NASA TECHNICAL MEMORANDUM

BRAZED BORSIC/ALUMINUM STRUCTURAL PANELS

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1. Report No. NASA TM X-3432	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle BRAZED BORSIC/ALUMINUM STRUCTURAL PANELS		5. Report Date March 1977
		6. Performing Organization Code
7. Author(s) Thomas T. Bales, H. Ross Wiant, and Dick M. Royster		8. Performing Organization Report No. L-11075
		10. Work Unit No. 743-01-22-01
9. Performing Organization Name and Address NASA Langley Research Center Hampton, VA 23665		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

H. Ross Wiant: Vought Corporation, Hampton, Virginia.

16. Abstract

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17. Key Words (Suggested by Author(s)) Borsic/aluminum Brazed Borsic/aluminum YF-12 flight panels Metal-matrix composites Shear panels		18. Distribution Statement Unclassified — Unlimited Subject Category 24	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 22	22. Price* $3.25

*For sale by the National Technical Information Service, Springfield, Virginia 22161.

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SUMMARY

A fluxless brazing process has been developed at the Langley Research Center that minimizes degradation of the mechanical properties of Borsic/aluminum composites. The process, which employs 718 aluminum alloy braze, is being used to fabricate full-scale Borsic/aluminum-titanium honeycomb-core panels for Mach 3 flight testing on the NASA YF-12 aircraft and ground testing in support of the Supersonic Cruise Aircraft Research (SCAR) Program. The manufacturing development and results of shear tests on full-scale panels are presented.

INTRODUCTION

The continuing demand for improved performance in advanced aerospace vehicles has increased the need for materials exhibiting higher structural efficiency. The high ratios of strength and stiffness to weight of composite materials are particularly attractive to meet these demands. In the metal-matrix class of materials, both boron/aluminum and Borsic/aluminum have undergone development for approximately 10 years but have had very few applications because of high material costs and limited fabricability. Factors which limit the fabricability are the hard and brittle nature of the boron and Borsic filaments and degradation of the mechanical properties of the composite due to thermal processing.

To realize the potential offered by metal-matrix composite materials, the Langley Research Center initiated a program to develop fabrication processes for incorporating these materials into advanced structures. Following an evaluation of the effects of joining on metal-matrix composites (ref. 1), brazing was selected as the process with the greatest potential for fabricating efficient complex structures. Borsic/aluminum (Bsc/Al) with a 6061 aluminum alloy matrix was selected rather than boron/aluminum (B/Al) because it is less affected by exposure to a brazing environment.

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In order to focus the research and show that brazing could be used to fabricate flight-quality hardware, upper wing panels of the Mach 3 NASA YF-12 aircraft (fig. 1) were selected for manufacturing development. Program plans include ground testing of panels following exposure to a simulated supersonic transport environment and flight service testing on the YF-12. Selection of the YF-12 panel also complements similar studies on titanium panels in support of Supersonic Cruise Aircraft Research (SCAR) Program (ref. 2). Reported herein are the manufacturing process development, non-destructive evaluation (NDE) techniques, and results of full-scale panel tests.

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 3.

DESIGN CONCEPT

The design concept selected for the YF-12 panel is shown in figure 2. The Bsc/Al-Ti honeycomb-core panel was designed to replace the original titanium integrally stiffened panel. The retrofit requirements imposed severe limitations on effective utilization of composite materials because of their inherent high stiffness and directional properties when compared with those of titanium. These requirements dictated that the panel measure 406 by 711 mm (16 by 28 in.), be capable of sustaining an ultimate shear flow of 0.525 MN/m (3000 lbf/in.), and have a maximum shear stiffness of 92 MN/m (0.525×10^6 lbf/in.) for ambient conditions. Test results indicated that to meet the shear load requirement and not exceed the shear stiffness allowable, the Bsc/Al skin of the panel should consist of seven plies with filaments oriented in the 45° direction. Consequently, the design concept shown employs an upper skin of four plies of Bsc/Al oriented in the +45°, -45°, -45°, +45° direction and a three-ply lower skin having an orientation of +45°, -45°, +45°. Four of the seven plies were oriented in the tensile direction because Bsc/Al is stronger in compression. Ti-6Al-4V titanium alloy was used to fabricate the frame assembly to carry the high bearing loads around the perimeter of the panel. The ramp of the frame was tapered at an angle of 30° to introduce load into the lower skin. Ti-3Al-2.5V titanium alloy honeycomb core was selected because of its good mechanical properties and its brazability using aluminum-base braze alloys. The braze alloy selected to join the Bsc/Al skins to the honeycomb core was 718 aluminum. This alloy was selected because of its favorable melting temperature and its good wetting and strength characteristics. In order to minimize the effect of interaction of the braze with the 6061 aluminum alloy matrix and Borsic filaments, the skin material was procured with a 0.127-mm-thick (0.005-in.) layer of 1100 aluminum alloy on the braze surface.

PANEL FABRICATION AND TESTS

Panel Components

The individual components of the panels are shown in figure 3. The four Ti-6Al-4V titanium alloy pieces for the frame were cold formed in a conventional power brake, machined to the required dimensions, and joined at the corners by electron-beam welding. Ti-6Al-4V titanium alloy sheet was also used for the flat rectangular-shaped doublers which were machined to length and then chem-milled to the proper thickness to mate the panel with the substructure of the YF-12 aircraft. The Ti-3Al-2.5V titanium honeycomb core had a 6.4-mm-square (1/4-in.), corrugated cell and a density of 72 kg/m^3 (4.5 lbm/ft^3). The honeycomb was stabilized for machining by bonding the core to a tooling plate by means of a water-soluble wax. A carbide "valve stem" cutter was used for machining the core to the required configuration. The Bsc/Al upper and lower skins were cut to size with a conventional metal shear.

Process Development

Initial tests were conducted to select a suitable brazing alloy and brazing parameters for joining Bsc/Al and titanium components. Based on these tests, a single-step brazing process using 718 aluminum braze alloy was selected for full-scale panel fabrication. Application of the process to full-scale panel fabrication resulted in difficulties with maintaining alinement of the numerous mating parts. Therefore, to simplify assembly and to improve the reliability, a two-step brazing process was developed. The first step consisted of brazing the titanium honeycomb core and doublers to the frame, and the second step involved brazing the Bsc/Al skins to the core and frame assembly. Because the first step involved brazing only titanium components, a braze alloy was selected that had a higher melting temperature and better wetting characteristics than 718 aluminum. The braze alloy selected was 3003 aluminum alloy which was used successfully by The Boeing Company to fabricate titanium sandwich structure for the Supersonic Transport Program. Second-step brazing employed 718 aluminum alloy to braze the Bsc/Al skins to the core-frame assembly.

Panel Assembly and Brazing

Prior to assembly for brazing, all components were chemically cleaned according to established procedures. Assembly for first-step brazing was initiated by positioning the core into the frame. Positioning was maintained by spot-welding titanium foil strips to both the cell walls of the honeycomb core and the frame. The titanium doublers and a 0.127-mm-thick (0.005-in.) strip of 3003 aluminum braze alloy were then spot-brazed to

the flanges of the frame. The assembled components are shown in the braze tooling in figure 4. The tooling consists of an upper and lower platen, a stainless-steel caul sheet, titanium release sheet, titanium honeycomb-core tooling, and a stainless-steel bladder. Strips of 0.25-mm-thick (0.010-in.) 3003 aluminum braze foil were placed between the honeycomb core and the release sheet during assembly. Brazing was accomplished by heating the tooling and components in a vacuum furnace to a temperature of 961 K (1270° F) for a period of 5 minutes at a pressure of 1.33 mPa (1×10^{-5} torr). Proper contact between mating parts was maintained during brazing by pressurizing the stainless-steel bladder with helium to 27.6 kPa (4 psi). The braze between the core and frame was established by the braze melting and flowing down the nodes of the honeycomb core. Capillary action of the molten braze resulted in uniform filleting between the core and frame. The core and frame assembly following brazing is shown in figure 5. Brazing of the core to frame in a separate operation provided for visual inspection for disbonds and facilitated rebrazing if necessary.

For the second-step braze assembly, the four-ply outer Bsc/Al skin and 0.2-mm-thick (0.008-in.) 718 braze foil were positioned on the core-frame assembly with resistance spot-brazes. The lower skin and braze alloy were held in place with titanium foil straps spot-welded to the frame across each corner. The assembled panel was placed in the tooling as shown in figure 6. The tooling was essentially the same as that shown in figure 4 with the exception that it was inverted and a larger pressurized bladder was used to apply pressure over the entire skin area. A photograph of the tooling in the vacuum furnace is shown in figure 7. The furnace has a circular, 838-mm-diameter (33-in.) hot zone and is capable of processing parts at temperatures up to 1922 K (3000° F). The equipment is capable of two-rate, two-soak programed resistance heating as well as closed recirculating inert-gas programed cooling. The temperature of the panel during brazing was monitored with nine thermocouples positioned at critical points. Suitable ports are provided through the chamber wall of the furnace to extend the thermocouples and pressure line of the bladder to the associated control and recording equipment. The time-temperature profile of the panel for brazing the Bsc/Al skins to the core-frame assembly is shown in figure 8. Following evacuation of the vacuum furnace to a pressure of 133 μPa (1×10^{-6} torr), the assembly was slowly heated to a temperature of 839 K (1050° F). Thermal equilibrium was established by holding the temperature at 839 K for approximately 10 minutes prior to heating to the brazing temperature of 864 K (1095° F). 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 minutes. Gas cooling was then discontinued and the panel was furnace cooled to ambient temperature.

Nondestructive Evaluation

Nondestructive evaluation (NDE) of the brazed panel consisted of ultrasonic C-scan and radiographic inspection. Suitable standards for each technique were established during development of the brazing process. These techniques were also used for inspection of panel components prior to brazing. Radiography was used to determine filament orientation, detect broken filaments in the skin material prior to panel fabrication, inspect the electron-beam welds in the frame, and inspect the braze between the Bsc/Al skin and the titanium frame. Ultrasonic C-scan was used to inspect the as-received Bsc/Al skins for delaminations and for filleting of the braze between the Bsc/Al skins and the titanium honeycomb core.

Panel Tests

Following inspection, the panel was trimmed with a diamond-impregnated wheel to final size and drilled with conventional high-speed drills to match the hole pattern in the test fixture. Following final quality assurance inspection, the panel was shipped to the Advanced Development Projects Division of Lockheed-California Company where it was instrumented with strain gages and mounted in the picture-frame test fixture having pinned corners as shown in figure 9. The panel was loaded in shear by applying a tensile load to diagonally opposite corners. Strain-gage readings indicated that a state of pure shear was achieved in the center of the panel on loading. The effects of slack in the load train were eliminated by zeroing the strain gages after the application of a load of 89 kN (20 kips). The panel was then loaded to the design limit load of 267 kN (60 kips) in 44.5-kN (10-kip) increments and then unloaded. On reloading the panel, 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 to failure. Strain-gage outputs were recorded following the application and stabilization of load for each increment.

RESULTS AND DISCUSSION

Shear Tests

The load-strain data obtained from the shear test of the first full-scale panel are presented in figure 10. The shear strains were calculated as the sum of the absolute readings of two 45° gages in the center of the panel. The panel was tested by loading to design limit load, unloading to 89 kN (20 kips), and reloading incrementally to failure. As shown in figure 10, the shear stiffness obtained on first loading was approximately 50 percent below that obtained on second loading. The stiffness on first loading was 15 percent below the design allowable whereas that obtained on second loading up to design limit was 30 percent above the design allowable. Extrapolation back to zero load

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of the data obtained on second loading to design limit indicated that the panel experienced a permanent strain of 1400 microunits. On second loading, from design limit load to failure, the panel responded according to the lower initial shear stiffness. Failure occurred suddenly at a load of 427 kN (96 kips) which corresponds to a shear stress of 0.338 GPa (49 ksi) and equals 93.2 percent of design ultimate. Although the shear stiffness values obtained from strain-gage readings indicate that the panel exceeded the stiffness allowable, examination of load displacement data showed that the shear stiffness based on overall deformation met the stiffness requirement with a 15-percent margin.

Photographs of the upper and lower surfaces of the panel following failure are shown in figure 11. Examination of the data obtained from strain gages located near the welded corner of the titanium frame, as shown in figure 11(b), indicated that the welded corner failed at a load less than 178 kN (40 kips). This local failure apparently permitted the outer skin to buckle and resulted in sudden catastrophic failure of the structure. Destructive evaluation of the panel verified the earlier findings of ultrasonic and radiographic inspection that a good quality braze was achieved in all joints.

The variation in panel stiffness between first and second loading can be attributed to the state of residual stress in the composite. As reported in reference 4, consolidation of the composite induces high residual tensile stresses in the aluminum matrix and residual compressive stresses in the Borsic filaments. Consequently, the application of a tensile load leads to yielding of the aluminum at low stress levels. This results in a low initial modulus for the composite due to limited elastic response of the matrix. Following tensile yielding of the matrix and unloading of the composite, the magnitude of the residual tensile stresses is reduced and may even become compressive in nature. Reloading of the composite results in an increased stiffness due to increased elastic contribution from the matrix.

Following detailed analysis of all test data and destructive examination of the failed panel, a second full-scale panel was fabricated which incorporated several minor design modifications to increase the ultimate shear load capability. The welded corners of the frame were strengthened by brazing 0.813-mm-thick (0.032-in.) preformed clips over the corners during first-step brazing. The density of the honeycomb core 127.0 mm (5 in.) in from each end of the panel was increased from 72 kg/m^3 (4.5 lbm/ft^3) to 128 kg/m^3 (8 lbm/ft^3) to increase resistance to transverse shear failure of the honeycomb in the vicinity of the corners of the panel. Finally, the number of plies of Borsic fibers in the lower skin was increased from three to four which increased the total number of plies in the panel from seven to eight and produced a balanced composite layup.

The modified panel was tested according to the procedures described previously, and the test results are presented in figure 12. The shear load-strain relationship exhib-

ited is similar to that obtained for the previous panel test in that the stiffness on second loading is approximately 50 percent greater than on initial loading to design limit load.

Failure occurred at a shear load of 0.512 MN (115 kips) which corresponds to 125 percent of design shear ultimate. The shear stiffness, based on overall panel deformation, equaled the upper limit of the design allowable and exceeded that of the first panel by approximately 15 percent. Based on these results the design modifications incorporated increased the shear strength of the panel by approximately 30 percent and resulted in a panel that fully met the ambient temperature design requirements.

Metallurgical Analysis

Failure analysis of the first panel included a metallurgical analysis of the brazed joint between the titanium honeycomb core and the Bsc/Al skins. One of the major problems normally encountered in fabricating Bsc/Al by brazing is the degradation in material properties caused by reaction of the braze alloy with both the matrix and filaments of the composite. The 6061 aluminum alloy matrix has an incipient melting temperature of 855 K (1080° F) which results in rapid liquid-liquid diffusion when brazed with 718 aluminum alloy at approximately 866 K (1100° F). In order to minimize the interdiffusion on interaction between the 6061 matrix and 718 braze, the Bsc/Al skins for the panel were procured with a 0.127-mm-thick (0.005-in.) layer of 1100 aluminum bonded to the braze surface. The incipient melting temperature of 1100 aluminum is approximately 311 K (100° F) higher than the 6061 aluminum matrix and, therefore, acts as a diffusion barrier to restrict reaction between the braze alloy and the constituents of the composite.

Photomicrographs of the brazed joint between the Bsc/Al skins and the titanium honeycomb core are shown in figure 13. The photomicrograph on the left shows good filleting between the honeycomb core and the composite, which indicates good wetting and flow of the braze alloy. The photomicrograph on the right depicts the effectiveness of the 1100 aluminum as a diffusion barrier. The silicon particles in the braze alloy stop fairly abruptly at the interface with the 1100 aluminum, which indicates little or no reaction between the braze alloy and the constituents of the composite. Perhaps more important is that the titanium honeycomb core has not been forced through the outer aluminum layer to damage the filaments as frequently occurs in pack brazing or when the pressure applied to maintain alinement is high. This problem was avoided by the use of the pressurized bladder which facilitated the application of a controlled pressure to mating parts during brazing.

CONCLUDING REMARKS

A satisfactory process has been developed for the fluxless brazing of Borsic/ aluminum (Bsc/Al) by using 718 aluminum braze. The degradation of properties normally caused by interaction of the braze alloy with the constituents of the composite was alleviated by using an 1100 aluminum alloy diffusion barrier and by establishing brazing parameters which minimize the time at the brazing temperature. Satisfactory nondestructive evaluation methods employing ultrasonic C-scan and radiographic inspection techniques were established for inspecting brazed Bsc/Al joints.

Full-scale Borsic/aluminum-titanium honeycomb-core structural panels, designed to meet the requirements of an upper wing panel for the YF-12 aircraft, have been successfully brazed. Test results obtained on an initial panel met the design requirements for bearing and shear stiffness and carried 93.2 percent of the design ultimate shear load. A second panel incorporating several design modifications complied with all the ambient temperature design requirements and carried 125 percent of design ultimate shear load. Additional panels are scheduled to be fabricated for ground testing following exposure to a simulated supersonic transport environment and for flight service on Mach 3 YF-12 aircraft.

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Hampton, VA 23665
December 2, 1976

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2. Hoffman, Edward L.; Payne, LeRoy; and Carter, Alan Lyman: Fabrication Methods for YF-12 Wing Panels for the Supersonic Cruise Aircraft Research Program. Materials Review '75, Volume 7 of National SAMPE Technical Conference Series, Soc. Advance. Mater. & Process Eng., c.1975, pp. 68-82.

3. Metric Practice Guide. E 380-72, American Soc. Testing & Mater., June 1972.

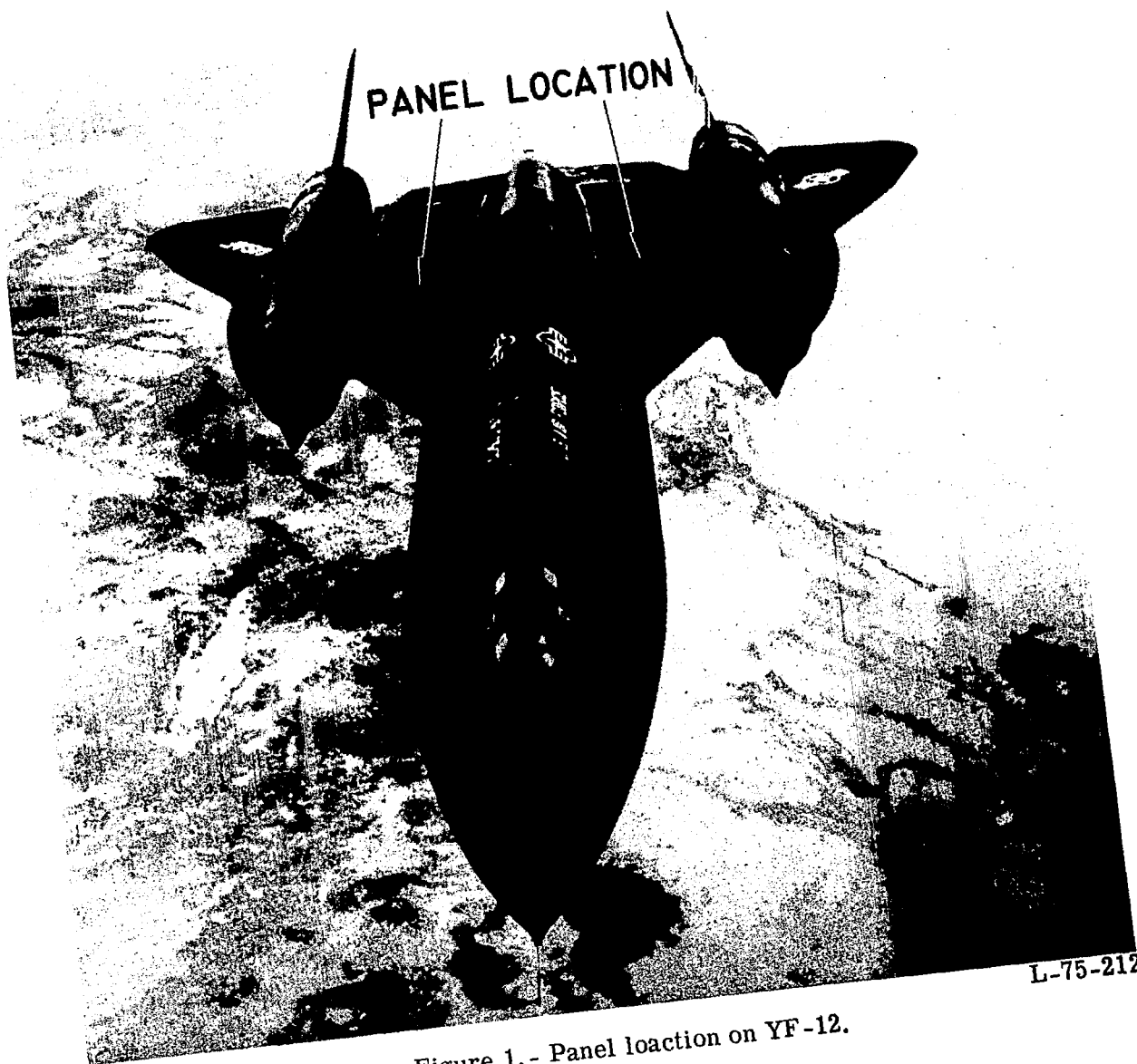4. Kreider, Kenneth G., ed.: Metallic Matrix Composites. Academic Press, Inc., 1974.

PANEL LOCATION

L-75-212

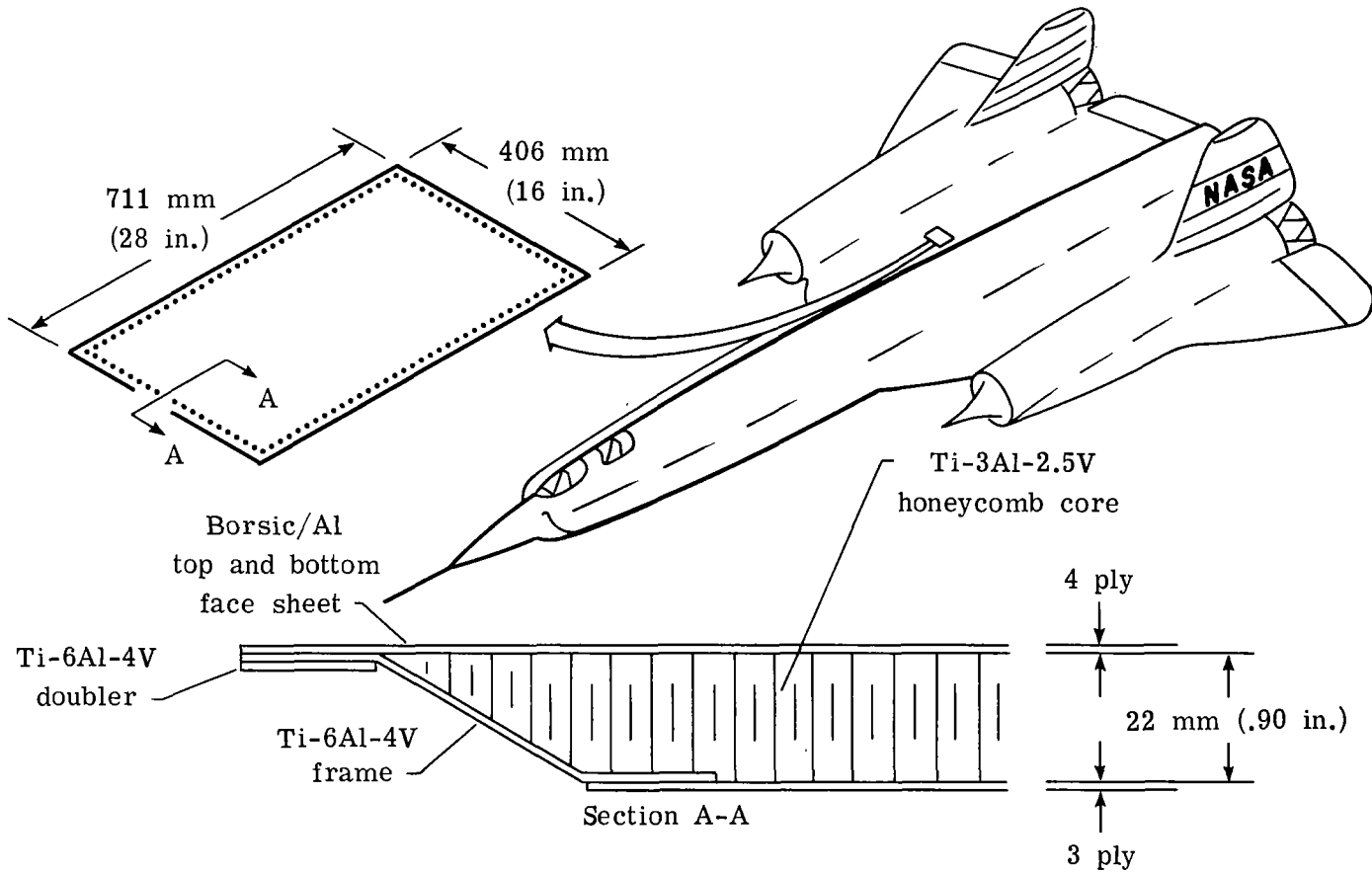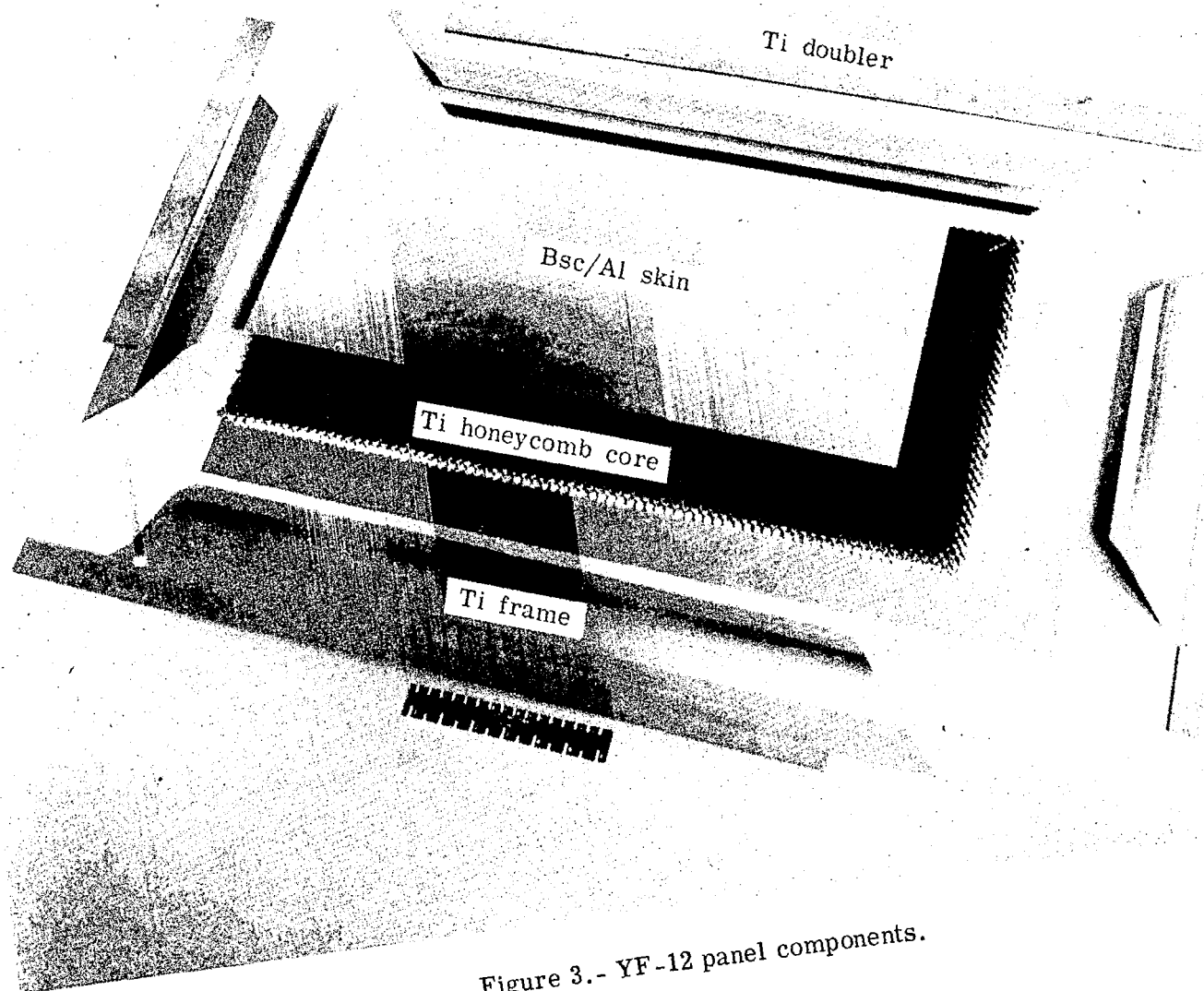Figure 1.- Panel loaction on YF-12.

Figure 2.- Bsc/Al-Ti honeycomb-core panel design.

Ti doubler

Bsc/Al skin

Ti honeycomb core

Ti frame

L-75-1328.1

Figure 3.- YF-12 panel components.

Platen

Caul sheet

Braze alloy

Release sheet

Tooling core

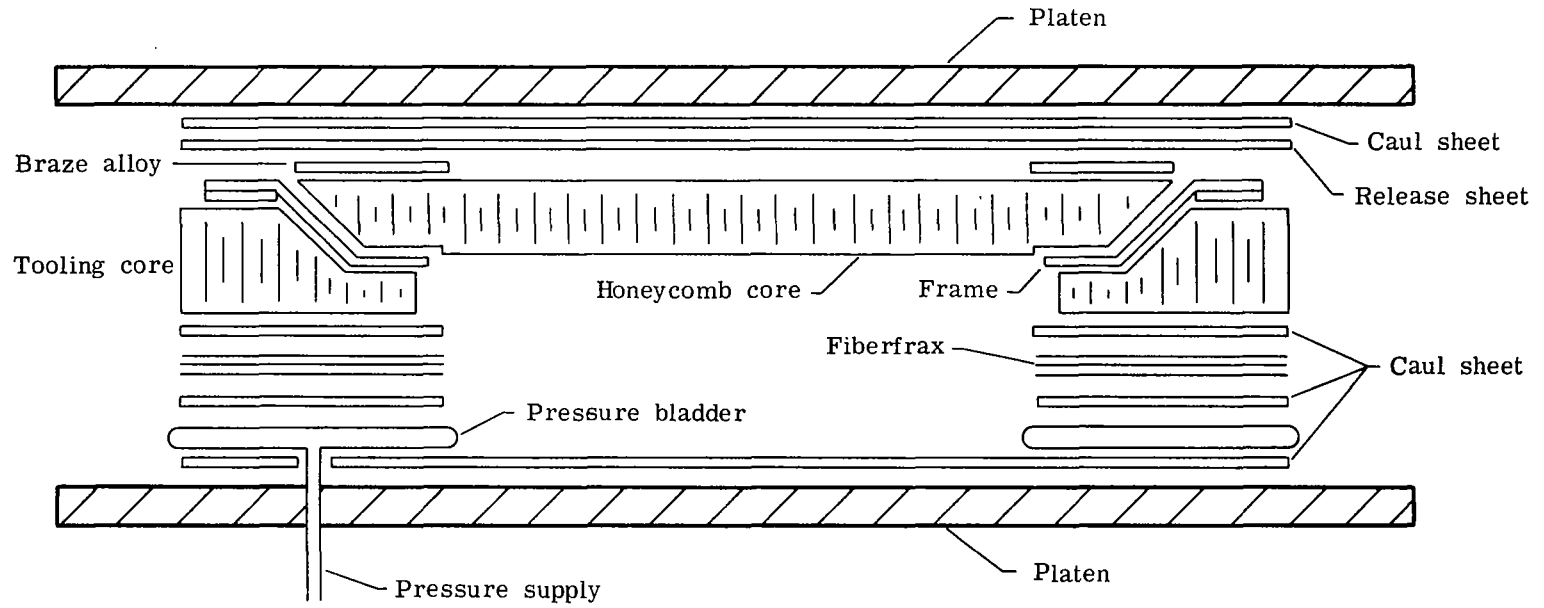Honeycomb core

Frame

Fiberfrax

Caul sheet

Pressure bladder

Platen

Pressure supply

Figure 4.- Cross section of tooling for first-step brazing.
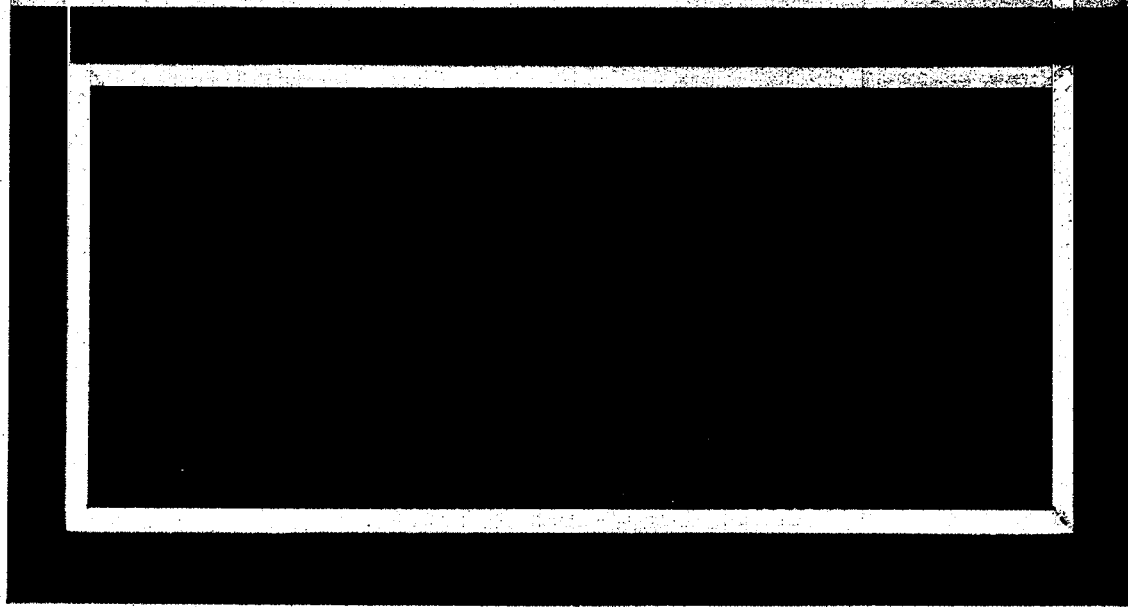
Figure 5.- Ti honeycomb-core and frame assembly following brazing.

L-76-353

Pressure supply

Stainless-steel caul sheet

Bladder

Steel platen

Fiberfrax

Bsc/Al skins

Steel platen
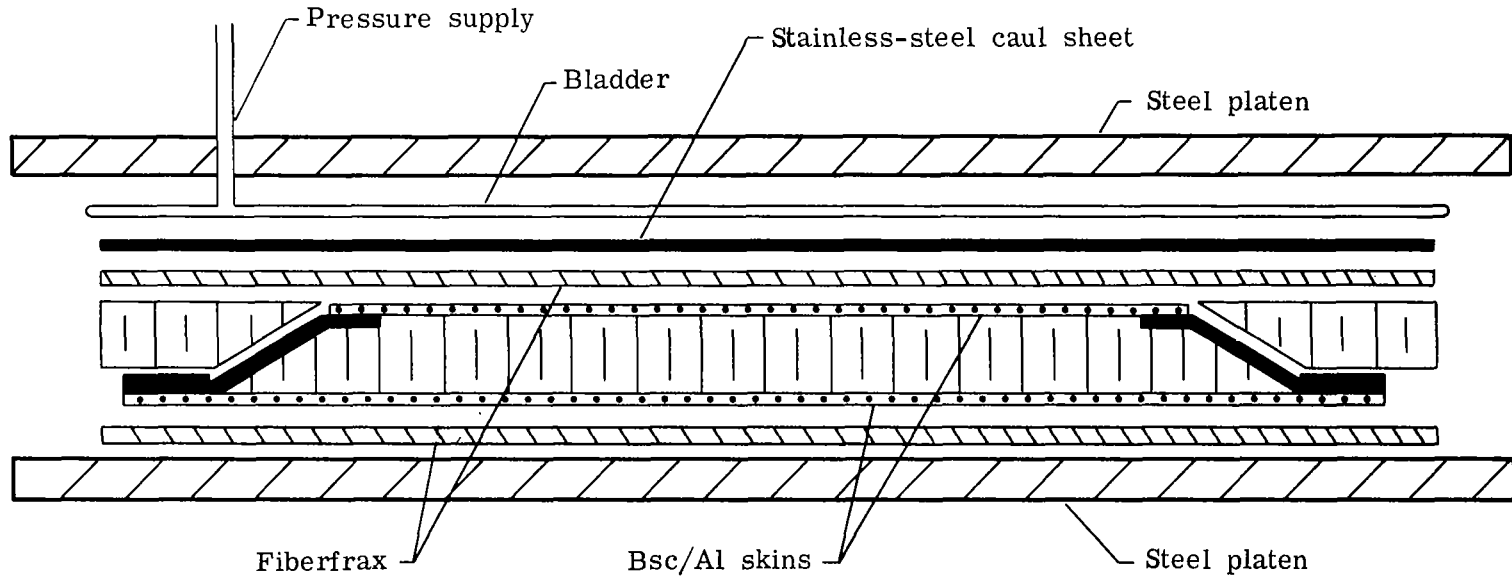
Figure 6.- Cross section of tooling for second-step brazing.
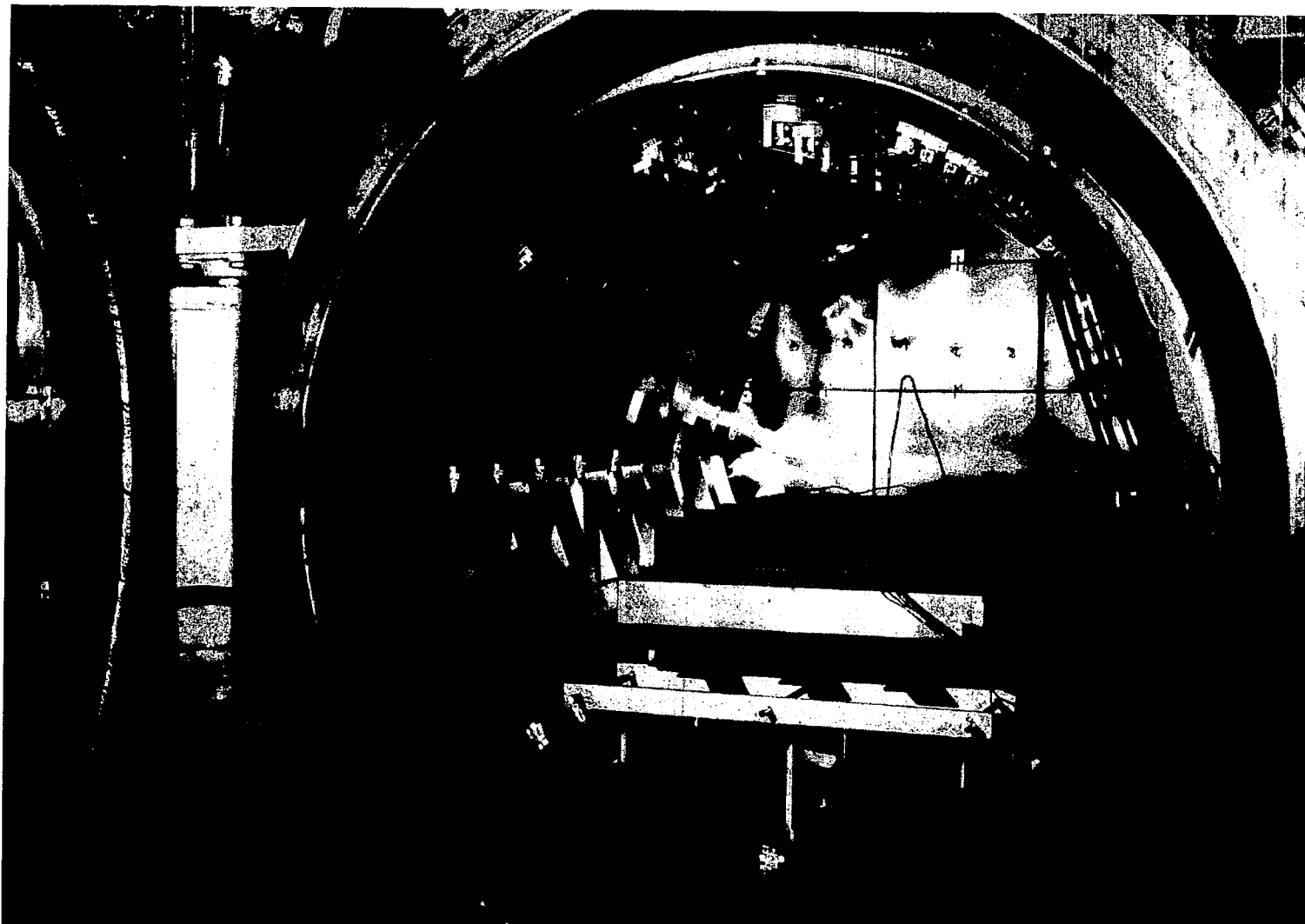
L-76-351

Figure 7.- Braze tooling in vacuum furnace.
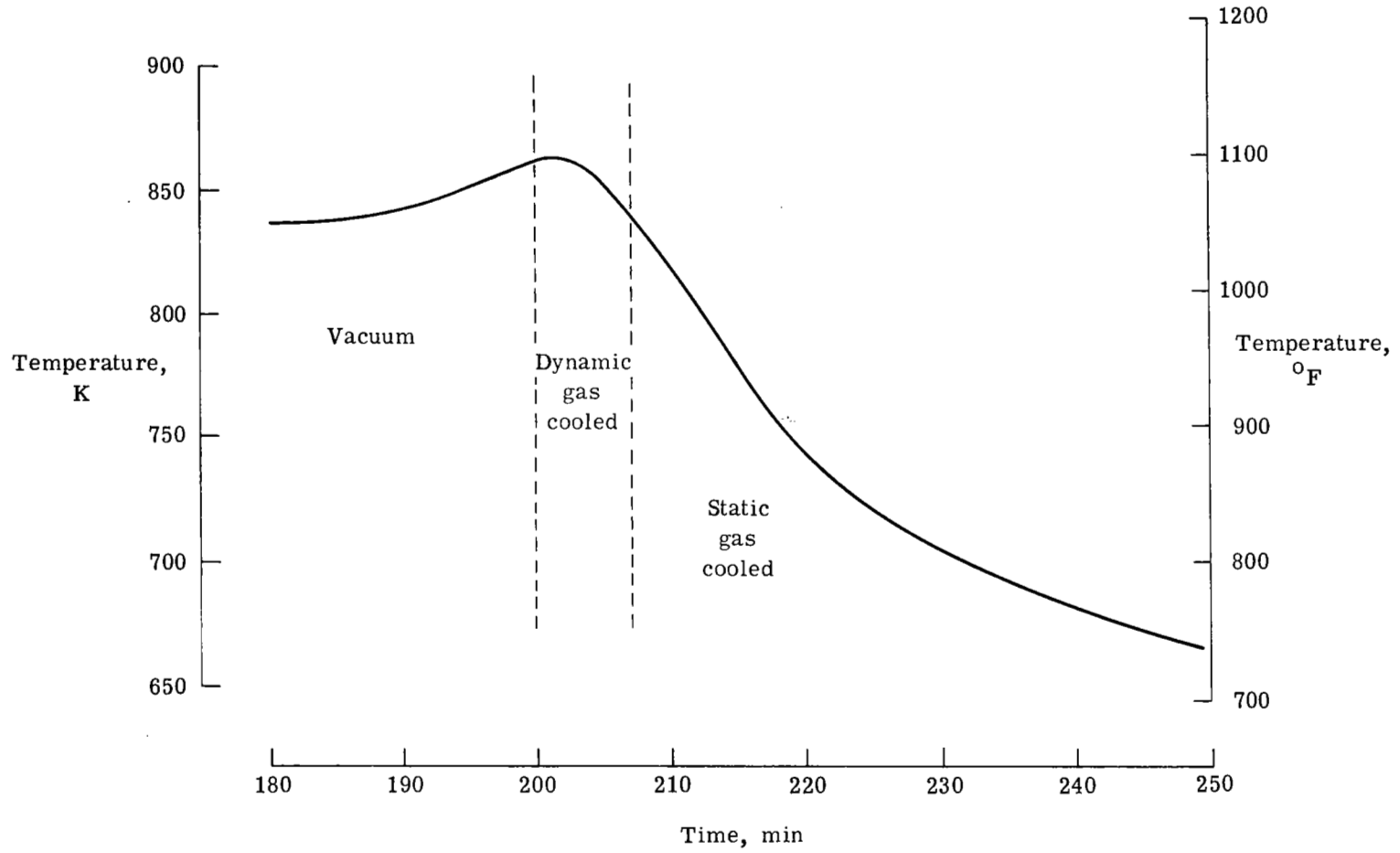
Figure 8.- Time and temperature brazing profile.
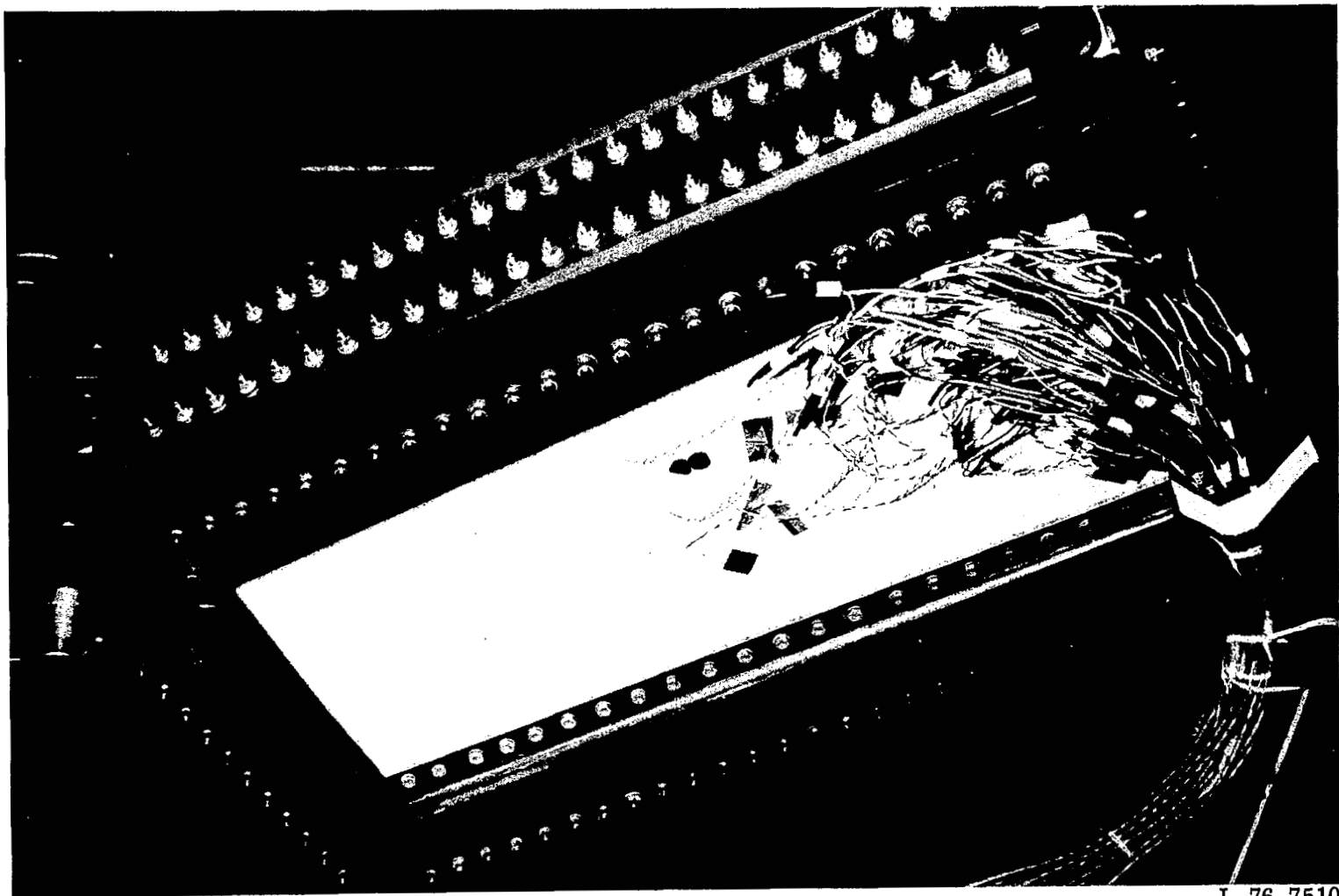
L-76-7510

Figure 9.- Panel shear test fixture.
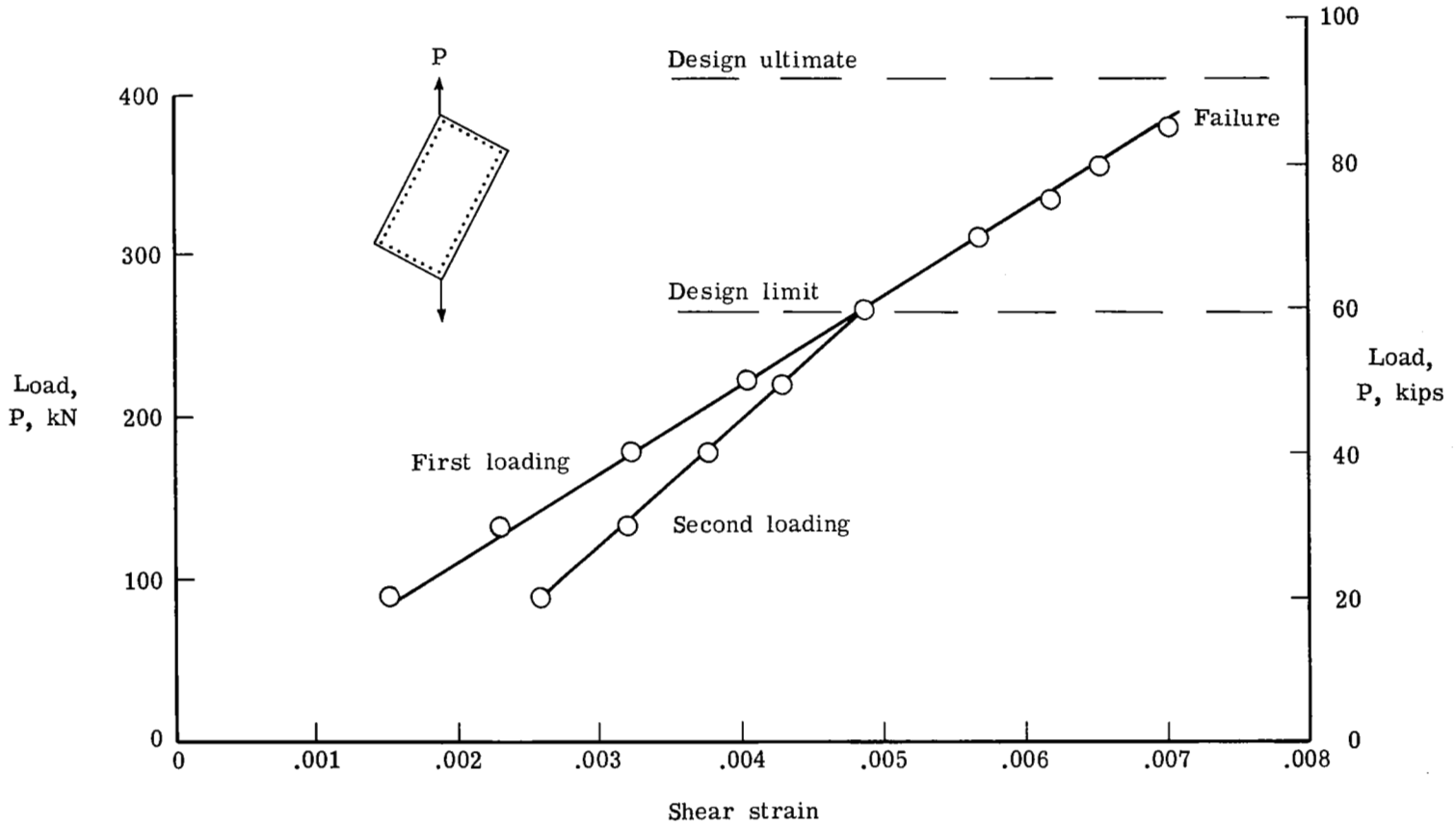
Figure 10.- Shear test data for first panel.

Load
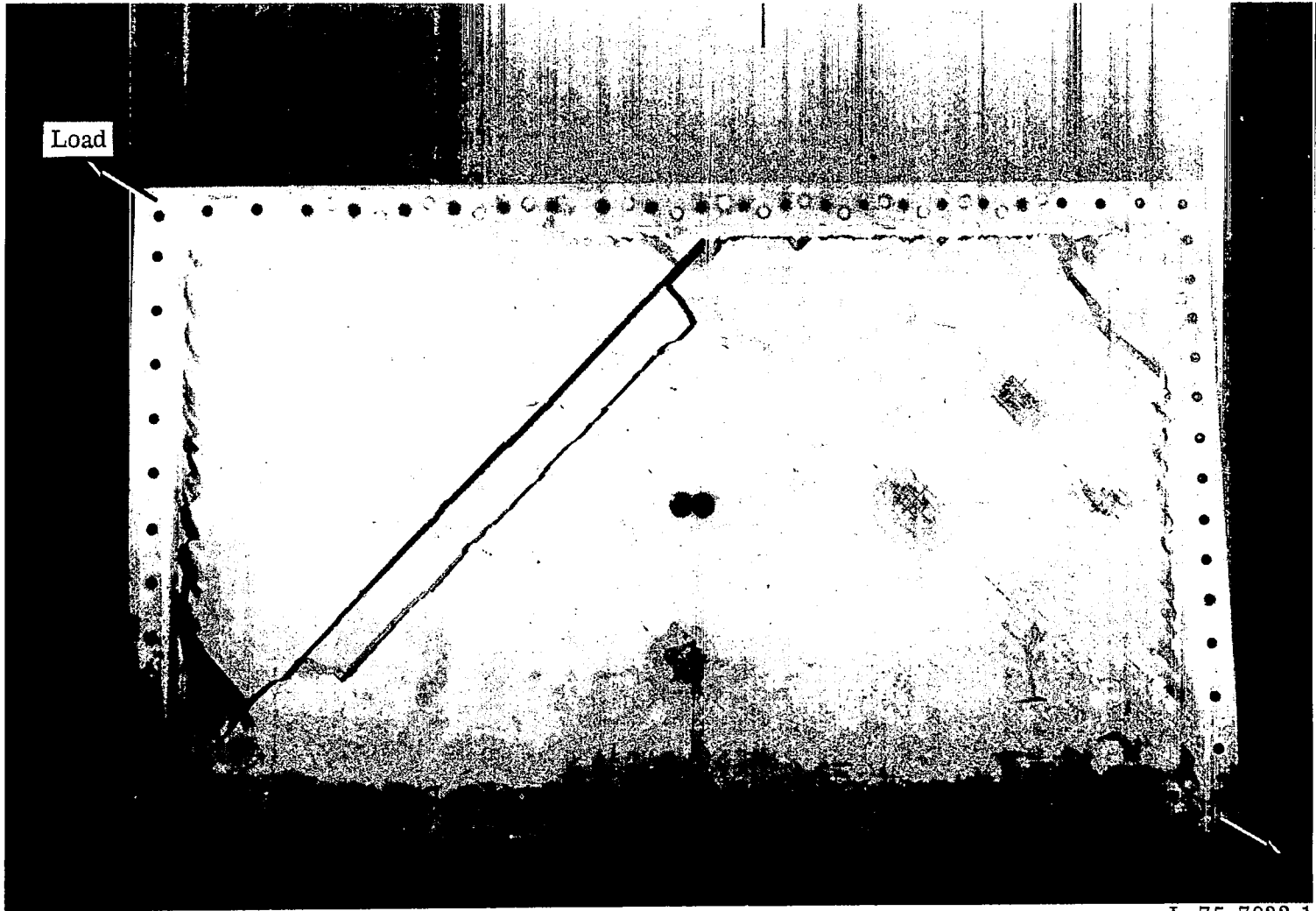
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(a) Upper skin.

Figure 11.- Panel after testing.
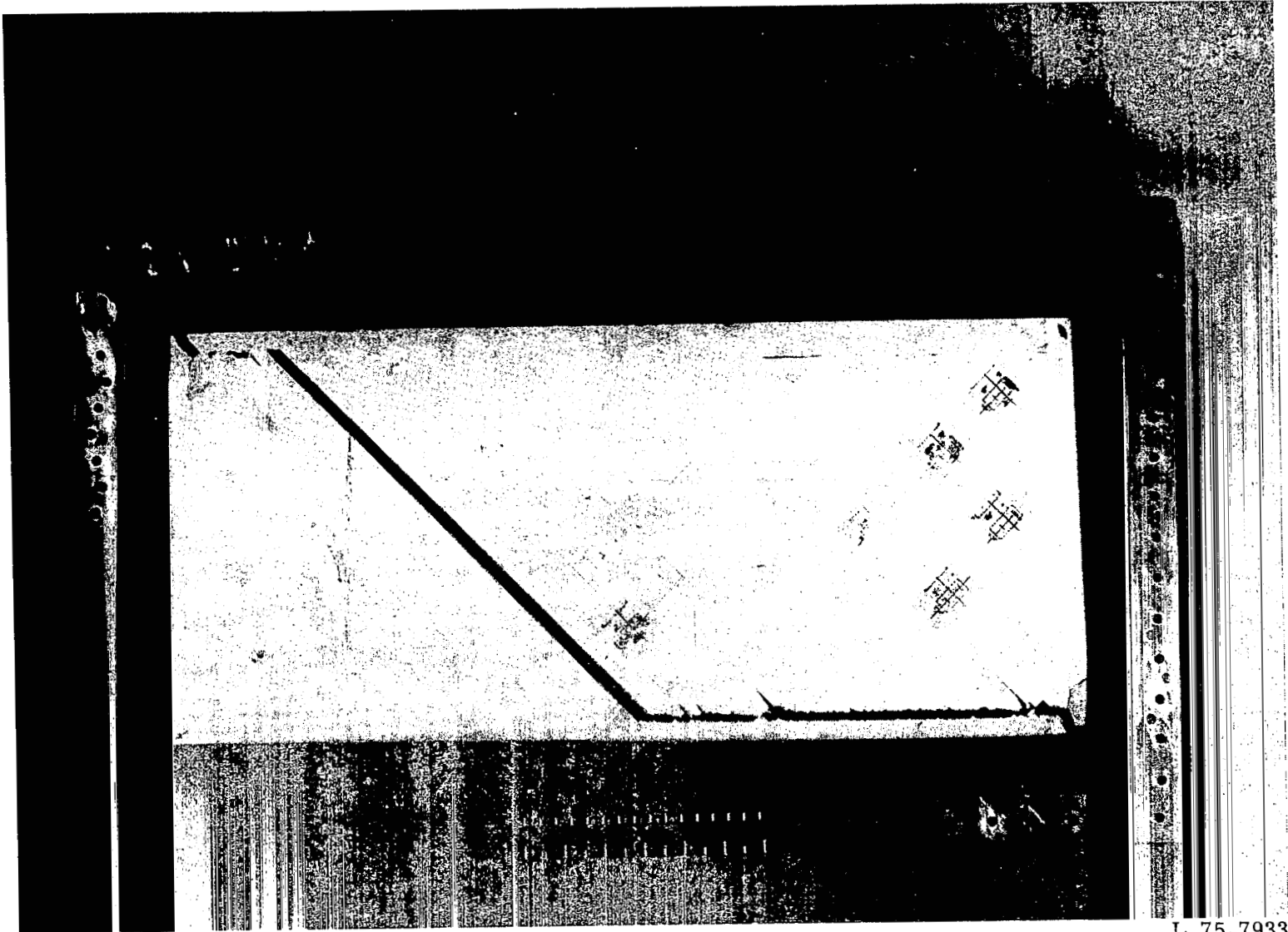
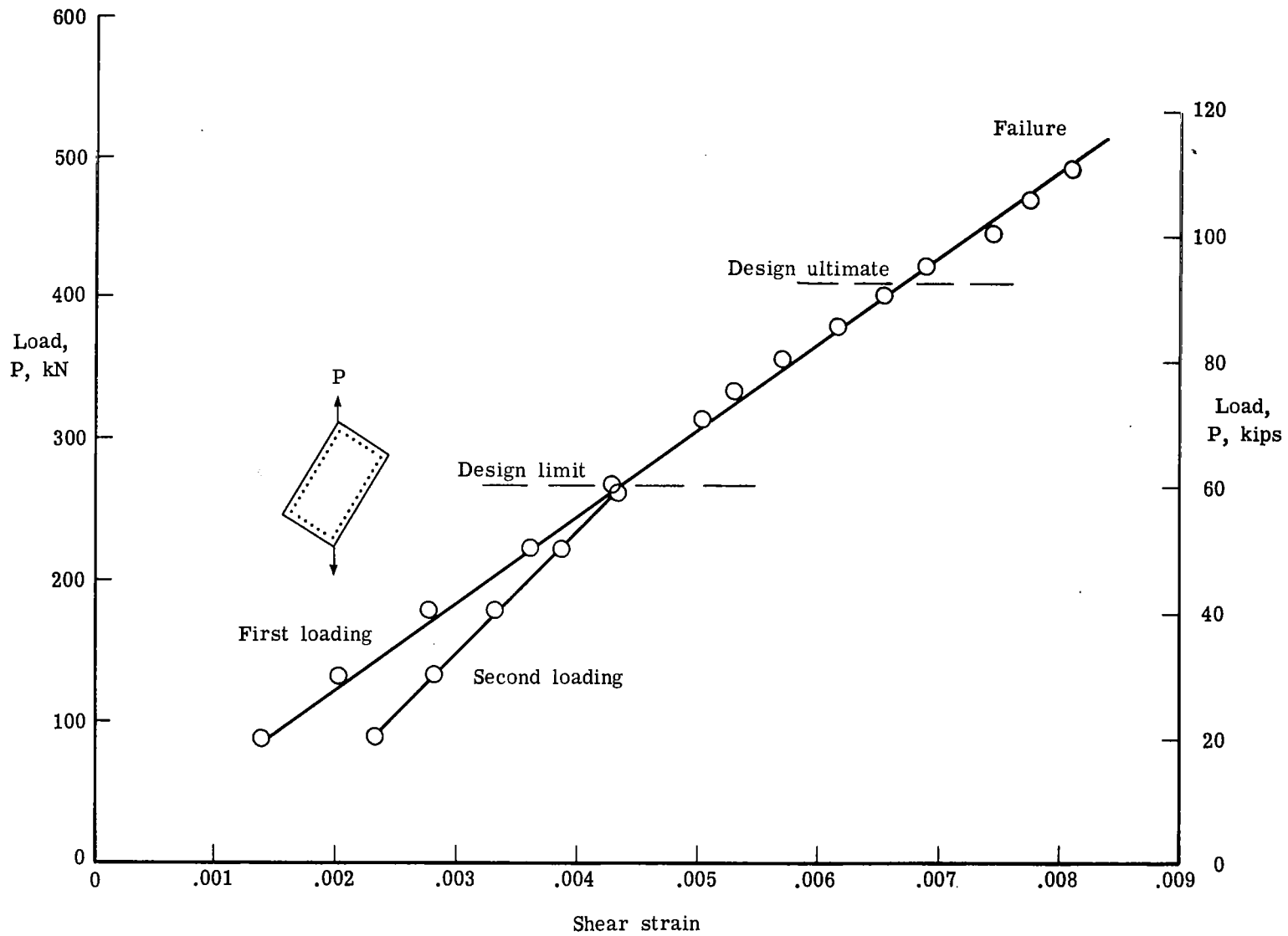L-75-7933

(b) Lower skin.

Figure 11.- Concluded.
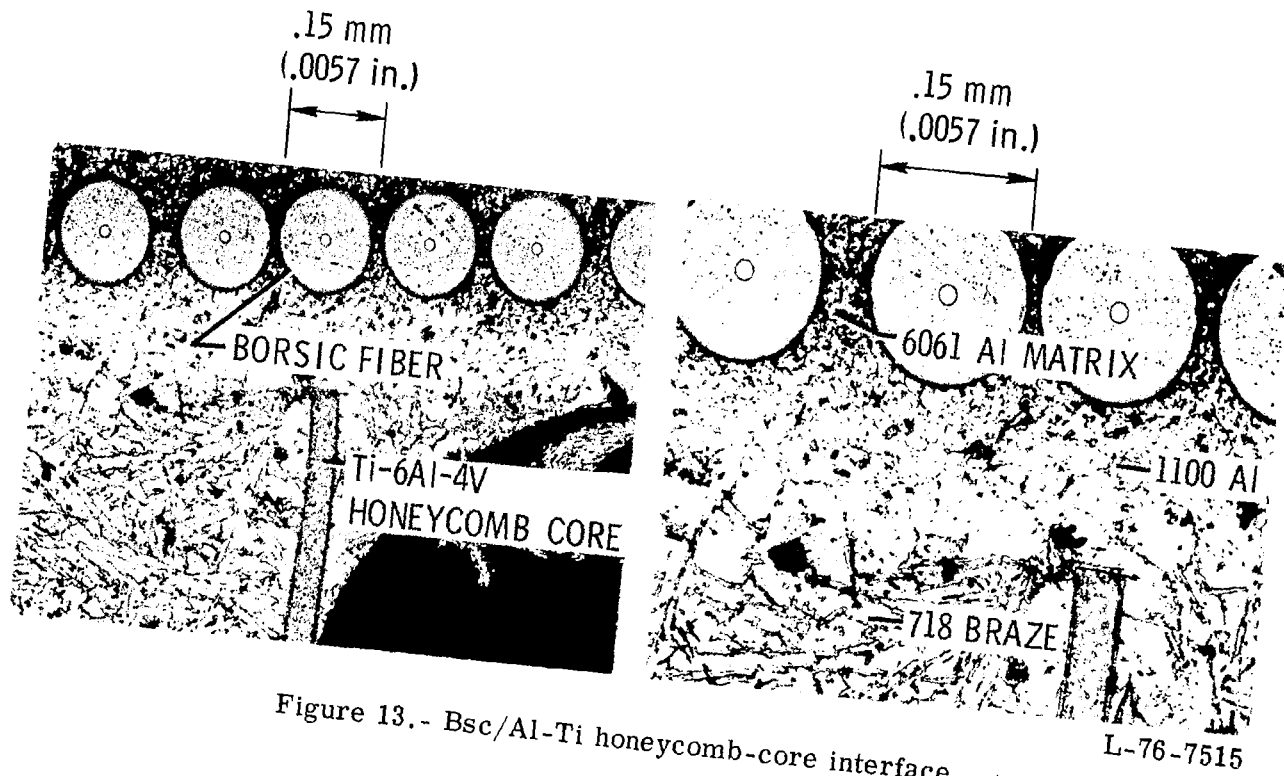
Figure 12.- Shear test data for second panel.

.15 mm
(.0057 in.)

.15 mm
(.0057 in.)

BORSIC FIBER

Ti-6AI-4V
HONEYCOMB CORE

6061 AI MATRIX

1100 AI

718 BRAZE

Figure 13.- Bsc/Al-Ti honeycomb-core interface.

L-76-7515

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."
—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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