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SUMMARY

The Langley Research Center, in support of the NASA Supersonic Cruise Research (SCR) program, initiated a study to develop fabrication processes for the efficient utilization of metal-matrix composites. Initial studies indicated that brazing was a viable process for fabricating Borsic[1]/aluminum structures when the braze surface of the composite was clad with a diffusion barrier which served to minimize interaction between the braze alloy and the constituents of the composite. A recent brazing study on a hybrid composite material with pure titanium foil cladding on the surfaces of Borsic/aluminum sheet material showed that the titanium cladding eliminated composite/braze interaction.

This report covers the development and evaluation of a brazing process to fabricate full-scale titanium-clad Borsic/aluminum (Ti-clad Bsc/Al) skin-stringer panels. A total of six panels were fabricated for inclusion in the program which included laboratory testing of panels at ambient temperature and 533 K (500° F) and flight service evaluation of one panel on the NASA YF-12 airplane. The unique panel design concept developed consisted of a hybrid composite skin, stiffened with capped honeycomb-core stringers. Panel test results showed that the brazing process developed was satisfactory for fabricating and joining Ti-clad Bsc/Al primary structure and that flight service evaluation of the panel on the YF-12 airplane had no deleterious effect on panel properties.

INTRODUCTION

In recent years, the rapidly increasing cost and limited supply of fuel has accentuated the need for stronger, lightweight aircraft structures. Metal-matrix composites is one class of materials which exhibit high ratios of stiffness and strength to density. To incorporate these materials into advanced aircraft structures, a technology base needs to be established.

To aid in the establishment of this technology base, the Langley Research Center, in support of the NASA Supersonic Cruise Research (SCR) program, initiated a study to develop fabrication processes for the efficient utilization of metal-matrix composites in aircraft structures. Initial studies (refs. 1 and 2) reported that the properties of boron/aluminum (B/Al) were degraded after exposure to temperatures above 811 K (1000° F). In contrast, when the boron fiber was coated with a thin layer of silicon carbide to form Borsic, the properties of Borsic/aluminum (Bsc/Al) were unaffected after a 1-hour exposure at a temperture of 866 K (1100° F). Further brazing studies resulted in the successful development of a brazing process for fabricating a Bsc/Al—titanium-honeycomb-sandwich structure (ref. 3). The success of this brazing process was largely attributed to the use of an 1100 aluminum alloy foil as a diffusion barrier on the braze surface of the Bsc/Al, which minimized interac-

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[1]Borsic: registered trade name of United Aircraft Products, Inc.

tion of the braze alloy with the constituents of the composite. A recent brazing study on a hybrid material having pure titanium foil cladding on the surfaces of Bsc/Al sheet material showed that the titanium cladding eliminated composite/braze interaction (ref. 4). This hybrid material was designated titanium-clad Borsic/aluminum (Ti-clad Bsc/Al).

The objective of this study was to develop and evaluate a brazing process for full-scale fabrication of Ti-clad Bsc/Al skin-stringer panels. The panel selected for manufacturing development was a right upper wing panel of the Mach 3 NASA YF-12 airplane. Five panels were fabricated for room-temperature and 533 K (500° F) laboratory testing and one panel was fabricated for flight service evaluation on the YF-12. Reported herein are panel design methodology, development of a satisfactory brazing process for panel fabrication, full-scale panel test results, and a metallurgical evaluation of the brazed panel joints.

Identification of commercial products in this report is used to adequately describe the materials. The identification of these commercial products does not constitute official endorsement, expressed or implied, of such products or manufacturers by the National Aeronautics and Space Administration.

SYMBOLS

The units for the physical quantities defined in this paper are given both in the International System of Units (SI) and parenthetically in the U.S. Customary Units. Measurements and calculations were made in the U.S. Customary Units. Factors relating the two systems are given in reference 5 and those used in this investigation are presented in appendix A.

EI	effective bending stiffness in direction of stiffeners
G	shear modulus
N_x	normal stress resultant
N_{xy}	shear stress resultant
p	pressure loading on panel
t	panel skin thickness
δ	out-of-plane deflection
ε	strain

Subscripts:

x,y	panel coordinates

DESIGN

The original wing panel on the YF-12 airplane was an integrally stiffened titanium panel. (See fig. 1.) The Ti-clad Bsc/Al panel was designed to meet the ultimate load, stiffness, and maximum strain requirements (listed in table I) for satisfactory flight service replacement of the titanium panel. Ultimate loads at room temperature and 533 K (500° F) are given by requirements 1, 2, 3, 7, and 8; stiffnesses by requirements 4, 5, and 9; and maximum strains by requirements 6 and 10.

The unique Ti-clad Bsc/Al composite stiffened panel designed to meet these specifications is shown in figure 2. The panel consisted of a titanium-clad Bsc/Al skin (six plies of $\pm 45^{\circ}$ fiber orientation) stiffened by 12 stringers of titanium honeycomb core that were capped with titanium-clad Bsc/Al (three plies of 0° fiber orientation). The perimeter of the panel was reinforced with titanium doublers to provide adequate bearing strength and to facilitate attachment to the YF-12 substructure. Details of the skin and cap laminates are shown in figures 3 and 4, respectively.

Structural element testing, that verified the adequacy of the mechanical properties of the skin laminate, is discussed in appendix B. Picture-frame shear tests at room temperature and elevated temperature indicated that the skin laminate shown in figure 3 was capable of supporting the shear loads in requirements 1 and 7 of table I (requirement 1 was the critical ultimate load) and was capable of meeting the shear stiffness requirements. Prior test results for [$\pm 45^{\circ}$] Bsc/Al laminates (ref. 3) showed that the allowable shear stiffness was exceeded if a sufficient number of plies were used to meet the ultimate shear strength requirement. Similarly, tensile tests of the skin laminate shown in figure 3 indicated that the laminate could sustain the maximum strains imposed by requirements 6 and 10 of table I.

Buckling analyses of the panel configuration shown in figure 2 indicated that requirement 1 was the critical buckling design requirement and this requirement governed stiffener spacing. The out-of-plane deformation restriction given by requirement 4 determined the cap laminate and stiffener width. Stiffener depth was the maximum allowed by the YF-12 configuration.

Local buckling modes between stiffeners and general panel buckling modes were examined in the analysis. For local buckling, the stiffeners were assumed to have sufficient stiffness to restrict local buckling to the skin material between stiffeners. For general panel buckling, the bending stiffness was assumed to be uniform over the panel and the panel was assumed to behave as an orthotropic plate. Simple support boundary conditions on all edges were used in the analysis.

PANEL FABRICATION

Components

The components used to fabricate the Ti-clad Bsc/Al skin-stiffened YF-12 panel are shown in figure 5 and consisted of the hybrid Ti-clad Bsc/Al skin, 12 prefabricated, capped, honeycomb-core stringers, 4 Ti-6Al-4V titanium alloy doublers, and strips of 718 aluminum[2] braze alloy used to join the components together. The weight fraction of the constituent elements in the 718 aluminum braze alloy was as follows: 0.12 silicon, 0.003 copper, 0.008 iron, 0.002 zinc, 0.001 magnesium, and 0.001 manganese.

Fabrication Methods

Stringer fabrication involved several operations. The Ti-3Al-2.5V titanium alloy honeycomb core, which was purchased having a premachined height of 23 mm (0.9 in.) and a density of 152 kg/m^3 (9.5 lbm/ft^3), was electrical-discharge machined to a size of 356 by 711 mm (14 by 28 in.). The honeycomb core was then immersed in a nitric-hydrofluoric acid solution to chemically clean the surface for brazing. After cleaning, the honeycomb core was placed on a 0.508-mm (0.020-in.) thick sheet of 3004 aluminum braze alloy and both were positioned between brazing platens coated with a graphite stop-off material. The entire assembly was then placed in a vacuum furnace and heated to a temperature of 950 K (1250° F) at a pressure of 1.33 mPa (1×10^{-5} torr). The assembly was held at temperature for 10 min and as the aluminum alloy melted it was drawn up the nodes of the honeycomb core by capillary action. Prewetting the nodal areas of the honeycomb core in this manner reduced the volume of 718 aluminum braze alloy subsequently used to braze the Ti-clad Bsc/Al cap material to the honeycomb core.

Following visual inspection, the prewet core was vapor degreased. A sheet of the 0.63-mm (0.025-in.) thick 3-ply unidirectional Ti-clad Bsc/Al was sheared to the approximate dimension of the honeycomb core and chemically cleaned. A sheet of chemically cleaned 0.25-mm (0.010-in.) thick 718 aluminum braze alloy was then positioned between the prewet core and the composite sheet and the assembly was positioned between brazing platens. Vacuum brazing was accomplished at a temperature of 864 K (1095° F), a pressure of 1.33 mPa (1×10^{-5} torr), and a time at brazing temperature of 3 min. The individual stringers were cut from the brazed assembly using electrical-discharge machining.

Prior to assembling the panel components for brazing, the Ti-clad skin and titanium doublers were chemically cleaned and the stringers were vapor degreased. Panel assembly was initiated by positioning the 1.4-mm (0.057-in.) thick Ti-clad Bsc/Al skin on the lower brazing platen using the panel edge alinement tool shown in figure 6. The alinement tool consisted of ground and machined flat bar stock that was drilled and pinned to the lower platen so that

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[2]No. 718 aluminum braze alloy: product of Aluminum Company of America. (Designated as 4047 by the Aluminum Association and as BalSi-4 by AWS-ASTM.)

the edges of the bars defined the perimeter of the panel. The 718 aluminum braze alloy foil strips and Ti-6Al-4V alloy doublers were positioned on the skin and butted against the alinement tool. The doublers were held in position by spot-welding titanium foil tie straps to both the doublers and the skin material. By using the edge alinement bars as a reference, the stringers and associated braze alloy strips were located on the skin with the aid of vernier calipers. Position of the stringers was maintained by spot-welding titanium foil tie straps to the honeycomb core and the skin material. Positioning of the final stringer is depicted in figure 7.

The braze tooling used to maintain contact between faying surfaces during brazing consisted of a network of titanium-capped titanium honeycomb-core cross members, a stainless-steel pressure bladder, and lower and upper platens as shown in figure 8. The tooling cross members were fabricated in a manner similar to that used to fabricate the panel stiffeners. The cross members were fabricated by first brazing a 330- by 355- by 0.81-mm (13- by 14- by 0.032-in.) sheet of Ti-6Al-4V titanium alloy to a slab of 72 kg/m^3 (4.5 lbm/ft^3) titanium honeycomb core using 3004 aluminum braze alloy. Individual cross members approximately 25.4 mm (1 in.) wide were then cut from the capped honeycomb-core panel. The corners of the cross members were chamfered at an angle of approximately 45° to avoid contact with the braze alloy strips used to join the panel stringers to the skin. Individually capped honeycomb-core tooling edge members were brazed and sized to fit over the doublers around the perimeter of the panel. After brazing and sizing of the individual components, the tooling was assembled into a single unit by drilling and pinning the edge members to the cross members which were spaced to nest between the panel stringers against the composite skin. The tooling was sized so that when positioned on the assembled panel, tooling and panel stringer heights were equal and the caps of the tooling and the stringers provided a flat, semicontinuous plane parallel to the composite skin.

After the capped honeycomb-core tooling was positioned on the assembled panel components, a 0.51-mm (0.020-in.) thick Ti-6Al-4V titanium alloy slip sheet was placed on the caps of the tooling cross members and stiffeners. The upper brazing platen with the stainless-steel pressure bladder suspended from it was then lowered in place. The upper and lower platens were bolted together and the entire assembly was placed in a vacuum furnace. A photograph of the tooling in the vacuum furnace is shown in figure 9. The furnace was capable of two-rate, two-soak programmed resistance heating as well as closed recirculating inert-gas programmed cooling. The temperature of the panel during brazing was monitored with eight thermocouples positioned at several points on the skin.

The time-temperature profile for brazing of the panel is shown in figure 10. Following evacuation of the vacuum furnace to a pressure of 133 μPa (1×10^{-6} torr), the assembly was heated to a temperature of 839 K (1050° F). Thermal equilibrium was established by holding the temperature at 839 K for approximately 10 min prior to heating to the brazing temperature of 864 K (1095° F). Proper contact between mating parts was maintained during brazing by pressurizing the stainless-steel bladder with helium to 13.8 kPa (2 psi).

When the skin temperature reached 864 K, power to the heating elements was turned off and the inert-gas cooling system was activated. Circulating helium gas cooled the panel to 839 K in approximately 10 min. Gas cooling was then discontinued and the panel was furnace-cooled to ambient temperature.

Thermal exposure of the composite material during brazing was minimized by the combination of prewet honeycomb core, lightweight honeycomb-core tooling, inert-gas cooling, and 718 aluminum braze alloy. Fabrication was completed by trimming the panel to size with a diamond-impregnated wheel and drilling with conventional high-speed drills. Following the fabrication of the process verification panel, five additional panels were fabricated using the same procedure. The average mass of the finished Ti-clad Bsc/Al panels was 2.73 kg (6.02 lbm) which represents a mass savings of 30 percent compared to the original titanium panel.

Nondestructive Evaluation

Visual, radiographic, and ultrasonic C-scan inspections were performed on the wing panel skins and the completed wing panels. Suitable standards were developed for both radiographic and ultrasonic inspection methods. The preconsolidated hybrid composite skin material was radiographically inspected for broken filaments and filament orientation and ultrasonically C-scanned to inspect for delaminations. The uniformity of prewetting of the honeycomb core used in stringer fabrication and the braze between the honeycomb core and the Ti-clad Bsc/Al caps of the stringers were inspected visually. Uniform filleting of the braze between the skin and capped honeycomb-core stiffeners was verified by ultrasonic C-scan. All panels were inspected by the above methods and no significant defects were detected.

Tests

Following final quality assurance inspection, all panels were shipped to the Advanced Development Projects (ADP) Division of the Lockheed-California Company for testing. The design verification panel was tested at ambient temperature. Flight qualification testing consisted of testing two panels at ambient temperature and two at 533 K (500° F). For each temperature, one was tested as fabricated and one after 100 hours of exposure at 533 K to simulate elevated temperature exposure during flight. The panel exposed to flight service on the YF-12 was tested at ambient temperature.

All panels were tested in shear using a picture-frame fixture having pinned corners and a loading frame, designed and fabricated by Lockheed ADP, as shown in figure 11 (ref. 6). For testing at 533 K (500° F), the fixture was enclosed in a removable oven employing quartz-lamp heaters. Six separate controllable heating zones, three on each side of the panel, were used to provide a temperature uniformity of ±11 K (±20° F) during testing.

The panels were loaded by applying a tensile load to two diagonally opposite corners. The design verification panel, the as-fabricated flight qualification panel tested at ambient temperature, and the flight panel were instrumented with 22 strain gages to determine loading uniformity and the shear stress-strain characteristics of the panels. Load displacement data were obtained on all panels. For the instrumented panels, the effects of slack in the loading train were eliminated by zeroing the strain gages after the application of a load of 89 kN (20 kips). Loading was then continued in 44.5-kN (10-kip) increments to the design limit load of 267 kN (60 kips) and then the panel was unloaded. On reloading the panels to failure, load was applied in 44.5-kN increments from 89 to 311 kN (20 to 70 kips) and in 22.2-kN (5-kip) increments from 311 kN (70 kips) to failure. Strain-gage and load-displacement data were recorded after the load had stabilized following each load increment.

RESULTS AND DISCUSSION

Panel Shear Tests

The design verification panel was tested at ambient temperature in the as-fabricated condition. After brazing, the panel possessed a longitudinal bow having a radius of curvature of approximately 203 cm (80 in.). The panel bow was attributed to residual strains induced during cool down from the brazing temperature because of differences in the thermal expansion coefficients of the titanium doublers and the hybrid composite skin. Flattening of the panel was easily accomplished using finger pressure. The panel was therefore tested to evaluate compliance with load and stiffness requirements. The remaining panels for the program were flattened prior to testing by clamping the panels to a wooden fixture having a reverse radius of curvature of 165 cm (65 in.), cooling to 200 K (-100° F), and heating back to ambient temperature.

The load-strain data obtained from the shear test of the design verification panel are presented in figure 12. As mentioned previously, the panel was loaded to 88.9 kN (20 kips) to negate the effects of slack in the load train and the strain gages were zeroed. The panel was then loaded to 133 kN (30 kips) and the strain-gage outputs recorded. The slope of the load-strain plot was then determined and extrapolated back through zero to locate the 88.9-kN data point on the plot. The shear strains were calculated as the sums of the absolute readings of two gages located in the center of the panel and oriented at $\pm 45^{\circ}$ to the longitudinal axis. As shown in figure 12, the shear stiffness on first loading to design limit load was approximately 50 percent below that obtained on second loading. Extrapolation of the data back to zero load indicates that the panel plastically deformed on loading to design limit load and exhibited a permanent strain of approximately 1.2 percent. On second loading from design limit to a load of approximately 400 kN (90 kips), the load-strain curve was linear and the panel responded according to the lower initial shear stiffness. Failure occurred suddenly at a load of 523 kN (117.5 kips) which corresponds to a shear flow of 669.3 kN/m (3822 lb/in.) and equals 127 percent of the design ultimate shear strength.

The inelastic behavior of the panel noted on first loading and the increased stiffness on second loading was essentially the same as that noted previously from data obtained on testing of the Bsc/Al titanium honeycomb-core sandwich panels designed to meet the same requirements (ref. 3). As stated in the reference, "the variation in panel stiffness between the first and second loading can be attributed to the state of residual stress in the composite." Differences in the thermal expansion coefficients of the aluminum matrix and the Bsc fibers result in high residual tensile stresses and strains in the matrix on cooling from the elevated temperature processing. As a result, the application of a tensile load leads to yielding of the matrix at low applied stress levels and a very limited elastic contribution to the overall stiffness of the composite. Following yielding of the matrix in tension and unloading of the composite, the matrix unloaded elastically to reduce the magnitude of the residual stresses in the matrix. On second loading, the composite exhibited an increased stiffness due to the increased elastic contribution of the matrix until the strain equaled that experienced on initial loading.

The stiffness exhibited on second loading to design limit load was used in determining compliance with the shear stiffness requirement because the flight panel was proof loaded prior to installation on the airplane. Based on the data obtained, the design verification panel was determined to fully comply with the ambient temperature design requirements.

Flight qualification testing consisted of testing four panels to determine compliance with design requirements. Two panels were tested at ambient temperature and two at 533 K (500° F). One panel at each temperature was tested in the as-fabricated condition and one was tested after 100 hours of exposure at 533 K to simulate the effects of thermal exposure during supersonic cruise on the YF-12 airplane.

Load-strain data obtained on the as-fabricated panel tested at ambient temperature are shown in figure 13 along with the data for the design verification panel. Load-strain response for the two panels on first loading was essentially identical. Permanent strain exhibited by the flight qualification panels following first loading was approximately 15 percent less than that of the design verification panel and the stiffness on second loading above design limit was slightly higher. The lower amount of permanent strain exhibited by the flight qualification panel, compared to the design verification panel, may be attributed to the flattening procedure employed for the flight qualification panels which should have reduced the residual stress in the matrix. Failure occurred at a load of 500 kN (112.5 kips) which is 122 percent of design ultimate.

The ultimate shear strengths of the design verification panel and the four flight qualification panels are shown in figure 14 and listed in table II. All the panels tested exceeded the shear flow design requirements of 525.4 kN/m (3000 lbf/in.) and 332.7 kN/m (1900 lbf/in.) at room temperature and 533 K (500° F), respectively. The failure loads of the two flight qualification panels and the design verification panel tested at room temperature were 122, 147, and 127 percent of design ultimate and the failure loads of the two flight qualification panels tested at 533 K were 183 and 186 percent of design ultimate.

The Ti-clad Bsc/Al flight panel, following its final operational flight, was removed from the airplane and returned to Langley Research Center for nondestructive evaluation. As a result of nine flights on the YF-12 the panel accumulated a total flight time of 15 hr 43 min. Of this total, 6 hr 3 min were at speeds in excess of Mach 2.6, with 3 hr 27 min at Mach 3.0. Ultrasonic C-scan and radiographic inspection revealed no evidence of changes in the panel as a result of flight exposure. After nondestructive evaluation, the panel was returned to Lockheed ADP for testing. The panel was then instrumented and tested to failure in the same manner as previous panels. The panel failed in shear at an applied load of 524.9 kN (118 kips) or a shear flow of 672.1 kN/m (3838 lbf/in.), which is 128 percent of the room-temperature design ultimate shear strength and indicates no detrimental effects of flight exposure. The panel failed in shear by diagonal tension of the skin as shown in the photographs of figure 15. This failure was typical of all the panels tested.

Metallurgical Investigation

After the panels were tested, they were returned to Langley for examination. The panels were sectioned in areas away from the fracture to examine the filleting of the braze between skin and capped honeycomb-core stiffeners. A typical cross section is shown in figure 16. The photomicrographs show good filleting between the honeycomb core and Ti-clad Bsc/Al, which is indicative of good wetting and flow of the braze alloy. The photomicrographs also indicate the effectiveness of the titanium alloy as a diffusion barrier. The silicon particles in the braze alloy stop abruptly at the interface with the titanium, which indicates no reaction between the braze alloy and the constituents of the composite.

CONCLUDING REMARKS

A hybrid composite, titanium-clad Borsic/aluminum was successfully used in the fabrication and testing of a unique skin-stringer panel for the NASA Supersonic Cruise Research (SCR) program. The following remarks are based on the fabrication development and structural evaluation of this panel concept.

A unique skin-stringer panel design employing capped honeycomb-core stiffeners was established.

A successful brazing process for fabricating and joining Ti-clad Bsc/Al was developed.

The brazing process was successfully used to fabricate six flight-quality wing panels for the Mach 3 YF-12 airplane.

The Ti-clad Bsc/Al skin-stringer panels exhibited a weight savings of 30 percent compared to the original titanium panel.

At panel failure, the average shear flow was 127 percent of design ultimate at room temperature and 186 percent at 533 K (500° F).

The titanium cladding of the Bsc/Al served as an effective diffusion barrier between the braze and the composite.

Flight service evaluation of the panel on the YF-12 airplane for a total flight time of approximately 16 hr with approximately 3.5 hr at Mach 3 had no deleterious effect on panel strength.

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APPENDIX A

CONVERSION OF U.S. CUSTOMARY UNITS TO SI UNITS

Conversion factors (from ref. 5) required for units used herein are given in the following table:

Physical quantity	U.S. Customary Unit	Conversion factor (a)	SI Unit (b)
Density	lb/in^3	2.768×10^4	kilogram/meter^3 (kg/m^3)
Force	kip = 1000 lbf	4.448×10^3	newton (N)
Force/length	lbf/in.	1.751×10^2	newton/meter (N/m)
Length	in.	2.54×10^{-2}	meter (m)
Mass	lbm	4.536×10^{-1}	kilogram (kg)
Stress	kip/in^2 (ksi) lbf/in^2 (psi)	6.895×10^6 6.895×10^3	pascal (Pa) pascal (Pa)
Temperature	^oF	$(^oF + 459.67)/1.8$	kelvin (K)
Pressure	torr	1.333×10^2	pascal (Pa)

[a]Multiply value given in U.S. Customary Unit by conversion factor to obtain equivalent value of SI Unit.

[b]Prefixes to indicate multiples of SI Units are as follows:

Prefix	Multiple	Symbol
micro	10^{-6}	μ
milli	10-3	M
kilo	10^3	k
mega	10^6	M
giga	10^9	G

APPENDIX B

STRUCTURAL ELEMENT TESTS

Specimen Fabrication

The main objective of the structural element testing was to determine the strength of the Ti-clad Bsc/Al material and to establish design allowables for the YF-12 wing-panel program. The configurations of the test specimens for determining the material strength are shown in figure 17. The tensile specimens (fig. 17(a)) were sheared to size with the Borsic fibers alined at $\pm 45^{\circ}$ with the specimen length for the skin material and 0° or 90° for the cap material. Specimens were sheared in both the longitudinal and transverse direction of the 432- by 737-mm (17- by 29-in.) preconsolidated sheets. The specimens were tested at room temperature using a 44-kN (10-kip) screw-driven testing machine at a crosshead speed of 1.27 mm/min (0.05 in/min). Failure loads were recorded from the dial of the test machine.

The bolt-bearing specimens shown in figure 17(b) were required to establish the bolt-bearing strength of the panel attachment area. The specimen was fabricated by brazing a sheet of 0.84-mm (0.003-in.) thick Ti-6Al-4V to a sheet of the Ti-clad Bsc/Al skin material. The specimens were then electrical-discharge machined from the fabricated sheet. Test specimens were machined from the sheet with the center line of the test specimen parallel to the sheet (0° orientation) or at an angle (27° orientation) that represents the direction of bolt hole bearing in the ground test of the skin-stringer panel. The following table gives dimensions and orientation of the bolt-bearing test specimens shown in figure 17(b):

BOLT-BEARING SPECIMENS

Edge distance, e, mm (in.)	Hole diam., mm (in.)	Orientation, deg
19.0 ± 0.25 (0.750 ± 0.010)	$6.3 \begin{smallmatrix} +0.1 \\ -0.0 \end{smallmatrix}$ $\left(0.250 \begin{smallmatrix} +0.005 \\ -0.000 \end{smallmatrix}\right)$	0, 27
12.7 ± 0.25 (0.500 ± 0.010)	$6.3 \begin{smallmatrix} +0.1 \\ -0.0 \end{smallmatrix}$ $\left(0.250 \begin{smallmatrix} +0.005 \\ -0.000 \end{smallmatrix}\right)$	0, 27

APPENDIX B

The bolt holes in the specimens were drilled with conventional high-speed drills and standard practice. The specimens were tested at room temperature using a 534-kN (120-kip) hydraulic testing machine at a load rate of 8.9 kN/min (2000 lbf/min). Failure loads were recorded from the dial of the test machine.

To establish design shear allowables for the skin material, Ti-clad Bsc/Al specimens were tested as depicted in figure 17(c). Two Ti-clad Bsc/Al face sheets were electrical-discharge machined to size, exposed to a simulated braze cycle, and then bonded to an aluminum honeycomb core. The specimens were strain-gaged, placed in a test fixture, and loaded in shear with the loads applied as shown by the arrows in the figure. The specimens were loaded at a rate of 44.5 kN/min (10 000 lbf/min). Specimen loads and strain were recorded every 5 sec on a central data recording system.

Results of Structural Element Tests

The tensile test results for the Ti-clad Bsc/Al skin and cap material are given in table III. The longitudinal and transverse tensile strengths of the cross plied skin material varied from 491 to 545 MPa (71.2 to 79.1 ksi). Some of the specimens were subjected to a simulated braze cycle, which had no apparent effect on tensile strength. The cap material had unidirectional strengths ranging from 1.16 to 1.27 GPa (167.7 to 184.9 ksi) and transverse strengths ranging from 257 to 276 MPa (37.4 to 40.1 ksi).

The test results for the bolt-bearing specimens are given in table IV. All specimens failed by bearing and pull out of material around the bolt hole. The load range for the specimens with 12.7 mm (0.5 in.) edge distance was 19.8 to 20.9 kN 4460 to 4700 lbf) which, at a minimum, was 45 percent more load-carrying ability than the 13.3 kN (3000 lbf) design requirement.

The test results of the picture-frame shear specimens are shown in figure 18 in which shear stress is plotted against shear strain. The tests were terminated at a load of 169 kN (38 000 lbf) to prevent damage to the test fixture. The shear strength developed was 460 and 482 MPa (66.7 and 69.9 ksi), respectively, for the two tests, which gave an average stress 27 percent higher than was required to meet the requirements of the test program.

These structural element tests verified the adequacy of the skin-stringer design to fulfill the design requirements of the YF-12 panels.

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TABLE I.- DESIGN REQUIREMENTS FOR YF-12 WING PANEL

Requirement	Temperature	Design condition
1	RT	$N_{xy} = 525$ kN/m (3000 lbf/in.) $N_x = 0$ $p = 0$
2	RT	$N_{xy} = 350$ kN/m (2000 lbf/in.) $N_x = -175$ kN/m (-1000 lbf/in.) $p = 0$
3	RT	$N_{xy} = 350$ kN/m (2000 lbf/in.) $N_x = 87$ kN/m (500 lbf/in.) $p = 0$
4	RT	$N_{xy} = 121$ kN/m (690 lbf/in.) $N_x = 83$ kN/m (472 lbf/in.) $p = 44.1$ kPa (6.4 psi) $\delta \leqq 0.914$ mm ($\leqq 0.036$ in.)
5	RT	$(EI)_x = 17.2$ kN-m (0.152×10^6 lbf-in.) $(Gt)_{xy} \leqq 91.9$ MN/m ($\leqq 0.525 \times 10^6$ lbf/in.)
6	RT	$\varepsilon_x = \pm 0.00085$ $\varepsilon_y = \pm 0.0067$
7	533 K (500° F)	$N_{xy} = 333$ kN/m (1900 lbf/in.) $N_x = 0$ $p = 0$
8	533 K (500° F)	$N_{xy} = 71$ kN/m (403 lbf/in.) $N_x = \pm 21$ kN/m (± 120 lbf/in.) $p = 5.2$ kPa (0.75 psi)
9	533 K (500° F)	$(EI)_x = 14.2$ kN-m (0.126×10^6 lbf-in.) $(Gt)_{xy} \leqq 77$ MN/m ($\leqq 0.44 \times 10^6$ lbf/in.)
10	533 K (500° F)	$\varepsilon_x = 0.000116$ $\varepsilon_y = 0.00410$

TABLE II.- TEST RESULTS OF Ti-CLAD BORSIC/ALUMINUM

SKIN-STIFFENED PANELS

Panel	Test temperature	Exposure prior to test	Failure load		Shear flow	
			kN	lbf	kN/m	lbf/in.
DVP	RT	None	522.7	117 500	669.3	3822
1	RT	None	500.4	112 500	640.8	3659
2	RT	100 hr at 533 K (500° F)	605.0	136 000	774.6	4423
3	533 K (500° F)	None	478.2	107 500	612.2	3496
4	533 K (500° F)	100 hr at 533 K (500° F)	467.1	105 000	598.1	3415
Flight	RT	Flight exposure on YF-12 airplane	524.9	118 000	672.1	3838

TABLE III.- TENSILE TEST RESULTS OF Ti-CLAD BORSIC/ALUMINUM MATERIAL

(a) Skin material

Specimen	Orientation to panel axis	Exposure prior to test	Area		Load		Stress	
			mm^2	in^2	kN	lbf	MPa	ksi
1	Trans.	Simulated braze cycle	36.8	0.0571	20.1	4515	545	79.1
2	↓	↓	36.8	.0570	19.3	4345	525	76.2
3	↓	↓	37.1	.0575	19.4	4365	523	75.9
4	↓	↓	37.0	.0574	18.6	4190	503	73.0
5	Long.	↓	36.5	.0565	18.9	4250	518	75.2
6	↓	↓	36.2	.0561	17.8	3995	491	71.2
7	↓	↓	36.5	.0565	18.6	4190	511	74.1
8	↓	↓	36.9	.0572	18.9	4240	511	74.1
9	Trans.	None	37.0	.0574	19.8	4450	534	77.5
10	↓	↓	37.1	.0575	19.1	4305	516	74.9
11	↓	↓	37.4	.0580	19.8	4460	530	76.9
12	↓	↓	37.3	.0578	19.5	4385	523	75.9
13	Long.	↓	37.4	.0579	18.8	4235	504	73.1
14	↓	↓	37.1	.0575	19.3	4350	521	75.6
15	↓	↓	37.5	.0582	18.9	4250	503	73.0

TABLE III.- Concluded

(b) Cap material

Specimen	Orientation	Exposure prior to test	Area		Load		Stress	
			mm^2	in^2	kN	lbf	GPa	ksi
1	Long.	None	16.5	0.0255	20.2	4550	1.23	178.2
2	↓	↓	16.1	.0249	19.2	4325	1.20	173.6
3			16.5	.0256	19.1	4300	1.16	167.7
4			16.1	.0249	19.3	4350	1.20	174.6
5			15.9	.0247	20.3	4575	1.27	184.9
6	↓		15.9	.0247	19.7	4420	1.23	178.9
7	Trans.		12.4	.0192	3.4	762	.27	39.8
8	↓		13.0	.0202	3.6	810	.28	40.1
9			12.5	.0193	3.3	740	.26	38.4
10			13.9	.0215	3.6	805	.26	37.4
11	↓	↓	14.2	.0220	3.8	850	.27	38.7

TABLE IV.- BOLT-BEARING TEST RESULTS OF

Ti-CLAD BORSIC/ALUMINUM MATERIAL

Specimen	Alinement, deg	Edge distance		Load	
		mm	in.	kN	lbf
1	0	12.7	0.5	20.9	4700
2				20.7	4650
3				20.8	4670
4				20.6	4620
5		19.1	.75	27.0	6070
6				26.9	6050
7				27.4	6160
8				26.6	5970
9	27	12.7	.5	19.9	4470
10				19.8	4460
11				20.5	4600
12				20.2	4535
13		19.1	.75	27.6	6200
14				27.1	6085
15				27.3	6130
16				27.7	6225

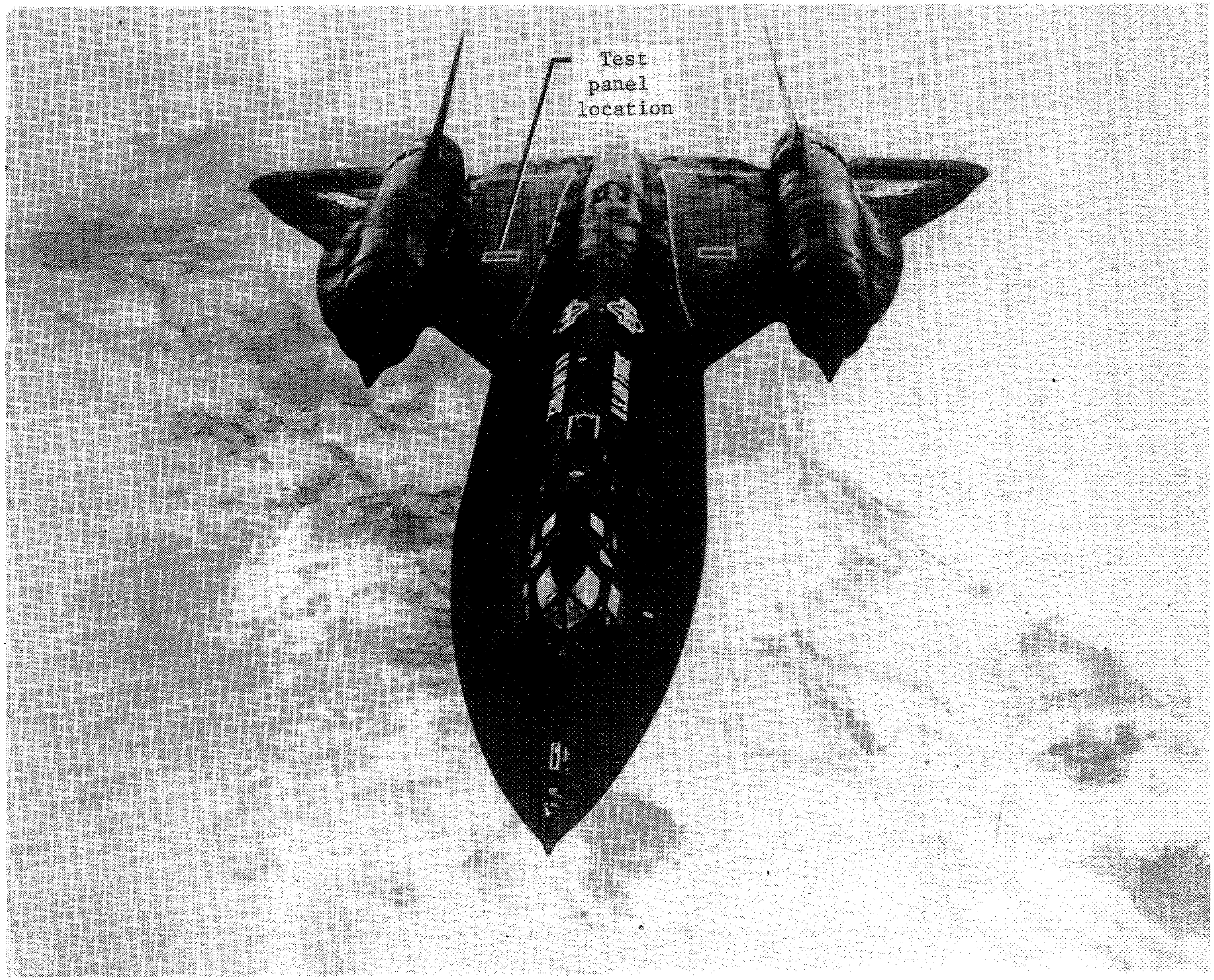

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Figure 1.- Test panel location on YF-12 airplane.

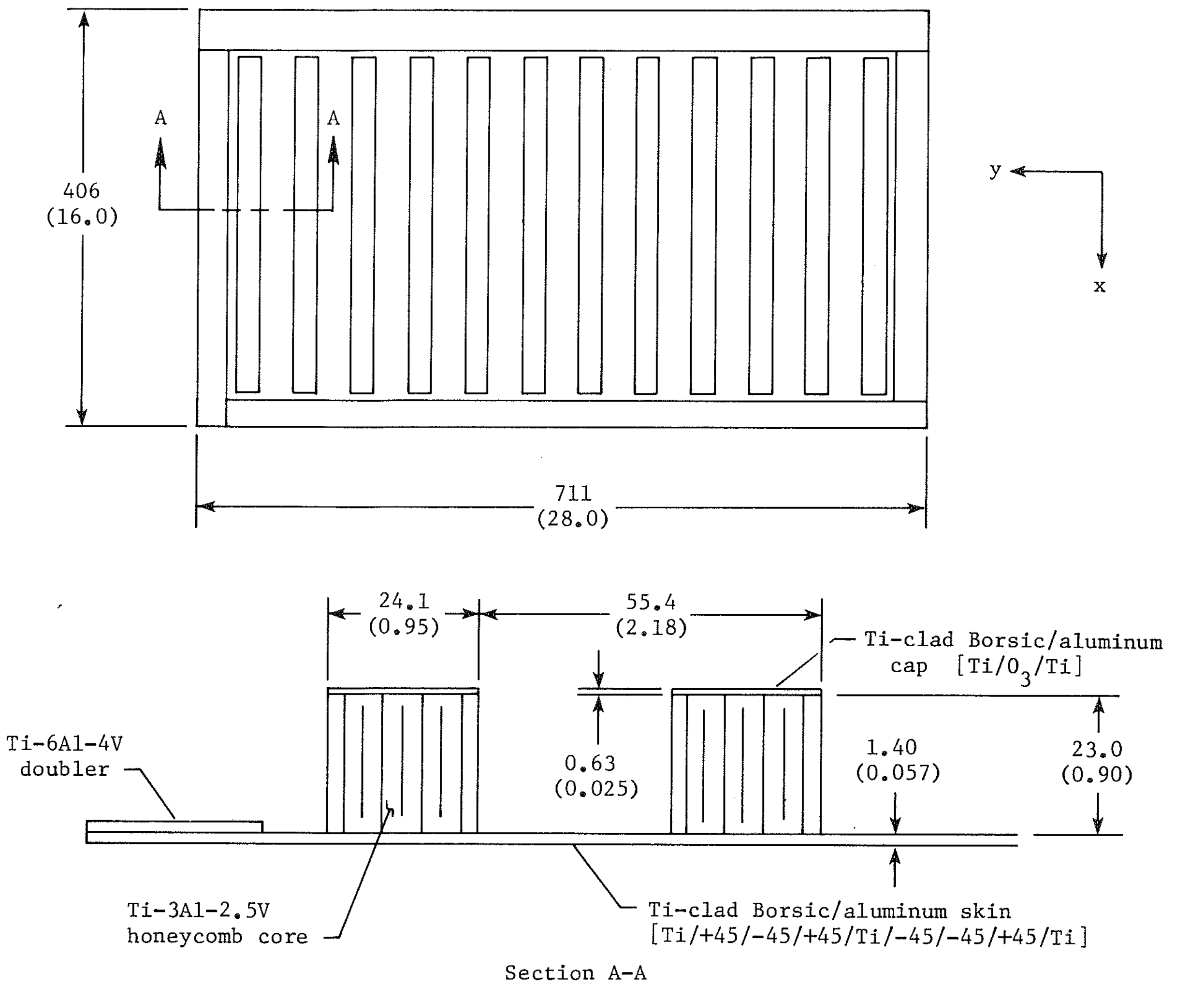

Figure 2.- Titanium-clad Borsic/aluminum skin-stringer panel design. Dimensions are given in millimeters (inches).

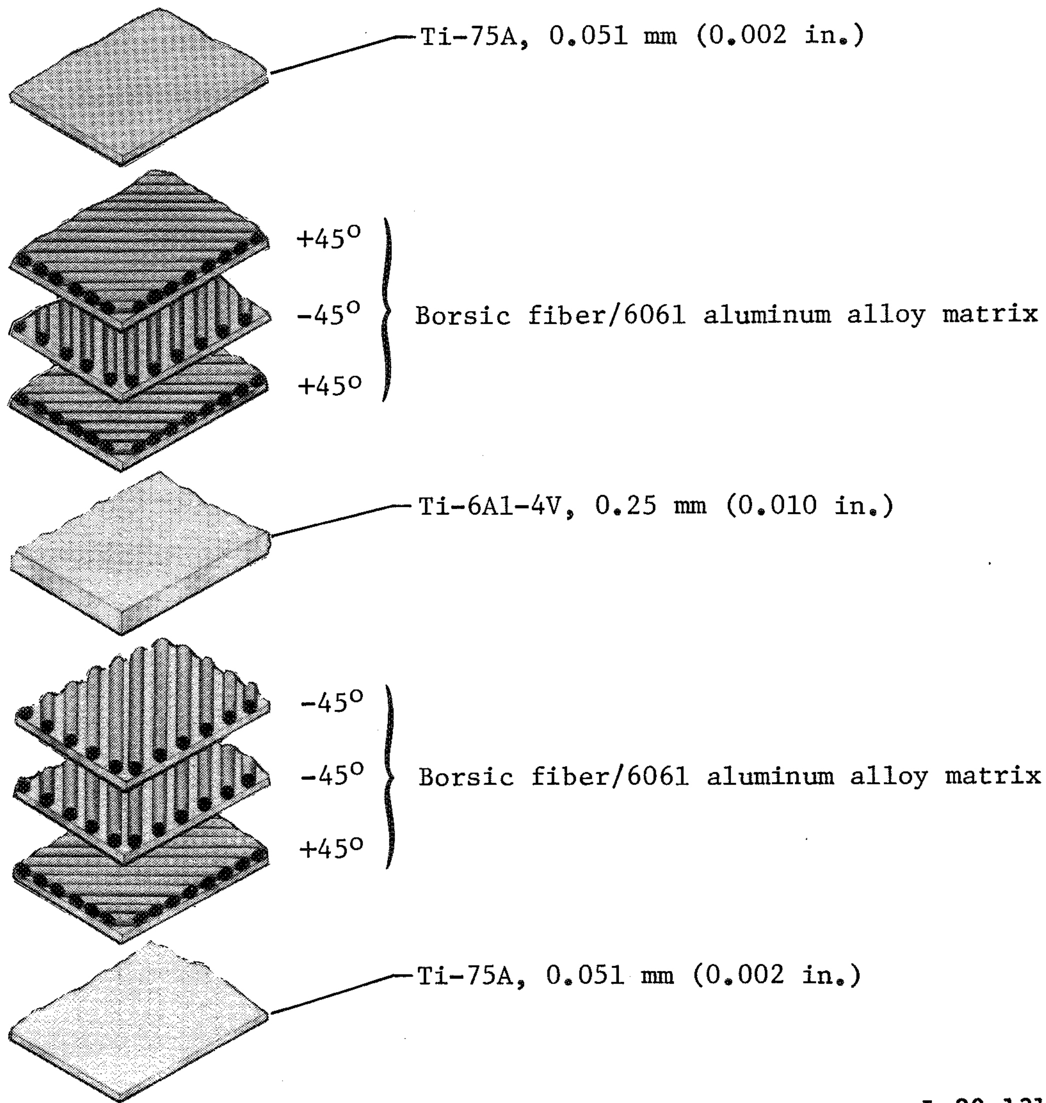

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Figure 3.- Fiber orientation in titanium-clad Borsic/aluminum skin material.

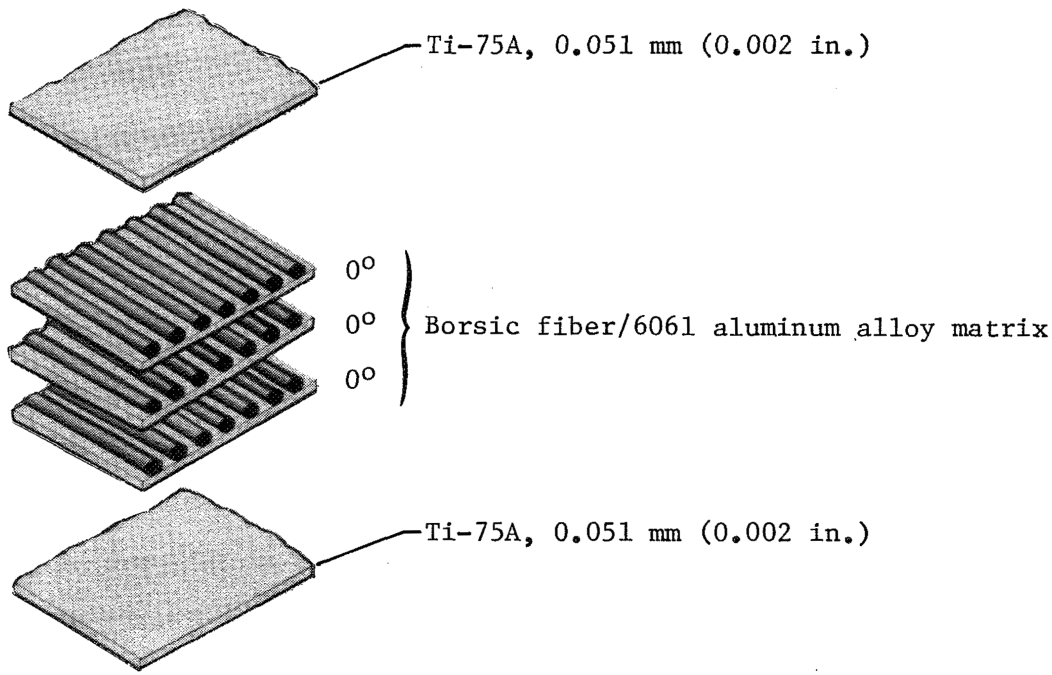

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Figure 4.- Fiber orientation in titanium-clad Borsic/aluminum cap material.

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Figure 5.- YF-12 panel components.

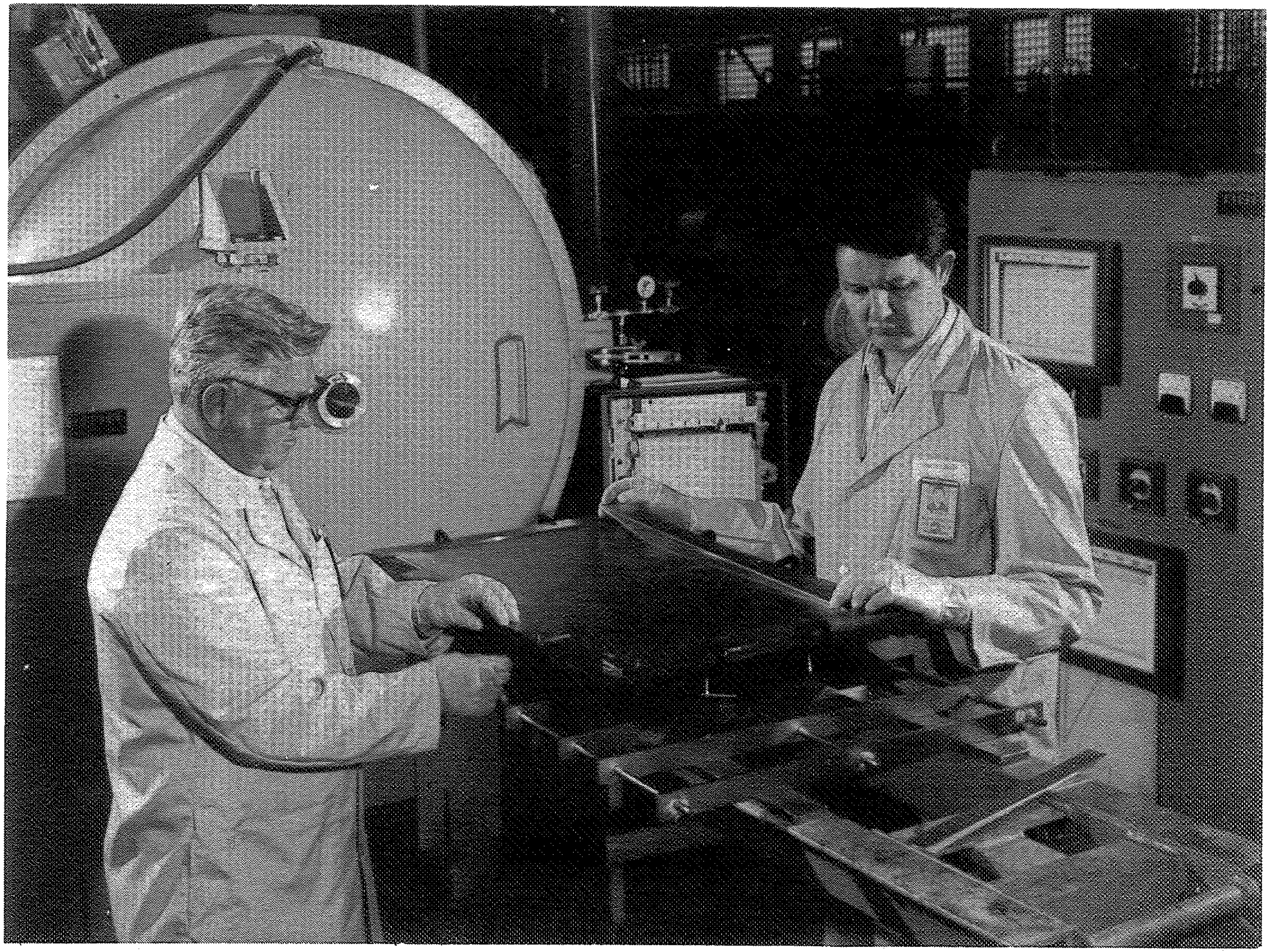

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Figure 6.- Panel alinement tool.

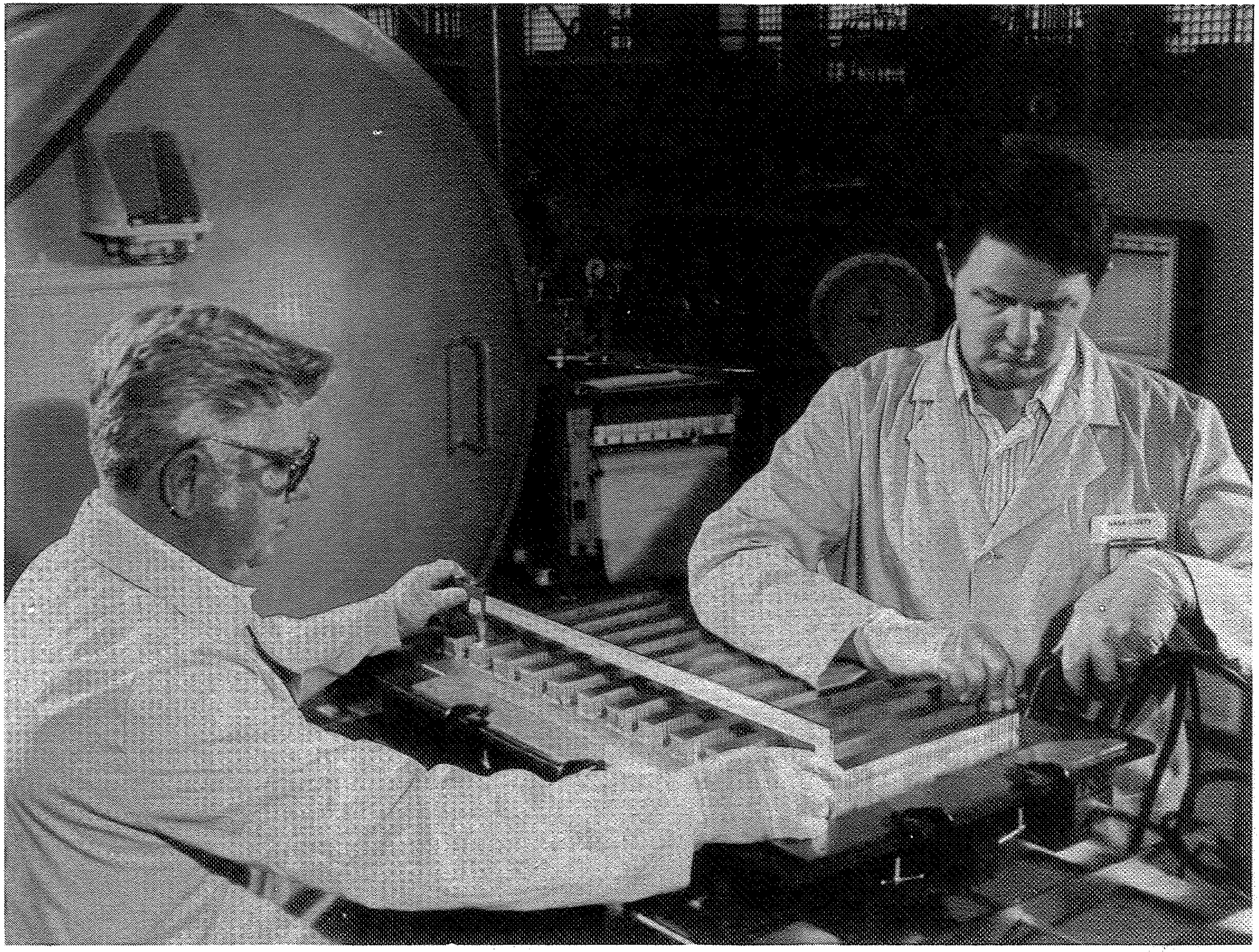

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Figure 7.- Assembly of stringers on skin.

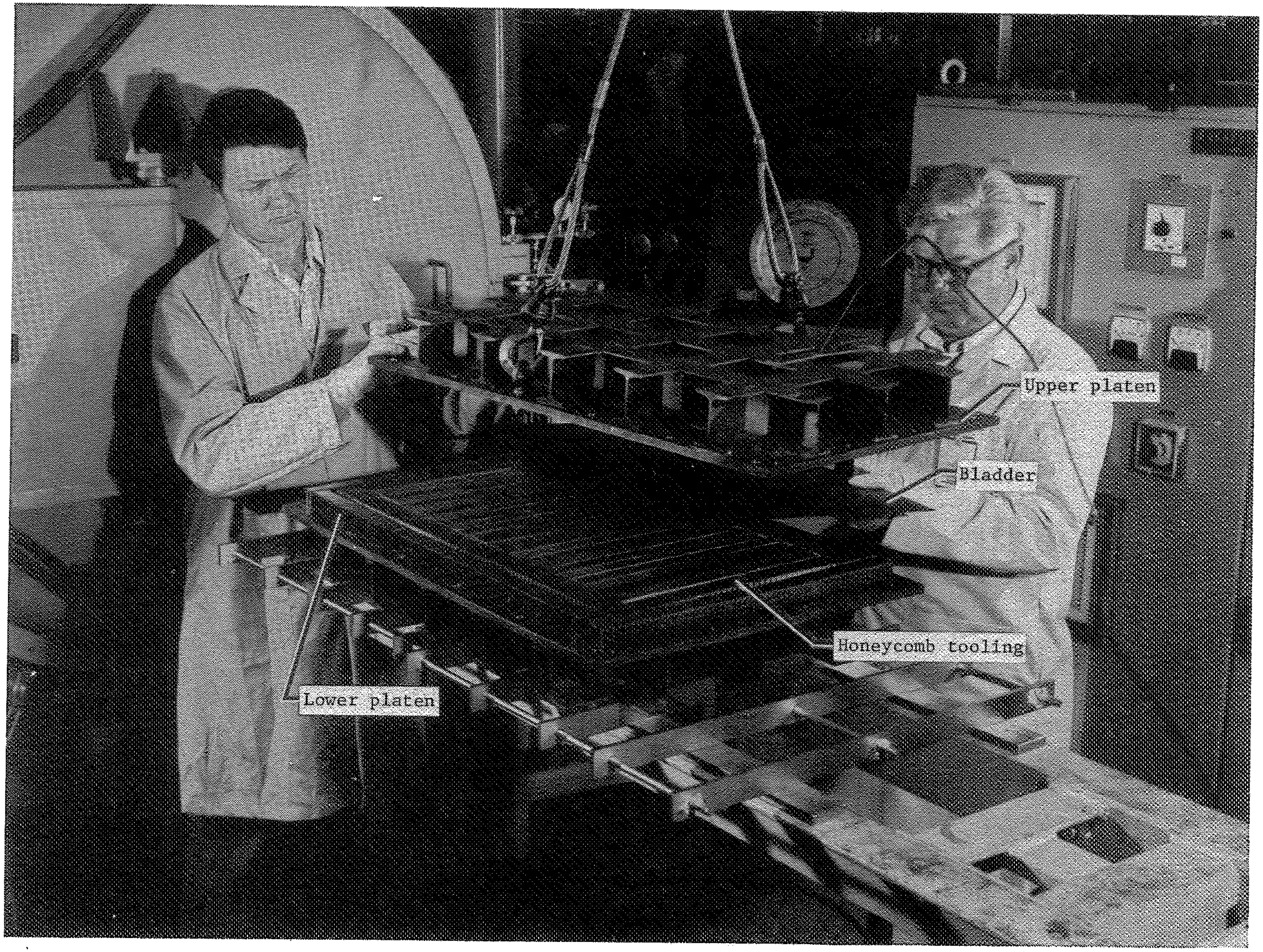

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Figure 8.- Titanium-clad Borsic/aluminum panel assembly and tooling for brazing.

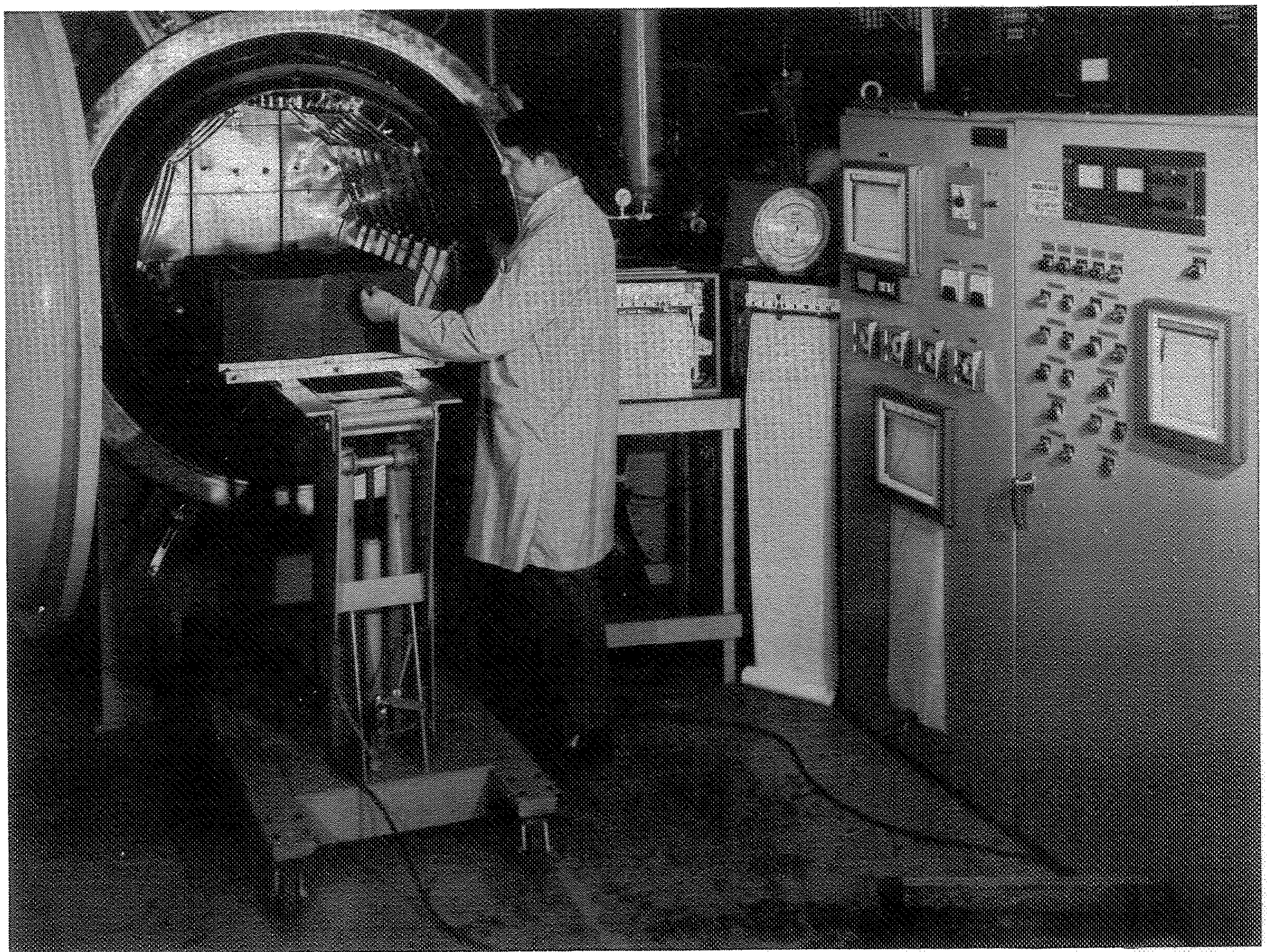

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Figure 9.- Braze tooling in vacuum furnace.

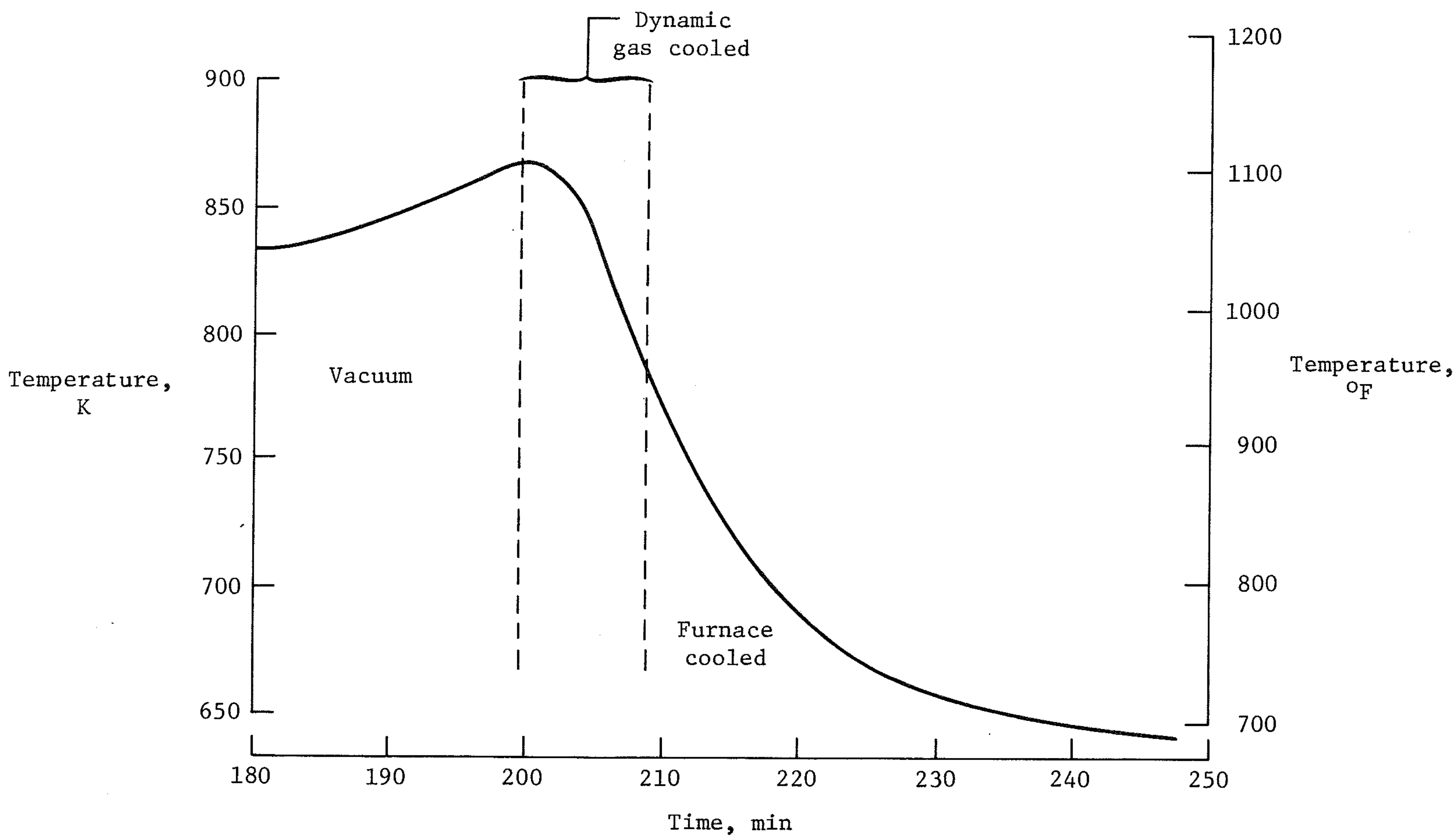

Figure 10.- Time and temperature brazing profile.

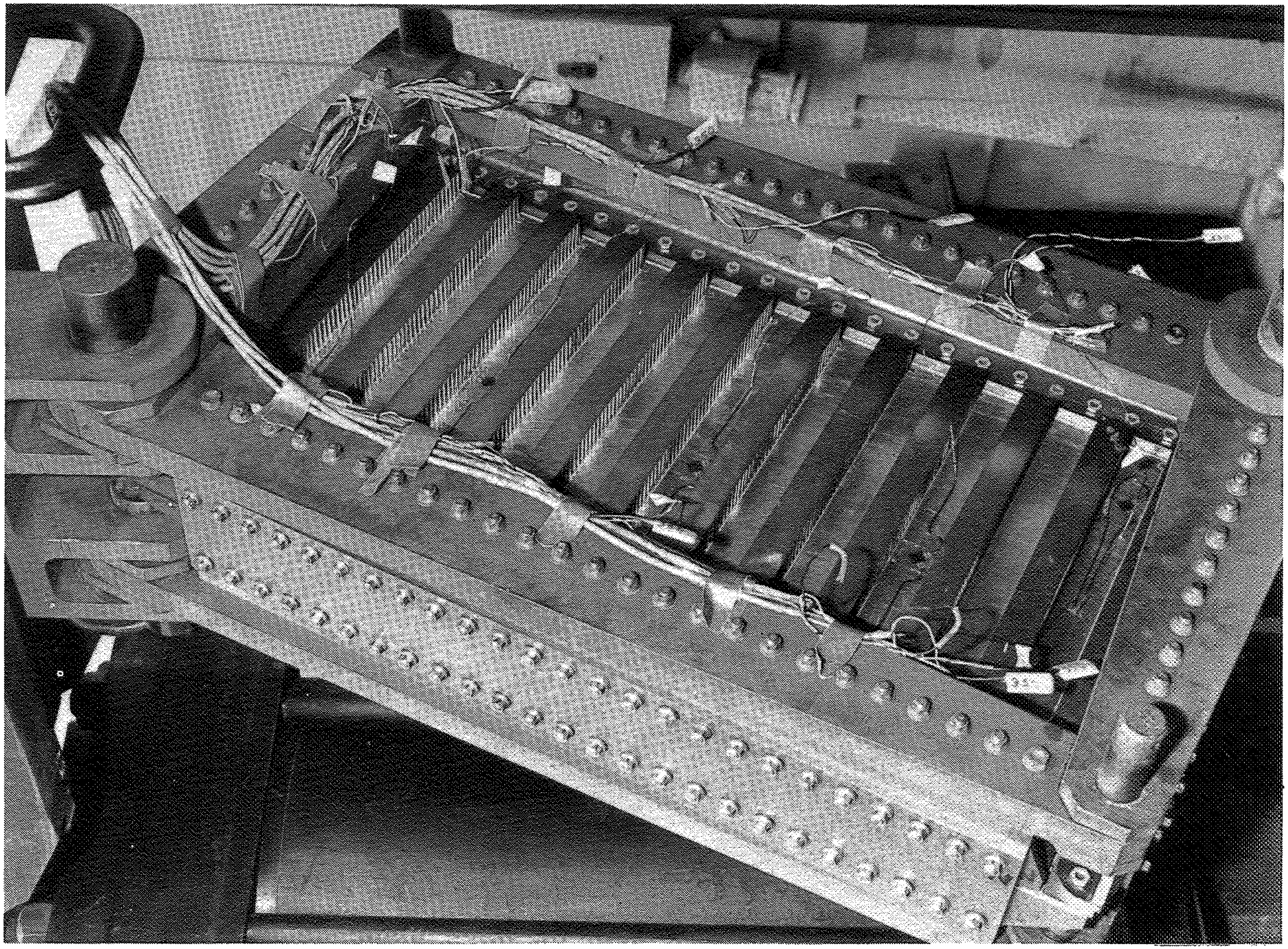

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Figure 11.- Panel in shear fixture for testing.

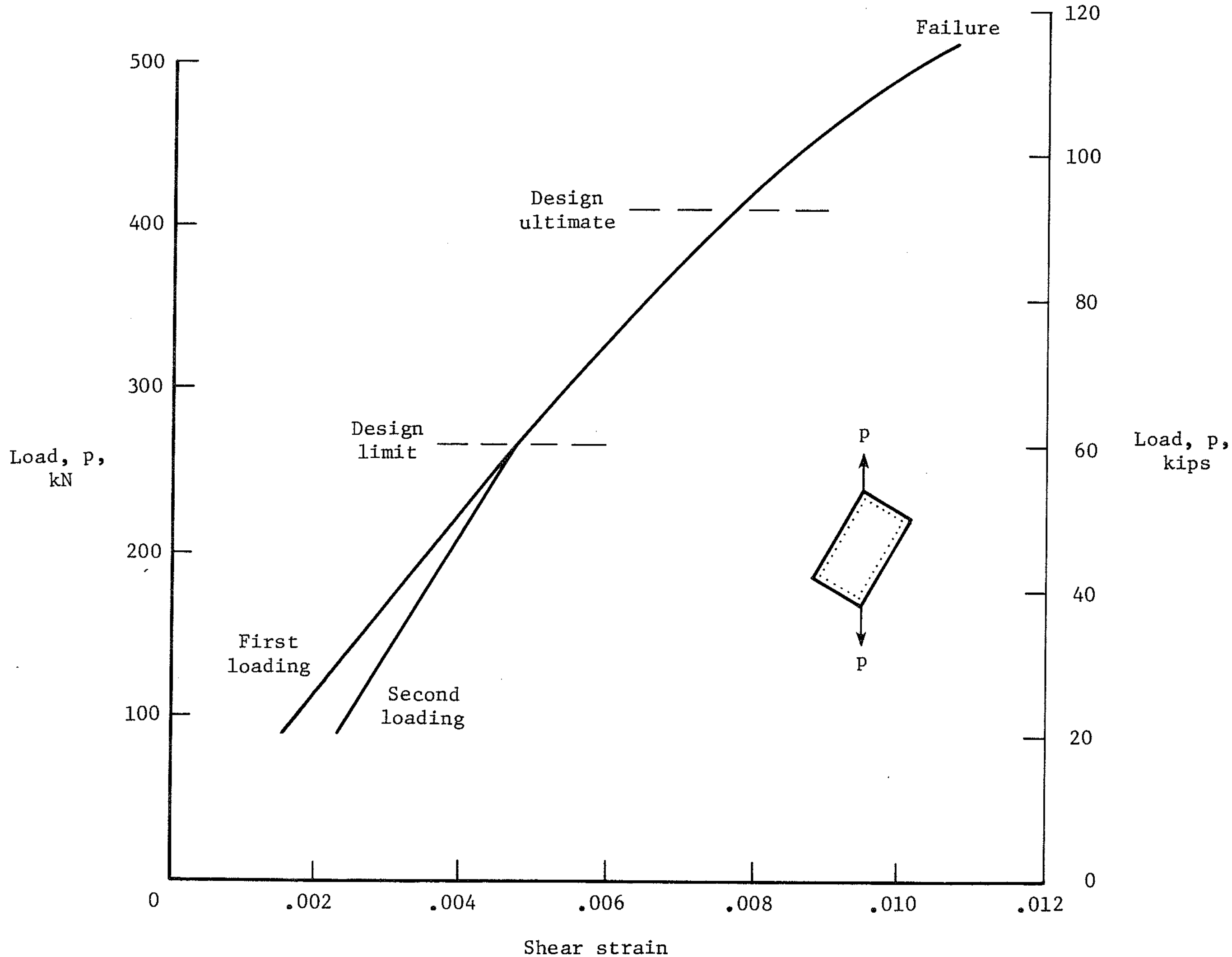


Figure 12.- Shear test data for design verification panel.

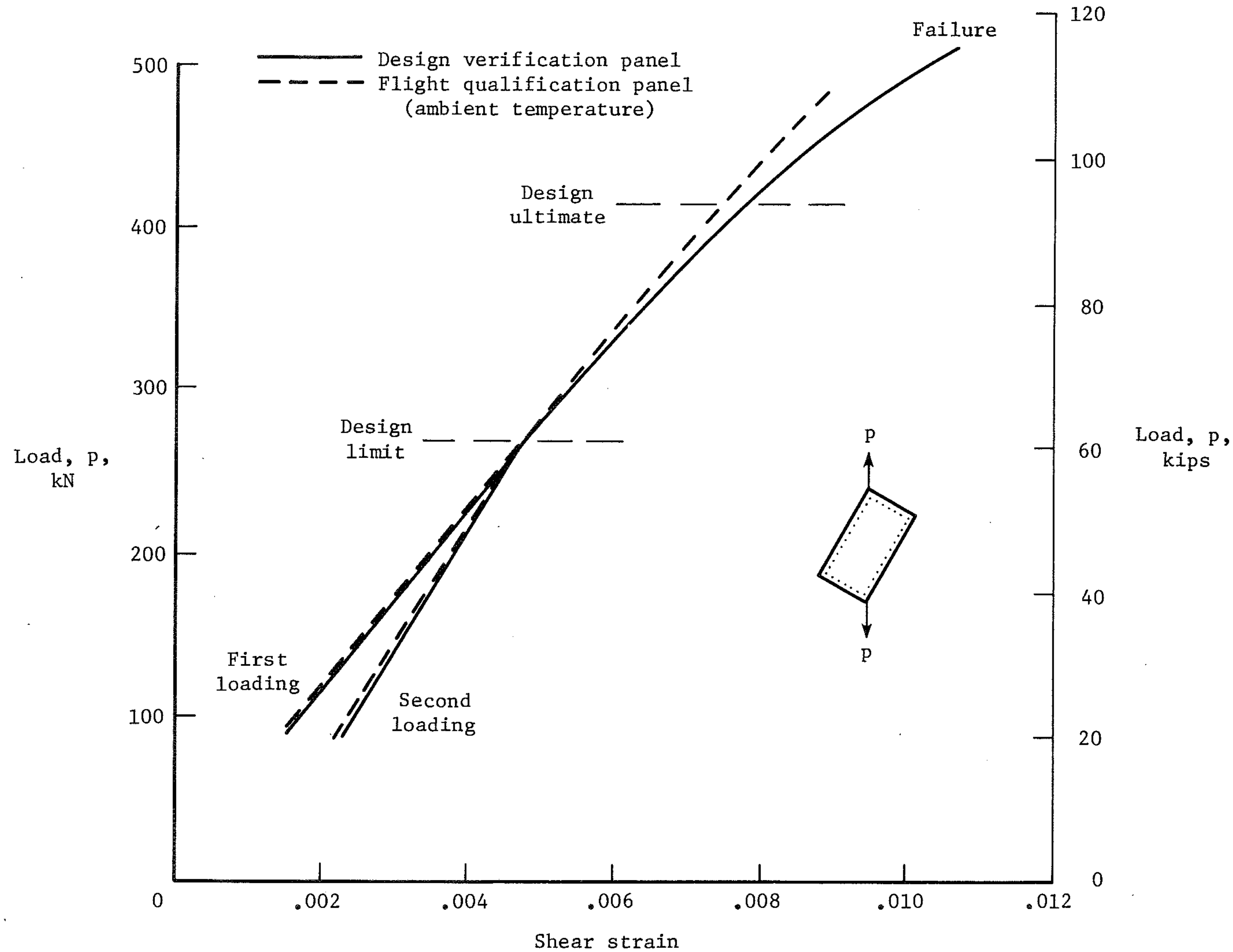


Figure 13.- Shear test data for design verification panel and flight qualification panel.

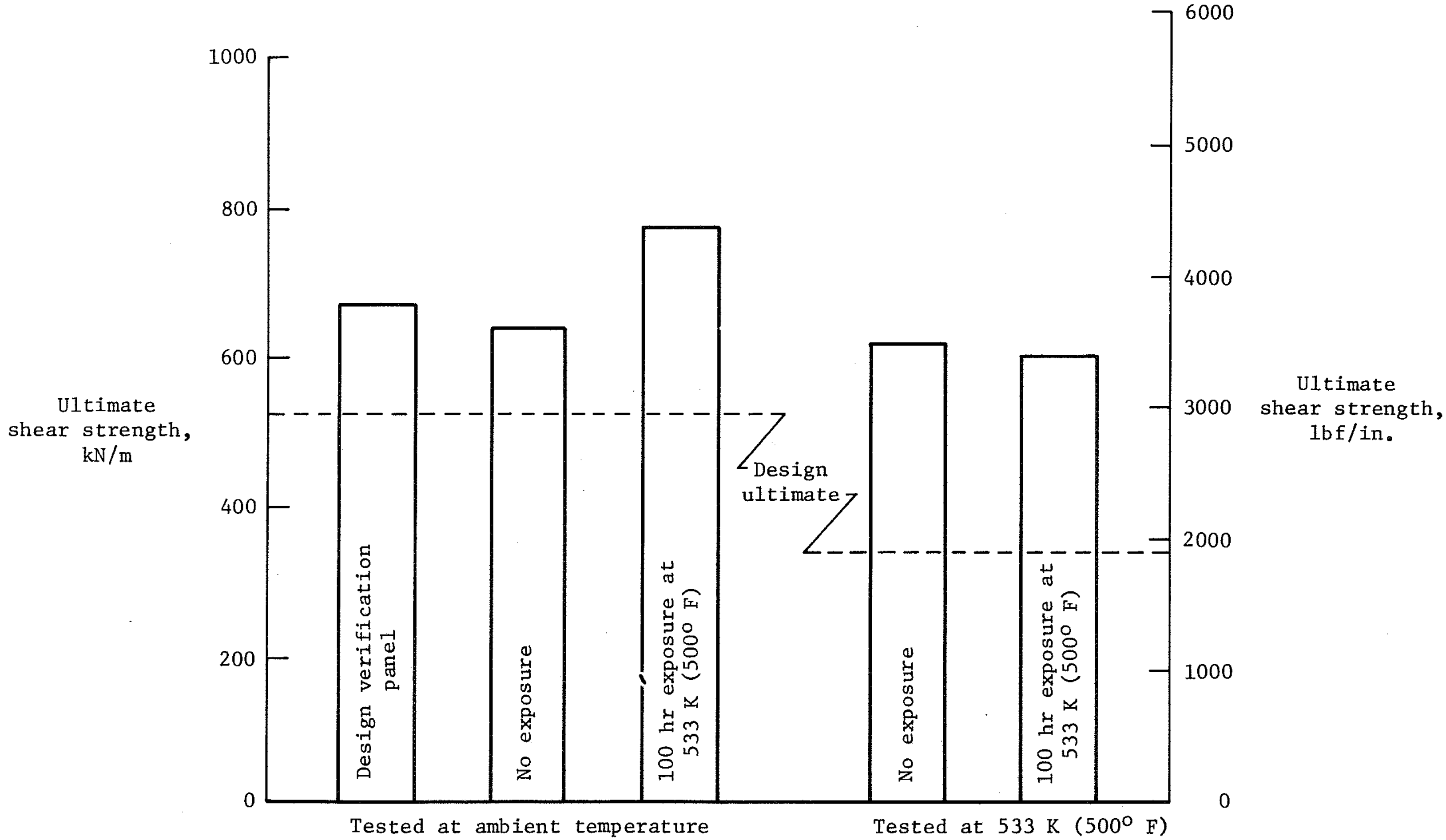
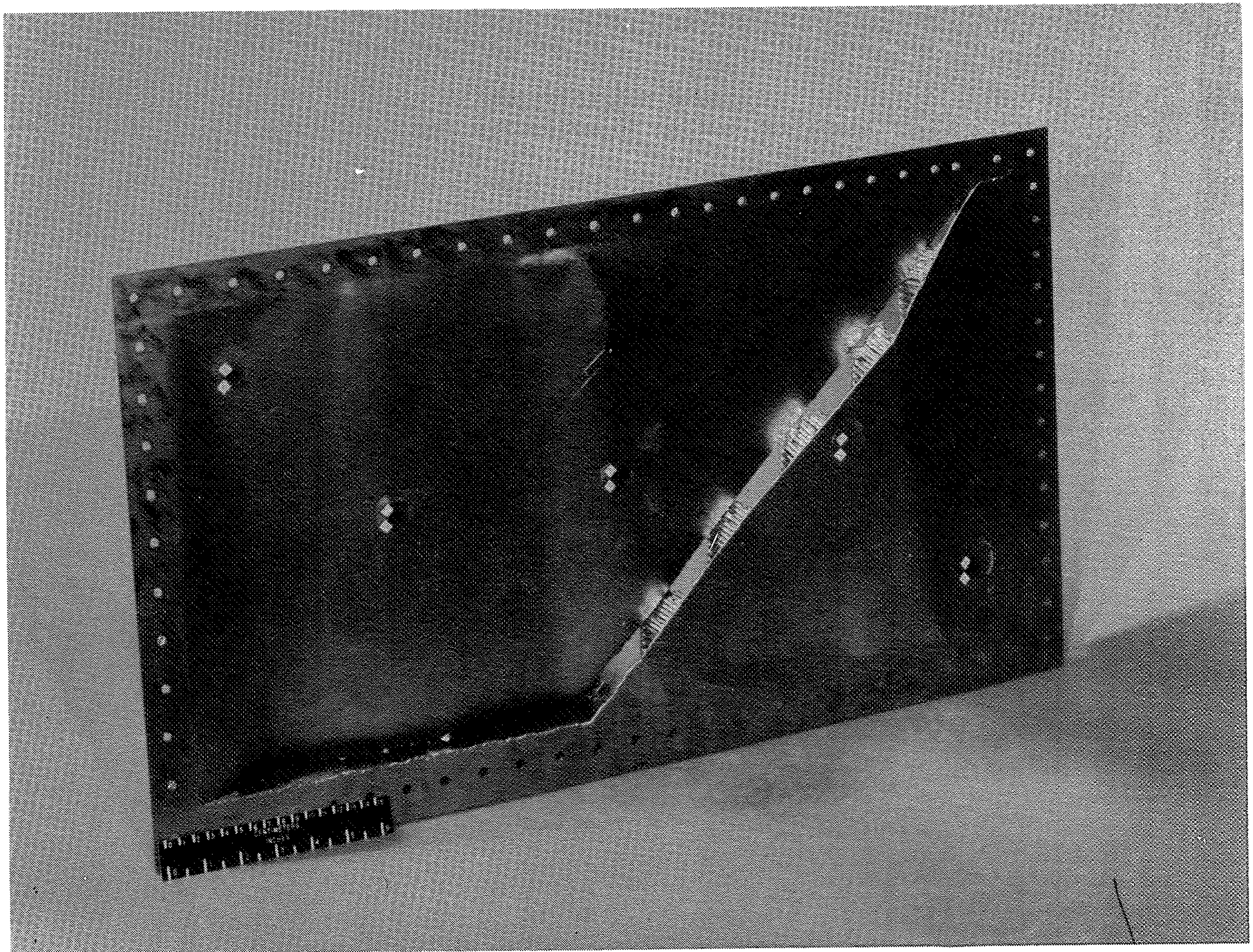


Figure 14.- Ultimate shear strengths of design verification panel and flight qualification panels.



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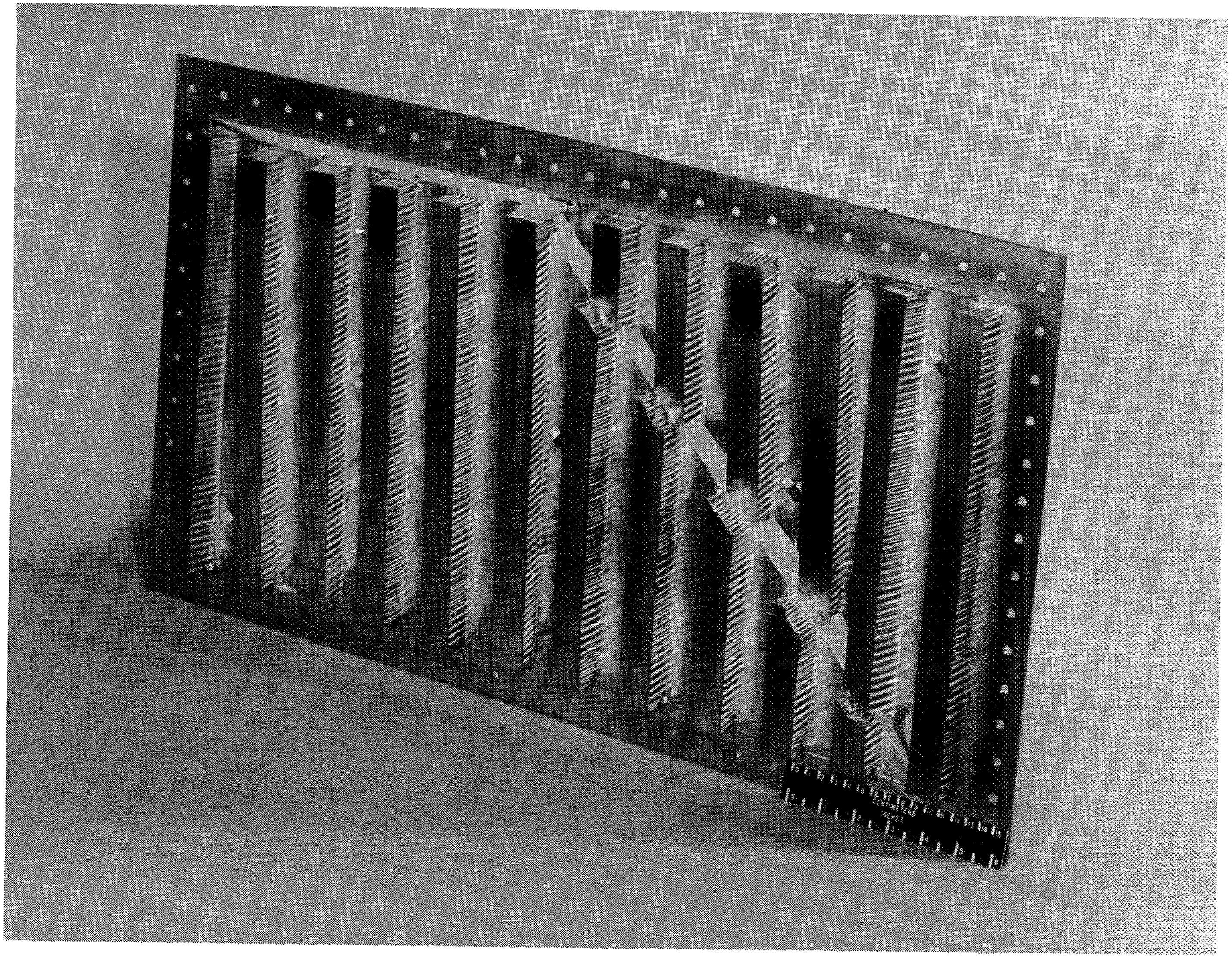
(a) Skin side.

Figure 15.- Flight service evaluation panel after test to failure.



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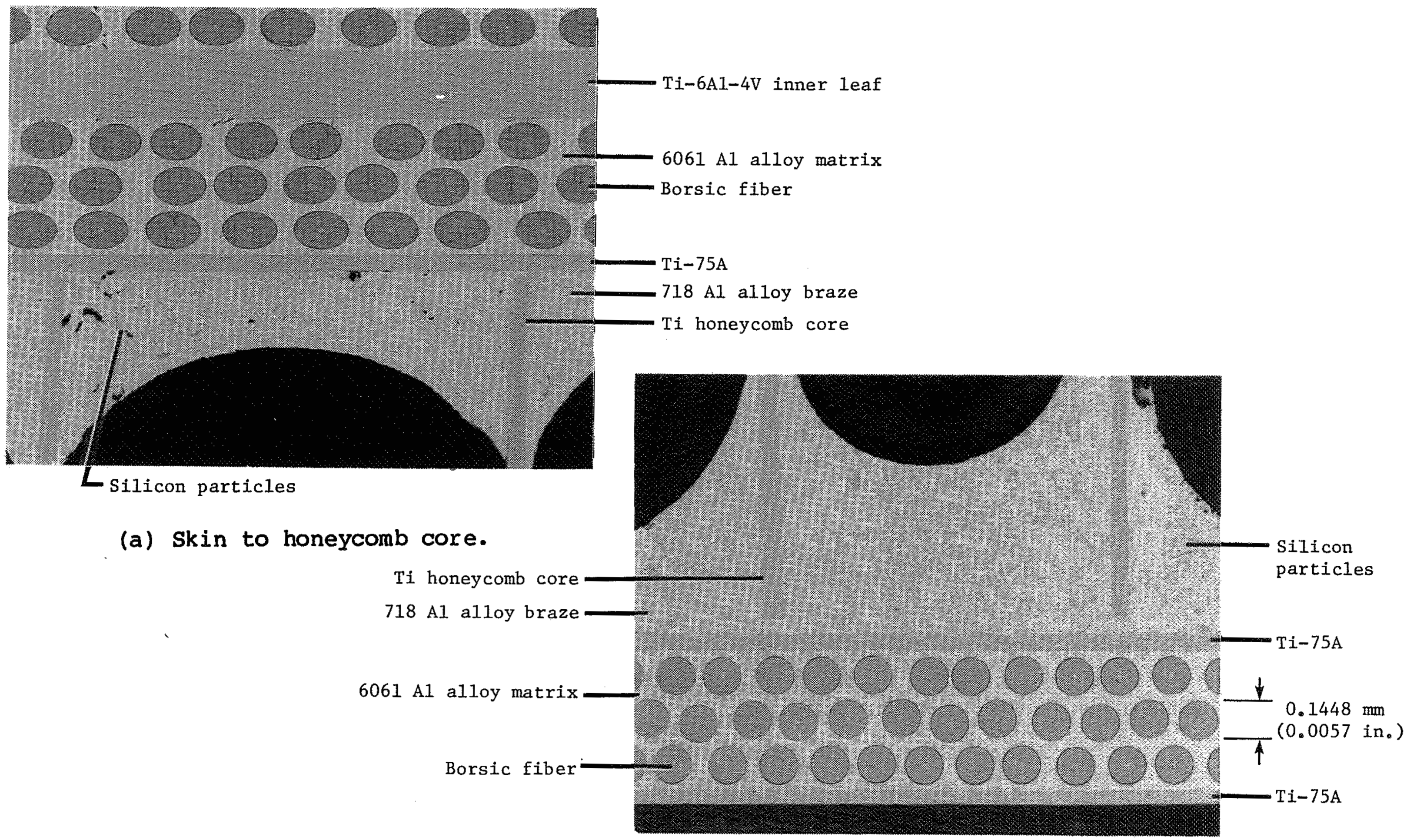
(b) Stringer side.

Figure 15.- Concluded.



(a) Skin to honeycomb core.

(b) Cap to honeycomb core.

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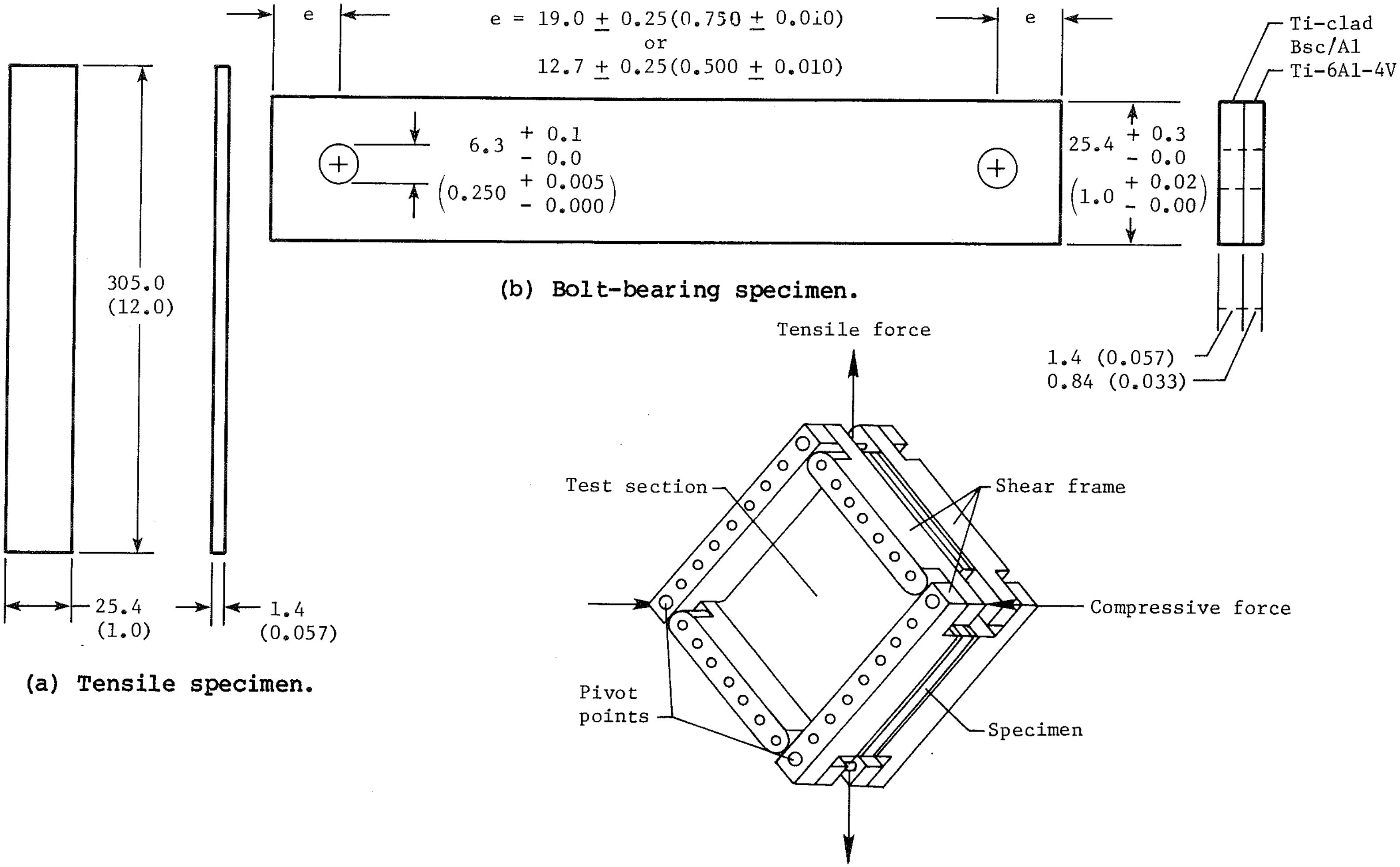
Figure 16.- Photomicrographs of typical skin-stringer braze joints.



(a) Tensile specimen.

(b) Bolt-bearing specimen.

(c) Picture-frame shear specimen.

Figure 17.- Structural element test specimens. Dimensions are given in millimeters (inches).

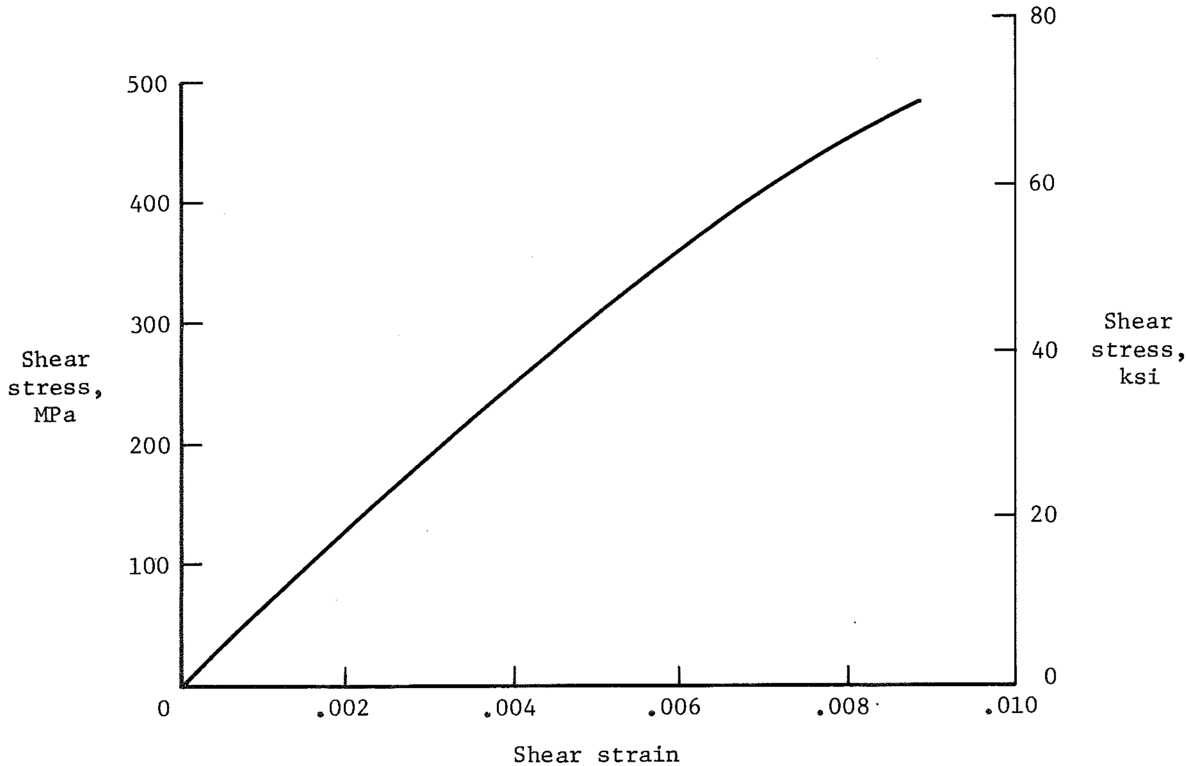


Figure 18.- Test results for structural element picture-frame shear specimen.

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15. Supplementary Notes

16. Abstract

A successful brazing process has been developed and evaluated for fabricating full-scale titanium-clad Borsic/aluminum skin-stringer panels. A unique panel design was developed consisting of a hybrid composite skin reinforced with capped honeycomb-core stringers. Six panels were fabricated for inclusion in the program which included laboratory testing of panels at ambient temperature and 533 K (500° F) and flight service evaluation on the NASA Mach 3 YF-12 airplane. All panels tested met or exceeded stringent design requirements and no deleterious effects on panel properties were detected following flight service evaluation on the YF-12 airplane.

17. Key Words (Suggested by Author(s))

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Picture-frame shear tests
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Skin-stringer panels
Hybrid metal-matrix composites
Brazed Ti-clad Bsc/Al Ti honeycomb core
Aluminum brazed titanium

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