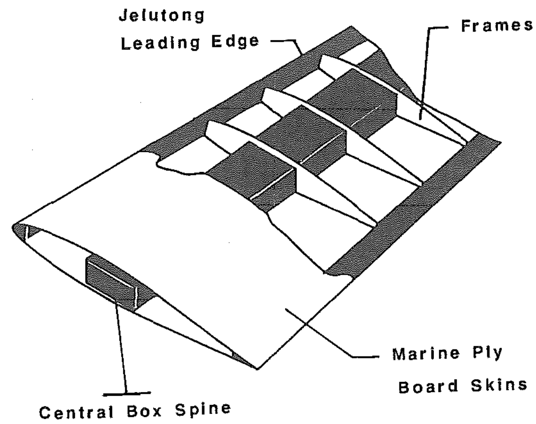

FIGURE 7: OVID WING MASTER PATTERN

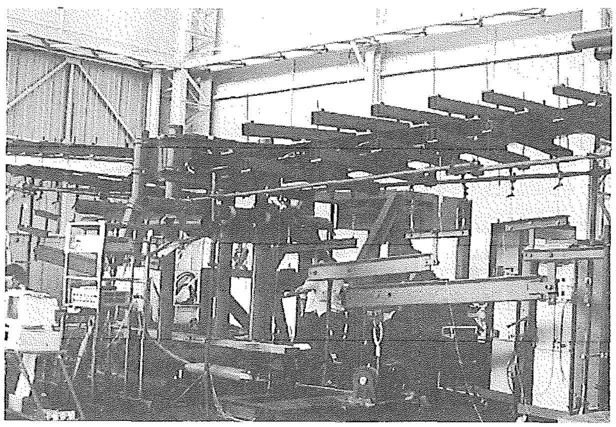

FIGURE 8: WING TEST

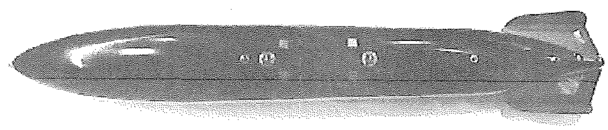

FIGURE 9: CARBON FIBRE DROP TANK

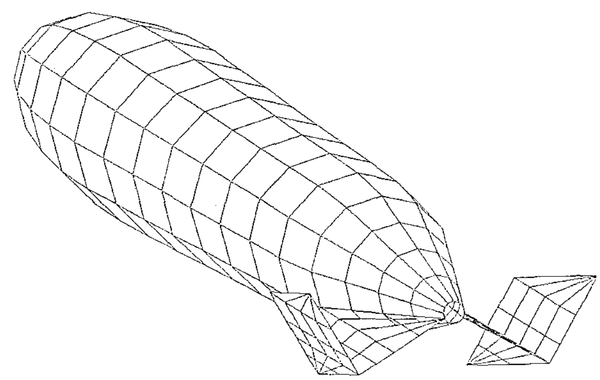

FIGURE 10: AERODYNAMIC MODEL OF DROP TANK

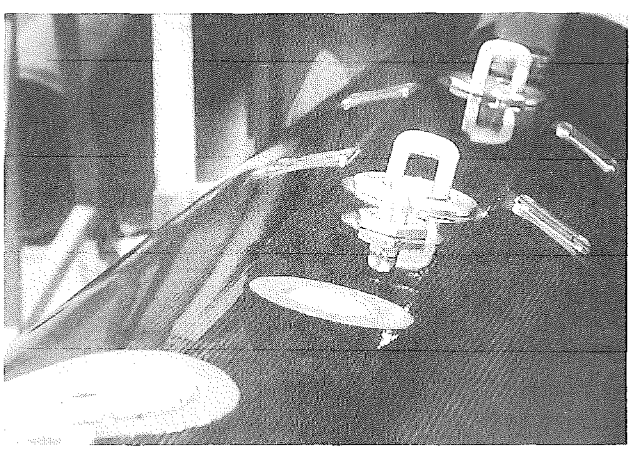

FIGURE 11: DROP TANK ATTACHMENT POINTS

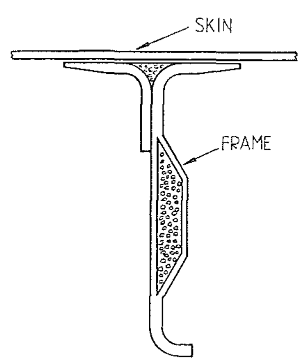

FIGURE 12: CROSS SECTION OF TANK FRAME

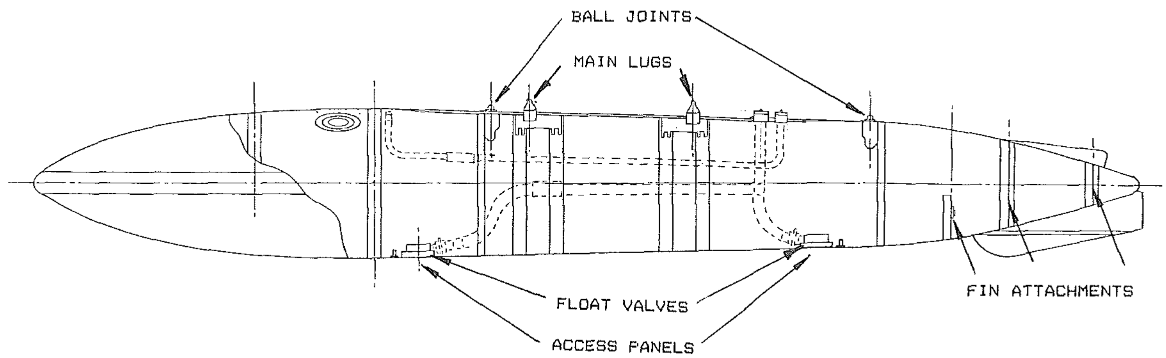

FIGURE 13: CROSS SECTION THROUGH TANK

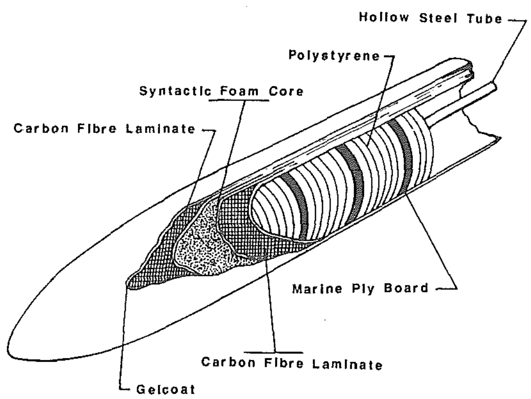

FIGURE 14: MASTER PATTERN FOR TANK

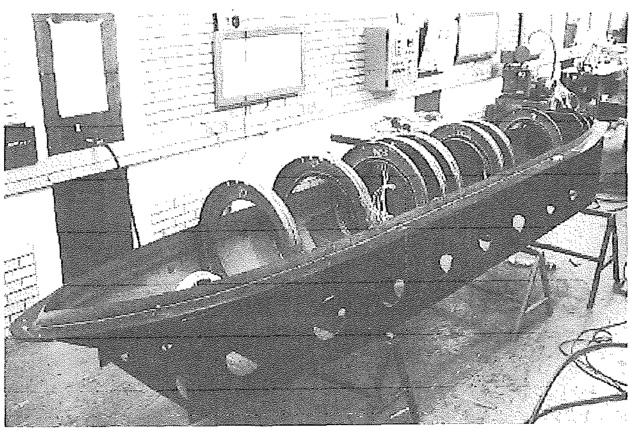

FIGURE 15: SHELL HALF PRIOR TO BONDING

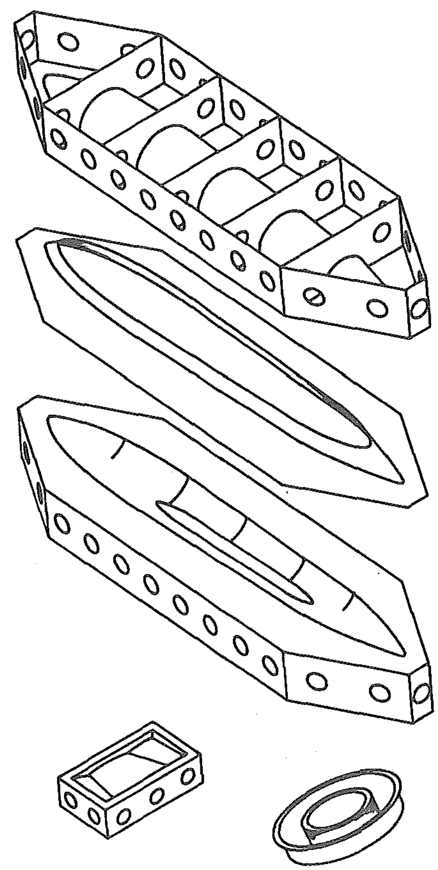

FIGURE 16: DROP TANK TOOLING