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SYMBOLS

cm	centimeter(s)
℄	centerline
dB	decibel: $20 \log_{10} \frac{p_1}{p_2}$
DB RE 0.0002 MICROBAR:	$20 \log_{10} \frac{p_3}{0.0002 \text{ MICROBAR}}$
ft	foot, feet
Hz	Hertz (cycles/second)
in	inch(es)
kg	kilogram(s)
kHz	kilohertz
kn	nautical miles/hour
lb	pound(s)
m	meter(s)
MICROBAR	sound pressure level equal to 0.1 N/m^2 and approximately equal to one-millionth of normal atmospheric pressure
N	Newtons
NH	High pressure compressor RPM
p_1	sound pressure level under evaluation, any units of pressure
p_2	reference sound pressure level, same units as p_1
p_3	sound pressure level under evaluation, MICROBARS
rms	root mean square
V	velocity, kn
Vmo	Velocity, maximum operational
^{o}C	degrees Celcius
^{o}F	degrees Fahrenheit
δf	flap angle, degrees (see Figure 3)
δch	choke angle, degrees (see Figure 3)
δn	conical nozzle angle, degrees (see Figure 2)

STATIC NOISE TESTS ON MODIFIED AUGMENTOR WING JET STOL RESEARCH AIRCRAFT

G. R. Cook and B. F. Lilley*

Ames Research Center

SUMMARY

Noise measurements were made in 1978 at the Ames Research Center on the U.S./Canadian C-8A Augmentor Wing Jet STOL Research Aircraft (AWJSRA) to determine if recent modifications made to the bifurcated jetpipe to increase engine thrust had at the same time reduced the noise level. The noise field was measured by a 6-microphone array positioned on a 30.5m (100ft) sideline between 90 and 150 degrees from the left engine inlet. Noise levels were recorded at three flap angles over a range of engine thrust settings from flight idle to emergency power and plotted in one-third octave band spectra. Little attenuation was observed at maximum power, but significant attenuation was achieved at approach and cruise power levels.

INTRODUCTION

Early flight-test activity at the NASA-Ames Research Center indicated excessive noise from the AWJSRA. Although the noise aggravation to local communities could be significantly reduced by operational changes, the two governments were interested in developing a detailed understanding of the noise characteristics to provide a firm technology base to reduce the noise level of this type of configuration.

Initial noise measurements of the AWJSRA were made in mid-1972 by Boeing at Paine Field, Washington, during the airworthiness flight test program prior to delivery to Ames. A more detailed series of tests jointly sponsored by the U.S. National Aeronautics and Space Adminstration (NASA) and the Canadian Department of Industry Trade and Commerce (DITC), were conducted at Ames during July and August 1973.

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The AWJSRA is a low cost proof-of-concept research aircraft designed to demonstrate the augmentor wing concept. The propulsion system consists of two Rolls Royce turbofan engines, formerly Spey Mk 511, modified to separate the fan air from the core air and redesignated Mk 801-SF (split flow). The fan air is ducted to the wing augmentor, and the core air is exhausted through a bifurcated jetpipe and nozzle vectoring system taken from a Pegasus (Harrier) engine. The mismatch between the cross sectional area of the Pegasus and Spey jetpipes was accommodated by a perforated (colander) plate located between the two. This effectively protected the engine from any disturbances within the oversized Pegasus bifurcated jetpipe or nozzle system. However, this low cost approach resulted in a thrust loss of about 10% and was suspected as the source of much of the noise.

The static tests conducted in July and August 1973 investigated the following areas:

- The noise contribution from the colander plate by testing configurations with and without the plate installed.

- The potential for noise reduction by running with over-area nozzles to reduce the core velocity.

- The potential for noise reduction by replacing the existing conical nozzles with multi-lobed nozzles.

- Engine case treatment.

- The effect of flap angle on sound directivity.

Tests with the multi-lobed nozzles and over-area nozzles (colander plate removed) showed some noise reduction but not as much as predicted. Engine case treatment showed little appreciable change.

Since these unusual noise-level characteristics are not associated with civil versions of the Spey engine, it was concluded by Rolls Royce that they must be associated with the bifurcated jetpipe and could be either generated internally within the jetpipe or externally as "modified" jet noise owing to the airflow profile ensuing from the nozzles. With either case an improved aerodynamic design of the jetpipe might be beneficial in reducing noise.

The opportunity to redesign the jetpipe came in 1977, when the U.S. and Canadian governments agreed to an extension program beyond the original 500 hours planned, requiring an extensive overhaul of the engines and airframe. A significant growth in the weight of the research aircraft combined with summer operations in high ambient temperatures led to the desirability of gaining extra thrust for improved engine out performance, in addition to reducing the noise level.

DeHavilland designed and built a "floating" nonload-carrying liner that fitted inside the Pegasus jetpipe and matched the cross sectional area of the aft end of the Spey to provide an improved aerodynamic path that eliminated the need for the colander plate. New nozzles were also required and they were designed slightly over-area, trading some of the potential thrust gain for reduced core velocity and reduced noise.

The new hardware was flight-qualified by Rolls Royce in test cell running at Montreal, demonstrating a thrust gain of about 7%. The overhauled engines and modified jetpipes were returned to Ames in March 1978.

Acoustic measurements to determine the magnitude of the noise reduction were made on November 3, 1978, approximately five (5) months after the research aircraft resumed its flight-test program.

AIRCRAFT DESCRIPTION

Configuration

The AWJSRA is a modified version of the high-wing, T-tail Buffalo turboprop transport manufactured by deHavilland Aircraft of Canada, Ltd., modified jointly by the Boeing Commercial Airplane Company and deHavilland. It is used to study the operational characteristics of a jet STOL aircraft using turbofan engines to provide both aerodynamic powered lift from an augmentor wing flap and vectored propulsive lift from rotatable nozzles. A description of the aircraft is given in table 1 and figure 1. Details of the flight characteristics of the aircraft are given in references 1 and 2. Several features of its powered lift and propulsion system are briefly described below, while a more complete description of the aircraft design features is given in reference 3.

Propulsion System

Two Rolls Royce Spey Mk 801-SF (Split-Flow) turbofan engines, one mounted in each nacelle as shown in figure 1, provide thrust for the aircraft as well as air for the augmentation system. These are hybrid engines modified by Rolls Royce from the Spey Mk 511-8 specifically for this application with a 0.6 bypass ratio and a maximum cold flow pressure ratio of 2.5. The engine hot flow is discharged into a Pegasus bifurcated jetpipe and out through two vectorable conical nozzles as shown in Figure 2, while the cold flow is collected and discharged through two 33 cm (13 in) diameter ducts located at the top of the engine, which supply the distribution system described below.

Air Distribution System

The distribution system directs the engine cold flow air to the augmentor nozzles, to the aileron nozzles and to the fuselage boundary layer blowing nozzles, as shown in figure 3.

A crossover ducting system directs approximately 64 percent of the bypass mass flow along the front of the wing and across the interior of the fuselage to the augmentor and aileron nozzles on the opposite side of the airplane and to half of the fuselage boundary layer blowing nozzles; the remaining 36 percent of the flow is directed back to the augmentor nozzles on the same side as the engine. Of the 64 percent of the engine mass flow carried by the crossover ducting system, approximately 7 percent is used for fuselage blowing, 44 percent by the augmentor nozzles, and the remaining 13 percent by the aileron boundary layer control (BLC) nozzles. This flow distribution is summarized in figure 4 for the left-hand engine.

Jetpipe Modification

The 5171 kg (11,400lb) thrust Spey 801-SF turbofan engines are fitted with a bifurcated jetpipe and nozzle vectoring system designed for the Pegasus (Harrier) engine of nearly twice the thrust. The mismatch in size between the Spey and Pegasus jetpipes was accommodated by a perforated (colander) plate located between the two that reduced the thrust by approximately 10%. In early 1978 (during a 500 hour engine overhaul), a floating nonload-carrying liner was fitted inside the Pegasus jetpipe to match the aft end of the Spey and provide a more ideal aerodynamic path, eliminating the need for the colander plate. New conical nozzles with an exit area of 1936 cm^2 (300 in^2) per pair were fitted in place of the original nozzles of 2290 cm^2 (355 in^2). These were slightly larger than optimum in order to trade some of the potential thrust gain for a slightly lower exhaust gas velocity and a correspondingly lower noise level. The resulting thrust levels are shown in figure 5 and represent approximately a 7% thrust gain.

ACOUSTIC TESTS

The AWJSRA near field noise levels were measured at the Ames Research Center's static test site, which is located away from the Center's main buildings.

Equipment and Installation

The noise data was measured with 6 Bruel and Kjaer 1.27 cm (0.5 in) diameter type 4134 condenser microphones and recorded on a Sangamo Sabre III tape recorder. The microphones were positioned as shown in figure 6. All of the microphones were equipped with a windscreen; however, no appreciable flow was observed over or around them during the test.

In order to eliminate multi-path effects and to facilitate comparison with previous test data, a measurement technique developed at Boeing (reference 4) was used with the microphones placed 1.27 cm (0.5 in) above a metal plate which was cemented to the concrete pad (figure 7). By placing the receiver close to the reflecting surface, the difference in path lengths for the direct and reflected signals becomes negligible for all frequencies of interest in these tests.

A detailed list of the data acquisition equipment used is presented in table 2. Each of the microphone cables was the same length, 305 m (1000 ft).

Calibration Procedure

The tape recorder was calibrated for frequency response, signal to noise ratio, and maximum undistorted signal level in a routine manner. The microphone amplifiers and associated cables were calibrated for frequency response by substituting a Hewlett-Packard mod. 204-B battery-powered oscillator for the microphone of each channel. The response was checked over the range of 50-10,000 Hz. Each microphone and associated amplifier was given an overall sensitivity calibration by means of a Bruel and Kjar pistonphone, before and after the aircraft acoustic measurements. The pistonphone is a mechanical device with a chamber into which the microphone is inserted for calibration. It produces a known, repeatable sound pressure at the microphone diaphram.

Test Procedure

During the test only the left-hand engine was operated. The ground electrical power and air start carts were removed from the vicinity of the aircraft after the engine had been started.

Each microphone channel was calibrated just prior to starting the engine. The engine was started and allowed to run for a few minutes at ground idle until its operating parameters stabilized and then the speed was advanced slowly to emergency power. At full engine power the amplification factor of each microphone amplifier was adjusted, in 10dB increments, to provide the maximum undistorted output voltage.

Data was taken at seven power settings between flight idle and emergency power, and at three flap angles. The nozzles were set at their minimum deflection of 6 degrees below the horizontal. A detailed list of engine operating parameters is given in Table 3. The engine was stabilized at each power setting and then data was recorded for approximately one minute.

Corrections

Deviations from uniform frequency response over the range of 50-10,000 Hz and deviations from constant sensitivity (vs. time) for the complete system including: microphones, amplifiers, cables, tape recorder and spectrum analyzer were negligible, so no corrections were necessary for these factors.

Test limits for weather, as established by Boeing Aircraft for their 1973 tests (page 31 of reference 6) are shown in figure 8. Since the attempt of this test was to duplicate the test conditions of the Boeing 1973 test, this test used the same weather test limits. Actual conditions were well within the limits and are given in Table 4. No weather-related corrections were made to the sound measurements.

The special microphone-holding fixture described under the previous "Equipment and Installation" section causes the microphone to have a response over the frequency range of interest, 50-10,000 hz, which is 6 db greater than it would have at the same position under free field conditions. To convert to free field levels, it is only necessary to subtract 6 db from the indicated level at any third octave band of interest. In order to be able to make direct comparisons with similarly obtained data from the 1973 tests, this 6 dB subtraction for free field conditions was not performed.

Data Processing

Each measurement was spectrum analyzed by a General Radio 1/3 octave analyzer using 16-second averaging time. The data analysis equipment that was used is listed in Table 5.

Discussion of Results

It should be noted that since the engine hot thrust power settings were chosen for comparison with earlier acoustic data, and since the modified engine has some 7% more thrust than before, the individual data points do not correspond exactly with current operating limits. The 590 kg (1300 lb) thrust (85% NH) may be considered as flight idle, 2177 kg (4800 lb) thrust (93.8% NH) is typical of approach power, 2722 kg (6000 lb) thrust (96% NH) maximum continuous and 2948 kg (6500 lb) thrust (97.2% NH) is a little below normal takeoff. The single measurement at 3220 kg (7100 lb) thrust (99% NH) is well below maximum emergency power.

Only 2 to 3 dB reduction in sound level is evident between the maximum thrust level measured 3220 kg (7100 lb) and maximum continuous 2722 kg (6000 lb) as may be seen in all of figures 9 through 14. At lower thrust levels the sound level falls off quite rapidly as may be seen in all of figures 9 through 29.

The maximum sound levels were recorded between 120 and 130 degrees from the engine inlet, with a fairly sharp fall off in sound level going from 140 to 150 degrees on the 30.5 m (100 ft) sideline. Further reduction in sound level is observed aft of the aircraft by lowering the flaps to 53 degrees.

The frequency spectrum of maximum sound intensity is shifted downward, progressing from the 90-degree position to 150 degrees. At 90 degrees it has a broad plateau between 500 Hz and 3 kHz, whereas at 150 degrees it is concentrated in the 100 Hz to 500 Hz range depending on flap angle. A peak (or tone) is frequently but not always present in the 1/3 octave band centered at 125 Hz.

The maximum sound level recorded in the 1973 tests of the 2290 cm^2 (355 in^2) conical nozzles (reference 6) was at 110 to 120 degrees to the engine inlet. By comparing data from the 1973 and 1978 tests (Figures 27 through 29) at 120 degrees which is close to the maximum sound intensity for both the old and new jetpipe configurations, it is apparent that there is little attentuation at the higher thrust settings but significant attenuation at the lower thrusts.

Conclusions

a. The jetpipe modification, including removal of the colander plate, the inclusion of a liner to improve the aerodynamic flow in the Pegasus "trouserpiece", and slight over-area conical nozzles, have resulted in little noise attenuation at maximum power but significant attenuation at approach power.

b. The maximum sound levels were recorded at 120-130 degrees from the engine inlet, which is about 10 degrees further aft than was found with the earlier jetpipe configuration. This is consistent with the smaller nozzle exit area, 1936 cm^2 vs 2290, (300 in^2 vs 355) and the correspondingly higher exhaust gas velocity.

c. The effect of flap angle on the sound directivity from the nozzles remained unchanged.

References

1. Quigley, Hervey C., Innis, Robert C., and Grossmith, Seth: A Flight Investigation of the STOL Characteristics of an Augmented Jet Flap STOL Research Aircraft. NASA TM X-62334, 1974.

2. Richard F. Vomaski, Robert C. Innis, Braan E. Swan, and Seth W. Grossmith: A Flight Investigation of the Stability Control and Handling Qualities of an Augmented Jet Flap Aircraft. NASA Technical Paper 1254, 1978.

3. Ashelman, R.H. and Shaudahl, H.: The Development of an Augmentor Wing Jet STOL Research Aircraft (Modified C-8A). NASA CR-114503, 1972.

4. Use of Ground Level Microphones to Acquire Static Free-Field Data. Boeing Document D6-40330.

5. Tanquchi, H.H. and Rasmussen, G.: Selection and Use of Microphones for Engine and Aircraft Noise Measurements. Sound and Vibration, February 1979, pp. 15 and 16.

6. C.C. Marrs, D.L. Harkonen and J.V. O'Keefe: Static Noise Tests on Augmentor Wing Jet STOL Research Aircraft (C-8A Buffalo). NASA CR-137520, 1974.

TABLE 1.- C-8A AUGMENTOR WING JET STOL RESEARCH AIRCRAFT CHARACTERISTICS

Weights kg (lb)		
Maximum gross	21,800	(48,000)
Maximum (STOL) landing	20,500	(45,000)
Operational empty	14,800	(32,600)
Maximum fuel	6,350	(14,000)
Areas, m^2 (ft^2)		
Wing area	80.36	(865)
Wing Flap area (including ailerons)	17.38	(187.1)
Horizontal tail area	21.65	(233)
Vertical tail area	14.12	(152)
Dimensions and General Data		
Wing m (ft)		
Root Chord	3.83	(12.58)
Tip chord	2.36	(7.74)
Mean aerodynamic chord	3.68	(12.1)
Sweepback at 40% chord, deg	0.0	
Dihedral, outer wing only, deg	5.0	
NOTE: Wing taper and dihedral each start 5.36 m (17.6 ft) from plane of symmetry		
Aspect ratio	7.2	

TABLE 1.- Concluded.

Horizontal tail m (ft)			
Root chord		2.54	(8.33)
Mean aerodynamic chord		1.91	(6.25)
Sweep of leading edge, deg		4.8	
Dihedral, deg		0.0	
Aspect ratio		4.4	
Vertical tail m (ft)			
Span		4.14	(13.6)
Root Chord		4.27	(14.0)
Mean aerodynamic chord		3.47	(11.41)
Sweep of leading edge, deg		22.6	
Aspect ratio		1.2	
Overall length (with 16ft nose boom), (m) ft		28.44	(93.32)

Control Surface Deflections	Symbol	
Flaps	δf	5.6° to 72° (Figure 3)
Conical nozzles	δn	6° to 100° from fuselage waterline (Figure 2)
Ailerons		±17° about + 35° max droop angle
Spoilers		-50°
Augmentor choke		55% choke gap area closure at 70° flap deflection
Rudder		±25°
Elevator		-15°, + 10°

TABLE 2.- DATA ACQUISITION EQUIPMENT

6 - Bruel and Kjaer microphones, type 4134
6 - Bruel and Kjaer preamps, type 2619
6 - Bruel and Kjaer type 141-B microphone amplifiers and line drivers
6 - Microphone baseplates and wind screens
1 - Bruel and Kjaer model 4220 pistonphone
1 - Datatron time code generator
1 - Sangamo Sabre III instrumentation tape recorder
1 - Hewlett-Packard model 141-A oscilloscope
1 - Hewlett-Packard frequency counter model 5233L
1 - Hewlett-Packard alternating current vacuum tube voltmeter model 400C
1 - Climet wind direction and velocity measuring instrument, models 011-1 and 014-102
1 - General Eastern remote reading thermometer and humidity indicator, model 400 CD

TABLE 3.- TEST PARAMETERS

RUN	TEST POINT	FLAPS (degrees)	% NH	HOT THRUST kg	(lb)
1	1	5.6	85	590	(1300)
1	2	5.6	90	1315	(2900)
1	3	5.6	91.7	1678	(3700)
1	4	5.6	93.8	2177	(4800)
1	5	5.6	96	2722	(6000)
1	6	5.6	97.2	2948	(6500)
1	7	5.6	99	3220	(7100)
2	1	30	90	1315	(2900)
2	2	30	93.8	2177	(4800)
2	3	30	96	2722	(6000)
2	4	30	97.2	2948	(6500)
3	1	53	90	1315	(2900)
3	2	53	93.8	2177	(4800)
3	3	53	96	2722	(6000)
3	4	53	97.2	2948	(6500)

TABLE 4.- METEOROLOGICAL DATA

RUN	TEST POINT	WIND SPEED km/hr	WIND SPEED kn	WIND DIRECTION DEGREES*	TEMP °C	TEMP °F	PERCENT RELATIVE HUMIDITY
1	1	7.8-10.4	4.2-5.6	346	18.6	61.8	52.0
1	2	8.9	4.8	353	18.6	61.8	52.0
1	3	8.4	4.5	332	18.8	62.0	52.0
1	4	6.7-9.9	3.6-5.4	345	18.8	62.0	52.0
1	5	7.2	3.9	5	18.8	62.0	51.4
1	6	10.6	5.7	9	18.4	61.4	51.3
1	7	10.7	5.8	0	18.3	61.2	51.2
2	1	14.1	7.6	332	19.4	63.0	49.8
2	2	13.2	7.2	352	19.4	63.0	49.4
2	3	13.4	7.3	347	18.8	62.0	49.1
2	4	10.3	5.6	345	18.8	62.0	49.0
3	1	11.7	6.3	362	18.9	62.3	46.8
3	2	10.9	5.9	350	18.9	62.3	46.8
3	3	10.9	5.9	351	19.0	62.5	46.1
3	4	9.4	5.1	363	19.1	62.6	46.0

* Aircraft heading was 335°

TABLE 5.- DATA ANALYSIS EQUIPMENT

1	-	Sangamo Sabre III instrumentation tape recorder
1	-	General Radio mod 1921, 1/2 octave analyzer
1	-	Hewlett-Packard mod 141A oscilloscope
1	-	Hewlett-Packard mod 5233L frequency counter
1	-	Hewlett-Packard mod 400 AC voltmeter
1	-	Hewlett-Packard 7046A X-Y plotter

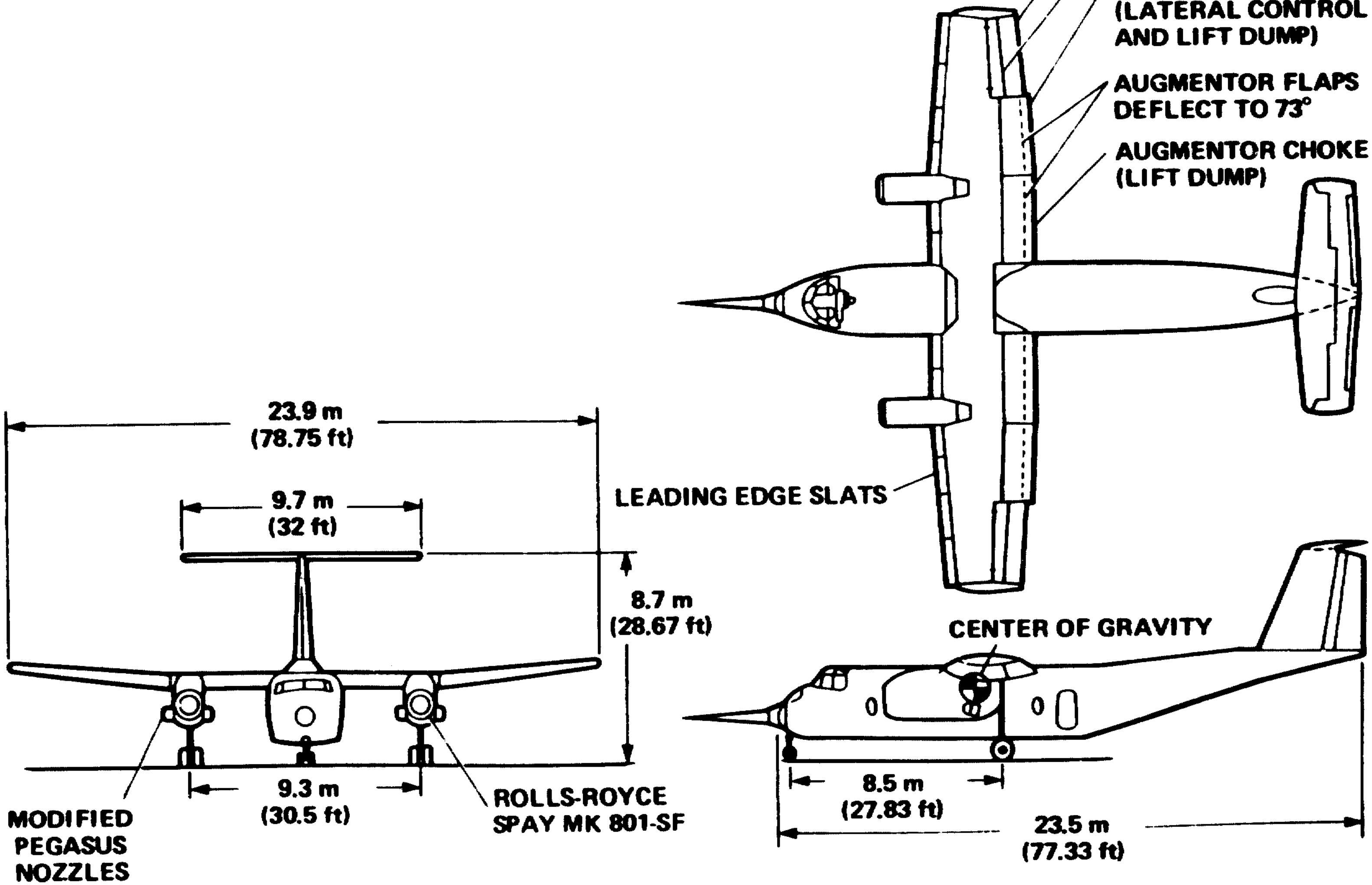
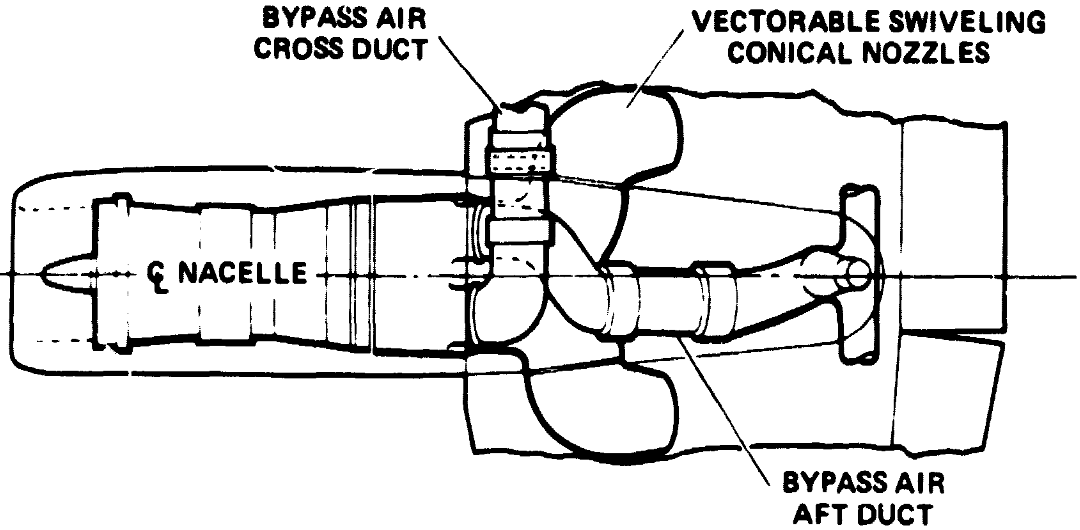


Figure 1.- Three-view drawing of the AWJSRA.



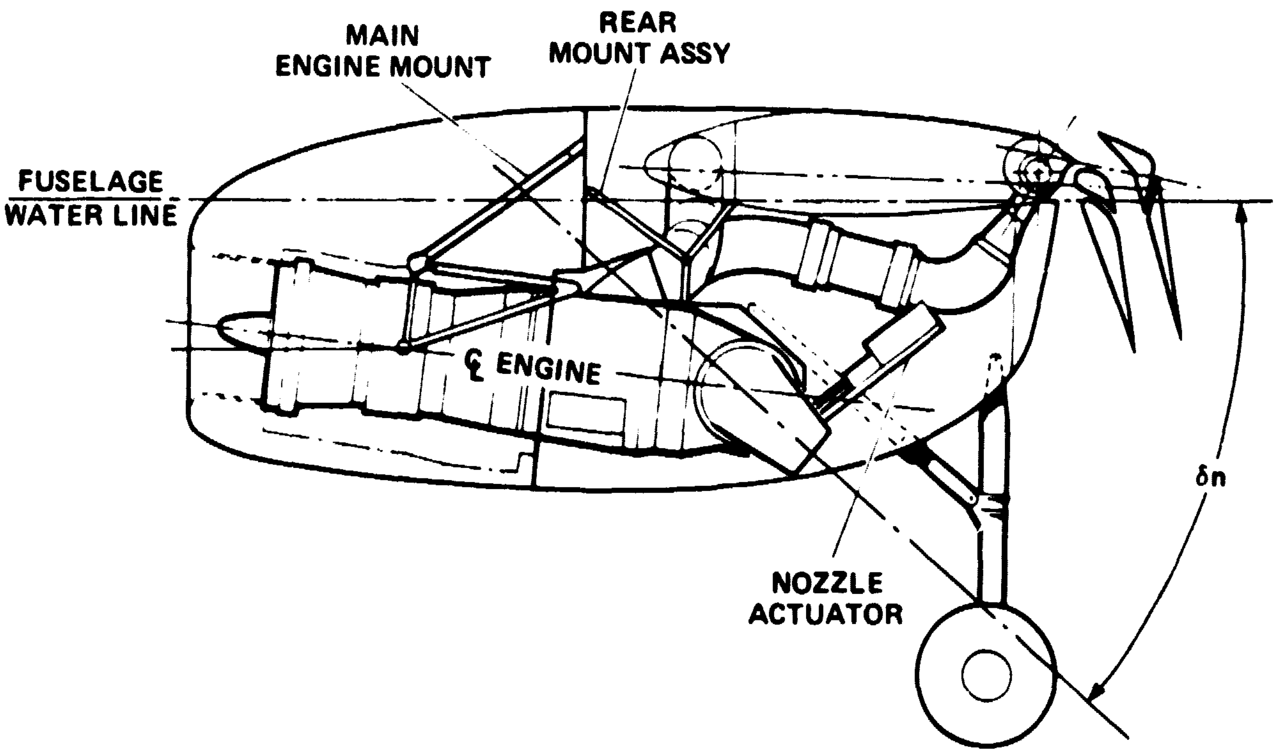


Figure 2.- AWJSRA propulsion system installation.

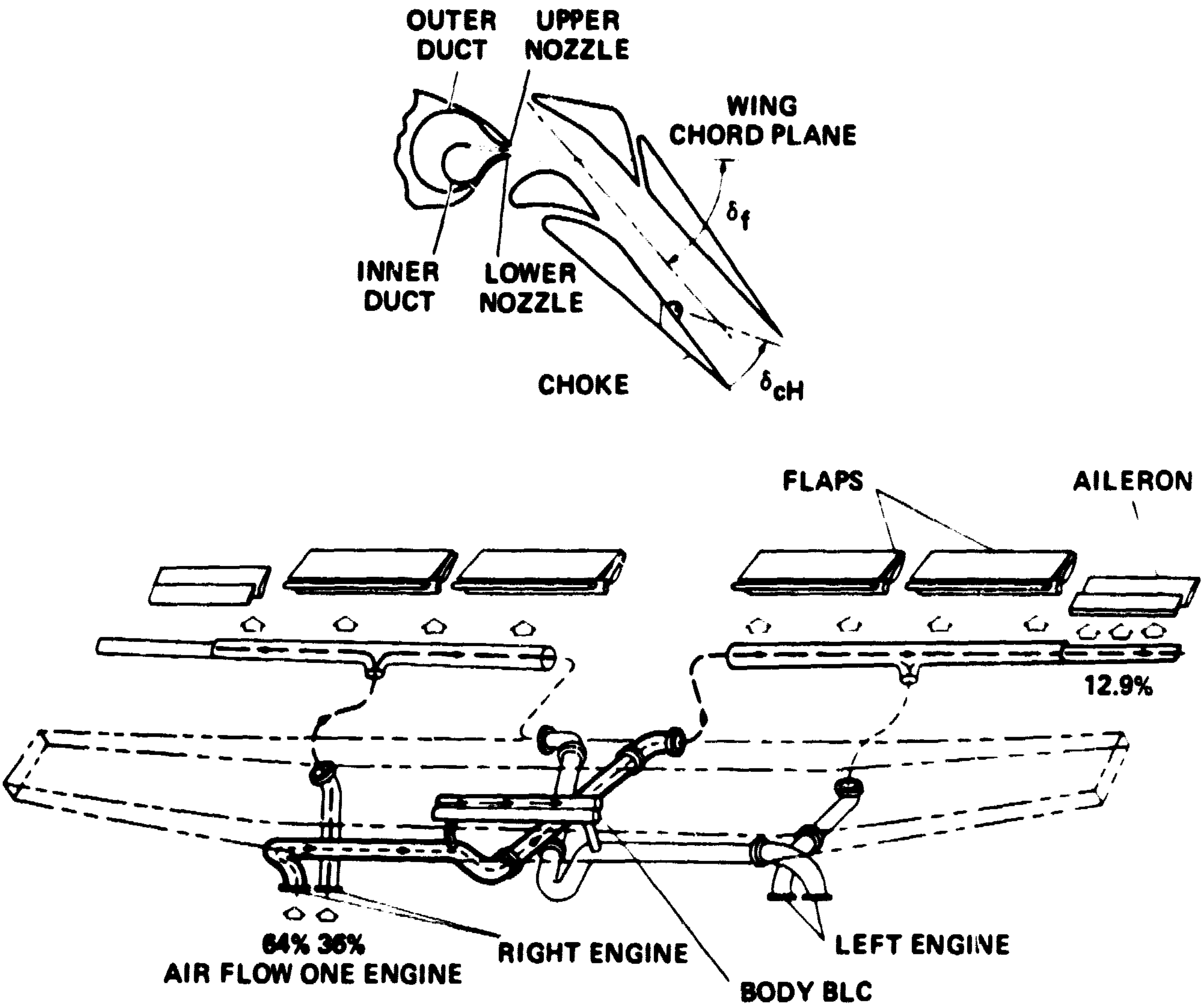


Figure 3.- AWJSRA cold flow air distribution system.

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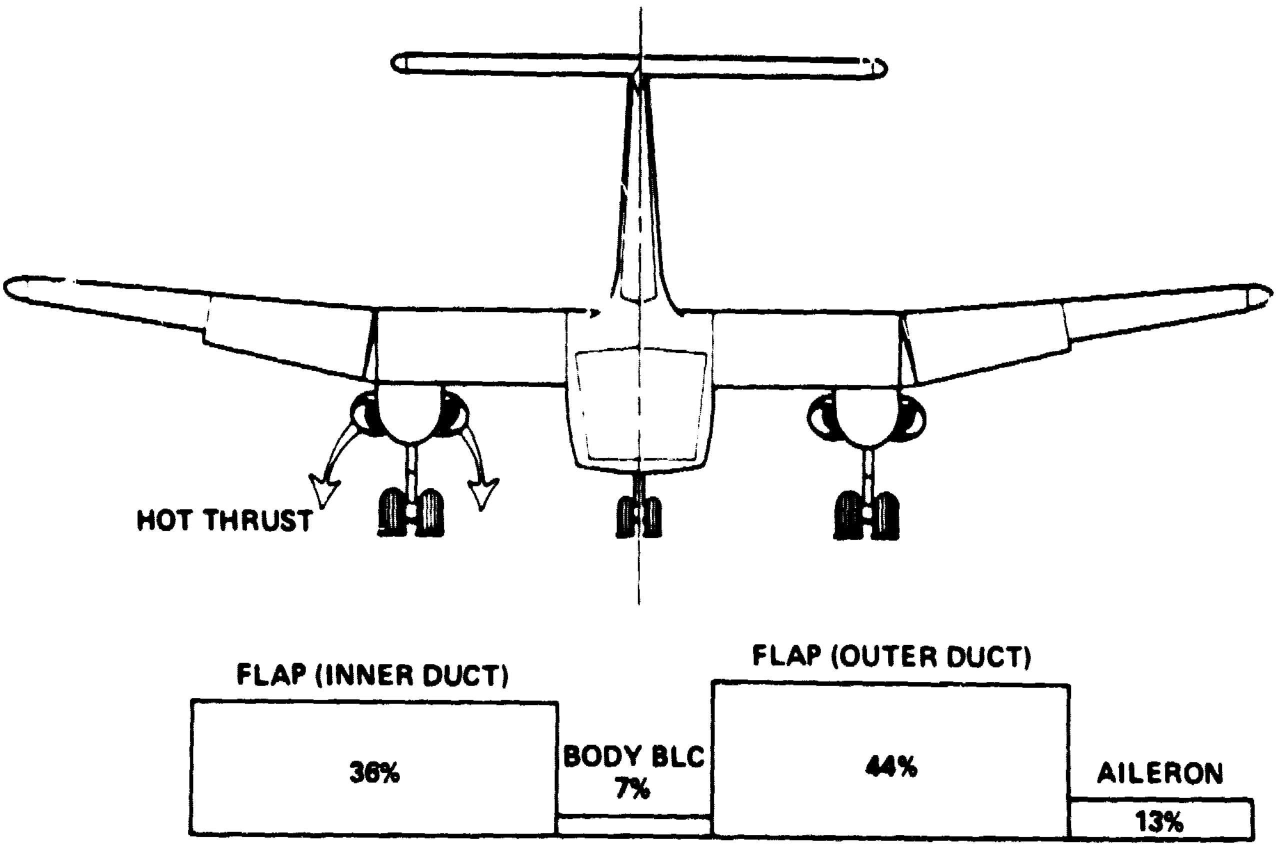

Figure 4.- Left-hand engine cold flow air distribution.

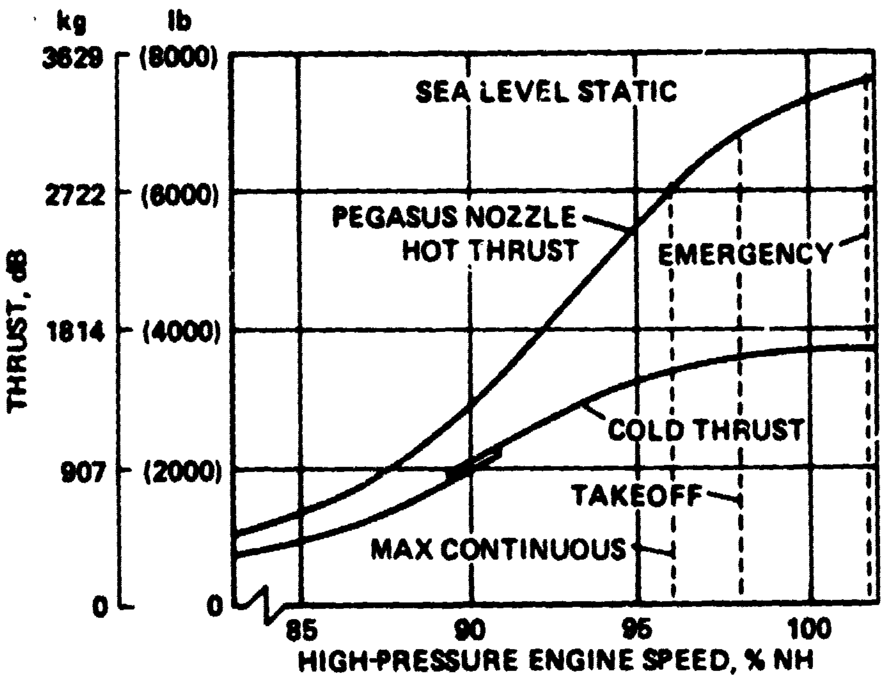

Figure 5.- Rolls Royce Spey 301-SF engine thrust with modified jetpipe.

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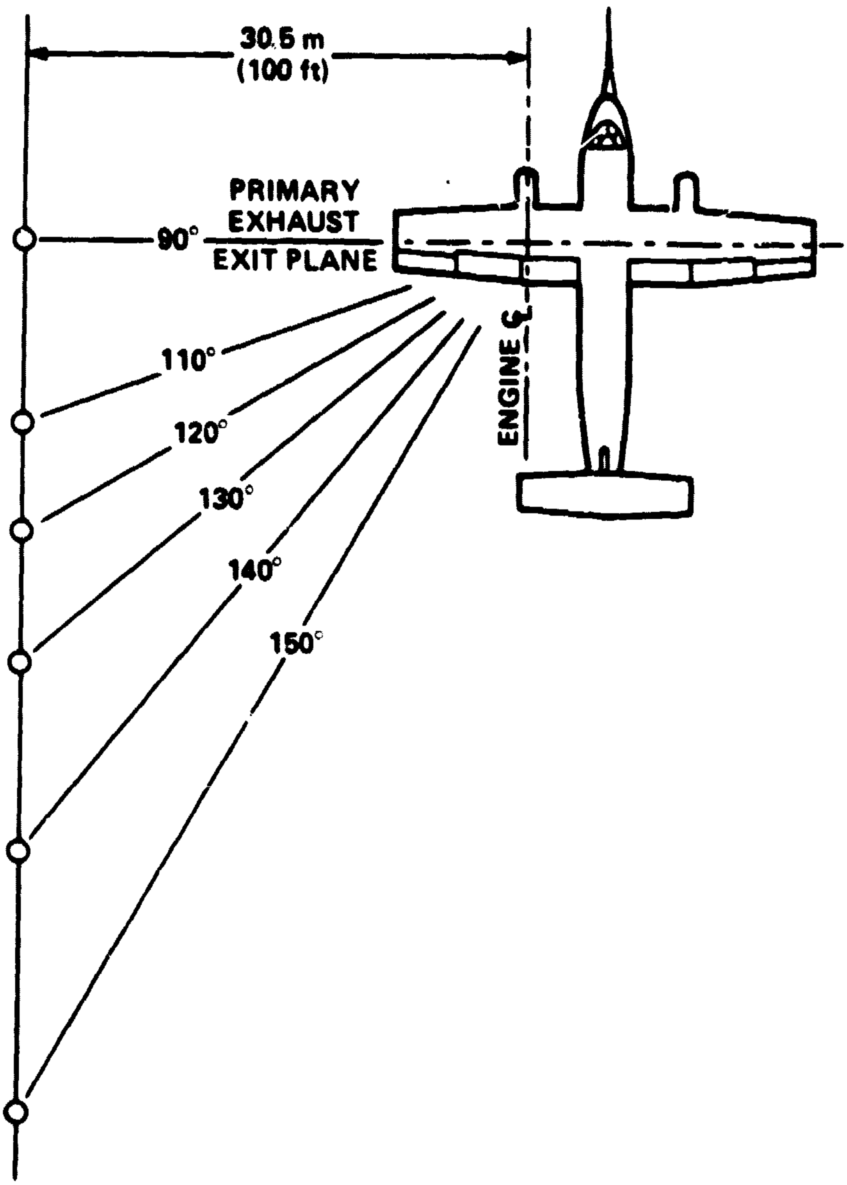

Figure 6.- Microphone array.

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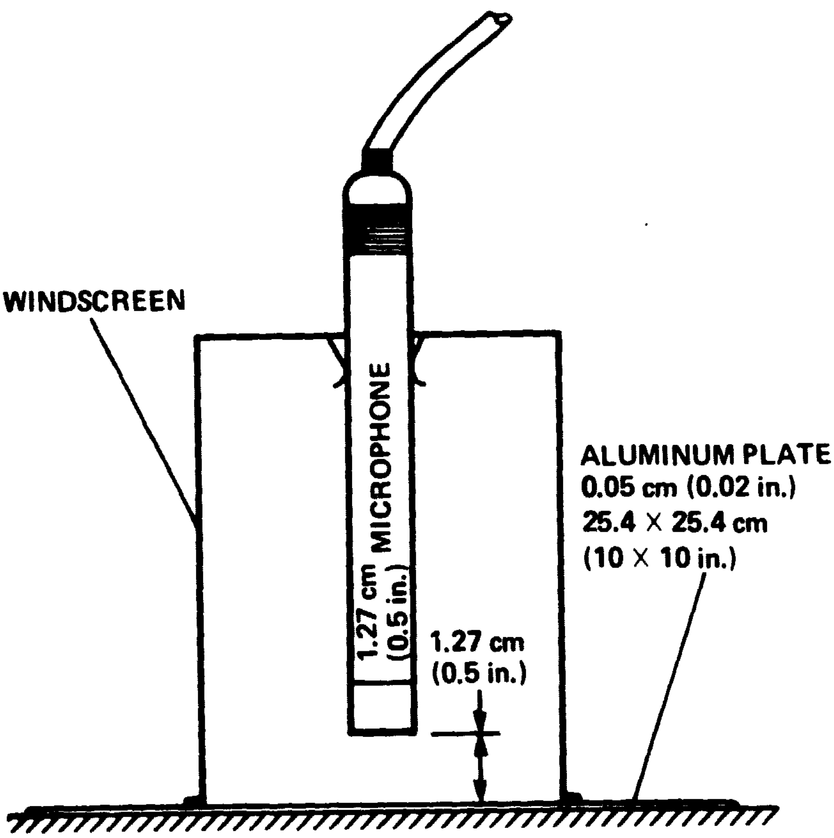


Figure 7.- Typical microphone installation.

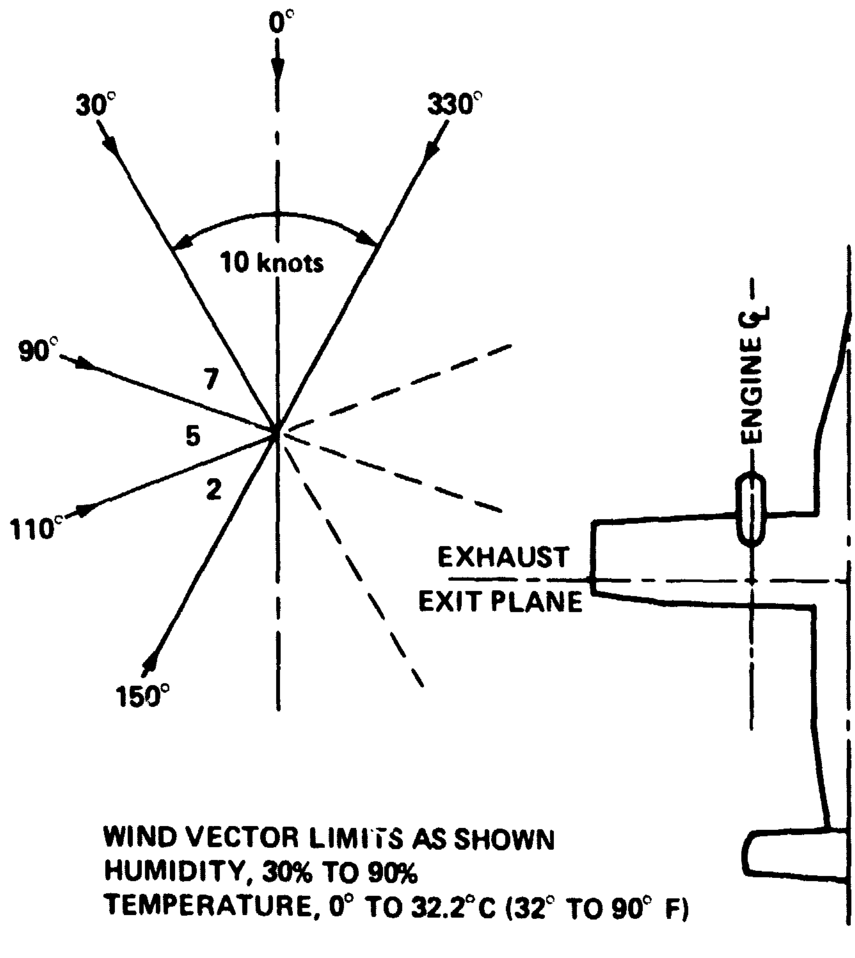


Figure 8.- Test limits, weather.

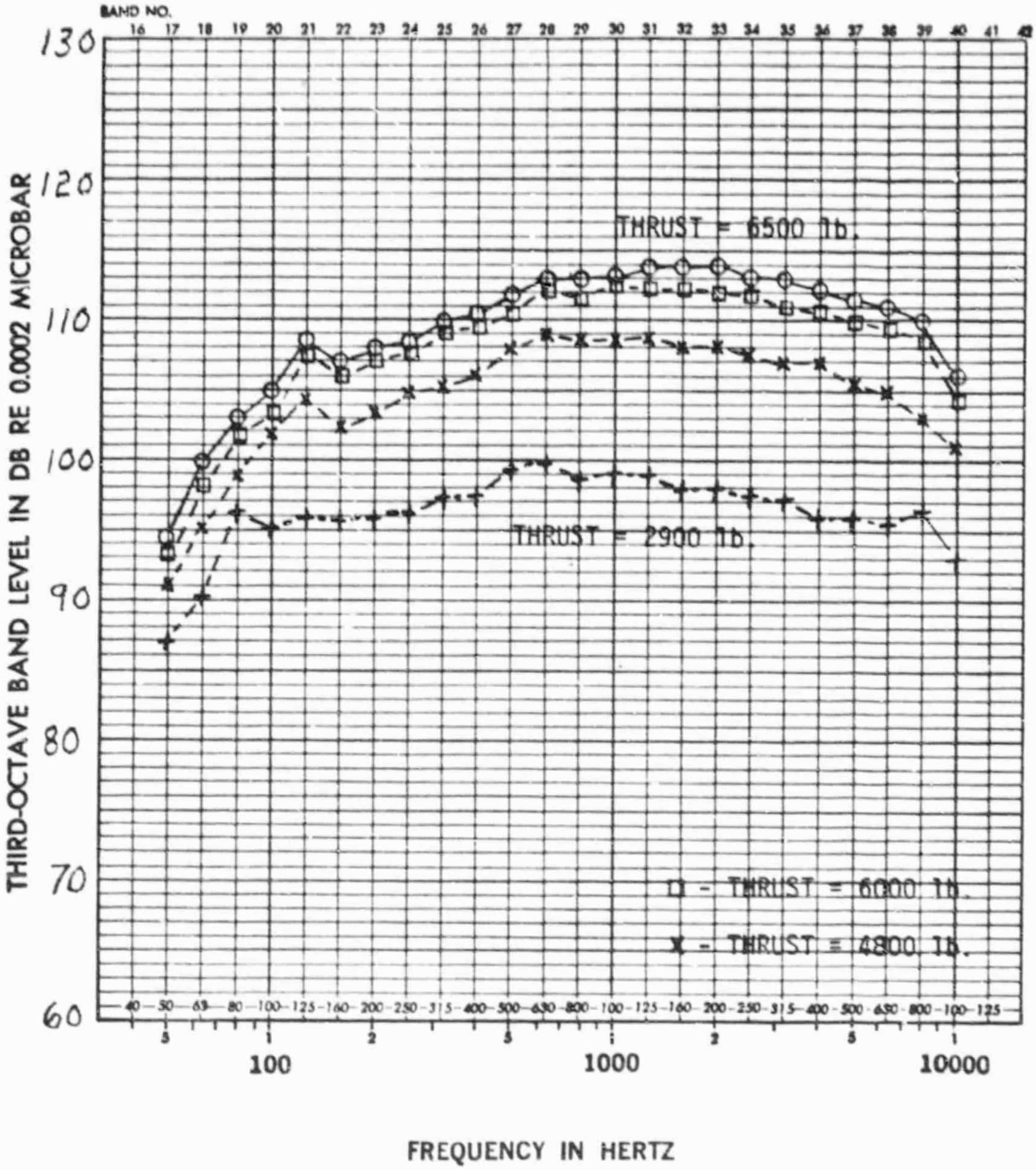


Figure 9. One third octave band sound pressure level at 90° microphone position. Flaps 5.6°.

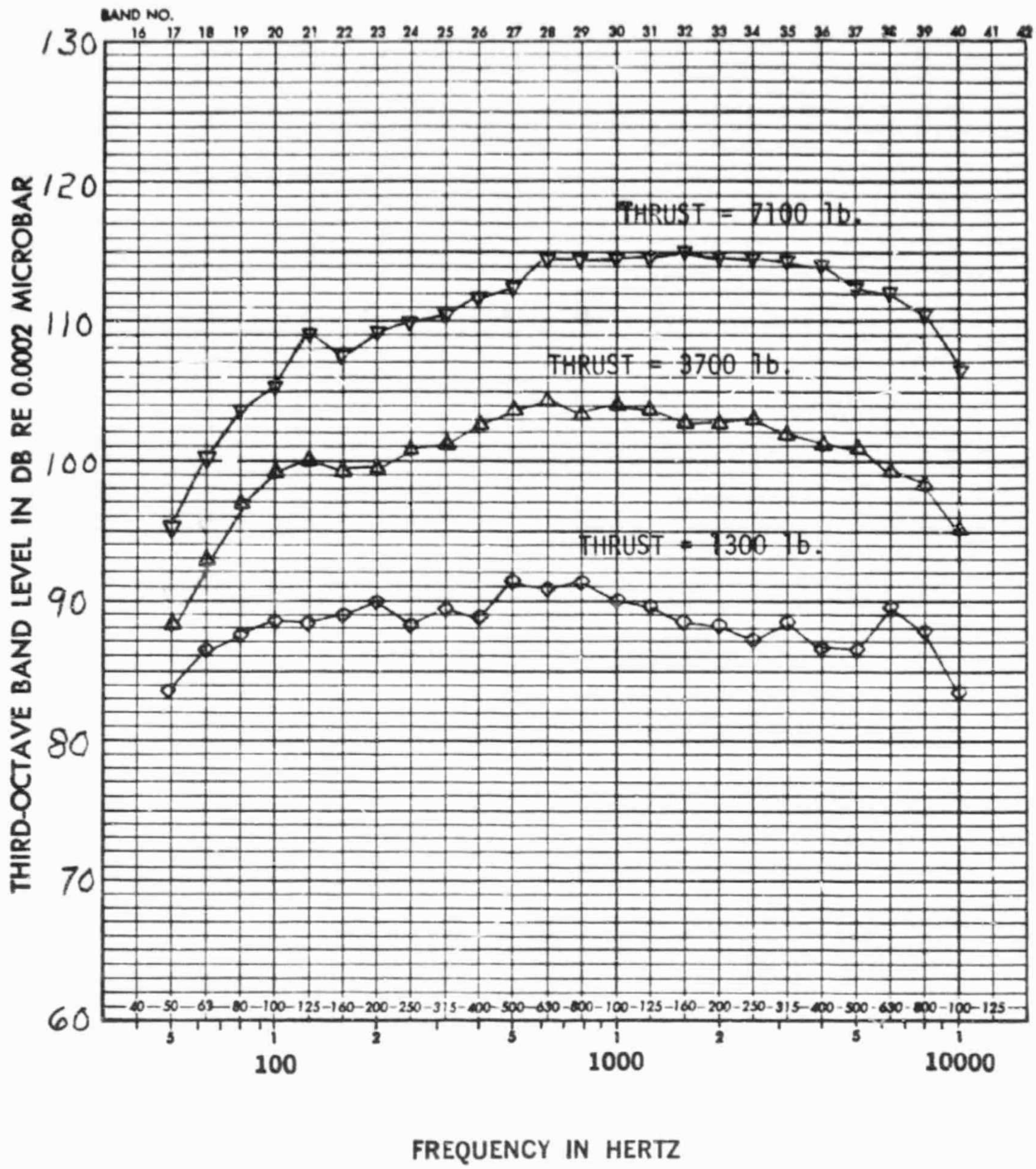


Figure 9. Concluded.

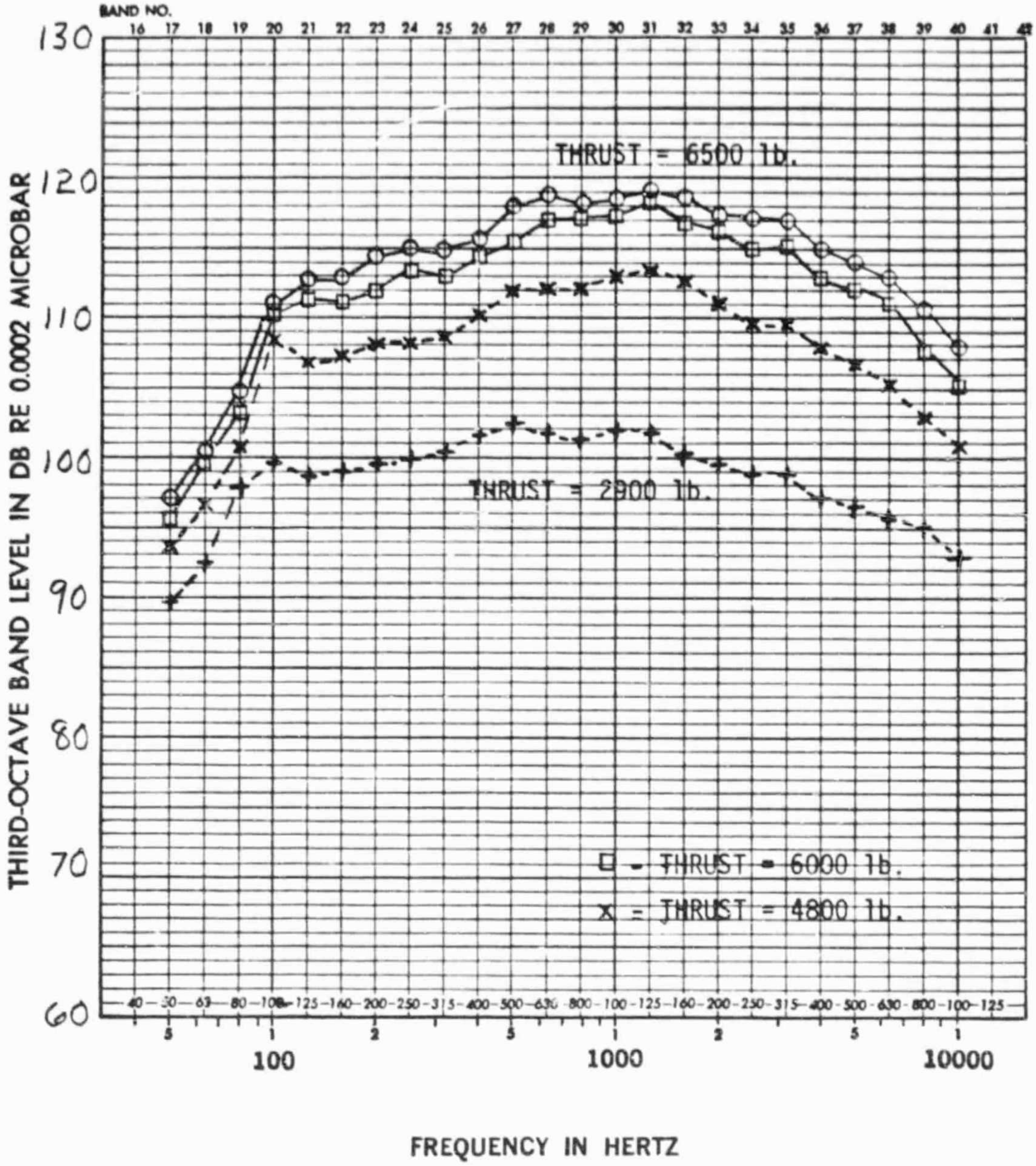
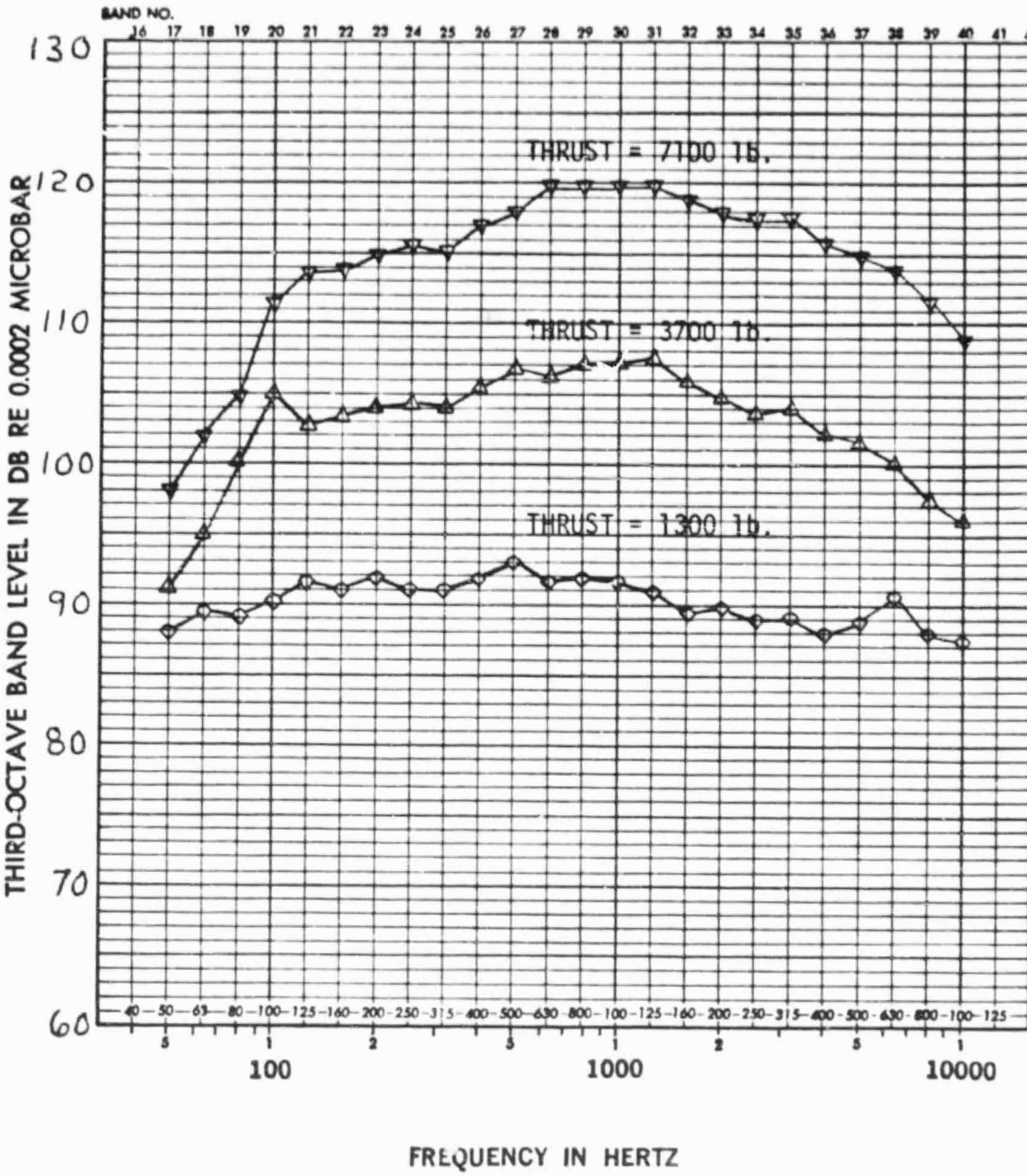


Figure 10. One third octave band sound pressure level at 110° microphone position. Flaps 5.6°.



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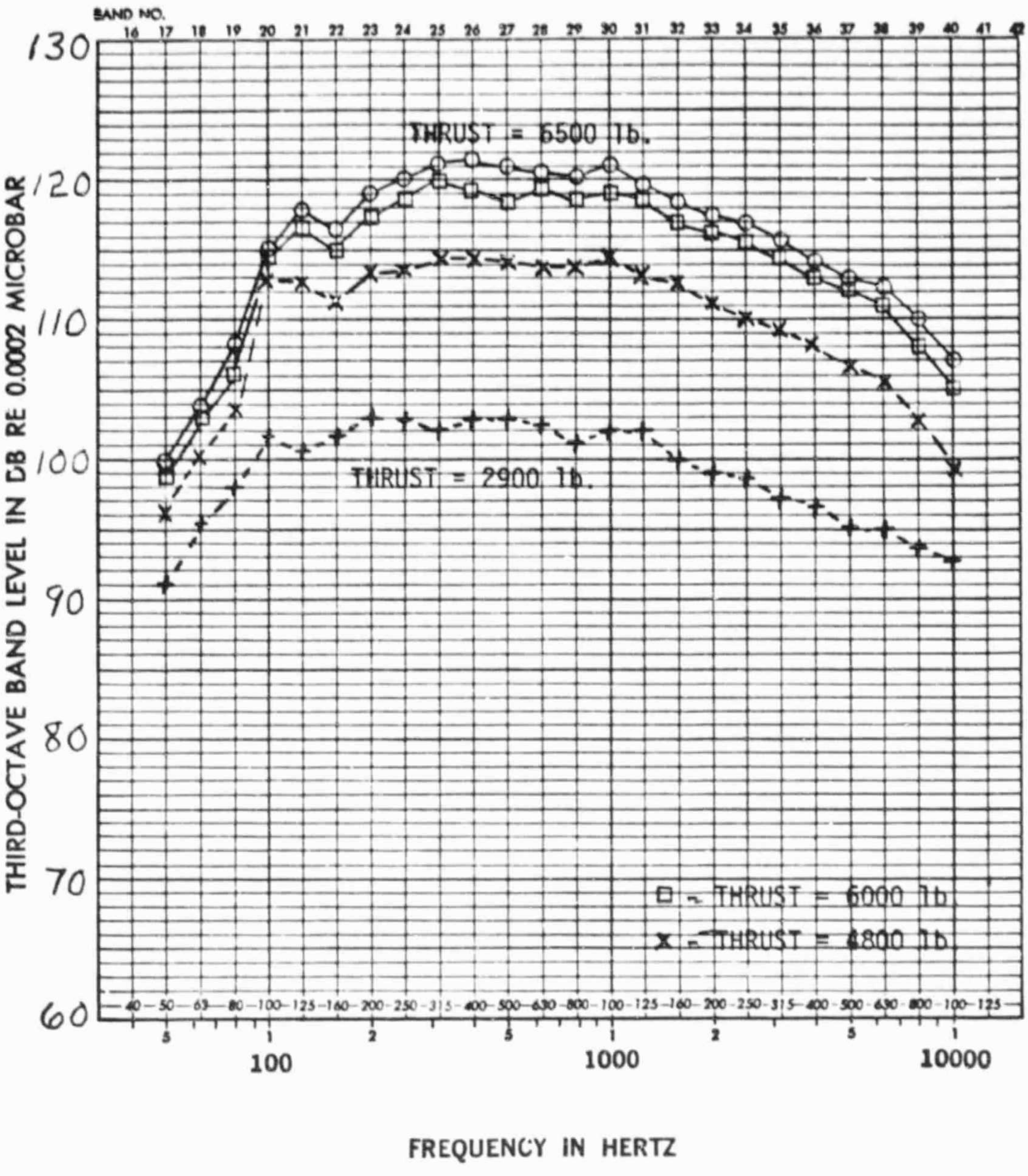


Figure 11. One third octave band sound pressure level at 120° microphone position. Flaps 5.6°.

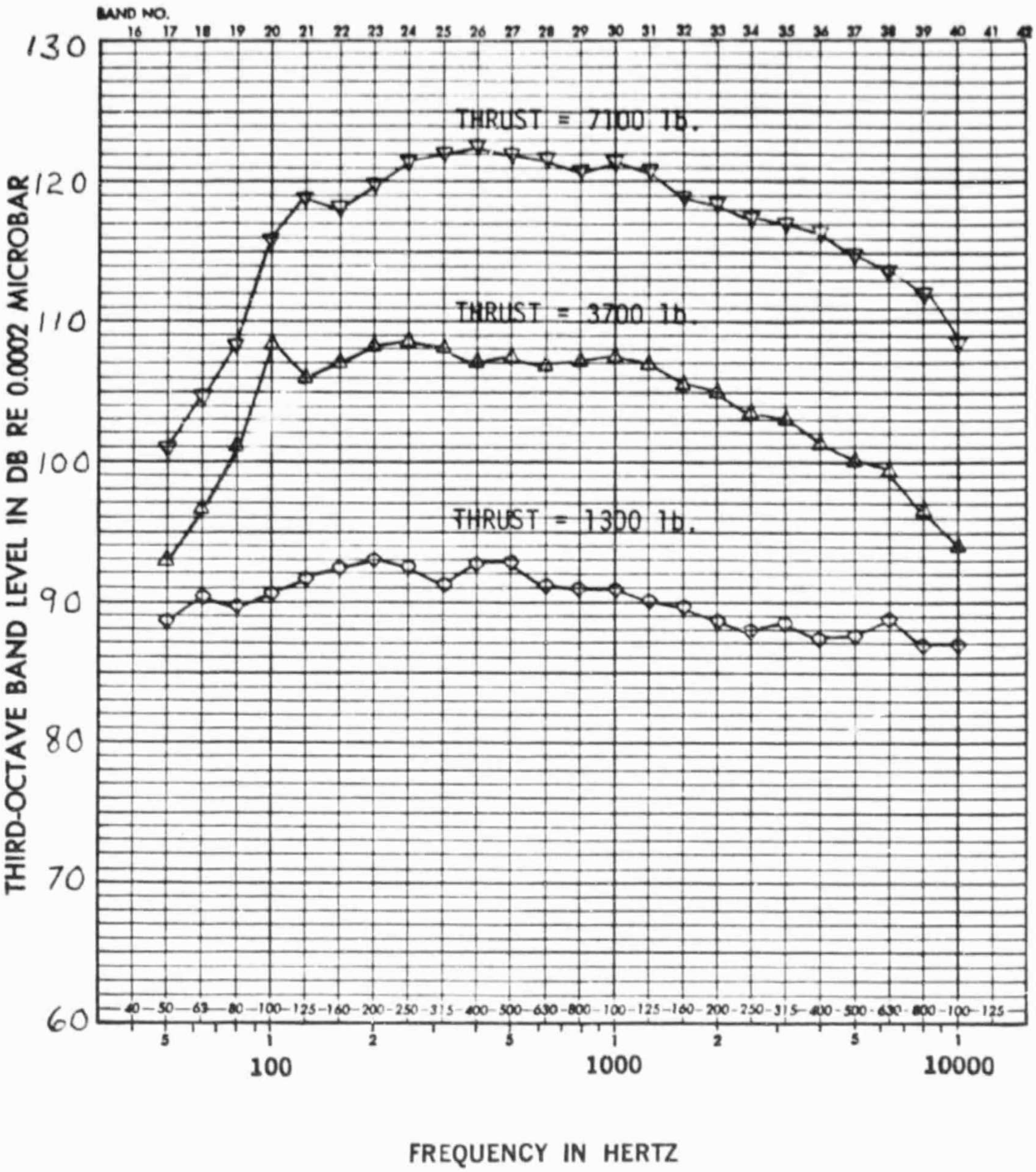


Figure 11. Concluded

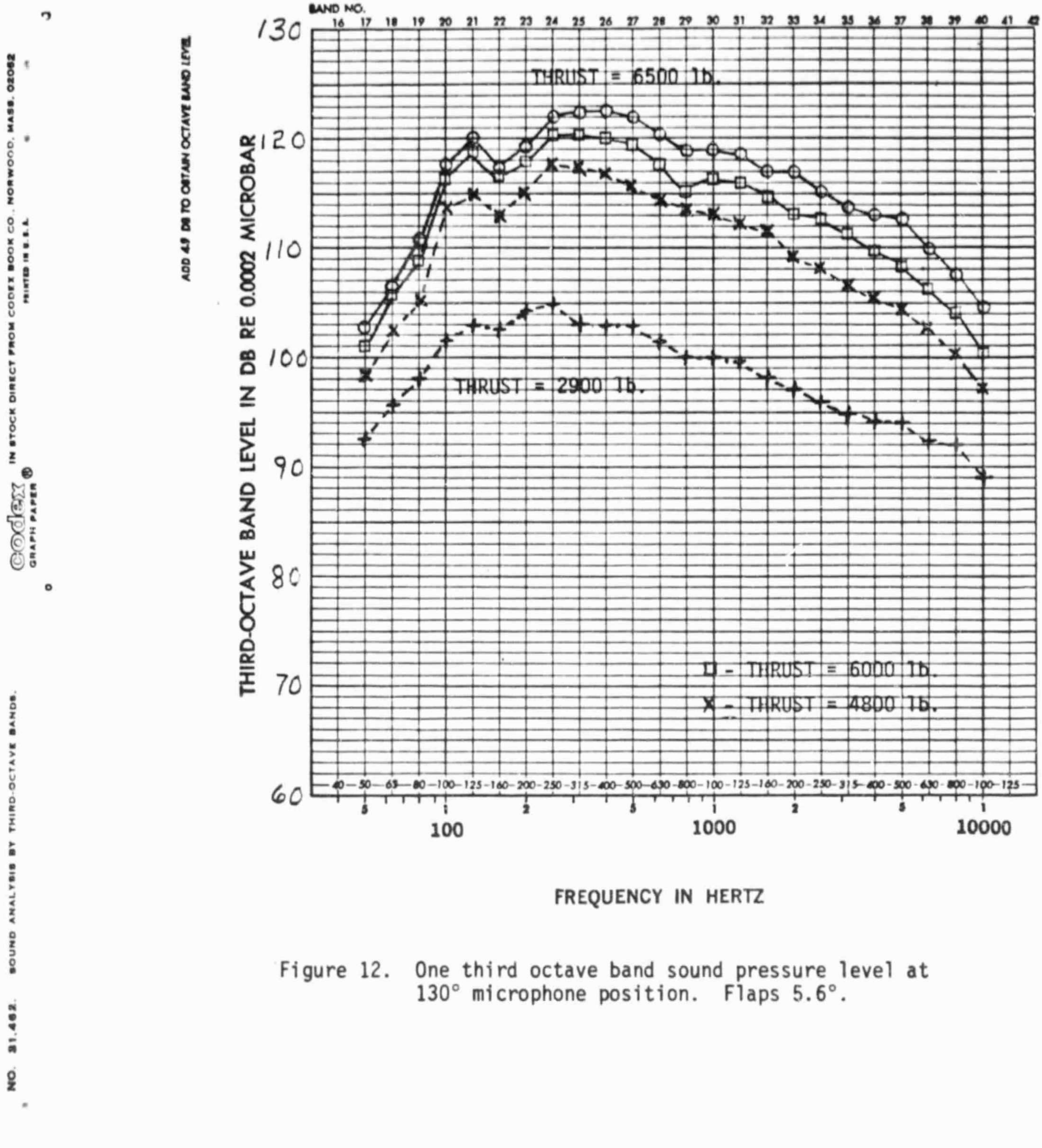


Figure 12. One third octave band sound pressure level at 130° microphone position. Flaps 5.6°.

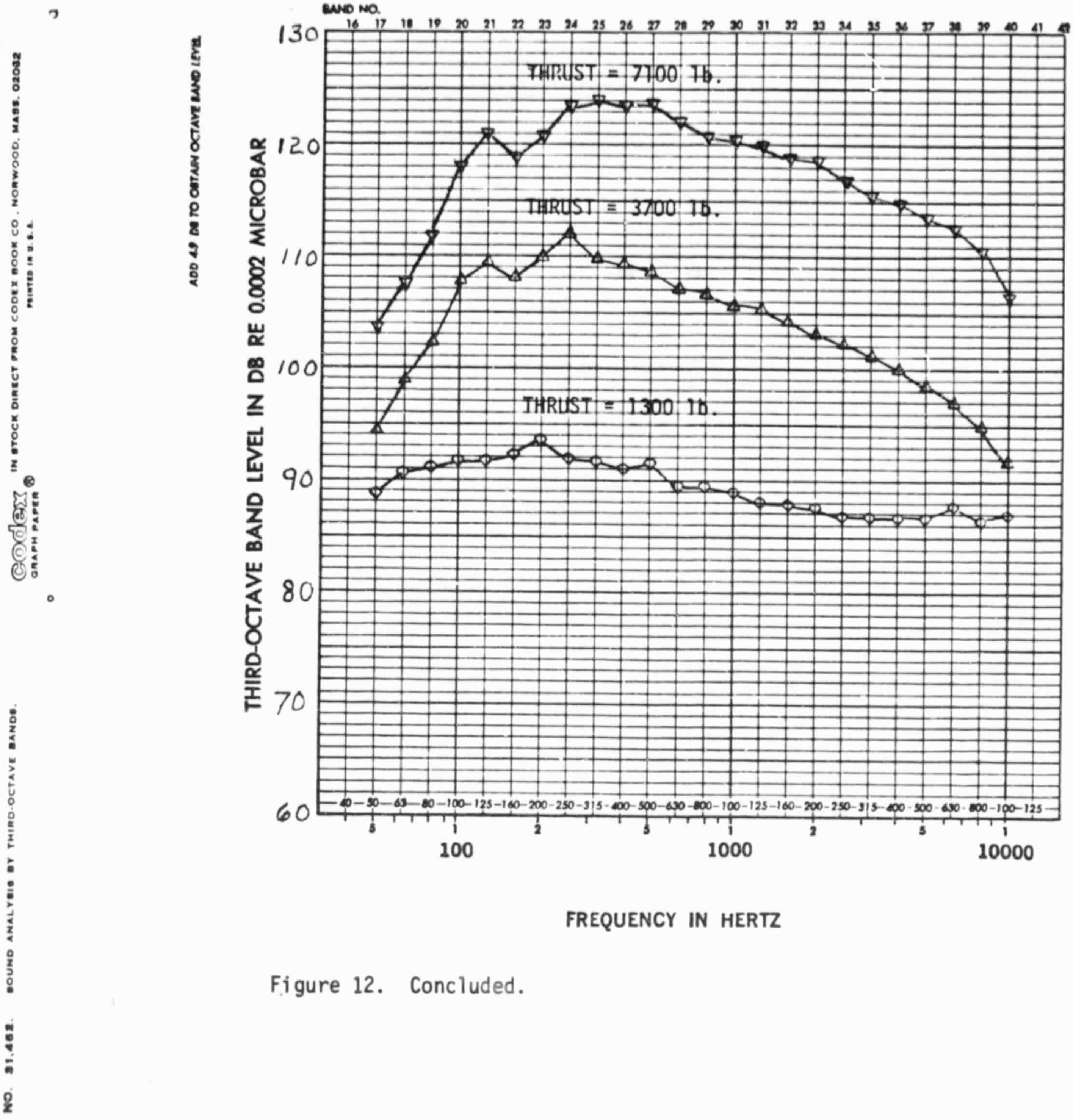


Figure 12. Concluded.

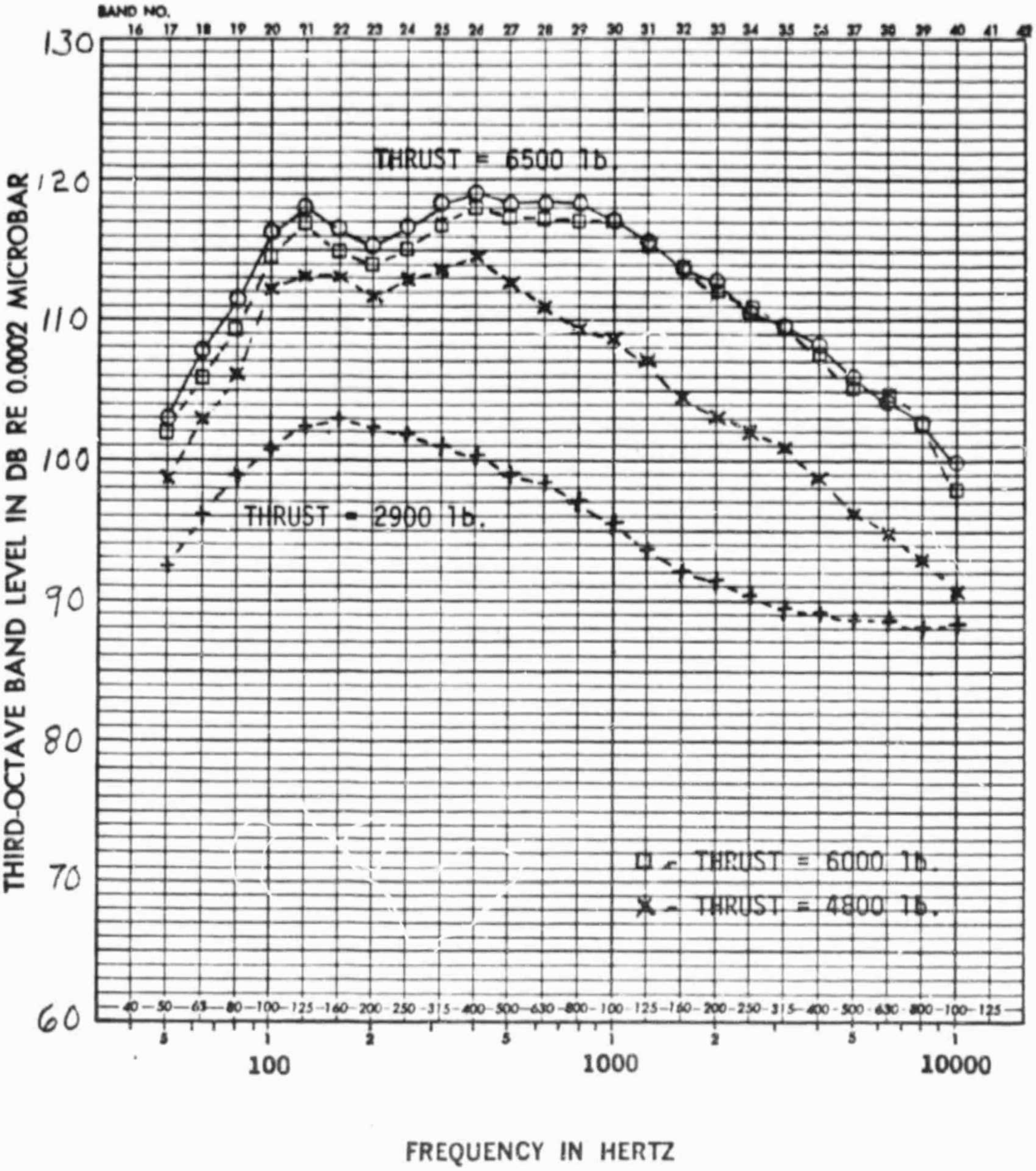
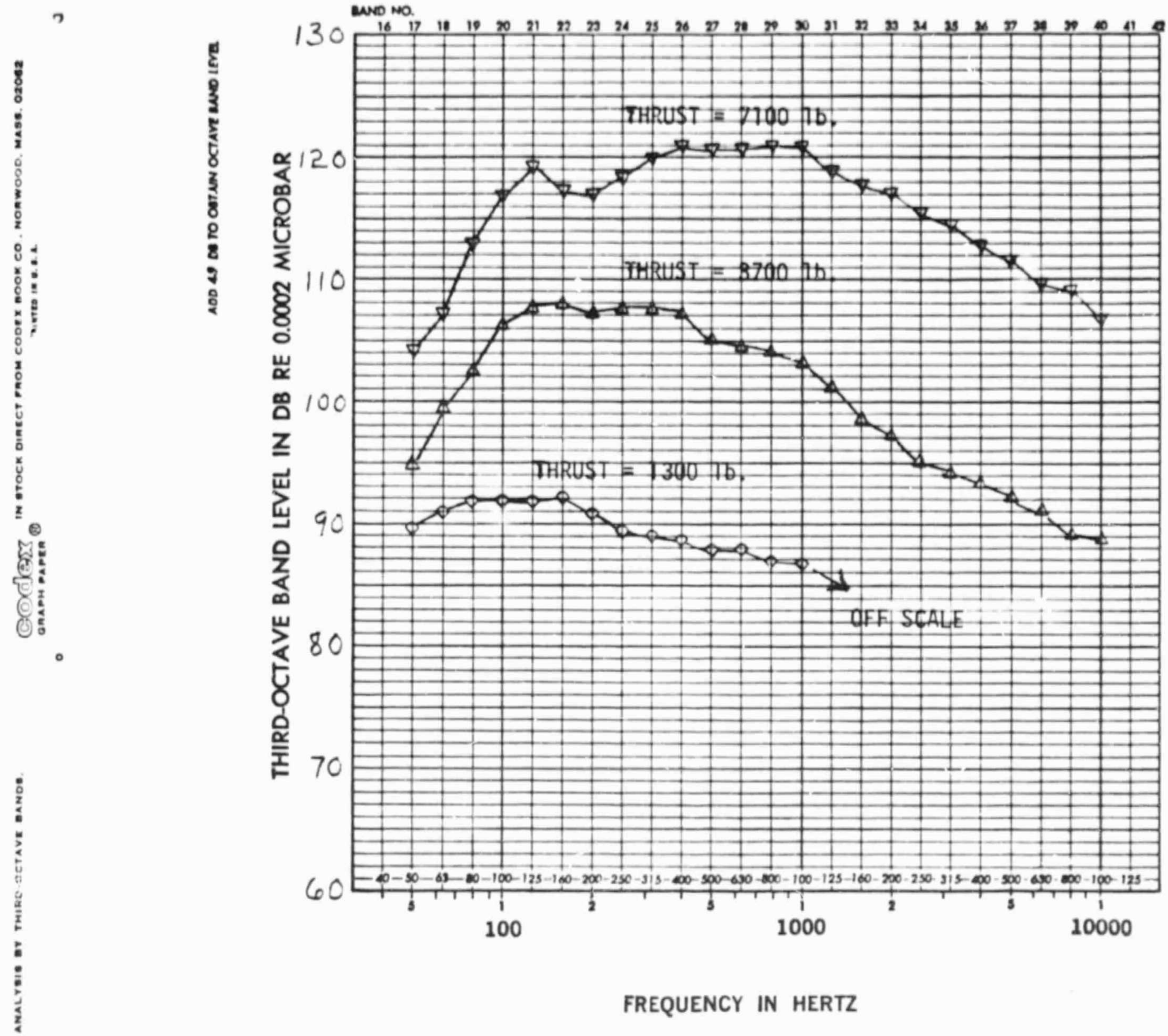


Figure 13. One third octave band sound pressure level at 140° microphone position. Flaps 5.6°.



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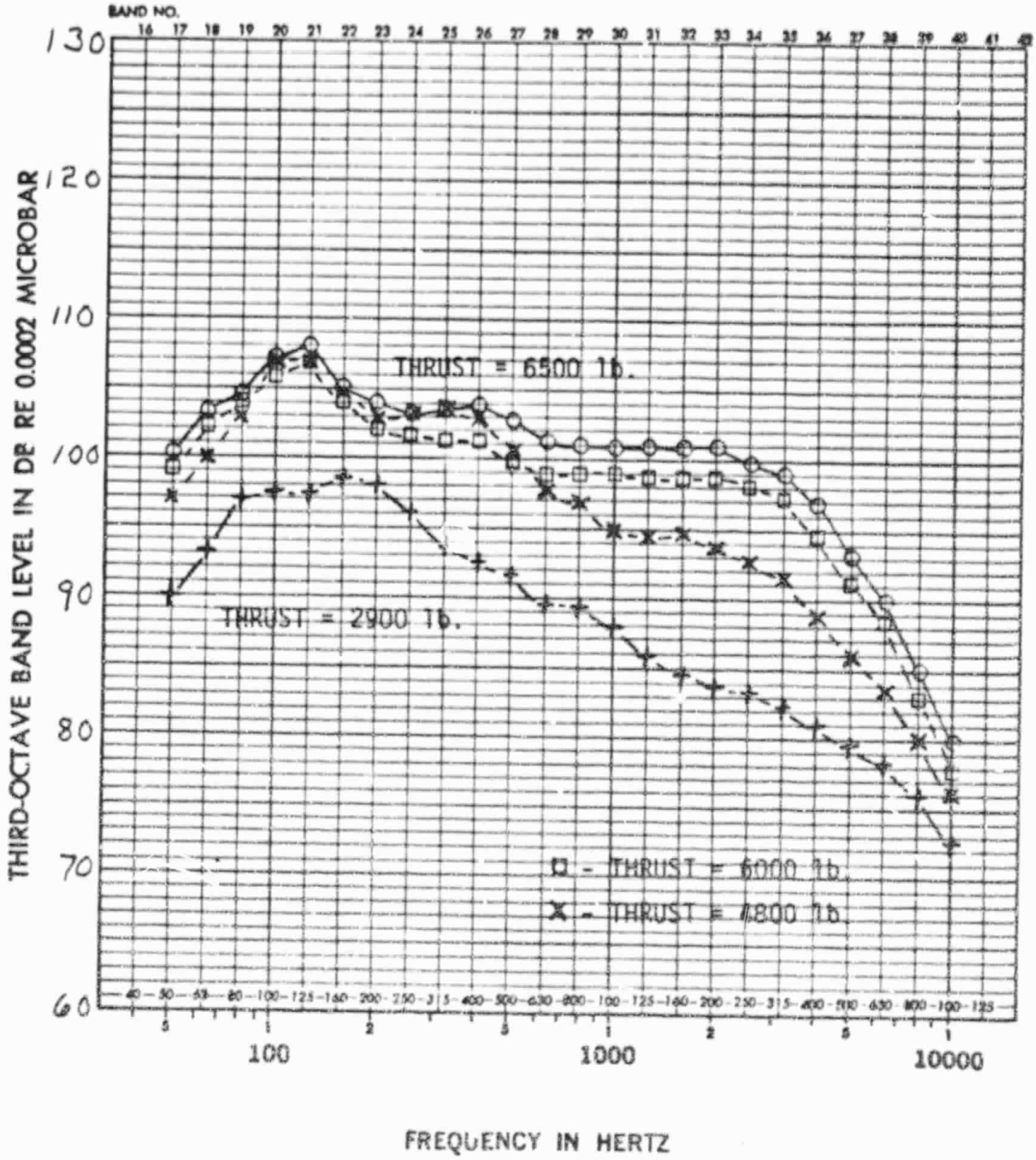
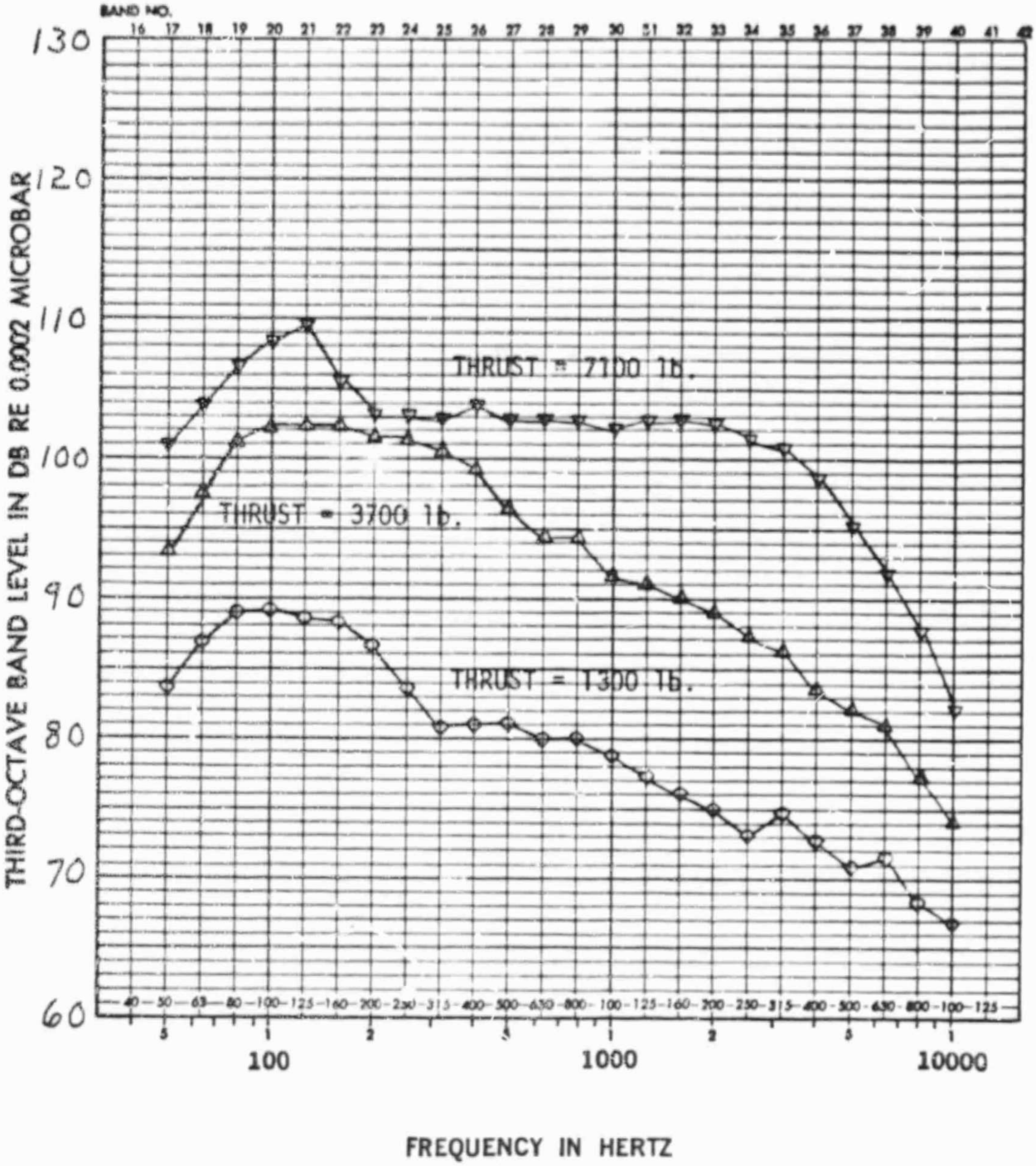


Figure 14. One third octave band sound pressure level at 150° microphone position. Flaps 5.6°.



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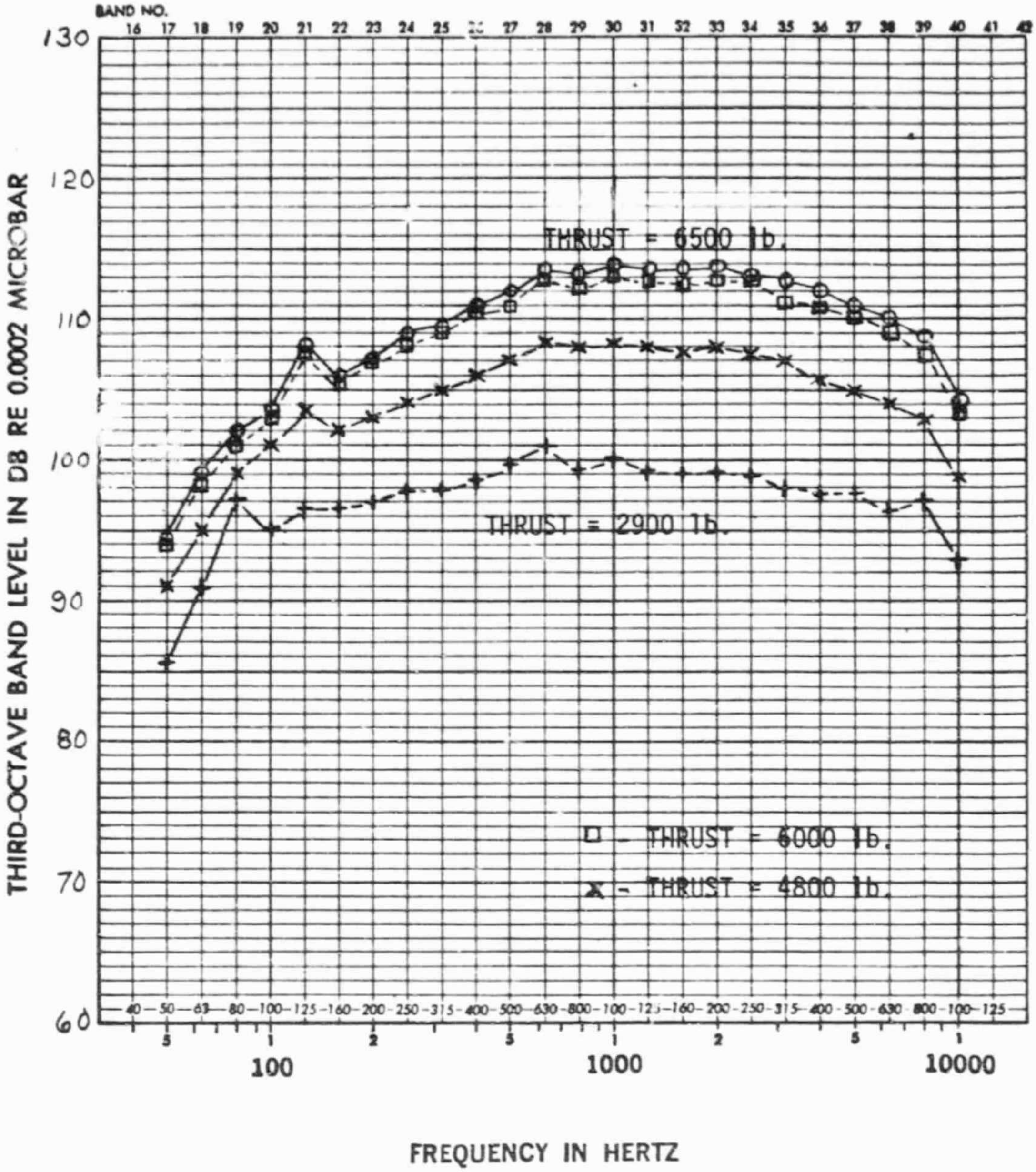

Figure 15. One third octave band sound pressure level at 90° microphone position. Flaps 30°.

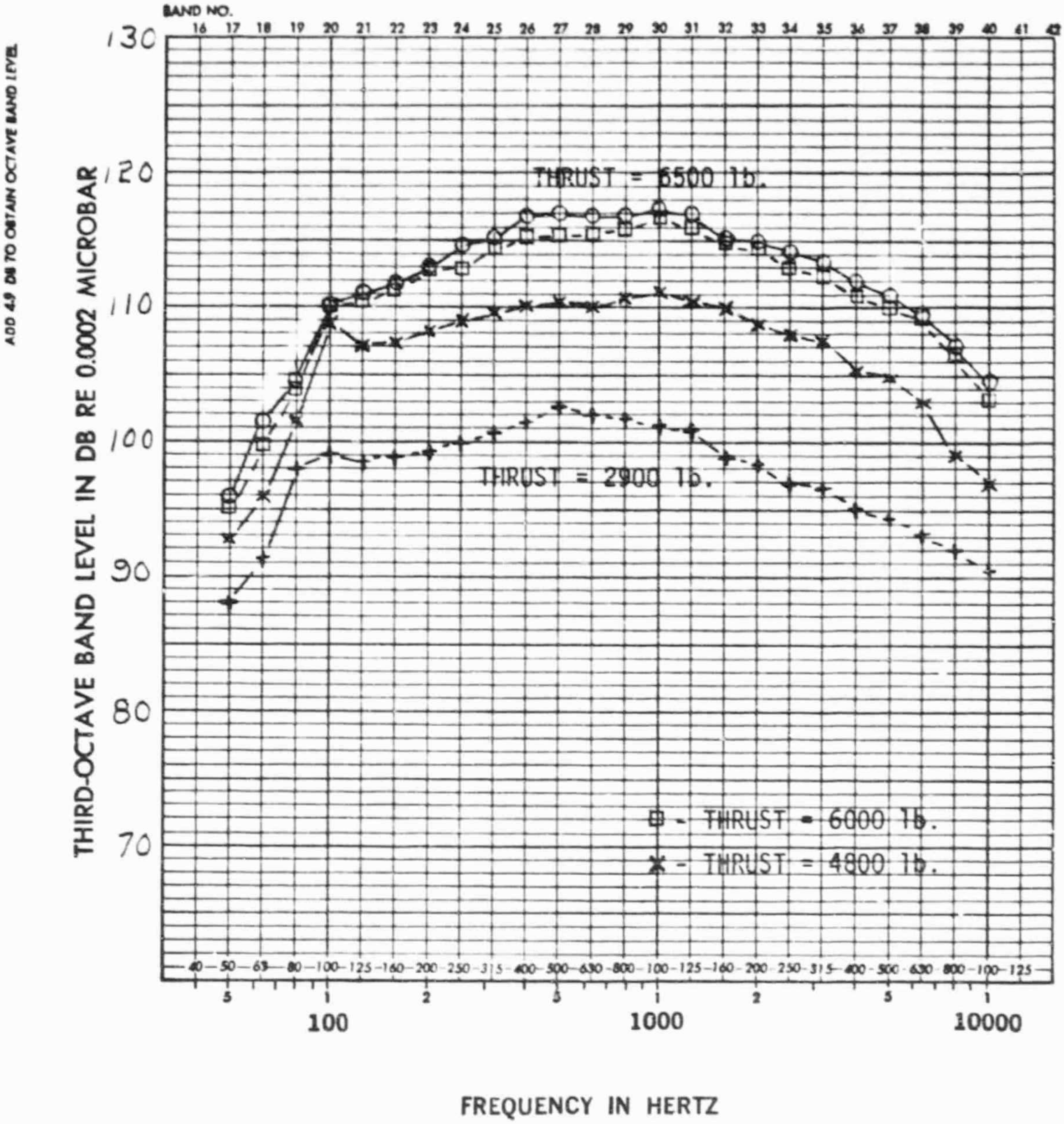

Figure 16. One third octave band sound pressure level at 110° microphone position. Flaps 30°.

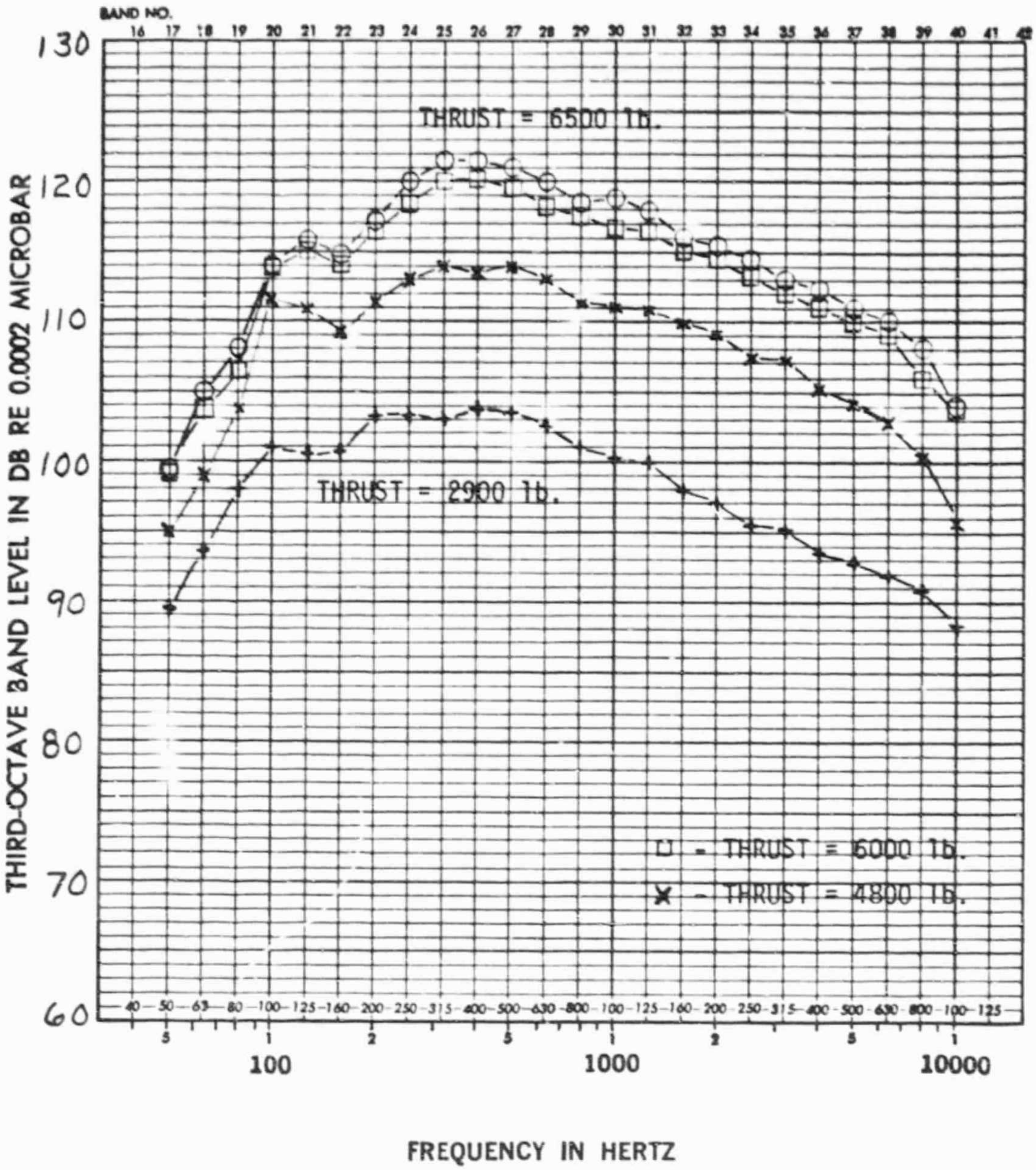

Figure 17. One third octave band sound pressure level at 120° microphone position. Flaps 30°.

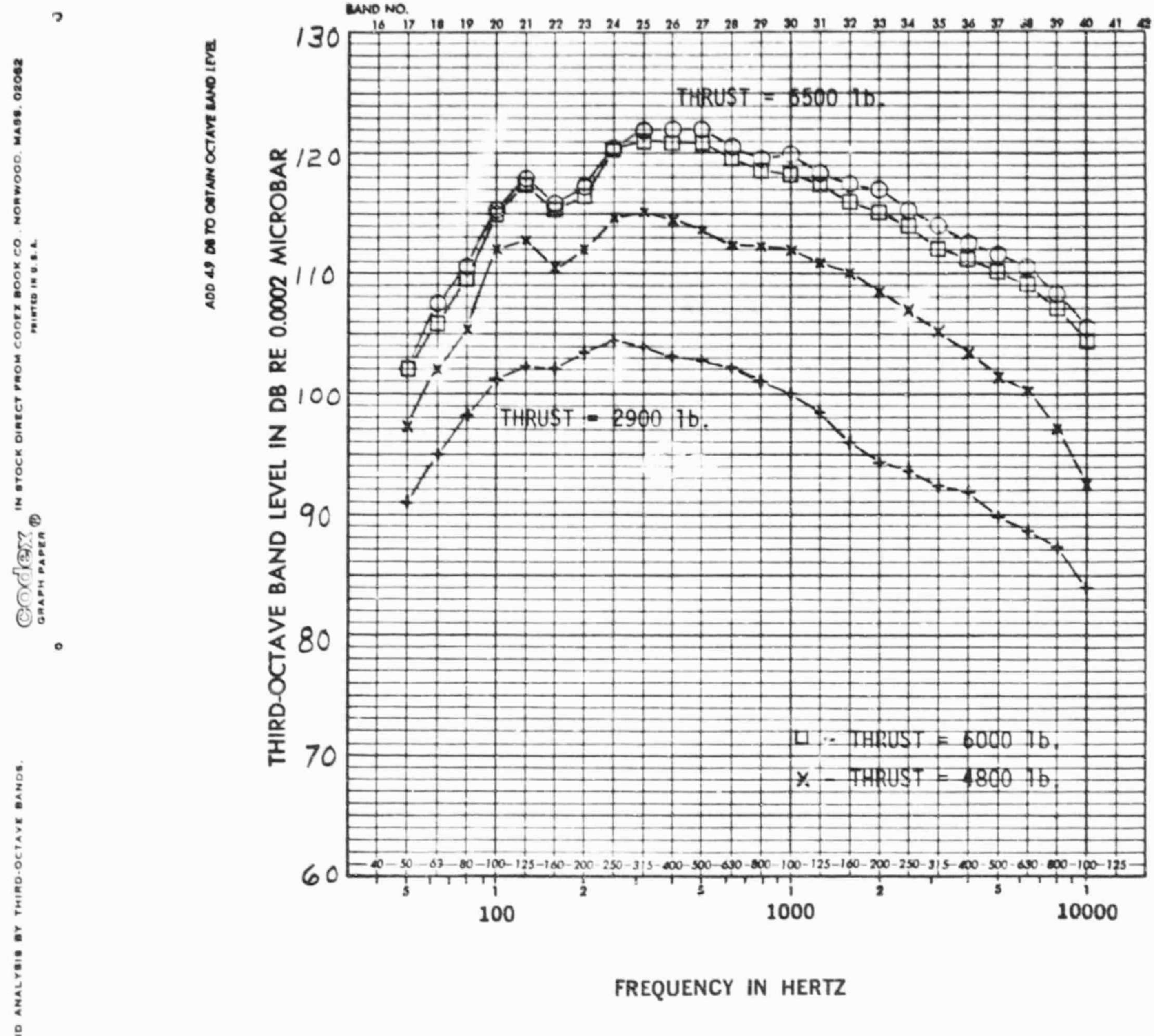

Figure 18. One third octave band sound pressure level at 130° microphone position. Flaps 30°.

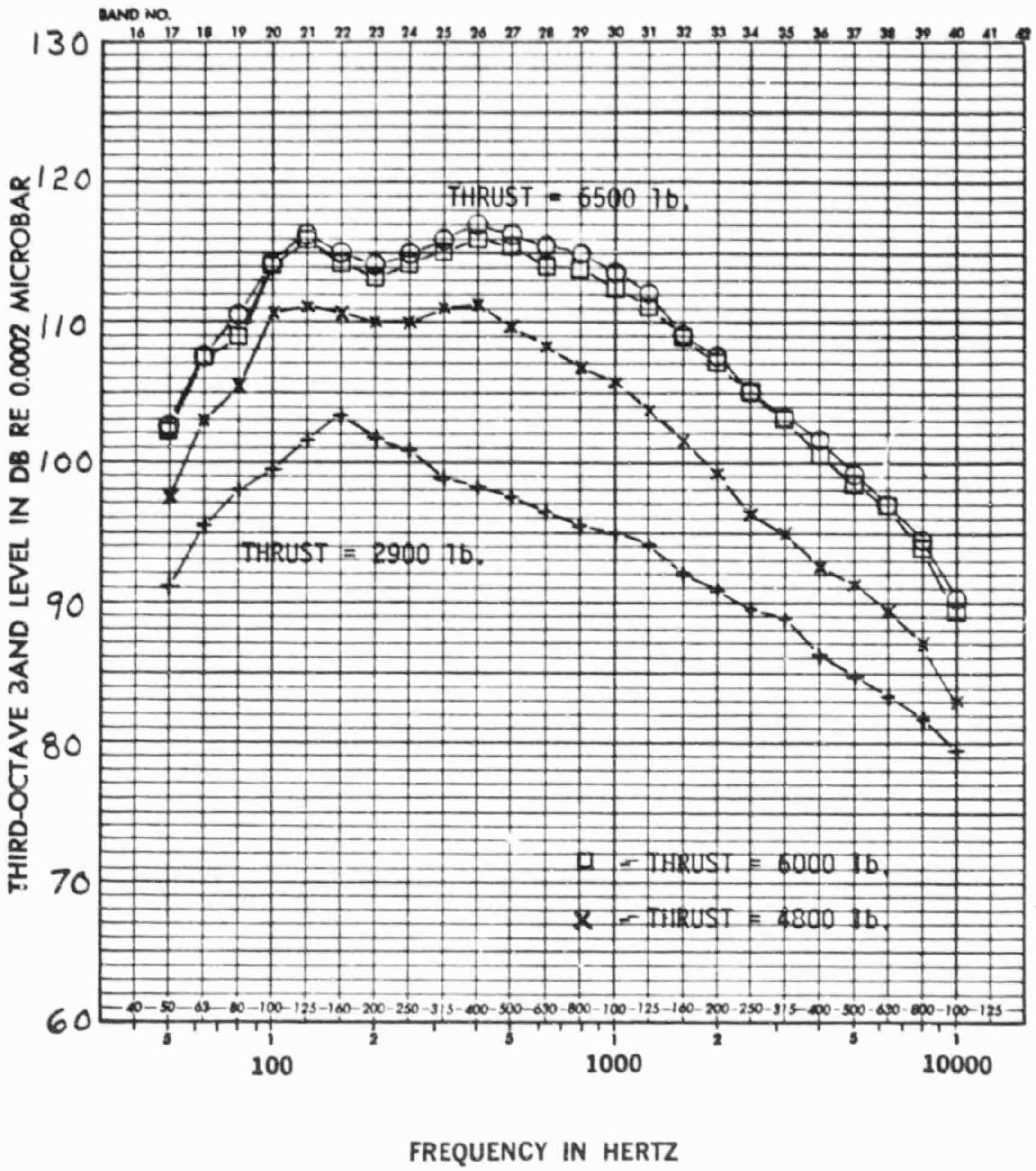

Figure 19. One third octave band sound pressure level at 140° microphone position. Flaps 30°.

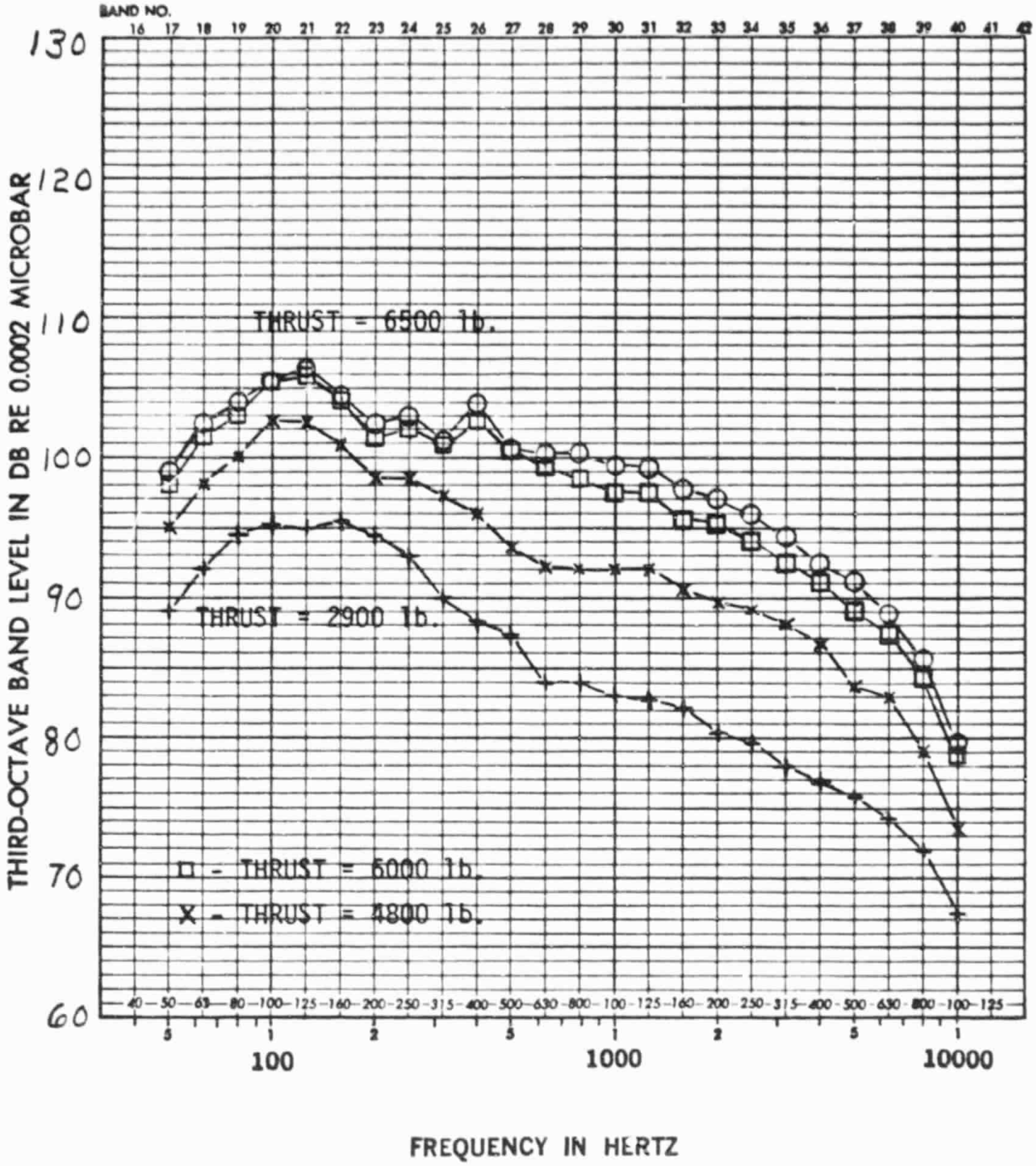

Figure 20. One third octave band sound pressure level at 150° microphone position. Flaps 30°.

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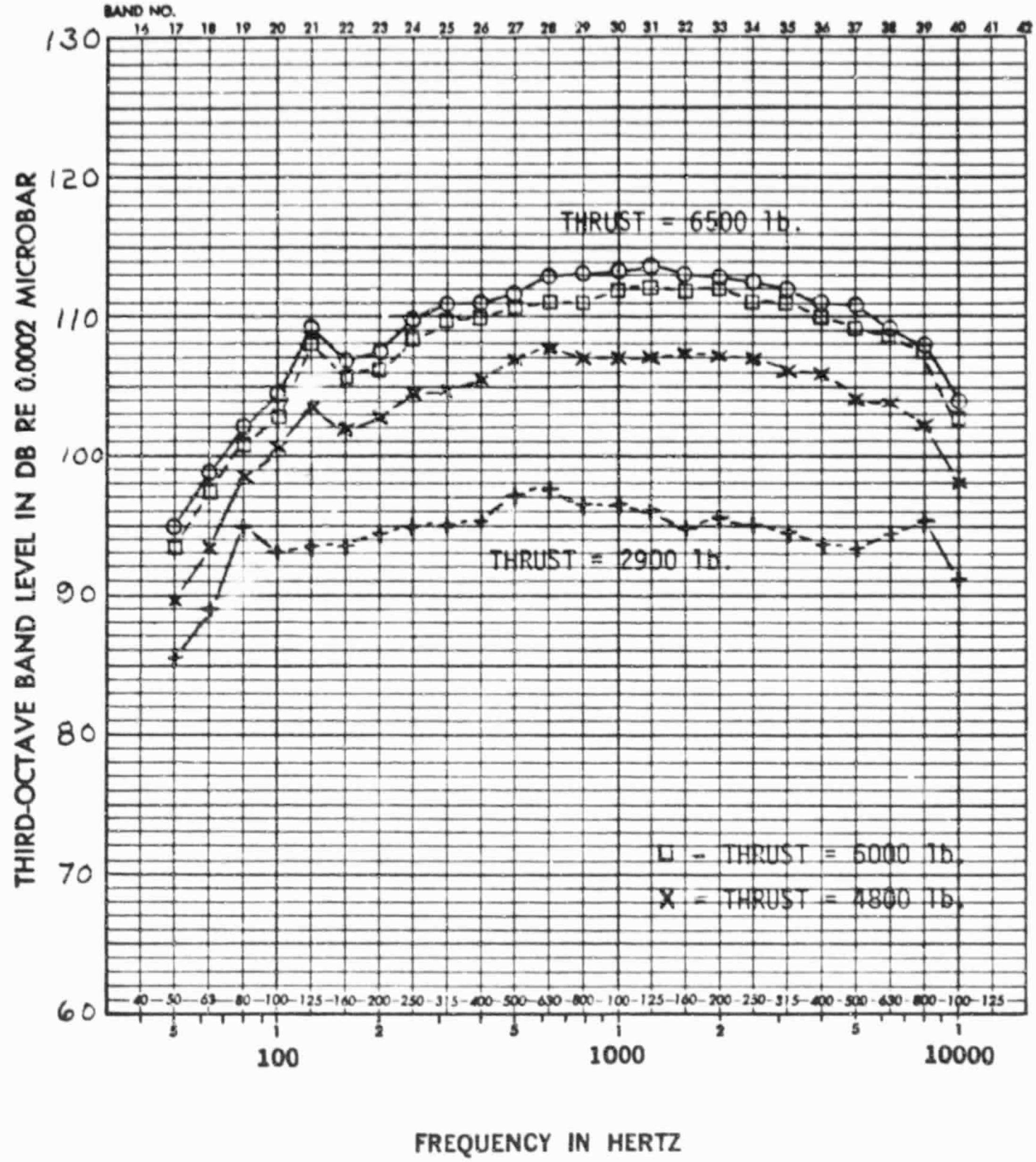

Figure 21. One third octave band sound pressure level at 90° microphone position. Flaps 53°.

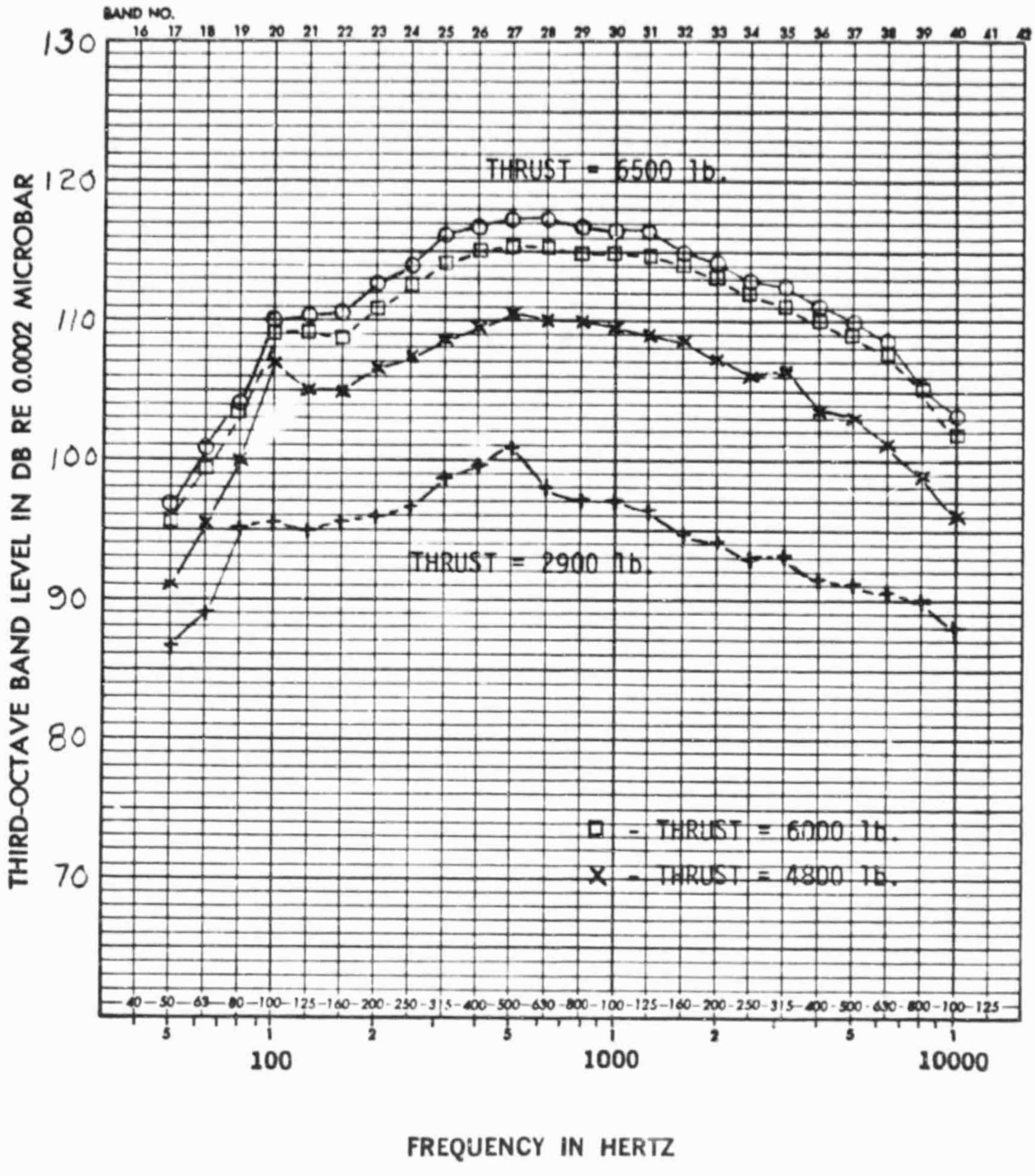

Figure 22. One third octave band sound pressure level at 110° microphone position. Flaps 53°.

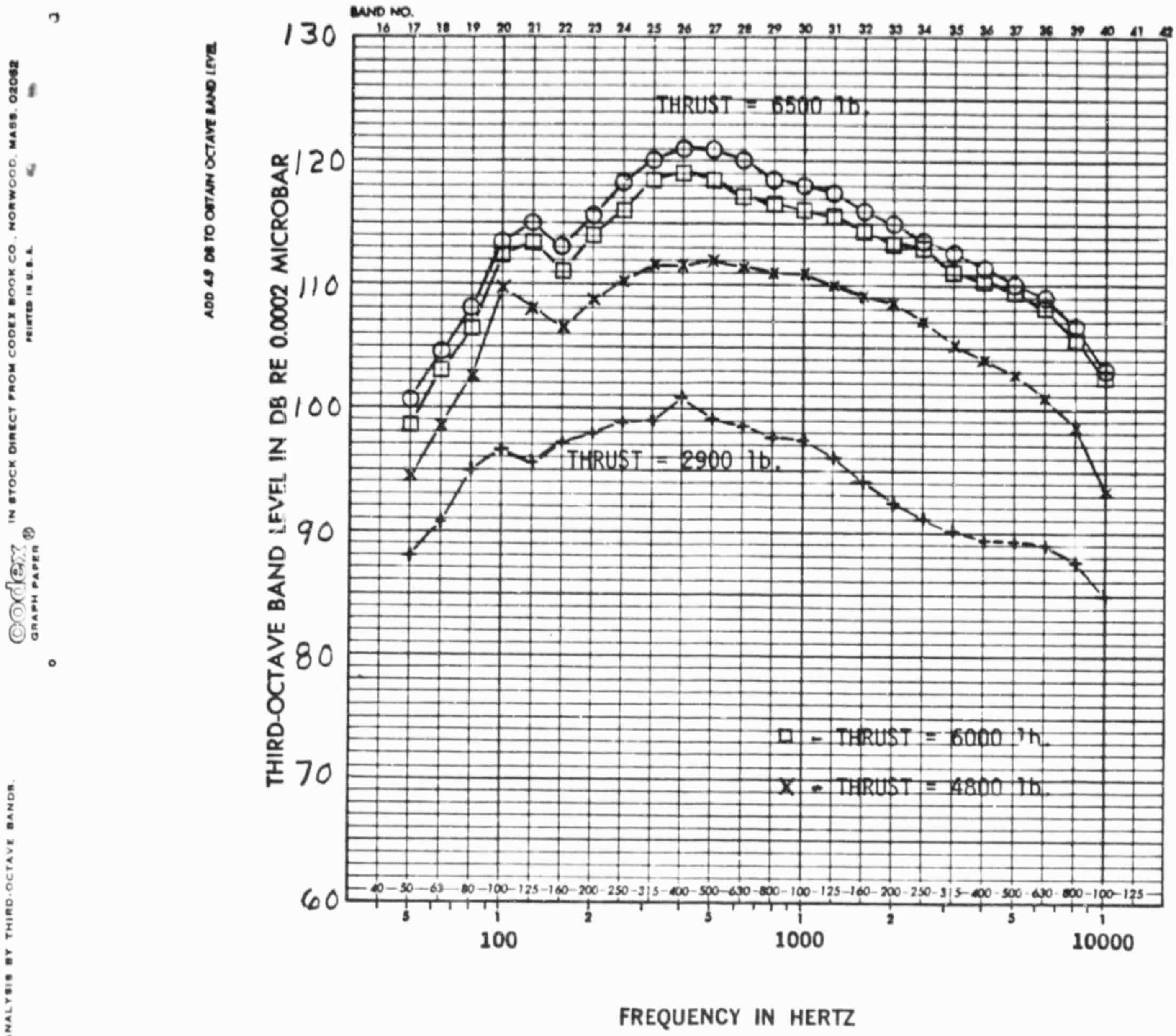

Figure 23. One third octave band sound pressure level at 120° microphone position. Flaps 53°.

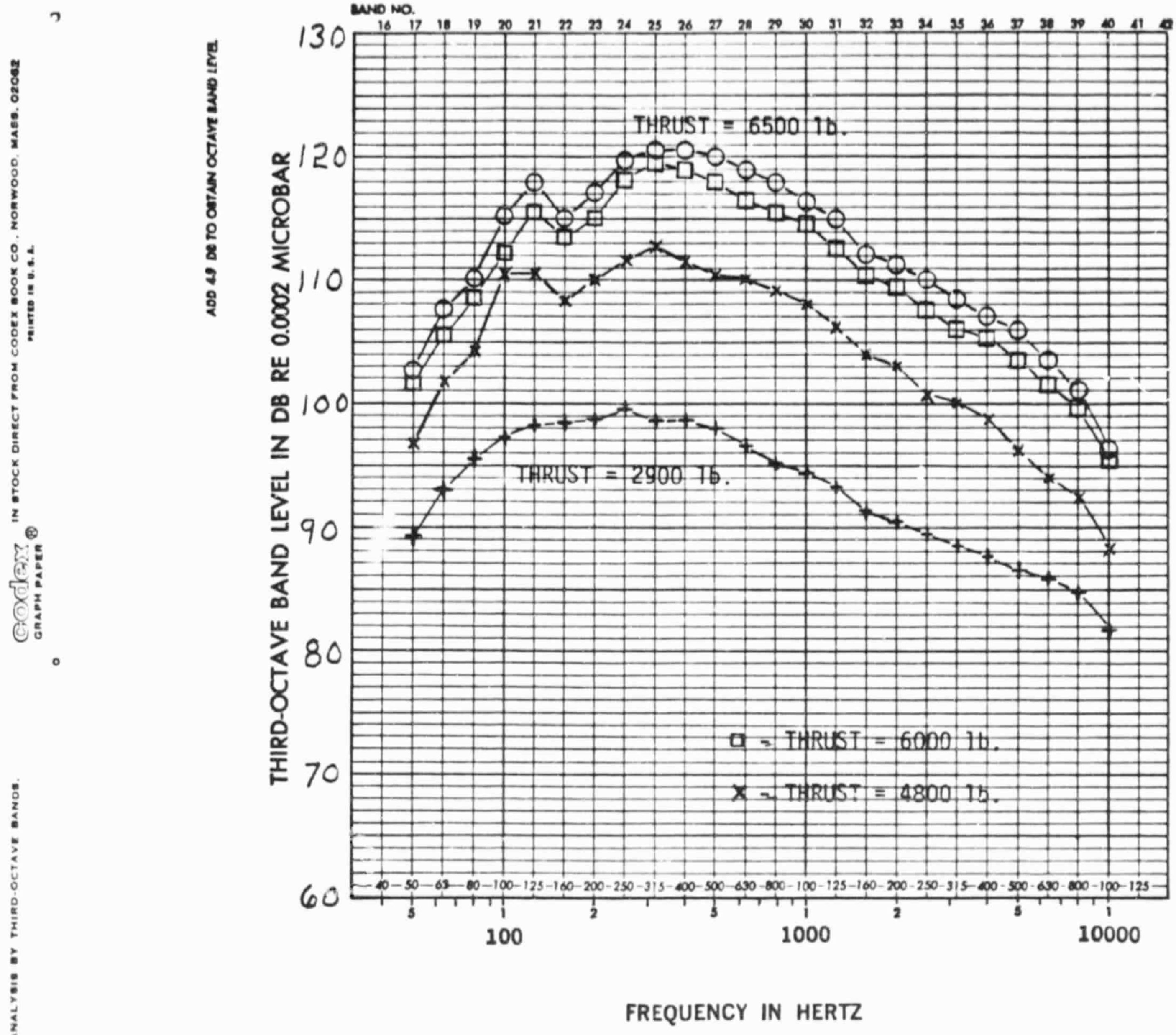

Figure 24. One third octave band sound pressure level at 130° microphone position. Flaps 53°.

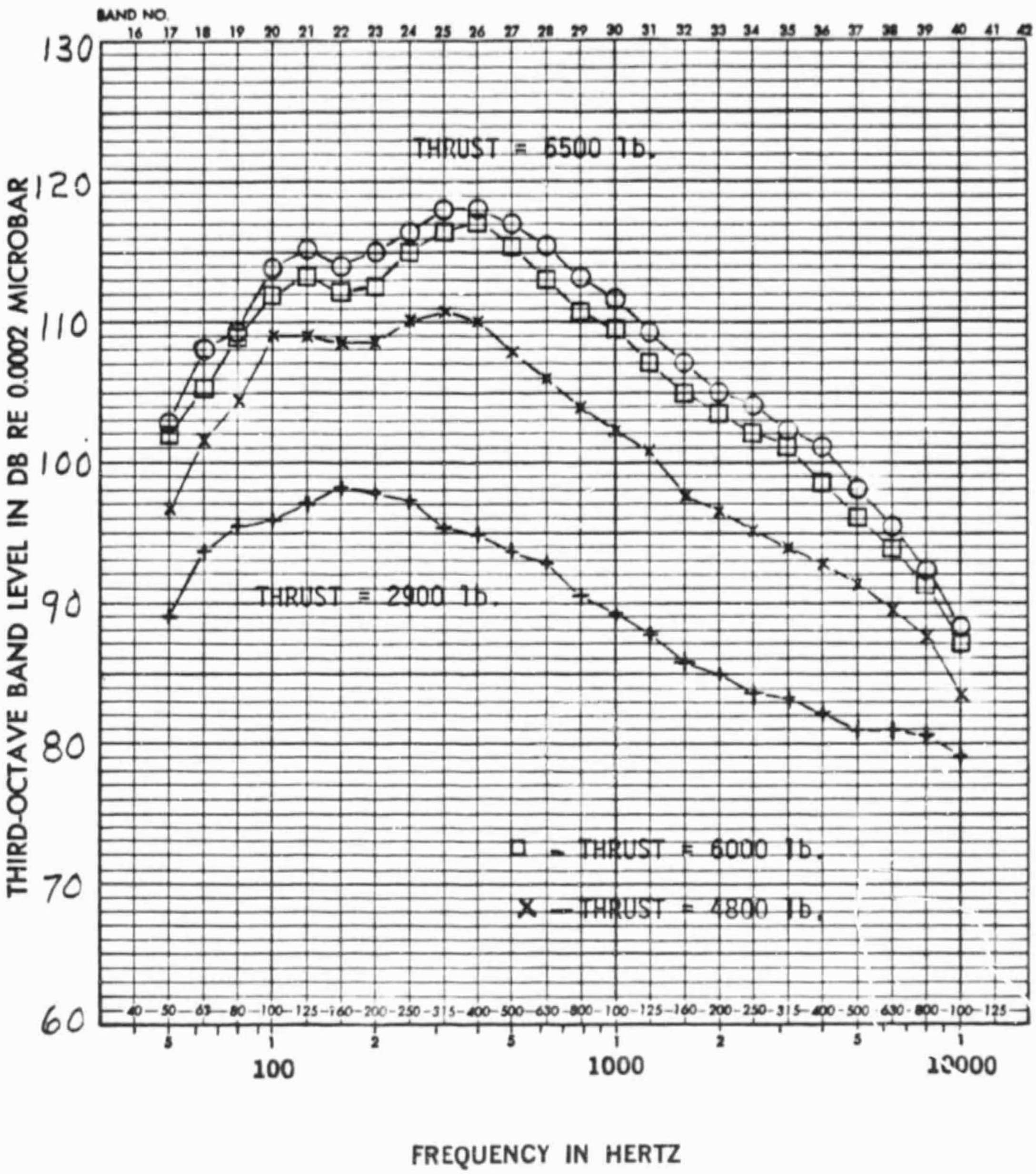

Figure 25. One third octave band sound pressure level at 140° microhpone position. Flaps 53°.

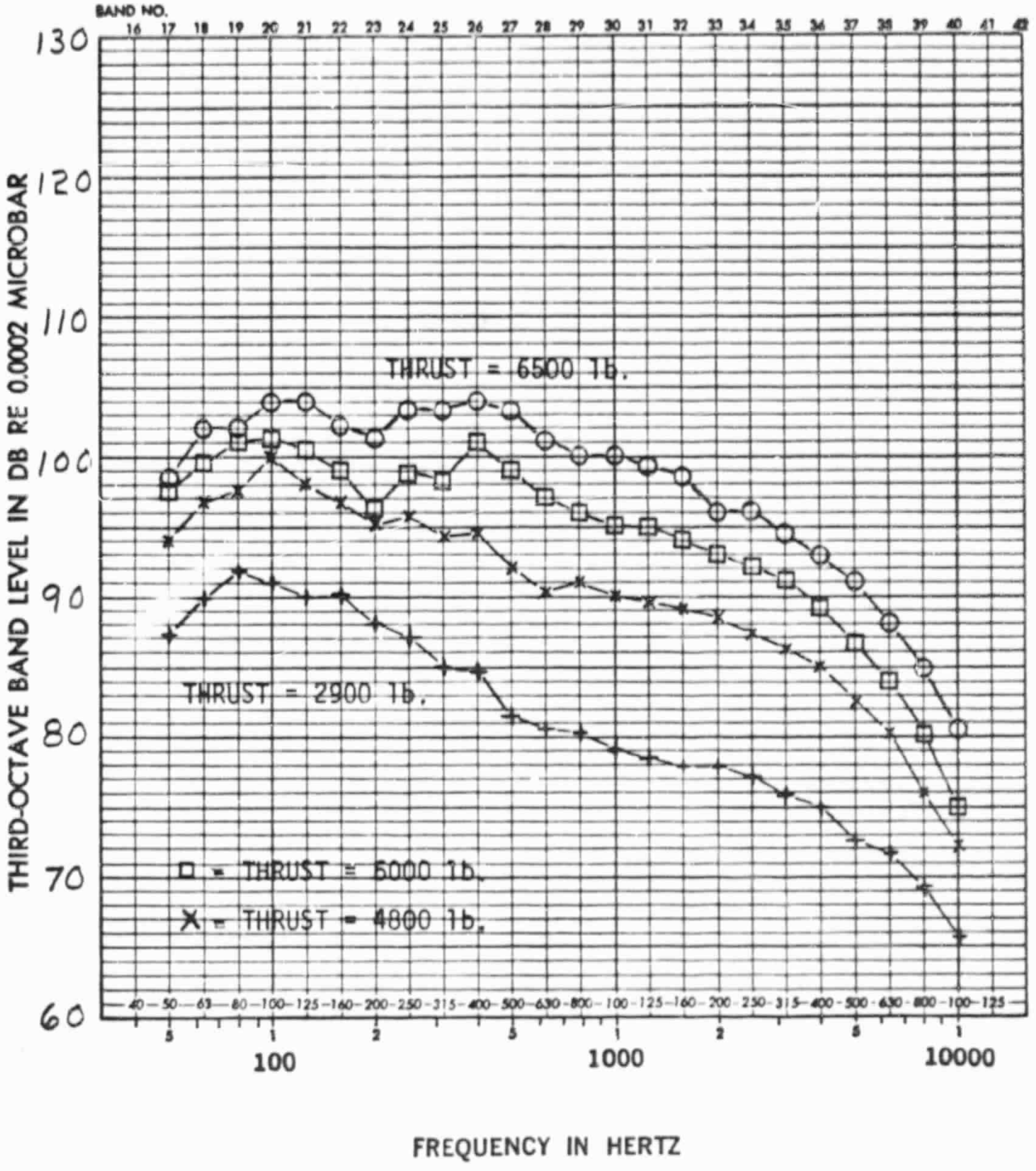

Figure 26. One third octave band sound pressure level at 150° microphone position. Flaps 53°.

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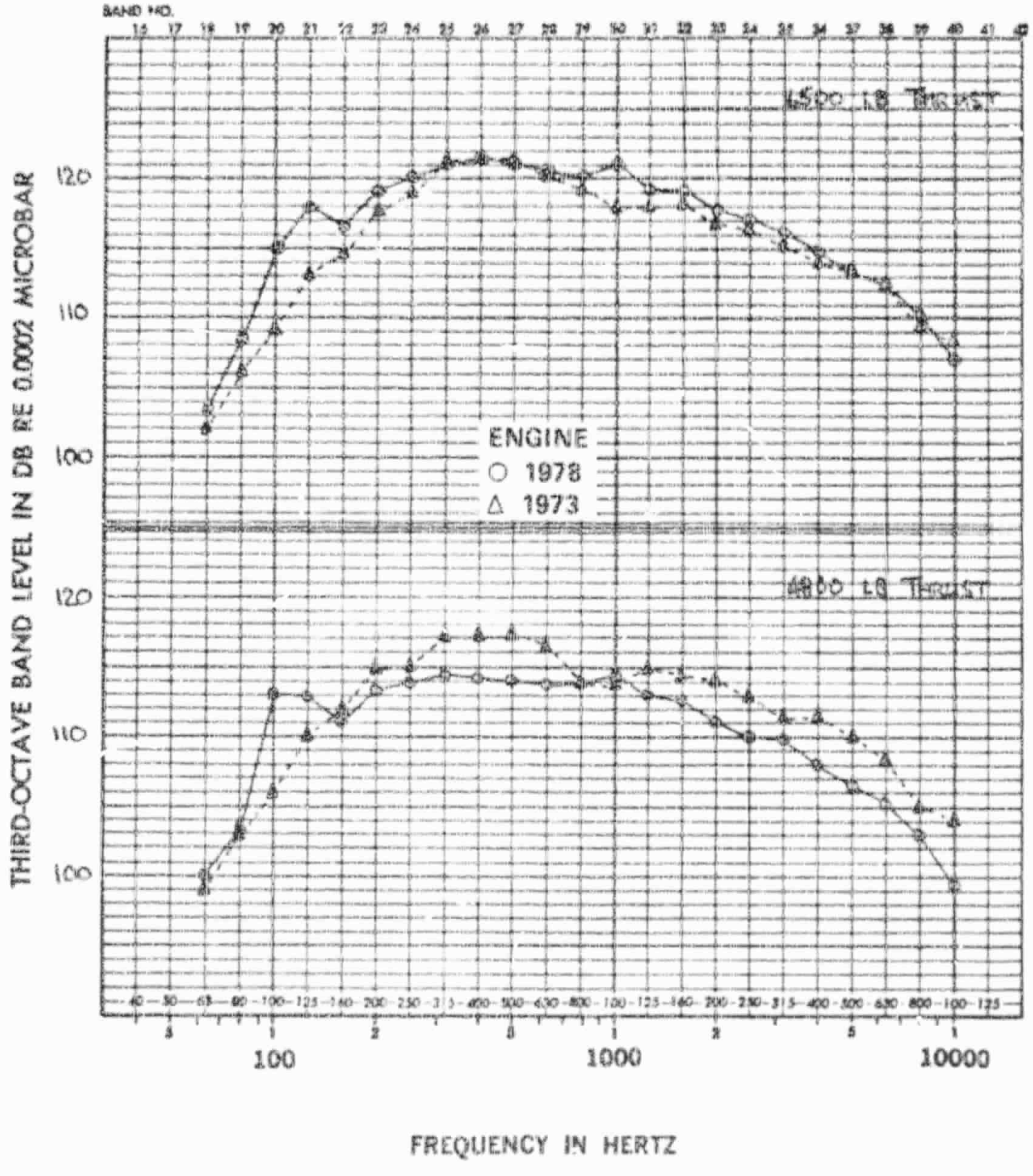

Figure 27. Comparison of one third octave band sound pressure level produced by 1978 and 1973 engines. Microphone position 120°. Flaps 5.6°.

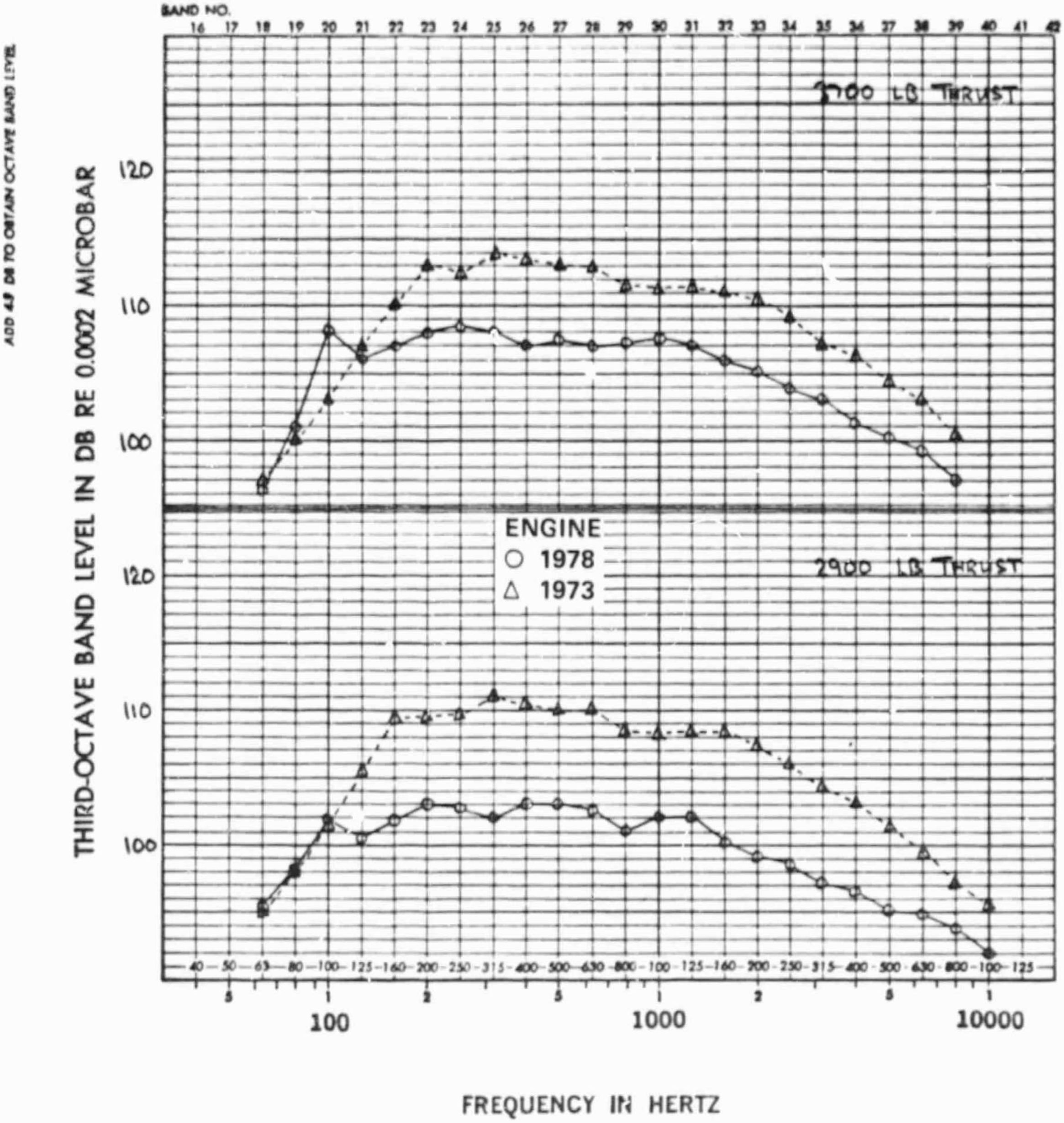

Figure 27. Concluded.

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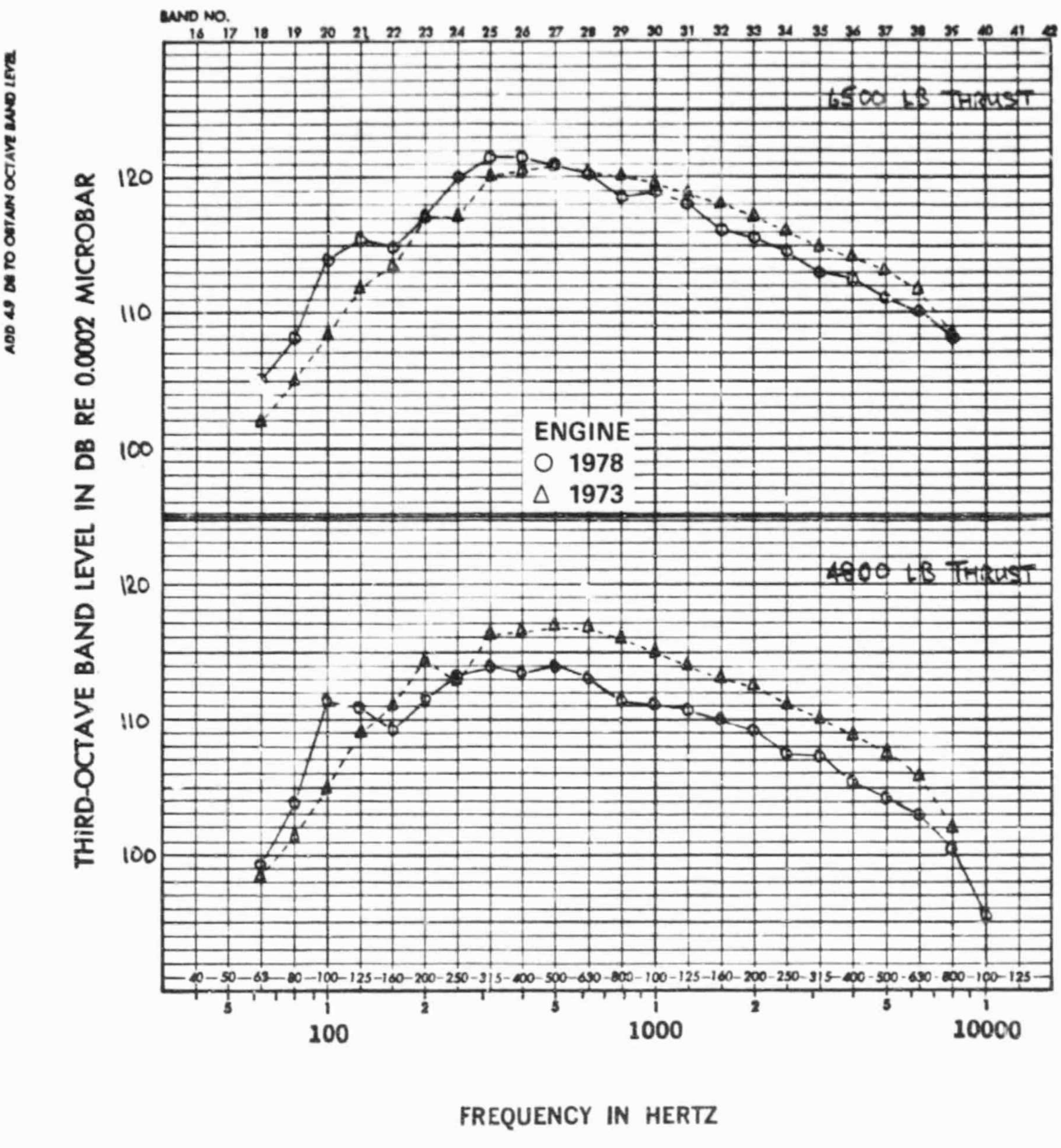

Figure 28. Comparison of one third octave band sound pressure level produced by 1978 and 1973 engines. Microphone position 120°. Flaps 30°.

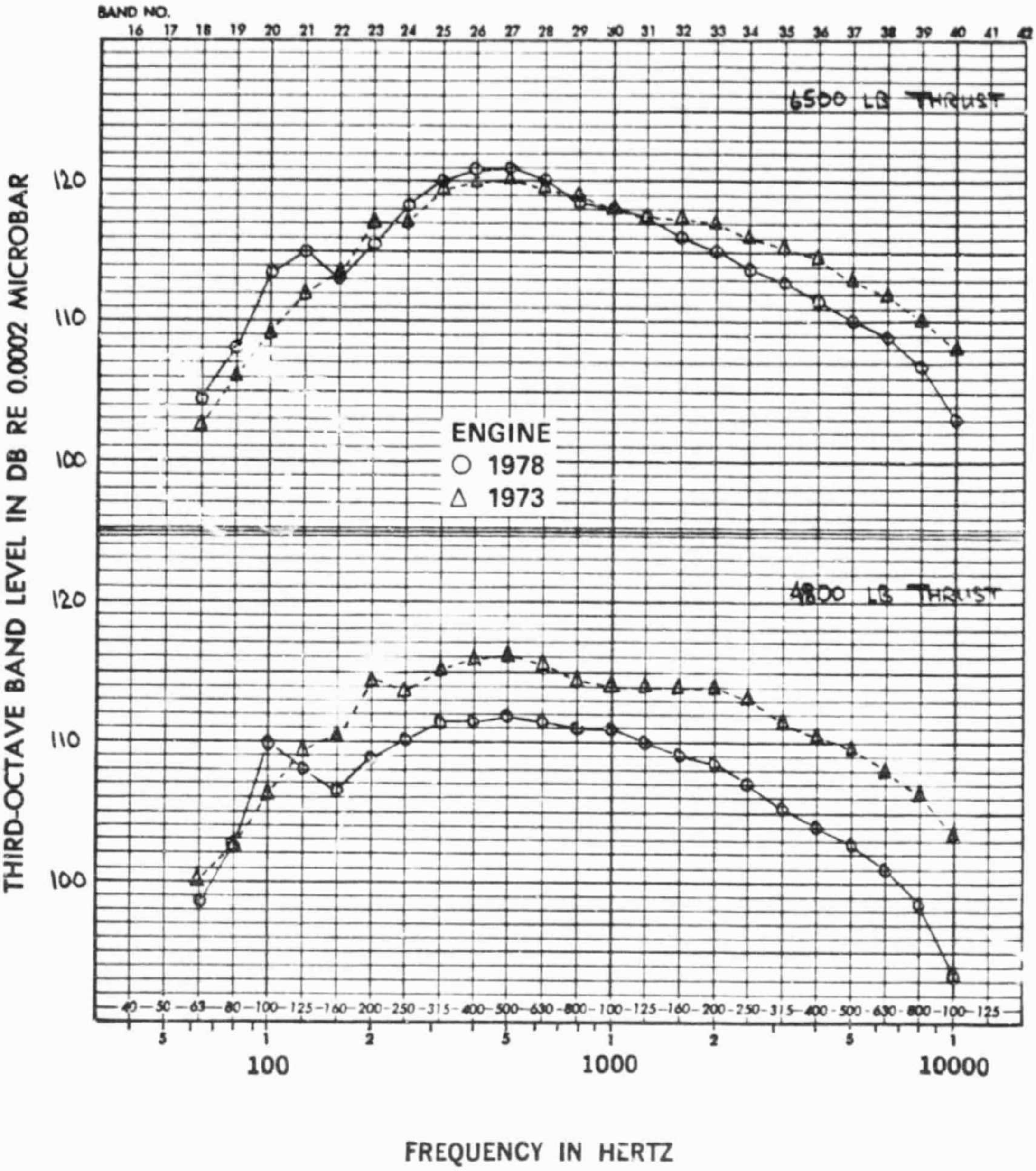

Figure 29. Comparison of one third octave band sound pressure level produced by 1978 and 1973 engines. Microphone position 120°. Flaps 53°.