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JOINING AND FABRICATION OF METAL-MATRIX COMPOSITE MATERIALS

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16. Abstract

Manufacturing technology associated with developing fabrication processes to incorporate metal-matrix composites into flight hardware is being studied at the Langley Research Center. The joining of composite to itself and to titanium by innovative brazing, diffusion bonding, and adhesive bonding requires such processes. The effects of the fabrication processes on the material properties and their influence on the design of YF-12 wing panels are discussed.

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JOINING AND FABRICATION OF METAL-MATRIX COMPOSITE MATERIALS

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SUMMARY

With increased emphasis being placed on the application of advanced metal-matrix composite materials to aerospace vehicles, a need for additional research on the joining and forming of these materials is required. Studies are being conducted on the joining of boron/aluminum and Borsic/aluminum composites to themselves and to titanium alloys. The processes being evaluated for fabricating flight quality hardware are spotwelding, brazing, adhesive bonding, and diffusion bonding. Selected processes are used to fabricate wing panels for the YF-12. The test results to date show that metal-matrix composites can be fabricated and joined by conventional processes. However, each process has its advantages and disadvantages. Borsic/aluminum is brazed quite readily at temperatures up to 876 K (1100° F), but an 1100 aluminum alloy barrier is used to prevent matrix degradation. The thermal processing associated with the brazing of boron/aluminum above 833 K (1040° F), results in serious fiber degradation. Spotwelding and adhesive bonding are shown to be viable joining processes with moderate strengths. Data indicate that diffusion bonding is a promising joining method for metal-matrix composites, the resulting joints exhibiting high strength.

INTRODUCTION

With increased emphasis being placed on the application of advanced metal-matrix composite materials to aerospace vehicles, additional research on the joining and fabrication of these materials is required. Questions still exist regarding the effects of joining and fabrication methods on the strength of the composite materials. Comparative studies of various joining methods for boron/aluminum and Borsic/aluminum composites are underway at the National Aeronautics and Space Administration (NASA) Langley Research Center. This report covers recent results of several programs and provides some insight into the advantages and disadvantages of various joining and fabrication methods.

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Studies were conducted on the joining of metal-matrix composites to themselves and to titanium alloys. The joining methods evaluated were spotwelding, brazing, adhesive bonding, and diffusion bonding. The purpose of these studies is not only to investigate the capabilities and limitations of composite structures, but also to develop the expertise necessary to fabricate full-scale flight-quality hardware for supersonic aircraft.

Test specimen configurations include single- and double-overlap tensile-shear specimens, honeycomb core-shear specimens, skin-stringer panels, and YF-12 wing panels. Figure 1 shows a YF-12 aircraft and the location for which wing panels of 406 mm by 711 mm (16 in. by 28 in.) are to be fabricated. Both industry and the Langley Research Center are involved in a program to fabricate panels for the YF-12 utilizing various materials and fabrication processes. One panel discussed in this report is being fabricated at the Langley Research Center. It consists of Borsic/aluminum face sheets brazed to a titanium frame and honeycomb core.

JOINT CHARACTERIZATION

Materials and Specimens

The main objective of the joining studies is to characterize the strength of the joints at room and elevated temperatures for the materials under investigation. Joints are being made from Borsic/aluminum and boron/aluminum, and these materials are either joined to themselves or to Ti-6Al-4V titanium alloy. Table I describes all the materials used in these studies.

The configurations of the test specimens for determining joint strengths are shown in figure 2. Single-overlap test specimens were fabricated by spotwelding and adhesive bonding, whereas double-overlap test specimens were fabricated by brazing. The single-overlap specimens were fabricated from a 1.67-mm-thick (0.066-in.) Borsic/aluminum sheet joined to a 1.34-mm-thick (0.053-in.) Ti-6Al-4V titanium sheet. (See fig. 2(a).) The overlap for the adhesive-bonded specimens was 25.4 mm (1 in.) and the overlap for the spotwelded specimens was 12.7 mm (0.5 in.). The spotwelded specimens were joined by two spotwelds spaced 25.4 mm (1 in.) apart across the width of the specimen.

The double-overlap specimens were fabricated from two strips of Ti-6Al-4V titanium alloy sheet brazed to two pieces of Borsic/aluminum. (See fig. 2(b).) The overlap for the brazed specimens was 6.3 mm (0.25 in.).

The single-overlap specimens for the stress rupture tests consisted of boron/aluminum strips joined by a single spotweld. (See fig. 2(c).) The overlap for the stress rupture specimen was 12.7 mm (0.50 in.).

Tensile-Shear Tests

The tensile-shear strength of Borsic/aluminum-titanium joints fabricated by spotwelding, adhesive bonding, and brazing is shown in figure 3. Maximum test load is plotted against test temperature, and data are presented for both single-overlap and double-overlap specimens. The symbols represent the average of three tests, and the scatter in the data is depicted by the bars. The curves are drawn through the average of the test data. For convenience in comparing the data, the bond areas for the single- and double-overlap specimens were normalized to 645 mm^2 (1 in^2). Regardless of the configuration or joining method, all the test specimens exhibited shear failures in the overlap.

The test results for the specimens are tabulated in table II. The spotweld data show no reduction in tensile strength for tests at room temperature (RT) and at 478 K (400° F) but do show a 25-percent reduction at 589 K (600° F).

The adhesively bonded, single-overlap specimens were joined by the use of a polyimide adhesive designated LRC-3A which was developed at the Langley Research Center. Adhesive bonds were cured at 611 K (640° F) and 1.38 MPa (200 psi) for 1 hour. Tests were conducted at room temperature, 450 K (350° F) and 533 K (500° F). The data obtained at 533 K (500° F) show a 30 percent less load-carrying ability than at room temperature.

The double-overlap specimens were brazed by using 718 aluminum alloy braze at a temperature of 856 to 861 K (1080° to 1090° F) for 5 minutes with a pressure of 34.5 kPa (5 psi). The data show that at 589 K (600° F) the load-carrying ability was 50 percent less than at room temperature. The loss in the load-carrying ability is attributed to the lower strength of the 718 braze alloy at 589 K (600° F). Although a direct comparison of the single-overlap to the double-overlap data is impossible, the brazed-joint data do show a greater magnitude in maximum load over the spotwelded and adhesive-bonded joints for the temperatures of the tests. However, other factors related to fabrication may dictate the use of spotwelding and adhesive bonding.

Stress-Rupture Tests

Stress-rupture properties of spotwelded boron/aluminum specimens (see fig. 2(c)) are shown in figure 4. The figure shows the applied load plotted against time to rupture on a log scale. The data shown are for specimens joined by a single spotweld and tested at 478 K (400° F) with applied loads equal to 75, 70, 60, and 50 percent of the maximum load obtained at room temperature. Heating was accomplished by using tube furnaces attached to conventional creep-testing machines (deadweight loading machines). The specimens were loaded and exposed continuously at the selected load until rupture occurred. Exposure time was monitored from initiation of the load to failure with the use of digital clocks. The data show a considerable loss in the ability of the spotwelds to carry the applied load at prolonged elevated temperature exposures. At 50 percent of the maximum room-temperature strength, the specimens failed in approximately 100 hours. From these data the use of spotwelded joints for boron/aluminum appears to be limited to a relatively low temperature because of poor stress-rupture properties.

Skin-Stringer Tests

In order to further characterize the joining methods, hat-shaped skin-stringer panels were fabricated and tested in end compression. The hat-shaped stringers consisted of 6 plies of either unidirectional boron/aluminum or Borsic/aluminum and were purchased from industry. The stringers were fabricated by several methods including either hot or cold forming or eutectic bonding. The stringers were joined to skins of 9-ply boron/aluminum or 9-ply Borsic/aluminum by spotwelding, brazing, and diffusion bonding.

The panel shown in figure 5 was fabricated by spotwelding. Each end of the 254-mm-long (10-in.) panel was "potted" with a room-temperature epoxy potting compound to facilitate machining the ends flat, parallel, and perpendicular to the face sheet. The potting was left on during testing to prevent crushing the panel ends and thus initiating a premature failure. The panel was strain-gaged to obtain strain data, and linear variable differential transformers were used to measure panel shortening. The panel dimensions were chosen so that the web, flange, and skin would bend as a result of local buckling in order to test the integrity of the joining process.

The load-shortening curves generated from some of the panels are shown in figure 6. Load is plotted against panel shortening. The data presented are for boron/aluminum panels fabricated by diffusion bonding and spotwelding and for a Borsic/aluminum panel fabricated by brazing. The curves shown are linear up to a point where buckling is initiated. The strain at buckling is given in the figure for each load-shortening curve. At the indicated maximum load, failure occurs by instantaneous crippling of the stiffener and skin associated with a dropoff in load. The diffusion-bonded panel developed the highest load. This result is not too surprising since the flanges of the hat are continuously bonded to the skin along the hat-skin interface, and no material degradation was apparent during the fabrication process. A longitudinal fracture of the stiffener in the radius of the flange occurred at maximum load while the diffusion bond remained intact. The parameters for making the diffusion bond are 811 K (1000° F) for 15 minutes at 10.3 MPa (1500 psi) pressure.

The panel fabricated by spotwelding developed the next highest load. Failure of the specimen did not destroy the integrity of the spotwelds.

The lowest load obtained was for the Borsic/aluminum panel, where the hat was continuously attached to the skin by brazing. The lower load-carrying ability of the brazed panel is attributed to composite degradation resulting from brazing. The composite degradation is discussed in detail in the section on "Shear Strength." The braze joint remained intact throughout loading and after failure of the panel. Tearing of the hat in the radius of the attachment flange was the failure mode.

COMPOSITE CHARACTERIZATION

Fiber Strength

Brazing and diffusion bonding are processes requiring elevated temperature exposure which can lead to degradation in composite properties. In order to determine the effect of thermal processing on the strength of the boron and Borsic filaments, pieces of the composites were exposed to temperature for various periods of time and the filaments were removed for testing by etching away the aluminum matrix in a sodium hydroxide solution. The strengths of the fibers were determined by using the test fixture shown in figure 7. This simple test fixture has mandrels with radii ranging from 12.7 mm (0.50 in.) at the top to 6.3 mm (0.25 in.) at the bottom. The stress values listed to the right of each mandrel correspond to the outer fiber tensile stress of a 0.15-mm (0.0057-in.) fiber that is wrapped around the mandrel. The strength of the filament was taken to be the stress associated with the smallest mandrel that the filament could be wrapped around without breaking.

Results of the fiber tests are shown in figure 8. The maximum fiber strength is plotted for several exposure conditions. Data are shown for boron fibers exposed at temperatures of 810, 820, 830, and 840 K (1000°, 1020°, 1040°, and 1060° F) and for Borsic fibers at 865 K (1100° F). Each point shown represents the average of approximately 40 fiber tests. After exposures of boron filaments for 4 to 8 minutes at 830 or 840 K (1040° or 1060° F), up to a 30-percent loss in fiber strength occurred. Brazing of metal-matrix composites using conventional braze alloys normally requires exposure to approximately 830 K (1040° F) for a period of at least 10 minutes; thus serious boron fiber degradation may be expected during brazing. However, the data for the Borsic fibers at 865 K (1100° F) show that for exposures up to 32 minutes there was no fiber degradation. These data clearly show the advantages offered by Borsic/aluminum as compared with boron/aluminum where brazing is employed for fabrication.

Shear Strength

In an attempt to establish design shear allowables for the current in-house YF-12 wing-panel program and to determine the degradation of Borsic/aluminum associated with brazing, Borsic/aluminum specimens having the configuration shown in figure 9 were tested. Two face sheets of 4-ply Borsic/aluminum with a -45°, +45°, +45°, and -45°

fiber orientation were joined to a titanium honeycomb core. Tests were conducted on both "as-received" and brazed Borsic/aluminum test specimens. The as-received specimens were those whose face sheets were attached to the titanium honeycomb core by the adhesive system Hysol 9602 which cures at 395 K (250° F) for 1 hour. 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 arrows in figure 9 also show the orientation of the fibers in the face sheets. Failure of the specimens occurred by fracture of both face sheets perpendicular to the applied tensile loads. (See fig. 9.) Doublers were bonded to the two corners of the specimen perpendicular to the tension direction to reduce stress concentration.

The test results obtained are shown in figure 10, in which shear stress is plotted against shear strain to failure. The as-received panel developed the highest stress of 503 MPa (73 ksi). The brazed panel which was brazed at 856 to 861 K (1080° to 1090° F) developed a stress of 338 MPa (49 ksi).

The difference of 33 percent in the ultimate shear strengths of the two panels again shows evidence of composite degradation as a result of brazing, as was noted for the hat-stiffened panels. These results contradict the earlier results obtained on the strength of fibers removed from material which was exposed to a similar thermal environment. One possible explanation for the reduced shear strength of the brazed Borsic/aluminum, assuming no filament degradation, is that the 718 aluminum braze alloy is diffusing into the 6061 aluminum matrix. Such diffusion might cause either a change in the metallurgical nature of the matrix or in the bond between the 6061 aluminum matrix and the Borsic fiber or a combination of the two. In order to circumvent this problem, Borsic/aluminum face sheets were purchased which have an 1100 aluminum alloy as an outer 0.13-mm (0.005-in.) layer to serve as a diffusion barrier between the braze and the matrix. The 1100 aluminum alloy has a 56 K (100° F) higher melting point than the aluminum matrix of the composite and therefore should retard diffusion of the braze alloy into the composite. The test results for a brazed shear specimen employing the 1100 aluminum alloy as a barrier are also shown in figure 10. Data are normalized to a standard fiber volume of 48 percent. The brazed panel with the barrier developed a stress of 414 MPa (60 ksi) or an increase of 20 percent over the brazed panel with no barrier. Hence the use of the 1100 aluminum alloy apparently retarded matrix degradation by preventing the braze from reacting with the matrix and fibers.

YF-12 FLIGHT PANELS

In order to achieve the expertise needed to fabricate full-scale flight hardware, a manufacturing research program is being conducted at the Langley Research Center to develop processes that will be used to fabricate wing panels for YF-12 supersonic aircraft. Figure 11 shows the location of the panel on the YF-12 aircraft. The width and length of the panel, respectively, are 406 mm and 711 mm (16 in. and 28 in.). Holes are drilled around the edges, and mechanical fasteners are used to attach the panel to the airplane. The top and bottom face sheets of the panel consist of Borsic/aluminum as shown. The top skin consists of 4 plies, $\pm 45^{\circ}$ fiber orientation; the bottom face sheet consists of 3 plies, $\pm 45^{\circ}$ fiber orientation. Both the upper and lower skins contain the 1100 aluminum diffusion barrier. The frame is Ti-6Al-4V titanium alloy and is continuous around the perimeter of the panel. The core is Ti-3Al-2.5V alloy with a cell wall thickness of 0.05 mm (0.002 in.) and a cell size of 6.3 mm (0.25 in.). All mating surfaces are brazed with 718 aluminum braze alloy.

The fixture for brazing the panel is shown in figure 12. For brazing convenience the panel is turned upside down. The panel is sandwiched between fiberfrax and a titanium caul sheet. Titanium honeycomb core tooling is used around the edges. Above the caul sheet is a formed 0.1-mm-thick (0.005-in.) stainless-steel bladder that can be pressurized to control the pressure on the panel components during brazing to insure proper fit. During the brazing operation the entire fixture is placed in a vacuum furnace and the chamber pressure is reduced to 133 μPa (1×10^{-6} mm Hg). The bladder is pressurized to exert 34.5 kPa (5 psi) pressure over the panel being brazed. The furnace is then heated to a temperature of 856 to 861 K (1080° to 1090° F) and the temperature is held constant for 10 minutes. The power is cut off and argon gas is introduced into the furnace to lower the furnace temperature below 811 K (1000° F) quickly.

The advantages of this process are as follows: (1) only one braze cycle is required to braze the entire panel; (2) the bladder applies a uniform pressure to the panel during brazing; (3) metal-to-metal contact is obtained during brazing without the use of expensive matched tooling; and (4) deadweight loading, which would add to the thermal mass, is eliminated.

The full-scale process verification panel shown in figure 13 was brazed by using the process previously discussed. A cross section taken through the panel revealed sound braze joints with generous fillets at the honeycomb core-skin interface. In addition, nondestructive evaluation tests and a shear test of a portion of the panel indicated that the brazing operation had been performed successfully.

CONCLUDING REMARKS

Although metal-matrix composites can be fabricated and joined by conventional processes, there are advantages and disadvantages for each process. Results indicate that the brazing of Borsic/aluminum can be accomplished quite readily at temperatures up to 856 to 861 K (1080° to 1090° F) without fiber degradation. The use of a diffusion barrier such as 1100 aluminum alloy retards matrix degradation by preventing the braze from reacting with the matrix and fibers. Serious boron fiber degradation may be expected during brazing at temperatures above 830 K (1040° F).

Spotwelding appears to be a viable joining process which results in moderate-strength joints with use of a minimum amount of tooling. However, its use appears to be limited to a relatively low temperature because of the poor stress-rupture properties obtained. Diffusion-bonded joints exhibit high strength with no fiber degradation.

The adhesively bonded joints using the LRC-3A polyimide adhesive, presently in the developmental stage, show moderate strength, and the joints are made with a minimum of tooling. However, the data indicate a 30-percent reduction in strength at 533 K (500° F) when compared with that at room temperature.

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August 27, 1975

TABLE I. - DESCRIPTION OF METAL-MATRIX, TITANIUM, AND BRAZE SHEET MATERIALS

Material	Sheet thickness		Number of plies	Heat treatment
	mm	in.		
Borsic/aluminum (B_{SC}/Al) (6061 Al matrix)	1.67	0.066	9 unidirectional	--------
Boron/aluminum (B/Al) (6061 Al matrix)	1.67	.066	9 unidirectional	--------
Ti-6Al-4V (Ti)	1.34	.053	--------------	Annealed
718 aluminum alloy braze	.13	.005	--------------	Annealed

TABLE II. - TENSILE-SHEAR DATA

Joining method	Load at –					
	Ambient temperature		478 K	400° F	589 K	600° F
	kN	lbf	kN	lbf	kN	lbf
Spotwelded (single overlap)	15.591 15.191 13.967	3505 3415 3140	16.458 12.744 14.523	3700 2865 3265	8.096 11.938 13.011	1820 2695 2925
Brazed (double overlap)	24.629 26.700 25.378	5984 6000 5703	22.517 18.158 23.452	5060 4080 5270	13.439 14.151 12.193	3020 3180 2740

Joining method	Load at –					
	Ambient temperature		450 K	350° F	533 K	500° F
	kN	lbf	kN	lbf	kN	lbf
Adhesive bonded (single overlap)	25.476 27.501 27.634	5725 6180 6210	24.653 25.632 24.475	5540 5760 5500	22.241 15.842 -----	4998 3560 ----

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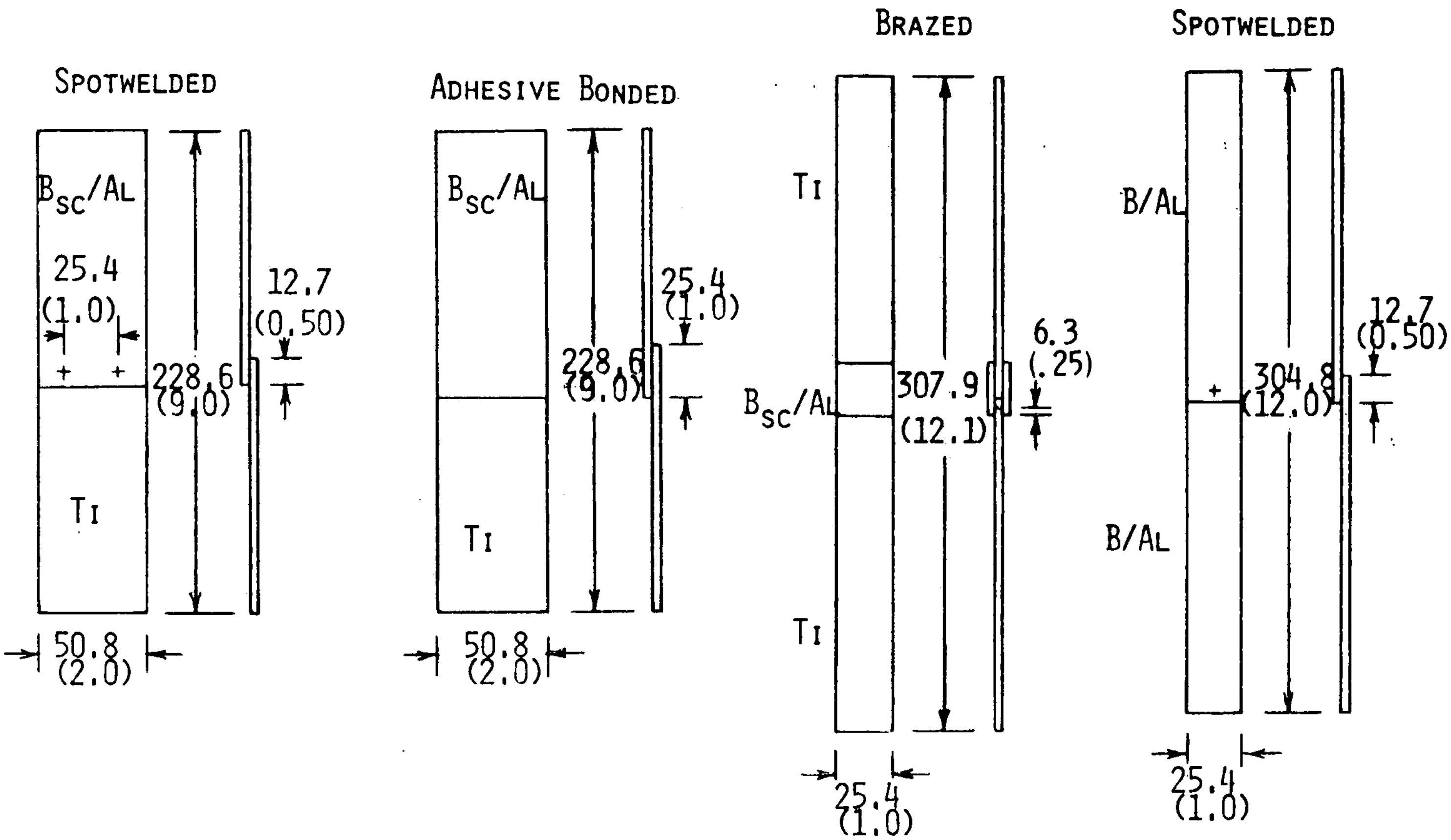
Figure 1.- Panel location for flight service on YF-12.



(a) Single overlap. (b) Double overlap. (c) Stress rupture.

Figure 2.- Joint characterization test specimens. Dimensions are given in millimeters (inches).

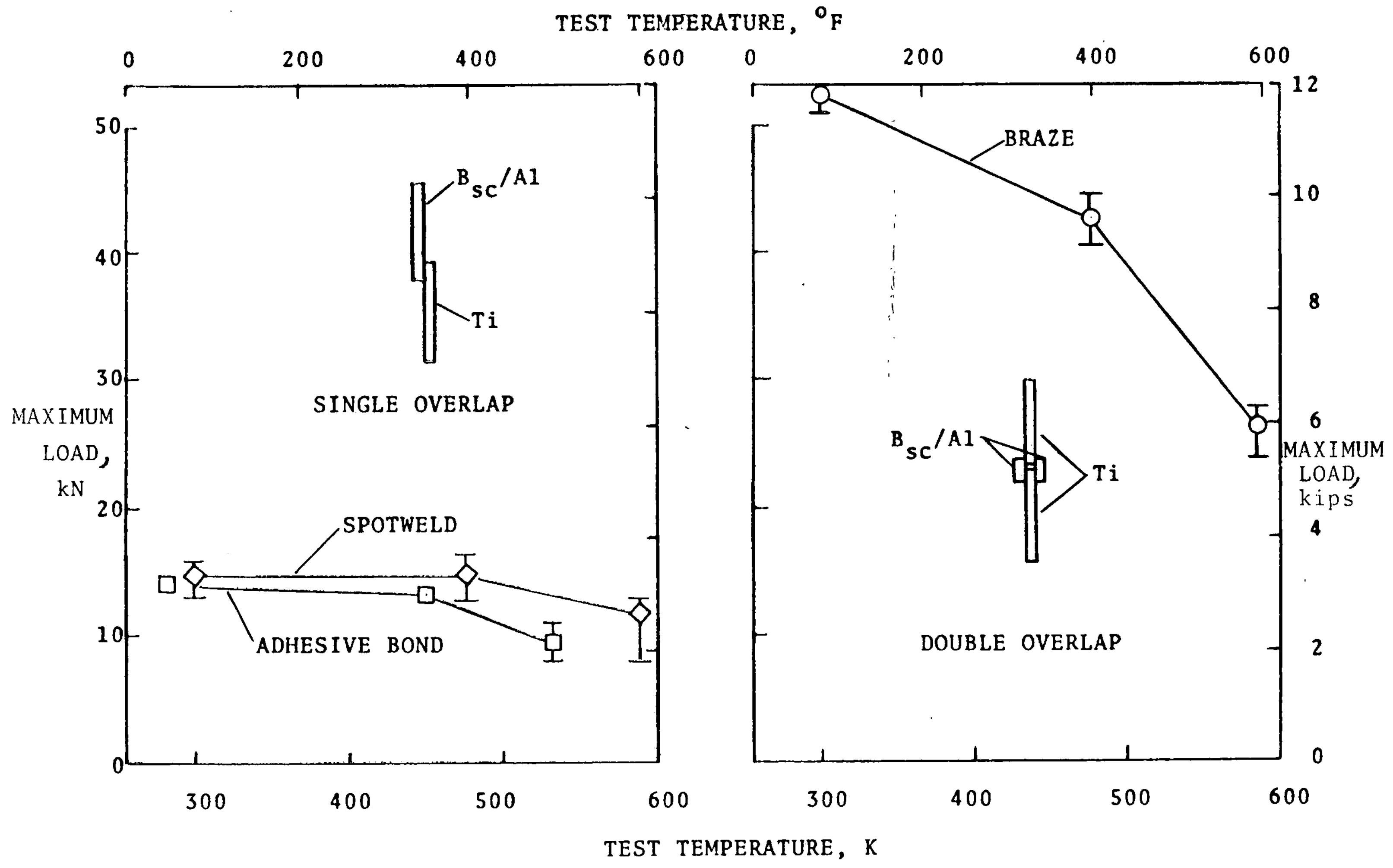


Figure 3.- Tensile-shear strength of Borsic/aluminum titanium joints.

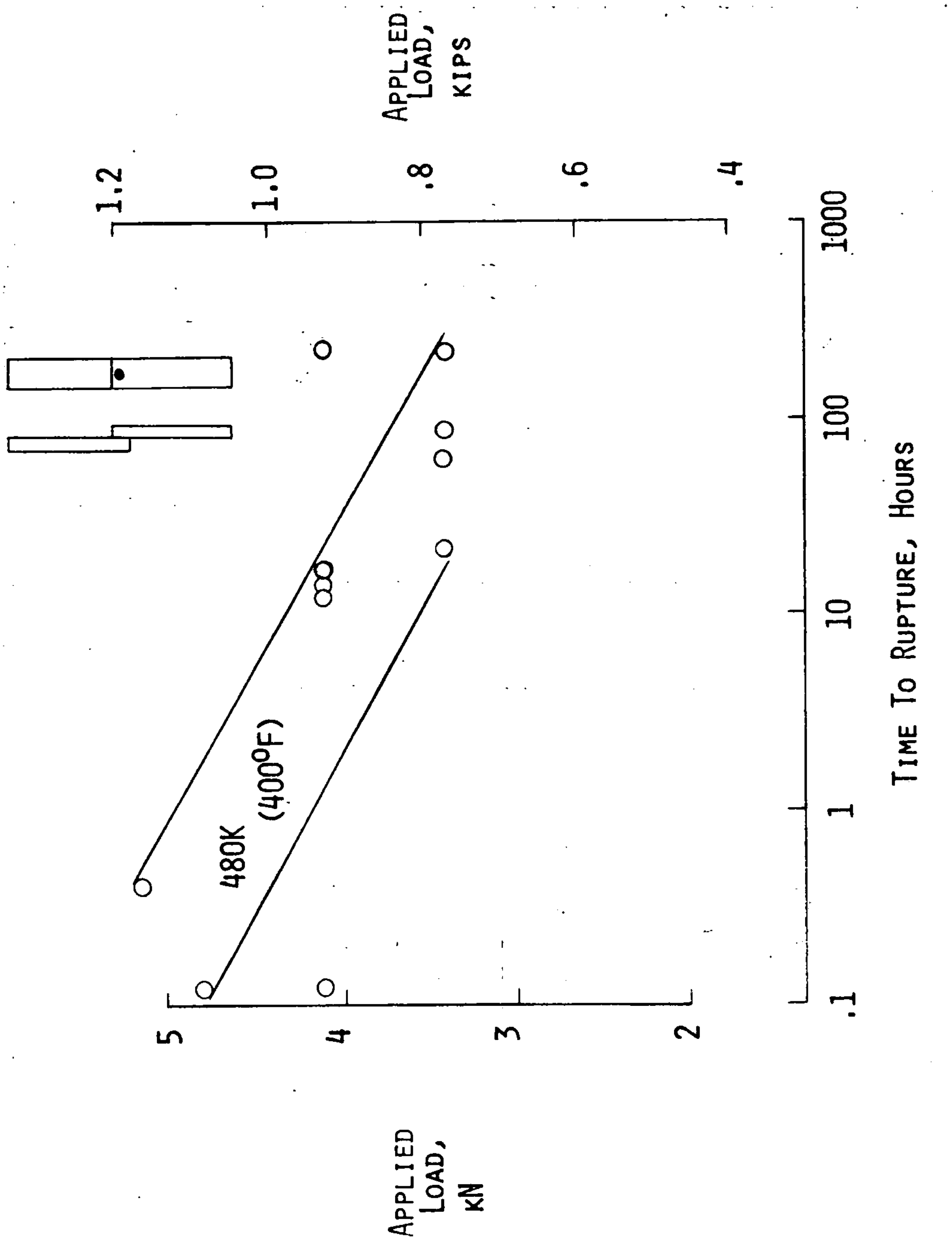


Figure 4.- Stress-rupture properties of spotwelded boron-aluminum specimens.

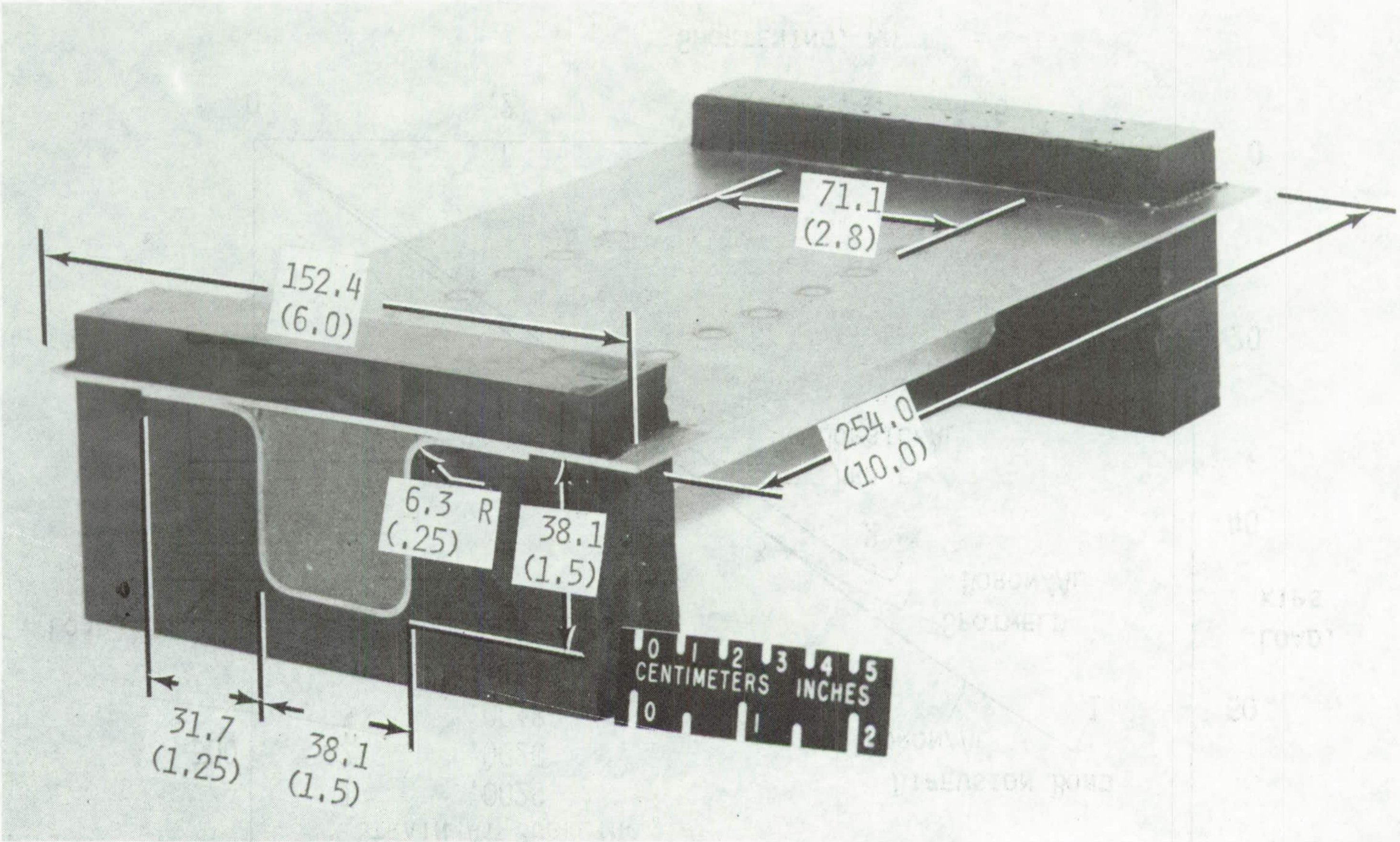


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Figure 5.- Skin-stringer panel. Dimensions are given in millimeters (inches).

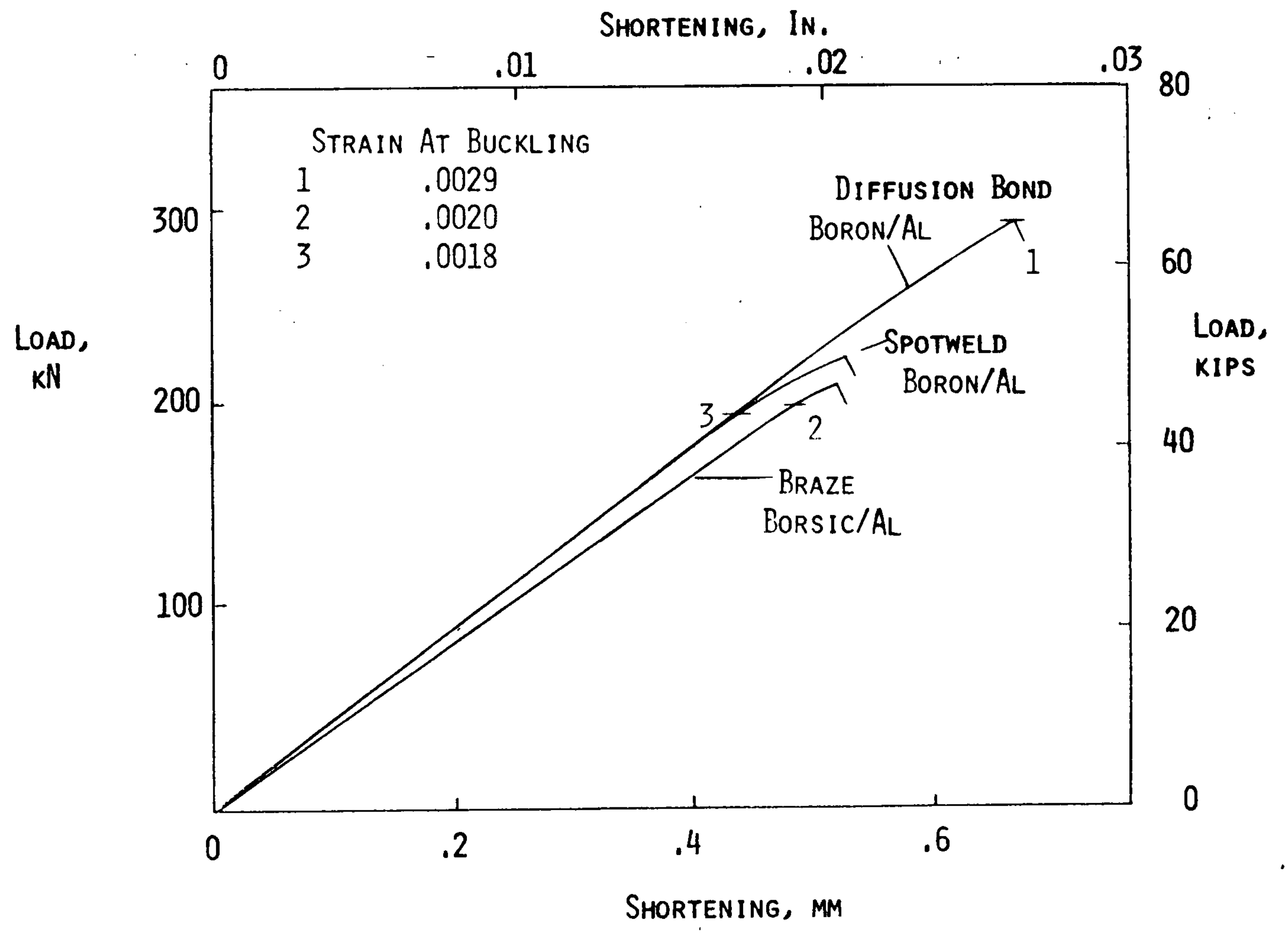

Figure 6.- Hat-stringer panel load shortening data.

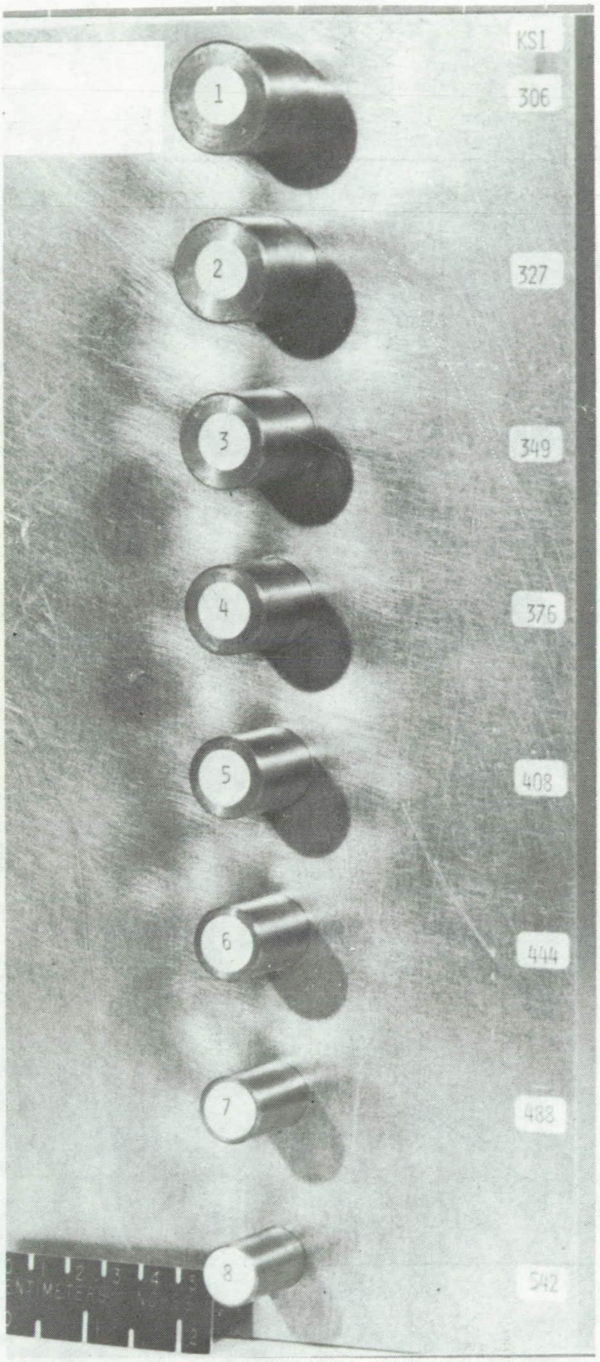

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Figure 7.- Fiber bend test fixture.

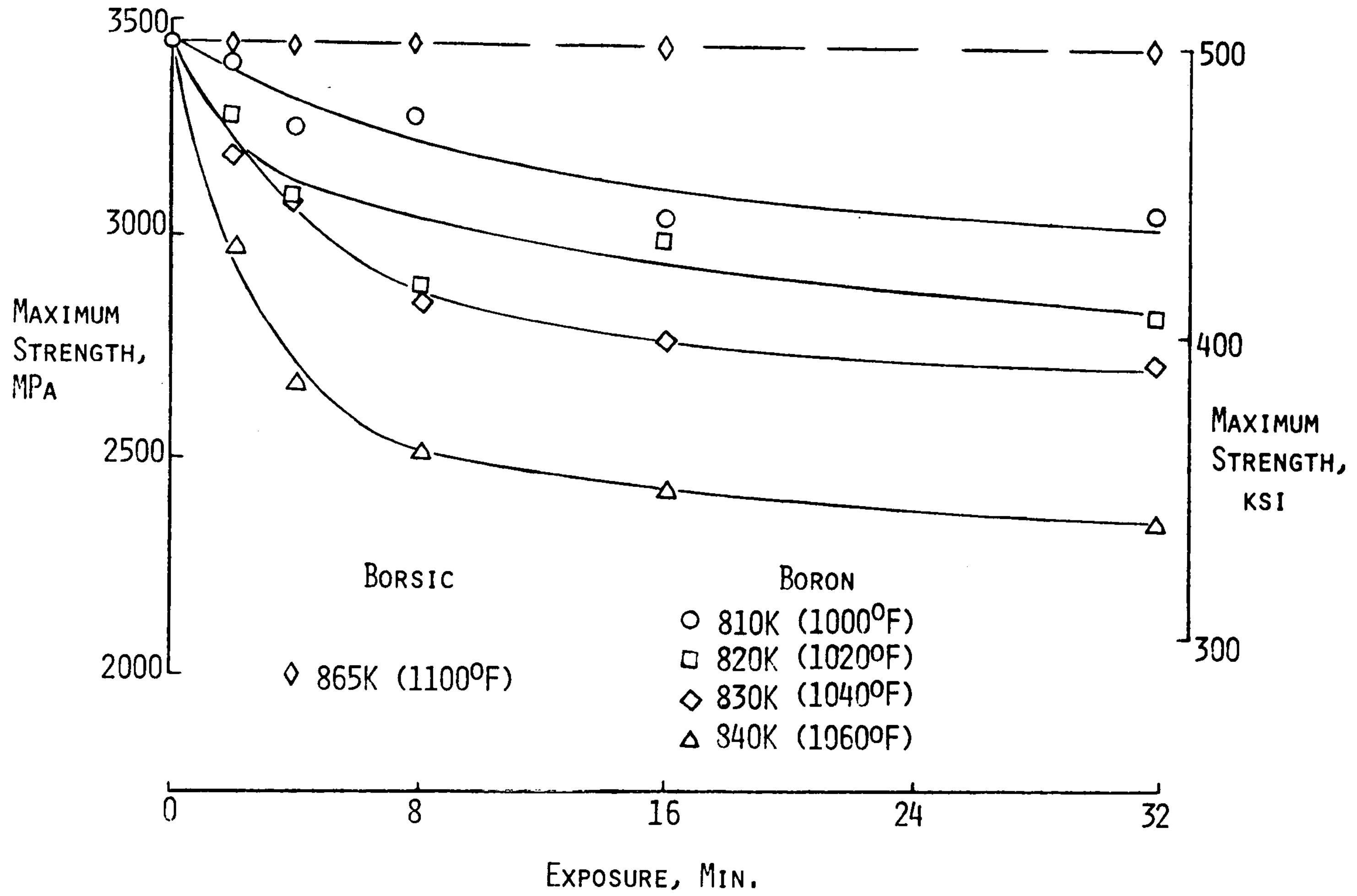

Figure 8.- Room-temperature fiber bend strength.

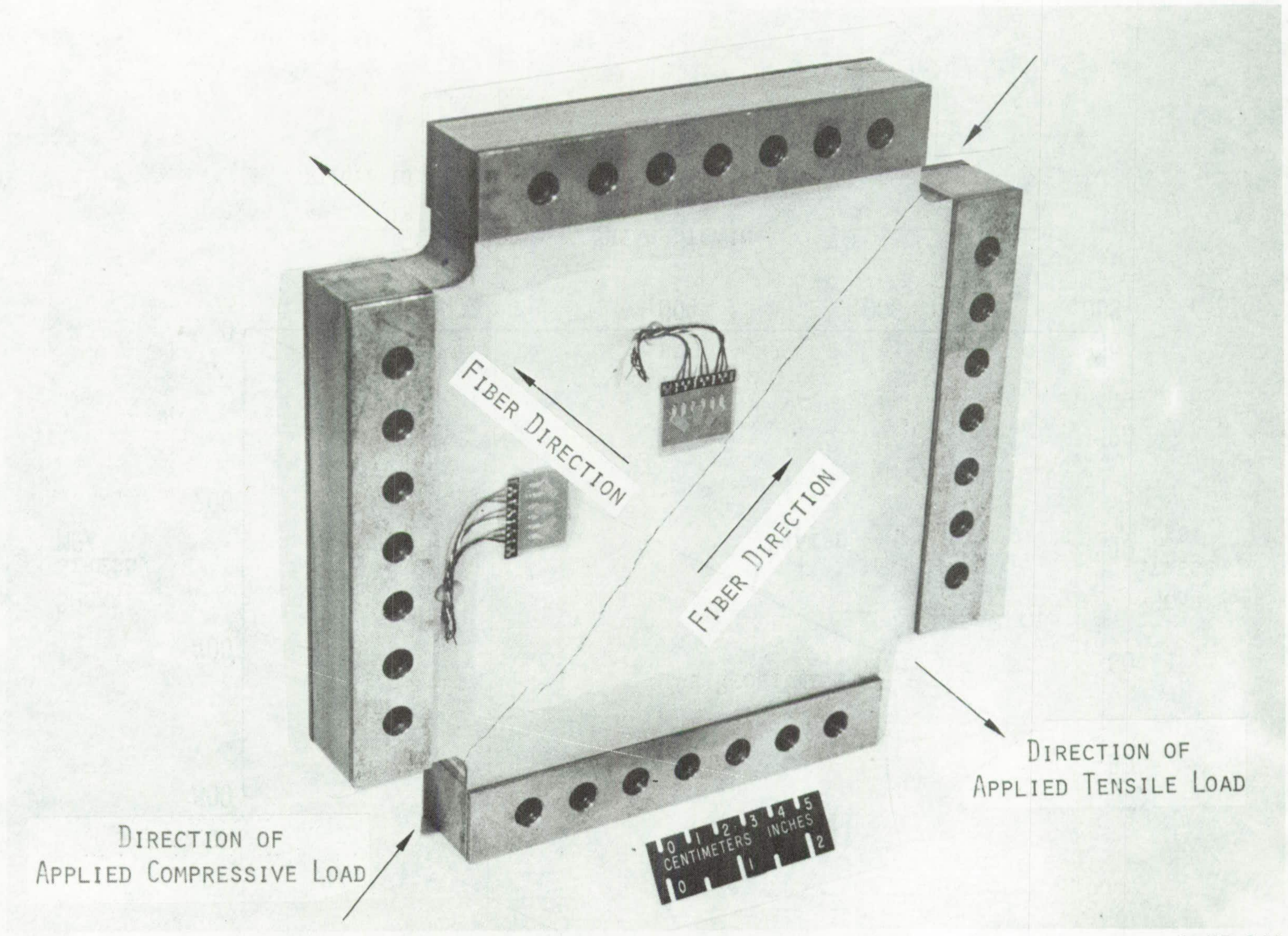

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Figure 9.- Honeycomb core shear specimen.

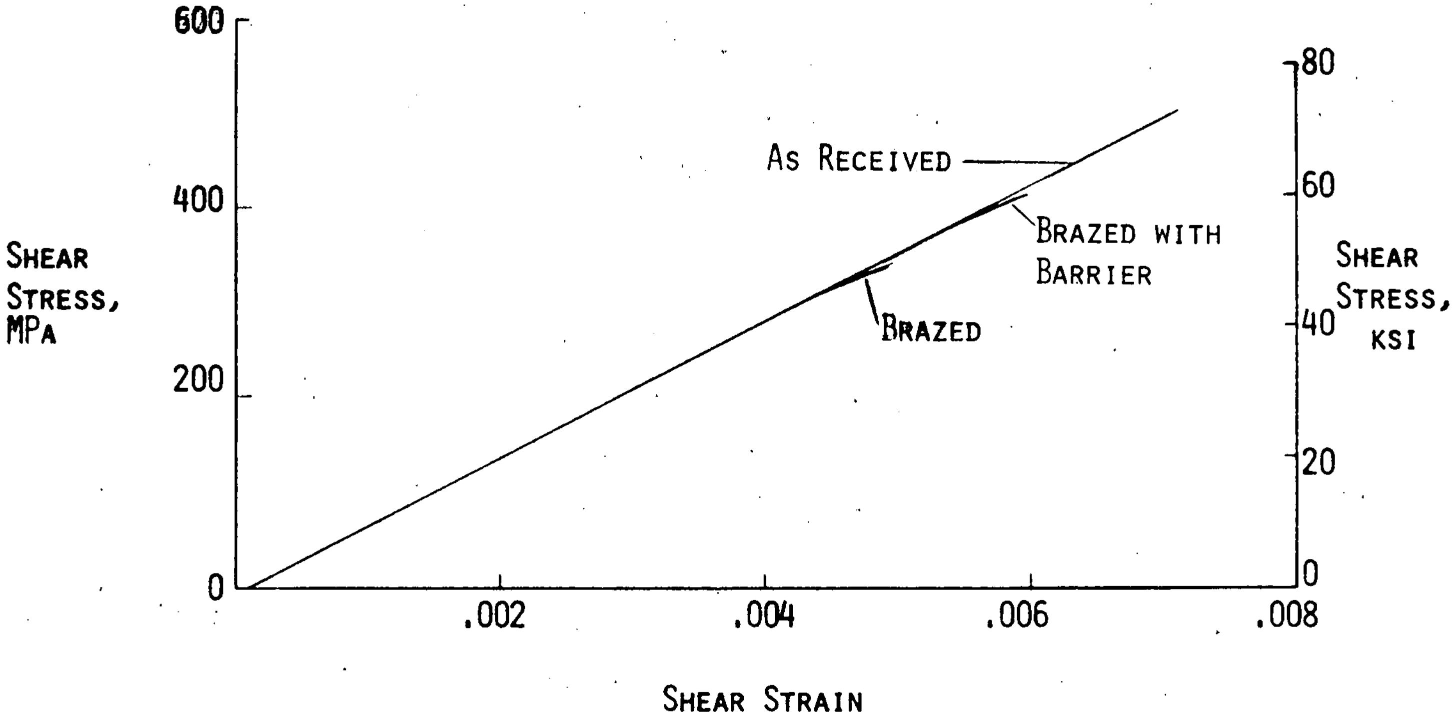

Figure 10.- Shear stress-strain data for Borsic/aluminum.

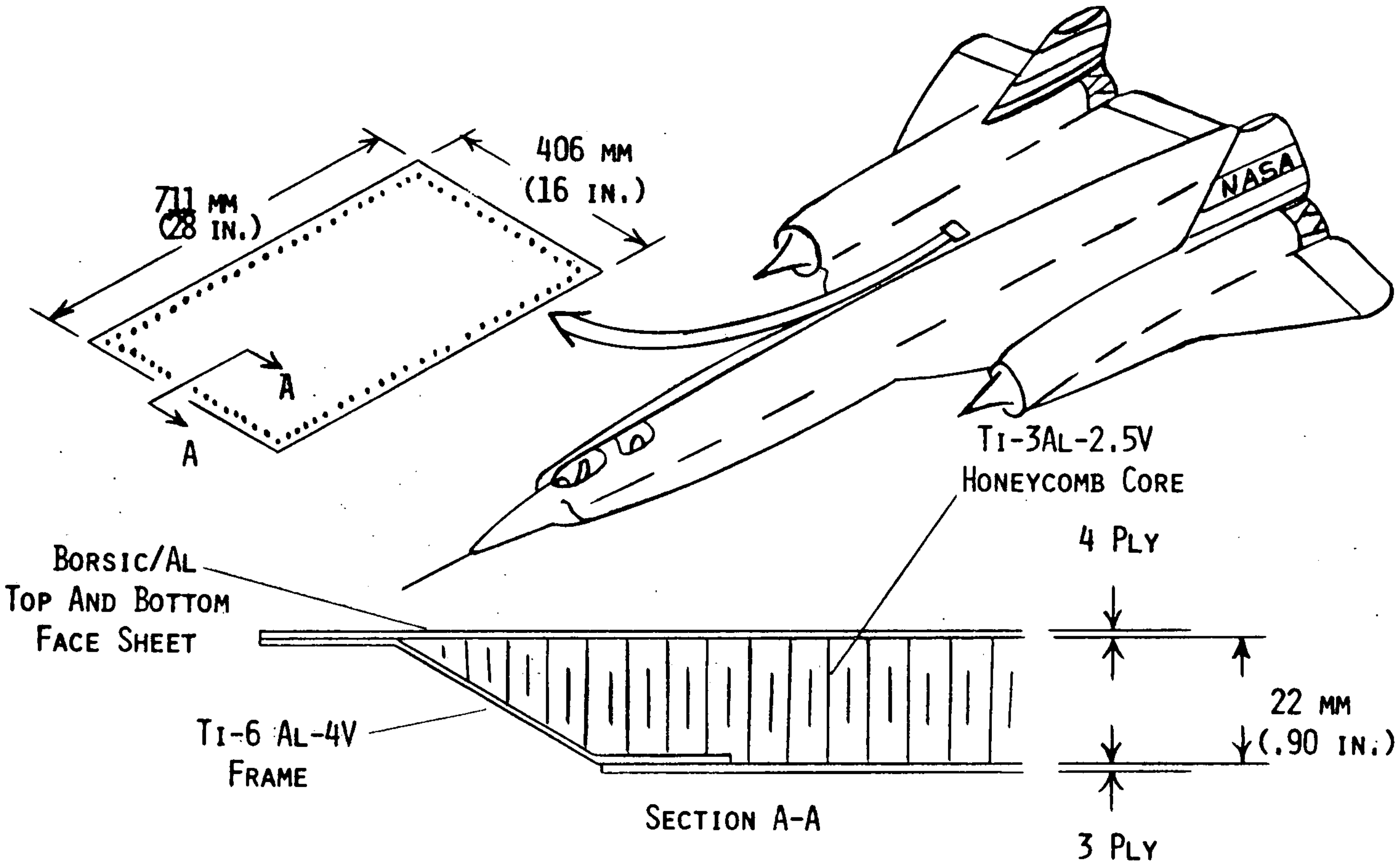

Figure 11.- Borsic/aluminum-titanium YF-12 panel design.

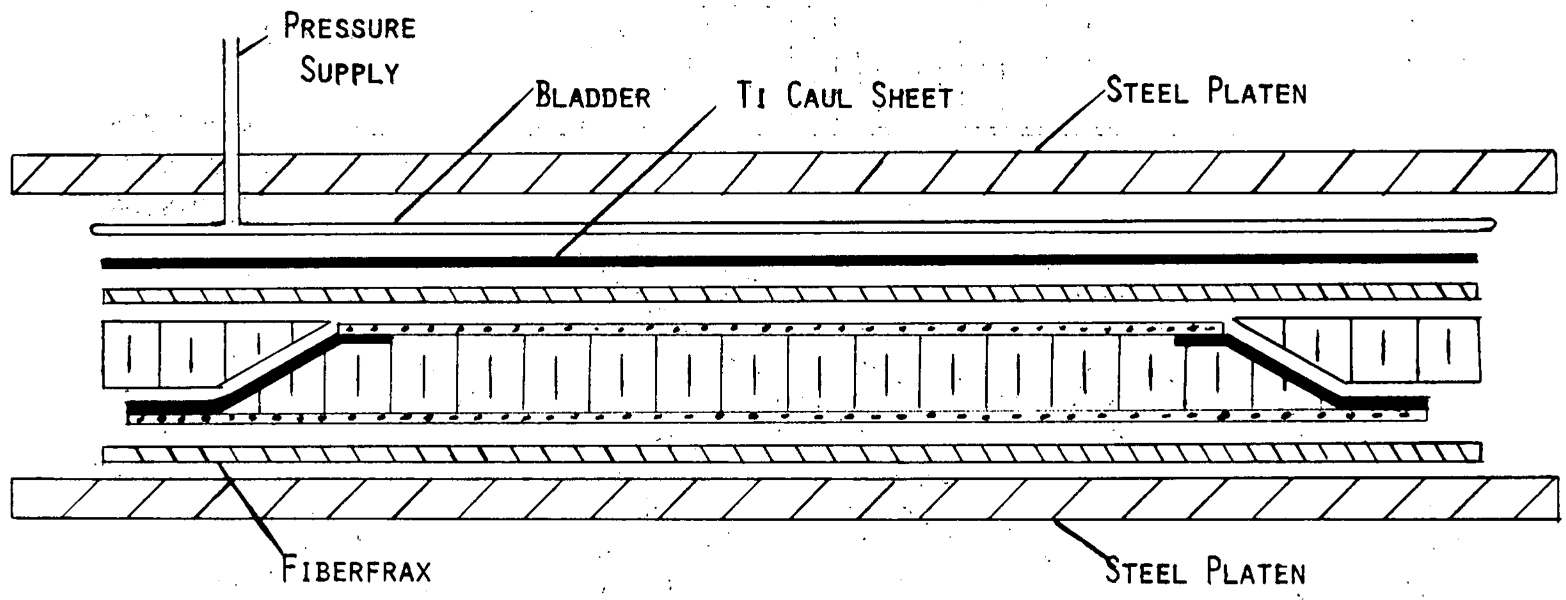

Figure 12.- Panel brazing fixture.

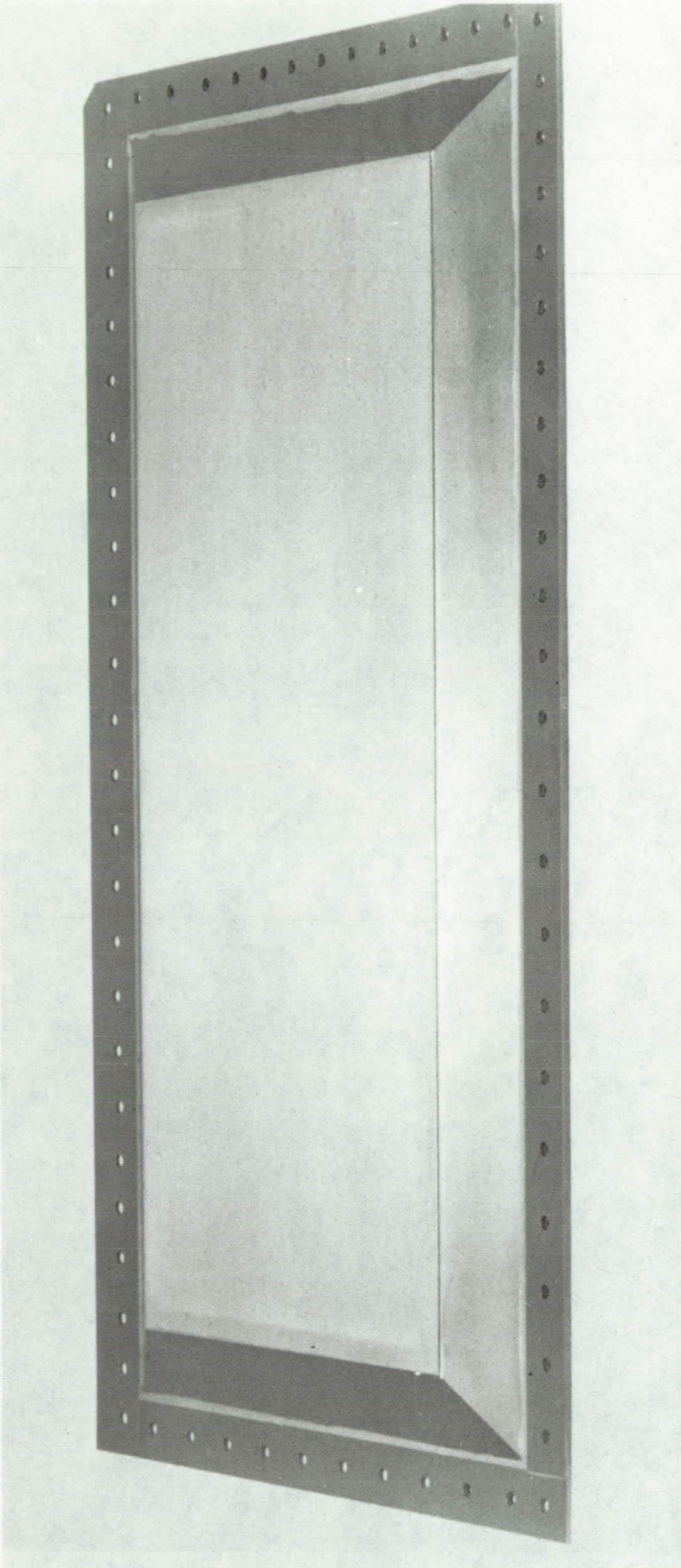

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Figure 13.- YF-12 Borsic/aluminum panel.

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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