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DESIGN AND FABRICATION OF AN AEROELASTIC FLAP ELEMENT FOR A SHORT TAKEOFF AND LANDING (STOL) AIRCRAFT MODEL

Final Report

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Prepared under contract NAS1-11767 by
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The work was monitored by Mr. Marvin D. Rhodes of the Structures Division of NASA Langley Research Center.

16. Abstract

A flap element typifying a third element in the flap system of a short takeoff and landing aircraft was designed, fabricated, and instrumented. It was delivered to NASA for flight-simulated testing. The flap element was aluminum skin-stringer-rib construction with adhesive laminated skins.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
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DESIGN AND FABRICATION OF AN AEROELASTIC FLAP ELEMENT FOR A SHORT TAKEOFF AND LANDING (STOL) AIRCRAFT MODEL

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SUMMARY

A flap element typifying a third flap element of a short takeoff and landing (STOL) airplane was designed and fabricated. The purpose of this program was to provide NASA with a representative aircraft part which could be evaluated by flight-simulated tests. The tests are expected to provide an insight to the loading and dynamic response of an airfoil section subjected to a flight airstream and jet engine exhaust impingement.

INTRODUCTION

The externally blown flap concept being considered in preliminary designs of a short takeoff and landing aircraft uses special high-lift devices to decrease runway requirements. One element which is critical to the successful operation of this concept is the trailing-edge flap. The location of this element exposes it to aerodynamic loads, elevated temperatures, jet engine exhaust impingement, boundary layer turbulence, and vortex shedding.

The object of this program is to design and fabricate a representative third flap element and provide it to NASA for flight-simulated testing. The tests will allow an assessment of the flap response to the flight environment.

SYMBOLS

a	panel length, mm (in.)
b	panel width, mm (in.)
B	bar width, mm (in.)
C	distance from neutral axis to outer fiber, mm (in.)
D	bar thickness, mm (in.)
E_T	Modulus of elasticity, tension, GPa, (psi)
E_C	Modulus of elasticity, compression, GPa (psi)
F_c	stringer stress, MPa (psi)
F_{tu}	allowable tensile ultimate stress, MPa (psi)
F_{ty}	allowable tensile yield stress, MPa (psi)
F_{cy}	allowable compression yield stress, MPa (psi)
F_{bru}	allowable bearing ultimate stress, MPa (psi)
G	modulus of rigidity, GPa (psi)
I	moment of inertia, μM^4 (in.^{4})
J	polar moment of inertia, μM^4 (in.^{4})
K_2	torsional shape constant
L	bar length, mm (in.)
M	bending moment, N·m (in.-lbf)
t	panel thickness, mm (in.)

ΔT	temperature differential, °K (°F)
W_e	effective skin width, mm (in.)
α	coefficient of thermal expansion
δ	deflection, mm (in.)
ϵ	expansion-contraction, mm (in.)
θ	angular rotation, rad

DESIGN CONCEPT

The overall arrangement and shape of the third-element flap was provided by NASA. This arrangement is shown in figure 1. The design loads and airfoil coordinates were also provided by NASA. The basic design considerations were the simultaneous application of a 36-m/sec (70 kn) airstream and exhaust impingement of a General Electric TF34 turbofan engine. The design life is to be 1000 hr, an environment of 155 dB noise level (table 1) and temperatures to 422° K (300° F) (table 2). It was further specified that the end attachment sections have a bending stiffness of approximately 3.18 mm/4.45 kN (0.125 in./1000 lbf) applied cantilever load and 3.29 rad/MN·m (0.327×10^{-6} rad/in.-lbf) torsional stiffness. This was accomplished by introducing the flap-end reaction loads into a solid aluminum bar 50.8 mm x 203 mm (2 in. x 8 in.), approximately 670 mm (26.4 in.) long (table 3). These end attachment sections were designed but not fabricated. They will be fabricated by NASA.

The end bars were designed with a nonstructural airfoil-shaped "boilerplate" shroud to maintain airfoil continuity. Schematics of the arrangement are shown in figures 2, 3, and 4.

The entire flap, except for the attachment fitting, was constructed with 2024-T3 aluminum alloy. This alloy was selected for the following reasons: (1) the flap was basically stiffness designed; (2) the alloy retains 95% of its modulus at 422°K (300°F); (3) it retains a high percentage of its tensile ultimate and compression yield after 1000 hr at (300°F); (4) it is a tough alloy with favorable ductility, and (5) it withstands sonic environments better than alloys such as 7075-T6. Material properties of 2024-T3 aluminum are listed in table 4.

The skins were laminated from three layers of 0.64 mm (0.025 in.) 2024-T3 sheet. The removable leading edge is a solid sheet aluminum 1.8 mm (0.071 in.).

The spars and ribs were built-up, riveted, formed aluminum, as were the leading-edge formers and trailing-edge riblets. The end ribs were machined out of aluminum plate with a profile mill. A simple aluminum sheet template was used for the airfoil profile.

The stringers are extruded aluminum. The basic structural concept is a conventional-type skin-stringer construction. The flap element evolved as a two-spar box construction with all skin effective in torsion and bending.

The laminated skins were autoclave bonded with Metlbond 329-7 adhesive. Manufacturer data on this material indicates a 90% strength retention after aging 1000 hr at 450°K (350°F).

All metal-to-metal contact surfaces were coated with Products Research and Chemical Corporation PR 1750-A-4. This material is a polysulphide composition that is used as a faying surface sealant. Boeing data indicates structures assembled this way are better in sonic environments to the extent of about 1 dB.

The total gross mass of the completed part was 45.5 kg (96 lb). The net mass minus instrumentation was 38.6 kg (85 lb).

The bottom skin of the flap element was instrumented with 22 axial strain gages and two rosettes. Eight accelerometers were positioned inside the flap.

An analysis of the element was conducted to assess the structural adequacy of the part. The aerodynamic loadings on the part were provided by NASA and result in the loads shown in figures 5, 6, and 7. Figures 8 to 11 show the physical properties of the flap element. Figures 12 and 13 show the Boeing design curves for sonic fatigue life that were used to check part details.

The fabrication sequence was straightforward. The skins were laminated and roll formed to contour. The ribs and spars were assembled in a simple jig tool. The upper skin was riveted on and the instrumented bottom skin was blind fastened with bulbed Cherrylock rivets. Lastly, the removable leading edge was installed with bolts into nutplates. Figures 14 to 24 show the subassembly and some details of the construction.

ANALYSIS

The design loads were provided by NASA. The flap is supported at each end by pinned connections and is therefore analyzed as a simply supported beam (tables 5 and 6). Various members were checked for their sonic fatigue capability (table 1), using Boeing-derived test data.

Boeing Commercial Airplane Company
P.O. Box 3707
Seattle, Washington 98124, December 14, 1973

TABLE 1.–SONIC CHECK

GEOMETRY

Rib spacing, a = 10 in.
Skin gage, t = 0.075 in.

Stringer spacing, b = 8 in. (maximum)
Spar gage, t = 0.063 in.

CALCULATIONS

Skin:

$$\frac{b}{t} = \frac{8}{0.075} = 107$$

$$\frac{a}{b} = 1.25$$

then, from figures 12 and 13,

$$162 - 1.7 = 160.3 \text{ dB} \longleftarrow$$

Spar:

$$\frac{b}{t} = \frac{4}{0.063} = 64$$

$$\frac{a}{b} = 2.5$$

then, from figures 12 and 13,

$$163 - 4.7 = 158.3 \text{ dB} \longleftarrow$$

Trailing-edge skin:

$$\frac{b}{t} = \frac{11}{0.063}$$

$$\frac{a}{b} = 1$$

then, from figures 12 and 13,

$$156 - 0 = 156 \text{ dB} \longleftarrow$$

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TABLE 2.—THERMAL EXPANSION CALCULATION

Assume third-element flap is fixed in expansion-contraction length by steel support structure.

Also assume, as worst case, that third element reaches 300°F uniform temperature; support structure remains at ambient 75°F. Then, maximum thermal expansion stress is

$$\begin{aligned}\sigma_e &= \alpha \Delta T E \\ &= (12 \times 10^{-6})(300 - 75)(10.1 \times 10^{6}) \\ &= 27{,}300 \text{ lbf/in.}^2\end{aligned}$$

Maximum bending stress is

$$\sigma_e = 7218 \text{ lbf/in.}^2$$

The total stress is then

$$27{,}300 + 7218 = 34{,}518 \text{ lbf/in.}^2 \text{ (237.97 MPa)}$$

Since yield stress is 35,000 lbf/in.^{2} (241.29 MPa), we have excessive thermal stress.

The expansion relief required to alleviate thermal stress is

$$\begin{aligned}\epsilon &= \alpha \Delta T L \\ &= (12 \times 10^{-6})(225)(80) \\ &= 0.216 \text{ in. (5.49 mm)}\end{aligned}$$

This relief is maximum due to conservative assumptions

NOTE: Calculations were performed in customary units and answers converted to SI units.

TABLE 3.—END FITTING STIFFNESS

Where desired end fitting stiffnesses are

Torsion: $\theta = 0.372 \times 10^{-6}$ rad/in.-lb

Bending: $\delta = 0.125$ in./1000 lb end load,

the length of a solid aluminum rectangle with assumed cross-sectional dimensions of 2 by 8 in. is calculated from *Strength of Materials* by F. R. Shanley, McGraw-Hill Book Company, Inc., p. 509, as follows:

$$\theta = \frac{K_2 ML}{BD^3 G}$$

where D/B = 0.25 and $K_2 = 3.6$.

Then, substituting desired torsion,

$$\theta = 0.372 \times 10^{-6} = \frac{(3.6)(1)(L)}{(8)(2)^3(4 \times 10^6)}$$

Solving for length,

L = 26.4 in. (670.6 mm)— length of a 2- by 8-in. (50.8- by 203- mm) aluminum bar required for torsional stiffness

Checking bending deflection for this length

$$\delta = \frac{PL^3}{3EI}$$

$$= \frac{(1000)(26.4)^3}{(3)(10.1 \times 10^6)\frac{(8)(2)^3}{12}}$$

= 0.114 in. (2.9 mm) vs desired 0.125 in. (3.18 mm)

NOTE: Calculations were performed in customary units and answers converted to SI units.

*TABLE 4.—MATERIAL PROPERTIES**

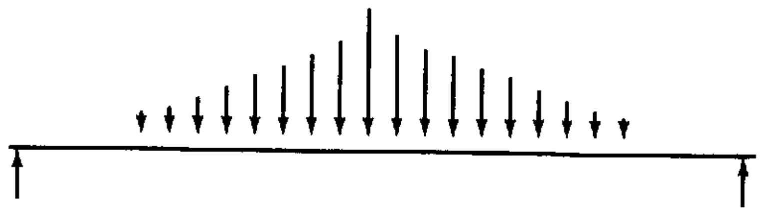
2024-T3 aluminum aged 1000 hr at 422° K (300° F) and tested at 422° K (300° F)

F_{ts}	=	333.71 MPa (48,400 psi)
F_{ty}	=	279.24 MPa (40,500 psi)
F_{cy}	=	241.32 MPa (35,000 psi)
F_{bru}	=	558.48 MPa (81,000 psi)
F_{bry}	=	434.37 MPa (63,000 psi)
E_t	=	69.64 GPa (10.1 x 10^6 psi)
E_c	=	71.02 GPa (10.3 x 10^6 psi)
G	=	26.48 GPa (3.84 x 10^6 psi)

*Reference: Boeing Design Manual, D-5000, volume 84A2

TABLE 5.–MAXIMUM BENDING STRESS IN SKINS

GEOMETRY



FLAP ELEMENT PROPERTIES

M_{max} = 75,465 in.-lbf at station 117 (see fig. 7)

I = 23 in.^{4} (see fig. 10)

c_{max} = 2.6 (tension side)

c_{max} = 2.2 (compression side)

CALCULATIONS

$$\sigma = \frac{Mc}{I}$$

$$= \frac{(75{,}465)(2.6)}{23}$$

$$= 8530 \text{ lbf/in.}^2 \text{ (58.81 MPa)(tension)}$$

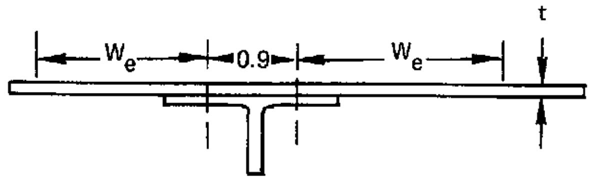
$$= \frac{(75{,}465)(2.2)}{23}$$

$$= 7218 \text{ lbf/in.}^2 \text{ (49.77 MPa)(compression)}$$

NOTE: Calculations were performed in customary units and answers converted to SI units

TABLE 6.—EFFECTIVE SKIN WIDTH AND STRINGER SPACING

GEOMETRY



CALCULATIONS

$$W_e = 0.85t\sqrt{\frac{E}{f_c}}$$

Where f_c = stringer stress,* and assuming that maximum compressive stress equals stringer stress, then (from table 5) f_c = 7218 lbf/in.^{2} and

$$W_e = (0.85)(0.075)\sqrt{\frac{10.3 \times 10^6}{7218}}$$

$$= 2.41 \text{ in.}$$

Total effective skin width = $2W_e + 0.9$

= (2)(2.41) + 0.9

= 5.72 in. (145.3 mm)

For all skin to be effective,

Stringer spacing $\leqslant$ 5.72 in. (145.3 mm)

Actual stringer spacing = 5 in. (127 mm)

Therefore, all skin between stringers is effective.

Note: Calculations were performed in customary units and answers were converted to SI units.

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*Reference *Boeing Stress Manual,* D6-22695, p. 11-11

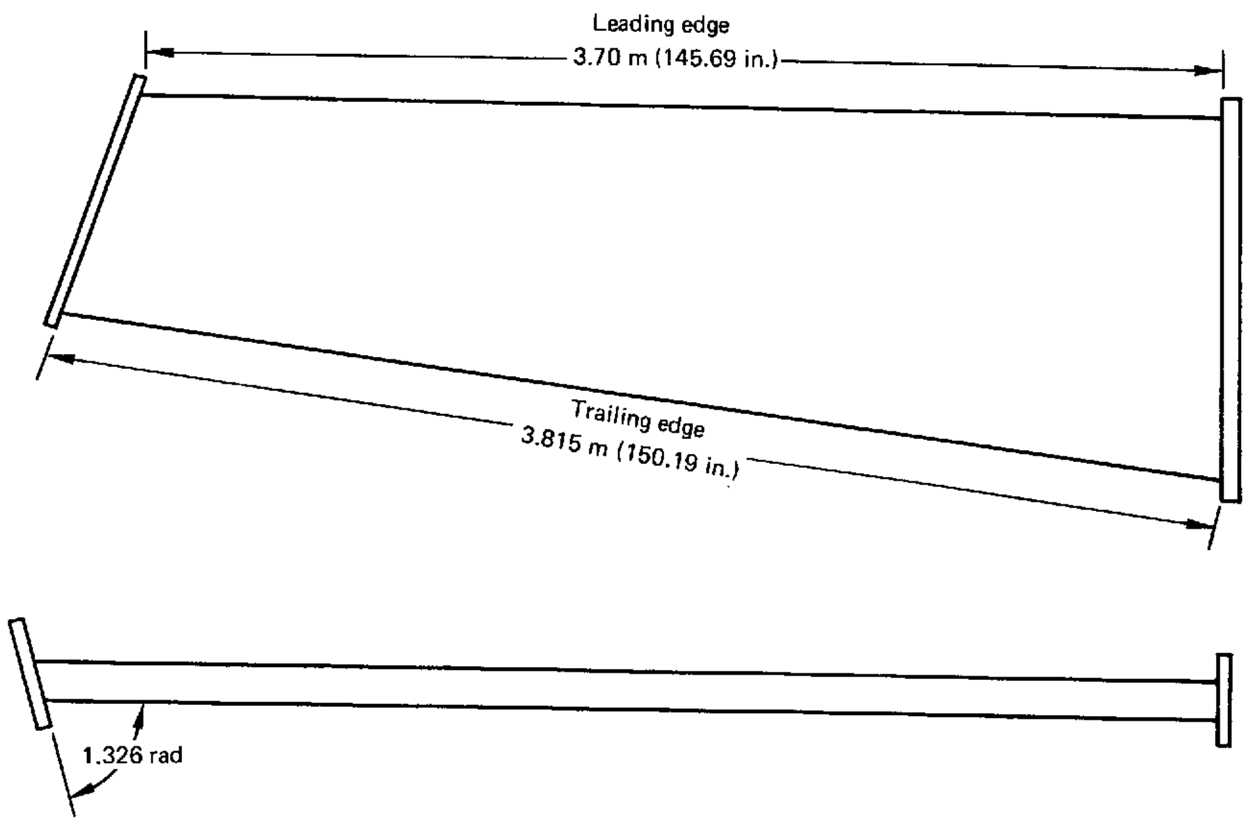

FIGURE 1.—DIMENSIONAL ENVELOPE

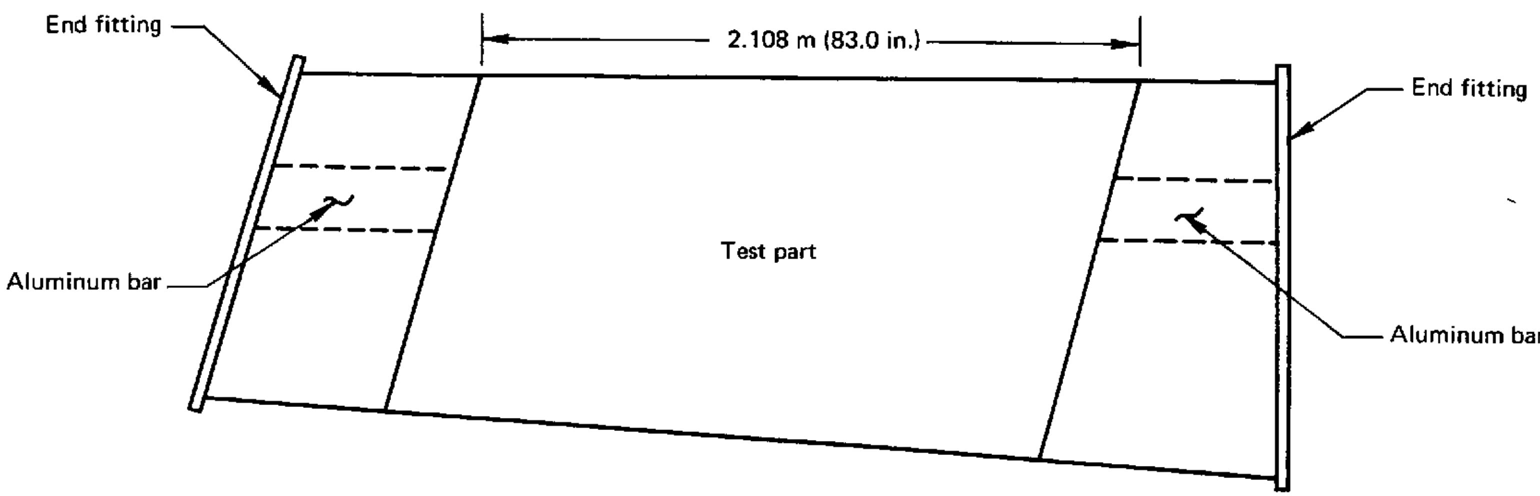

FIGURE 2.—GEOMETRICAL CONCEPT

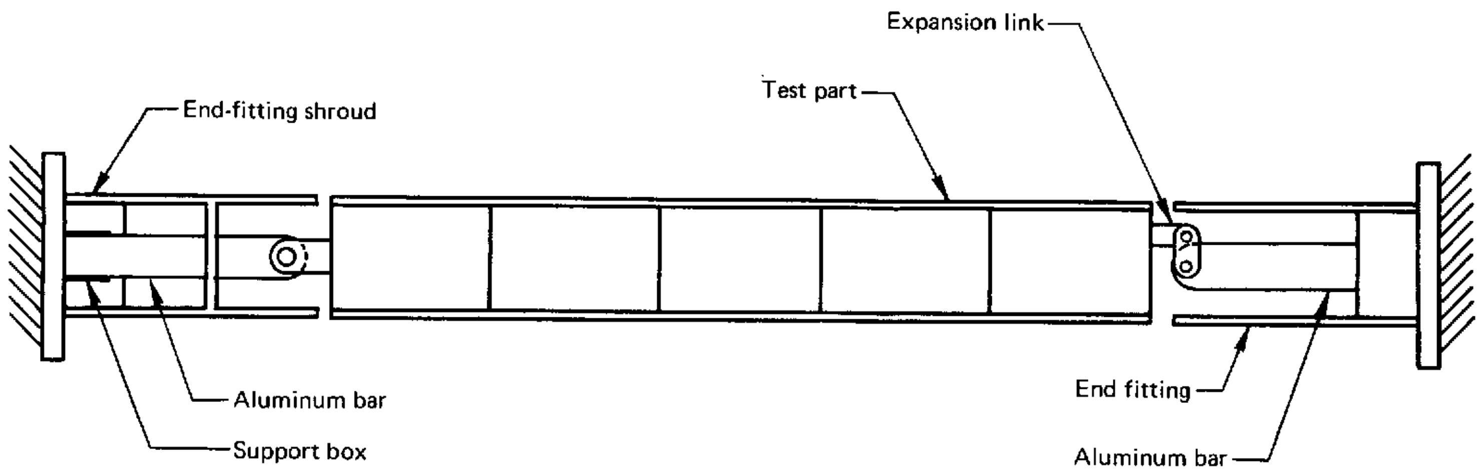

FIGURE 3.–ATTACHMENT SCHEMATIC

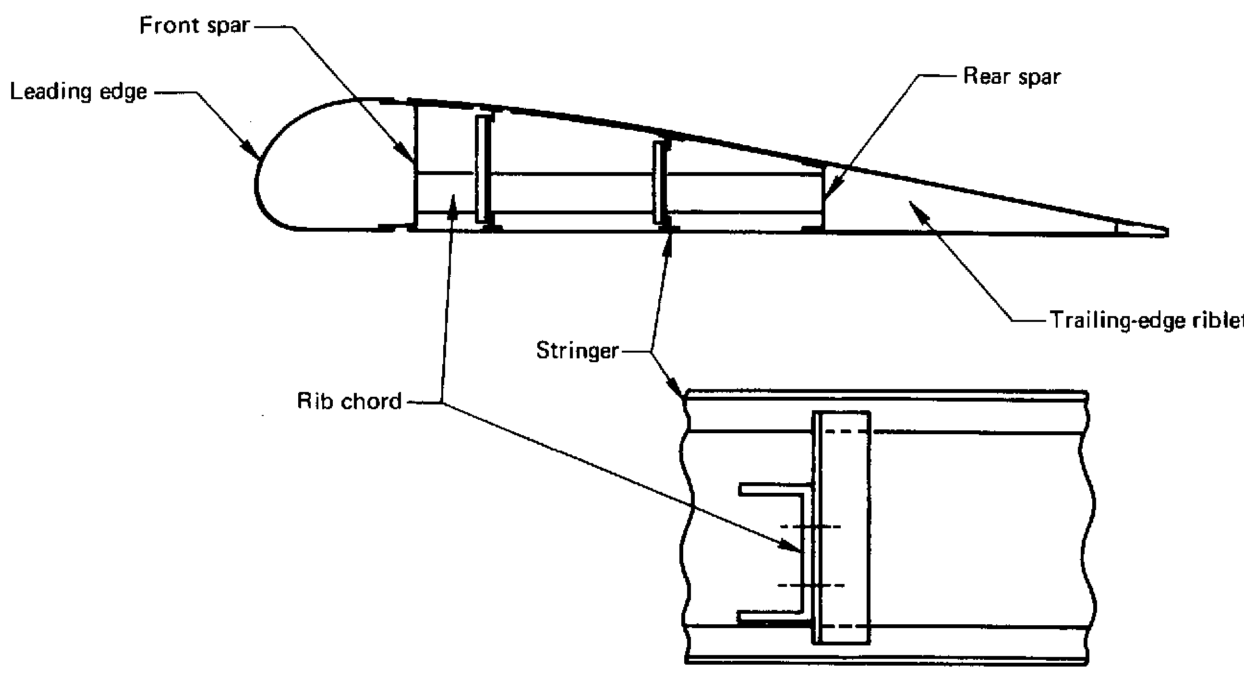

FIGURE 4.—SPANWISE ARRANGEMENT

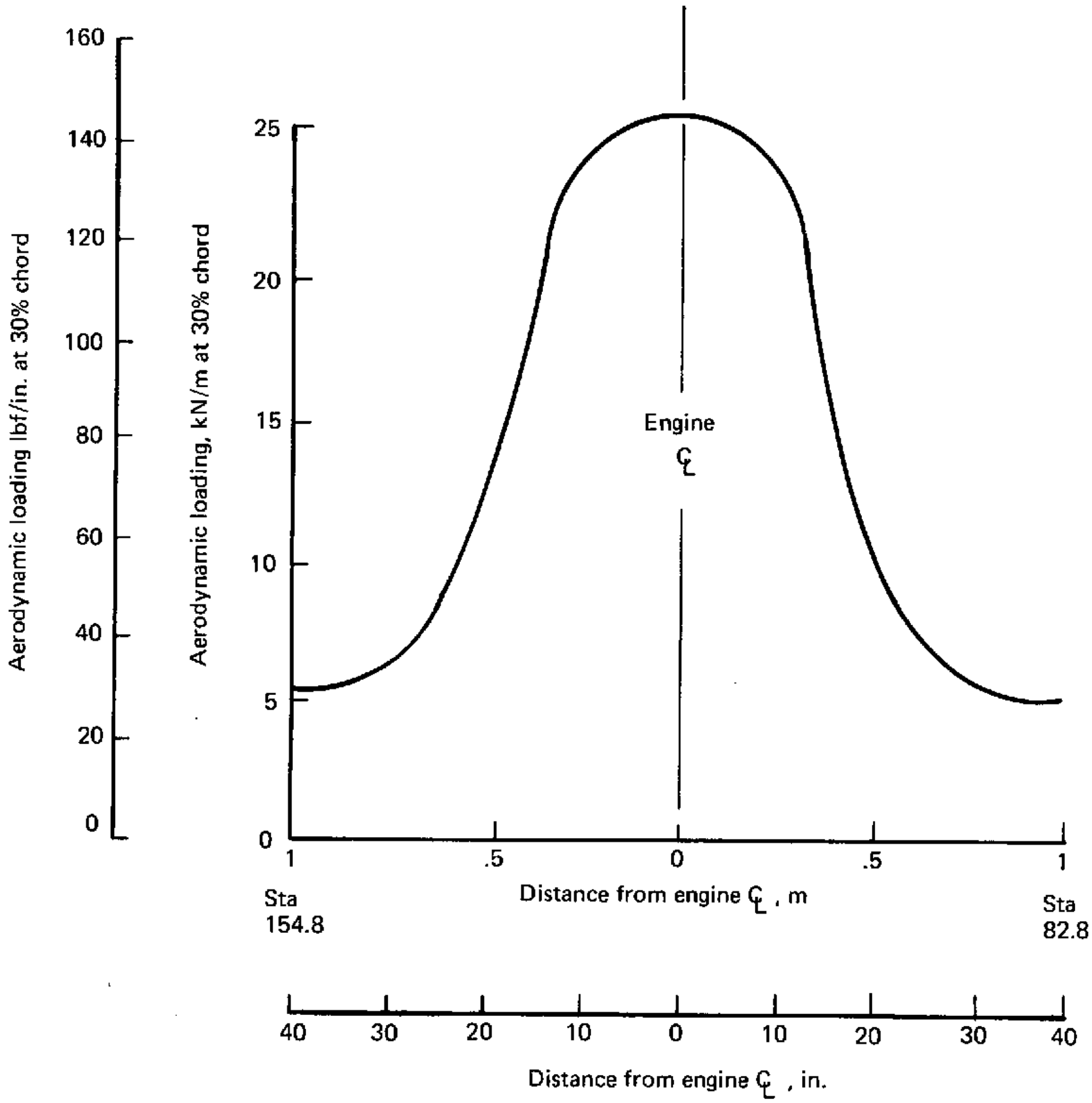

FIGURE 5.—AERODYNAMIC LOADING ON THIRD-ELEMENT STOL FLAP (REF. NASA A0332)

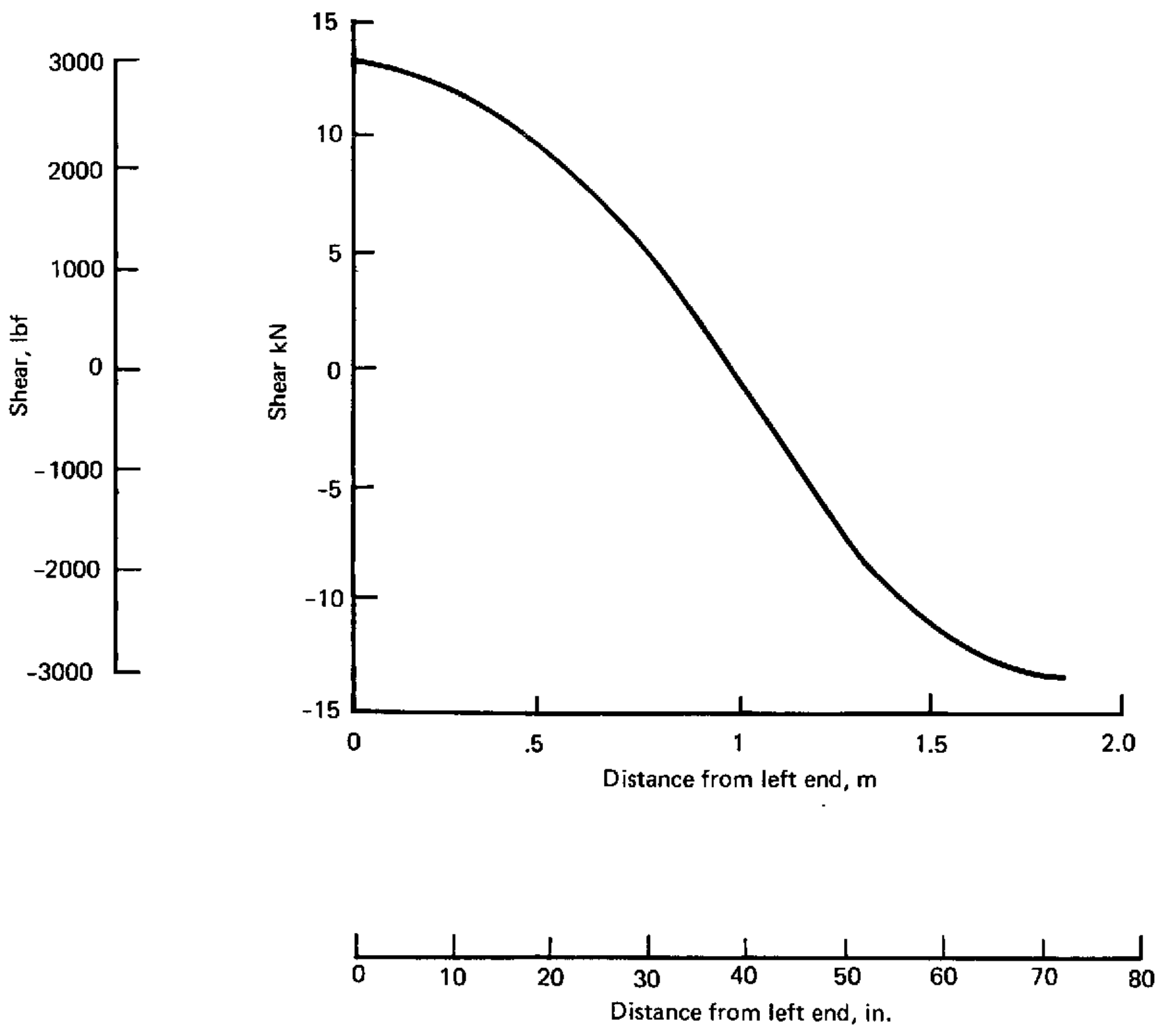

FIGURE 6.—SHEAR DISTRIBUTION, THIRD-ELEMENT STOL FLAP

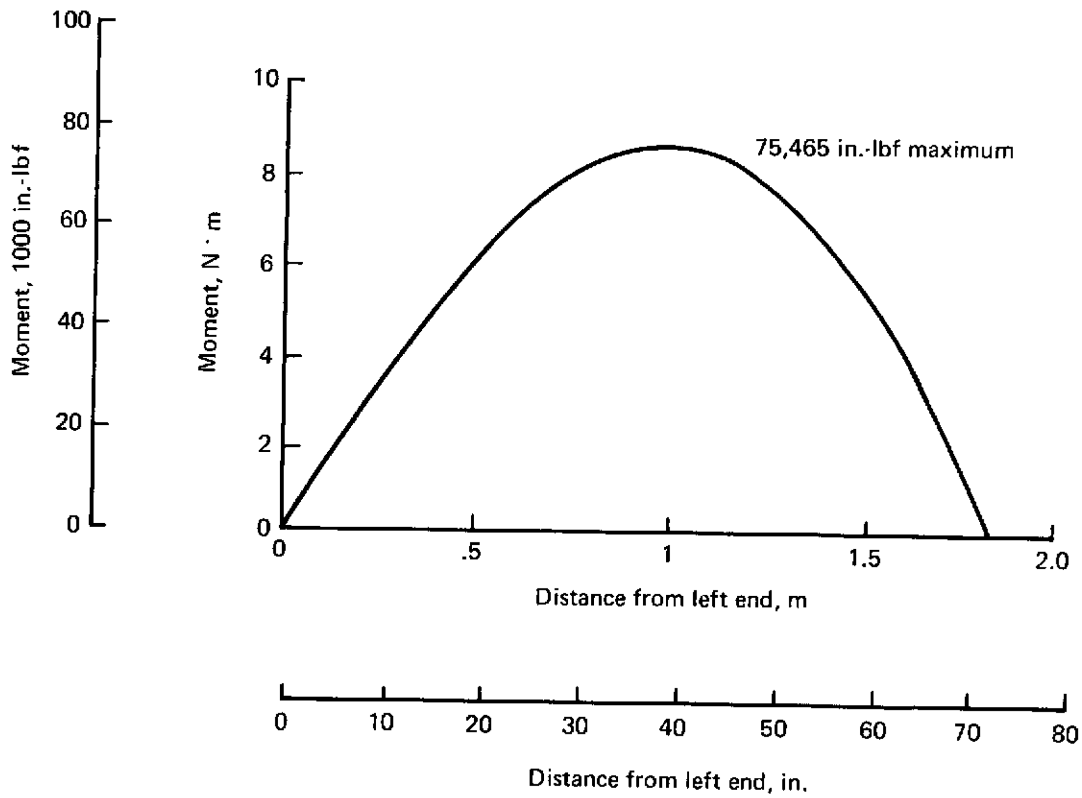

FIGURE 7.—MOMENT DISTRIBUTION, THIRD-ELEMENT STOL FLAP

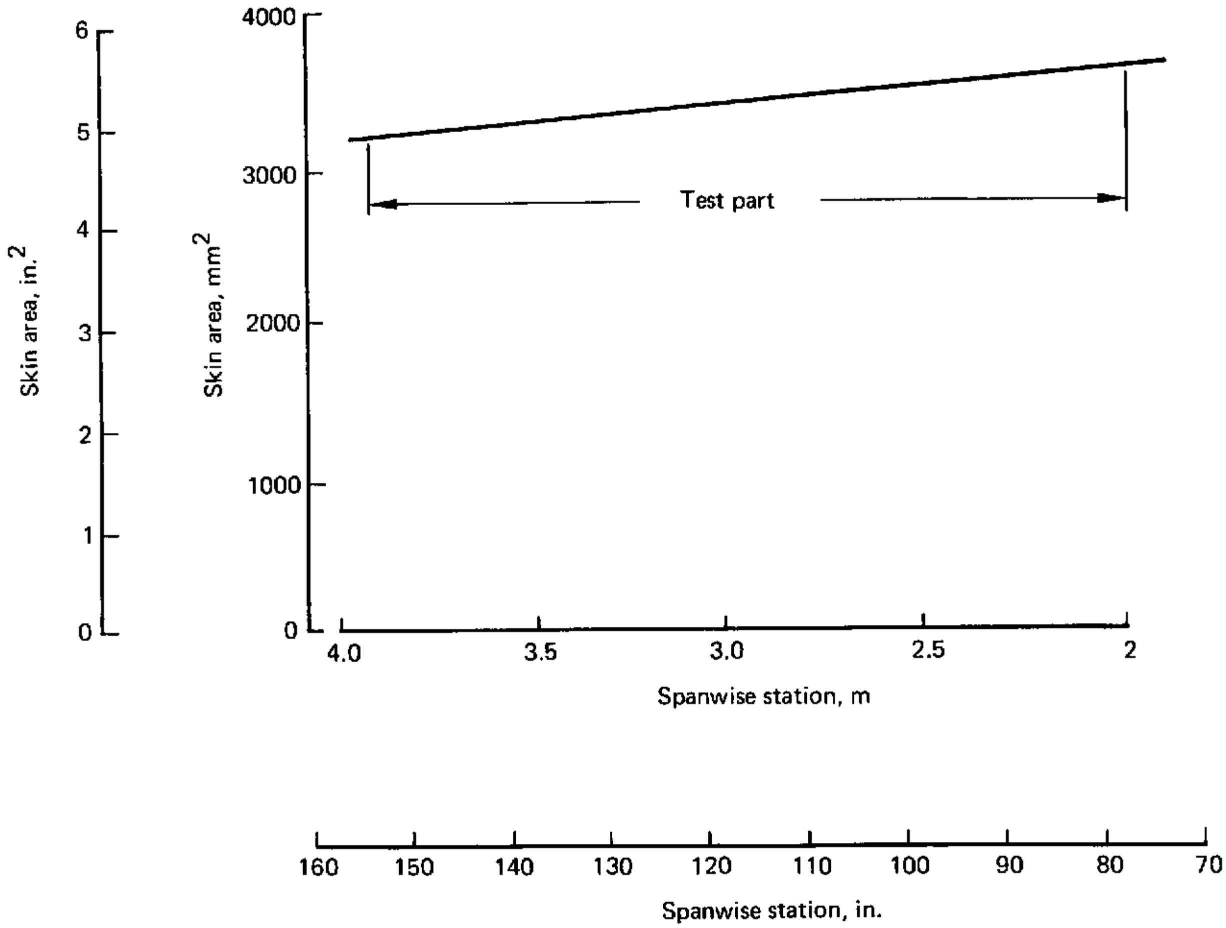

FIGURE 8.—AIRFOIL SKIN AREA, 1.91-mm (0.75-IN.) SKINS

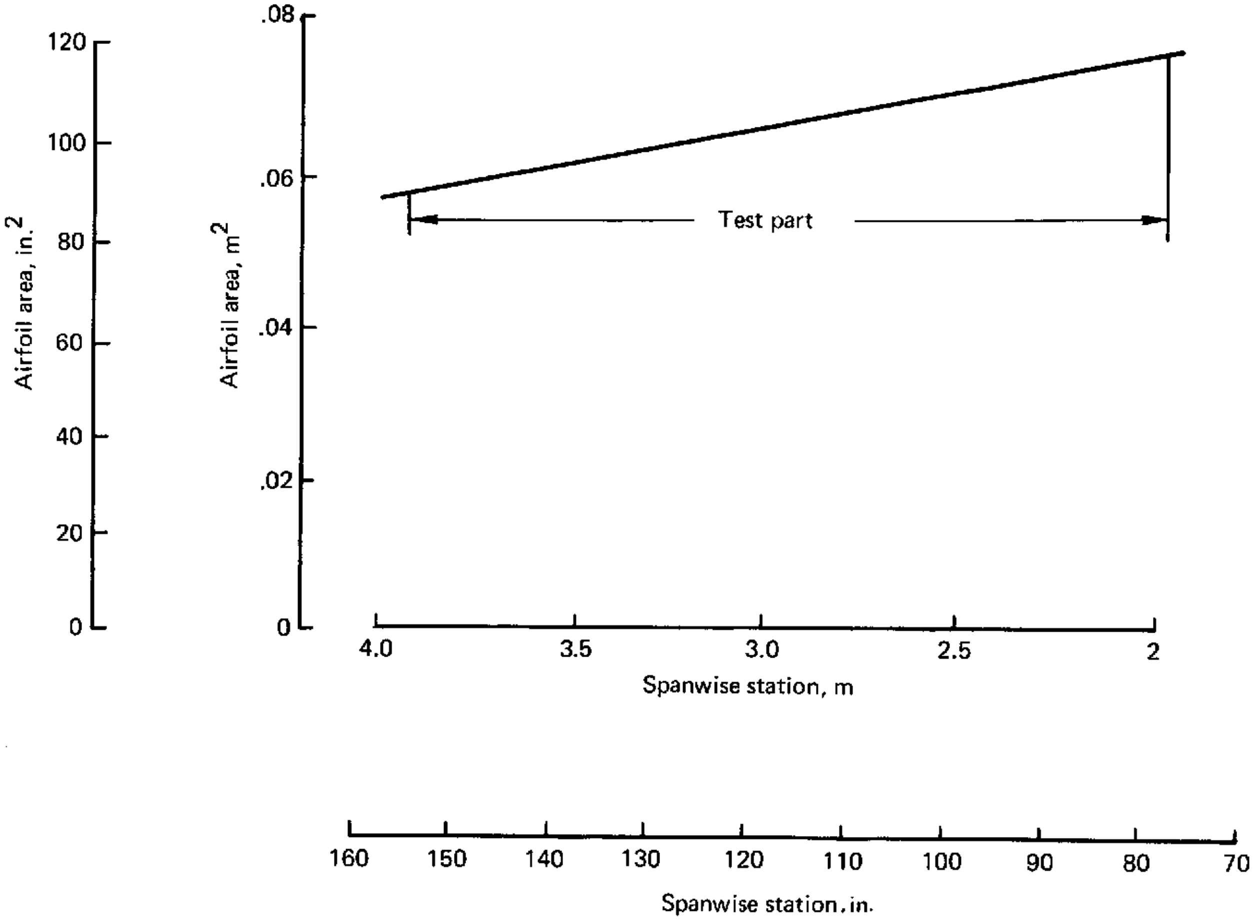

FIGURE 9.—AIRFOIL CROSS-SECTIONAL AREA

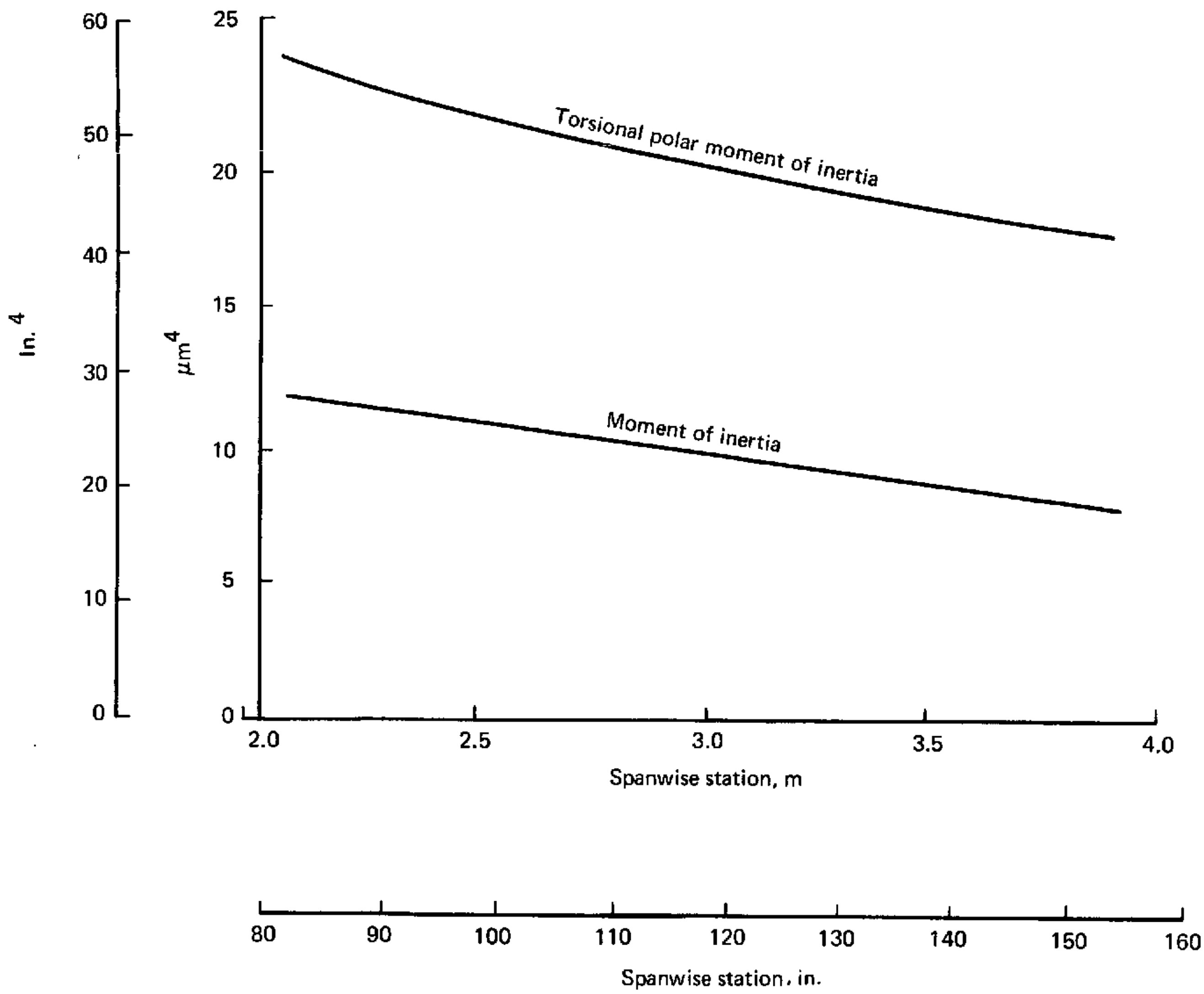

FIGURE 10.—TORSIONAL AND BENDING MOMENT OF INERTIA

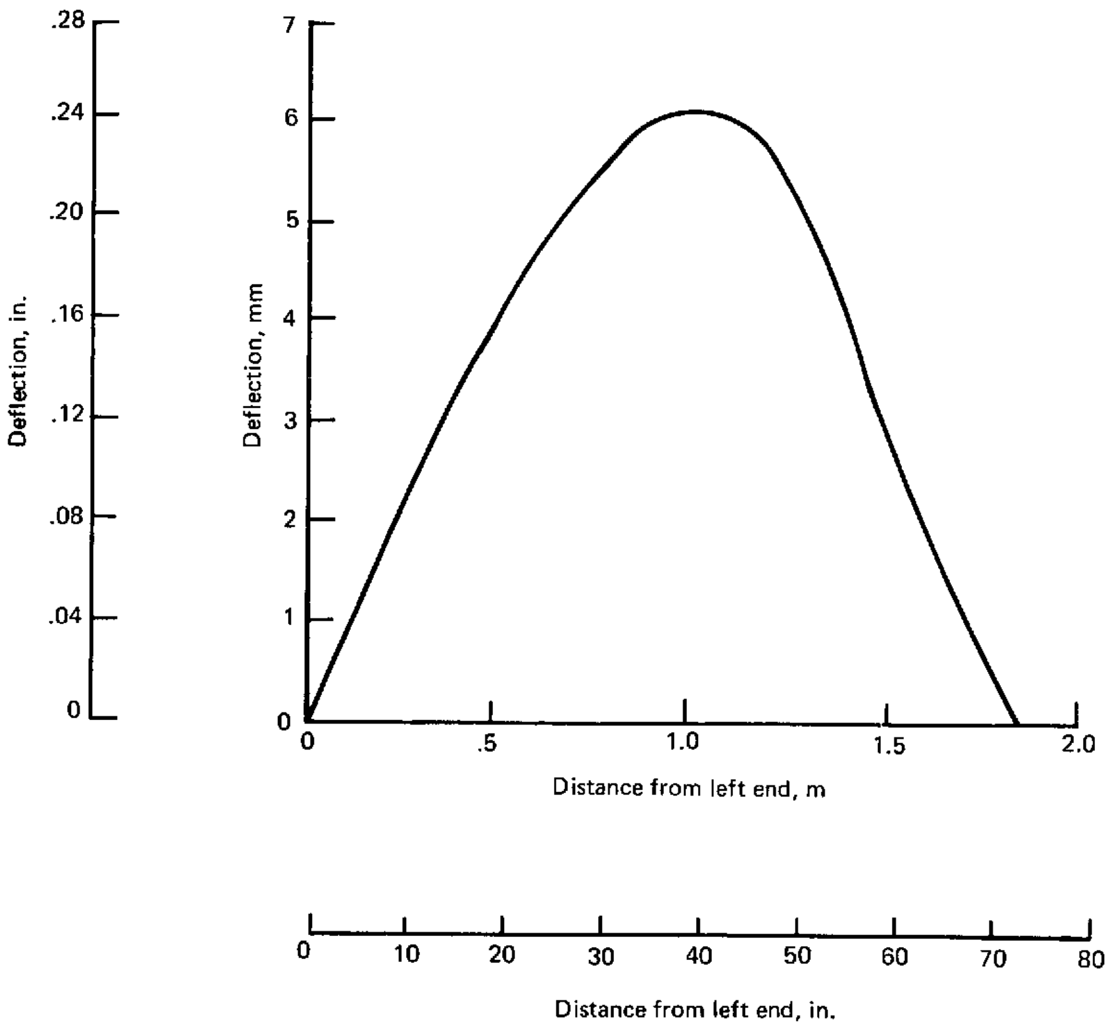

FIGURE 11.—FLAP DEFLECTION

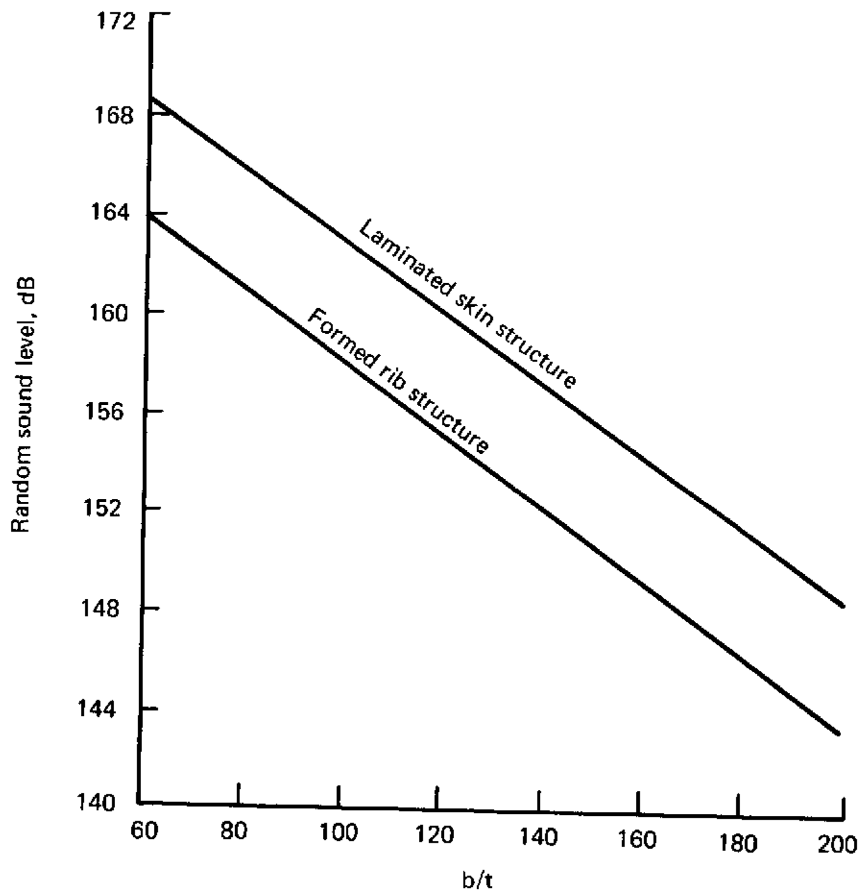

FIGURE 12.—SONIC FATIGUE DESIGN, SKIN/STRINGER STRUCTURE, 1000-HR LIFE AT SOUND

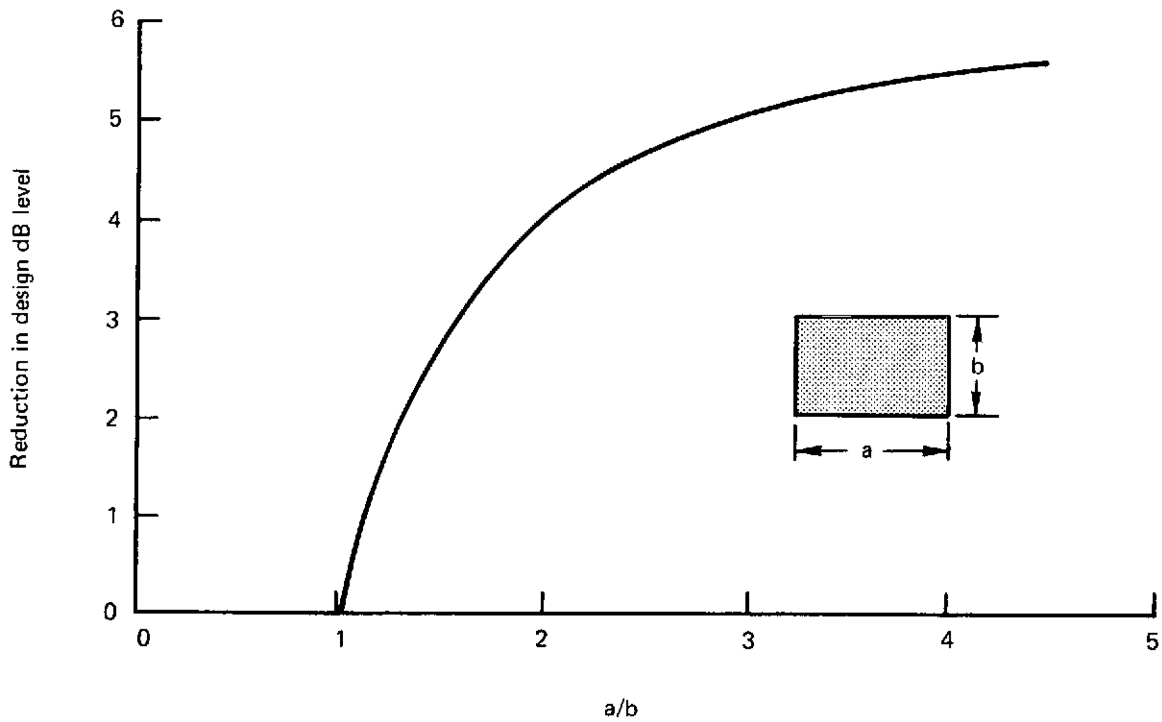

FIGURE 13.—LEVEL CORRECTION FACTOR FOR RECTANGULAR PANELS IN SONIC FATIGUE

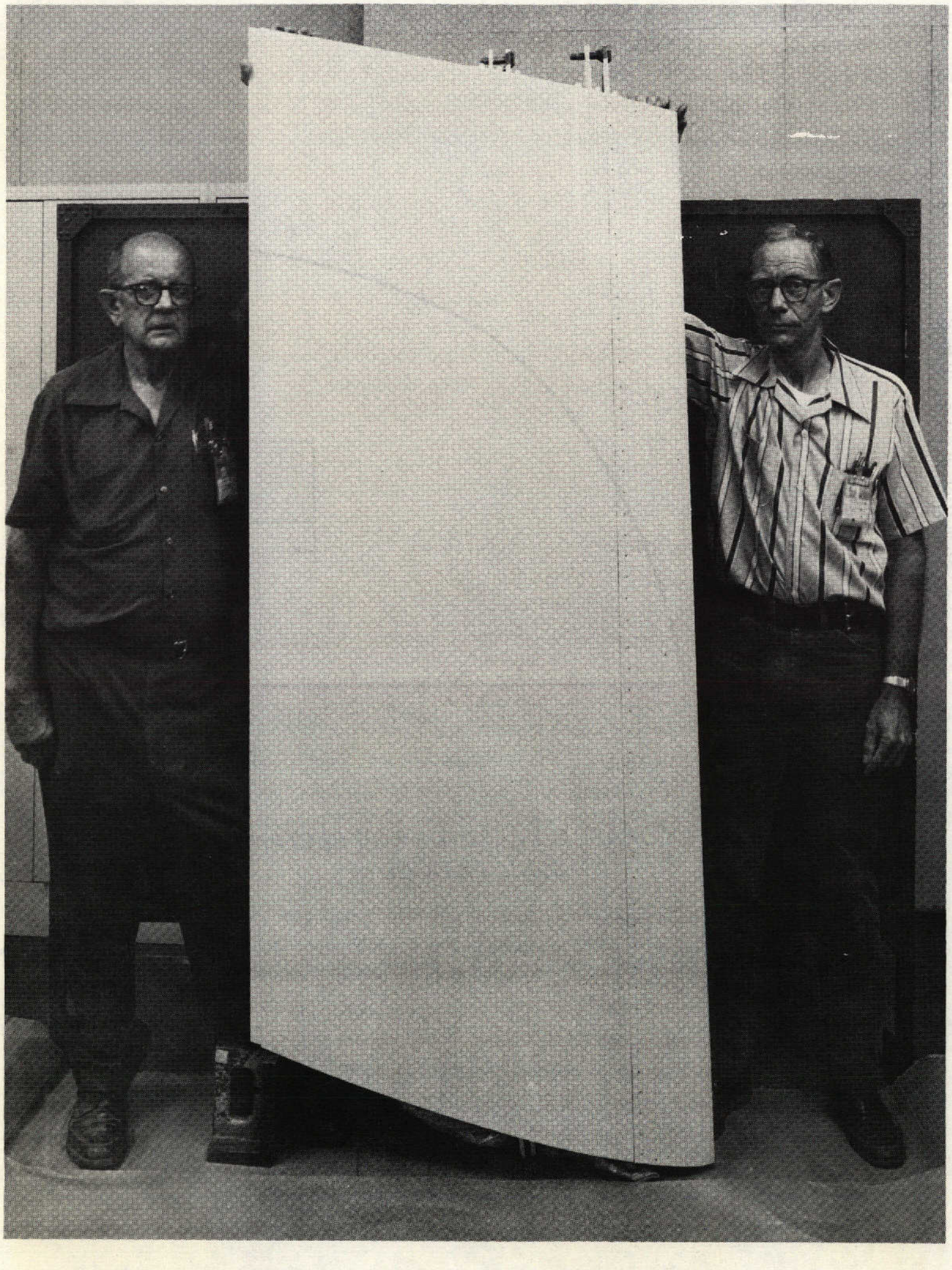

FIGURE 14.—COMPLETED FLAP ASSEMBLY

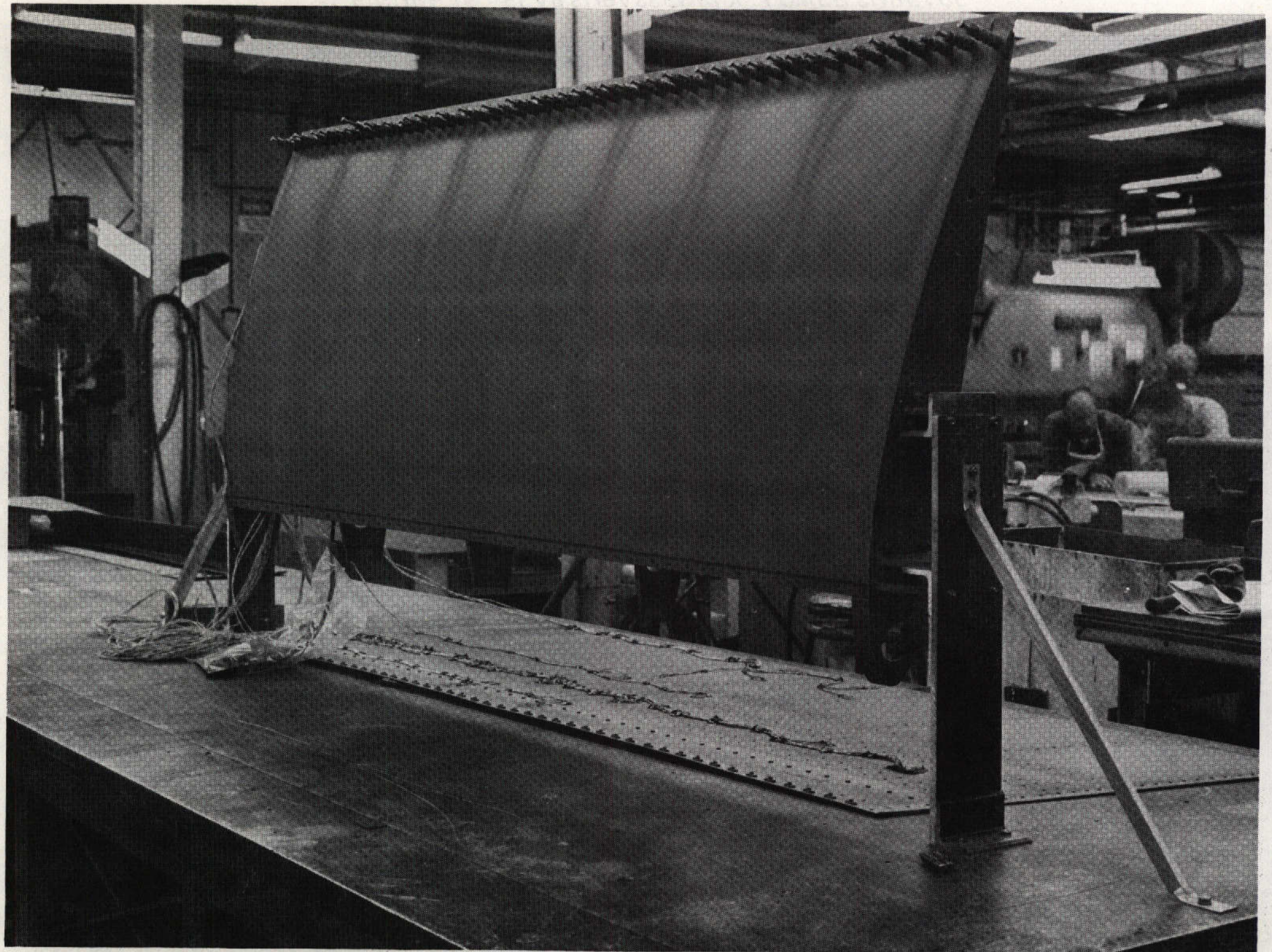

FIGURE 15.—FLAP FINAL ASSEMBLY

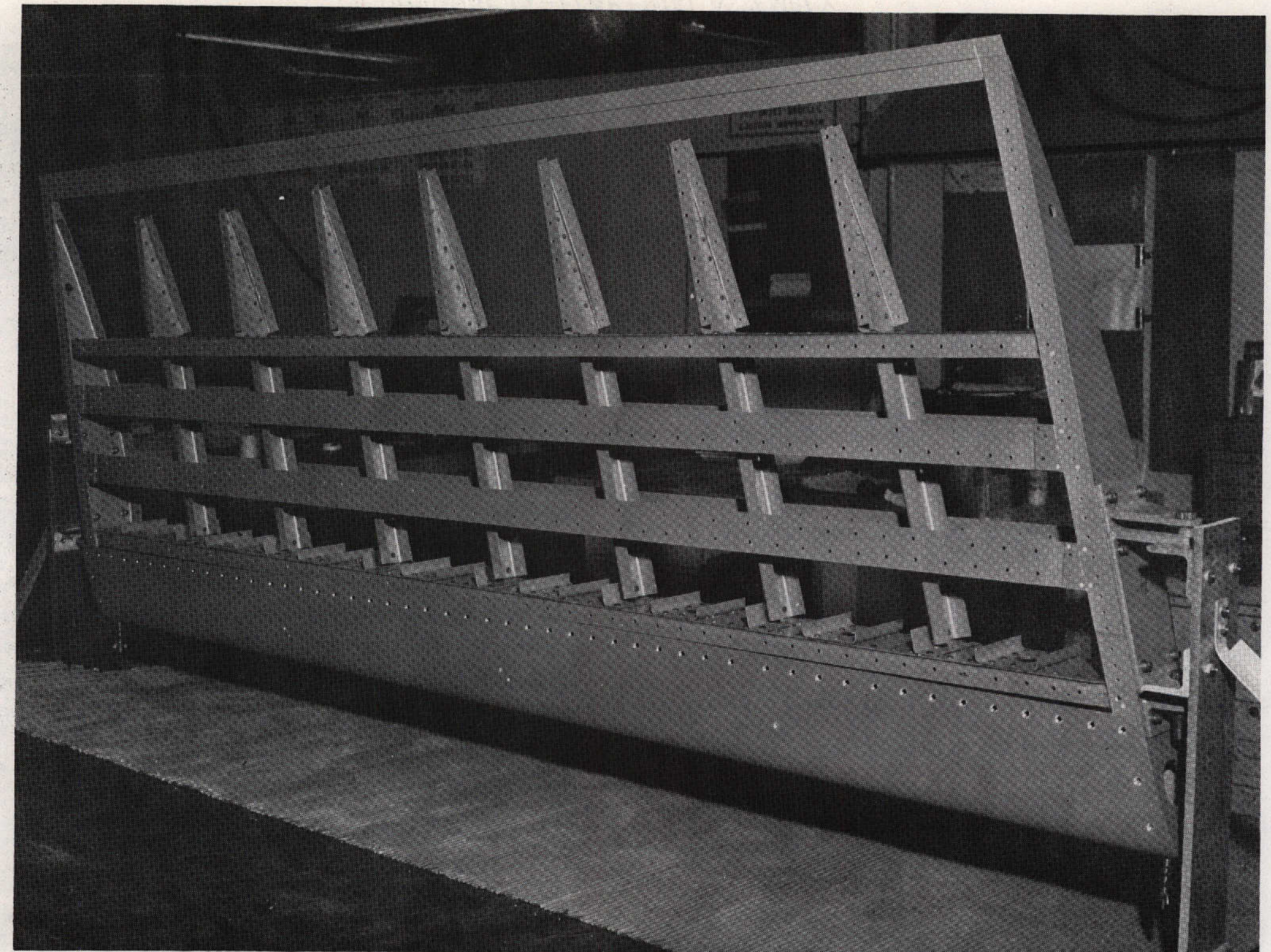

FIGURE 16.—FLAP SUBASSEMBLY—LOWER SURFACE

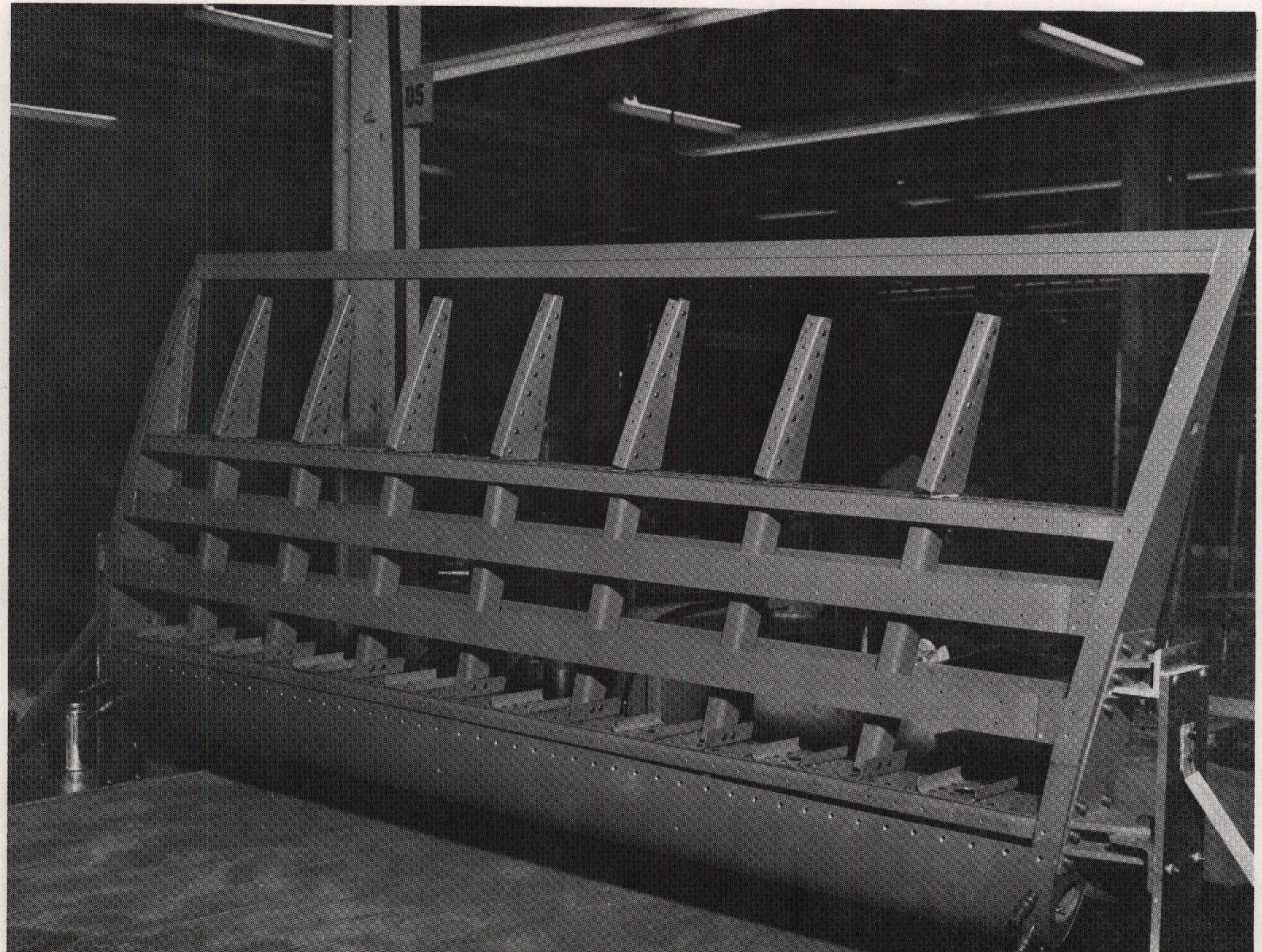

FIGURE 17.—FLAP SUBASSEMBLY—TOP SURFACE

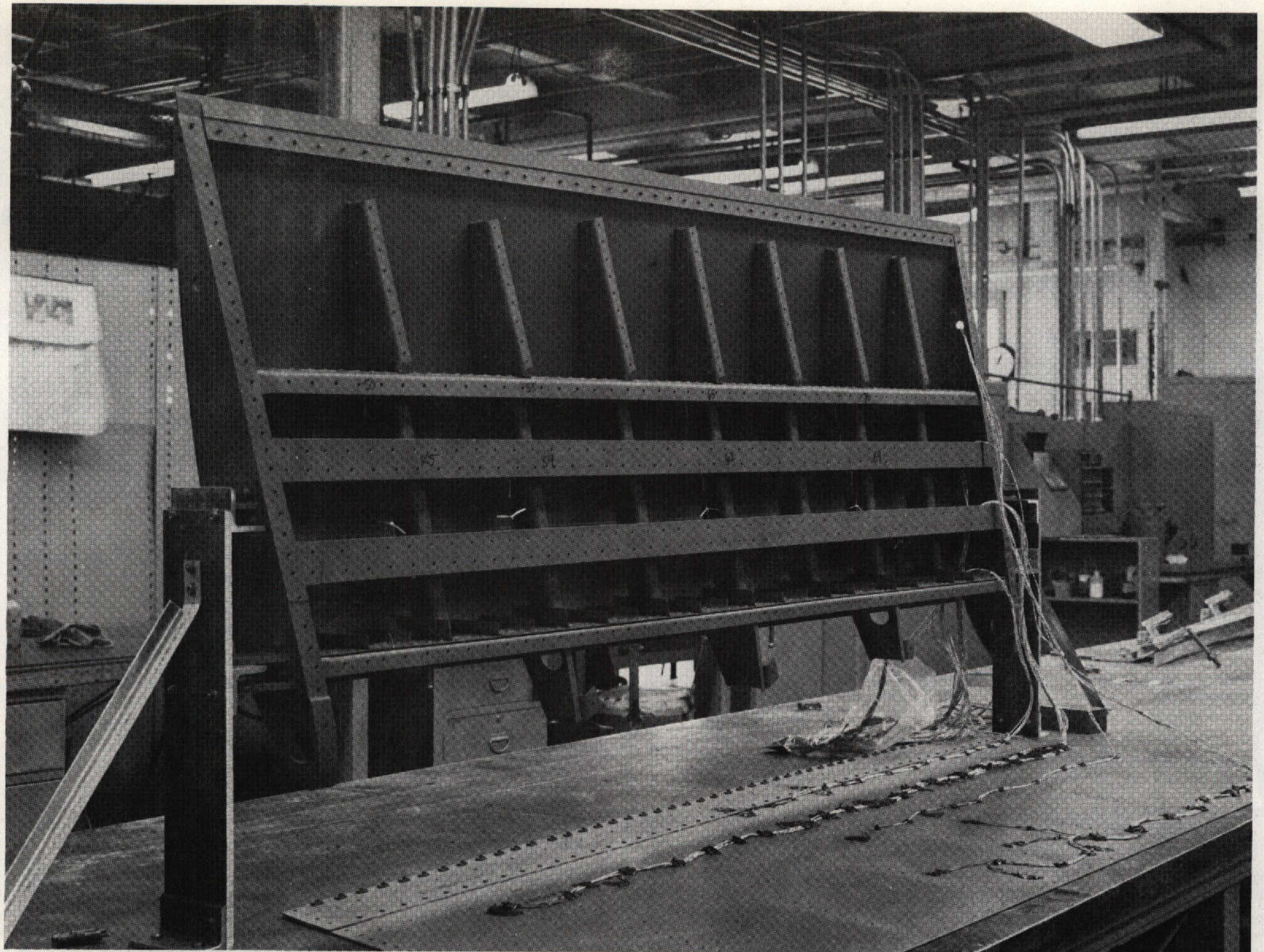

FIGURE 18.—FLAP SUBASSEMBLY AND INSTRUMENTED LOWER SKIN

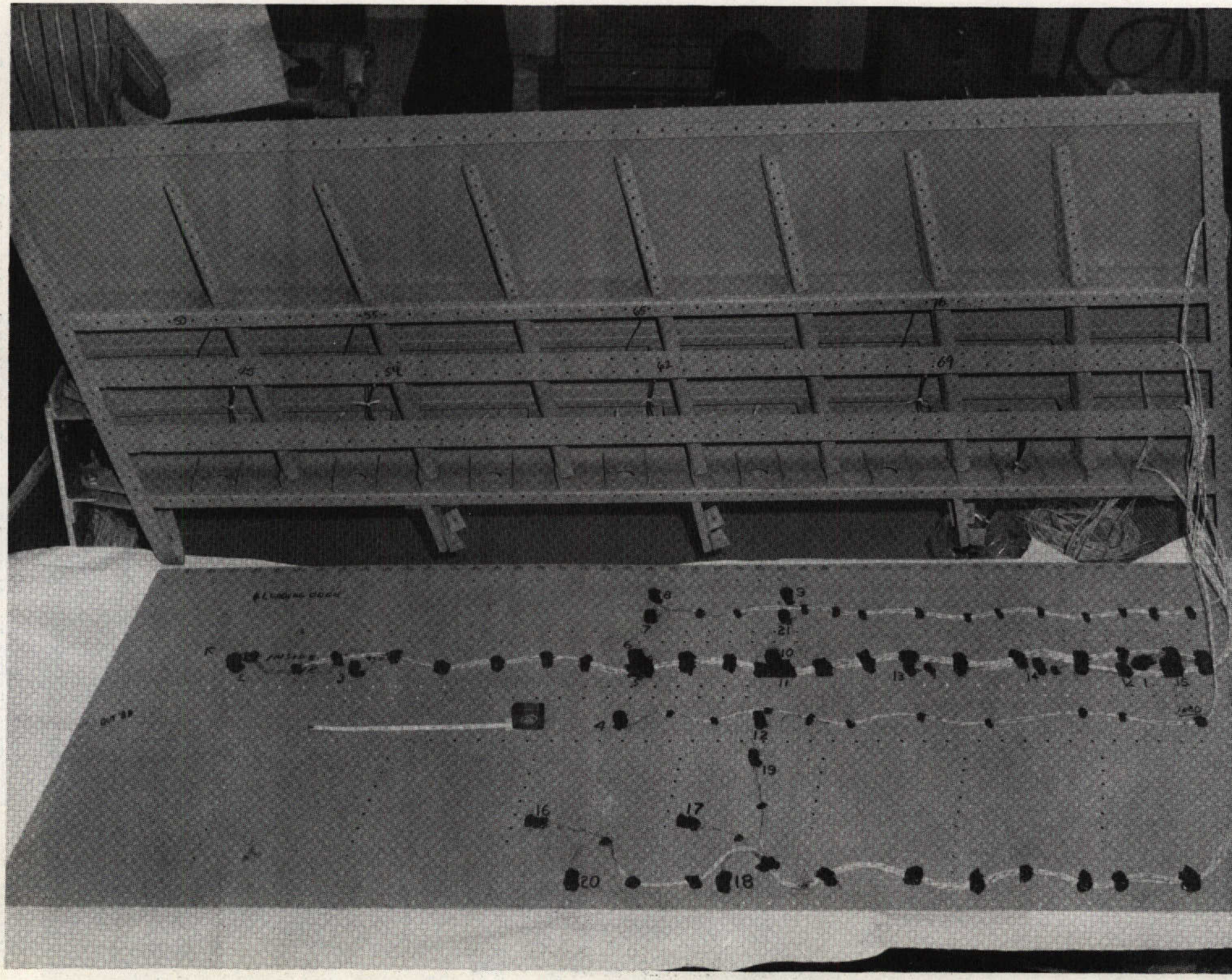

FIGURE 19.—INSTRUMENTATION LAYOUT

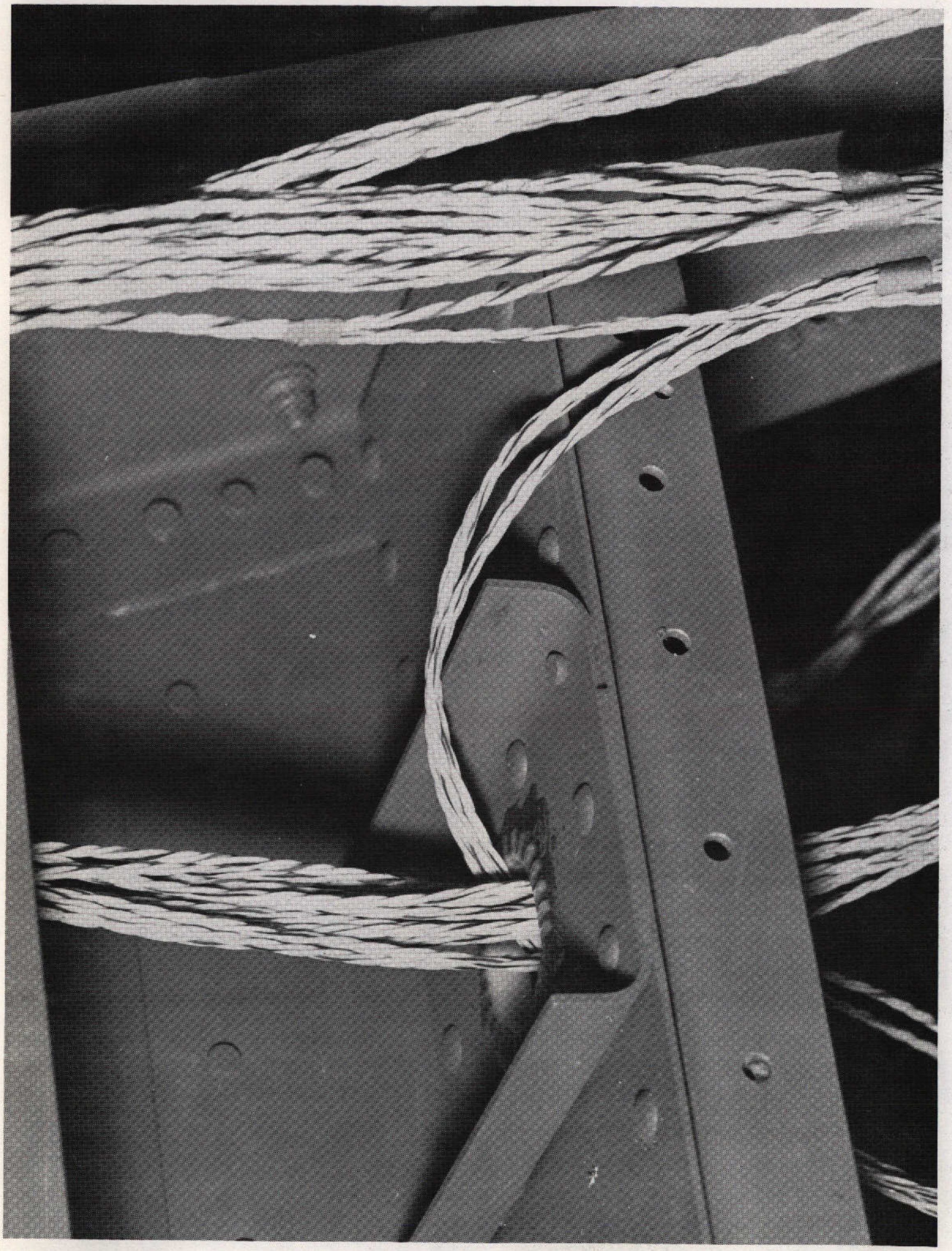

FIGURE 20.–INSTRUMENTATION WIRE ROUTING THROUGH FORWARD SPAR

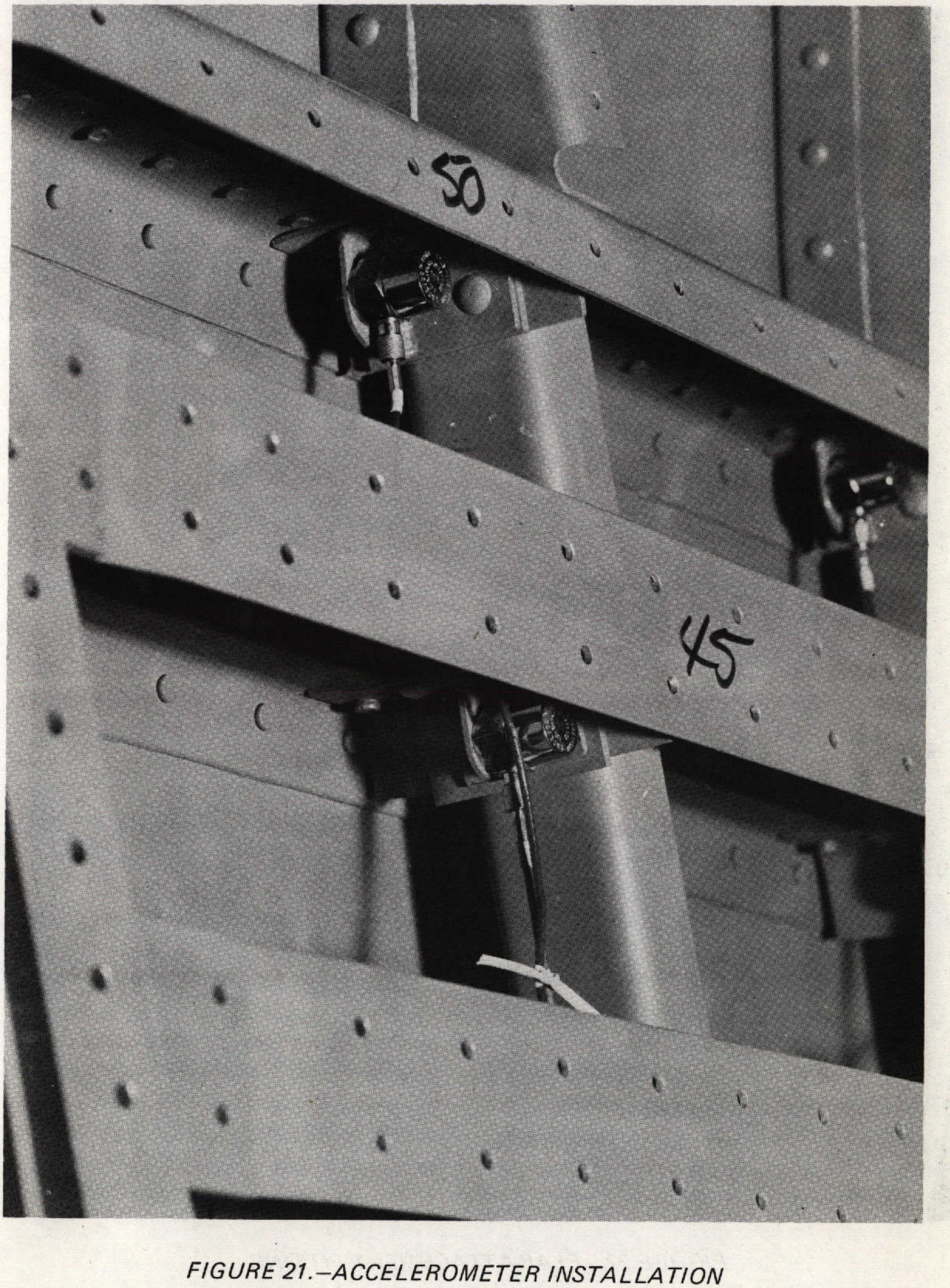

FIGURE 21.—ACCELEROMETER INSTALLATION

FIGURE 22.—FLAP ATTACHMENT FITTING

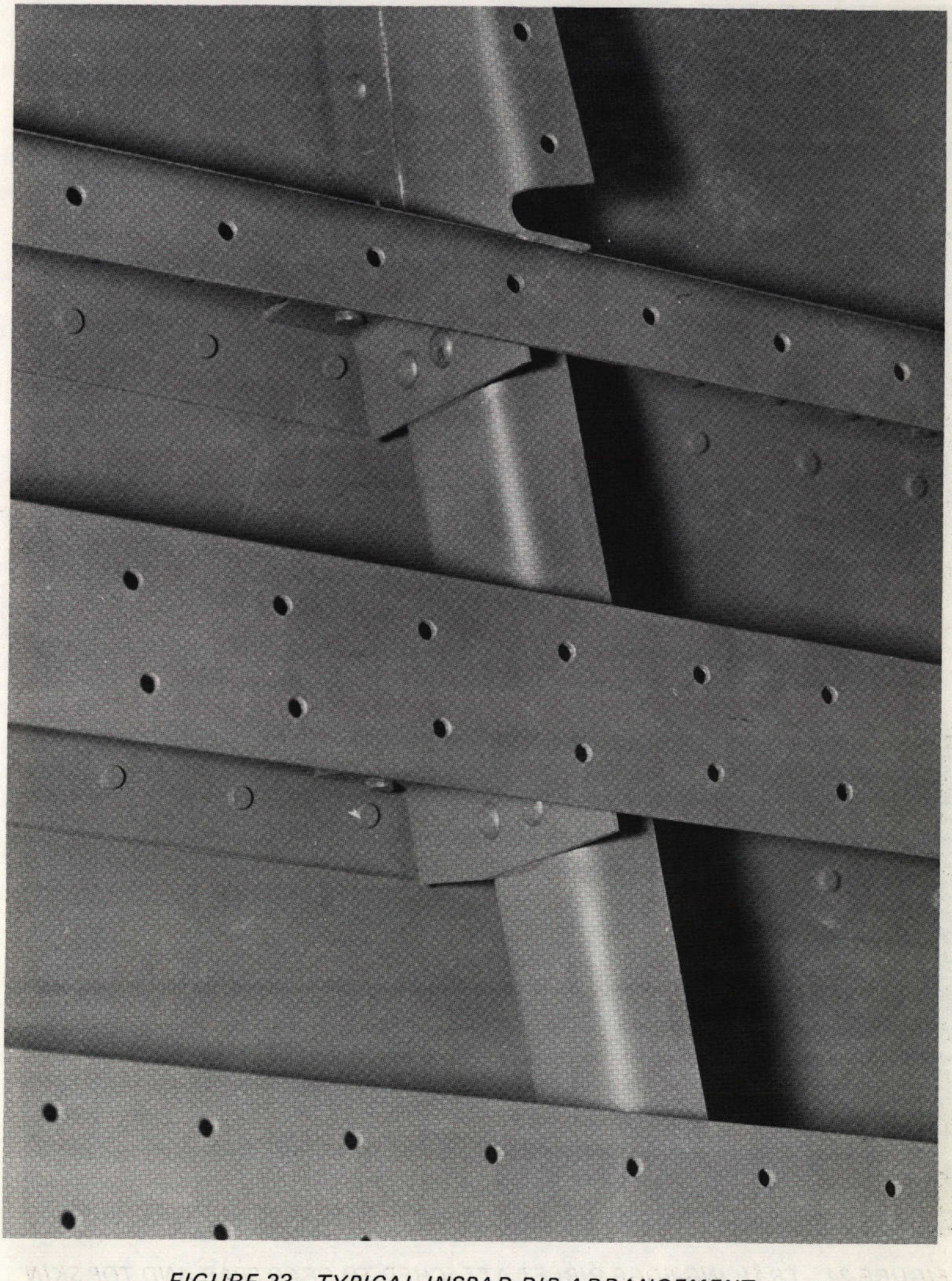

FIGURE 23.—TYPICAL INSPAR RIB ARRANGEMENT

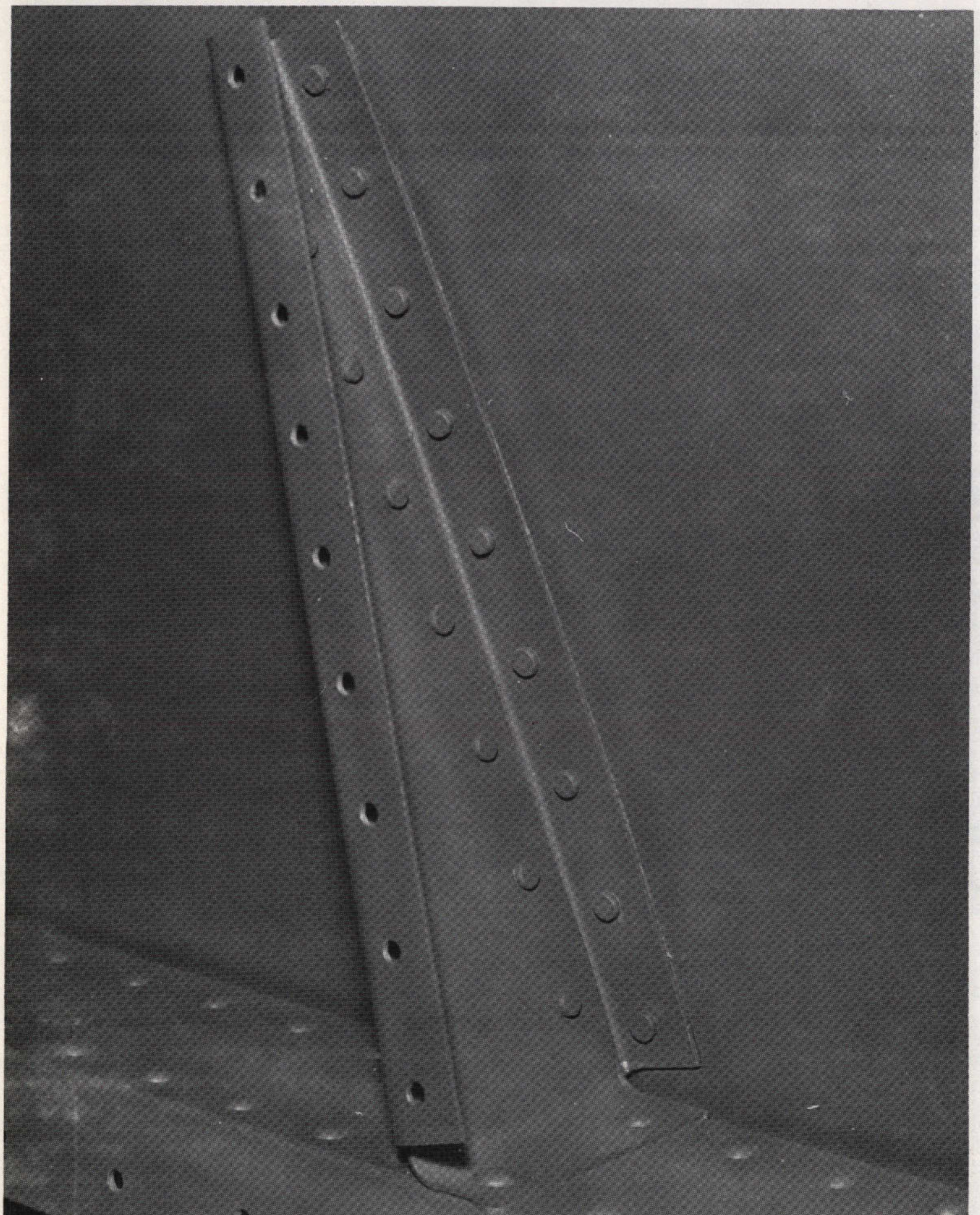

FIGURE 24.—TRAILING-EDGE RIBLET ATTACHED TO REAR SPAR AND TOP SKIN