FABRICATION AND EVALUATION OF ADVANCED

TITANIUM AND COMPOSITE STRUCTURAL PANELS

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SUMMARY

Advanced manufacturing methods for titanium and composite material struc-
tures are being developed and evaluated by NASA in support of the Supersonic
Cruise Aircraft Research Program. The focus for the manufacturing effort is
the fabrication of full-scale structural panels which replace an existing shear
panel on the upper wing surface of the NASA YF-12 aircraft. The program, with
the Lockheed-California Company, Advanced Development Projects Division, as
prime contractor and support from the Dryden Flight Research Center, involves
design, fabrication, ground testing, and Mach 3 flight service of full-scale
structural panels and laboratory testing of representative structural element
specimens.

Results discussed include the manufacturing methods and test results for
weldbrazed and RohrBond titanium panels fabricated by aerospace contractors and
the development of fabrication methods for producing Borsic/aluminum and
graphite/PMR-15 polyimide panels at LaRC. Test data presented on the titanium
panels include results obtained from flight service on the YF-12 aircraft and
from ground-exposure to 589 K (600°F) for 10,000 hours. In-house fabrication
studies include the development of a satisfactory brazing process for Borsic/
aluminum and an expandable rubber process for forming graphite/polyimide
components. Test results are also presented for a Borsic/aluminum design
verification panel.

INTRODUCTION

The development of efficient future supersonic aircraft is dependent on an
advanced technology base in many areas. In order to develop this technology,
NASA has initiated a multidisciplined Supersonic Cruise Aircraft Research
(SCAR) program. In support of SCAR, the Manufacturing Technology Section at
the Langley Research Center has implemented studies on the development and
evaluation of advanced fabrication methods for titanium and high temperature

composite materials. To insure that viable fabrication methods were being studied, primary structure shear panels for the upper wing surface of the Mach 3 NASA YF-12 aircraft (fig. 1) were selected for manufacturing development. Panels were designed to the YF-12 criteria and were extensively ground tested and evaluated in flight service on the YF-12 aircraft. Exposure and testing of specimens was a joint activity of NASA Langley Research Center, NASA Dryden Flight Research Center, and Lockheed's Advanced Development Projects Division (ADP). Results discussed include the manufacturing methods and test results for weldbrazed and RohrBond titanium panels fabricated by aerospace contractors and the development of fabrication methods for producing Borsic/ aluminum and graphite/PMR-15 polyimide panels at LaRC.

SPECIMENS AND TEST CONDITIONS

The test program consists of exposure of element-type specimens and full-scale wing panels to various temperature environments for up to 10,000 hours and subsequent static testing. The types of specimens and tests are depicted in figure 2. For skin-stringer panel designs, lap shear and compression crippling specimens were investigated to determine the integrity of the joining process used to attach the stringers to the panel face sheet. For honeycomb-core panel designs, flatwise tension specimens were used. The full-scale 406 x 711 mm (16 x 28 in.) wing panels were tested in shear, which is the primary type of loading experienced on this component on the YF-12 aircraft.

The ground exposure program and test schedule are outlined in figure 3. The element type specimens were tested at room temperature after constant temperature or cyclic exposure. The titanium specimens were exposed at constant temperatures of 478, 589, 700, and 811 K (400°, 600°, 800° and 1000°F) for periods up to 10,000 hours. Exposure temperatures for the composite specimens were 431, 478, 533 and 589 K (300°, 400°, 500° and 600°F) because this class of materials is considered to have a lower use temperature than titanium. For cyclic exposure, the titanium specimens were exposed for 1000 cycles of the temperature profile shown on the lower portion of figure 3. This temperature profile simulates a three hour flight with two hours at supersonic cruise. Composite specimens were exposed to a similar profile with the exception that the maximum temperature was 533 K (500°F) rather than 589 K (600°F). Cyclic exposures were conducted both in an environmental chamber at constant sea level pressure and in a chamber which simulates the pressure associated with supersonic flight.

Full-scale wing panels were tested in shear at room temperature following exposure for up to 10,000 hours at a constant temperature of 589 K (600°F) for the titanium panels and 533 K (500°F) for the composite panels and after cyclic exposure for 1000 cycles. In addition, flight qualification panels were tested at ambient and elevated temperature to verify compliance with design requirements.

FABRICATION METHODS

The program to date includes four different YF-12 wing panel designs; two of titanium and two of high temperature composites as follows: (1) weldbrazed titanium skin-stringer, (2) RohrBond titanium with honeycomb-core, (3) Borsic/aluminum with titanium honeycomb-core, and (4) graphite/PMR-15 with polyimide glass honeycomb-core. Fabrication and testing of the two titanium panel designs have been completed while the fabrication of the composite panels are still under development.

Titanium Panels

Weldbrazed skin-stringer panel.- Weldbrazing is a process developed in the Manufacturing Technlogy Section at the Langley Research Center which combines resistance spotwelding and brazing to produce a continuous high strength joint (ref. 1). Following in-house development, the technology was transferred to Lockheed-ADP who successfully designed and fabricated the weldbrazed skin-stringer panels which are represented schematically in figure 4. The titanium "Z" stiffeners were first spotwelded to the face sheet, aluminum braze alloy was placed along the edge of the stiffeners, and the assembly was brazed in a vacuum furnace at a brazing temperature of 950 K (1250°F) for 10 minutes in a vacuum of 1.3 mPa (10^{-5} torr). Under these conditions, the braze alloy melts and flows by capillary action into the faying surface gap. The weldbrazed panel weighed 3.9 kg (8.5 lb), the same as the integrally stiffened panel it was designed to replace.

RohrBond titanium honeycomb-core panel.- RohrBond is a Rohr Industries, Inc. titanium joining method that uses a "proprietary" Liquid Interface Diffusion (LID) process consisting of selectively electroplating the components to be joined with several layers of material which act as an eutectic to aid diffusion bonding (ref. 2). Bonding, or diffusion, is accomplished in a vacuum of 6.7 mPa (5 x 10^{-5} torr), at 1208 K (1715°F) for 1.5 hours. Construction details for the YF-12 panels, designed and fabricated by Rohr Industries, are shown in the lower portion of figure 4. The panel consists of a titanium frame, fabricated from four machined edge members and fusion welded at the corners, a titanium honeycomb-core, and titanium face sheets. The honeycomb-core is positioned in the frame, cell walls are resistance welded to the frame, face sheets are positioned, and the assembly is placed on the vacuum furnace hearth. Tungsten pellets are used for dead weight loading to provide the pressure required for bonding. The RohrBond panels weighed 3.2 kg (7.1 lb), 12 percent less than the weldbrazed and integrally stiffened panels.

Composite Panels

Borsic/aluminum titanium honeycomb-core panels. The design and fabrication of the Borsic/aluminum (B_{SC}/Al) panels is an in-house program at the Langley Research Center. The design concept selected to meet the YF-12 panel requirement is shown in the upper portion of figure 5. The concept consists of upper

and lower skins of B_{SC}/Al brazed to a titanium honeycomb-core and frame assembly. The orientation of the Borsic filaments in the composite is +, -, -, +, $\pi/4$ rad (45°) to sustain the shear load requirements. The ramp of the Ti-6Al-4V titanium alloy frame was capered at an angle of $\pi/6$ rad (30°) to aid in the introduction of load into the inner skin. Titanium Ti-3Al-2.5V alloy honeycomb-core was selected because of its mechanical properties and its amenability to brazing. The braze alloy selected to join the B_{SC}/Al skins to the honeycomb-core frame assembly was 718 aluminum. This alloy was selected because of its favorable melting temperature and its wetting and strength characteristics. In order to minimize the effect of interaction of the braze alloy with the 6061 aluminum alloy matrix and Borsic filaments of the composite, the skin material contained a 0.127 mm (0.005 in.) layer of 1100 aluminum alloy on the braze surface to serve as a diffusion barrier. A panel of this design is approximately 30 percent lighter than the original YF-12 titanium panel.

Graphite/PMR-15 polyimide glass honeycomb-core panel.- Design and fabrication of graphite/PMR-15 polyimide (Gr/PI) composite panels are also underway in-house at Langley. The design concept is similar to that of the Borsic/aluminum panel and is shown in the lower portion of figure 5. The panel consists of an HT-S graphite/PMR-15 polyimide upper skin and pan assembly bonded to a glass/polyimide honeycomb-core using LaRC-13AI adhesive which was developed at LaRC. The ply orientations shown in the figure were chosen to carry the required loads. The estimated weight of this design is approximately 1.9 kg (4.25 lb) or about 50 percent less than the original YF-12 titanium panel.

Borsic/aluminum braze tooling.- The tooling used for the successful fluxless brazing of the B_{SC}/Al titanium honeycomb-core panel is shown in figure 6. Shown are an upper and lower platen, a caul sheet, a pressure bladder, fiber-frax insulation, and titanium honeycomb-core tooling. Following assembly, the panel and tooling are placed in a vacuum furnace for brazing. Brazing is accomplished in a vacuum of a 0.67 mPa (5 x 10^{-6} torr) and a temperature of 860 K (1090°F). During brazing the bladder was pressurized using inert gas to a pressure of 27.6 kPa (4 psi) to maintain alinement of mating parts. Since the bladder concept has a low thermal mass compared to other concepts, such as dead weight loading, the time at temperature and degradation of the composite materials from the thermal exposure were minimized.

Gr/PI rubber compaction process.- The tooling utilized to consolidate the contoured Gr/PI pans using the rubber compaction process is shown in figure 7. The graphite/PMR-15 polyimide is first "B" staged over the male die and the RTV expandable rubber is precast to fit the tooling. Following assembly of components the tooling is placed between heated platens in a hydraulic press. As the assembly is heated, the rubber expands and exerts a uniform hyrostatic pressure on the composite. At the cure temperature of 589 K (600°F) a pressure of approximately 4.1 MPa (600 psi) was applied. This concept has the advantage of applying uniform pressures over a contoured surface and is less expensive than matched die molding for a limited number of parts.

RESULTS AND DISCUSSION

The results reported herein consist of those obtained from full-scale panel tests and observations concerning the development of manufacturing methods for composite structures. Data obtained from the representative element tests for the weldbrazed and RohrBond processes are reported in references 3 and 4.

Titanium Panels

Weldbrazed panels.- Results obtained from shear tests of the full-scale weldbrazed panels are presented in figure 8. Shear tests were conducted at ambient temperature and 589 K (600°F) for exposed and as-fabricated panels. Exposures for the ambient temperature tests were as follows:

> no exposure
> 100 hr at 589 K (600°F)
> 1000 cycles at 219 K to 589 K (-65° to 600°F) constant pressure
> 1000 cycles at 219 K to 589 K (-65° to 600°F) varying pressure
> 10,000 hr at 589 K (600°F)
> 106 hr flight service on YF-12, 31.7 hr above Mach 2.6

As shown in the figure the ambient-temperature strength of all panels exceed the design ultimate shear strength of 680.3 kN/m (3885 lb/in.) by approximately 30 percent and the elevated temperature strengths exceeded the design allowable by approximately 70 percent. Based on these results, weldbrazing is considered to be a satisfactory process for fabricating structures suitable for use on supersonic aircraft.

RohrBond panels.- The results of shear tests of the full-scale RohrBond honeycomb-core sandwich panels are shown in figure 9. Exposure conditions for the panels were the same as for the weldbrazed panels with the exception of flight service testing. The panel flown on the YF-12 aircraft accumulated 40 hours of flight service exposure, with 14.3 hours at speeds above Mach 2.6. Panel strengths at ambient temperature exceeded the design ultimate by approximately 20 percent while those tested at 587 K (600°F) exceeded the design value by approximately 30 percent. These results indicate no degradation following either long term thermal exposures or flight service.

Panel cost estimates.- Cost estimates for machined titanium integrally stiffened panels, weldbrazed panels, and RohrBond panels are compared in figure 10 in terms of unit selling price in 1975 dollars. Data for the machined panel are based on a learning curve of approximately 90 percent, while data for the weldbrazed and Rohrbond panels are based on learning curves of 88 percent. For 100 panels, the data show the Rohrbond panels cost approximately 3.5 percent more than the machined-integrally stiffened panels, while the weldbrazed panels cost approximately 15 percent less. However, the mass of the RohrBond panel (3.2 kg (7.1 lb)) was less than that of the weldbrazed and original machined panels (3.9 kg (8.5 lb)).

Borsic/aluminum titanium honeycomb-core panels.- Development of a successful brazing process for fabricating Borsic/aluminum has only recently been completed; therefore, the only full-scale test to date is for a design verification panel. The results of this test are shown in figure 11 where the applied load is plotted against shear strain. The shear strains were calculated as the sum of the absolute readings of two $\pi/4$ rad ($\pm$ 45°) gages located in the center of the panel. The panel was tested by loading to design limit 267 kN (600 kips), unloading to 89 kN (20 kips), and reloading incrementally to failure. As shown, the shear stiffness on first loading was approximately 50 percent below that obtained on second loading. This inelastic behavior is apparently characteristic of crossplied metal matrix composites and has been attributed to the state of residual stresses in the composite material (ref. 5). On second loading, the panel failed at 125 percent of design ultimate load or approximately 434 MPa (63 ksi) ultimate shear stress in the composite skin. This panel failure shear stress compares favorably with test results on as-received material which indicates that the B_{SC}/Al was not degraded by exposure to the brazing environment.

The adequacy of the brazing process for fabricating B_{SC}/Al was also investigated by metallurgical analysis. Two photomicrographs showing the brazed joint between the B_{SC}/Al skin and the titanium honeycomb-core of the panel are shown in figure 12. The needlelike or acicular silicon particles of the 718 aluminum braze alloy are shown to stop at the 1100 aluminum surface of the B_{SC}/Al, indicating that interaction between the braze material and the constituents of the composite has been avoided. Therefore, degradation of the composite properties associated with exposure to a brazing environment have been minimized. A more detailed description of the braze-composite interaction is presented in reference 6.

Based on these results the brazing process developed for joining B_{SC}/Al composite appears satisfactory. Using this process, a B_{SC}/Al panel has been fabricated which fully complies with the ambient temperature design requirements of the YF-12 aircraft.

Gr/PMR-15 PI glass honeycomb-core panels.- The technology development for Gr/PI panels is at the process verification panel fabrication stage. A design verification panel has not been fabricated to date due to problems associated with obtaining consistent quality prepreg. However, recent results indicate that the problems can be rectified. Employing the technology developed to date, the process verification panel shown in figure 13 has been fabricated. This technology includes development and application of the rubber compaction process to polyimide materials for consolidating the Gr/PMR-15 PI inner pan of the panel and the development and application of LaRC-13AI polyimide adhesive for bonding Gr/PI to glass/polyimide honeycomb-core. NDE results indicate that consolidation of the composite by the rubber compaction process and the bonding between the composite laminates and honeycomb-core were successful.

CONCLUDING REMARKS

An overview of the SCAR YF-12 Panel program has been presented. This program is a joint activity between NASA Langley Research Center, NASA Dryden Flight Research Center and Lockheed. Manufacturing methods for fabricating advanced titanium structures have been evaluated and methods for fabricating high temperature composite materials are being developed. Evaluation of the structures fabricated include both flight service on the Mach 3 YF-12 aircraft and ground testing after exposure to a simulated supersonic aircraft environment. The following comments are based on the results obtained to date:

Evaluation of two advanced titanium panel concepts has been completed. Panels fabricated by weldbraze and Rohrbond processes have performed successfully in Mach 3 flight service and in ground tests after ground exposure to 587 K (600°F) for 10,000 hours.

A satisfactory process has been developed in-house at LaRC for the fluxless brazing of B_{sc}/Al using 718 aluminum alloy braze which minimizes degradation of the composites mechanical properties. This process has been used to fabricate a B_{sc}/Al titanium honeycomb-core structural panel that meets the ambient temperature design requirements of the SCAR YF-12 panel program.

Fabrication processes have been developed that appear capable of producing full-scale graphite/PMR-15 polyimide glass honeycomb-core panels for inclusion in the structural panel program.

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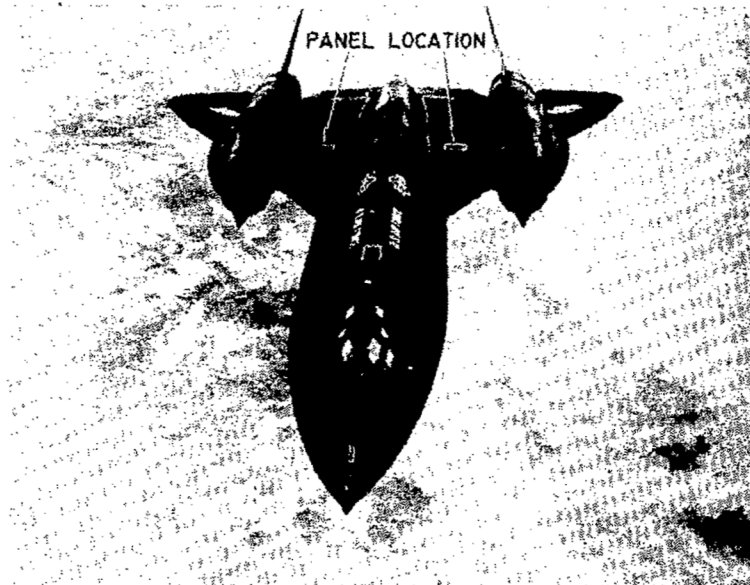
Figure 1.- Panel location on NASA YF-12.
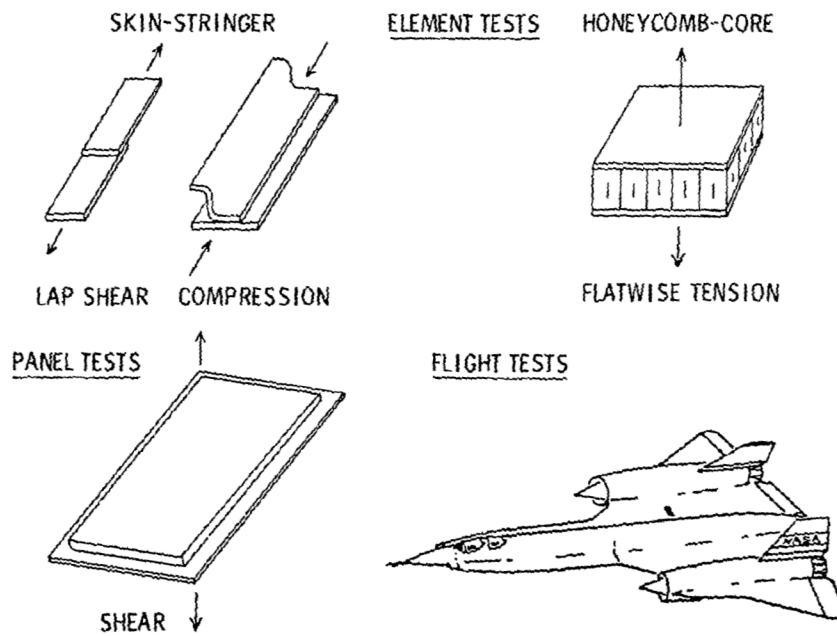
Figure 2.- Specimen types and test methods.

CONSTANT EXPOSURE

SPECIMEN TYPE	TEST TEMP., K (°F)	EXPOSURE TEMP., K (°F)	EXPOSURE TIME, hr
STRUCTURAL ELEMENTS	297 (75)	498, 589, 700, 811 (400, 600, 800, 1000)	0, 100, 1000, 5000, 10 000
FULL-SCALE PANELS	297 (75) 589 (600)	589 (600) 589 (600)	0, 100, 10 000 0, 100

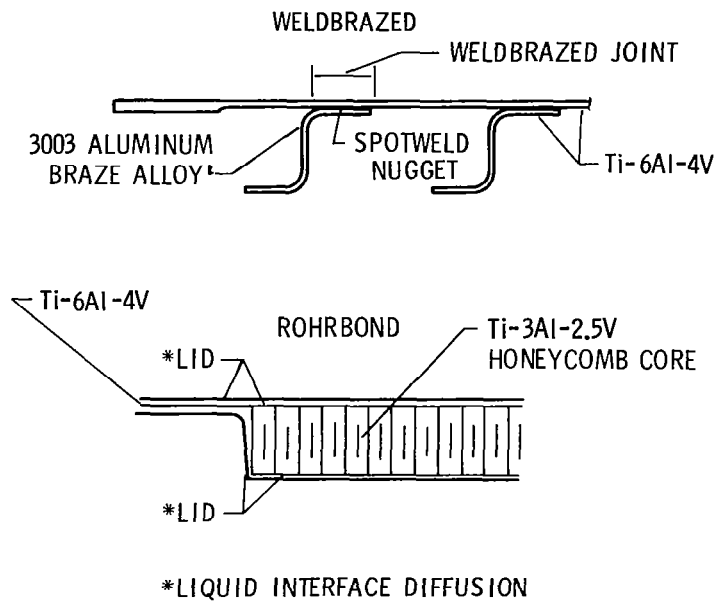CYCLIC EXPOSURE
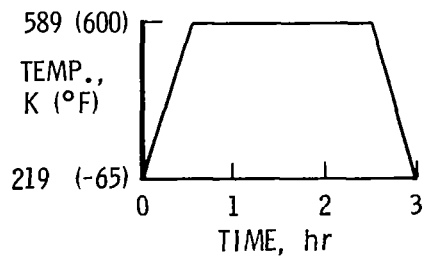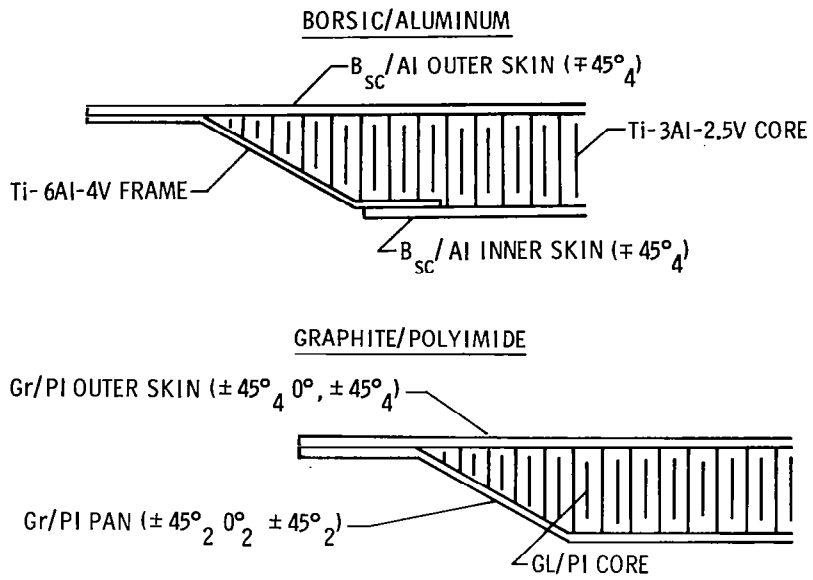

Figure 3.- Ground exposure and test schedule.

Figure 4.- Titanium panel concepts.

BORSIC/ALUMINUM

B_{sc}/Al OUTER SKIN ($\mp 45°_4$)

Ti-3Al-2.5V CORE

Ti-6Al-4V FRAME

B_{sc}/Al INNER SKIN ($\mp 45°_4$)

GRAPHITE/POLYIMIDE

Gr/PI OUTER SKIN ($\pm 45°_4\ 0°, \pm 45°_4$)

Gr/PI PAN ($\pm 45°_2\ 0°_2\ \pm 45°_2$)

GL/PI CORE

Figure 5.- Composite panel concepts.

PRESSURE SUPPLY

BLADDER

Ti CAUL SHEET

STEEL PLATEN

FIBERFRAX

STEEL PLATEN
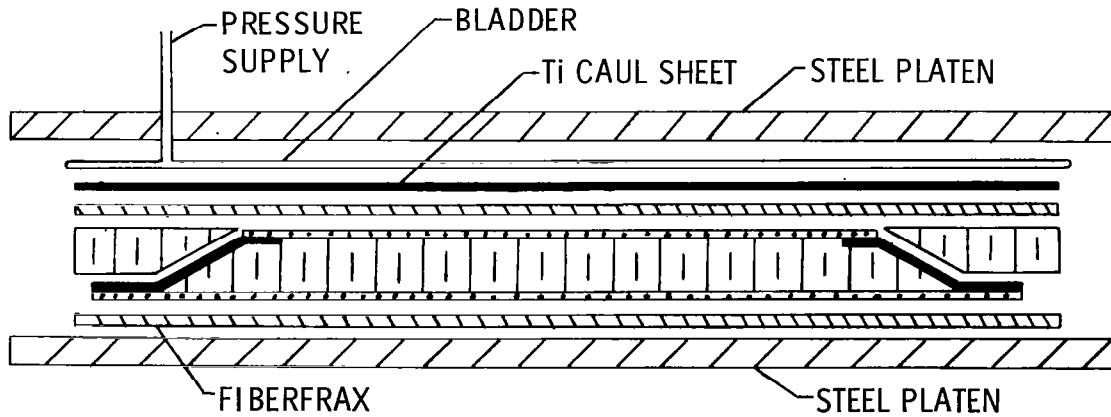
Figure 6.- Brazing fixture for B_{sc}/Al panels.
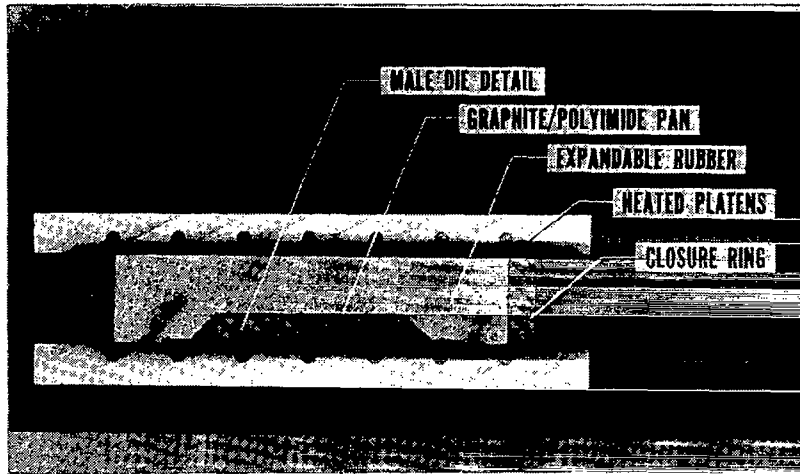
Figure 7.- Rubber compaction process for Gr/PI.
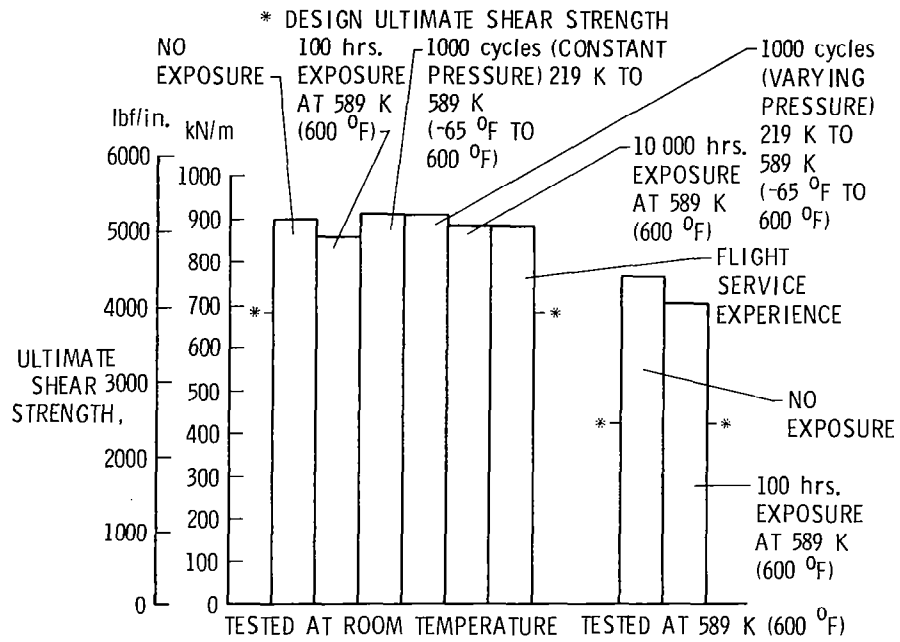
Figure 8.- Weldbrazed panel shear test results.
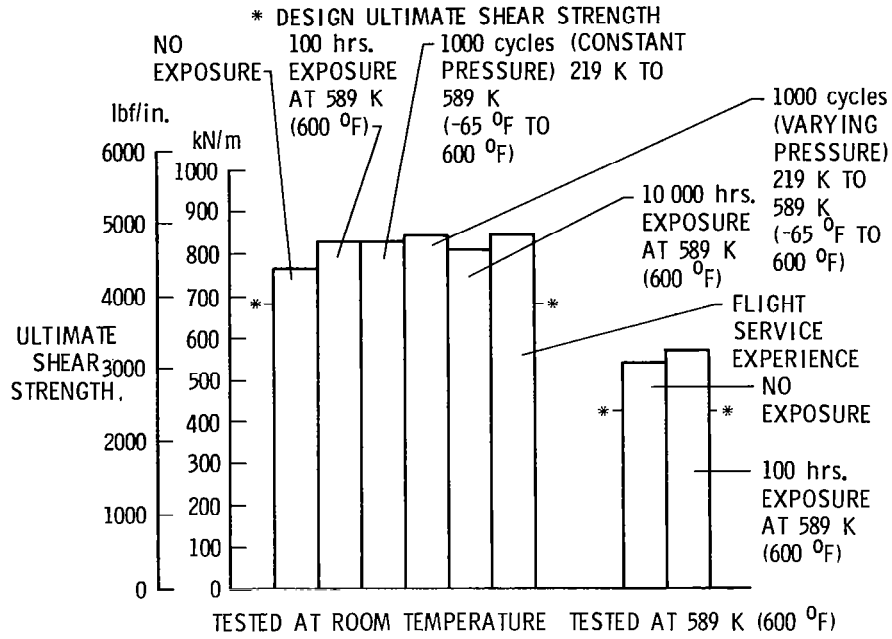
Figure 9.- RohrBond panel shear test results.
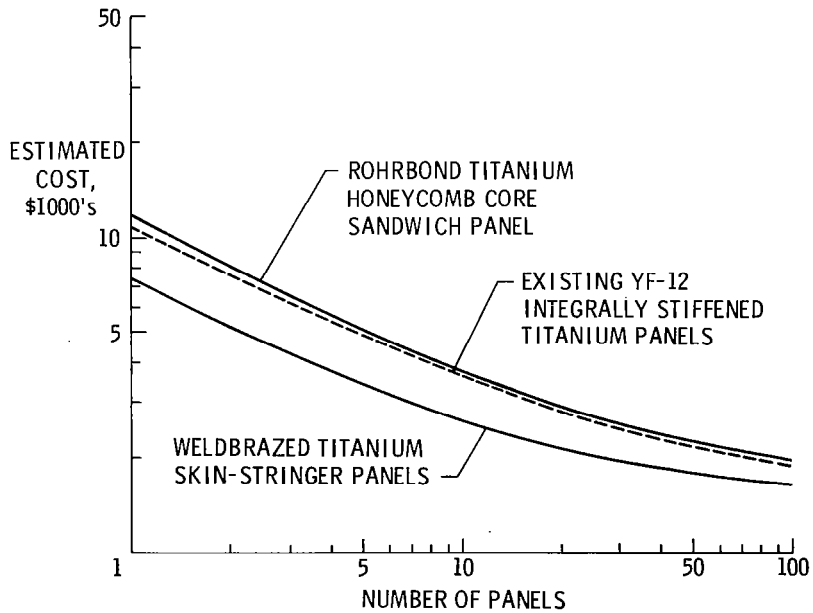
Figure 10.- Estimated costs of titanium panels.
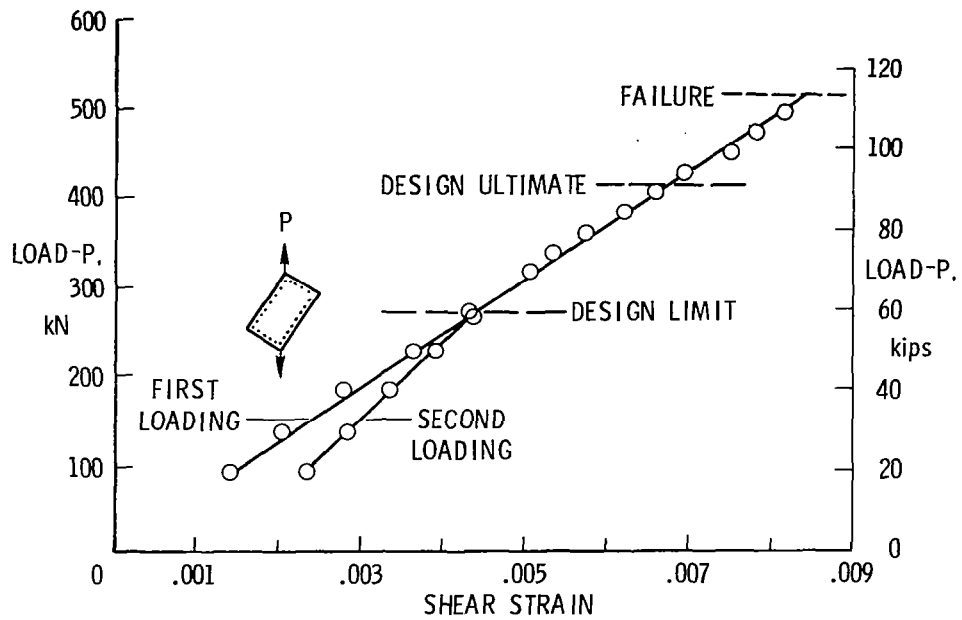
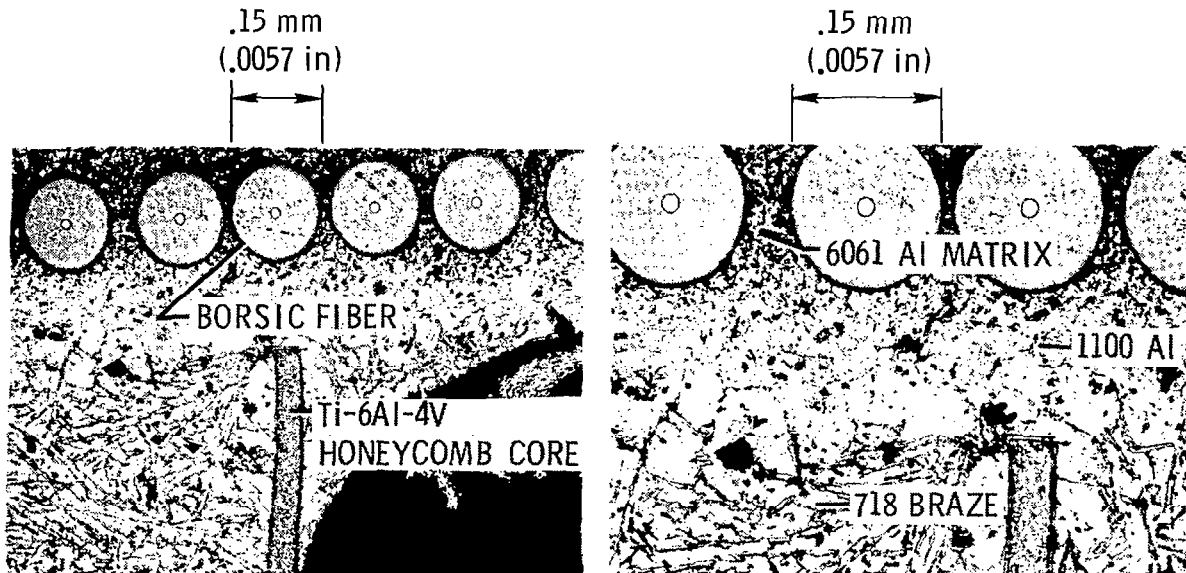Figure 11.- B_{sc}/Al panel shear test results.
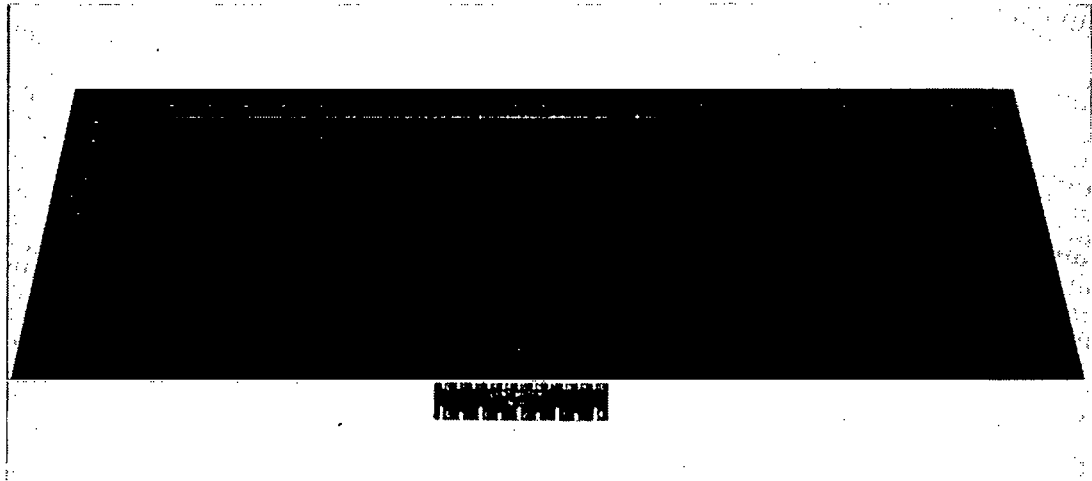

Figure 12.- B_{sc}/Al Ti honeycomb-core joint.

Figure 13.- Gr/PMR-15 PI process verification panel.