

Research Work of the All-composite Fuselage Structure for HondaJet

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ABSTRACT

HondaJet utilizes an all-composite fuselage. The main feature of this fuselage structure is that the stiffened panel and the sandwich panel are co-cured integrally to reduce both weight and cost. Another key feature is that a buckling tolerance design is adopted for the skin. This means that shear buckling is allowed beyond 50% limit load. Several affordable process technologies were developed in the field of lay up and structural bonding. The material and structural tests were conducted applying the concept of “Building Block Approach” to prove safety of flight. First the design allowables of the materials were decided. Environmental, damage tolerance, and fatigue conditions were considered. Three major component tests were conducted.

1. Introduction

An important issue in the structures of aircraft is reducing weight. This has driven an increase in the use of composites in place of aluminum. Typically, a sandwich-panel structure is selected in the design of general aviation aircraft. Sandwich panel structure is effective in reducing the cost and development time due to a simple manufacturing process and a reduction in the number of molds and other manufacturing tools. However, the weight savings compared with conventional aluminum construction are not enough. In addition, there are difficulties with quality control and installation of some equipment.

The MH-02, an all-composite experimental aircraft developed by Honda, made its maiden flight in 1993⁽¹⁾. Both its fuselage and wing were designed and successfully manufactured using sandwich panels. Later, a fuselage barrel

was manufactured with co-cured stiffened panel structure with the objective of further reducing weight and evaluating the manufacturing process. This structure was indeed lighter, but it became clear that applying this method to components with three-dimensional contours would result in higher manufacturing costs.

This report describes the concepts and process technology behind the structural revisions accomplished in the lightweighting programs and cost-cutting measures of the HondaJet that achieved her first flight in December 2003⁽²⁾.⁽³⁾ The procedures for assuring flight safety of the fuselage are also described.

2. Fuselage Structure

The fuselage of the HondaJet is constructed entirely of composites of carbon fiber reinforced plastics (below,

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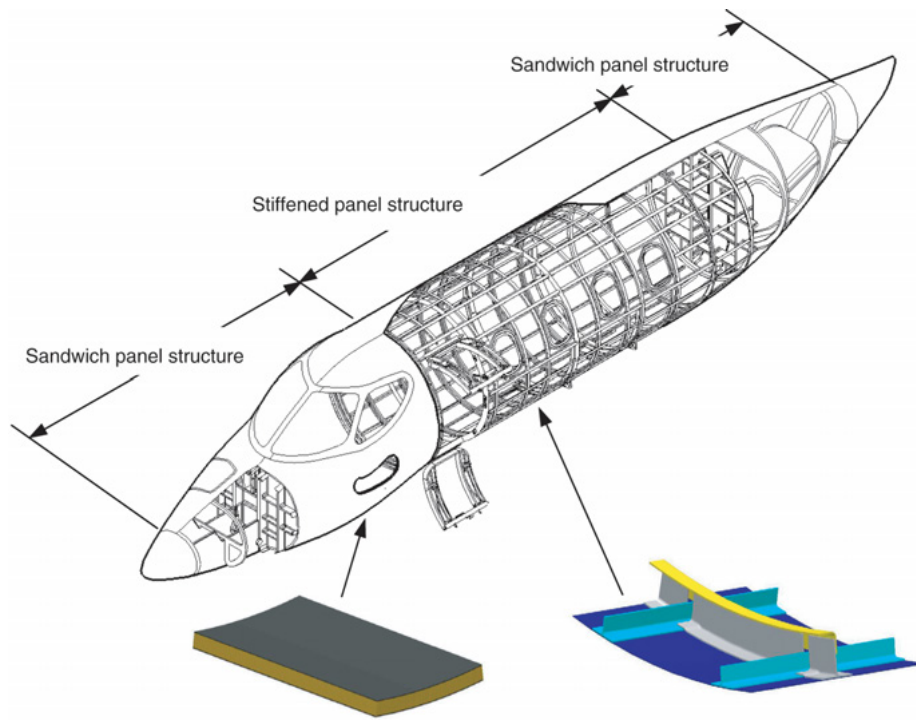


Fig. 1 Structural concept of all-composite fuselage

composite), with the exception of the windshields, the windows, and the metal fittings installed at locations subject to concentrated loads. The cabin is pressurized and has a clamshell type passenger door and a plug-type emergency exit. The design requirements are based on U. S. Federal Aviation Regulations (FAR) Part 23, a set of U. S. Federal Aviation Administration (FAA) standards for small aircraft.

The structural design was conducted to reduce weight and manufacturing costs while assuring adequate strength. Therefore, two structural configurations are used depending on shape: a stiffened panel structure and a sandwich panel structure (Fig. 1). Each panel type is assembled into the appropriate locations for curing in an autoclave as a large integrated whole as shown in Fig. 2. This co-curing process allows the fusing of multiple components into an integrated mass in a single cure step, without adhesives or fasteners. The structure of stiffened panel consists of a skin, stringers, and frames. While the structure of sandwich panel consists of a core material sandwiched between two sheets of skin. The stiffened panel structure, a typical construction for aircraft structures, is quite efficient from the structural view point and well suited for simple shapes such as the cylinder of an aircraft cabin. The sandwich panel structure, a very cost-efficient composite structure, is used for the three-dimensional contours of the nose, cockpit, and tail. In general, the curing pressure for sandwich panels is less than half of that for stiffened panels, which contain no honeycomb, to avoid collapsing the honeycomb core material. To realize this new concept structure, it was essential to cure the sandwich panels at the same pressures as the stiffened panels, so a new curing method was developed.

The second notable feature of this structure is that the buckling tolerant design was adopted for the skin of the stiffened panel beyond 50% of the limit load (the maximum possible load during flight). In decided contrast with other structures, aircraft structures, designed under severe weight requirements, are generally designed to allow some buckling.

In composite structures, however, few examples of

buckling tolerant design exist. Therefore, shear buckling tests⁽⁴⁾ were carried out on sub-components of stiffened panels to verify the post-buckling behavior and the final strength. From these results, a buckling tolerant design method for composite structure was established to select the numbers of skin laminations and fiber orientation necessary to meet design buckling conditions.



Fig. 2 Co-cured skin pane

The left and right skin panels are quite large, measuring 9.74 m in overall length, and are bonded to each other along the top and bottom single lap splices. This means that there are no splices running circumferentially except for the radome and the tail-cone splices. The cross sections for the stringers and the frames of the stiffened panels were designed in consideration of structural efficiency and the manufacturing process. Figure 3 shows those cross sections and the general configuration of intersections of them. The skin, stringers, and T-shaped frame members (white portion in the figure) making up the stiffened panels and the sandwich panels were integrally co-cured into a single, large skin panel. The L-section ring frames (yellow portion) and C-section ring frames (not shown), used where large concentrated loads were applied, were bonded at the same time that the left and right skin panels were bonded together. The height of the frame was limited to 45 mm to maximize available cabin space. Thus, the frame flange was laminated to preserve a modulus of elasticity of at least 80 000 MPa. The honeycomb thickness of the sandwich panels was set at 20 mm to provide the bending strength necessary to withstand cabin pressurization.

A trade study was conducted to compare weights of three fuselages for the reference model of a small business jet.

- (1) Conventional aluminum structure (stiffened panel structure)
- (2) Composite sandwich panel structure
- (3) A new concept involving a sandwich panel and a stiffened panel structure

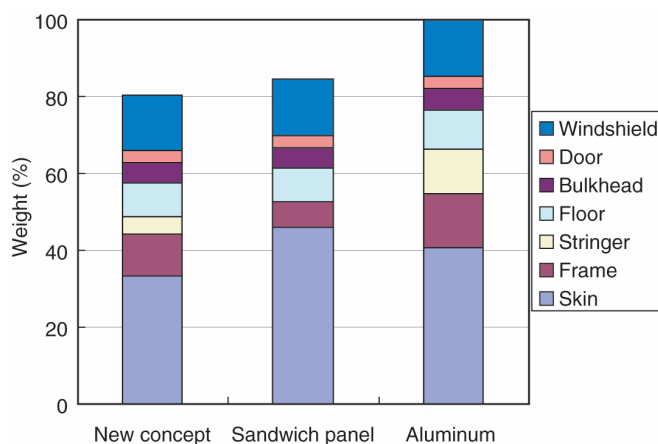
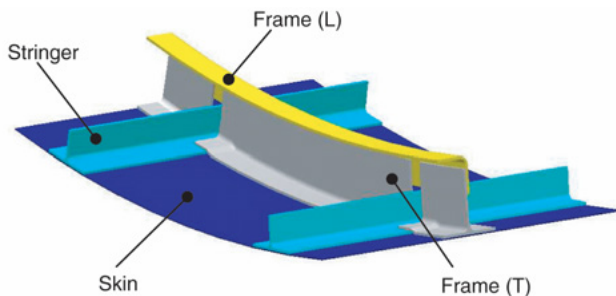


Fig. 4 Weight comparison of fuselage structures

The results of this study are shown in Fig. 4. The sandwich panel structure showed a reduction of 15% in weight over the conventional aluminum construction, and the new concept was predicted to show a reduction of about 20%.

3. Materials and Processes

3.1. Materials

The main material used for the fuselage structure was composite materials. The resin was a toughened epoxy type (Cytec 5276-1), cured at 180 °C and have high toughness and the capability to withstand hot-wet environments. The

Fig. 3 Configuration of stiffened panel

reinforced fibers were unidirectional and plain woven, high strength, and intermediate modulus carbon fiber (Toho G30-50). The raw material is sheet, called a “prepreg”, and consists of a mixture of resin and fiber in a predetermined ratio. The components are designed with fiber-orientation angles and layup number of the prepreg (thickness) to satisfy the required characteristics. Once a component has been laid up on the mold, it is heated and pressurized in the autoclave to complete the forming and curing process. A honeycomb manufactured by Nomex was used as the core for the sandwich panels, and the core and skin were bonded with an epoxy film adhesive (3M FM300) that cures at 180 °C. Also, a resin film containing an expanded copper foil was laid on the outer surface to protect the skin from lightning damage.

3.2. Fabrication

To reduce labor-intensive operations from all phases of composite fabrication for the new concept structure, several new manufacturing processes were developed as follows:

- Press forming technique for pre-compaction parts with pre-laminate
- Step forming technique for joggle of pre-compaction parts
- Edge trimming technique for honeycomb with plugged freeze gel
- Fabrication technique for glass prepreg tool which considered thermal expansion
- Co-curing technique of stiffened panel and sandwich panel

3.2.1. Press forming technique for pre-compaction parts with pre-laminates

The stiffened panels in the fuselage were co-cured simultaneously with the skin, stringers, and frame into a single mass. The stringers of the stiffened panel were prepared by pre-compaction before co-curing. In general, composite parts are laminated layer by layer by hand lay-up, then vacuum bagged and heated. Labor costs of these processes are high. The process shown in Fig. 5 was developed to replace this. In this process, a wide, five-lamina sheet was fabricated in advance, and the machine-cut five layers were press-formed into the desired configuration. The pre-cut pieces were warmed at 90 °C, and the press-forming operation was carried out at 40 °C. This approach eliminated the inconvenience of single-layer lay-ups, vacuum bags, and heating in an oven. Required labor time has been cut to about one-fifth of the previous level.

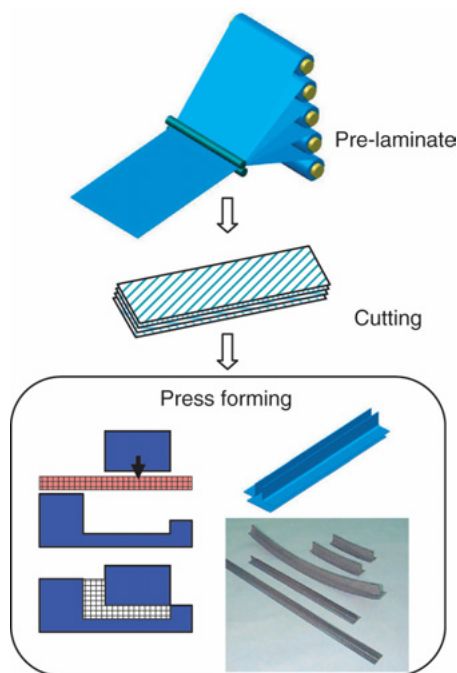


Fig. 5 Pre-laminate and press forming

3.2.2. Co-curing technique of stiffened panels and sandwich panels

Integrally co-cured composite structures are effective to reduce the number of parts and work load in manufacturing. Thus, it was necessary to combine the sandwich panels for the three-dimensional contours in an integral way with the stiffened panels for the cylindrical body (Fig. 6). The stiffened panels were formed under a high pressure (0.6 MPa) to assure quality and avoid interlaminar voids, but this pressure crushed the core of the sandwich panels. So usually, both configurations of panels could not be manufactured together. Potting compound is normally filled into the core to prevent collapse, but this increases weight. The method shown in Fig. 7 was developed to avoid these issues. Picture frame shaped plates 120 mm wide were placed around the edges of the honeycomb core to prevent slippage of the prepreg and stop collapse of the core. This approach allowed formation of an integrated, highly reliable skin panel without increasing weight.

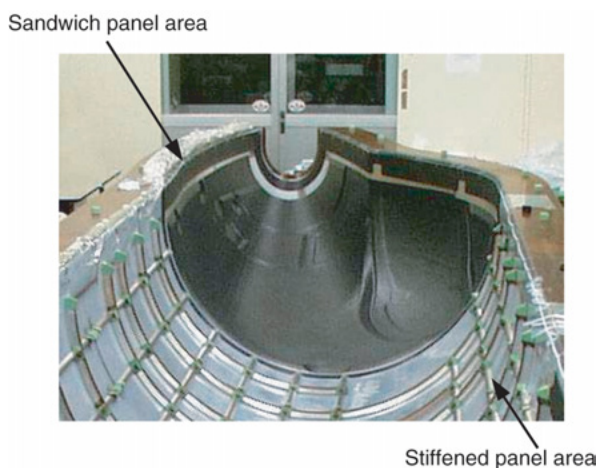


Fig. 6 Skin panel under fabrication

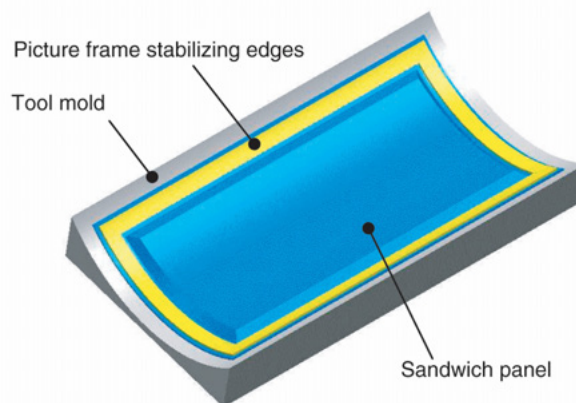


Fig. 7 Method to prevent core crush

3.3. Airframe Assembly

The primary structure of the fuselage was bonded with 120 °C cured epoxy film adhesive. The assembly method/jigs to keep the dimensional accuracy and contact pressure during bonding processes were developed. Figure 8 shows the fuselage assembly jig which was designed to keep accurate geometry of the fuselage and to maintain sufficient contact pressures during cure at 120 °C.

To address issues caused by different thermal expansion between the composite components and the steel jigs during the 120 °C curing process, the assembly jigs were designed to slide longitudinally. At the wing and fuselage joint, where relatively tight dimensional tolerances were called for, jigs were constructed of composite pipe to eliminate differential thermal expansion. These made it possible to hold, for example, a tolerance within 0.25 mm on a dimension of 1 360 mm between the two pins in critical metal fittings that fasten the wing to the fuselage.

Figures 9 and 10 show methods developed to ensure adequate contact pressure between adjoining components during the bonding process. The magnetic force of neodymium magnets were used at right and left skin bonded joint, and the hoop tension of composite bands were used at bonded ring frame assembly. These methods cut the labor time to one-third of that necessary for the same kind of procedure using clecos. Figure 11 shows the fuselage assembly.

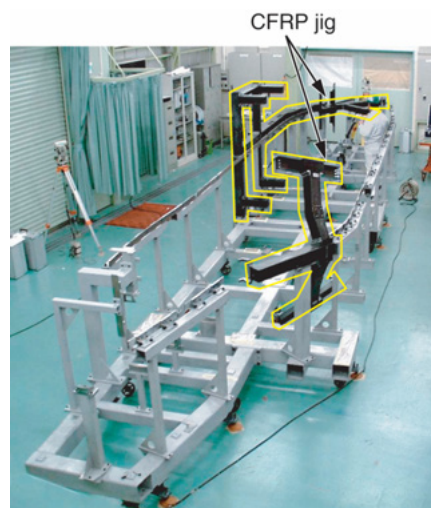


Fig. 8 CFRP/steel assembly jig

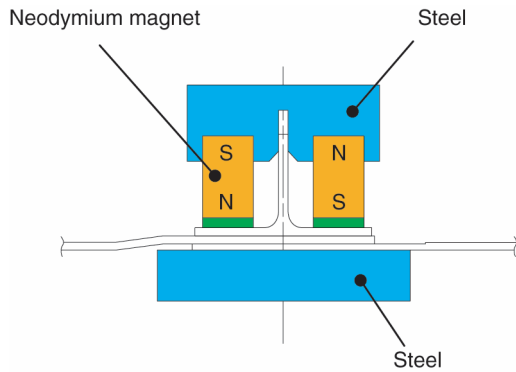


Fig. 9 Bonding method with neodymium magnet

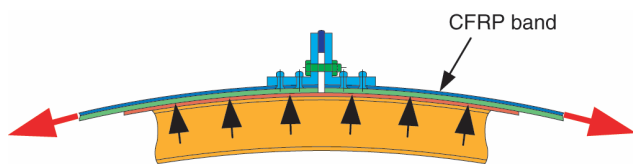


Fig. 10 Bonding method with CFRP band



Fig. 11 Fuselage assembly

4. Flight Safety Assurance

4.1. Procedure

HondaJet is an experimental aircraft, but the material and structural tests of the composite fuselage were conducted applying the concept of “Building Block Approach” based on FAA advisory circular AC20-107⁽⁵⁾. They comprised coupon, element, sub-component, and component tests. The purposes of these material and structural tests were to determine design allowables, to verify the accuracy of the FEA and other analyses, and finally to prove safety of flight for the composite fuselage. Furthermore, the full-scale proof test of the fuselage for flight was conducted to confirm the repeatability of the strain level to compare with component test results.

4.2. Design Allowables and Coupon Test

We considered “Damage tolerance”, “Environmental condition (temperature and humidity)”, and “No-growth damage under the fatigue” to determine material design allowables. A lot of data from coupon tests were obtained for these purposes. The coupon test specimens were subjected to impact damage, and the strength tests were conducted under hot-wet condition. Figure 12 shows the relationship between compressive failure strain and impact damage under hot-wet condition. The compressive design allowable was determined with these test data by considering barely visible impact damage and the worst environmental condition.

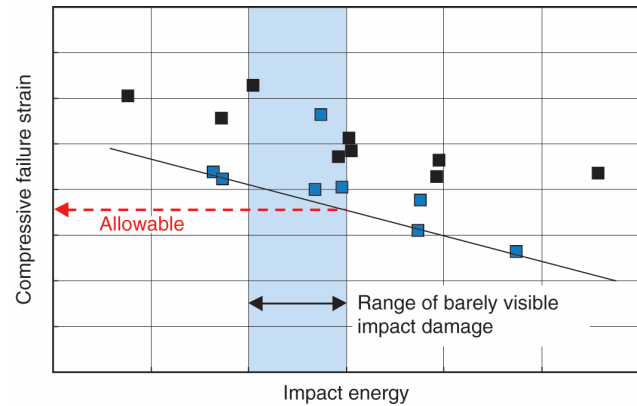


Fig. 12 Compression design allowable

4.3. Element Tests

The 17 strength-critical elements in the structure were selected and tested. The main objective in element tests is to verify the strength of already impact-damaged critical bonded joints in hot-wet environments. Other specific elements were tested such as cutouts in the heavily loaded frame web, skin joggles around the windshields, and transition area between sandwich panels and stiffened panels. The lightning protection system in the skin was also tested. The resin layer containing the expanded copper foil described above showed good protection capability with a minimal weight penalty.

4.4. Sub-component Tests

In the building block approach, before the component tests, which are the highest level, the following five essential structures were selected for testing at the sub-component level.

- (1) Stiffened panel (tests for shear/compressive buckling, static strength)
- (2) Stiffened panel with window (tests for shear buckling, static strength)
- (3) Passenger door (tests for static strength, fatigue strength, fail safe strength)
- (4) Wing and fuselage joint (tests for static strength, fatigue strength)
- (5) Windshield (tests for static strength, fail safe strength)

All composite sub-components were subjected to barely visible impact damage to verify damage tolerance at critical points.

The shear buckling test of the stiffened panel was conducted to evaluate the post-buckling design and load carrying capacity after buckling (Fig. 13). The sub-components were 0.8 m squared curved panels with the same

structural detail and manufacturing process as the actual structure. The picture frame type jig to apply the shear load on the panel was designed to avoid warping due to curvature of the panel during the test.

Buckling load of finite element analysis (FEA) showed good agreement with the test result. The final strength shown by the panels was at least 4.5 times greater than their buckling load, and the failure mode was compressive failure of the skin. No clear delamination of the stiffener from the skin was observed until the collapse.

The objective of the passenger door sub-component test (Fig. 14) was to assure pressurization fatigue strength under three lifetimes, i. e., 60 000 cycles and air-tightness. The door was attached to a steel pressure vessel, and the pressure was cycled with a blower. Pressurization fatigue strength was confirmed for all components except the metal lugs connecting the upper and lower doors. The lugs were then re-designed, and fatigue strength was confirmed after additional element test.

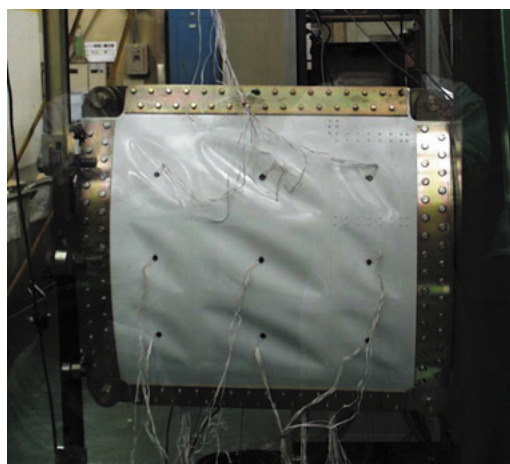


Fig. 13 Shear buckling test of stiffened panel



Fig. 14 Sub-component test of passenger door

4.5. Component Tests

The objective of component tests was to prove the strength of the fuselage by full-size components. This corresponds to the top of the building block. The fuselage was divided into the three following components in consideration of manufacturing and testing facilities and the functions of the fuselage.

- (1) Nose component (proof of nose landing gear attachment)
- (2) Cabin component (proof of pressurized cabin for bearing flight loads and pressurized loads)
- (3) Tail component (proof of vertical tail attachment)

Barely visible impact damage was applied to critical points of the components. The tests were carried out at standard room temperature conditions due to the limited test facilities. A knockdown factor was added to the test load to compensate for the expected decrease in strength of the components in hot-wet environments.

The nose component included the installation of a dummy nose landing gear attached to the front bulkhead. A hydraulic actuator applied three cases of landing loads through the dummy nose landing gear. The nose component carried these loads successfully.

The cabin component was the pressurized section between front and rear bulkheads of 5.8 m length including the passenger door, emergency exit, windshields, and windows. The main objectives of the cabin component test were to measure the strains, displacements, reaction forces, and skin buckling load to verify the accuracy of FEA, and to prove the static strength and three lives pressurized fatigue strength. The test procedure was as follows:

- (1) Limit load test of four cases of flight and pressurized loads
- (2) Pressurized fatigue test for three lifetimes (60 000 cycles of normal operating load) after impact damage in critical points of the composite
- (3) Ultimate load test (1.5 times the limit load), residual strength (final strength) test

Five servo-controlled hydraulic actuators simulated flight loads and a blower applied pressurization (Fig. 15). Figure 16 shows buckling behavior under the limit load. Shear buckling occurred at the design target, 50% limit load, as shown in Fig. 17. Observation of strain revealed less than 10% error in the prediction by FEA. These tests also showed that the impact damages did not grow under three lives cyclic pressurized load. On the other hand, cracking occurred in a few fasteners, metal parts of the door, and the acrylic windshields. These were repaired or replaced during testing. The ultimate load test and residual strength test were carried out after the pressurized fatigue test.

The tail component was 2.6 m long with a dummy aluminum empennage. The main objectives of the tail component tests were to prove static strength of the fuselage to empennage joints. Front end of the component was clamped to the test frame, and loads were applied through the dummy empennage by six hydraulic actuators (Fig. 18). First, limit load tests were carried out at the critical three load cases. Because impermissible buckling occurred during some loading conditions near the empennage attachment, the affected locations were reinforced with laminate plate, and testing was continued to prove the strength of the component. This concluded the tail component test.

The necessary reinforcement during the component test was reflected in revisions to the design of the fuselage.



Fig. 15 Cabin component test



Fig. 18 Tail component test



Fig. 16 Skin buckling at limit load

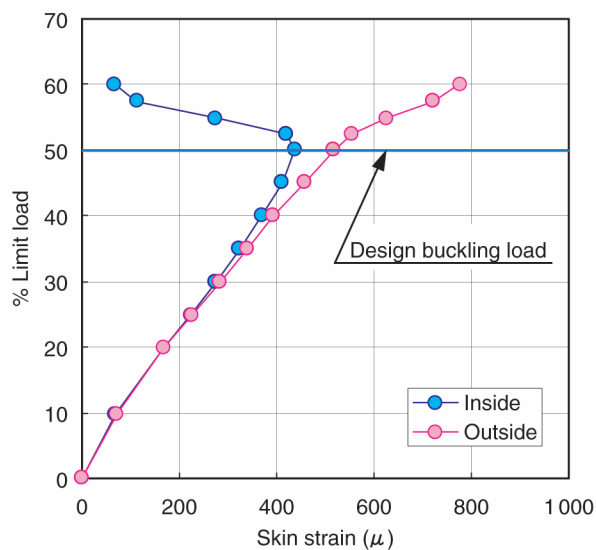


Fig. 17 Load vs. skin strain

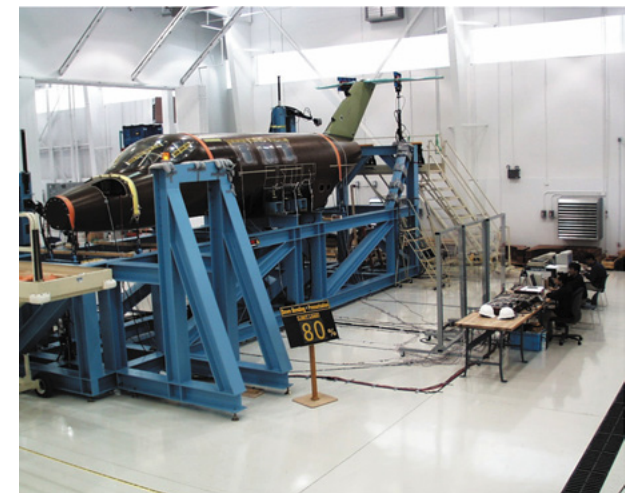


Fig. 19 Fuselage proof test

4.6. Proof Test

Proof test was carried out to confirm that the results of the three component tests would be reproduced in the entire fuselage used for the actual flight. The fuselage structure used for the test is used for the flight program so the test load was limited up to 80% limit load. Six load cases were selected: pressurization, positive gust, positive gust with pressurization, rudder maneuvers, rudder maneuvers with pressurization, and two-point landing. These were applied by the six actuators of MTS Aero90 system. It simulated design flight load as close as possible (Fig. 19), and the fuselage bore all load cases.

The strains, displacements, and reaction forces were measured at representative locations, and these results were compared with those from both component tests and FEA. The reproducibility of the component tests and validity of FEA were confirmed. The strain values extrapolated from strain observed at the critical points were also confirmed to be within design allowables.

5. Conclusion

An all-composite fuselage was developed using a new concept involving the integral co-curing of assembled stiffened panel and sandwich panel. And the first flight was performed successfully. The following results were reported.

- (1) The new structural concept and buckling tolerant design permitted the structure to be lightened by 20% in comparison to conventional aluminum structure.
- (2) Newly developed process technology cut manufacturing costs in both fabrication of components and assembly with adhesives.
- (3) Structural analyses and tests from the coupon level to the component level were carried out, in the same “building block approach” used for certification of an aircraft. Furthermore, the proof test of the fuselage for flight was conducted. As a result, safety of flight for the composite fuselage was proven.

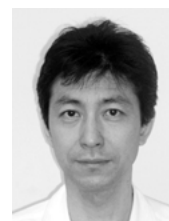
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