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NASA CR-134518

DESIGN AND FABRICATION OF AN AUGMENTOR WING MODEL FOR ACOUSTIC TESTS

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16. Abstract

This report covers the design and fabrication of a full-scale section of an augmentor wing to be used for acoustic testing at the Lewis Research Center.

The design of the hardware built under this contract was configured from data obtained during tests run at the Boeing Commercial Airplane Company on augmentor wing models designed and built under NASA-Ames contract NAS2-6344 and reported in NASA report CR-114534.

This hardware will be used primarily to investigate scaling effects of acoustic data obtained during the Boeing-run model tests. Typical model test data is shown in this report, together with predictions on both performance and acoustics that can be expected from the full-scale section built under this contract.

The major sections of this report cover the aerodynamic and acoustic criteria of the flap system and nozzles, detail discussion on the hardware built, a test system operation procedure, and a complete stress analysis of the entire test system.

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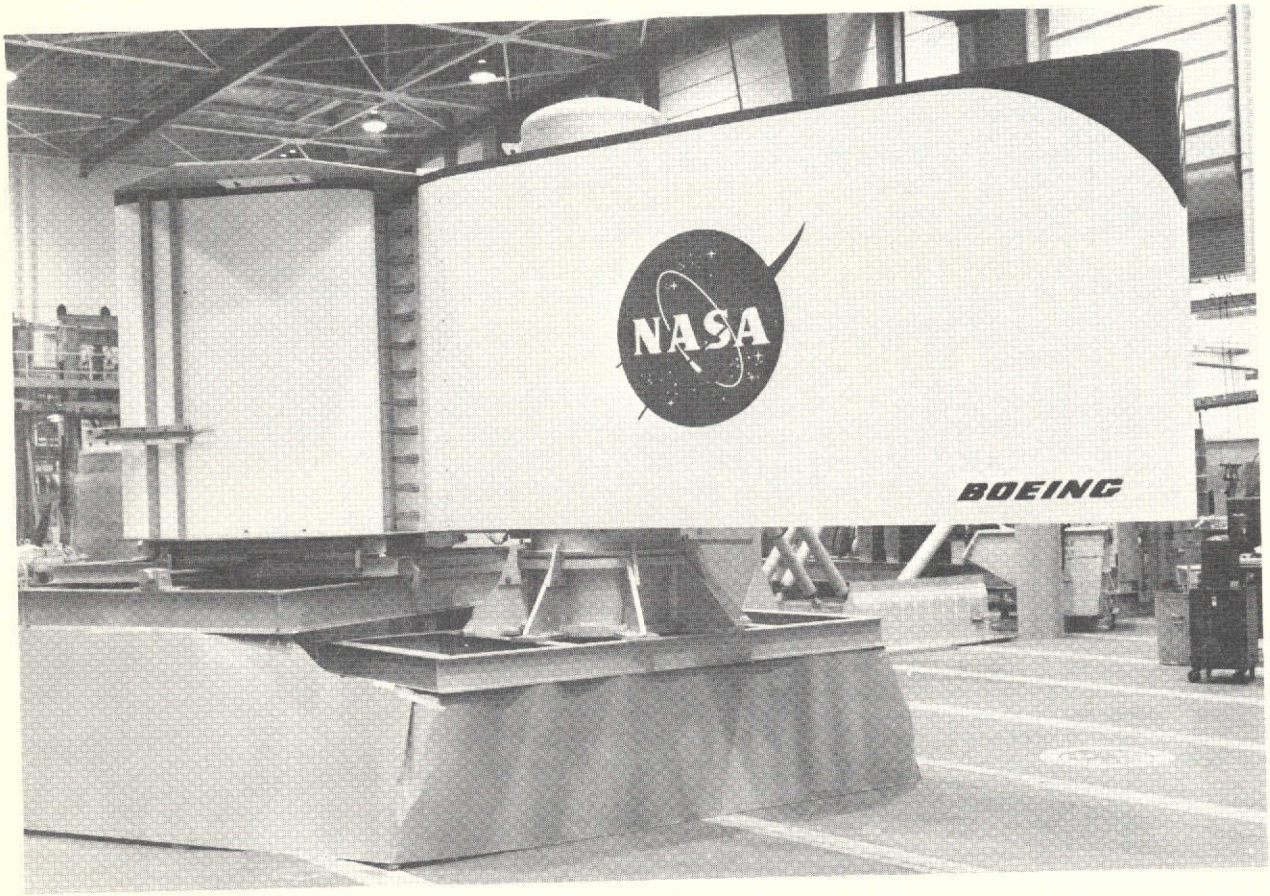
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DESIGN AND FABRICATION OF AN AUGMENTOR WING MODEL FOR ACOUSTIC TESTS

by John Jackson, R. W. Schedin, and J. M. Campbell

1.0 SUMMARY

This technical report covers the design analysis, fabrication, checkout, and delivery of a full-scale section of an augmentor wing test system to the NASA-Lewis Research Center under NASA-Lewis Research Center contract NAS3-17362. This system is to be used primarily to investigate acoustic scaling effects of an augmentor wing system developed by Boeing under NASA contract NAS2-6364 and recorded in NASA report CR-114534 (Boeing document D6-60174).

An overall photograph of this test system is shown in the frontispiece of this report.

The system design was based on data obtained from the previously built smaller scale models and it was integrated with the NASA-Lewis Research Center augmentor wing test facility. During the design phase, optimum configurations of the flap system, suppressor nozzle, and wing trailing edge were taken from the results of the previous work noted above.

The hardware built for this contract represents a full-scale section of wing of the augmentor wing airplane established under NASA contract NAS2-6364 and also reported in CR-114534. This resulted in the final hardware having a scale of 1:2.93. The previously tested model components were scaled directly by this factor of 2.93, with the exception of the span of the wing and flap test section. The air supply of the Lewis Research facility limited the wing section span to 75 in.

The acoustic panel design was not scaled directly but was designed with additional refinements to increase the noise suppression over those tested and recorded in CR-114534. Three sets of acoustic panels were built, one at the design point, and one on each side of the design point to allow a wide scope of acoustic testing.

The optimum nozzle (suppressor) chosen was that of a corrugated lobe-exit configuration. The difference existing between this and the previously tested smaller scale nozzle was that 12 lobes instead of 20 were used. During the following three months, complete detail design and stress analysis was carried out, and as each part was detailed the final drawings were sent to NASA-Lewis for approval prior to the start of manufacture.

Upon completing the manufacturing of all details and major assemblies, the final model assembly and checkout were executed under close engineering supervision with support by Boeing Quality Control. Planned order inspection records of all parts and assemblies are available for review at Boeing. At this point, the model hardware was broken down into suitable packages and shipped to the NASA-Lewis Research Center at Cleveland, Ohio.

The design analysis and functional tests, which included the pressure testing of all pressure vessel hardware to the nozzle interface, showed that the model met or exceeded all structural requirements of the specifications.

The acoustic panels, of a double layer polyimide honeycomb construction, required careful material selection and inprocess inspections to meet the specified Rayl numbers. This intensely supervised assembly procedure slowed production of the panels so that the two alternate panel sets were shipped subsequent to delivery of the augmentor wing model. The model stress analysis in the appendix of this report shows that the hardware is designed for maximum model test operating loads, and will withstand wind velocities to 100 mph.

2.0 INTRODUCTION

The potential for application of the augmentor wing concept to commercial STOL airplanes depends on the achievement of large amounts of noise suppression. This must be done while maintaining high thrust augmentation in augmentors that fit within wing geometries acceptable to STOL airplanes. It was determined during task III* that a noise objective of 95 PNdB at 500-ft sidelines can be met and static thrust augmentation levels above 1.40 are attainable.

STOL airplane configuration design studies have indicated that there are installation and weight advantages for augmentors using single-row, full-height lobe primary nozzles rather than the multirow lobe nozzles used to achieve the above objective. New nozzle exit shapes, such as high-aspect ratio lobes with a central splitter (splitter lobe) corrugated exit lobes and cross-mixing (or hypermixing) lobe nozzles, show promise for more favorable augmentor operation with lighter and less complex installations.

Advances in lining technology have led to the concept of multilayer linings to improve the sound absorption in the augmentor. The multilayer linings reduce the peak noise level by more than the single-layer lining; they also broaden the frequency range where attenuation can be achieved.

Some promising augmentor wing airplane designs incorporate lower nozzle pressure ratios (NPR = 1.4 to 1.6). Acoustic and thrust performance data at low nozzle pressure ratios are also needed for related applications, parametric information, and for a better understanding of the relation between noise performance and jet velocity.

Static rig performance and noise tests, along with related systems studies, were formulated to determine the potential of augmentor wing designs. The above ideas were implemented in model designs and system studies; the results are documented in CR-114534.

*Campbell, J. M., Lawrence, R., and O'Keefe, J. V., Design Integration and Noise Studies for Jet STOL Aircraft, final report volume III, Static Test Program, D6-40552-3 (NASA CR-114285), The Boeing Company, May 1972.

3.0 DESIGN ANALYSIS

The following paragraphs describe the basic design approach and technical considerations used in establishing the configuration, size, and operating characteristics of the augmentor wing model hardware.

3.1 AERODYNAMIC

The aerodynamic lines of the wing section, flap system, and nozzles were based upon optimum configurations obtained after extensive testing on smaller-scale augmentor wing hardware that was designed, built, and tested at Boeing facilities in Seattle, Washington. Results of these previous tests are documented in CR-114534.

The wing section used in this present program varies slightly from a direct scaleup of the previous model. The thickness-to-chord ratio was increased from 12% to 15% to accommodate the 30-in. diameter vertical plenum which was sized for low Mach numbers to feed the nozzles. This increase in t/c will not affect the static performance of the test system. The augmentor, consisting of the flap and shroud, has a length of approximately one quarter that of the wing chord. While internal aerodynamic lines of the augmentor represent the optimum configuration found during smaller-scale tests as mentioned above, the leading-edge curvatures of the flap and shroud have a larger radius than would be used for a true airplane configuration. These shapes provide good secondary air entry for static test conditions and eliminate the possibility of internal flow separation.

Several augmentor design variations were tested during earlier Boeing tests. The performance results are discussed in CR-114285, volume III, section 4.3.3.3. The internal design variations tested included a curved shroud surface (nonsymmetrical) and three symmetrical configurations; namely, two-stage diffusion, constant-area throat (mixing), and constant diffusion. Comparison of the four variations tested at different flow turning angles clearly shows that the highest peak thrust augmentation levels are produced by the symmetrical designs. The configuration identified as constant diffusion, which uses straight surfaces from the throat to the augmentor trailing edge, was chosen for all future tests due to its combined high-thrust performance and design simplicity.

Many tests have been conducted to determine the optimum diffuser angle. As reported in both the above reference and the task V report (CR-114534), the maximum thrust augmentation is consistently measured at 5° total diffuser angle.

Figures 1 and 2 show typical augmentor exit Mach numbers that can be expected for both takeoff power setting and approach power setting with the flap angle of 20° and 50°, respectively. Figure 3 shows thrust augmentation versus nozzle pressure ratio for a typical set of model hardware that was discussed in CR-114534.

The augmentor length of the model described in CR-114534 was equal to 55 equivalent slot heights of the baseline slot nozzle having an aspect ratio of 100. (See CR-114285, vol. III, fig. 20, p. 55.) It was noted in the above report that the longest practical length is only 46 equivalent slot heights and that this shorter augmentor as built for this current program will cause a predictable decrement in both performance and noise suppression. Two nozzles were built for this current program; one, a slot nozzle of aspect ratio 60, and the other, a corrugated lobe nozzle. These nozzles were scaled directly by using a scale factor of 1:2.93. The slot nozzle was taken from CR-114285, vol. III, figure 20, page 55, while the corrugated lobe nozzle was taken from CR-114534, figure 27, page 39. The facility airflow capability was the factor that established the augmentor test section span. This span resulted in the corrugated nozzle having 12 lobes compared to the 20 lobes of the smaller-scale model.

A screech shield was added to the slot nozzle built for this program to give a predicted increase in noise suppression over the previously tested smaller-scale model hardware. A discussion of the screech shield is also covered in CR-114534 (sec. 4.2).

Figures 4a and 4b show the basic augmentor geometry, while table 1 shows expected thrust augmentation at optimized geometric augmentor relationships with the array area ratio = 8 corrugated lobe nozzle installed for both takeoff and approach power settings at flap angles of 20 and 65 , respectively.

Tests with the augmentor on a facility that lacks thrust measurement capability must rely on the position settings that are listed in table 1. To provide a feeling for the sensitivity of these position settings on thrust augmentation, figures 5, 6, and 7 are shown. It can be seen from these figures that the Z position is a more sensitive parameter than the ℓ_Z position. At high turning angles, the Z position is critically important as flow attachment on the flap must be maintained. Total pressure rakes at the exit of the augmentor are a useful tool in determining that flow attachment is maintained. Use of the exit rakes for thrust integration across the exit of a short augmentor (ejector) is not recommended. As indicated in figure 7, the optimum Z position is plotted as a function of flap deflection angle, δ_F. It can be seen that as the turning angle is increased, the Z position becomes more negative; i.e., the flap coanda surface moves closer to the nozzle jet, and finally flow impingement occurs at the steepest turning angles.

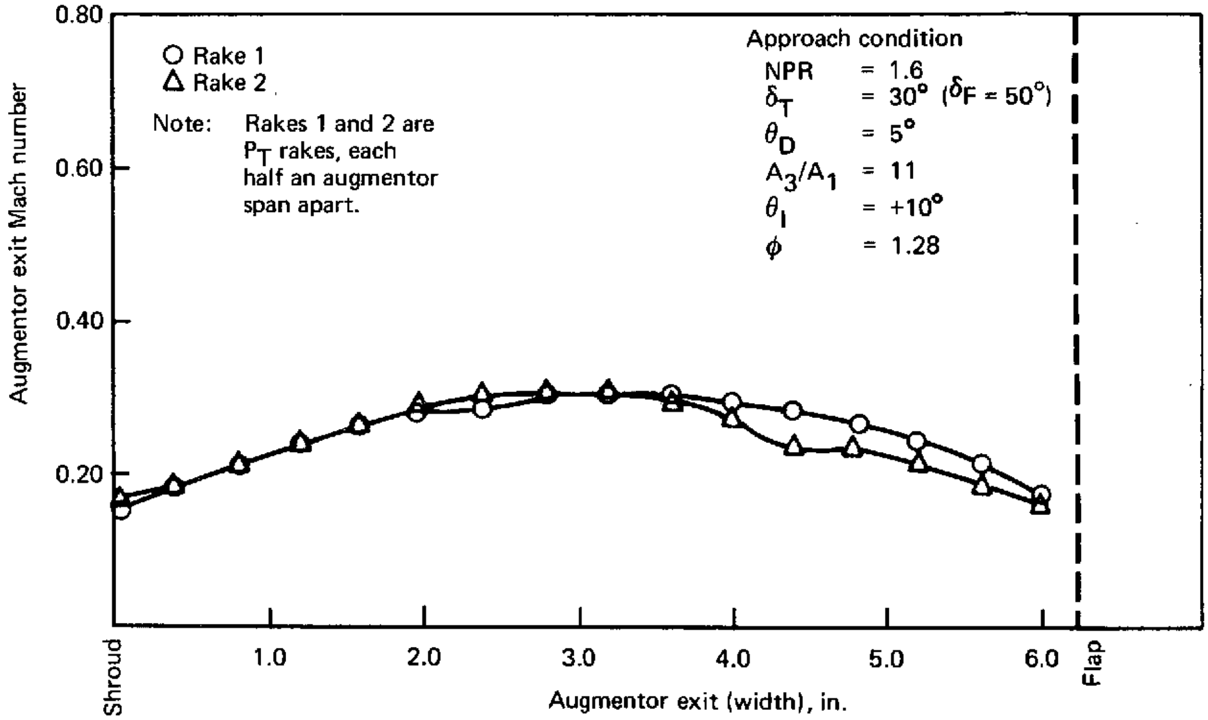


FIGURE 1.–TYPICAL EXIT MACH NUMBER FOR MULTIROW TUBE NOZZLES DATA FROM REPORT CR-114534–RUN 814

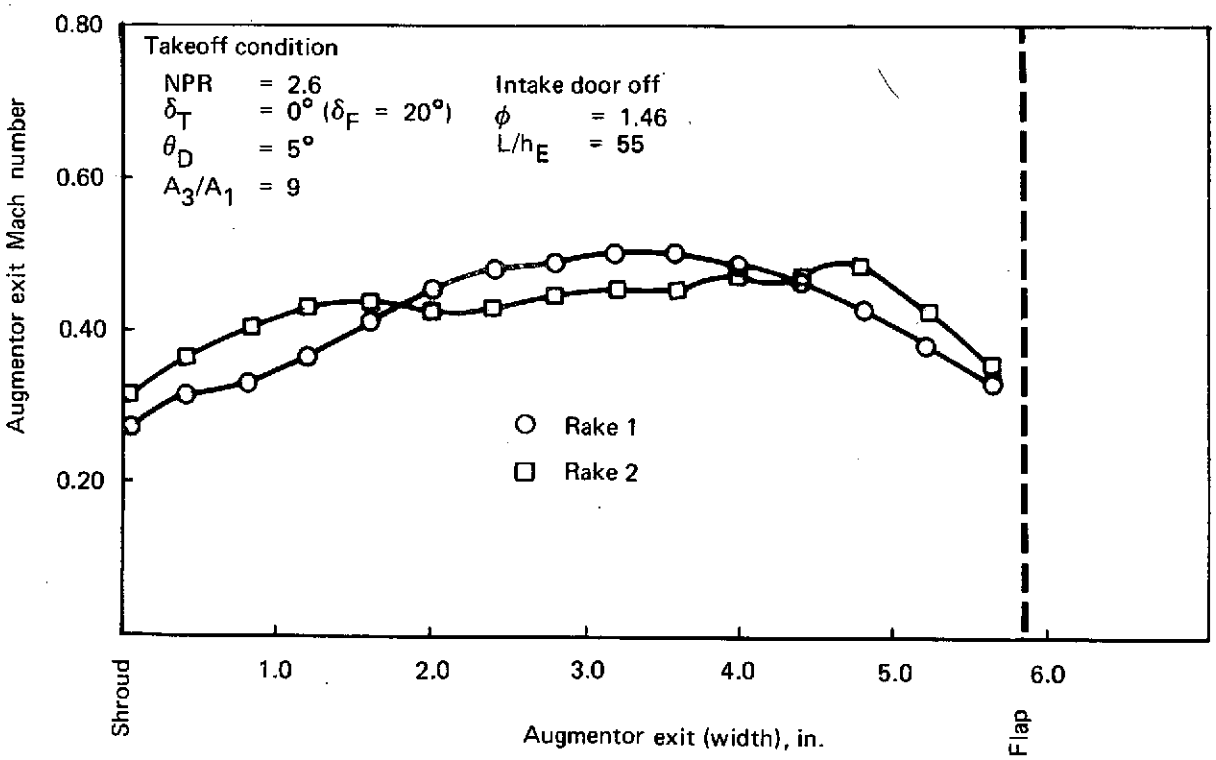


FIGURE 2.–TYPICAL EXIT MACH NUMBER FOR MULTIROW LOBE NOZZLE DATA FROM REPORT CR-114534–RUN 1040

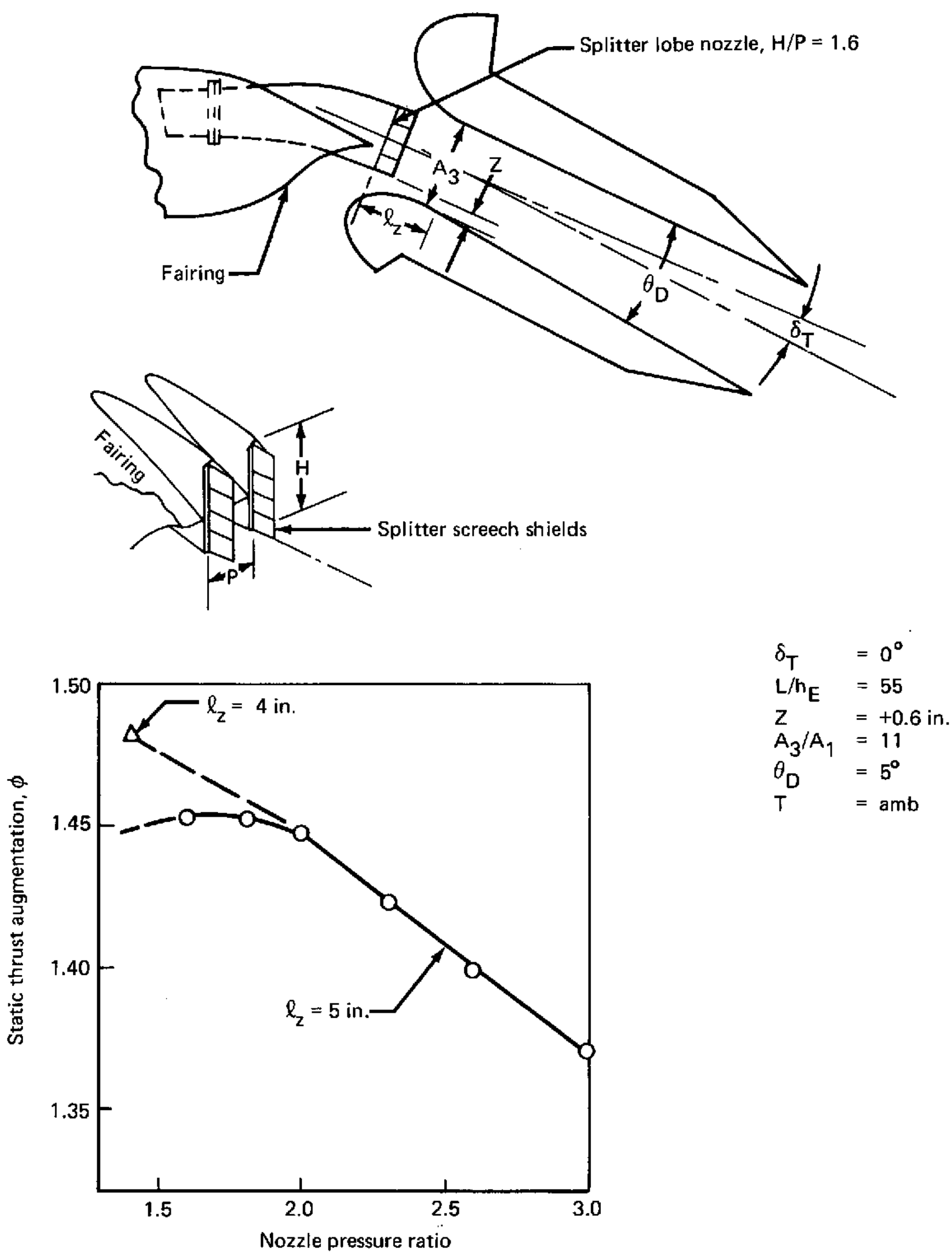
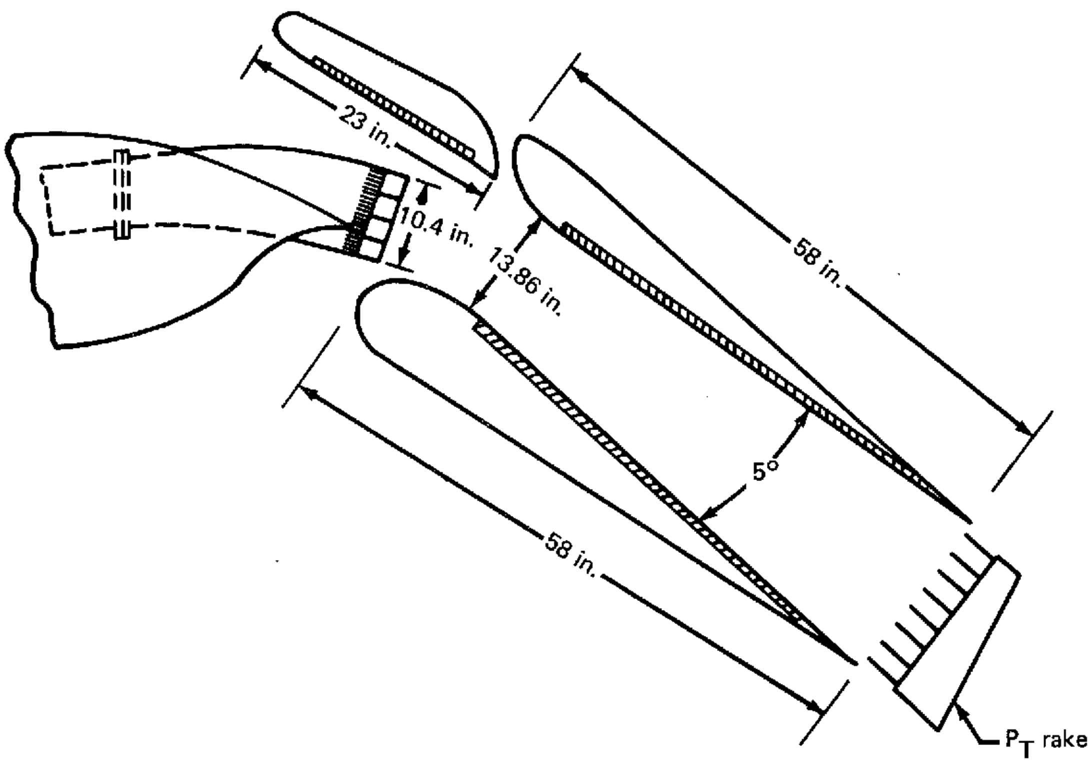


FIGURE 3.—TYPICAL EFFECT OF NOZZLE PRESSURE RATIO ON AUGMENTOR PERFORMANCE

a) GENERAL ARRANGEMENT (SPAN = 75 IN.)



b) BASIC GEOMETRY

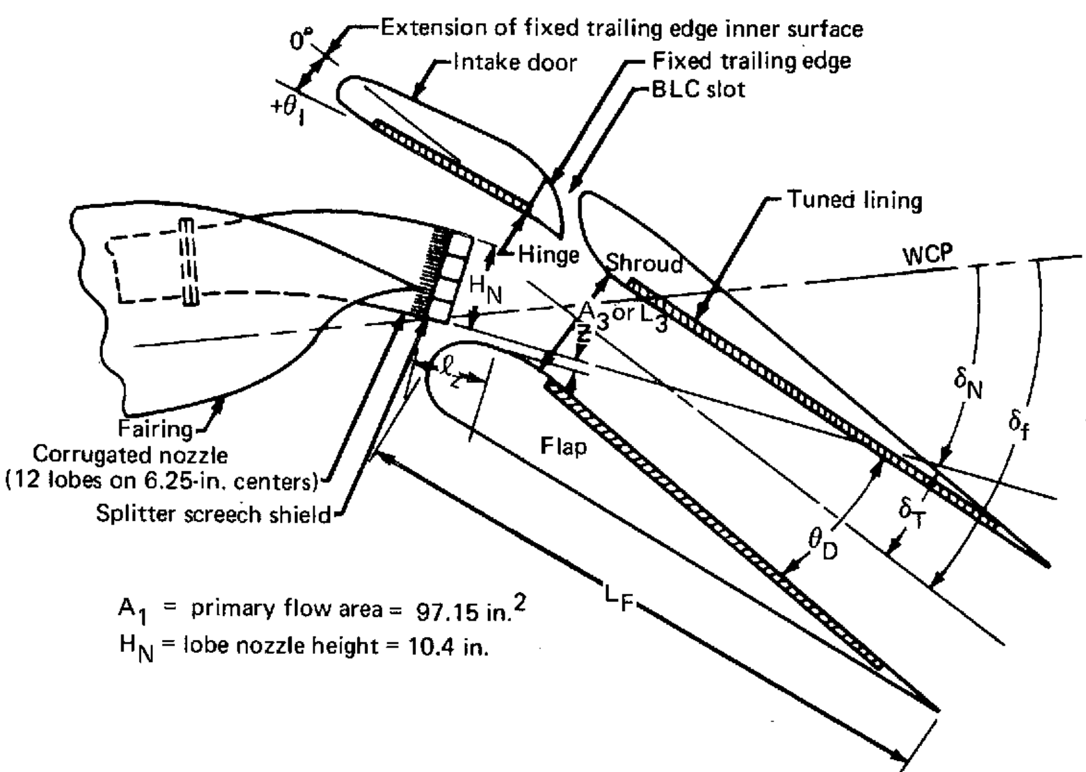


FIGURE 4.– AUGMENTOR

TABLE 1.—EXPECTED THRUST AUGMENTATION, ϕ

δ_F	ℓ_z	Z	θ_I	Thrust augmentation ϕ,	
Takeoff power setting and flaps					
20^o	14.6 in.	+1.76 in.	-20^o *	1.36 *	NPR = 2.6, intake off, no lining
20^o	14.6 in.	+1.76 in.	-20^o *	1.32 *	NPR = 2.6, intake off, with lining
Approach power setting and flaps					
65^o	14.6 in.	-1.172 in.	20^o	1.16	NPR = 1.6, intake on, no lining
65^o	14.6 in.	-1.172 in.	20^o	1.13	NPR = 1.6, intake on, with lining

* Subtract 0.05 from ϕ when intake door is installed

Optimized $\frac{A_3}{A_1} = 11$ (L_3 = 13.86 in.) Optimized $\theta_D = 5^o$

NOTE: These optimum conditions were found by extensive testing on smaller-scale augmentor wing models, results of which can be found in report CR-114534.

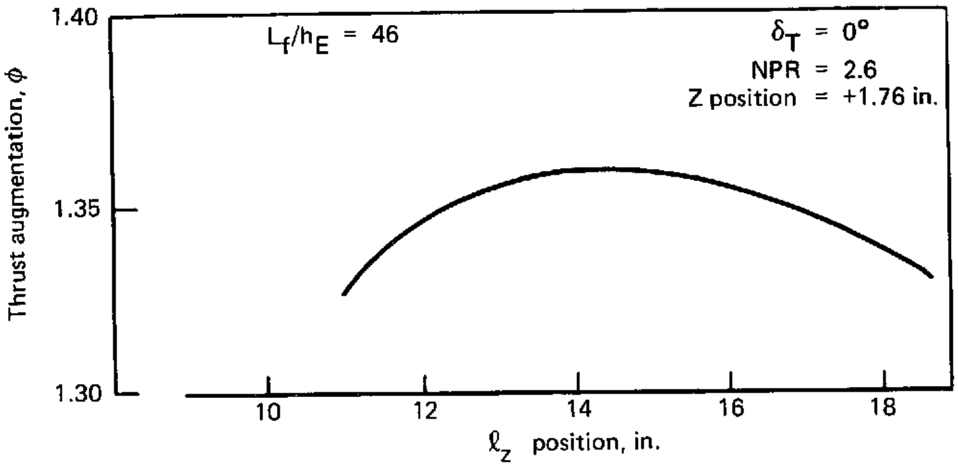


FIGURE 5.—THRUST AUGMENTATION ϕ AS A FUNCTION OF ℓ_2

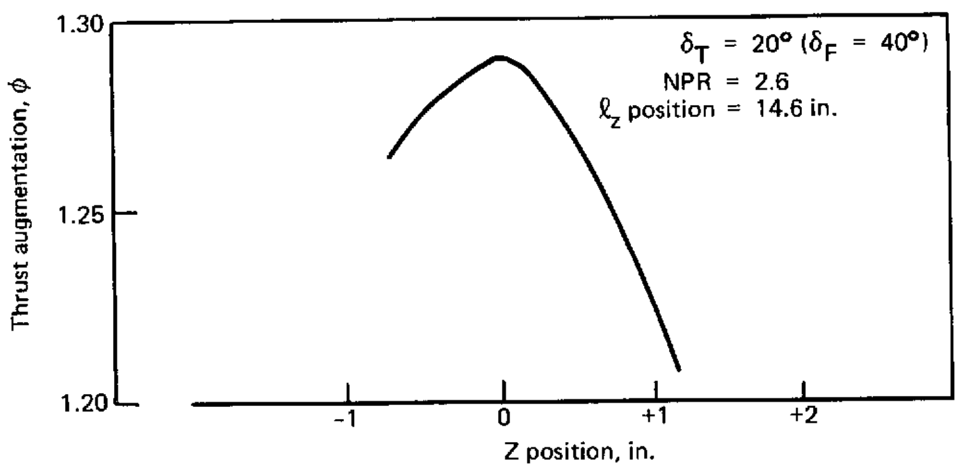


FIGURE 6.—THRUST AUGMENTATION ϕ AS A FUNCTION OF Z

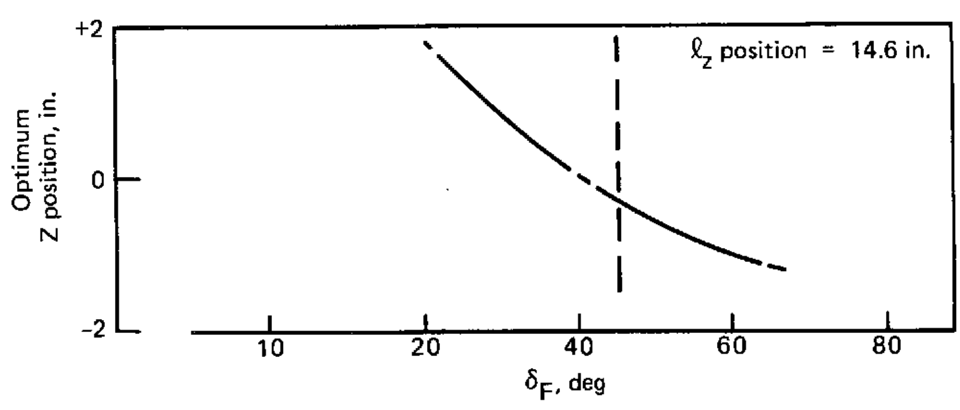


FIGURE 7.—TYPICAL SENSITIVITY OF Z POSITION ON δ_F

3.2 ACOUSTICS

The acoustic characteristics of the nozzle and the augmentor acoustic linings were designed to supplement each other. The nozzle design used was optimized during testing of smaller-scale augmentor wing models at Boeing. The results of this prior work are documented in CR-114534. The overall sound pressure level spectrum from this particular nozzle, in an unlined augmentor, will peak at from 2 to 3 kHz, and hence the lining design was optimized to provide maximum noise reduction at this frequency.

An acoustic lining is basically a transducer; i.e., the lining converts acoustic energy (motion) into heat. This is accomplished by placing an appropriate porous material in a location of high particle velocity. The core and impervious backing sheet of the lining act somewhat like a number of closed organ pipes. The particle velocity at the closed end is minimum while the pressure is maximum; at the open end, the conditions are reversed. When a material of the appropriate acoustic resistance is placed over the open side of the lining, a reduction in sound pressure level is observed over a broad band of frequencies. The acoustic resistance is not only a function of the material and construction procedure, but also a function of its environment; i.e., sound pressure level, temperature, and grazing Mach number of the jet flow.

To broaden the attenuation spectrum (and thus provide the maximum PNL reduction), two different linings tuned at two different frequency bands are required. These requirements can be met by using a "two-layer lining" (fig. 8). The septum sheet, inner core, and the backing sheet taken together form one lining, which is tuned to the high end of the spectrum. The second lining consists of the face sheet, both cores, and the backing sheet. The second lining is tuned at a lower frequency. This lining has been designed to provide 7 PNdB reduction referenced to an unlined augmentor level measured on a 500-ft sideline during takeoff.

The design of the lining represents Boeing's latest technology in lining design and is an improved design over those developed during the smaller-scale augmentor wing model tests mentioned above.

Table 2 shows the acoustical qualities in terms of Rayl numbers for the completed three sets of panels that were built for the flap system.

Set no. 2 is the design point set, while sets 1 and 3 have acoustic qualities covering an impedance band above and below that of the design point panels. The additional two sets of panels were constructed to allow testing at other than the design point.

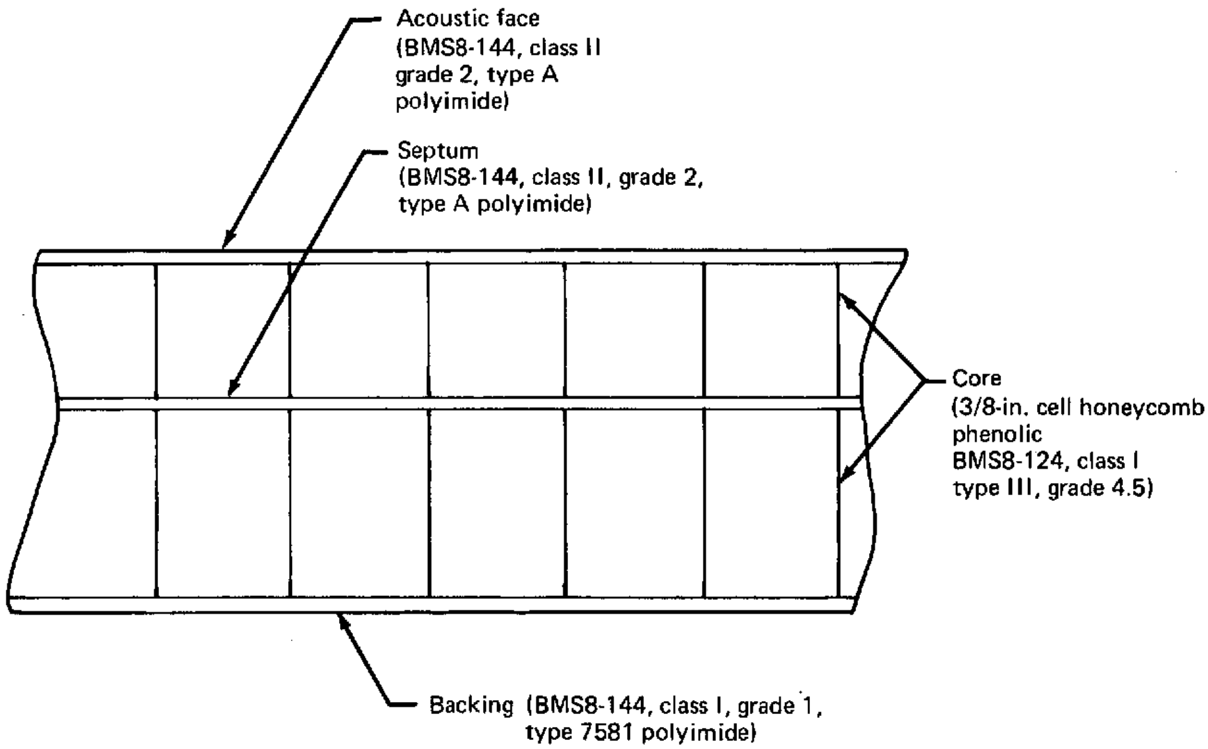


FIGURE 8.—ACOUSTIC PANEL CONSTRUCTION

TABLE 2.–ACOUSTIC PANEL RAYL SUMMARY RECORD

Panel serial no.	Ref only, face sheet Rayl at 132 c/s	Ref only, septum sheet, Rayl at 54 c/s	Design criteria, face/core, Rayl at 132 c/s	Design criteria, core/septum/core, Rayl at 54 c/s
	Drawing specifications (average, with upper and lower limits shown)			
	17 21.25 12.75	23 28.75 17.25	24 30 18	30 37.5 22.5
	Set 1			
-1, M09	19.2	13.5	26.3	28.8
, M010	9.9	11.7	12.5	22.2
, M011	14.1	13.2	19.3	24.8
, M012	14.3	12.0	18.3	28.0
, M013	14.9	12.7	19.5	33.9
, M014	13.7	14.0	18.1	27.7
, M015	13.0	14.0	18.0	39.0
, M016	14.9	12.3	18.7	29.3
-4, M03	15.6	17.0	18.2	32.6
, M04	13.6	21.8	17.5	42.7
	Drawing specifications (average, with upper and lower limits shown)			
	29 36.25 21.75	42 52.5 31.45	37 46.2 27.75	65 39
	Set 2			
-2, M01	28.3	29.6	36.1	67.2
, M02	19.2	23.0	36.6	39.3
, M03	24.2	20.9	30.6	41.6
, M04	20.5	28.1	30.1	57.8
, M05	22.4	21.5	33.6	51.2
, M06	19.7	20.7	27.6	54.7
, M07	20.6	21.1	32.8	50.4
, M08	26.0	21.8	43.6	61.4
-5, M01	17.4	26.7	25.7	45.1
, M02	17.8	23.9	27.4	46.6
	Drawing specifications (average, with upper and lower limits shown)			
	44 55 33	67 83.6 50.2	54 67.5 40.4	81 101.2 60.6
	Set 3			
-3, M01	44.90	57.6	51.7	103.2
, M02	32.15	34.3	55.2	89.5
, M03	40.3	58.5	45.3	103.5
, M04	31.5	60.4	45.2	92.0
, M05	33.1	55.6	43.2	87.2
, M06	36.1	49.9	45.7	93.0
, M07	24.2	27.2	58.4	97.0
, M08	29.1	44.2	46.0	92.2
-6, M01	35.1	41.2	41.0	62.0
, M02	36.6	41.8	44.5	70.0

In table 2, four sets of Rayl numbers are shown for each set of panels. The Rayl number for the face sheet and the septum sheet are reference figures only, while the actual design criteria is for the face sheet with the core bonded and for the septum sheet with both cores bonded. These design criteria Rayl numbers represent the impedance of the panel after the bonding operation. The reference Rayl numbers, prior to core bonding, are essential since the final core bonding technique can then be adjusted if necessary to give the desired impedance results after the core bonding operation.

The Rayl numbers shown for the face sheet, septum sheet, and face sheet/core bond are average numbers of four readings taken at random over the panel area. The Rayl number of the core/septum/core bond is an average of the readings obtained from two separate 4-in.-diameter test disks. These latter test specimens were taken from the excess edge material that was trimmed from each completed panel.

3.2.1 Acoustic Prediction Procedure

The free-field predicted PNdB, OASPL, and typical 1/3-octave spectrum for the indicated configurations are shown in figure 9. These values were derived using the following procedure: From the acoustic data recorded during the DNS program and reported in CR-114623 and CR-114534 appropriate spectra were abstracted. For the takeoff configuration of the suppressor, run 2100 was used, while run 2102 was used for approach. For the slot nozzle, run 1600 was chosen. (See tables 3, 4, 5, and 6 for the following steps.) Next, all four spectra were adjusted to atmospheric conditions of 77 F and 70% relative humidity.

For the slot nozzle configuration, a correction must be made to the model data to remove the "screech" prior to scaling this data to full scale. The reason for removing the screech is that the full-scale hardware has a splitter installed while the model did not. The philosophy used in designing the screech shield is derived from CR-114285, section 4.3.2. The "screech" was removed from the appropriate 1/3-octave bands by using an engineering estimate of the decrement required. The lobe nozzle/augmentor configuration does not require a screech correction but, as the L/h_E of the model was 55 and the full-scale L/h_E is 46, an investigation into the acoustic effect of changing this ratio was made.

The model test data includes a series of runs which vary the L/h_E ratio and hold all the remaining parameters constant. Investigation of these test results show that for a change in L/h_E from 55 to 46, the acoustic effect is negligible.

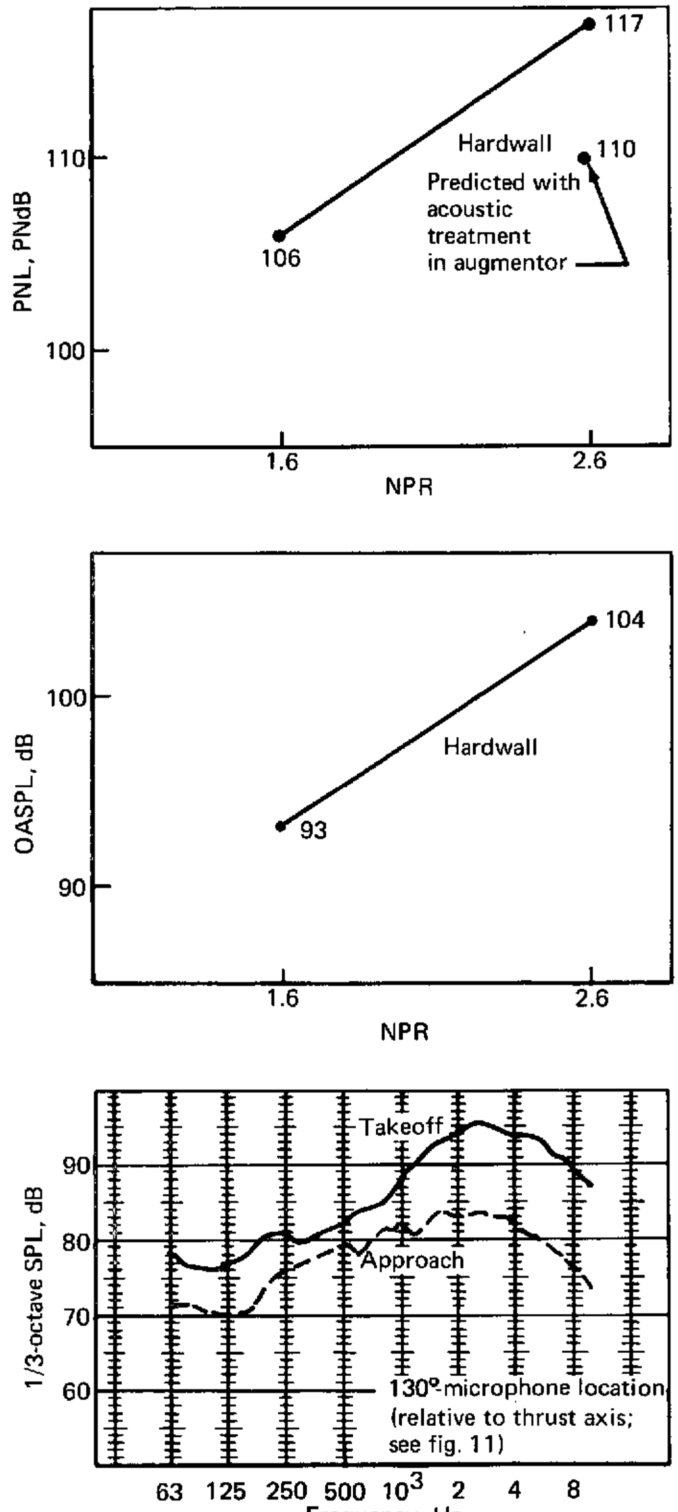


FIGURE 9.–PREDICTED ACOUSTIC VALUES AT 100-FT RADIUS SUPPRESSOR–CORRUGATED, LOBED NOZZLE

TABLE 3.—STEP-BY-STEP RESULTS OF ACOUSTIC PREDICTION PROCEDURE TAKEOFF POWER—SUPPRESSOR AT 130 MICROPHONE, $\delta_F = 20$, NPR = 2.6

Model scale (ref. CR-114623)						Full scale			
Band number (a)	SPL at 50-ft radius, dB re: 0.0002 microbar					Band number (a)	SPL at 150-ft radius, dB re: 0.0002 microbar		
	As measured (run 2100)	Correction for:					Scaled-up spectrum	Correction for:	
		Standard acoustic day (b)	Free field (-3 dB)	Span (-2.3 dB)	L/h_E (0 dB)			Velocity and density (-2.5)	100-ft radius
23	83.8	83.8	80.8	78.5	78.5	18	78.5	76.0	79.5
24	80.4	80.4	77.4	75.1	75.1	19	75.1	72.6	76.1
25	80.1	80.1	77.1	74.8	74.8	20	74.8	72.3	75.8
26	81.5	81.5	78.5	76.2	76.2	21	76.2	73.7	77.2
27	83.0	83.0	80.0	77.7	77.7	22	77.7	75.2	78.7
28	84.8	84.8	81.8	79.5	79.5	23	79.5	77.0	80.5
29	84.9	84.9	81.9	79.6	79.6	24	79.6	77.1	80.6
30	84.2	84.2	81.2	79.9	79.9	25	78.9	76.4	79.9
31	85.3	85.3	82.3	80.0	80.0	26	80.0	77.5	81.0
32	87.1	87.1	84.1	81.8	81.8	27	81.8	79.3	82.8
33	87.8	87.8	84.8	82.5	82.5	28	82.5	80.0	83.5
34	90.9	90.9	87.9	85.6	85.6	29	85.6	83.1	84.6
35	92.9	92.9	89.9	87.6	87.6	30	87.6	85.1	88.6
36	95.1	95.1	92.1	89.8	89.8	31	89.8	87.3	90.8
37	97.4	97.4	94.4	92.1	92.1	32	92.1	89.6	93.2
38	98.5	98.8	95.8	93.5	93.5	33	93.5	91.0	94.6
39	99.2	99.7	96.7	94.4	94.4	34	94.4	91.9	95.6
40	98.2	99.0	96.0	93.7	93.7	35	93.7	91.2	95.0
41	97.0	98.0	95.0	92.7	92.7	36	92.7	90.2	94.0
42	95.9	97.4	94.4	92.1	92.1	37	92.1	89.6	93.5
43	92.9	94.8	91.8	89.5	89.5	38	89.5	87.0	91.0
44	91.0	93.5	90.5	88.2	88.2	39	88.2	85.7	90.0
45	87.3	90.6	87.6	85.3	85.3	40	85.3	82.8	87.3

OASPL = 104
PNL = 117

[a]Band number = $10 \log_{10} f_c$

f_c = band center frequency

[b]Standard acoustic day = 77° F, 70% relative humidity.

TABLE 4.—STEP-BY-STEP RESULTS OF ACOUSTIC PREDICTION PROCEDURE APPROACH POWER—SUPPRESSOR AT 130 MICROPHONE, $\delta_F = 65$, NPR = 1.6

Model scale (ref. CR-114623)						Full scale			
	SPL at 50-ft radius, dB re: 0.0002 microbar						SPL at 150-ft radius,dB re:0.0002 microbar		
		Correction for:						Correction for:	
Band number (a)	As measured (run 2102)	Standard acoustic day (b)	Free field (-3 dB)	Span (-2.3 dB)	L/h_E (0 dB)	Band number (a)	Scaled-up spectrum	Velocity and density (-2.5)	100-ft radius
23	75.6	75.6	72.6	70.3	70.3	18	70.3	67.8	71.3
24	70.0	70.0	67.0	64.7	64.7	19	64.7	62.2	65.7
25	74.4	74.4	71.4	69.1	69.1	20	69.1	66.6	70.1
26	74.3	74.3	71.3	69.0	69.0	21	69.0	66.5	70.0
27	74.3	74.3	71.3	69.0	69.0	22	69.0	66.5	70.0
28	78.4	78.4	75.4	73.1	73.1	23	73.1	70.6	74.1
29	80.0	80.0	77.0	74.7	74.7	24	74.7	72.2	75.7
30	81.3	81.3	78.3	76.0	76.0	25	76.0	73.5	77.0
31	81.6	81.6	78.6	76.3	76.3	26	76.3	73.8	77.3
32	83.9	83.9	80.9	78.6	78.6	27	78.6	76.1	79.6
33	83.1	83.1	80.1	77.8	77.8	28	77.8	75.3	78.8
34	85.7	85.7	82.7	80.4	80.4	29	80.4	77.9	81.4
35	85.0	85.0	82.0	79.7	79.7	30	79.7	77.2	81.7
36	84.9	84.9	81.9	79.6	79.6	31	79.6	77.1	80.6
37	87.2	87.2	84.2	82.9	82.9	32	82.9	80.4	84.0
38	87.3	87.6	84.6	82.3	82.3	33	82.3	79.8	83.4
39	87.2	87.7	84.7	82.4	82.4	34	82.4	79.9	83.6
40	86.3	87.1	84.1	81.8	81.8	35	81.8	79.3	83.1
41	84.8	85.8	82.8	80.5	80.5	36	80.5	78.0	81.8
42	82.9	84.5	81.5	79.2	79.2	37	79.2	76.7	80.6
43	80.2	82.2	79.2	76.9	76.9	38	76.9	74.4	78.4
44	78.0	80.6	77.6	75.3	75.3	39	75.3	72.8	77.1
45	73.6	77.1	74.1	71.8	71.8	40	71.8	69.3	73.8

OASPL = 93
PNL = 106

[a]Band number = $10 \log_{10} f_c$
f_c = band center frequency

[b]Standard acoustic day = 77° F, 70% relative humidity.

TABLE 5.—STEP-BY-STEP RESULTS OF ACOUSTIC PREDICTION PROCEDURE TAKEOFF POWER—SLOT NOZZLE ONLY AT 130 MICROPHONE, NPR = 2.6

Model scale (ref. CR-114534)						Full scale			
	SPL at 50-ft radius, dB re: 0.0002 microbar						SPL at 150-ft radius, dB re: 0.0002 microbar		
		Correction for:						Correction for:	
Band number (a)	As measured (run 1600)	Standard acoustic day (b)	Free field (-3 dB)	Span (-2.3 dB)	Screech	Band number (a)	Scaled-up spectrum	Velocity and density (-4.0)	100-ft radius
23	91.0	91.0	88.0	85.7	85.7	18	85.7	81.7	85.2
24	82.7	82.7	79.7	77.4	77.4	19	77.4	73.4	76.9
25	87.6	87.6	84.6	82.3	82.3	20	82.3	78.3	81.8
26	88.0	88.0	85.0	82.7	82.7	21	82.7	78.7	82.2
27	89.3	89.3	86.3	84.0	84.0	22	84.0	80.0	83.5
28	92.0	92.0	89.7	86.7	86.7	23	86.7	82.7	86.2
29	94.2	94.2	91.2	88.9	88.9	24	88.9	84.9	88.4
30	96.5	96.5	93.5	91.2	91.2	25	91.2	87.2	90.7
31	98.5	98.5	95.5	93.2	93.2	26	93.2	89.2	92.7
32	100.8	100.8	97.8	95.5	95.5	27	95.5	91.5	95.0
33	104.2	104.2	101.2	98.9	97.4	28	97.4	93.4	96.9
34	107.4	107.4	104.4	102.1	99.6	29	99.6	95.6	99.1
35	110.5	110.5	107.5	105.2	101.7	30	101.7	97.7	101.2
36	112.0	112.0	109.0	106.7	101.7	31	101.7	97.7	101.2
37	111.9	111.9	108.9	106.6	103.6	32	103.6	99.6	103.2
38	109.2	109.2	106.2	103.9	103.9	33	103.9	99.9	103.5
39	111.3	111.3	108.3	106.0	102.5	34	102.9	98.9	102.6
40	108.2	108.2	105.2	102.9	102.4	35	102.4	98.4	102.2
41	106.9	106.9	103.9	101.6	101.6	36	101.6	97.6	101.4
42	104.9	104.9	101.9	99.6	99.6	37	99.6	95.6	99.5
43	102.4	102.4	99.4	97.1	97.1	38	97.1	93.1	97.1
44	99.4	99.4	96.4	94.1	94.1	39	94.1	90.1	94.4
45	97.2	97.2	94.2	91.9	91.9	40	91.9	87.9	92.4

OASPL = 112
PNL = 125

[a]Band number = $10 \log_{10} f_c$
f_c = band center frequency

[b]Standard acoustic day = 77° F, 70% relative humidity.

TABLE 6.—STEP-BY-STEP RESULTS OF ACOUSTIC PREDICTION PROCEDURE
APPROACH POWER—SLOT NOZZLE ONLY AT 130 MICROPHONE, NPR = 1.6

Model scale (ref. CR-114534)						Full scale			
Band number (a)	SPL at 50-ft radius, dB re: 0.0002 microbar					Band number (a)	SPL at 150-ft radius, dB re: 0.0002 microbar		
	As measured (run 1600)	Correction for:					Scaled-up spectrum	Correction for:	
		Standard acoustic day (b)	Free field (-3 dB)	Span (-2.3 dB)	Screech			Velocity and density (-4.0)	100-ft radius
23	74.2	74.2	71.2	68.9	68.9	18	68.9	64.9	68.4
24	72.7	72.7	69.7	67.4	67.4	19	67.4	63.4	66.9
25	75.6	75.7	72.6	70.3	70.3	20	70.3	66.3	69.8
26	78.2	78.2	75.2	72.9	72.9	21	72.9	68.9	72.4
27	79.6	79.6	76.6	74.3	74.3	22	74.3	70.3	73.8
28	82.5	82.5	79.5	77.2	77.2	23	77.2	73.2	76.7
29	84.0	84.0	81.0	78.7	78.7	24	78.7	74.7	78.2
30	86.0	86.0	83.0	80.7	80.7	25	80.7	76.7	80.2
31	87.5	87.5	84.5	82.2	82.2	26	82.2	78.2	81.7
32	89.3	89.3	86.3	84.0	84.0	27	84.0	80.0	83.5
33	90.7	90.7	87.7	85.4	85.4	28	85.4	81.4	84.9
34	92.7	92.7	89.7	87.4	87.4	29	87.4	83.4	86.9
35	93.5	93.5	90.5	88.2	88.2	30	88.2	84.2	87.7
36	94.3	94.3	91.3	89.0	89.0	31	89.0	85.0	88.5
37	95.4	95.4	92.4	90.1	90.1	32	90.1	86.1	89.7
38	95.2	95.2	92.2	89.9	89.9	33	89.9	88.9	92.5
39	94.8	94.8	91.8	89.5	89.5	34	89.5	84.5	88.2
40	93.5	93.5	90.5	88.2	88.2	35	88.2	84.2	88.0
41	91.6	91.6	89.6	86.3	86.3	36	86.3	82.3	86.1
42	88.9	88.9	85.9	83.6	83.6	37	83.6	79.6	83.5
43	86.2	86.2	83.2	80.9	80.9	38	80.9	76.9	80.9
44	81.9	81.9	79.9	76.6	76.6	39	76.6	72.6	76.9
45	78.0	78.0	75.0	72.7	72.7	40	72.7	68.7	73.2

OASPL = 99
PNL = 112

[a]Band number = $10 \log_{10} f_c$
f_c = band center frequency

[b]Standard acoustic day = 77° F, 70% relative humidity.

At this point, the data is scaled up by a factor of 3. The data is scaled simply by changing the band designation for each 1/3 octave. As the model data were measured at 50-ft radius, a scaleup of 3 now makes the distance 150 ft.

When scaling the model data up by 3, it is assumed that all dimensions of the nozzle increase uniformly and, therefore, straight increase to a factor of 3 can be used. However, in this particular hardware configuration, a change occurs in the wingspan dimension, making it necessary to add a correction of

$$-10 \log \frac{\text{Span (model) x scale factor x } h_E \text{ (model) x scale factor}}{\text{Span (full scale) x } h_E \text{ (full scale)}}$$

which reduces to a correction of -3.2 dB.

A correction must be made to the data to compensate for the fact that the DNS model was tested at 300° F total temperature, and the full-scale rig will be operated at ambient temperatures. This correction incorporates a density and velocity term while holding the nozzle pressure ratio constant. For a round convergent nozzle, the hot-to-cold correction would be -4.5 dB; however, measurements of a 100/1 slot nozzle (runs 255 and 260) with a long lip (screech suppressed) indicate a correction of only -4.0 dB. In the case of a suppressor in an augmentor, measurements (runs 284 and 386) show that a factor of only -2.5 dB is required.

Next, the 1/3-octave sound pressure levels were moved from 150-ft radius to 100 ft by using a correction of 20 log 150/100 plus atmospheric corrections. This correction brings the data to the location of the microphone array recommended for the NASA Lewis program.

The final columns in tables 3, 4, 5, and 6 show the predicted free-field spectra: OASPL and PNL for both the reference slot nozzle and suppressor at 100-ft sideline.

If a ground microphone test setup is used (see recommended microphone placement) during the full-scale tests, a 6-dB increase in all the 1/3-octave bands will occur. If a high microphone test setup is used, a ground reflection/cancellation factor will be required, and the procedures outlined in *Use of Ground Level Microphones to Acquire Static Free Field Data*** are recommended.

**McKaig, M. B.: D6-40330, The Boeing Company, January 19 1972.

Acoustic data for additional microphone angles can be derived by referring to the microfiche data transmittal and applying the correction factor shown in tables 3 through 6. (All acoustic and propulsion data, including test logs for the NASA test reported in CR-114534, were supplied to NASA, along with the report.)

3.2.2 Recommended Microphone Placement

Normally, for a full-scale test such as this, a 200-ft polar microphone array would be used. However, due to the size of the hard-surfaced acoustic area at NASA Lewis, the use of a 100-ft polar microphone array is recommended. This array is shown in figure 10. If the exhaust exit centerline is maintained at the location shown in the figure, an array of six microphones on 10° spacing would be feasible for the minimum case. Of course, more microphone locations could be installed, say at 5° increments, but we recommend not going to a greater angle than 150° nor to any angle less than 100°. The lesser angle location is close to the edge of the hard-surfaced area and close to airpipes; therefore, we feel it is the furthest forward location which would give reliable information.

If acoustic information over a larger number of angles is required, we recommend rotating the exhaust centerline 60° clockwise and using the 100-ft polar array shown in figure 11.

The microphone height for figures 10 and 11 would be at the centerline of the exhaust slot with the microphone diaphragm parallel to the ground surface. If the surface in the arena is hard and very smooth, we would recommend using ground-mounted microphones, as shown in figure 12, in conjunction with the high microphones. These ground microphones would be located approximately 5 ft in front of the high microphones (95-ft polar radius).

Figure 13 shows a 20-microphone array which could be used to record near-field data, which in turn can be plotted on a grid and a more accurate location of the octave band sound sources established relative to the exhaust exit plane. With the sources established, extrapolation of the data could be accomplished by measuring on a 100-ft polar arc from the actual sound source, instead of from an assumed point source.

Finally, we recommend covering all the microphone connector boxes in the acoustic area with 4-in. thick fiberglass to prevent feedback reflection effects.

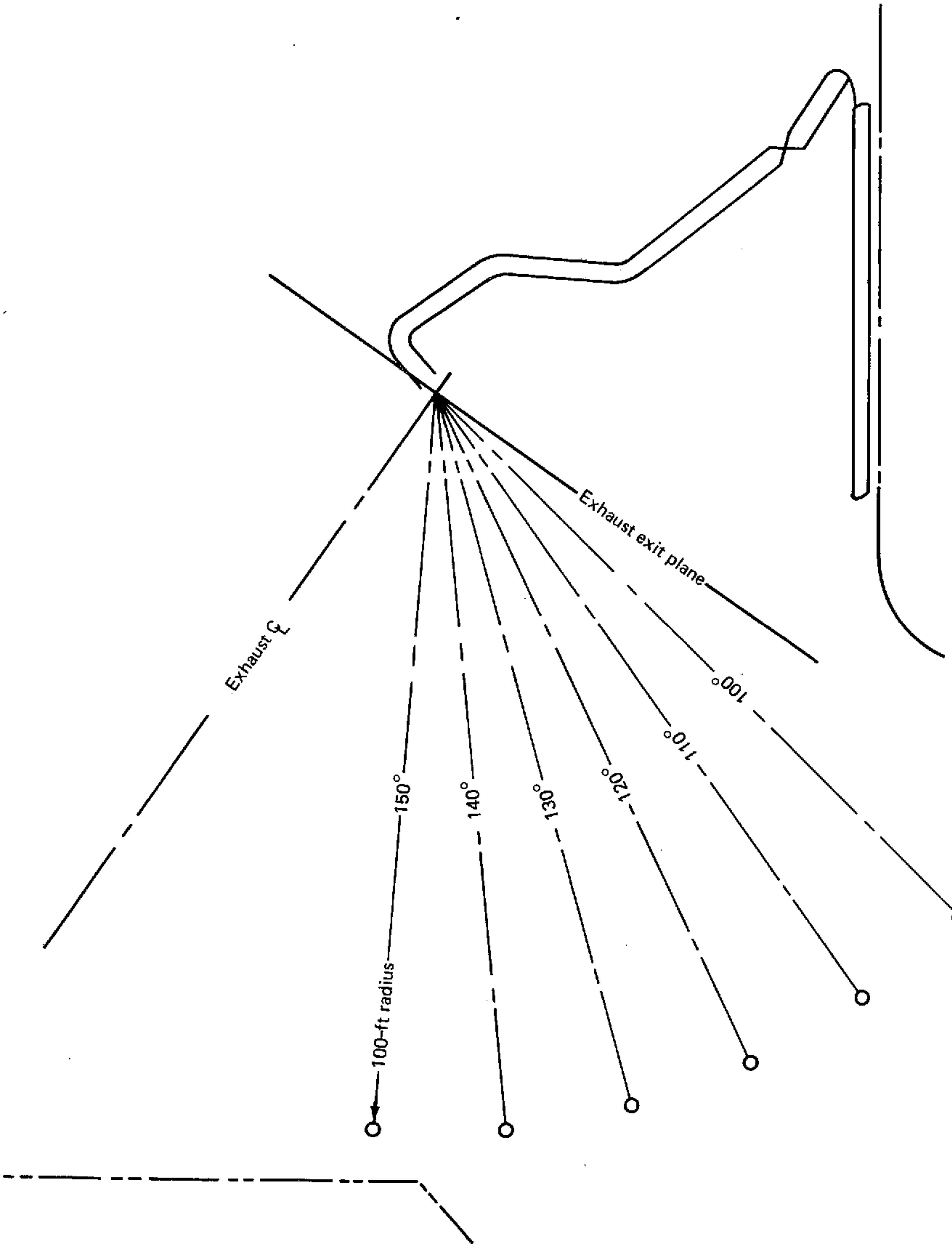


FIGURE 10.—MICROPHONE LAYOUT PER EXISTING EXHAUST CENTERLINE

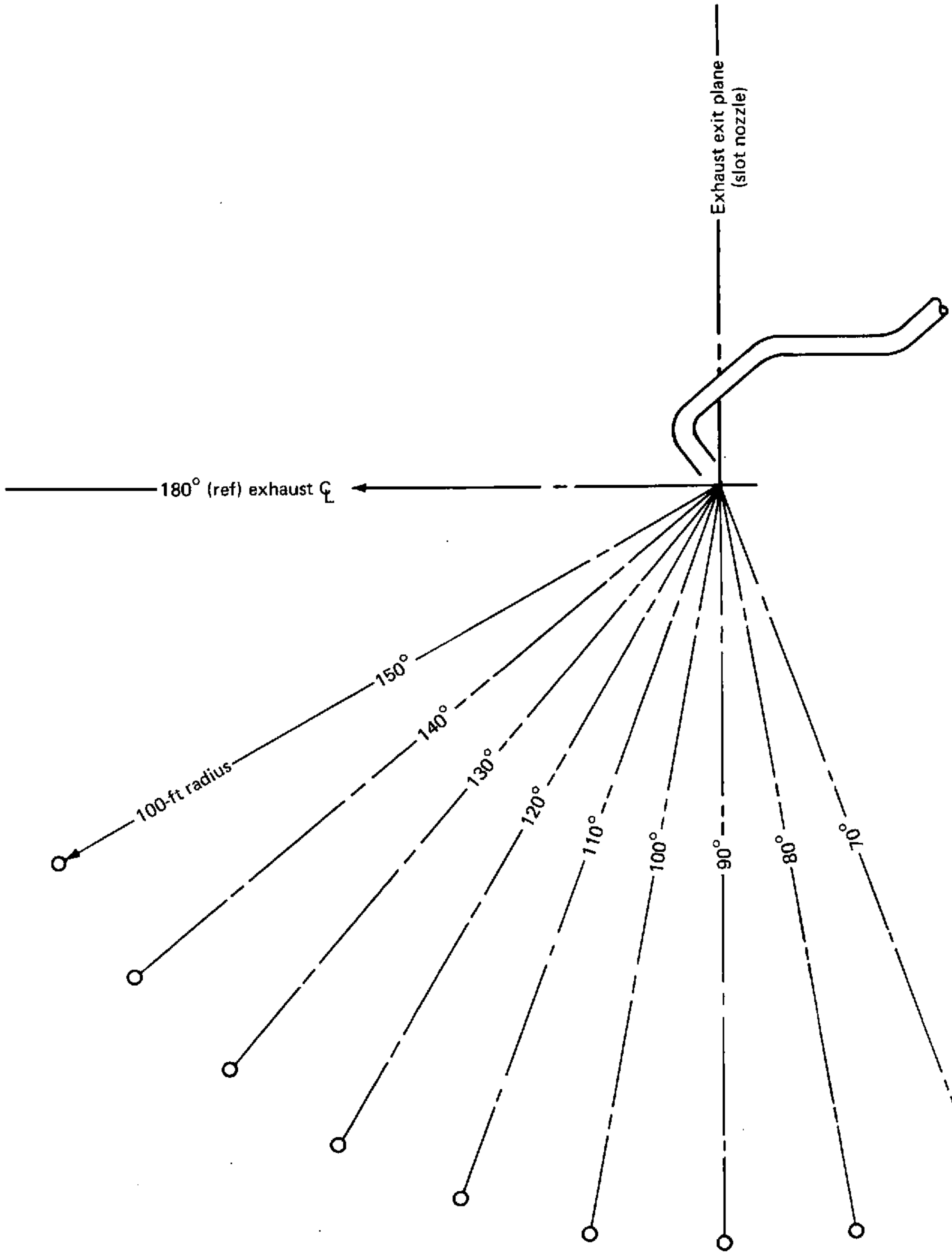


FIGURE 11.—MICROPHONE LAYOUT WITH 60°-CLOCKWISE ROTATION OF EXHAUST CENTERLINE

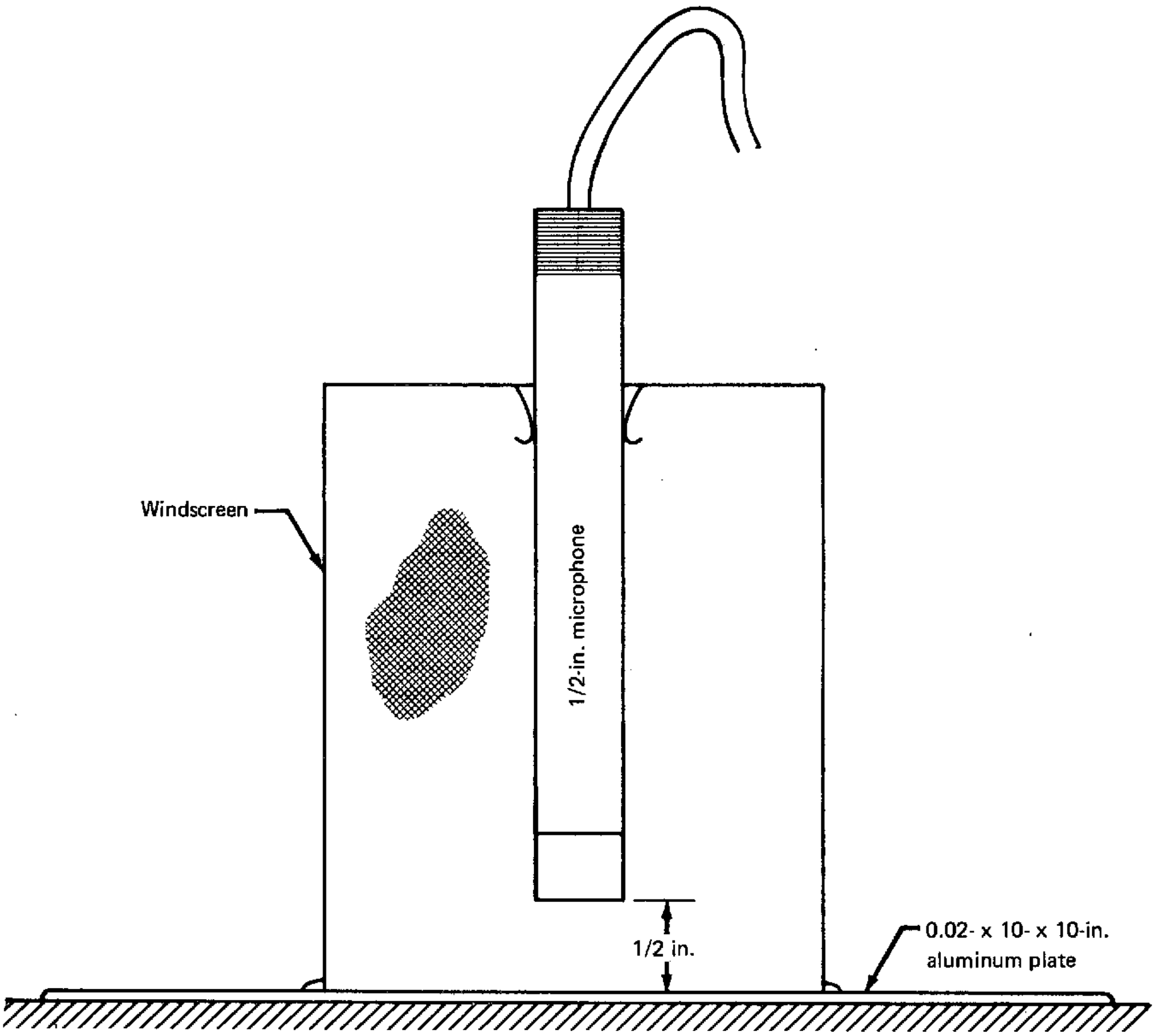


FIGURE 12.–GROUND LEVEL MICROPHONE INSTALLATION

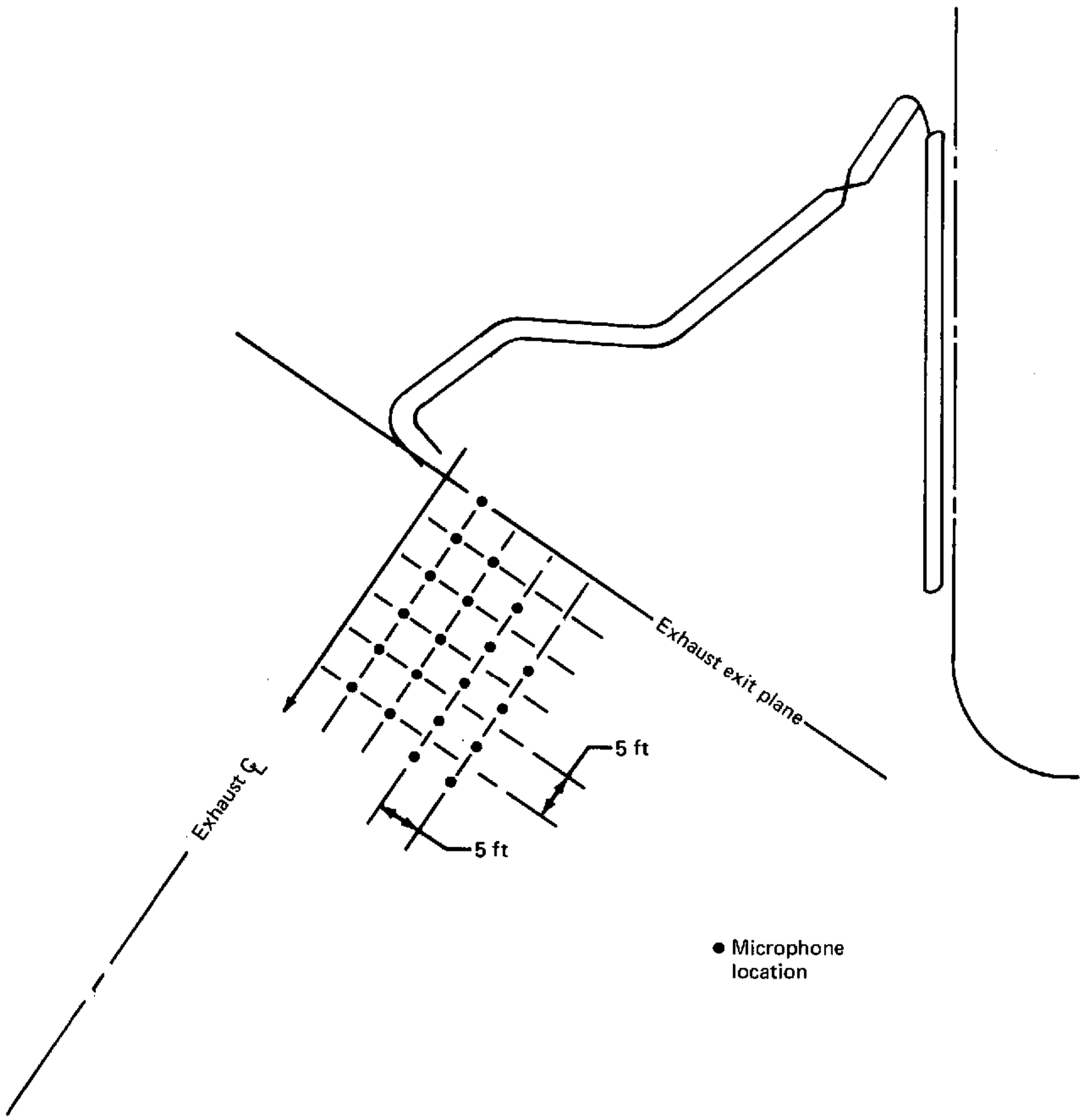


FIGURE 13.–20-MICROPHONE GRID LAYOUT

3.3 STRUCTURE INTEGRITY

System operating loads at full mass flows were computed and are shown in figures 14 and 15.

With one exception, the stress analysis contained in appendix A of this report shows all components of the augmentor wing model and mounting structure to meet or exceed the contract-specified structural requirement for factors of safety of three to the yield strength of the material. The single exception is the corrugated nozzle duct wall, with a calculated safety factor to yield of 1.46. The analysis used to size the duct wall is simple but very conservative as verified by previous experience. A similar nozzle built and successfully tested under NASA-Ames contract NAS2-6344 was less conservatively designed than the nozzle built for this contract.

In addition to the design analysis, the following pressure vessels were pressure-proof-tested to 90-psi gage:

a)	Baffled plenum tank	P/N 5461-40-1
b)	Nozzle feed plenum	P/N 5461-3-1
c)	Nozzle extension	P/N 5461-4-1

This structural verification, at three times the design pressure differential, was completed and certified by Boeing Quality Control.

3.4 FACILITIES INTERFACE

The augmentor wing model test system described in section 4.0 in this report was designed to interface with the existing NASA-Lewis augmentor wing support stand and with the air supply ducting that feeds the stand.

The following data was given in the general specifications of the contract.

a)	Maximum airflow rate	105 lb/sec
b)	Pressure ratio	3:1

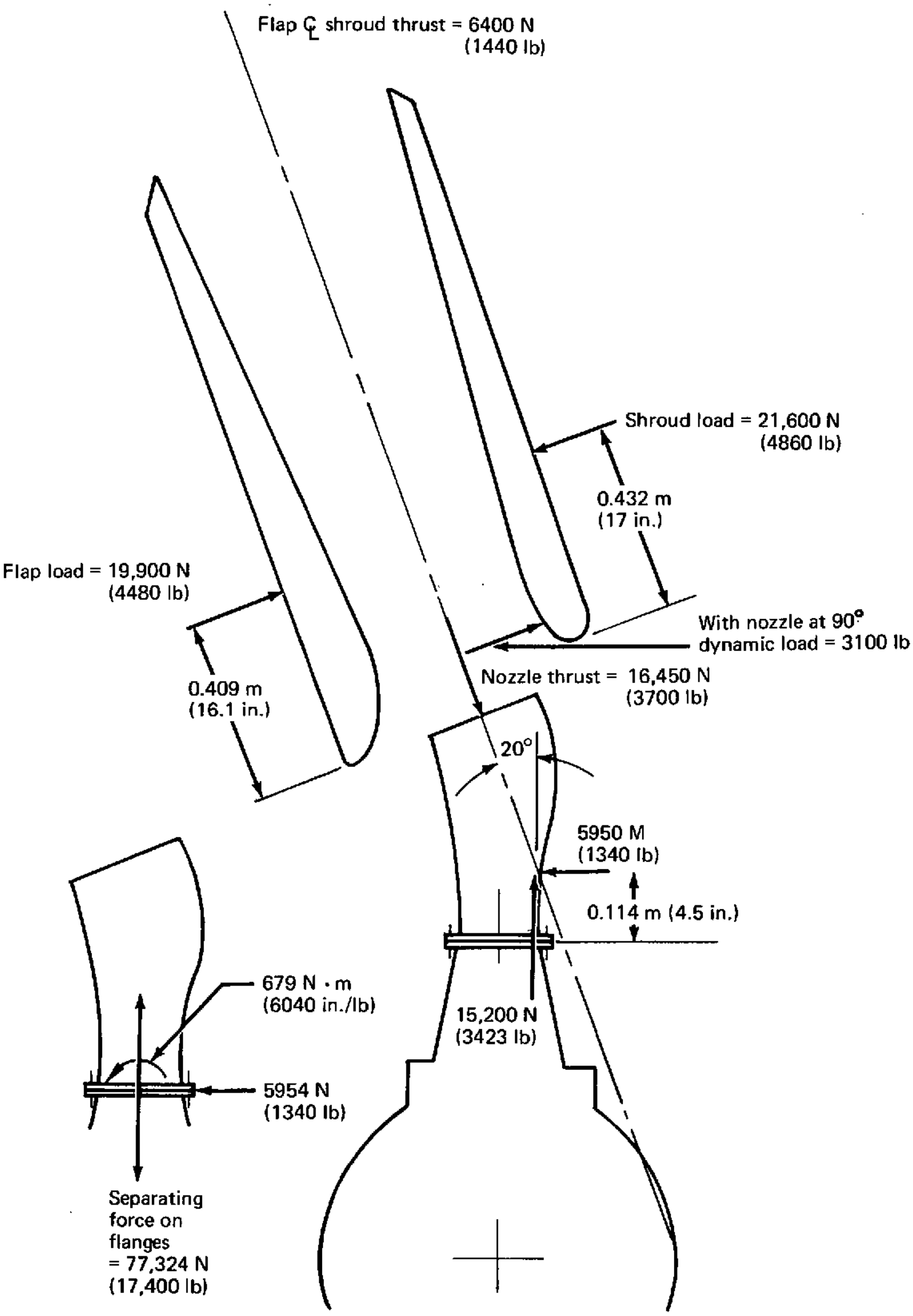


FIGURE 14.–OPERATING LOADS (FLAP SYSTEM AND NOZZLE)

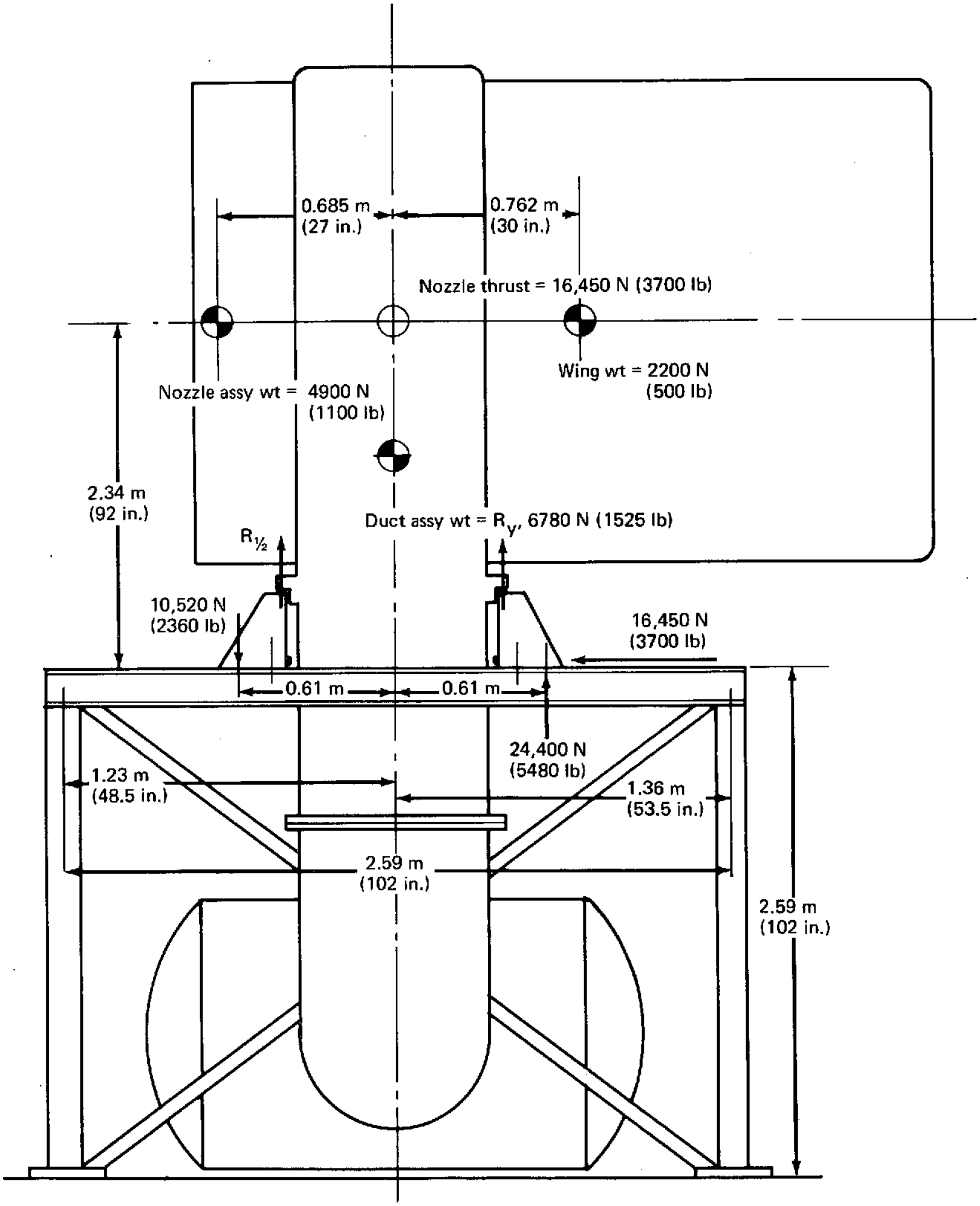


FIGURE 15.—SYSTEM OPERATING LOADS

c) Air temperature ambient to 350°F

d) Supply duct size 30-in. diameter

NASA-Lewis Research Center also provided the following drawings of the existing government support stand on which the model will be mounted.

a) CR-651408–Support Stand

b) SKAW-1 (revision of 7-18-73), Piping Layout

c) SKAW-2 (revision of 7-18-73), Piping Layout

d) SKAW-3 (revision of 7-18-73, Piping Layout

With the above data supplied, and combining this with data obtained from previous Boeing studies and tests on a smaller scale augmentor wing test system, a model scale was chosen to represent a full-scale wing and flap chord with a 75-in. span. This corresponds generally to the augmentor wing type aircraft developed and shown in CR-114534.

3.5 DESIGN REVIEW

Upon completing the design analysis and preliminary layout drawings, a design review was held at the NASA-Lewis Research Center. Design approval was obtained from the NASA technical monitor and authorization given to proceed with the detail design and procurement of long-lead hardware items.

4.0 DESIGN

The augmentor wing model that was built is illustrated in figure 16 and described in the following paragraphs. Complete sets of Boeing model 5461 shop fabrication and assembly drawings are on file at the NASA-Lewis Research Center. All of these detail and assembly drawings were approved by the NASA technical monitor prior to their release for hardware manufacture.

4.1 NOZZLE FEED PLENUM

The nozzle feed plenum assembly (fig. 17) consists of two major sections: the feed plenum and the nozzle extension. The assembly was sized to produce extremely low Mach number conditions and to provide uniform flow over a 75-in. span section that feeds the nozzle assembly. Internal plenum design was such to minimize any noise generation while maintaining design and fabrication simplicity.

4.2 NOZZLES

Two nozzles (also shown in fig. 17) were built. One is a simple rectangular slot nozzle; the other is a multielement breakup design of 12 lobes with a corrugated exit that corresponds to the optimum design developed and documented in CR-114534. Photos (fig. 18) show the corrugated nozzle installed with the flap assembly.

4.3 FLAP SYSTEM

The flap system (figs. 19 and 20) has three main components: flap, shroud, and intake. They are also of 75-in. span and have removable polyimide honeycomb acoustic linings. Three sets of these linings were built to configuration specifications provided by Boeing. Reversability of one set provides hardwall linings, and the acoustical qualities of these linings are shown in table 2.

As was mentioned in section 3.2, the Rayl numbers of the face sheet and septum sheet are reference Rayls only but are essential inasmuch that the bonding operation can be adjusted to give more accurate Rayl numbers of a completely bonded panel; i.e., if a low reference Rayl is found to exist on the face sheet alone, a heavier coating of adhesive can be used than would normally be used

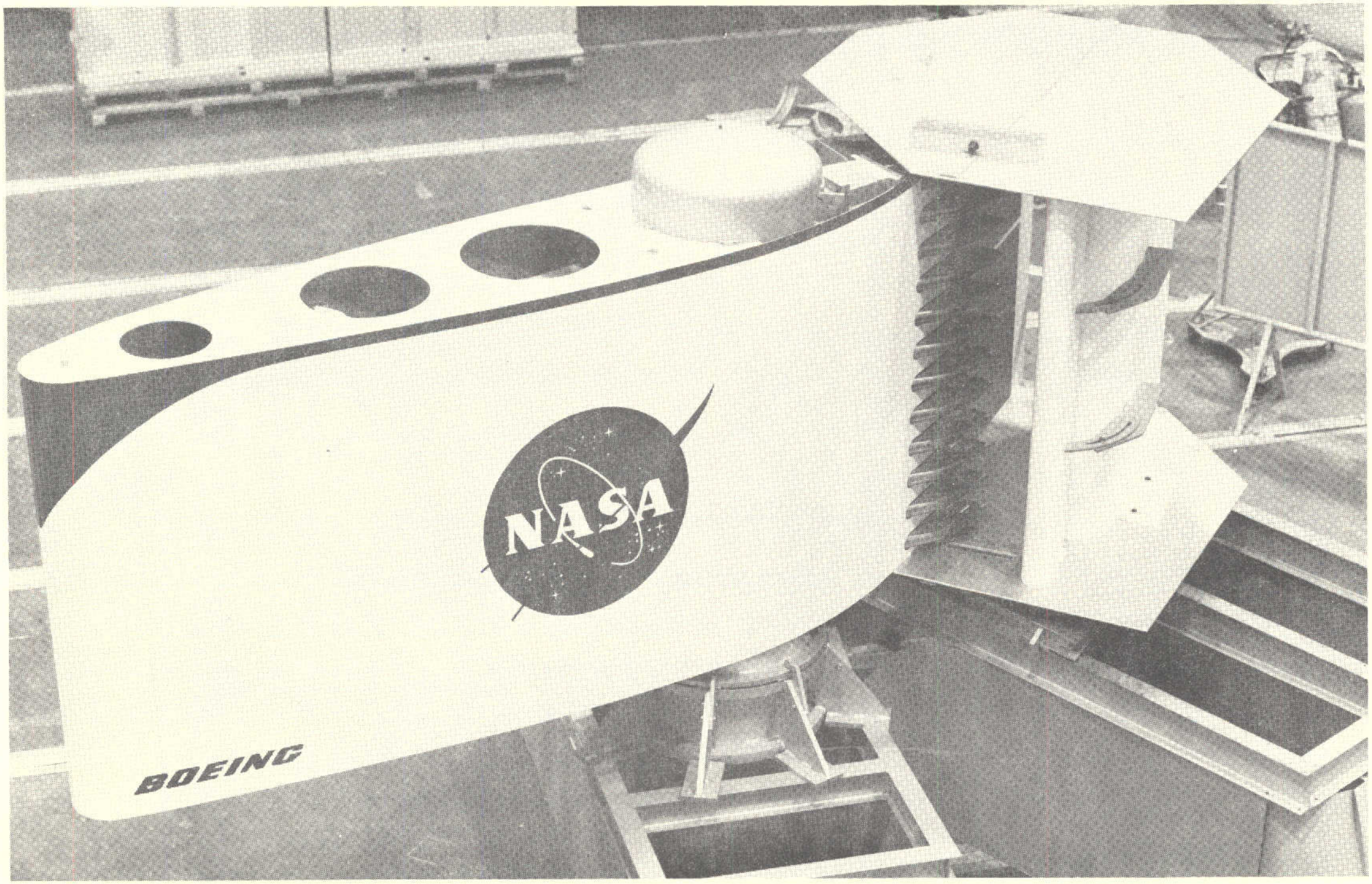


FIGURE 16.—AUGMENTOR WING MODEL FOR ACOUSTIC TESTS

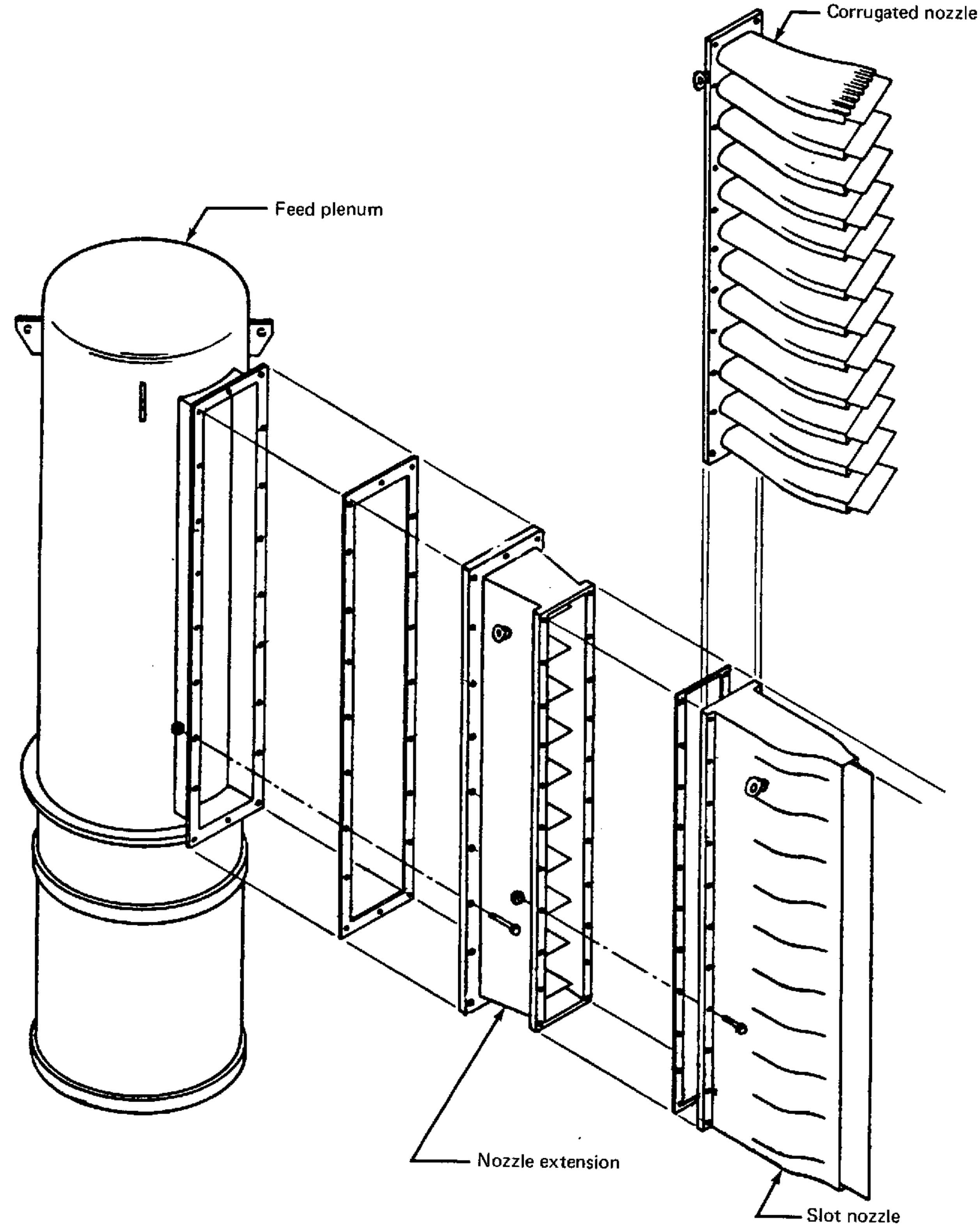


FIGURE 17.–NOZZLE FEED PLENUM AND NOZZLE ASSEMBLY

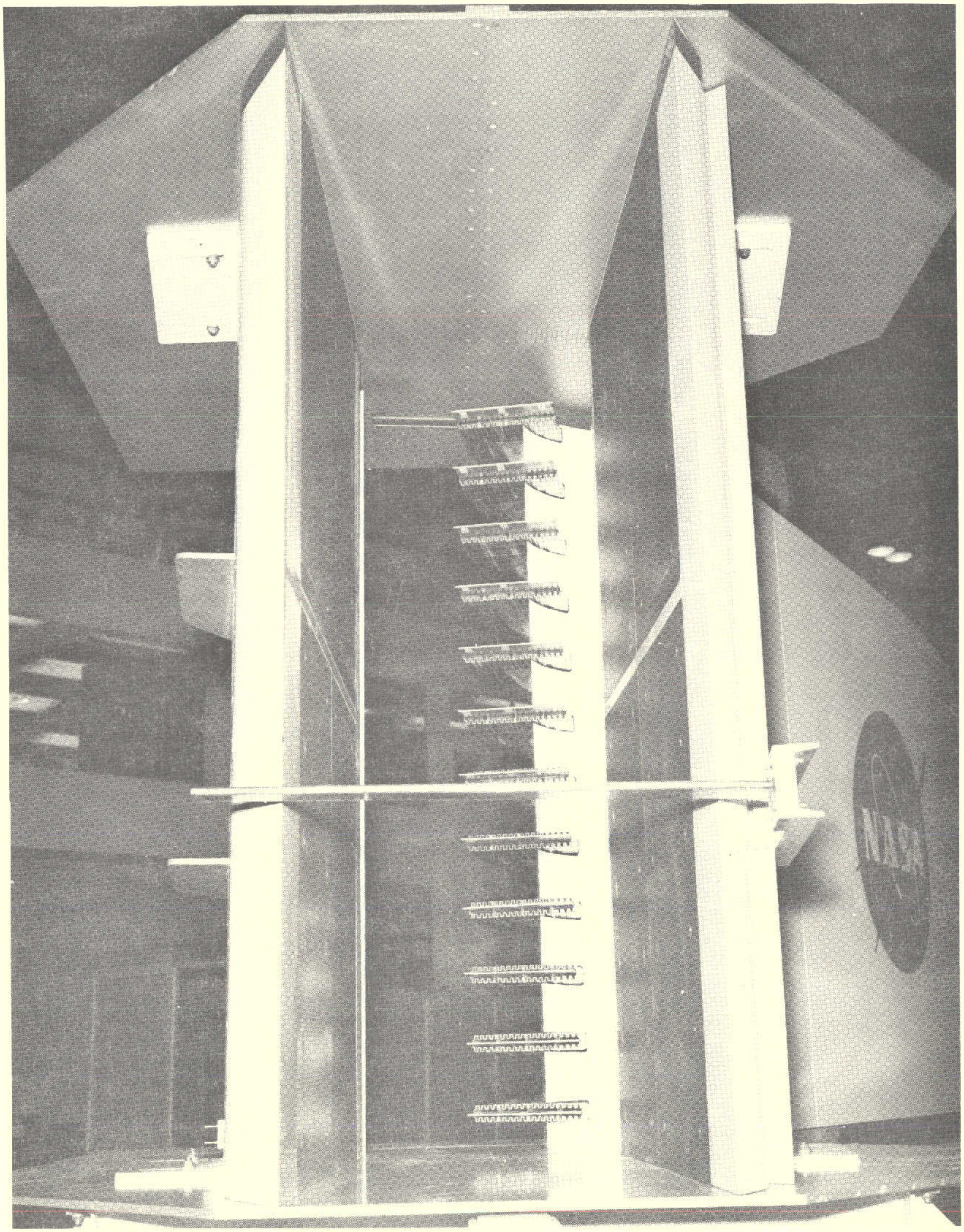


FIGURE 18.–CORRUGATED NOZZLE INSTALLED

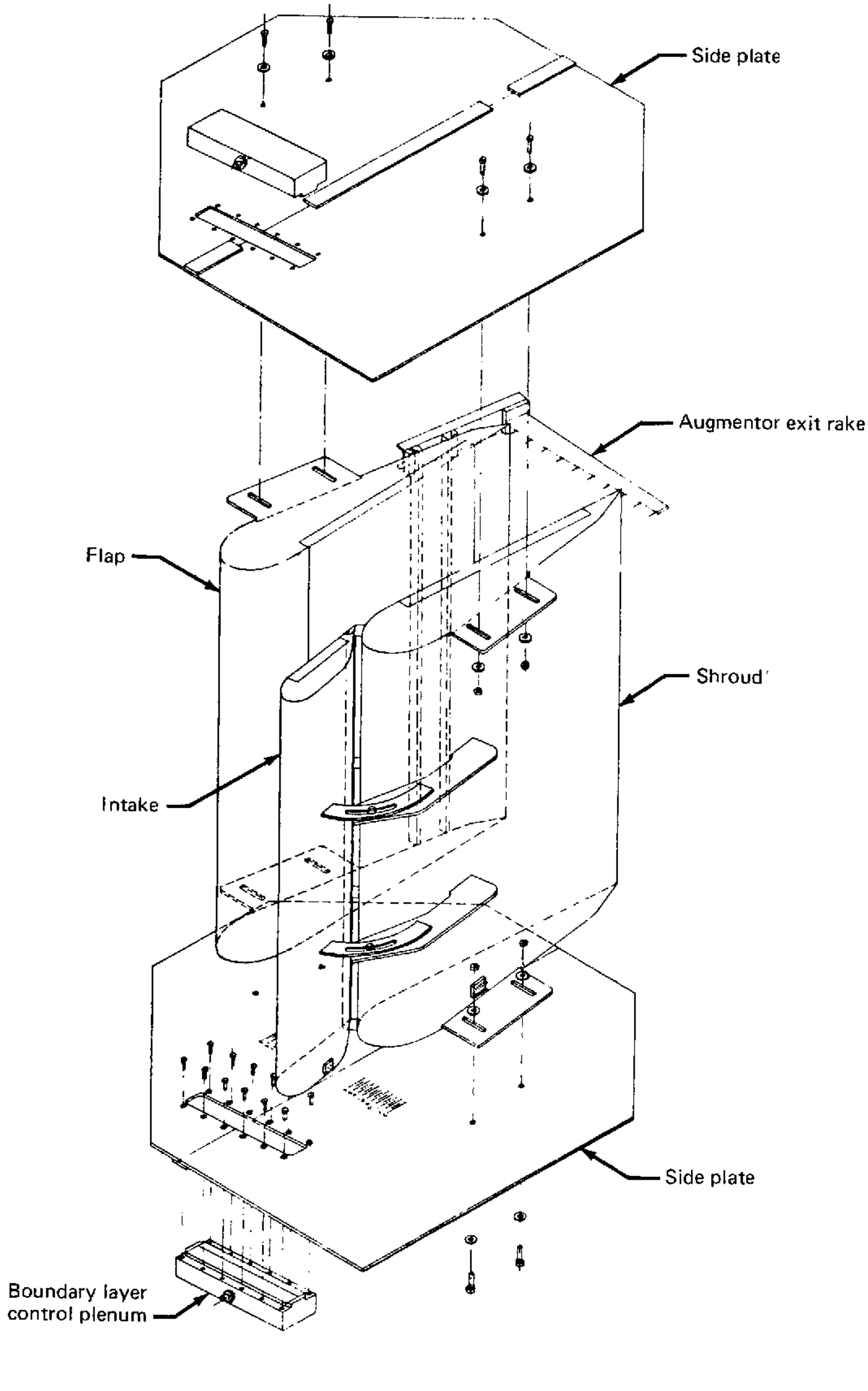


FIGURE 19.–FLAP SYSTEM ASSEMBLY

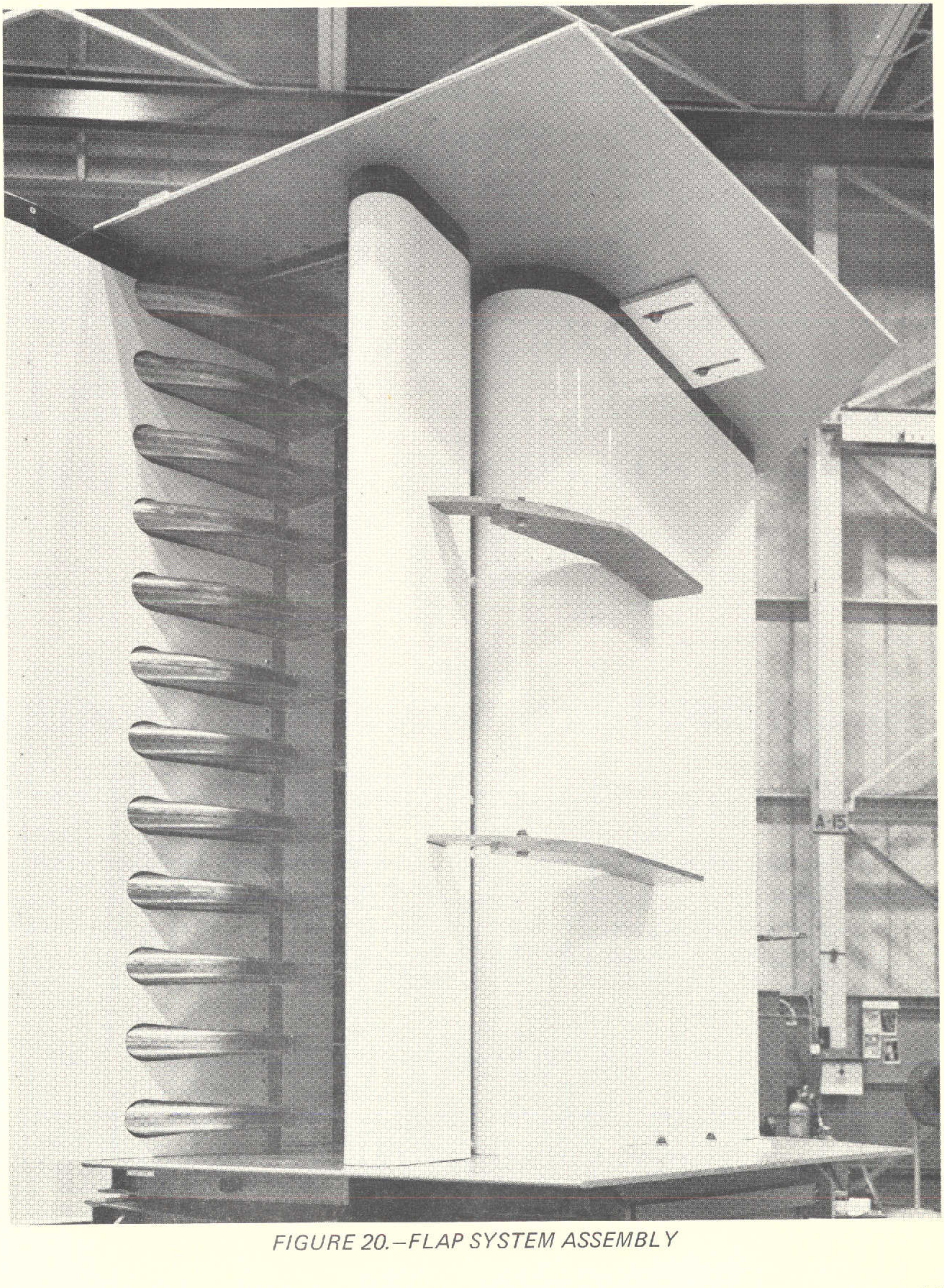


FIGURE 20.–FLAP SYSTEM ASSEMBLY

for a face sheet with a nominal reference Rayl number. This is also partially true in the case of the septum sheet, however, because there are two core bonds that take place on the septum sheet, and mismatch of the two cores on either side of the septum sheet can result in final Rayl numbers that vary greatly from panel to panel. This has proven to be the greatest difficulty in the panel construction, and it can be seen that careful examination of the reference Rayl of the septum sheet must be made prior to core bonds to achieve final Rayls which fall within the drawing tolerance bond.

This drawing tolerance is ±25% of the design point, which is as much as ±10% higher than would be expected for production of normal single-layer polyimide acoustic panels; i.e., panels with no septum sheet. At the time of writing this report, no heated flow is available to the system, and although model hardware is designed structurally for the hot (350°F) condition, panel acoustic design has been established at 70°F. This flap system has the capability of providing the following geometric changes:

a) Throat dimension (between flap and shroud)

b) Exit dimension (between flap and shroud)

c) Intake angle (relative to shroud)

d) Flap angle (relative to wing chord plane)

4.4 SIDE PLATES

A pair of side plates (fig. 19) were built to entrain the flap system at the ends. They are mounted at all times to the flap system and move with the flaps during flap angle changes. Provision within these side plates has been made to blow air at an expected pressure of up to 100 psi parallel to and in the same direction as the nozzle flow to minimize boundary layer end effect.

4.5 FLAP SYSTEM SUPPORT AND TRANSLATION STRUCTURE

Structure (figs. 21 and 22) was built to give position changes of the flap system. This consists of two frames that move in X-Y coordinates to cater to different flap angle requirements (20°-80°). A ground-supported fixed structure was built to support these translation frames. This fixed structure is ground-mounted and completely independent of the existing NASA/Lewis test stand.

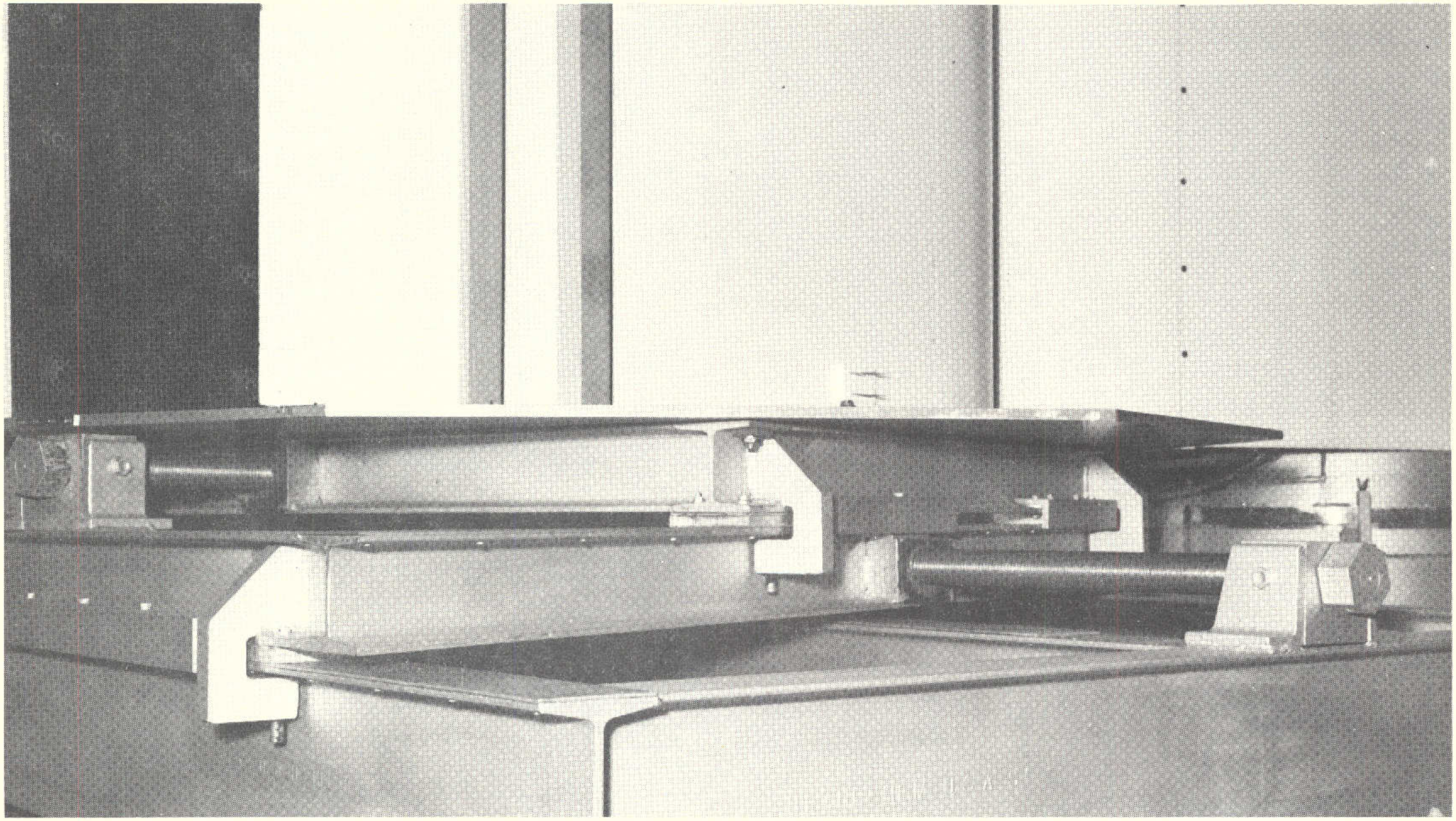


FIGURE 21.—FLAP SYSTEM TRANSLATING STRUCTURE

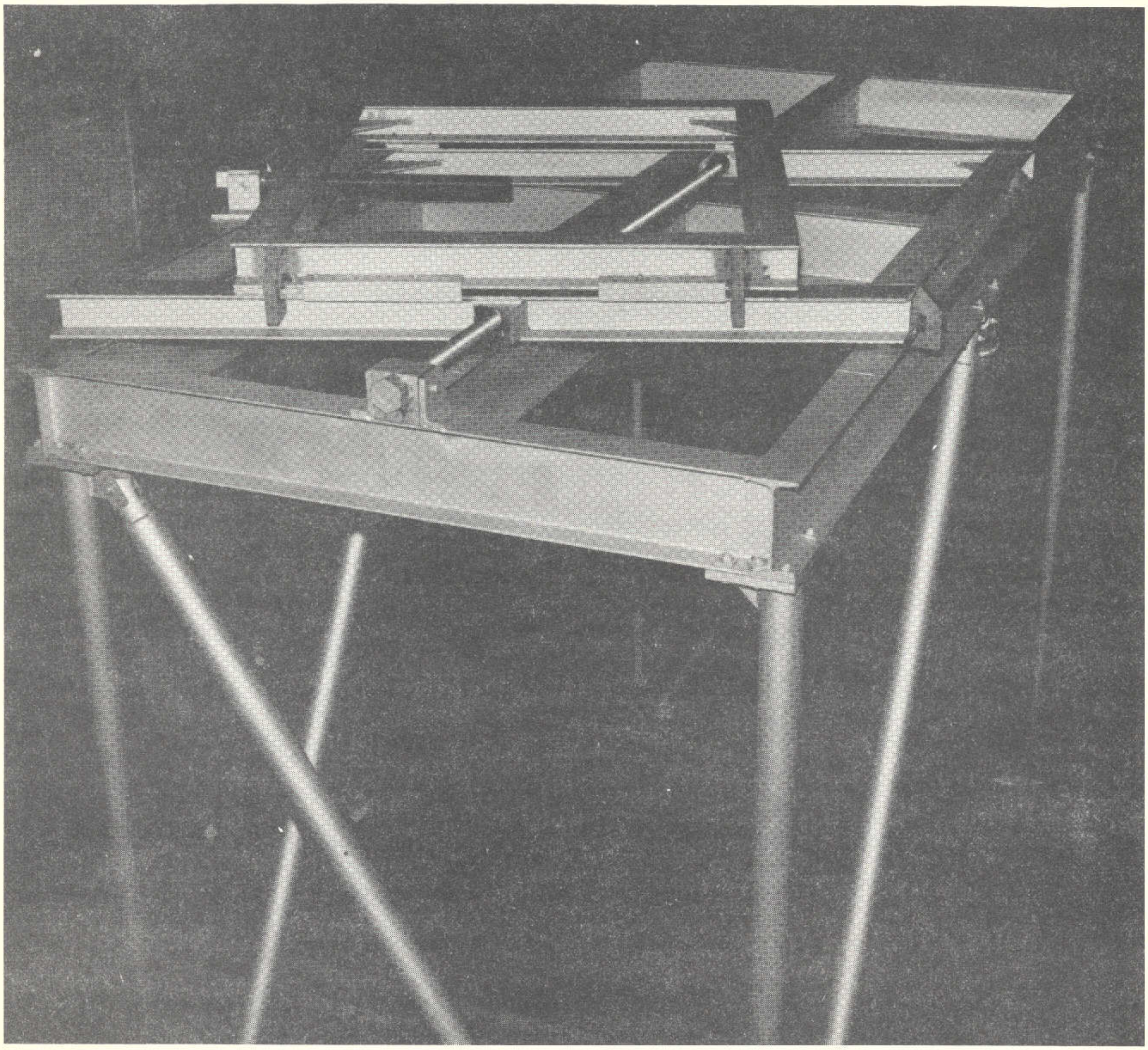


FIGURE 22.—FLAP SYSTEM SUPPORT AND TRANSLATING STRUCTURE

4.6 ACOUSTIC MUFFLER PLENUM

An acoustically treated plenum (fig. 23) was provided to install in the 30-in. diameter supply line just upstream of the nozzle feed plenum. This is used to eliminate any accumulation of valve and pipeline noise and is of sufficient volume to reduce the flow to low Mach numbers to maximize the efficiency of the plenum. The acoustic treatment consists of fiberglass matting contained within a perforated sheetmetal shroud. A completely lined plenum with a cross baffle, also acoustically treated, provides adequate design.

4.7 SIMULATED WING

A simulated wing section of 75-in. span and 14-ft chord (fig. 24) was designed, built, and attached to the nozzle feed plenum. This wing plane has the last 3 ft of the trailing edge removable to facilitate different nozzle changes.

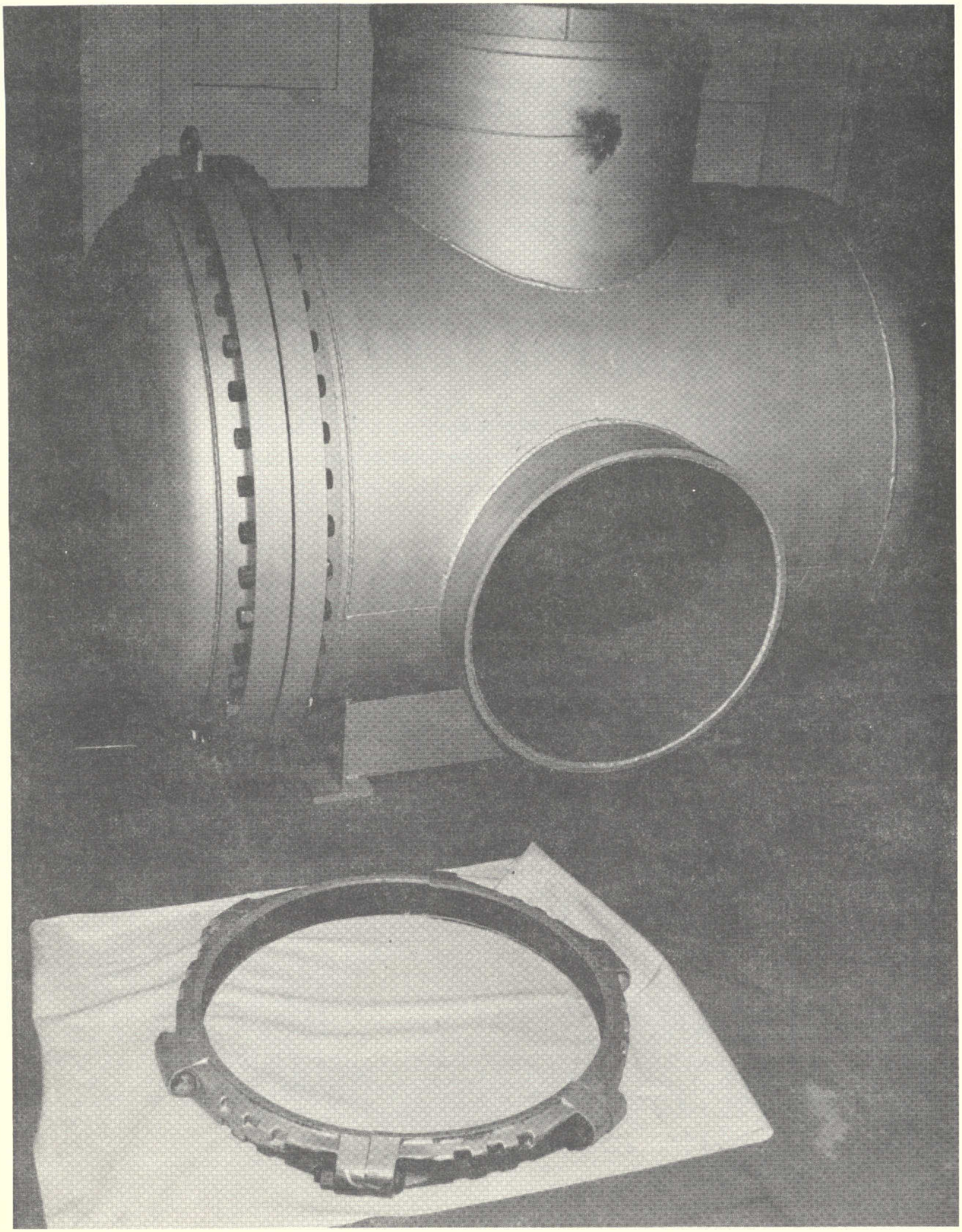


FIGURE 23.–ACOUSTIC MUFFLER

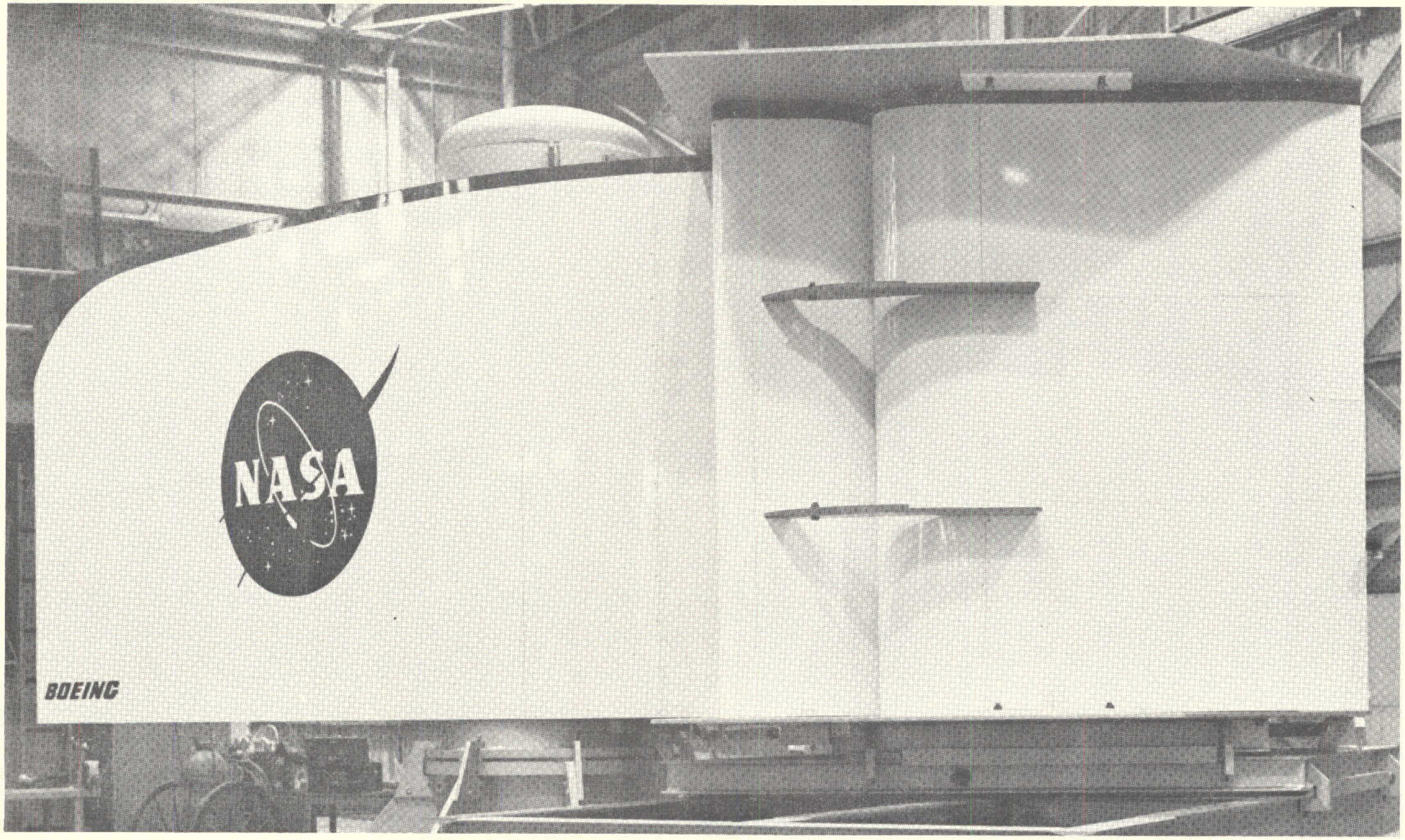


FIGURE 24.—SIMULATED WING PLANE

5.0 PROCUREMENT, FABRICATION, AND DELIVERY

The procurement of materials and the fabrication and assembly of all model hardware was accomplished at the Boeing manufacturing facilities, Seattle, Washington.

Model system checkout procedures were provided to NASA on an engineering information sheet, 5461-1007, items 1 and 2. These two items were accomplished and accepted by the Boeing engineering organization.

The other major checkout was the hydrostatic pressure test of the entire test system from the NASA interface through to the nozzle interface at 90 psi gage. This also was completed and accepted by Boeing engineering personnel and certified by Boeing Quality Control.

On completion of the model hardware assembly and checkout by Boeing personnel, the model was shipped to NASA-Lewis Research Center, Cleveland, Ohio, marked for Receiving. In addition to the model hardware, 10 copies of all engineering drawings, along with one reproducible copy, were shipped to the NASA technical monitor.

6.0 RELIABILITY AND PRODUCT ASSURANCE

From the detail design phase, the product assurance of all model hardware was initiated by careful tolerancing of interfacing details, nozzle exit control dimensions, contour surfaces of flaps and wing trailing edge, and important location control dimensions. During manufacture, quality control of each detail was maintained by standard Boeing inspection procedures for test hardware systems, and deviation from drawing requirements when occurring, were evaluated by the Engineering Department to provide part status, rework, acceptance, etc. As detail parts were completed and assembly and checkout started, interfacing parts, translation details, and relative locations were inspected, and pressure tests were carried out as specified on the face of drawings or by specific engineering instructions. Each phase of the product assurance step was accepted either by Boeing Quality Control personnel or by Engineering personnel as required. Close tolerance dimensions specified by Engineering were measured and recorded by Quality Control. These records, as well as the completed manufacturing records, are available for review.

All inspection apparatus, micrometers, pressure gages, etc. used during all stages of product assurance were validated Boeing Quality Control equipment.

7.0 OPERATING PROCEDURES

This section covers the operating conditions, the recommended environmental conditions, and the adjustment techniques for the augmentor wing model for acoustic testing. For complete model definition, materials used, dimensions, and fasteners, reference should be made to Boeing 5461 drawings.

7.1 STATIC MODEL CONDITION

At all times while the model hardware is in the static (nonoperating) condition it is imperative that guy wires be used to secure the wing plane and that the safety pin be installed at the vertical plenum bearing support. This requirement is necessary to avoid excessive system loading due to high wind conditions.

7.2 SYSTEM OPERATING CONDITIONS

The operating design conditions for the subject model hardware are shown below. They are design limitations and are the basis for the design analysis (shown in the appendix).

a) Maximum flow rate of 105 lb/sec

b) Pressure ratio of 3:1

c) Maximum operating temperature of 350° F

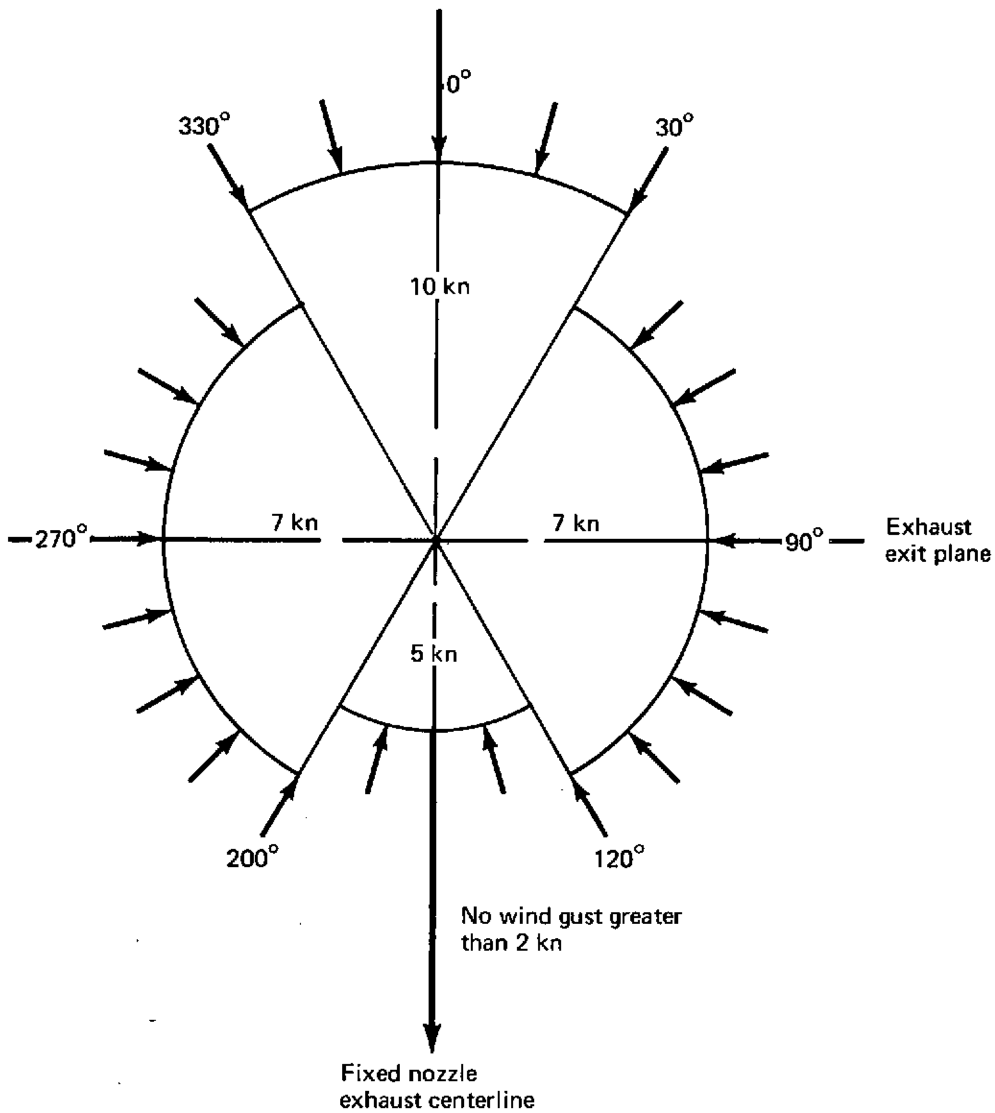
7.3 ENVIRONMENTAL RUNNING CONDITION

The following recommendations are given for the environmental conditions at which acoustic data can be taken.

a) Humidity range–30%-90%

b) Temperature range–32°-90° F

c) Wind conditions (see fig. 25)



Test limits—weather:

Wind direction and speed—as shown above
Humidity—30% to 90%
Temperature—32° to 90° F

FIGURE 25.—RECOMMENDED WEATHER TEST WINDOW

For best acoustic data the flap exit rake must be removed.

7.4 ADJUSTMENT TECHNIQUES

There are six areas of adjustment that are necessary to make a configuration change; these are listed below, together with their definitions. (See fig. 26.)

a) Flap angle—angle between wing chord plane and augmentor flap exit centerline

b) Augmentor throat size—minimum dimension between flap and shroud

c) Augmentor exit size—dimension between flap trailing edge and shroud trailing edge

d) Intake angle-relative angle between intake flap and shroud

e) Z dimension—distance between a line drawn parallel to the nozzle exit centerline from the lower edge of the nozzle exit and the closest point on the flap surface

f) ℓ_Z dimension—distance between nozzle exit plane and the tangent point of the Z dimension on the flap.

The following gives the techniques that should be adopted in making test configuration changes to the subject model hardware.

To make flap angle changes, three steps must be accomplished:

a) The nozzle plenum must be rotated. This is accomplished by releasing the eight clamps at the nozzle plenum bearing, located at the lower end of the wing plane, and releasing the Victaulic coupling at the lower end of the nozzle plenum. The wing and nozzle plenum can now be rotated to the desired flap angle by using the flap angle indicator at the lower end of the nozzle plenum and the flap angle pointer which is attached to the acoustic muffler. When the desired angle is obtained, the Victaulic coupling and bearing clamps can now be torqued. Bearing clamps should be torqued to 10-15 ft/lb.

   Caution: It will be necessary to move the flap system away from the nozzle exit before making the above adjustment. This is done by releasing the clamps on the top flap

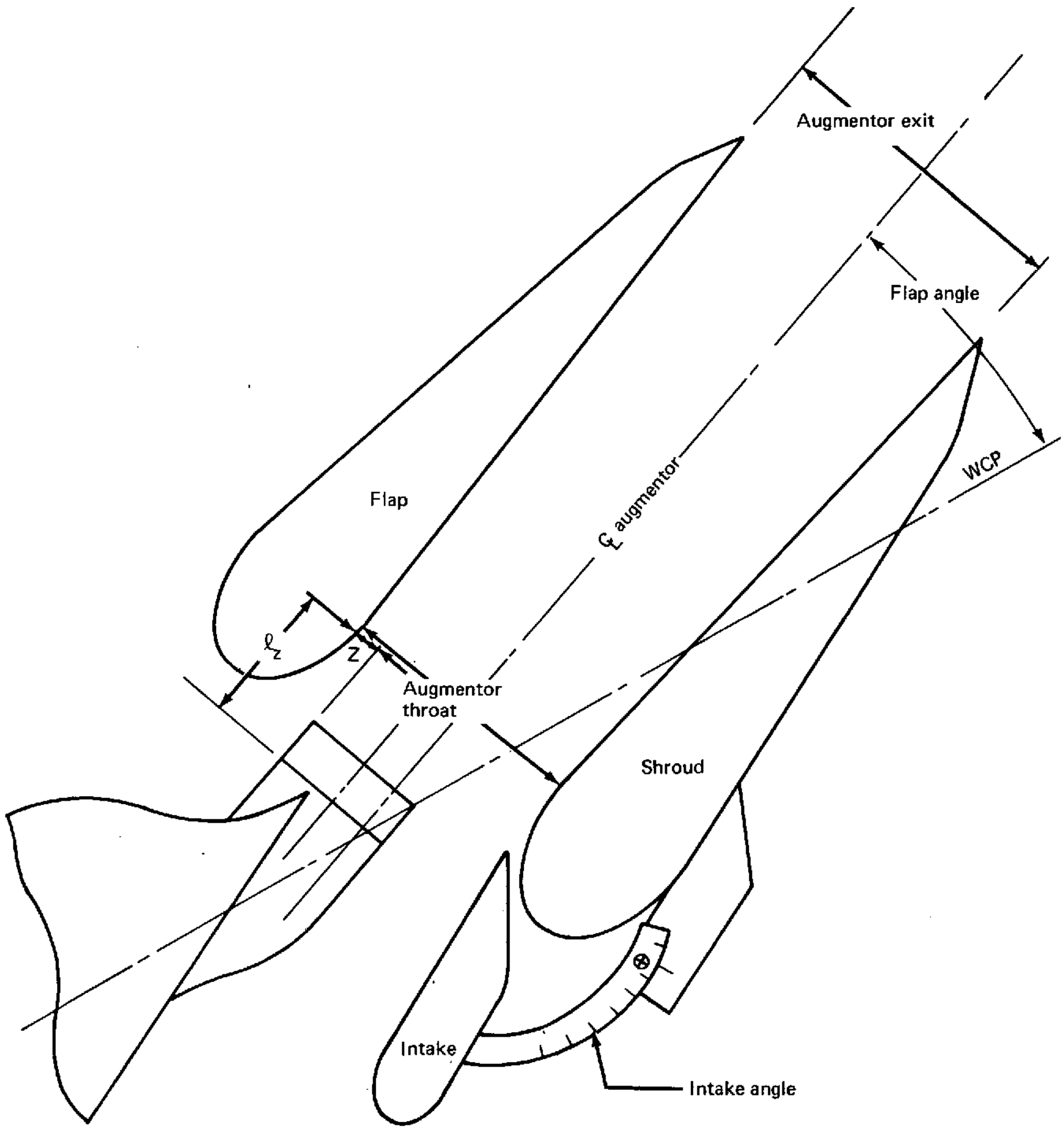


FIGURE 26.—FLAP SYSTEM VARIABLES

support frame (Y-axis) and by using the Y-axis traversing screw. Traverse the flap system away from the nozzle in the Y-direction.

b) After operation (a), the flap system must now be reoriented to the nozzle exit. This can be done by utilizing the X-axis and Y-axis traversing screws. With all flap frames unclamped, the flap system can be positioned relative to the nozzle exit by traversing in either direction until the desired location is obtained (Z and ℓ_Z dimensions). A new flap system "throat" and/or exit dimension may be required. This should be made prior to reorienting the flap system relative to nozzle and can be done by releasing the clamp screws at both ends of the flap and shroud, (5461-23 and 5461-21). Throat dimensions are engraved on both side plates 5461-25, and the new throat dimension required can be achieved by lining up the internal surface of the flap and shroud accordingly. Direct measurement can be made at the flap system exit. Reorientation to the nozzle exit can now be made (Z and ℓ_Z dimensions).

Note: After reorientation is made, ensure that all frame and flap clamps are retorqued to 15 to 20 ft-lb.

c) After operations (a) and (b) have been completed, a new intake angle will be required. This can be accomplished by releasing the two clamp screws at the brackets that join the shroud (5461-21) to the intake flap (5461-22) and by moving the intake to the new angle. Angle marks are engraved on the brackets in 5° increments (+35° to -10°). When the intake is in the new desired location, the clamp screws can be retorqued to 15-20 ft/lb.

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APPENDIX A

STRESS ANALYSIS

The following pages show the stress analysis of all major components that make up the augmentor wing model test system.

The calculations shown are for maximum model operating conditions given in the general specifications of the contract.

This stress analysis shows that the structural integrity of the hardware meets the structural requirements for the test system.

For ease of reference, the analysis is presented in the following order:

AUGMENTOR WING TEST RIG - NASA-LEWIS.

METRIC CONVERSION UNITS - REFERENCE NASA SP 7012

ACCELERATION

$g = 9.81\ m/sec^2$

$ft/sec^2 = .3048\ m/sec^2$

$in/sec^2 = 2.54 \times 10^{-2}\ m/sec^2$

AREA

$ft^2 = 9.290 \times 10^{-2}\ m^2$

$in^2 = 6.4516 \times 10^{-4}\ m^2$

DENSITY

$lbm/in^3 = 2.768 \times 10^4\ \frac{Kg}{m^3}$

$lbm/ft^3 = 1.602 \times 10^1\ \frac{Kg}{m^3}$

ENERGY

ft-lbf = 1.356 JOULES = 1.356 N-m

FORCE

lbf = 4.448 NEWTONS

LENGTH

INCH = 2.54×10^{-2} m

ft = .3048 m

MASS

lbm = .4536 Kg

PRESSURE

$$\text{ATMOSPHERE} = 1.013 \times 10^{5} \frac{N}{m^2}$$

$$lbf/ft^2 = 4.788 \times 10^{1} \frac{N}{m^2}$$

$$lbf/in^2 = 6.895 \times 10^{3} \frac{N}{m^2}$$

TEMPERATURE

$$^{\circ}\text{CELSIUS} = 5/9 \left(^{\circ}F - 32\right)$$

$$^{\circ}\text{KELVIN} = 5/9 \, ^{\circ}R = ^{\circ}C + 273$$

VOLUME

$$ft^3 = 2.832 \times 10^{-2} \, m^3$$

$$in^3 = 1.639 \times 10^{-5} \, m^3$$

GENERAL LOADS CALCULATIONS

NOZZLE FLOW AND DIMENSIONS

GIVEN: PRESSURE RATIO MAX. = 3:1

$$P_o = 3\left(1.013\times10^5\right) = 3.04\times10^5 \frac{N}{m^2} = (44.1 \text{ psia})$$

$$\dot{m}_{MAX} = 47.6 \frac{Kg}{Sec} = (105 \text{ lbm/sec})$$

FOR A RESERVOIR TEMPERATURE $T_o = 15.56°C$ $(60°F)$

$$T_o = 15.56 + 273 = 288.56\,°K \quad (520°R)$$

FROM AIR PROPERTIES TABLE:

$$\frac{P_{EXIT}}{P_o} = .52828 \qquad \frac{T_{EXIT}}{T_o} = .833 \qquad T_{EXIT} = 240°K \;(434°R)$$

$$\overline{V}_{EXIT} = C = \left\{1.4\times 9.81\times R\times 240\right\}^{1/2}$$

WHERE $R = \dfrac{.3048\times 53.34}{\frac{5}{9}} = 29.22$

$$\therefore\ \overline{V} = \left\{9.64\times10^4\right\}^{.1/2} = 311\ m/sec = (1020\ ft/sec)$$

$$\rho_{EXIT} = \frac{P_{EXIT}}{R\,T_{EXIT}} = \frac{3.04\times10^5\times.528}{29.22\times240\times9.81} = 2.33\frac{Kg}{m^3} = \left(.1452\frac{lbm}{ft^3}\right)$$

$$A_{EXIT} = \frac{\dot{m}}{\rho\overline{V}} = \frac{47.6}{2.33(311)} = .0658\ m^2 \text{ MAX. } (.709)\,ft^2$$

IF $A_{EXIT} = .0612\ m^2$ $(95\ in^2)$ $\quad \dot{m} = 44.3\dfrac{Kg}{Sec}$ $\left(97.8\dfrac{lbm}{Sec}\right)$

$$\text{DUCT AREA} = \frac{\pi}{4}\left(30 \times 2.54 \times 10^{-2}\right)^2 = .456\ m^2 = \left(708\ in^2\right)$$

$$V_{DUCT} = \frac{43.3}{\rho\,(.456)}$$

$$\rho_{DUCT} = \frac{3.04 \times 10^5}{29.22 \times 288 \times 9.81} = 3.68\ \frac{kg}{m^3} = \left(.229\ \frac{lbm}{ft^3}\right)$$

$$V_{DUCT} = \frac{43.3}{3.68\,(.456)} = 25.8\ m/sec = \left(84.6\ \frac{ft}{sec}\right)$$

NOZZLE THRUST

SINCE THE EXIT IS AT $20°$ FROM THE INLET THERE WILL BE x AND Y COMPONENTS :

$$P_x = \frac{\dot{m}g}{g_c}\left(\bar{V}\cos 20° - V_{DUCT}\right) + \left(P_{EXIT} - ATM.\right)A_{EXIT}\cos 20°$$

$$P_y = \frac{\dot{m}g}{g_c}\left(\bar{V}\sin 20°\right) + \left(P_{EXIT} - ATM.\right)\sin 20°\,A_{EXIT}$$

$$P_x = \frac{44.3}{9.81}g\left(311\,[.94] - 25.8\right) + \left(1.605 \times 10^5 - 1.013 \times 10^5\right)(.0612).94$$

$$P_x = 11800 + 3400 = 15200\ \text{NEWTONS} = (3423\ lbs)$$

$$P_y = \frac{44.3}{9.81}g\left(311\,[.342]\right) + \left(1.605 \times 10^5 - 1.013 \times 10^5\right)(.0612)(.342)$$

$$P_y = 4710 + 1244 = 5954\ \text{NEWTONS} = (1340\ lbs)$$

RESULTANT : $P = \left\{ (15200)^2 + (5954)^2 \right\}^{1/2} = 16330$ NEWTONS
(3695 lbs)

BENDING MOMENT ON DUCT :

$$M_{MAX} = \frac{1}{2}(16330)\left(75 \times 2.54 \times 10^{-2}\right) = 15{,}550 \text{ N-m}$$
(11,480 ft-lb)

TORQUE ON DUCT :

$$(2.875 \times 12)(2.54 \times 10^{-2})(5954) = 5220 \text{ N-m} = (3850 \text{ ft-lb})$$

WITH THE DUCT ROTATED IN THE 90° POSITION SO THAT THE NOZZLE JET IS IMPINGING ON THE SHROUD (AN EXTREME CASE)

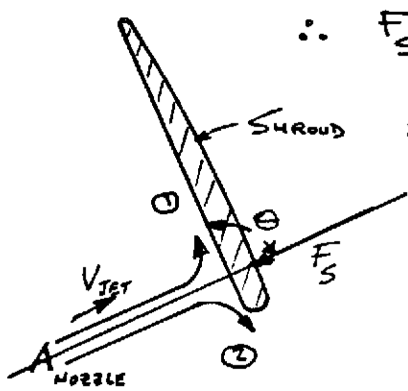
FORCE ON SHROUD : $\dot{m} = \rho A V$

$$\Sigma F_S = \dot{m}\,\Delta V_x = \rho A V_{JET}^2 \sin\theta$$

AND $\dot{m}_1 = \frac{\dot{m}}{2}(1+\cos\theta)$ $\dot{m}_2 = \frac{\dot{m}}{2}(1-\cos\theta)$

WITH $\theta = 90°$ $\sin\theta = 1$

$$\therefore \; F_S = \rho_E A_N V_{JET}^2 \quad \dot{m}_1 = \dot{m}_2 = \frac{\dot{m}}{2}$$



$$F_S = 2.33(.0612)(311)^2$$

$F_S = 13800$ NEWTONS (3110 lbs)

DUCT DEFLECTION DUE TO NOZZLE THRUST :

$$y = \frac{1}{8}\,\frac{16400\left(79 \times 2.54 \times 10^{-2}\right)^3}{30 \times 10^6 \times 6.895 \times 10^3 \times I}$$

$$I = .05\left[\left(30 \times 2.54 \times 10^{-2}\right)^4 - \left(29.25 \times 2.54 \times 10^{-2}\right)^4\right]$$

$$I = 1.66 \times 10^{-3}\ m^4 \quad \left(4000\ \text{IN}^4\right)$$

$$\therefore\ y = 4.82 \times 10^{-5}\ m \quad (.0019\ \text{IN.})$$

RESULTANT MOMENT DUE TO WING & NOZZLE ASSEMBLY

$$2220\left(30 \times 2.54 \times 10^{-2}\right) - 4890\left(27 \times 2.54 \times 10^{-2}\right) = 1660\ N.m$$

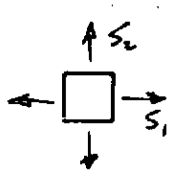
(1225 ft.-lbs)

DUCT DEFLECTION DUE TO MOMENT:

$$y = \frac{1660\left(41.5 \times 2.54 \times 10^{-2}\right)}{30 \times 10^6 \times 6.895 \times 10^3 \times 1.66 \times 10^{-3}}\left(79 - 20.75\right)2.54 \times 10^{-2} = 755 \times 10^{-8}\ m.$$

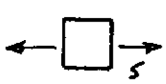
$\left(297 \times 10^{-6}\ \text{IN}\right)$

STRESSES IN THE DUCT :

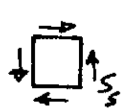


$$S_1 = \frac{pR}{2t} \qquad \text{DUE TO PRESSURE}$$

$$S_2 = \frac{pR}{t}$$



$$S = \frac{Mc}{I} \qquad \text{DUE TO BENDING}$$



$$S_s = \frac{2TR}{\pi\left(R^4 - R_o^4\right)} \qquad \text{DUE TO TORSION}$$

$t = 9.52 \times 10^{-3}\ m$ (.375 IN

$R = .381\ m$ (15 IN.),

$R_o = .371\ m$ (14.625

$$S_1 = \frac{.381\left(2.027\times10^5\right)}{2\times 9.52\times10^{-3}} = 4.1\times10^6\ \frac{N}{m^2} \quad (600\ psi)$$

$$S_2 = 2S_1 = 8.2\times10^6\ \frac{N}{m^2} \quad (1200\ psi)$$

$$S = \frac{\left[16400\left(42\times2.54\times10^{-2}\right) - 1660\right].381}{1.66\times10^{-3}} = 3.63\times10^6\ \frac{N}{m^2}$$

$$(526\ psi)$$

$$S_s = \frac{2\left(5220\right).381}{\pi\left(.381^4 - .371^4\right)} = 6.34\times10^5\ \frac{N}{m^2} \quad (92\ psi)$$

$$\therefore \quad S_2 = 8.2\times10^6\ \frac{N}{m^2} \quad (1200\ psi)$$

$$S_1 = 4.1\times10^6 + 3.63\times10^6 = 7.73\times10^6\ \frac{N}{m^2} \quad (1126\ psi)$$

$$S_s = 6.34\times10^5\ \frac{N}{m^2} \quad (92\ psi)$$

$$\text{MAX. SHEAR STRESS} = \left\{\left[\frac{(7.73-8.2)10^6}{2}\right]^2 + \left[6.34\times10^5\right]^2\right\}^{1/2}$$

$$\cong 6.9\times10^5\ \frac{N}{m^2} \quad \doteq (100\ psi)$$

$$\text{MAX. NORMAL STRESS} \quad \frac{(8.2+7.73)10^6}{2} + \text{MAX. SHEAR STRESS}$$

$$\simeq 8.\times10^6 + .69\times10^6 = 8.69\times10^6\ \frac{N}{m^2}$$

$$\doteq (1263\ psi)$$

WEIGHT BREAKDOWN OF ASSEMBLIES SUPPORTED BY STAND

NOZZLE ASSY : 4890 NEWTONS (1100 lbs)

WING ASSY : 2220 NEWTONS (500 lbs)

DUCT ASSY : 6780 NEWTONS (1525 lbs)

IN THE STATIC CONDITION (NON-OPERATIVE)
THRUST BEARING REACTIONS:

$$\Sigma M_B = 2200\left(30-17.2\right)2.54\times10^{-2} - 6780\left(17.2\right)2.54\times10^{-2}$$
$$- 4890\left(27+17.2\right)2.54\times10^{-2} + R_{y_2}\left(34.5\right)2.54\times10^{-2} = 0$$

$$\uparrow R_{y_2} = 8770 \text{ NEWTONS} = \left(1975 \text{ lbs}\right)$$

$$\Sigma M_{y_2} = 6780\left(17.2\right)2.54\times10^{-2} + 2220\left(17.2+30\right)2.54\times10^{-2}$$
$$- 4890\left(27-17.2\right)2.54\times10^{-2} - R_{y_1}\left(34.5\right)2.54\times10^{-2} = 0$$

$$\uparrow R_{y_1} = 4950 \text{ NEWTONS} = \left(1122 \text{ lbs}\right)$$

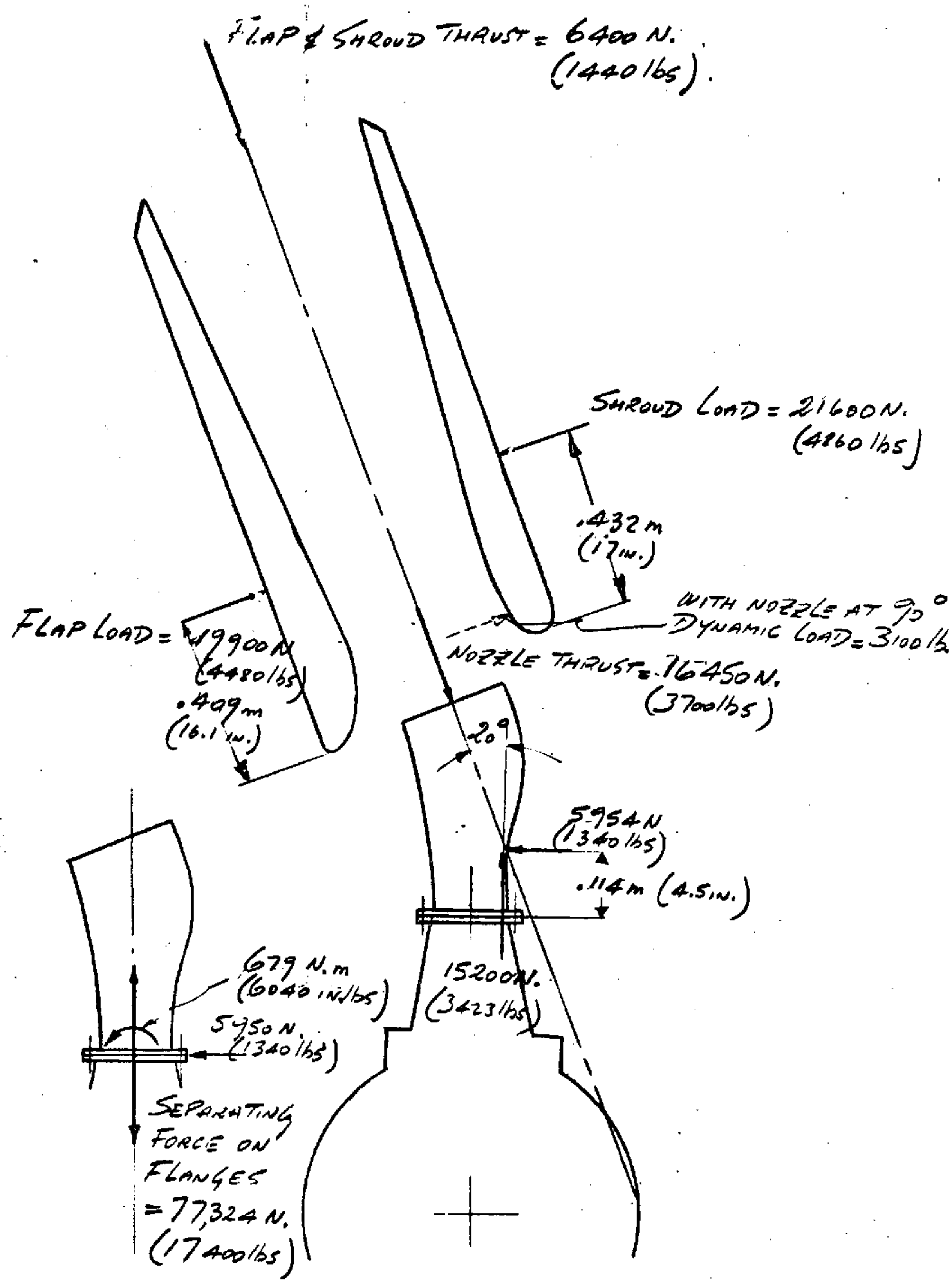
IN THE OPERATING CONDITION - NOZZLE THRUST ≈ 16400 N. (3700 lbs)

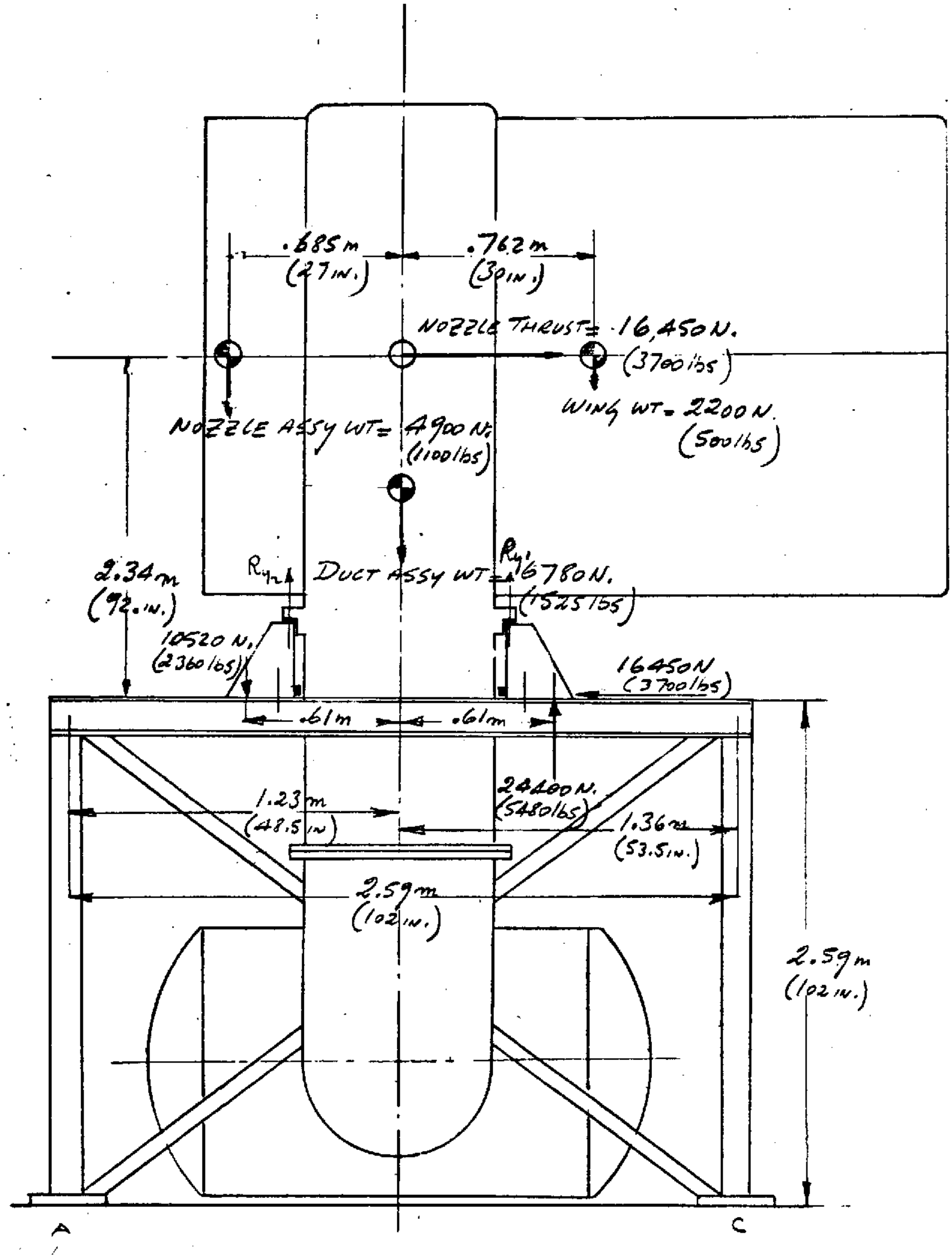
$$\Sigma M_{y_2} = \left[6780\left(17.2\right) + 2220\left(47\right) - 4890\left(10\right) + 16400\left(42\right) - R_{y_1}\left(34.5\right)\right]2.54\times10^{-2} = 0$$

$$\downarrow R_{y_1} = 25200 \text{ N.} = \left(5660 \text{ lbs}\right)$$

$$\uparrow R_{y_2} = 11300 \text{ N.} = \left(2550 \text{ lbs}\right)$$

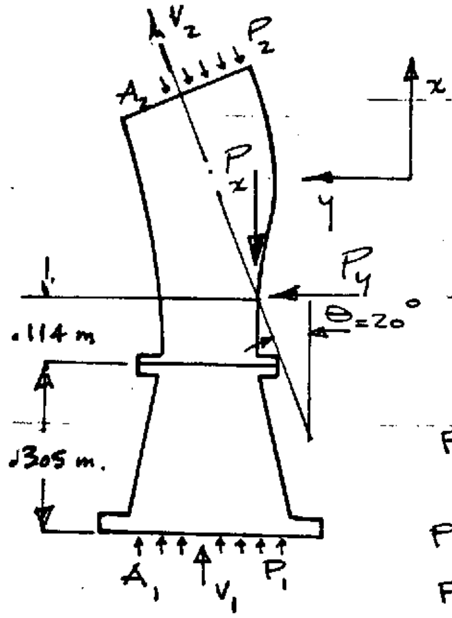
$$\overleftarrow{\Sigma F_x} = 16400 \text{ N.} \left(3700 \text{ lbs}\right)$$ HORIZONTAL SHEAR ON CLAMP AND FRICTION SURFACES


FLAP & SHROUD THRUST = 6400 N.
(1440 lbs).
SHROUD LOAD = 21600 N.
(4860 lbs)
.432 m
(17 IN.)
WITH NOZZLE AT 90°
DYNAMIC LOAD = 3100 lb
FLAP LOAD = 19900 N
(4480 lbs)
.409 m
(16.1 IN.)
NOZZLE THRUST = 16450 N.
(3700 lbs)
20°
5954 N
(1340 lbs)
.114 m (4.5 IN.)
679 N.m
(6040 IN/lbs)
15200 N.
(3423 lbs)
5950 N.
(1340 lbs)
SEPARATING
FORCE ON
FLANGES
= 77,324 N.
(17400 lbs)



.685 m (27 IN.)
.762 m (30 IN.)
NOZZLE THRUST = 16,450 N. (3700 lbs)
WING WT = 2200 N. (500 lbs)
NOZZLE ASSY WT = 4900 N. (1100 lbs)
DUCT ASSY WT = 6780 N. (1525 lbs)
2.34 m (92. IN.)
10520 N. (2360 lbs)
16450 N (3700 lbs)
.61 m
.61 m
24400 N. (5480 lbs)
1.23 m (48.5 IN)
1.36 m (53.5 IN.)
2.59 m (102 IN.)
2.59 m (102 IN.)
A
C

NOZZLE SEPARATING LOAD FROM DUCT



x DIRECTION

$$P_1 A_1 - P_2 A_2 \cos\theta - P_x = \frac{\dot{m}}{g_c}\left(V_2 \cos\theta - V_1\right)$$

y DIRECTION

$$P_y - P_2 A_2 \sin\theta = \frac{\dot{m}}{g_c}\left(V_2 \sin\theta\right)$$

P_x & P_y FORCES OF NOZZLE ON FLUID.

$P_1 = 3.040 \times 10^{5} - 1.013 \times 10^5 = 2.027 \times 10^5 \ \frac{N}{m^2}$

$P_2 = 1.605 \times 10^5 - 1.013 \times 10^5 = .592 \times 10^5 \ \frac{N}{m^2}$

$A_1 \approx A_{DUCT} = .456 m^2 \ (708 in^2)$

$A_2 = .0612 \ m^2 \ (95 in^2)$

$V_1 = 25.8$ m/sec (84.6 ft/sec)

$V_2 = 311$ m/sec (1020 ft/sec)

$\dot{m} = 44.3 \ \frac{kg}{sec} \ \left(97.8 \ \frac{lbm}{sec}\right)$

$\cos\theta = .94 \qquad \sin\theta = .342$

$$P_x = -44.3\left(311 \times .94 - 25.8\right) - .592 \times 10^5 (.0612) .94$$
$$+ \ 2.027 \times 10^5 \ (.456) = 7.7324 \times 10^4 \ N. \quad (17400 \ lbs)$$

$$P_y = 44.3\left(311 \times .342\right) + .592 \times 10^5 \ (.0612) .342 = 5.95 \times 10^3 \ N.$$
(1340 lbs)

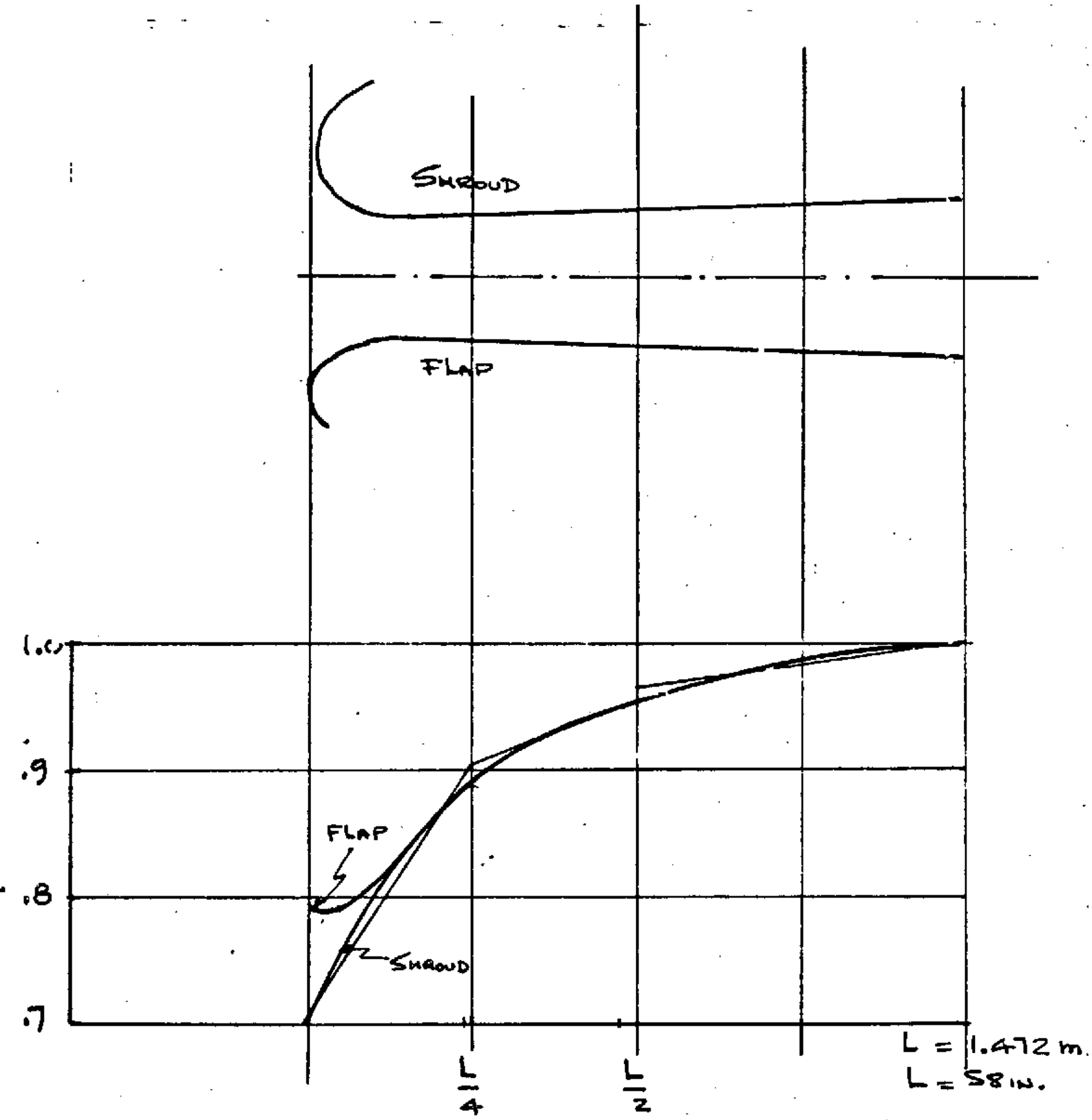
MOMENT ON NOZZLE FLANGE :

$$5.95 \times 10^3 \times .114 = 679 \ N.m. \quad (6040 \ in \ lbs)$$

MOMENT ON NOZZLE EXTENSION FLANGE :

$$5.95 \times 10^3 \times (.305 + .114) = 2.49 \times 10^3 \text{ N.m.} \quad (22100 \text{ in lbs})$$

FORCE CALCULATIONS ON FLAP & SHROUD.



STATIC PRESSURE PROFILES

ON SHROUD: WITH THE AREAS DIVIDED AS SHOWN:

$$F_S = \left[(1.0-.7)\frac{L}{4} - \left(\frac{.91-.7}{2}\right)\frac{L}{4} + (1.0-.7)\frac{L}{4} - \left(\frac{.95-.7+.91-.7}{2}\right)\frac{L}{4} + (1.0-.7)\frac{L}{2} - \left(\frac{1.0-.7+.96-.7}{2}\right)\frac{L}{2}\right] P_a \times \text{WIDTH}$$

$$\therefore F_S = \left[.3\frac{L}{4} - .105\frac{L}{4} + .3\frac{L}{4} - .23\frac{L}{4} + .3\frac{L}{2} - .28\frac{L}{2}\right] P_a \times \text{WIDTH}$$

$$F_S = .076\, L P_a \ell \qquad \ell = \text{WIDTH OF FLAP} = 1.905\text{ m. } (75\text{ IN})$$

$$F_S = .076\,(1.472)\,(1.013\times10^5)\,1.905 = 21600\text{ N.} = (4860\text{ lbs})$$

ON FLAP:

ADDITIONAL AREA UNDER THE CURVE:

$$.224\,L + \left(\frac{.8-.7+.78-.7}{2}\right)\frac{L}{16}$$

$$\Delta A = (.224 + .00562)L = .23\,L \qquad .3L - .23L = .07\,L$$

$$\therefore \quad F_{FLAP} = .07\, L \ell P_a = 19900\text{ N.} = (4480\text{ lbs}).$$

THRUST CALCULATIONS:

TAKE A 5° DIVERGENT ANGLE

ALREADY ESTABLISHED AVG $\frac{P_s}{P_a} = .7 + .224 = .924$

$$T = (1-.924) P_a A_{AVG} = .076\,(1.013\times10^5)\,A_{AVG}$$

$$A_{EXIT} = (21\times75)(2.54\times10^{-2})^2$$

$$A_{THROAT} = (13.5\times75)(2.54\times10^{-2})^2$$

$$A_{AVG} \cong \left(\frac{21 \times 13.5}{2}\right) 75 \left(2.54 \times 10^{-2}\right)^2 = .833 \text{ m}^2 \quad \left(1290 \text{ IN}^2\right)$$

$$\therefore \quad T = .076 \left(1.013 \times 10^5\right) .833 = 6.4 \times 10^3 \text{ NEWTONS} \quad (1440 \text{ lbs})$$

DIRECTED TOWARDS THE THROAT.

RESULTANT LOCATION: FROM LEADING EDGE x

$$F_S\, x = P_a \ell \left\{ .195 \frac{L}{4} x_1 + .07 \frac{L}{4}\left(\frac{L}{4} + x_2\right) + .02 \frac{L}{2}\left(\frac{L}{2} + x_3\right)\right\}$$

$$x_1 = \frac{L}{4}\left(\frac{2}{3}\right) = \frac{L}{6} = 9.66$$

$$x_2 = \frac{L}{4}\left\{\frac{1.9 + .91}{3(.95 + .91)}\right\} = \frac{L}{4}\left\{\frac{2.81}{5.58}\right\} = .126 L$$

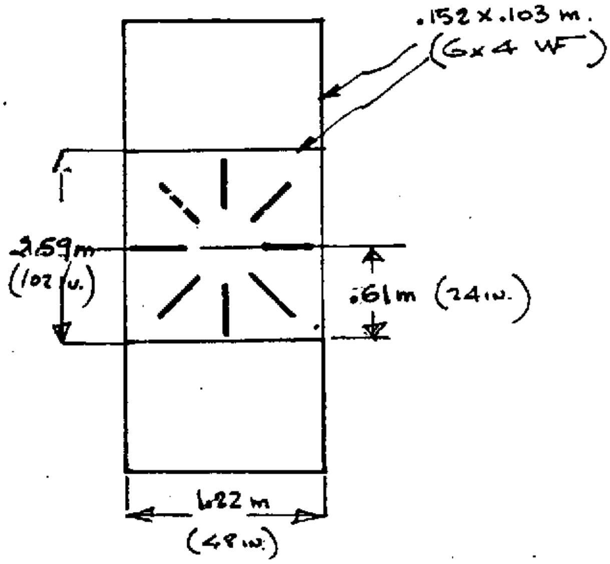
$$x_3 = \frac{L}{2}\left\{\frac{2 + .96}{3(1 + .96)}\right\} = \frac{L}{2}\left\{\frac{2.96}{5.88}\right\} = .252 L$$

$$F_S\, x = P_a \ell \left\{ .195 \frac{L}{4} \frac{L}{6} + .07 \frac{L}{4}\left(\frac{L}{4} + .126L\right) + .02 \frac{L}{2}\left(\frac{L}{2} + .252L\right)\right.$$

$$x = \frac{8850 \text{ N.m}}{21600 \text{ N.}} = .41 \text{ m} \quad (16.1 \text{ IN}).$$

BEARING SUPPORT FRAME

MATERIAL : STRUCTURAL STEEL, 1020 H. ROLLED YIELD $= 2.48 \times 10^{8} \frac{N}{m^2}$ (36 KSI)



SECTION MODULUS $Z = 11.85 \times 10^{-5} m^3$ (7.24 in^3)

FRAME LOADS :

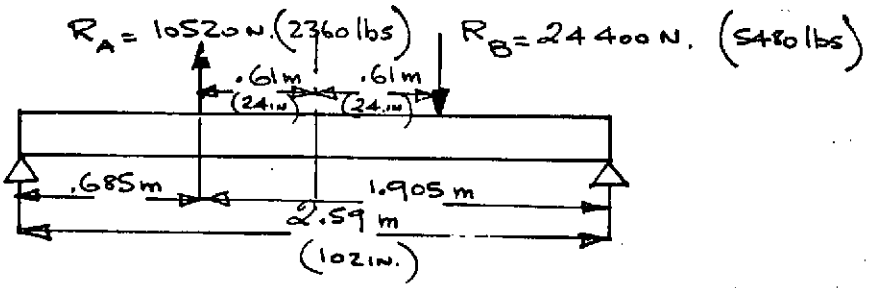
$\Sigma F_y = 4890 + 2220 + 6780 = 13890$ NEWTONS (3125 lbs)

$$\Sigma M_A = \left[4890(27-24) + 6780(24) + 16400(55) + 2220(24+30) - R_B(48)\right] 2.54 \times 10^{-2} = 0$$

$$\uparrow R_B = \frac{29800}{1.22} = 24400 \text{ N.} \quad (5480 \text{ lbs})$$

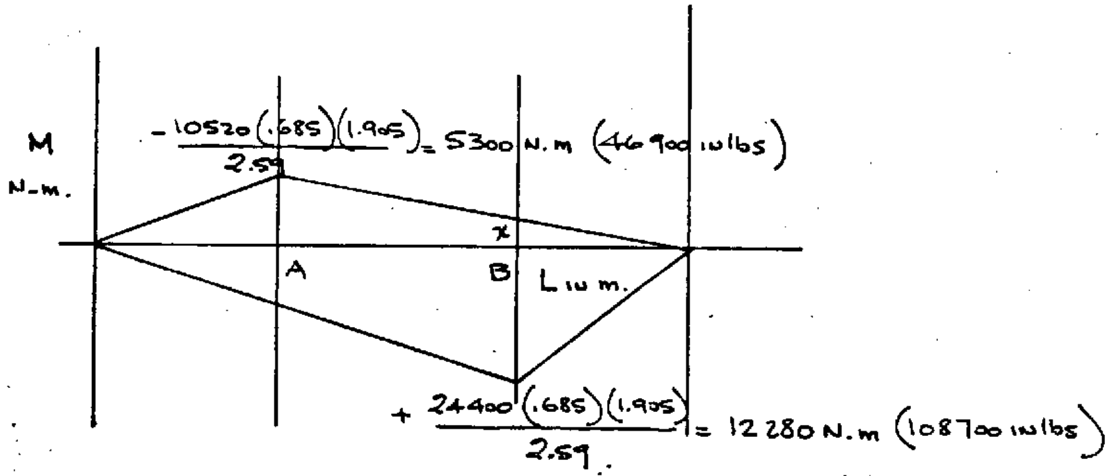
$$\Sigma M_B = \left[-4890(51) - 6780(24) + 16400(55) + 2220(6) - R_A(48)\right] 2.54 \times 10^{-2} = 0$$

$$\downarrow R_A = \frac{12830}{1.22} = 10520 \text{ N.} \quad (2360 \text{ lbs})$$



WITH THE LOAD (THRUST) ACTING ALONG THE LONGITUDINAL AXIS OF THE FRAME:

BENDING MOMENT DIAGRAM:



BENDING MOMENT $x = 5300 \frac{.685}{1.905} = 1906$ N.m.

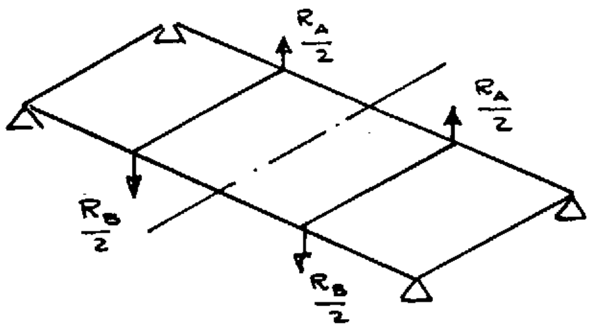
$M_{MAX} = 12280 - 1906 = 10374$ N.m (91700 in lbs)

TWO BEAMS ARE LOADED

$\therefore \; S_{MAX} = \frac{M_{MAX}}{2Z} = \frac{10374}{2\,(11.85 \times 10^{-5})} = 4.37 \times 10^{+7} \frac{N}{m^2}$ (6350 psi)

AT POINT B

$$FS = \frac{2.485 \times 10^{8}}{4.37 \times 10^{7}} = 5.7$$



WITH THE NOZZLE ROTATED 90° THE REACTIONS ARE AS SHOWN AND THE MAXMUM BENDING MOMENT IS NOT AS LARGE

$$M_{A\&B} = \frac{12200(.685)(1.905)}{2.59}$$

$$= 6140 \text{ N.m}$$

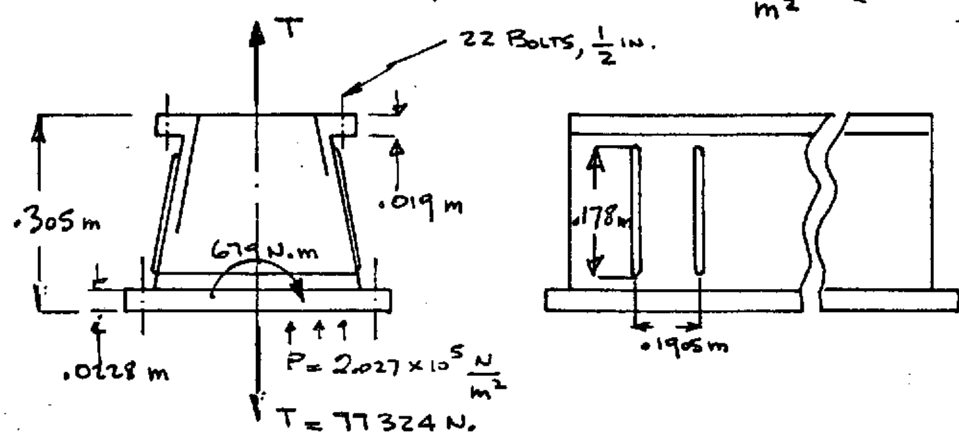
IF THE CROSS MEMBERS ARE CONSIDERED RIGID THEN THE FORCES ACTING CAN BE CONSIDERED AS $\frac{R_B}{2} - \frac{R_A}{2}$ ON THE TWO BEAMS : 24400 - 10520 = 13880 N.

AND THE MAX. BENDING MOMENT IS : $\frac{13880}{2}(.685) = 4750 \text{ N.m}$

THE FIRST CASE CONSIDERED IS THEREFORE THE MOST CRITICAL.

NOZZLE EXTENSION

MATERIAL : 1018   YIELD = $3.24 \times 10^8 \frac{N}{m^2}$ (47 KSI)



FOR THE SECTION .1905 x .178 m. x .00965 m THICK.

CASE [41] ROARK, UNIFORM PRESSURE, ALL EDGES FIXED.

$$S_{max} = \frac{.5\,(2.027\times10^5)(.178)^2}{(.00965)^2\left(1+.623\left[\frac{.178}{.1905}\right]^6\right)} = 2.43\times10^7\ \frac{N}{m^2}\ (3550\,psi)$$

$$FS = \frac{3.24\times10^8}{2.43\times10^7} = 13.3$$

SEPARATING LOAD = 77324 N., 22 BOLTS, .0127 m DIA.

$8.3\times10^8 \frac{N}{m^2}$ YIELD. (120 KSI)

$$LOAD/_{BOLT} = \frac{77324}{22} = 3520\ N.\ (792\ lbs/_{BOLT})$$

STRESS CONCENTRATION FACTOR ≈ 3

WITHOUT PRELOAD ON BOLTS $S = \frac{3520}{.1597\,(6.45\times10^{-4})} = 3.42\times10^7\ \frac{N}{m^2}$

WITH STRESS CONCENTRATION $S = 1.024\times10^8 \frac{N}{m^2}$ (14850 psi)

SHEAR STRESSES ON FLANGE ARE NEGLIGIBLE

PLENUM TANK ASSEMBLY

DESIGN ACCORDING TO "ASME BOILER & PRESSURE VESSEL CODE"

PROOF PRESSURE : $1.034 \times 10^6 \frac{N}{m^2}$ (150 psi)
TANK DIAMETER : 1.22 m (48 in)
ALLOWABLE STRESS FROM CODE : $1.12 \times 10^8 \frac{N}{m^2}$ (16250 psi)
EFFICIENCY OF JOINT E = .7

REQUIRED HEAD THICKNESS FROM CODE :

$$t = \frac{PD}{2SE - .2P} = \frac{1.034 \times 10^6 (1.22)}{2(1.12 \times 10^8).7 - .2(1.034 \times 10^6)}$$

$$t = .00805 \text{ m. } (.317 \text{ in})$$

MAKE $t = .00952$ m (.375 in.)

HEAD LOAD : $\frac{\pi (1.22)^2}{4}(1.034 \times 10^6) = 1.21 \times 10^6$ N. (2.72×10^5 lbs)

44 BOLTS .0286 m DIA. $A_{ROOT} = 5.52 \times 10^{-4} m^2$

$$\text{LOAD/BOLT} = \frac{1.21 \times 10^6}{44} = 2.75 \times 10^4 \text{ N. } (6190 \text{ lbs}).$$

STRESS CONCENTRATION FACTOR OF 3

STRESS WITHOUT PRELOAD $\frac{2.75 \times 10^4}{5.52 \times 10^{-4}} \times 3 = 1.495 \times 10^8 \frac{N}{m^2}$ (21700 psi)

WITH 160 KSI BOLTS

$$F_S > 5.5$$

LONGITUDINAL WELD JOINT :

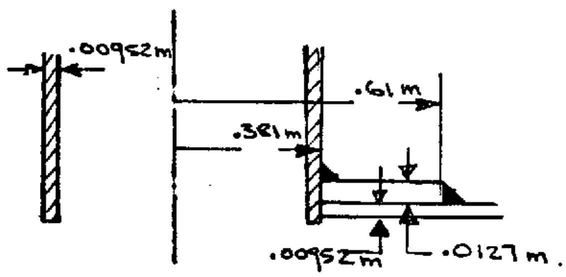
ACCORDING TO THE CODE CIRCUMFERENTIAL STRESS :

$$S = \frac{P(R + .6t)}{Et}$$

$$S = \frac{1.034\times10^{6}\left(.61 + .6\times .00952\right)}{.7\times .00952} = 1.034\times10^{8}\,\frac{N}{m^{2}}\ (15000\,psi)$$

REINFORCEMENT AROUND OPENINGS.

VERTICAL DUCT :



WALL THICKNESSES REQUIRED :

TANK $\quad t = \dfrac{1.034\times10^{6}\times .61}{1.12\times10^{8} - .6\left(1.034\times10^{6}\right)} = .0081\ m.$

DUCT $\quad t = \dfrac{1.034\times10^{6}\times .381}{1.12\times10^{8} - .6\left(1.034\times10^{6}\right)} = .00508\ m.$

INNER FILLET WELD : $3.58\times10^{-2}\times .7\times 1.27\times10^{-2} = 1.27\times10^{-2}\ m.$

REINFORCEMENT PLATE FILLET WELD :

$$3.58\times10^{-2}\times .7\times 1.27\times10^{-2} = 1.27\times10^{-2}\ m.$$

AREA OF REINFORCEMENT REQUIRED :

$$A = 74.4\times10^{-2}\times .0081 = 6.02\times10^{-3}\ m^{2}\ (9.34\,in^{2})$$

AREA OF REINFORCEMENT PROVIDED :

$$A_1 = 74.4\times10^{-2}\left(.00952 - .00805\right) = 1.059\times10^{-3}\ m^{2}$$

$$A_2 = 2\times 4.13\times10^{-2}\left(.00952 - .00508\right) = 3.68\times10^{-4}\ m^{2}$$

$$A_3 = 2\times .0127\left(.0127^{2} + .0127^{2}\right) = 3.63\times10^{-4}\ m^{2}$$

$$A_4 = .0127\left(47.5\times10^{-2}\right) = 58\times10^{-4}\ m^2$$

TOTAL AREA PROVIDED : $75.5\times10^{-4}\,m^2$

VERSUS REQUIRED : $60.2\times10^{-4}\,m^2$

HORIZONTAL DUCT :

TANK $t_r = .0081\ m$ $\quad$ $t_{ACTUAL} = .00952\ m.$

DUCT $t_{rm} = .00502\ m$ $\quad$ $t_{ACTUAL} = .00952\ m.$

$$P = 1.034\times10^{6}\ \frac{N}{m^2}$$

AREA OF REINFORCEMENT REQUIRED :

$$A = 74.4\times10^{-2}\times .0081 = 6.02\times10^{-3}\ m^2\ \left(9.34\ in^2\right)$$

DOUBLER SAME AS VERTICAL DUCT, SAME STRESS.

$$A_4 = .0127\left(47.5\times10^{-2}\right) = 58\times10^{-4}\ m^2\ +.\ OK.$$

SIZE OF WELD REQUIRED :

$$t_c = .7\left(.00952\right) = .00666\ m.\ \left(.2625\ in.\right)\ OK.$$

NOZZLE FEED PLENUM

$P = 1.034 \times 10^6 \frac{N}{m^2}$ (150 psi)

$D = .762$ m (30 in.)

$S = 1.12 \times 10^8 \frac{N}{m^2}$ (16250 psi) FROM ASME CODE AND CRS C-1018

$E = .7$

HEAD THICKNESS : $t = \frac{PD}{2SE - .2P} = \frac{1.034 \times 10^6 (.762)}{2(1.12 \times 10^8).7 - .2(1.034 \times 10^6)}$

$t = 5.03 \times 10^{-3}$ m. (.198 in)

USE $t = 9.52 \times 10^{-3}$ m. (.375 in).

RECTANGULAR OUTLET :

DOUBLER $\ell = \frac{.762}{2}\left(\frac{10.4 \times 10^{-1}}{360}\right) 2\pi = 27.2 \times 10^{-2}$ m EACH SIDE

ADDED AREA : $27.2 \times 10^{-2} (.0127) = 34.8 \times 10^{-4} m^2$

ADDED AREA SHORT SIDE : $7.52 \times 10^{-2} \times .0127 \times 2 = 19.3 \times 10^{-4} m^2$

SHELL : $t = \frac{PR}{SE - .6P} = \frac{1.034 \times 10^6 (.381)}{(1.12 \times 10^8)1 - .6(1.034 \times 10^6)}$

$t = 3.53 \times 10^{-3}$ m (.139 in)

USE $t = 9.52 \times 10^{-3}$ m. (.375 in.)

AREA OF REINFORCEMENT REQUIRED :

LONGITUDINAL : $190 \times 10^{-2} (3.53 \times 10^{-3}) = 67 \times 10^{-4} m^2$

CIRCUMFERENTIAL : $25 \times 10^{-2} (3.53 \times 10^{-3}) = 9.15 \times 10^{-4} m^2$

ADDED AREA IS $34.8 \times 10^{-4} m^2$

FOR LONGITUDINAL ADDED AREA : $19.3 \times 10^{-4} m^2$ + RIB AREA

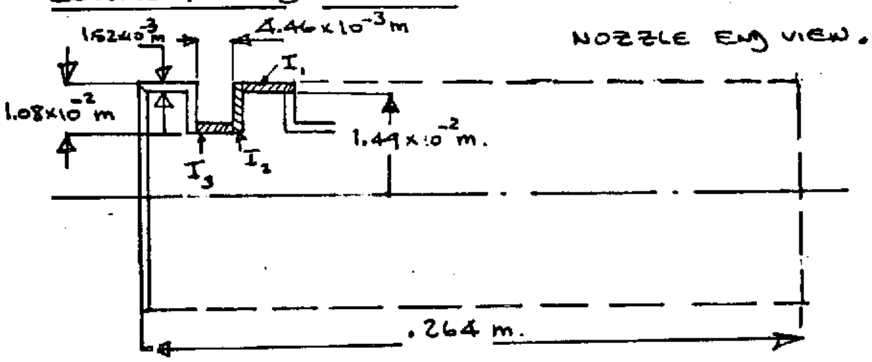
$+ (9.52 \times 10^{-3} - 5.03 \times 10^{-3}) 6$

ADDED AREA : $19.3\times10^{-4} + 6.84\times10^{-4} + \left(2.54\times10^{-2}\right)^{2}3$
$+ \left(6.35\times10^{-2}\right)^{2}1 =$

$= 19.3\times10^{-4} + 6.84\times10^{-4} + 48.4\times10^{-4}\ m^2$

$= 74.5\times10^{-4}\ m^2\ \left(11.56\ in^2\right).$

CORRUGATED NOZZLE



$$\text{PITCH} = 9.75 \times 10^{-3} + 4.46 \times 10^{-3} = 14.21 \times 10^{-3}\ \text{m}.$$

$$\text{CONVOLUTIONS} = \frac{26.5 \times 10^{-2}}{14.21 \times 10^{-3}} \times 2 = 37.$$

EQUIVALENT THICKNESS OF CONVOLUTIONS:

$$I_{EQ.} = \Sigma I_2 + \Sigma I_1 + \Sigma A_1 d^2 + \Sigma I_3 + \Sigma A_3 d^2$$

$I_1 = 20.2 \times 10^{-13}\ m^4 \qquad I_3 = 13.5 \times 10^{-13}\ m^4$

$A_1 = 10.2 \times 10^{-6}\ m^2 \qquad A_3 = 6.81 \times 10^{-6}\ m^2$

$\Sigma I_2 = 16.2 \times 10^{-11}\ m^4 \times 37 = 60 \times 10^{-10}\ m^4$

$\Sigma I_1 = 74.7 \times 10^{-12}\ m^4 \qquad \Sigma I_3 \doteq 50 \times 10^{-12}\ m^4$

$\Sigma A_1 d^2 = 8.1 \times 10^{-9}\ m^4 \qquad \Sigma A_3 d^2 = 5.3 \times 10^{-9}\ m^4$

$\therefore I_{EQ.} = 196 \times 10^{-10}\ m^4$ AT NOZZLE TIP

4.4 IN. BACK WHERE THE CONVOLUTIONS DISSAPPEAR

$I_{EQ} = 1.19 \times 10^{-10}\ m^4$

USING A SECTION LENGTH OF 6.67×10^{-2} m.

$$I_o = \frac{6.67 \times 10^{-2}}{2.64 \times 10^{-1}}\ 1.96 \times 10^{-10} = 49.6 \times 10^{-10}\ m^4$$

I VARIES FROM $L=0$ TO $L = 4112$ m.

IN THE FASHION $I = \dfrac{1}{z L^3 + a}$

AND $I_{AVG} = \dfrac{1}{L} \displaystyle\int_0^L \dfrac{1}{z L^3 + a}\, dL$

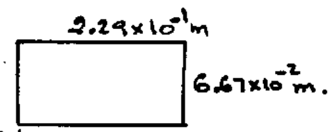
$I_{AVG} \approx 10.4 \times 10^{-10}\, m^4 \approx \dfrac{b t_{EQ.}^3}{12}$

$\therefore\ t_{EQUIV.} = .554 \times 10^{-2}\, m = .00554\, m \quad (.218\ \text{IN})$.

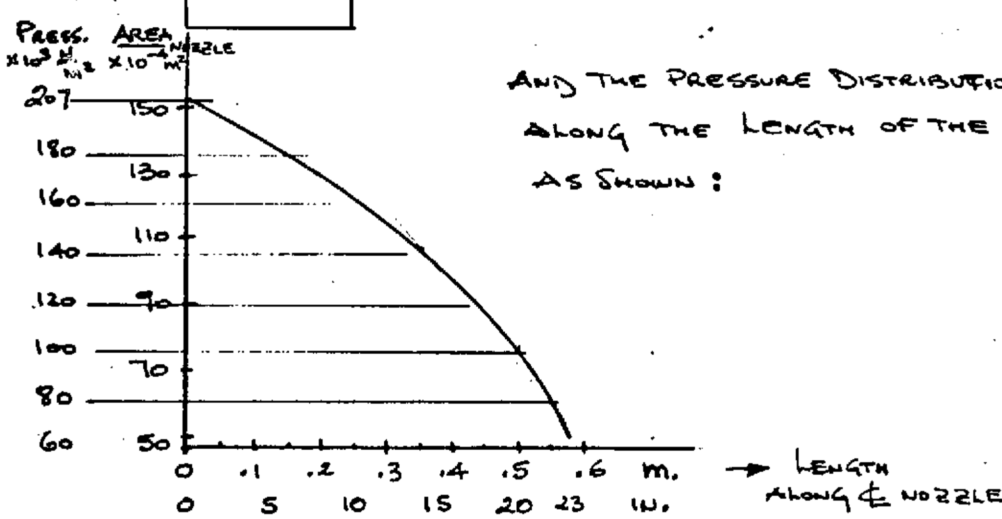
NOZZLE AREA CHANGE UNDER PRESSURE.

$A_{NOZZLE} = 19\left(1.49\times10^{-2} \times .67\times10^{-2}\right)2 + 18\left(.558\times10^{-2} \times .752\times10^{-2}\right)2$
$= 53 \times 10^{-4}\ m^2 \quad (8.22\ \text{IN}^2)$

ASSIMILATE A NOZZLE SECTION TO A PLATE AS SHOWN:



AND THE PRESSURE DISTRIBUTION ALONG THE LENGTH OF THE NOZZLE AS SHOWN:

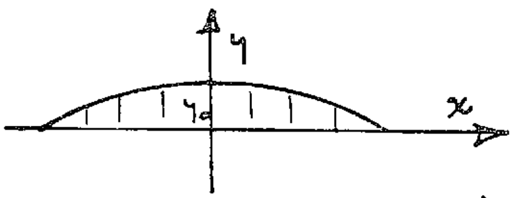


THE AVERAGE PRESSURE IN THAT REGION IS $100 \times 10^3 \frac{N}{m^2}$

WALL DEFLECTION:

$$y = .167 \frac{(10^5)\left(6.67\times10^{-2}\right)^4}{3\times10^7\left(6.9\times10^3\right)\left(.554\times10^{-2}\right)^3} = 92.5\times10^{-7}\ m.$$

FOR AREA CHANGE CALCULATIONS, CONSIDER THE DEFLECTION CURVE TO BE PARABOLIC AS SHOWN:



$$y = y_o\left(1 - \frac{4x^2}{\ell^2}\right)$$

$$A = 2\int_0^{\ell/2} y_o\left(1 - \frac{4x}{\ell^2}\right)dx$$

$$= 2\left[y_o\left(\frac{\ell}{2} - \frac{4\ell^3}{24\ell^2}\right)\right]$$

$$A = 2\left[y_o \frac{\ell}{3}\right] = 2\left[92.5\times10^{-7}\ \frac{6.67\times10^{-2}}{3}\right] = 41.1\times10^{-9}\,m^2$$

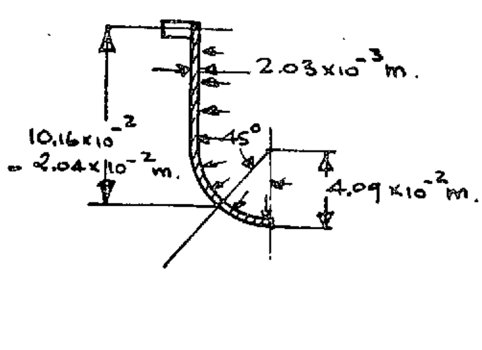
FOR 2 SIDES $\Delta A = 82.2\times10^{-9}\,m^2$

AND FOR 4 SPANS $\Delta A = 328.8\times10^{-9}\,m^2$ ($.00051\ in^2$)

$$\%\ \text{CHANGE} = \frac{32.88\times10^{-8}}{53\times10^{-4}}\times100 = .0062\%$$ NEGLIGIBLE.

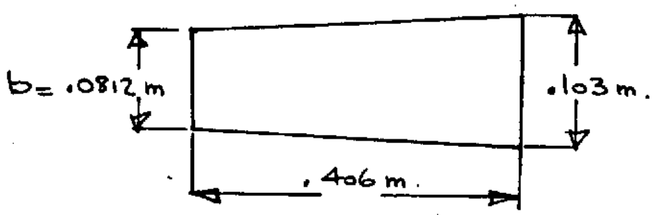
<u>NOZZLE STRESSES</u>, <u>BENDING STRESS</u>

THE NOZZLE FLANKS CAN BE ASSIMILATED TO TWO FLAT PLATES ON EACH SIDE.



THE EFFECTIVE WIDTH OF PLATE IS 8.12×10^{-2}m AT L=0 AND 10.3×10^{-2}m AT L=40.6×10^{-2}m.

FOR A CONSERVATIVE ESTIMATE NOTE THE PRODUCT Pb^2 CAN BE TAKEN AS CONSTANT



CONSIDERING ALL EDGES FIXED WITH MAX. PRESSURE

$$P = 207 \times 10^3 \frac{N}{m^2}$$

$$S_{MAX} = \frac{.5\left(207 \times 10^3\right)\left(.0812\right)^2}{\left(2.03 \times 10^{-3}\right)^2} = 1.655 \times 10^8 \frac{N}{m^2}$$

(24000 psi)

<u>RADIAL</u> <u>STRESS</u> :

PROJECTED AREA : 2rL

$$F = 4.09 \times 10^{-2} (2)(.406)\, 207 \times 10^3 = 6900 \text{ NEWTONS}$$

$$F/_{SIDE} = \frac{6900}{2} = 3450 \text{ N.}$$

CROSS SECTION AREA OF PLATE :

$$2.03 \times 10^{-3} (.406) = 8.26 \times 10^{-4} \, m^2$$

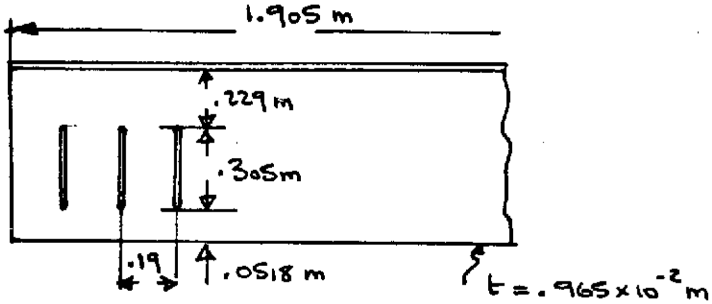
RADIAL STRESS : $\frac{3450}{8.26 \times 10^{-4}} = 4.18 \times 10^6 \frac{N}{m^2}$ (605 psi) NEGLECT

USING CRS 321, YIELD = $2.41 \times 10^8 \frac{N}{m^2}$

$$FS = \frac{2.41}{1.655} = 1.46$$

From previous nozzles of similar configuration & loading this method of analysis has proven to be conservative. The above nozzle with a calculated FS = 1.46 to yield is conservatively loaded compared to similar hardware successfully tested at Boeing facilities.

SLOT NOZZLE   MAT: CRS, YIELD $2.41 \times 10^8 \frac{N}{m^2}$ (35 KSI)



FOR THE UPPER SECTION 1.905 x .229 m. ALL EDGES FIXED

$$S_{MAX} = \frac{.5\,(207 \times 10^3)(.229)}{(.965 \times 10^{-2})\left\{1 + .623\left(\frac{.229}{1.905}\right)^6\right\}} = 57.9 \times 10^6 \frac{N}{m^2} \quad (8400\,psi)$$

FOR THE SECTION .19 x .305 m. ONE SHORT EDGE FREE

ROARK [CASE 48].

$$S_{MAX} = .77\,\frac{(207 \times 10^3)(.19)^2}{(.965 \times 10^{-2})^2} = 62.1 \times 10^6 \frac{N}{m^2} \quad (9000\,psi)$$

FOR THE NOZZLE END .0518 x 1.905 m, ONE LONG EDGE FREE

ROARK [CASE 45].

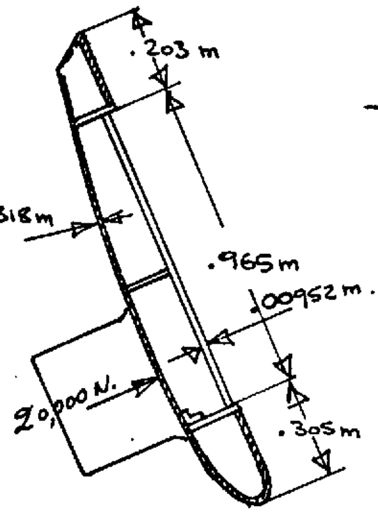
$$S_{MAX} = 3\,\frac{(280 \times 10^3)(.0518)^2}{(.482 \times 10^{-2})^2} = 27.8 \times 10^6 \frac{N}{m^2} \quad (4040\,psi)$$

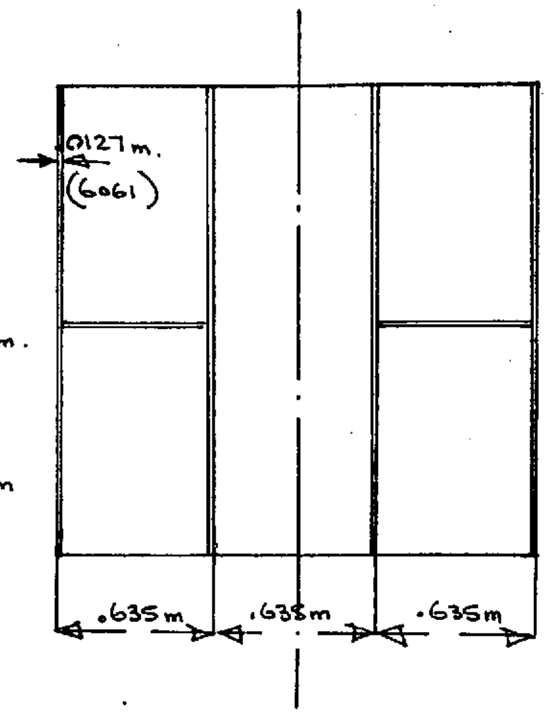
DEFLECTION AT NOZZLE END:

$$y = 1.37\,\frac{(80 \times 10^3)(.0518)^4}{3 \times 10^7 \times 6.9 \times 10^3\,(.482 \times 10^{-2})^3\left(1 + 10\left\{\frac{.0518}{1.905}\right\}^3\right)} = 3.27 \times 10^{-5}\,m \quad (.00128\,in)$$

NEGLIGIBLE CHANGE IN AREA.

FLAP ASSEMBLY.



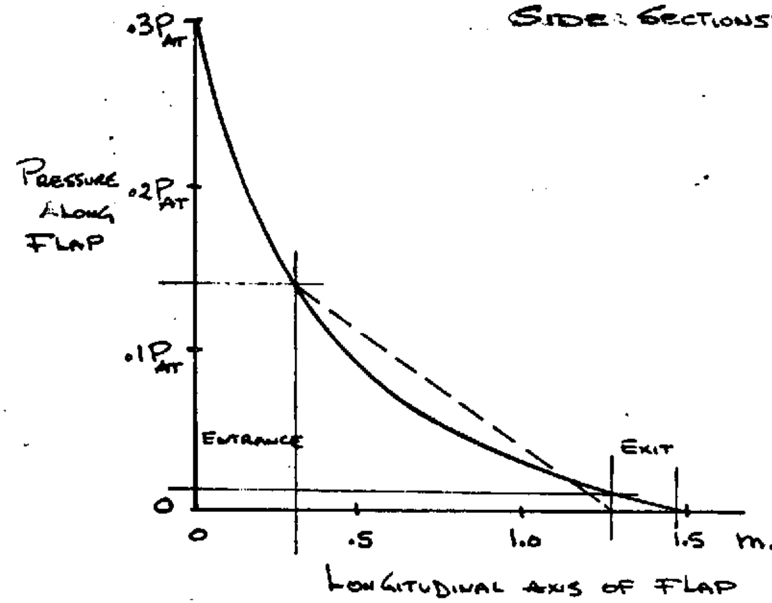


SKIN 2024_T3

YIELD : $3.45 \times 10^{8} \frac{N}{m^2}$

SIDE SECTIONS: $\frac{.965}{2} \times .635$ m.

(.482 x .635 m.)



• THE WORST CASE IS WHEN THE PRESSURE IS ACTING UPON THE SKIN. IF THE SKIN SURFACE IS NOT SEALED, THEN THE PRESSURE WILL ACT ON THE CENTER SECTION (WORST CASE) SIZE .965 x .635 m.

$\frac{b}{a} = \frac{.635}{.965} = .66$ ROARK [CASE 70] $\alpha = .0035$

$\beta_1 = .14$ $w = .14 P_{AT.}$ $t = .00952$ m. $a = .965$ m.

$$S_{b\,MAX} = \beta_1 \frac{wa^2}{t^2} = .14 \frac{.14 (1.013 \times 10^5)(.965)^2}{(.00952)^2} = 2.04 \times 10^7 \frac{N}{m^2}$$

(2960 psi)

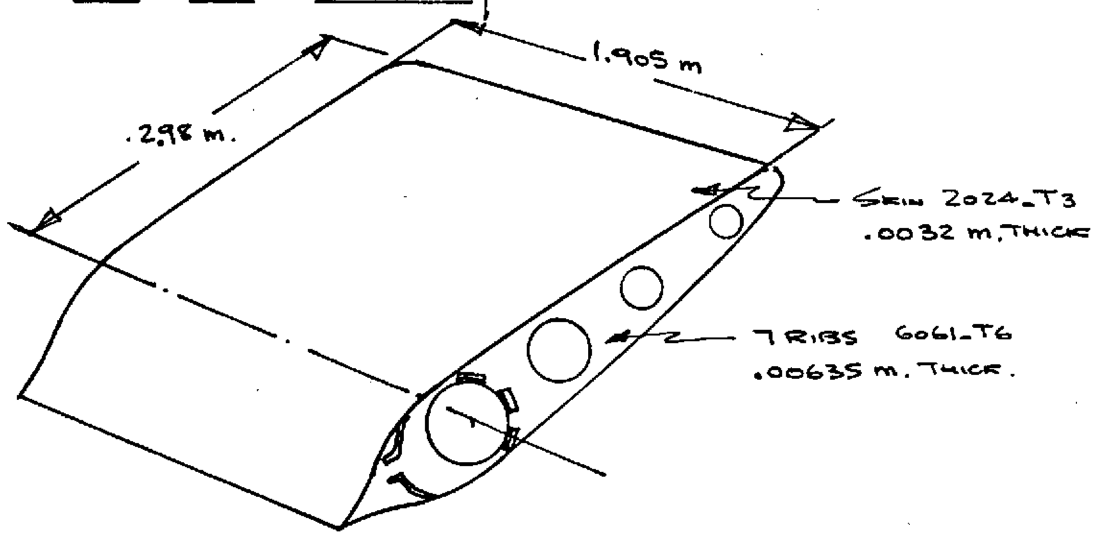
MAX. DEFLECTION :

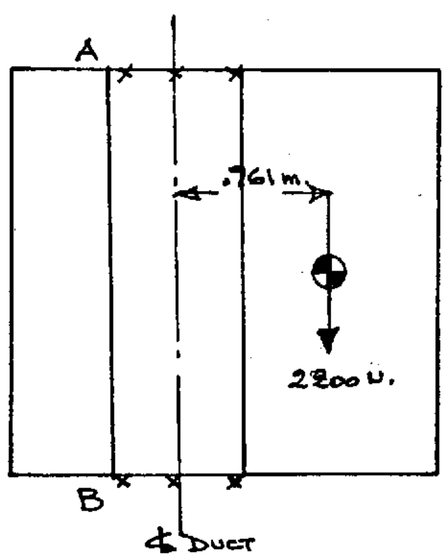
$$y = \alpha \frac{wa^4}{Et^3} = \frac{3.5 \times 10^{-3} (.14)(1.013 \times 10^5)(.965)^4}{\cancel{3} \times 10^7 \times 6.9 \times 10^3 (.00952)^3} = 7.26 \times 10^{-9} \text{m}$$

$(2.86 \times 10^{-7} \text{IN})$

FS > 10.

WING PLANE ASSEMBLY





HORIZONTAL FORCE @ A

$$A_H = \frac{.761 \times 2200}{1.905} = 880 \text{ N.}$$

RESULTANT @ A

$$P_A = \sqrt{1100^2 + 880^2} = 1420 \text{ N.}$$

WELD @ A $\quad P = S_a A$

$$S_a = 78 \times 10^6 \frac{N}{m^2}$$

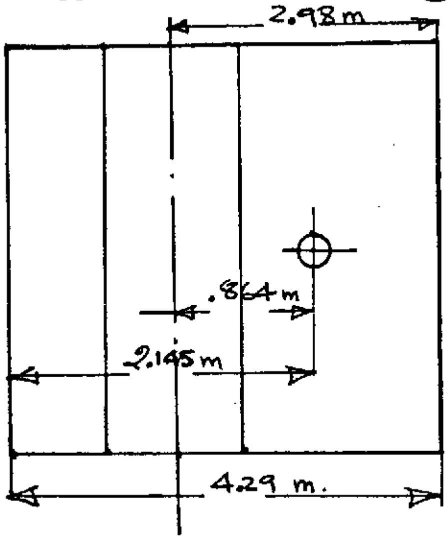
$A = .707\, bL$

$b = .00508$ m.

ALLOWABLE LOAD : $78 \times 10^6 (.707)(.00508)(.0254) = 5150$ N.

$$FS = \frac{5150}{1420} = 3.62$$

ASSUME A 100 MPH WIND BLOWING PERPENDICULAR TO THE WING.



$F_w = \rho A V^2$ $\frac{F_w}{A} = \rho V^2 = (.075 \times 16.02)(100 \times .447)^2$

$= 2405 \frac{N}{m^2}$ (.349 psi).

$F_w = 2405(4.29)(1.905) = 19700$ NEWTONS (4430 lbs)

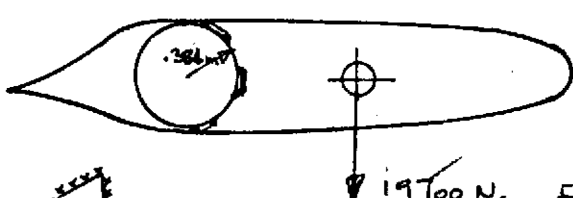
TORQUE ON DUCT: $T = 19700(.864) = 17000$ N-m. (150,200 in.-lbs)

STRESS ON SKIN DUE TO WIND VELOCITY:

$$S_{MAX} = \frac{.5(2405)(.318)^2}{(.00318)^2} = 12.1 \times 10^6 \frac{N}{m^2}$$

STRESS IN WELDS:

WELDS 3 PLACES 6 LOCATIONS



FORCE @ WELDS:

$$\frac{19700 \times .864}{.384} = 44600 \text{ N.}$$

$F_{TOT} = 44600 + 19700 = 64300$ N.

6 WELDS ARE IN TENSION, 6 IN COMPRESSION & 6 IN SHEAR.

$$S_a \approx 89.8 \times 10^6 \frac{N}{m^2} \left(13000 \, psi\right)$$

$$S_{ACTUAL} = 1.618 \frac{64300}{(.00508)(.0254)12} = 67.1 \times 10^6 \frac{N}{m^2} \left(9740 \, psi\right)$$

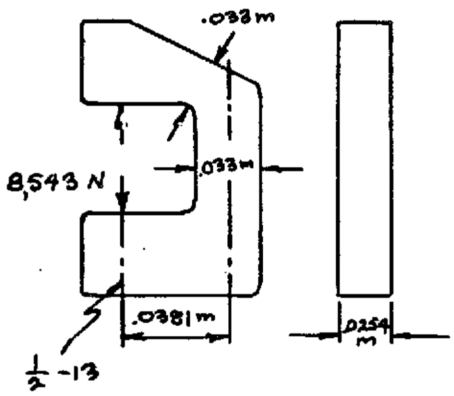
WITH THE ADDITIONAL SHEAR WELDS $FS \approx 2$.

TRANSLATING FRAME ASSEMBLY

Fram Clamp HRS ASTM A-7 Yield $= 2.48 \times 10^{8} \frac{N}{m^2}$

with 16 ft-lbs bolt torque : 21.7 N-m

Separating Load $Q = \frac{(5)(21.7)}{.0127} = 8{,}543\ N$
(1921 lbs)



Section area $= .033(.0254) = 8.38 \times 10^{-4}\ m^2$

$$I = \frac{(.0254)(.033)^3}{12} = 7.60 \times 10^{-8}\ m^4$$

$$S_{direct} = \frac{8543}{8.38 \times 10^{-4}} = 1.019 \times 10^{7} \frac{N}{m^2}$$

$$S_{bending} = \frac{8543 \times .0381 \times .0165}{7.60 \times 10^{-8}} = 7.066 \times 10^{7} \frac{N}{m^2}$$

$$S_{max} = S_{direct} + S_{bending} = 1.019 \times 10^{7} + 7.066 \times 10^{7} = 8.085 \times 10^{7} \frac{N}{m^2}$$

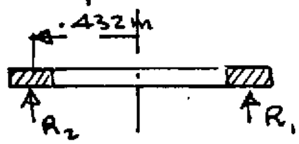
$$FS = \frac{2.48}{.8085} = 3.06$$

BEARING AND SUPPORT ASSEMBLY.

8 PADS .0127 m THICK × .0518 × .1036 m. LONG.
(.5 × 2 × 4 IN.)

$$\text{AREA/PAD} = .0518 \times .1036 = 5.37 \times 10^{-3}\ m^2.$$
$(8\ \text{IN}^2)$

WITH SYSTEM IN STATIC CONDITIONS (NO THRUST)



$R_1 = 8800$ N. $R_2 = 5000$ N.

MAX. BEARING STRESS ASSUMING R_1 CONCENTRATED ON ONE PAD :

$$\frac{8800}{5.37 \times 10^{-3}} = 1.64 \times 10^{6}\ \frac{N}{m^2} \quad (240\ \text{psi})$$

IN OPERATING CONDITION (WITH THRUST)
WHEN CLAMPED :

$R_1 = 25200$ N + CLAMPING F.

$R_2 = -11350$ N + CLAMPING F.

IF CLAMPING F = 12600 N.
+ LOAD IS DISTRIBUTED OVER 4 PADS AND SO IS THE
– LOAD

$$\therefore \text{MAX. LOAD/PAD} = \frac{25200}{4} + 12600 = 18900\ \text{N.}$$

$$\text{MAX. BEARING STRESS} \quad \frac{18900}{5.37 \times 10^{-3}} = 3.53 \times 10^{6}\ \frac{N}{m^2} \quad (516\ \text{psi})$$

$$\text{AVG. LOAD/PAD} : \frac{\frac{25200}{4} + 12600 - \frac{11350}{4} + 12600}{2} = 14300\ \text{N.}$$
(3218 lbs)

THE FRICTIONAL TORQUE RESISTING ANY TWISTING MOMENT ON THE DUCT IS:

$$M = \frac{1}{2} f(\text{LOAD}) \left(R_o + R_i \right)$$

SINCE $\frac{R_o + R_i}{2} = R_m$ $\quad R_o + R_i = 2R_m$

$$\therefore M = \frac{1}{2} f (\text{LOAD}) \, 2R_m$$

WITH $f = .2$

$$M = \frac{1}{2} \cdot .2 \, (14300)(2)(.432)(8) = 9860 \text{ N.m}$$

$$(7280 \text{ ft-lb})$$

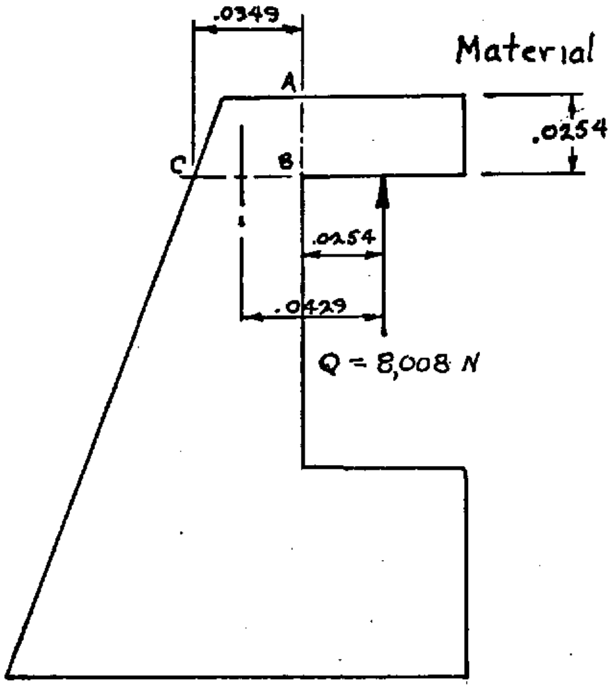
THE TORQUE ON DUCT DUE TO OFFSET NOZZLE EXIT WAS 5220 N.m (3850 ft.-lb.).

THE TORQUE DUE TO 100 MPH WIND ACTING ON THE WING WAS 17000 N.m, IN EXCESS OF THE CLAMPING CAPABILITIES

NOTE:

The operating procedure requires the wing plane to be supported by guy wires whenever the model is not in use and the model is not used in winds that exceed 7 knots. Therefore, all loads that appear exessive due to the assummed 100 mph. wind do not apply.

CLAMP BRACKET



Material 4340 yield $= 5.17 \times 10^8 \frac{N}{m^2}$

Bolt Torque = 20.34 N-m

CLAMP FORCE:

$$Q = \frac{5T}{D} = \frac{5(20.34)}{.0127} = 8{,}008 \ N$$

(1,800 lb)

thickness $= t = .01905$ m

$$I_{A\text{-}B} = \frac{(.01905)(.0254^3)}{12} = 2.6 \times 10^{-8} \ m^4$$

$$\sigma_b = \frac{(.0254)(8{,}008)(.0127)}{2.6 \times 10^{-8}} = .9935 \times 10^8 \ \frac{N}{m^2}$$

(14,400 psi)

$$F.S. = \frac{5.17}{.9935} = 5.20$$

$$I_{B\text{-}C} = \frac{(.01905)(.0349^3)}{12} = 6.75 \times 10^{-8} \ m^4$$

$$\sigma_b = \frac{(.0429)(8{,}008)(.01745)}{6.75 \times 10^{-8}} = .89 \times 10^8 \ \frac{N}{m^2}$$

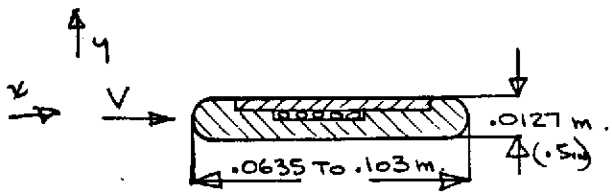
$$\sigma_t = \frac{8{,}008}{(.01905)(.0349)} = .12 \times 10^8 \ \frac{N}{m^2}$$

$$\sigma_{max} = (.89 \times 10^8) + (.12 \times 10^8) = 1.01 \times 10^8 \ \frac{N}{m^2}$$

(14,700 psi)

$$F.S. = \frac{5.17}{1.01} = 5.12$$

Flap Exit Rake Assembly



.866 m Long.

Force on Rake : $F_D = C_D A \dfrac{\rho U^2}{2}$

$A = .0127(.866) = .011\ m^2$

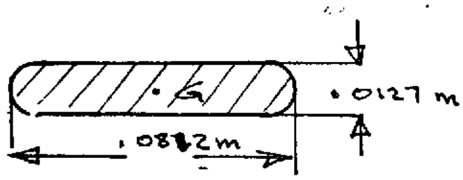
$V = 213\ m/s$ (700 ft/sec)

$C_D \approx .2$

$\rho = 2.33\ \frac{kg}{m^3}$

$F_D = .2\,(.011)\,\dfrac{2.33}{2}(213)^2 = 116.5$ Newtons (26.2 lbs)
(Neglect)

Flutter Calculations . Torsion-Bending.



$V^2 < \dfrac{4K_\theta}{\pi \rho c^2}$

$\ell = .866\ m.$ $C = .0832\ m.$

$\rho = 2.33\ \frac{kg}{m^3}$

$K_\theta = \dfrac{K}{\ell} = \dfrac{JG}{\ell}$ $\quad K = ab^3\left\{\dfrac{16}{3} - 3.36\dfrac{b}{a}\left(1 - \dfrac{b^4}{12a^4}\right)\right\}G$

$a = .0406\ m$
$b = .0063\ m.$
$G = 26.6\times10^9\ \frac{N}{m^2}$

$K = 4.9\times10^{-8}\ m^4 \times 26.6\times10^9$

$K_\theta = 1500.$

$V^2 < \dfrac{4\times1500}{\pi\ 2.33\,(.0832)^2} < 11.85\times10^4 \quad V < 344\ m/s.$

SINCE $V_{ACTUAL} = 213\ m/s$ THERE IS NO FLUTTER PROBLEM.

<u>BENDING STRESS</u>

IN x DIRECTION: $w = \dfrac{116.5}{.866} = 134.5\ N/m$

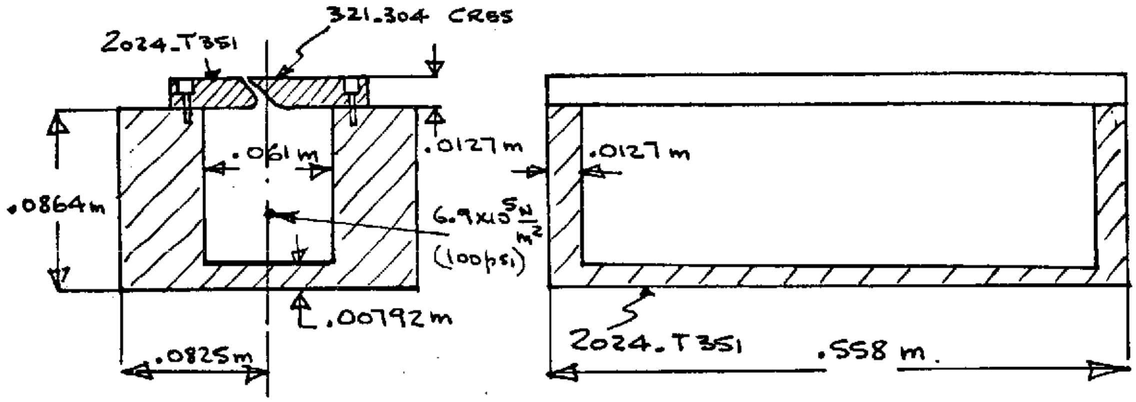
$$M_{MAX} = \frac{116.5 \times .866}{2} = 50.5\ N.m.$$

$$I = \frac{.0127}{12}(.0812)^3$$

$$S_{MAX} = \frac{50.5}{I}(.0406) \approx 2.37 \times 10^6 \frac{N}{m^2} \quad (340\ psi)$$

$$y_{MAX} = \frac{116.5\,(.866)^3}{8 \times 10^7 \times 6.9 \times 10^3\, I} = 12 \times 10^{-5}\ m \quad (.00472\ in.)$$

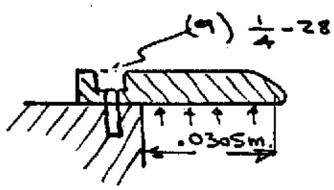
FLAP SYSTEM, SIDE PLATE.



FORCE ON NOZZLE LIPS

$$.061\,(.5326)\,6.9\times10^5 = .224\times10^5\ N. \quad (5040\ lbs)$$

$$\text{FORCE ON ONE LIP} = \frac{.224}{2}\times10^5 = .112\times10^5\ N \quad (2540\ lbs)$$



$$y_{max} = \frac{1}{8}\,\frac{w\ell^3}{\frac{E}{1-\nu^2}\,I} \qquad w = .112\times10^5\ N. \quad \ell = .0305\ m \quad \nu = .3$$

$$I = \frac{.5326}{12}(.0127)^3 = 9.1\times10^{-8}\ m^4$$

$$y_{max} = 4.6\times10^{-5}\ m \quad (.00181\ in.)$$

$$S_{max} = \frac{\frac{w\ell}{2}\,(.0063)}{9.1\times10^{-8}} = 11.9\times10^6\ \frac{N}{m^2} \quad (1725\ psi)$$

$$FS = \frac{3.45\times10^8}{11.9\times10^6} = 29.$$

BOTTOM OF PLENUM:

SIZE: .5326 m LONG × .061 m × .00792 THICK.

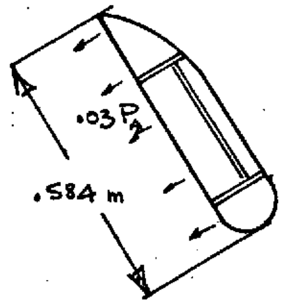
ALL EDGES FIXED, ROARK [CASE 41]

$$\frac{b}{a} = \frac{.061}{.5326} = .1145$$

$$S_{max} = \frac{.5\,(6.9 \times 10^5)(.061)}{.00792} = 2.21 \times 10^7 \ \frac{N}{m^2} \quad (3200\,psi)$$

$$FS = \frac{3.45 \times 10^8}{2.21 \times 10^7} = 15.6$$

INTAKE FLAP ASSEMBLY

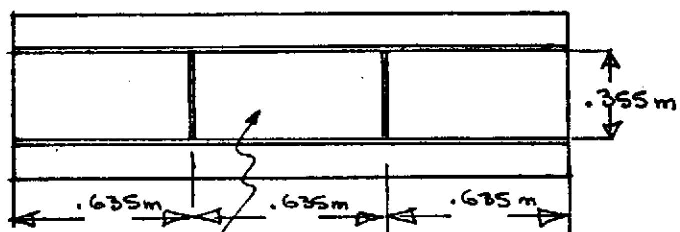


ASSUME .03 $P_{ATM.}$ ALONG THE FULL LENGTH

$$.03\left(1.013\times10^5\right)= 3.04\times10^3\ \frac{N}{m^2}$$

$$F=\left(.635\times3\right)\left(.584\right)3.04\times10^3 = 3.38\times10^3\ N.$$

$$F/_{PANEL} = \frac{3.38\times10^3}{3} = 1.127\times10^3\ N.$$



6061_T651 t = .00635 m

S_{MAX} FROM ROARK [CASE 70]

$$\frac{b}{a}=\frac{.635}{.355}= 1.7$$

$$S_{MAX} = .19\ \frac{\left(3.04\times10^3\right).355}{(.00635)^2} = 17.4\times10^5\ \frac{N}{m^2} \quad (252\ psi)$$

STAND WEIGHTS BREAKDOWN:

TOP PLATE : 6,000 NEWTONS (1346 lbs)

LEGS WT :

ENDS 11.4×10^{-2} m OD , $.602 \times 10^{-2}$ m WALL , 2.28 m LENGTH

4 TUBES + 1 VERT. TUBE = 1780 NEWTONS (400 lbs).

CROSS MEMBERS

4 TUBES + 2 HORIZONTAL MEMBERS : 1425 N. (321 lbs)

SUPPORT BEAMS

BJ6, 6x4

2 BEAMS + 2 CROSS MEMBERS : 466 N. (105 lbs)

BEARING PADS

1780 N. (400 lbs)

BEARING SUPPORT STRUTS

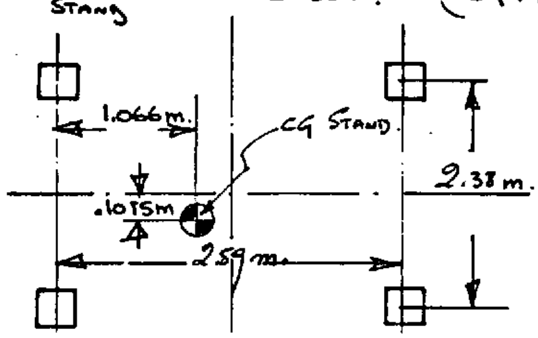
854 N. (192 lbs)

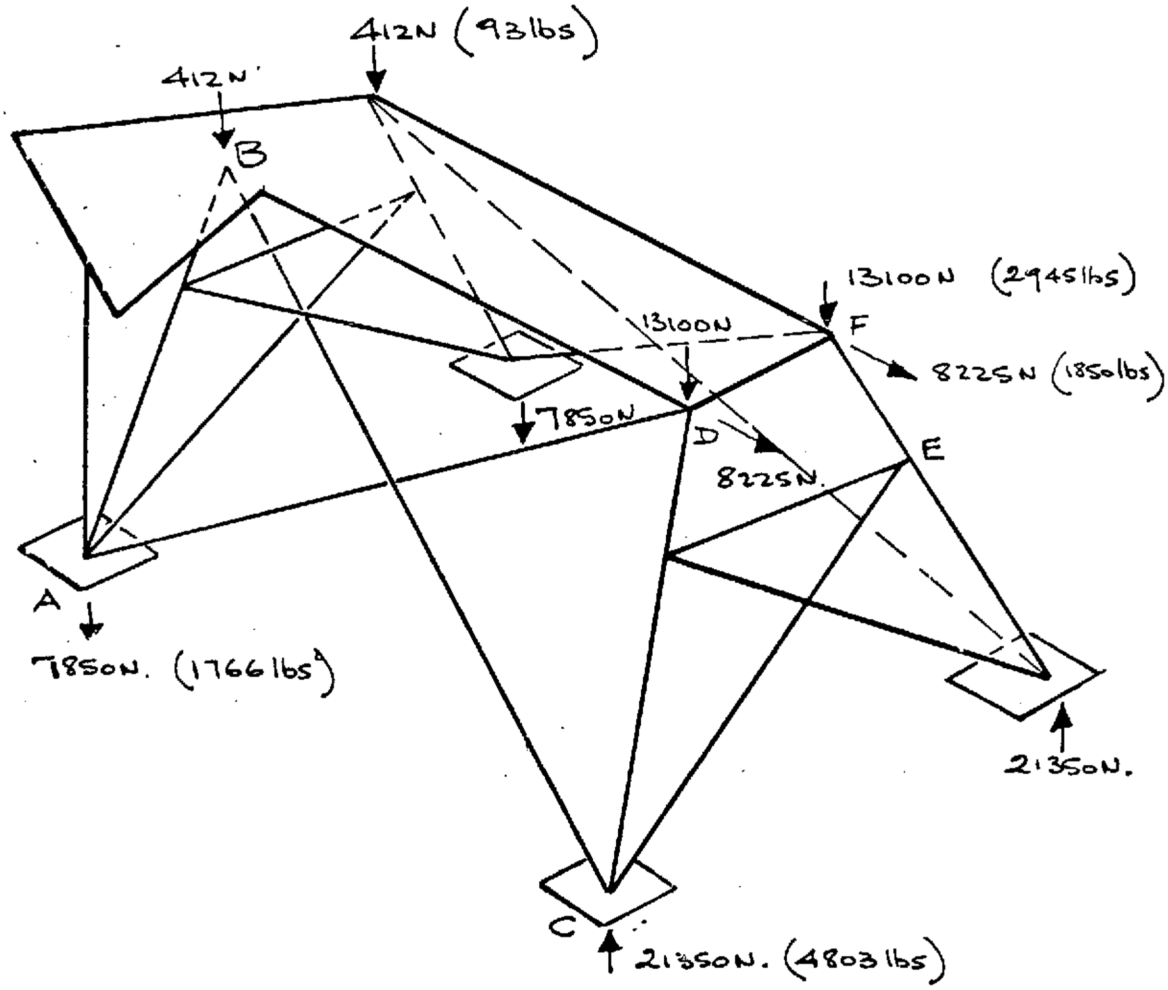
PADS 734 N. (165 lbs)

TOTAL WT : 7050 N. (1583 lbs)

+ TOP PLATE 6000 N. (1346 lbs)

WT_{STAND} : 13050 N. (2949 lbs)





FORCES IN SUPPORT STAND.

FORCES ACTING:	NOZZLE THRUST	16450 N.	(3700lbs).
	NOZZLE ASSY WT.	4900 N.	(1100lbs).
	DUCT ASSY WT.	6780 N.	(1525lbs).
	WING ASSY. WT.	2200 N.	(500lbs).
	SUPPORT STAND WT.	13100 N.	(2949lbs).

FLOUR REACTIONS :

$$\Sigma M_A = 16450\left(194 \times 2.54 \times 10^{-2}\right) + 2200\left(78.5 \times 2.54 \times 10^{-2}\right)$$
$$+ 4900\left(21.5 \times 10^{-2} \times 2.54\right) + 13100\left(1.066\right) - R_C$$
$$\left(102 \times 2.54 \times 10^{-2}\right) = 0$$

$\uparrow R_C = 42700N.$ (9605 lbs) FOR TWO LEGS.

TOTAL ASSY WT : $4900 + 6780 + 2200 + 13100 = 27000N$ (6074 lbs)

$\therefore \downarrow R_A = 42700 - 27000 = 15700$ N. (3531 lbs) FOR TWO LEGS

TOP PLATE FORCES :

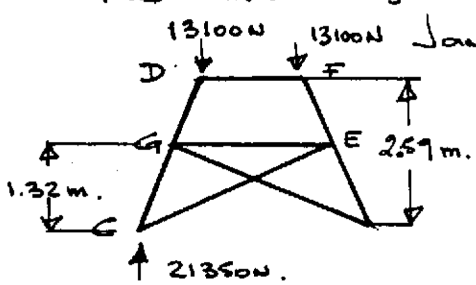
$$\Sigma M_A = 16450\left(92 \times 2.54 \times 10^{-2}\right) + 2200\left(78.5 \times 2.54 \times 10^{-2}\right)$$
$$+ 4900\left(21.5 \times 10^{-2} \times 2.54\right) + 13100\left(1.066\right) - R_D$$
$$\left(102 \times 2.54 \times 10^{-2}\right) = 0$$

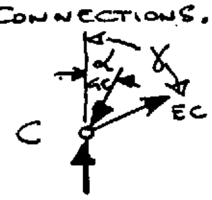
$\downarrow R_D = 26200$ N. (5890 lbs) FOR TWO LEGS

$\downarrow R_B = 823$ N. (185 lbs) FOR TWO LEGS

SINCE THE FRAME IS BOLTED TO THE FLOOR THE SYSTEM IS INDETERMINATE AND THE FORCES IN THE VARIOUS MEMBERS CANNOT BE CALCULATED.

TO ESTIMATE THE MAXIMUM LOADED MEMBER CONSIDER THE RIGHT HAND SECTION CDFE AND ASSUME PIN JOINT CONNECTIONS.





$\tan\alpha = \frac{30}{102} = .294 \quad \alpha = 16.4^{\circ} \qquad \cos\alpha = .959 \quad \sin\alpha = .266$

$\tan\gamma = \frac{78.7}{52} = 1.515 \quad \gamma = 56.5^{\circ} \qquad \cos\gamma = .55 \quad \sin\alpha = .834$

AT JOINT C: $EC\,(.834) = GC\,(.266) \qquad EC = .319\,GC$

$GC\,(.959) - EC\,(.55) = 21{,}350\ N.$

$\therefore \quad GC = \frac{21350}{.784} = 27300\ N.$ MAX. LOAD. (6130 lbs)

$EC = 8700\ N.$

THE LEGS ARE 4 in. PIPE SCH. 40. : .1141 m OD

.1024 m. ID.

$$A_{SECTION} = \frac{\pi}{4}\left(.1141^2 - .1024^2\right) = 20.5 \times 10^{-4}\ m^2.$$

CHECK FOR BUCKLING

$I = 29.8\ m^4 \times 10^{-8}$

$$k = \sqrt{\frac{I}{A}} = .0381\ m.$$

$$\frac{L}{k} = \frac{52\,(2.54 \times 10^{-2})}{.0381\,(.959)} = 36.2$$

WITH A YIELD STRENGTH OF $2.07 \times 10^8 \frac{N}{m^2}$ (30000 psi)

THE L.B. JOHNSON FORMULA YIELDS:

$$\frac{P_{cr}}{A_{SECT.}} = S_y \left\{ 1 - \frac{S_y \left(L/k \right)^2}{4C\pi^2 E} \right\}$$

FOR $C = 1 \qquad \frac{P_{cr}}{A} \approx 1.93 \times 10^8 \frac{N}{m^2}$

$\therefore \quad P_{cr} = 1.93 \times 10^8 \times 20.5 \times 10^{-4} = 39.6 \times 10^4\ N.$

(89000 lbs).

$FS = \frac{39.6}{2.73} = 14.5$

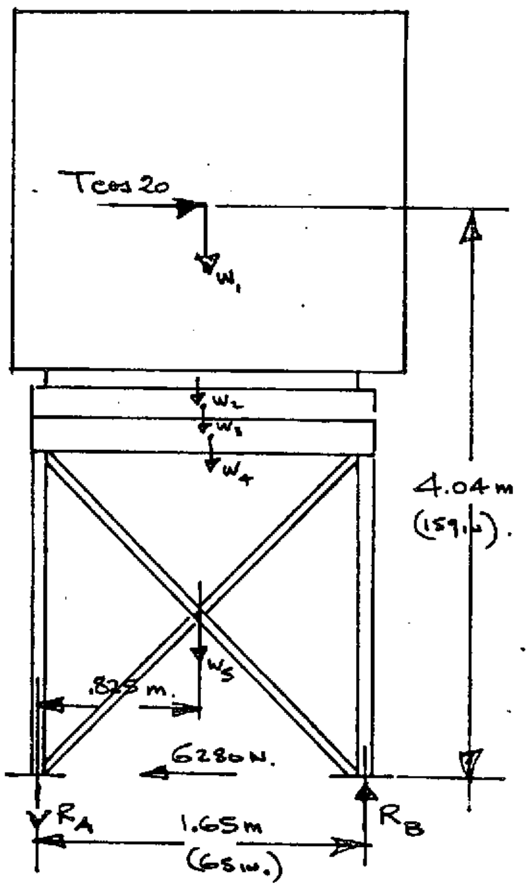
SUPPORT FRAME.

WEIGHT BREAKDOWN:

SIDE PLATES		7070 N.	1554 lbs
MAIN FRAME			
	LONG BEAM	2950	663
	SHORT BEAM	837	184
	CROSS MEMBERS	418	94
LEGS	4 VERTICAL	1125	253
	4 CROSS MEMB.	1200	270
	2 SIDE CROSS MEM.	578	130
X FRAME	SHORT BEAMS	480	108
	LONG BEAMS	725	163
Y FRAME	SHORT BEAMS	392	88
	LONG BEAMS	458	103
TOTAL		16080 N.	3610 lbs.
FLAPS + SHROUD		2670 N	600
		18750 N.	4210 lbs.

+ BRACKETS, PAD, MISCELLANEOUS

TAKE 20,000 N (4500 lbs.).



$W_1 + W_2 + W_3 + W_4 + W_5 = 20{,}000 \text{ N}.$

$T \cos 20 = 6670(.94) = 6280 \text{ N}.$

GROUND REACTIONS

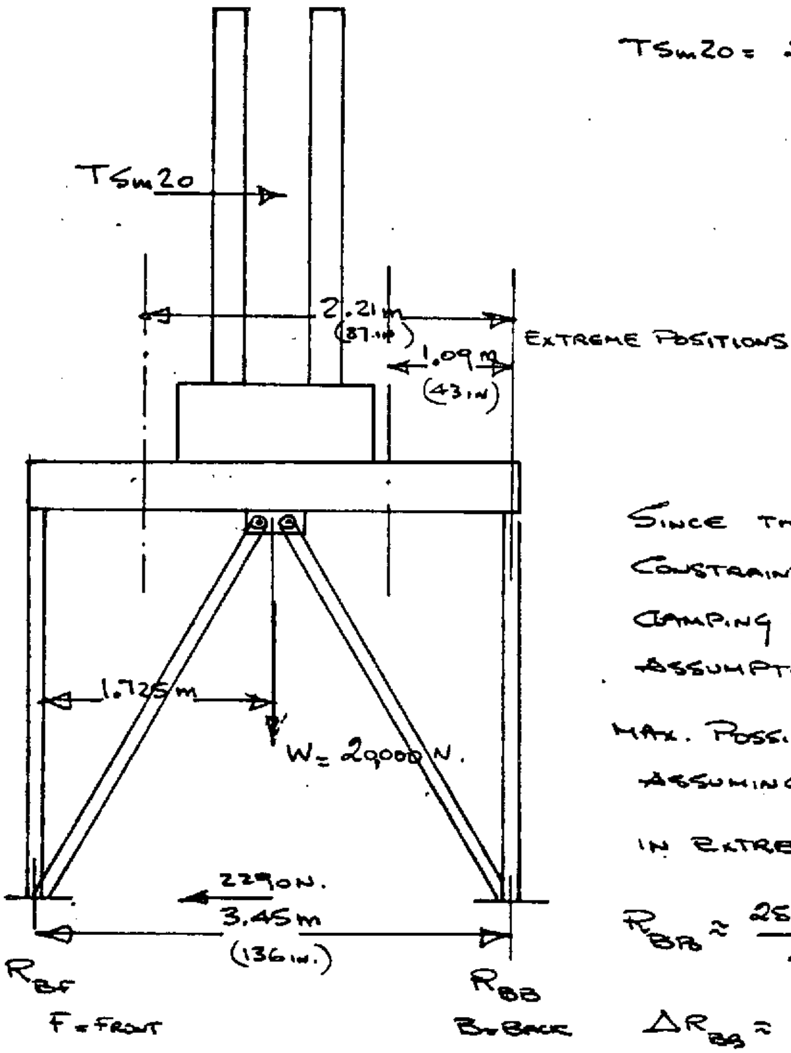
$\Sigma M_A = 20{,}000(.825) + 6280(4.04)$
$\quad - 1.65 R_B = 0$

$\uparrow$
$R_B = 25{,}400 \text{ N} \quad (5700 \text{ lbs}).$

$\Sigma M_B = -20000(.825) + 6280(4.04)$
$\quad - 1.65 R_A = 0$

$\downarrow$
$R_A = 5{,}400 \text{ N}. \quad (1200 \text{ lbs}).$

TSm20 = 2290 N.



SINCE THERE ARE REDUNDANT CONSTRAINTS, BECAUSE OF CLAMPING TO THE FLOOR, ASSUMPTIONS ARE NECESSARY

MAX. POSSIBLE REACTION ASSUMING PIN CONNECTIONS:

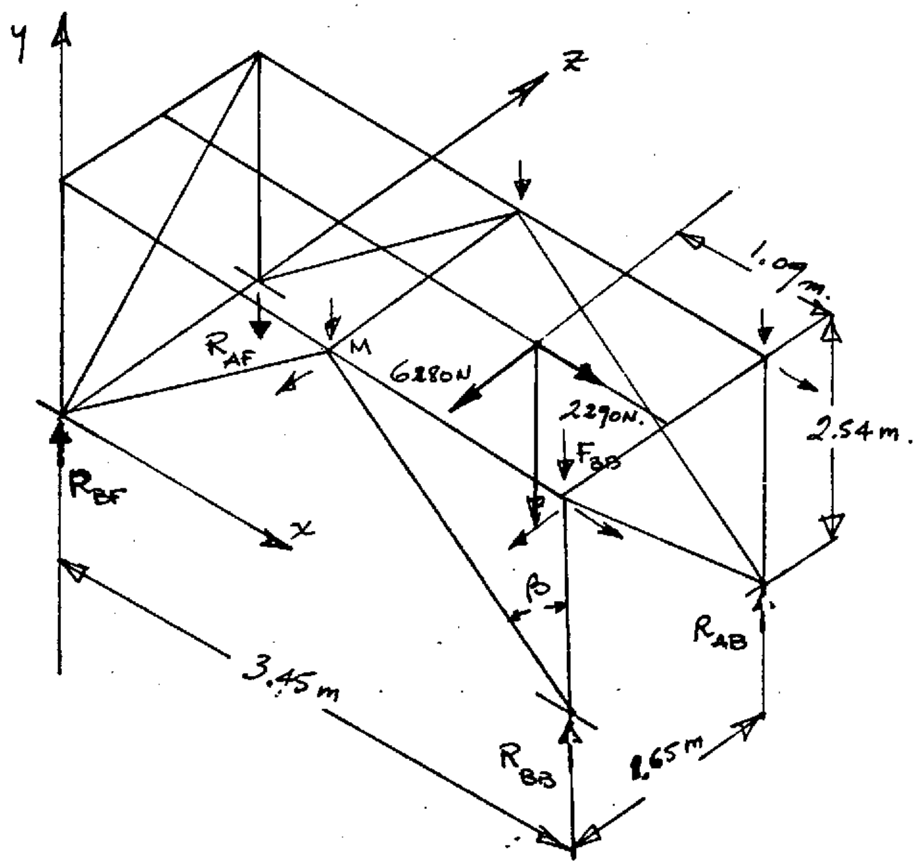
IN EXTREME POSITION TO THE RIGHT

$$R_{BB} \approx \frac{25400}{2} + \Delta R_{BB}$$

$$\Delta R_{BB} \approx 2290(4.04) + (20000-8000)(1.725-1.09)$$

$$\approx 4900\ \text{N}/2$$

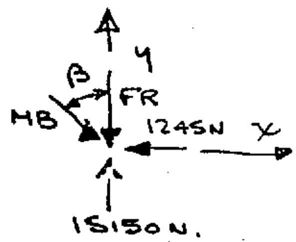
$$\therefore R_{BB} \approx 12700 + 3450 = 15150\ \text{N.}\ (3400\ \text{lbs})\ \text{MAX.}$$



FOR A CONSERVATIVE ESTIMATE OF MAX. LOADS, CONSIDER THE LOADS CONCENTRATED ON THE TOP RIGHT HAND SECTION.

$F_{BB} \approx 3000 + 1420 + 500 + 3970 \approx 8890N.$ (2000 lbs) MAX.

WHICH SHOWS THAT THE LOADS ARE RELATIVELY LIGHT.



IF WE ASSUME THAT AT BB THE JOINT IS PINNED.

$\tan\beta = \frac{1.725}{2.54} = .68 \quad \beta = 34.2^\circ$

$\cos\beta = .826$

$\sin\beta = .562$

$\Sigma F_y = 15150 - FR - MB(.826) = 0$

$\Sigma F_x = MB(.562) - 1245$

$\therefore$ MB = 2220 N. (500 lbs). NEGLECT.

FR = 13318 N. MAX. (3000 lbs) NEGLIGIBLE STRESS.

APPENDIX B

SYMBOLS AND ABBREVIATIONS

NPR	nozzle pressure ratio
δ_T	augmentor flap air turning angle, $\delta_F - \delta_N$, degrees
θ_D	augmentor diffuser angle, degrees
A_3/A_1	augmentor throat area/nozzle exit area
θ_1	augmentor intake angle, degrees
ϕ	thrust augmentation ratio, flaps-on thrust/flaps-off thrust
L/h_e	augmentor length/equivalent slot nozzle height
h_e	equivalent slot nozzle height
δ_f	flap rotation angle with respect to WCP, degrees
δ_N	augmentor primary nozzle deflection angle with respect to WCP, degrees