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ACOUSTICALLY TREATED GROUND TEST NACELLE FOR THE GENERAL ELECTRIC TF34 TURBOFAN

FINAL REPORT

by D.P. Edkins

GENERAL ELECTRIC COMPANY
AIRCRAFT ENGINE GROUP
LYNN, MASSACHUSETTS/CINCINNATI, OHIO

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
January 31, 1972

NASA Lewis Research Center
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CONTRACT NAS 3-14338 Modification No. 1

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TABLE OF CONTENTS

ABSTRACT

Report describes the ground test quiet nacelle for the TF34 engine. The suppression treatment consists of cylindrical splitters in the inlet and fan exhaust ducts plus duct wall treatment and core exhaust wall treatment. Aerodynamic design analysis of the inlet and exhaust ducts and overall engine performance with pressure losses from the acoustic treatment is included. The objectives of the test program are to obtain noise data for a heavily suppressed high bypass turbofan with various arrangements of exhaust systems and acoustic treatment, and to provide a basis for the powerplants for the Quiet Experimental STOL Aircraft (Questol).

SUMMARY

This report describes a quiet nacelle designed for the TF34 bypass 6 turbofan which will be put on test at Edwards AFB during the second quarter of 1972. The nacelle is a ground test vehicle with features for maximum flexibility to evaluate the noise and performance characteristics of various suppressive treatments, of separate and mixed flow, and of externally-blown flap STOL propulsion systems. The basic ultimate objective for the Questol aircraft, which would have a flight version of the ground test nacelle, is a noise level at a 150 m (500 ft) sideline, maximum of 95 PNdB for bulk absorber and single degree of freedom (SDOF) treatment.

The noise treatment is as follows:

	Inlet		Fan Exhaust		Core Exhaust
	Wall	Splitters	Wall	Splitters	
Geometry	Cylindrical	(3) Cylindrical	Cyl/conical	(2) Cylindrical	Cylindrical
Type	SDOF	SDOF	Bulk and SDOF		Bulk absorber
Estimated Attenuation PNdB	16-18		26(bulk)	22.5(SDOF)	16

The estimated inlet and exhaust system aerodynamic data is as follows:

	Inlet	Fan Exhaust	Core Exhaust
M. No. in suppressed passage	.58	.45	.56
Pressure drop, %	3.2	6.0	2.3

Fan and core areas and jet velocities are adjustable as shown below:

	Fan Exhaust	Core Exhaust
Area range, m^2 (in.^{2})	.40 - .50 (630-790)	.11 - .18 (180-280)
Velocity range, isentropic, m/s (ft/s)	220-255 (720-838)	303-418 (995-1372)
Velocity range, exit plane	220-255 (720-838)	243-372 (797-1222)

Various mixed flow cycles have been investigated under another section of the overall study and are reported fully separately. For reference, the basic (i.e. with the fan operating line the same as in a standard TF34) mixed flow cycle compares as follows:

Exhaust area	m^2 (in.^{2})	.65 (1007)
Exhaust velocity	m/s (ft/s)	248 (813)

The nacelle is designed to be mounted in the General Electric outdoor acoustic test stand at EAFB, with provisions for supporting the inlet and nacelle (and also the velocity decayer exhaust expected to be added later) without loading up the engine. Both acoustic and engine performance instrumentation is provided in the basic initial design.

LIST OF SYMBOLS

A	area
AE	effective area
F	thrust
H	boundary layer shape factor
H	height
L	length
M, M.No.	Mach number
N	rotational speed
P	pressure
R	radius
T	temperature
V	velocity
W	airflow
f	frequency
f_o	characteristic frequency
q	dynamic pressure
Δ	change in parameter
δ	pressure correction factor, P/reference P
δ^*	displacement thickness of boundary layer
λ	wave length
θ	temperature correction factor, T/reference T
θ	boundary layer momentum loss

Subscripts

F	fan
I	isentropic
L	local
S	static
T	total
n	net

NOTE: Numerical subscripts refer to engine stations as defined in Figure

Abbreviations

BPR	bypass ratio
EAFB	Edwards Air Force Base
EFTC	Edwards Flight Test Center
F	Fahrenheit
Hz	hertz
K	Kelvin
L. E.	leading edge
MDOF	multiple degree of freedom
N	newton
PNdB	perceived noise level in decibels
PNL	perceived noise level
SDOF	single degree of freedom
SLS	sea level static
SPL	sound pressure level
STOL	short takeoff and landing
dB	decibel
ft	foot
in.	inch
kg	kilogram
m	meter
mm	millimeter
rad	radian
s	second

INTRODUCTION

Recent emphasis on the reduction of aircraft noise in the vicinity of airports has led to a concerted effort on noise reduction. This has been directed along three basic paths: (1) reduction of source noise, (2) suppression of noise (3) reduction of noise at the observation point by flight path manipulation.

The TF34 ground test nacelle in this report addresses to path (2) above. The acoustic suppression system is designed to provide a much larger amount of sound attenutation than has been previously attempted or achieved.

The nacelle is designed with maximum flexibility to accept a variety of exhaust systems, noise treatments, nozzle areas and duct areas.

This report is a summary technical description of the nacelle covering mechanical design, acoustic treatment, aerodynamic analysis and engine performance as installed in the nacelle. Closely related programs sponsored by NASA are: a velocity decayer design study, which will lead to the procurement of nozzles designed to reduce the jet velocity rapidly before impingement in a high lift STOL type wing flap; manufacture of a wing and flap section, and Quiet Experimental STOL Aircraft program which will use flight type nacelles based on the ground test version described herein.

1. GENERAL DESCRIPTION

Figure 1 shows the general arrangement of the nacelle. It consists of:

a. A separate inlet assembly. This has a separate bellmouth of standard contour, a cylindrical main section and a shorter cylindrical spacer section. There are three acoustically treated splitters which as an assembly can be placed in each of the three axial positions. The splitters can also be removed one at a time. The cylindrical sections have acoustically treated walls. Acoustic design data is given in Section 2.

b. A main nacelle assembly consisting of the fan exhaust duct and nozzle and the core engine cowling. The fan exhaust duct walls are both fully acoustically treated, and there are two treated splitters. All surfaces are cylindrical over the center portion, and maximum use is made of conical surfaces for transition from the fan to the main duct and to the nozzle. This results in minimum manufacturing cost. There is a main support pylon at the 12 o'clock position and a narrow pylon at 6 o'clock so that each main duct consists of a separate annular arc. Each of these annuli is hinged to the pylon and can be lifted to gain access to the core engine. The nacelle is supported independently of the engine to avoid excessive engine flange loads. The splitter leading and trailing edges are contoured to provide flow turning at each end of the cylindrical annulus. A 12.7 mm (.5") thick space between the inner and outer treatment on each splitter allows either the use of thicker treatment or thinner splitters to provide lower duct air velocity with the same thickness of treatment.

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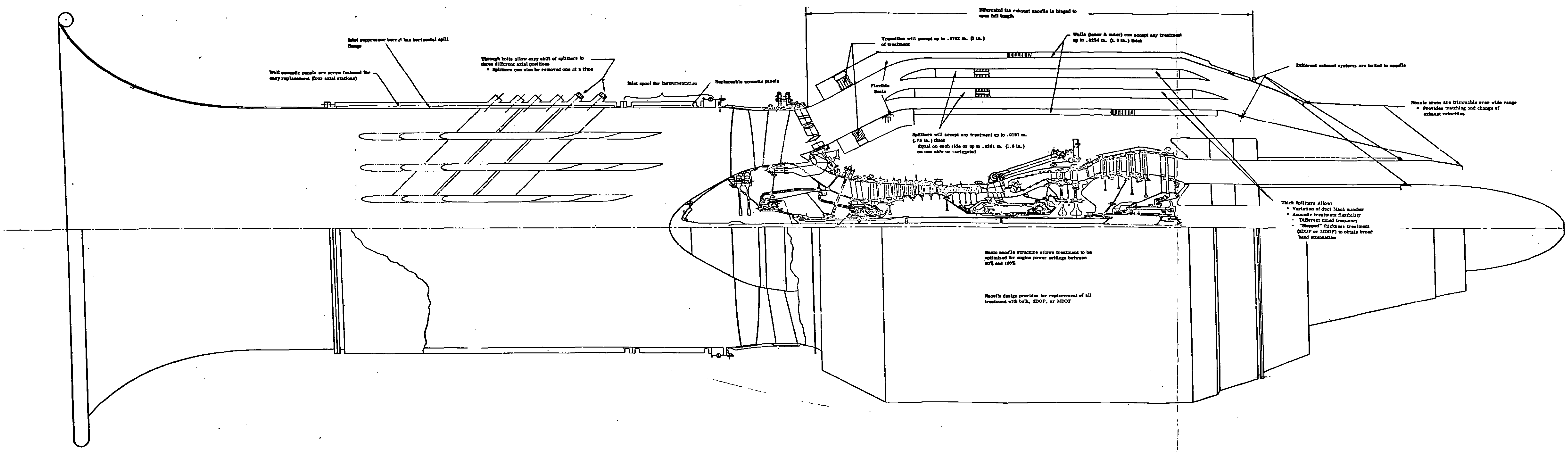

Figure 1. Features of TF34 Quiet Nacelle.

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Provisions are made for cooling flow inside the core cowling in a manner similar to that in the standard TF34. Acoustic and aerodynamic design data are given in Sections 2 and 3 respectively. The acoustic panels are removable. The basic nacelle structure is 3 mm (.125") thick aluminum.

c. The core exhaust system, consisting of an annular duct lined with low and high frequency suppressive material.

Provisions are made to bolt on different exhaust systems. It is thus possible to test the following:

a) Separated flow

b) Mixed confluent flow

c) Mixed flow with a core mixer

d) Mixed flow with core mixer and velocity decayer

The velocity decayer is being designed under a separate contract.

The various flow areas for all of these parts have been carefully selected to be adequate for possible cycle variations such as different duct losses from nominal, requirements for reduced jet velocities, etc. See Section 4 for details.

2. ACOUSTIC TREATMENT

Description

The inlet suppressor uses SDOF (Single Degree of Freedom) acoustic treatment on all surfaces. The perforated face sheet is 0.5 mm (0.020 in.) thick aluminum with 1.27 mm (0.050) diameter holes giving 6.8 percent open area. The perforated face sheet is bonded to honeycomb which is 12 mm (0.480 inch) deep. The honeycomb is bonded to an unperforated backing sheet 0.5 mm (0.020 inch) thick for the wall treatment and 6 mm (0.250 inch) thick for the three splitters.

The splitters are arranged to give a passage height of 140 mm (4.5 inches) in the three outer passages. The I.D. of the inner splitter is 0.24 m (9.5 inches). Table I summarizes the acoustic treatment at the inlet.

Figure 2 shows the inlet suppressor which consists of two basic assemblies: 1) Inlet Suppressor, NASA LeRC, Dwg CR 501945 and 2) Inlet Spool Piece, NASA LeRC Dwg CF 501947.

The wall of the Inlet Spool Piece is acoustically treated like the suppressor and provides about 24 cm (9.5 inches) of treated length. The Inlet Spool Piece provides mounting pads for fan inlet distortion rakes.

Fan Exhaust acoustic treatment is defined by GE drawing 17A111-782, sheet 1, which is reproduced in Figure 3 . The nacelle will be built with two different kinds of treatment material:

1) Bulk suppressor (Scottfelt 3-900)

2) SDOF (6 mm (.250 inch) core aluminum honeycomb)

Scottfelt is a polyurethane open cell foam which is heat processed to obtain the desired cell size and density. Scottfelt 3-900 has 3.5 pores per millimeter (90 pores per linear inch) and weighs 28 kg/m^3 (1.73 lb/ft^3).

Table II shows the important data on the fan exhaust acoustic treatment. By using the nacelle drawing, Figure 1 and Table II , a complete definition of the acoustic treatment is obtained.

The SDOF treatment uses honeycomb having 6 mm (0.250 inch) cell size. This is chosen to minimize closure of the perforated sheet holes. Fabrication of the treatment requires care to assure no reduction in the open area during bonding where the bond material can reduce hole size.

The Core Exhaust Suppressor is a strictly ground test device designed to suppress the turbine and combuster noise to a level sufficiently low to allow accurate assessment of the performance of the other more challenging suppression devices (fan exhaust, velocity decayer, etc.). A cross-section of the suppressor can be seen in Figure 1 as part of the Quiet Nacelle.

The suppressor uses a bulk suppression material called Cerafelt. Two different depths are used: 1) 100 mm (4 inches) deep for the low frequency combuster noise, and 2) 13 mm (0.5 inch) deep for the high frequency noise.

The total length of suppression is 0.9 m (36 inches); 30 cm (12 inches) of length are 10 cm (4 inches) deep and the other 0.6 m (24 inches) are 13 mm (0.5 inch) deep.

The "Cerafelt CR-400", 64 kg/m^3 (4 lb/ft^3) is used with 1 mm (0.040 inch) thick perforated sheet of Inco 625 material. Behind the perforated sheet, 0.13 mm (0.005 in. dia.) wire screen (31 x 28 per cm (80 x 72 per inch)) of Inco 600 is used to contain the Cerafelt. The perforated sheet has an open area of 23.3% using holes of 3 mm (0.127 inch) diameter.

The outer shell of the suppressor is a structural member which is required to carry the hot mixer, one end of inner suppressor, and the tail cone. The aft flanges of the suppressor are designed to readily adapt to the various exhaust configurations.

TABLE I - FAN INLET SUPPRESSOR

	Perforated Sheet			SDOF	Total Treated	Passage	Splitter	
	Holes	Open Area %	Thickness	Depth	Length	Height	Thickness	Radius
OUTER WALL	1.27 mm dia (0.05 in.)	6.8	0.5 mm (0.020 in.)	12.2 mm (0.480 in.)	1.35 m* (53 in.)	114 mm (4.5 in.)	-	0.56 m (22.0 in.)
OUTER SPLITTER	1.27 mm dia (0.05 in.)	6.8	0.5 mm (0.020 in.)	12.2 mm (0.480 in.)	0.71 m (28 in.)	114 mm (4.5 in.)	32 mm (1.25 in.)	0.43 m (16.2 in.)
MIDDLE SPLITTER	1.27 mm dia (0.05 in.)	6.8	0.5 mm (0.020 in.)	12.2 mm 480 in.)	0.71 m (28 in.)	114 mm (4.5 in.)	32 mm (1.25 in.)	0.28 m (11.1 in.)
INNER SPLITTER	1.27 mm dia (0.05 in.)	6.8	0.5 mm (0.020 in.)	12.2 mm (0.480 in.)	0.57 m (22.5 in.)	120 mm (4.75 in.)	32 mm (1.25 in.)	136 mm (5.4 in.)

* The 1.35 m (53 in.) includes 24 cm (9.5 in.) of treated wall in the Inlet Spool Piece.

TABLE II - FAN EXHAUST SUPPRESSOR

Ref: G. E. Drawing 17A111-782

Bulk Treatment

1. Scottfelt 3-900
2. Perforated sheet
   Hole size 1.5 mm (0.060 in.) dia
   Open area 30 percent
   Thickness 0.76 mm (0.030 in.)

SDOF Treatment

1. Honeycomb 6 mm (0.250 in.) cell size
2. Perforated sheet
   Hole size 1.5 mm (0.060 in.) dia
   Open area 22.5 percent
   Thickness 0.76 mm (0.030 in.)

DATA APPLICABLE TO BOTH TYPE TREATMENTS

	Treatment Depth	Splitter Thickness	Treatment Length SDOF	Treatment Length Scottfelt	Passage Height
	50 mm (2.0 in.)	-	0.43 m (17.0 in.)	0.43 m (17.0 in.)	
OUTER WALL	25 mm (1.0 in.)	-	1.17 m (46 in.)	1.17 m (46 in.)	52 mm (2.06 in.)
	6 mm (0.250 in.)	-	0.22 m (8.5 in.)	.22 m (8.5 in.)	
OUTER SPLITTER	12.7 mm (0.500 in.)	41 mm (1.61 in.)	1.07 m (42 in.)	1.15 m (45.3 in.)	52 mm (2.06 in.)
INNER SPLITTER	12.7 mm (0.500 in.)	41 mm (1.61 in.)	1.19 m (47 in.)	1.27 m (50 in.)	52 mm (2.06 in.)
	50 mm (2.0 in.)	-	0.34 m (13.5 in.)	0.34 m (13.5 in.)	
INNER WALL	25 mm (1.0 in.)	-	1.32 m (52 in.)	1.32 m (52 in.)	52 mm (2.06 in.)

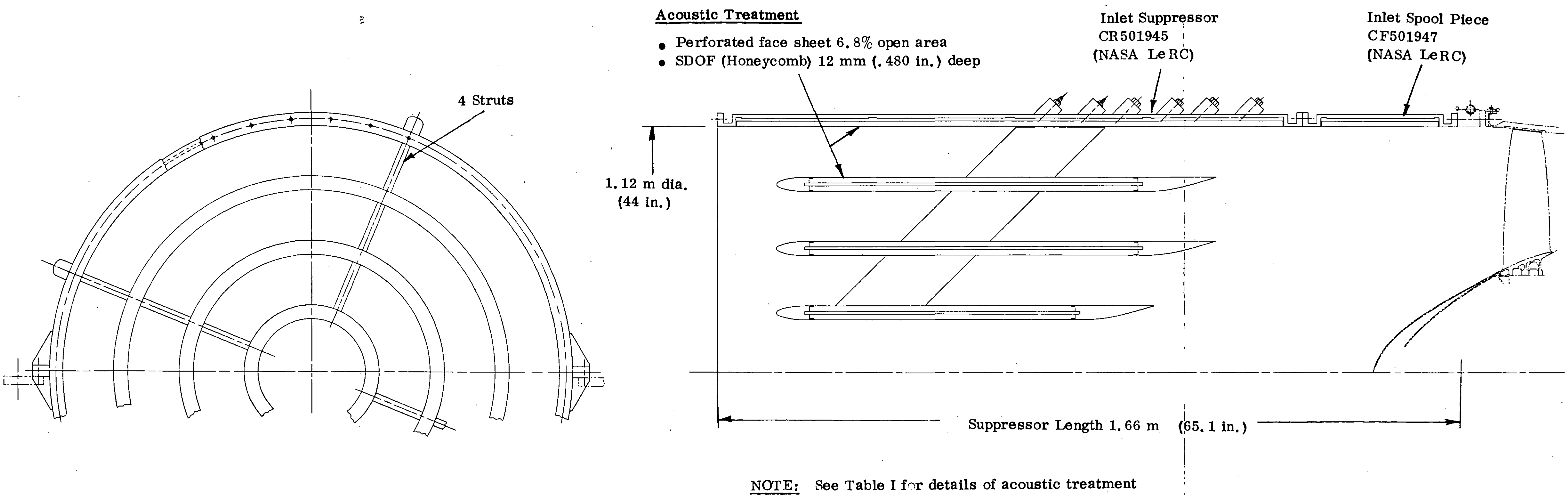

Figure 2. Acoustic Inlet Assembly NASA TF34 Quiet Nacelle.

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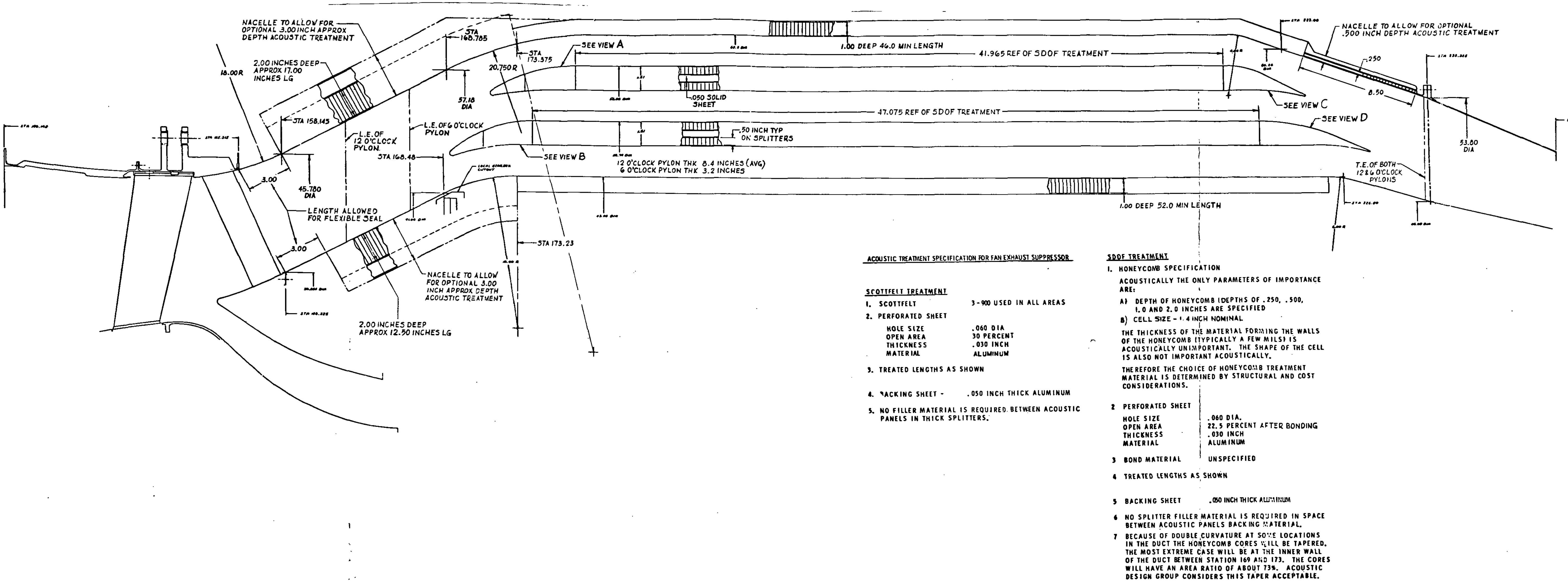

Figure 3. NASA Quiet Nacelle TF34 Ground Test Control Drawing.

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Performance

The measured perceived noise level (PNL) of a single unsuppressed engine extrapolated to the 150m (500') sideline is shown in Figure 4 as a function of thrust for the maximum noise angles in the forward and aft directions. The spectral distribution versus sound pressure level (SPL) for the maximum fore and aft angles at 100% thrust is shown in Figure 5. Also shown are the calculated spectra of the several sources and their respective PNL values.

The basis for these unsuppressed noise estimates is as follows:

Fan Noise - The unsuppressed fan noise predictions are based on data recorded from the TF34/GE2 engine (201-005) tested at Edwards Flight Test Center in March of 1971. Several test runs (plus repeats) were made at the various power settings. The maximum forward and maximum aft 60m (200') sideline spectra, for the power settings nearest to 100%, 80%, and 50%, were obtained by averaging repeated runs. Figure 6 is an example, for the maximum aft spectrum at 103% power. The 60m (200') sideline static jet noise, at the appropriate angle, was predicted from scale model coannular nozzle acoustic data as discussed under "Jet Noise", below. This jet noise, and a ground reflection peak at 125 Hz, were then subtracted to obtain "fan only" spectra, as is illustrated in Figure 6. These fan spectra were further adjusted to the desired power setting, this adjustment being determined from the curve of measured maximum PNL versus percent thrust (Figure 7).

Turbine Noise - Unsuppressed turbine broadband noise was predicted by a semi-empirical method based on published (Rolls-Royce) turbine rig noise data. Turbine pure tone noise levels were determined by a GE analytical method. Measurements of TF34 core nozzle noise were made during the Edwards Center tests, using a directional acoustic array. At the lower power settings, where jet noise did not totally mask out turbine noise, the predicted turbine noise levels compared favorably with those actually measured.

Jet Noise - The jet noise predictions are based on noise data recorded from scale model nozzles of similar configuration, tested in a model facility. The data was scaled up to TF34 size, based on standard scaling procedures and on data recorded from actual engine testing.

The attenuation predicted for the various treatments described earlier in this section is shown in Figures 8, 9, 10 and 11 as plots of Δ SPL vs. frequency. The different sections of treatment indicated in the exhaust configuration are employed to give the required suppression bandwidth. Each section is designed to peak at frequencies such that the total suppression in terms of maximum Δ PNdB is optimized. The inlet suppressor is designed to peak at 3150 Hz, the optimum tuning frequency based on the Noy weighted unsuppressed forward radiated fan noise spectrum. The core exhaust configuration is designed to give at least 10 dB suppression over a frequency range of 400 Hz to 10000 Hz. The two indicated sections of core treatment are designed such that the suppression bandwidth requirements are met.

The predicted peak attenuation values and suppression bandwidth were established by using design curves such as those given in Figures 12 - 14. These type design curves are based on combinations of jet engine data and acoustic duct data that has been made available over the past few years. Engine parameters such as geometry, temperature and flow velocity are reflected in these curves.

The suppressed engine spectra resulting from the various acoustic treatments are shown in Figure 15 - 17, together with the resulting PNL. These results are based on engine cycle data presented in Table III.

TABLE III

JET NOISE PARAMETERS

TF34 QUIET NACELLE

			Baseline Unsuppressed Engine		Fully Suppressed Engine	
Core Nozzle Area A_8, m^2 $(in.^2)$			.135	(209)	.81	(281)
Fan Nozzle Area A_{28}, m^2 $(in.^2)$			.40	(622)	.51	(790)
Core Isentropic Velocity, V_{91}	m/s	(ft/s)	367	(1204)	321	(1055)
Fan Exhaust Velocity, V_{28}	m/s	(ft/s)	267	(881.5)	231	(759)
Core Airflow, W_8	Kg/s	(lb/s)	21	(46.7)	20	(44.9)
Fan Airflow, W_{28}	Kg/s	(lb/s)	128	(281.9)	134	(296.5)

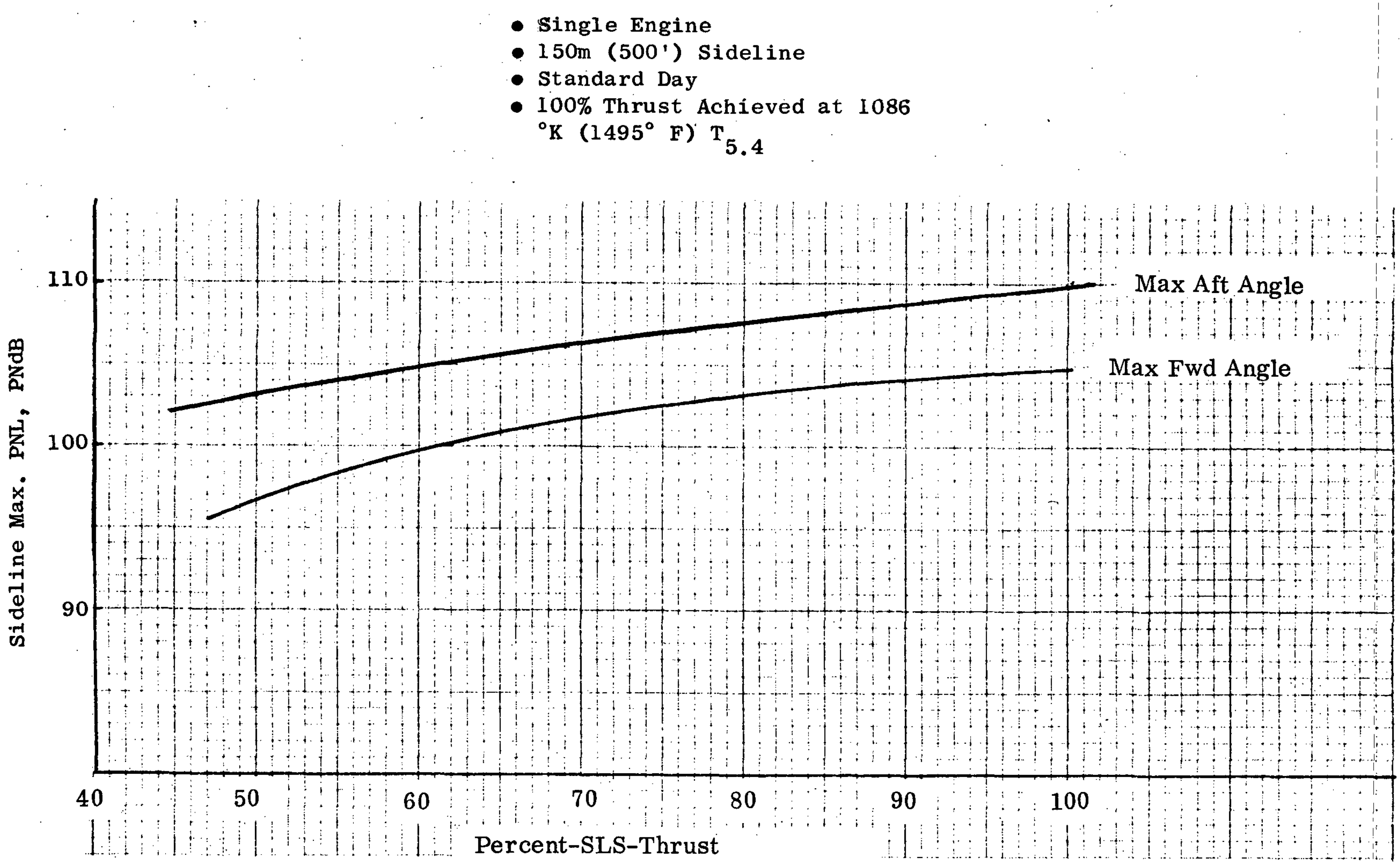

Figure 4. Unsurppressed TF34 Baseline Engine Estimated Max. PNL vs. Percent Thrust.

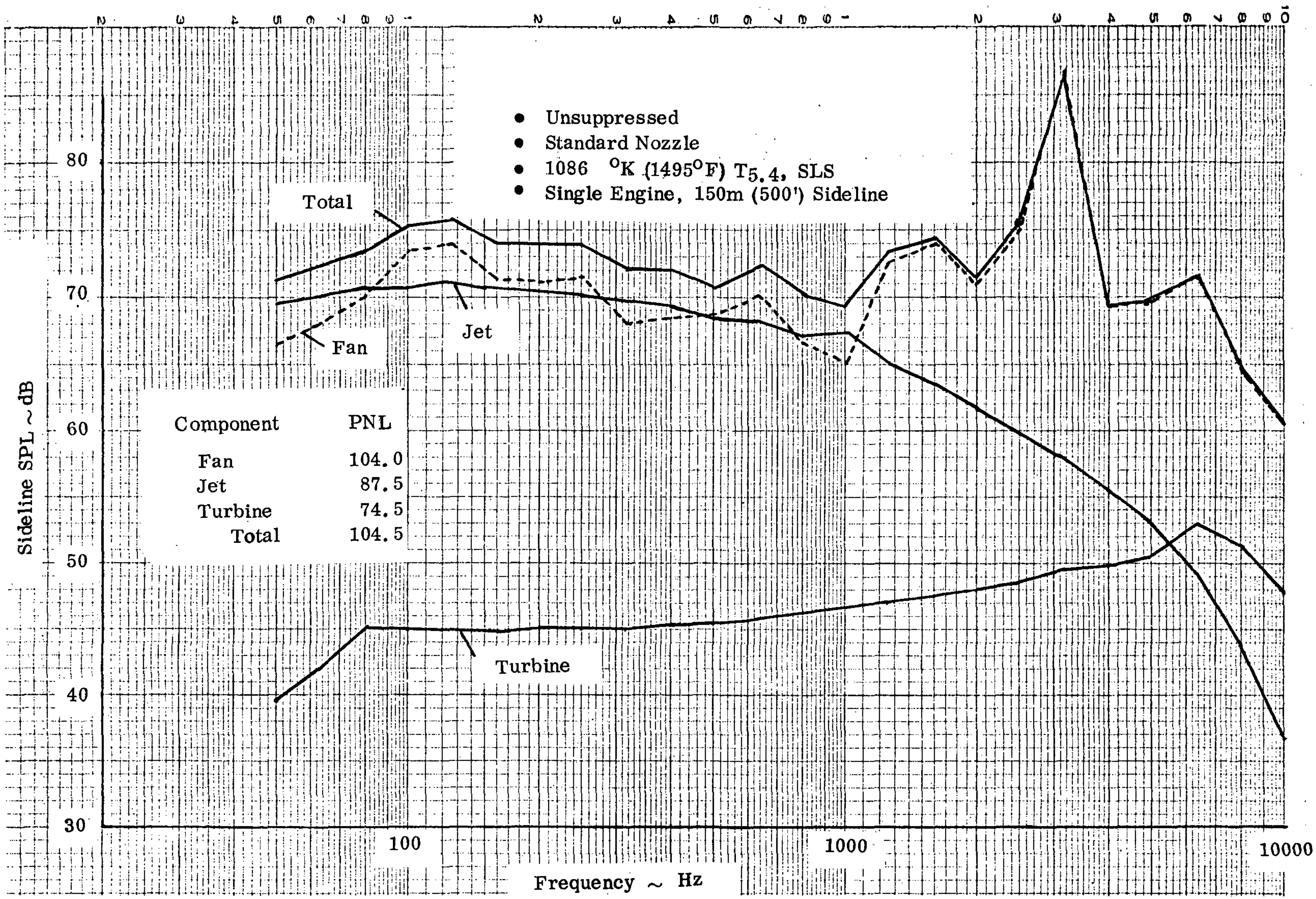

Figure 5. TF34 Baseline Engine Estimated Frequency Spectrum at Max. Forward Angle, 50°.

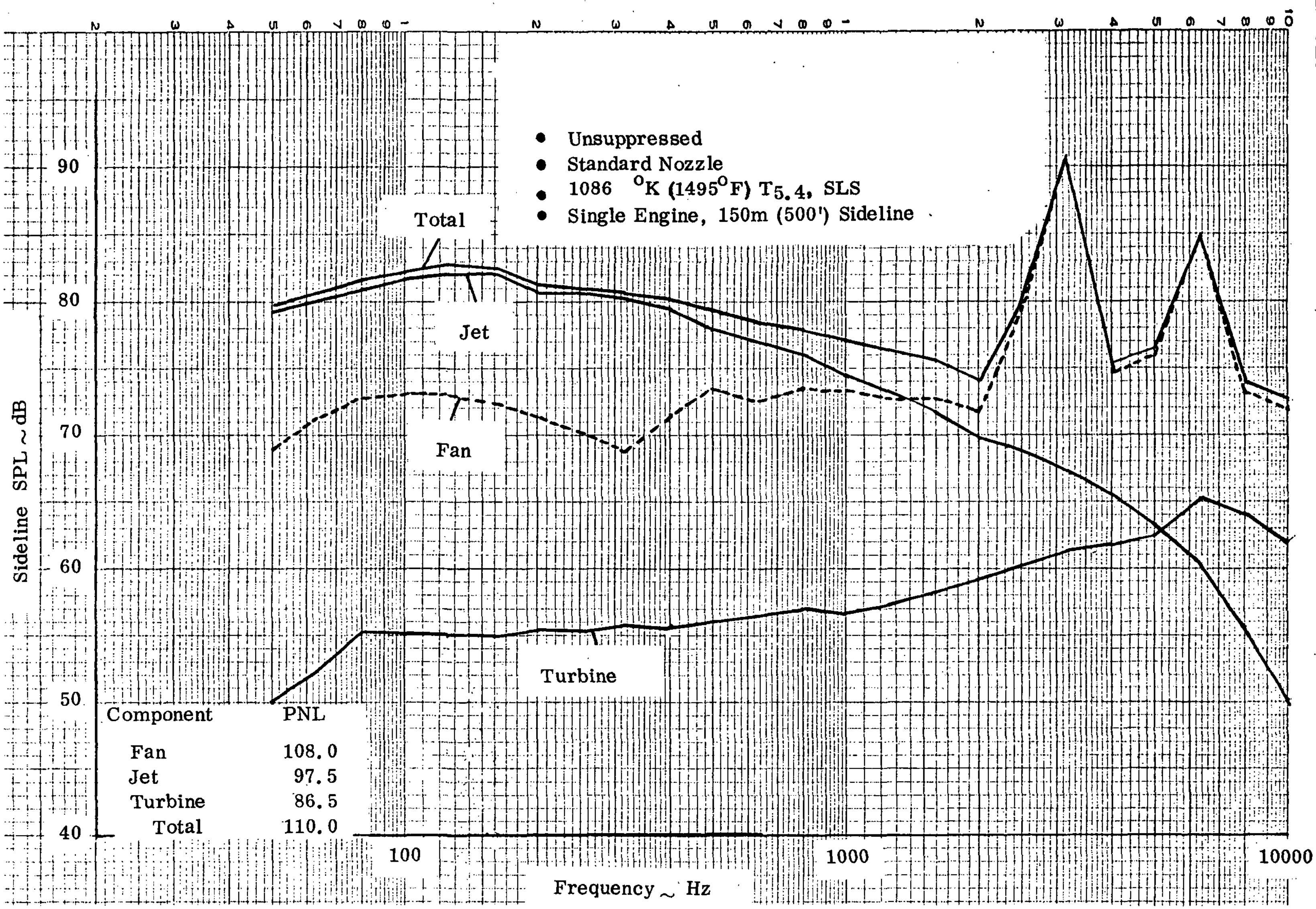

Figure 5. TF34 Baseline Engine Estimated Frequency Spectrum at Max. Aft Angle, 110° (Concluded).

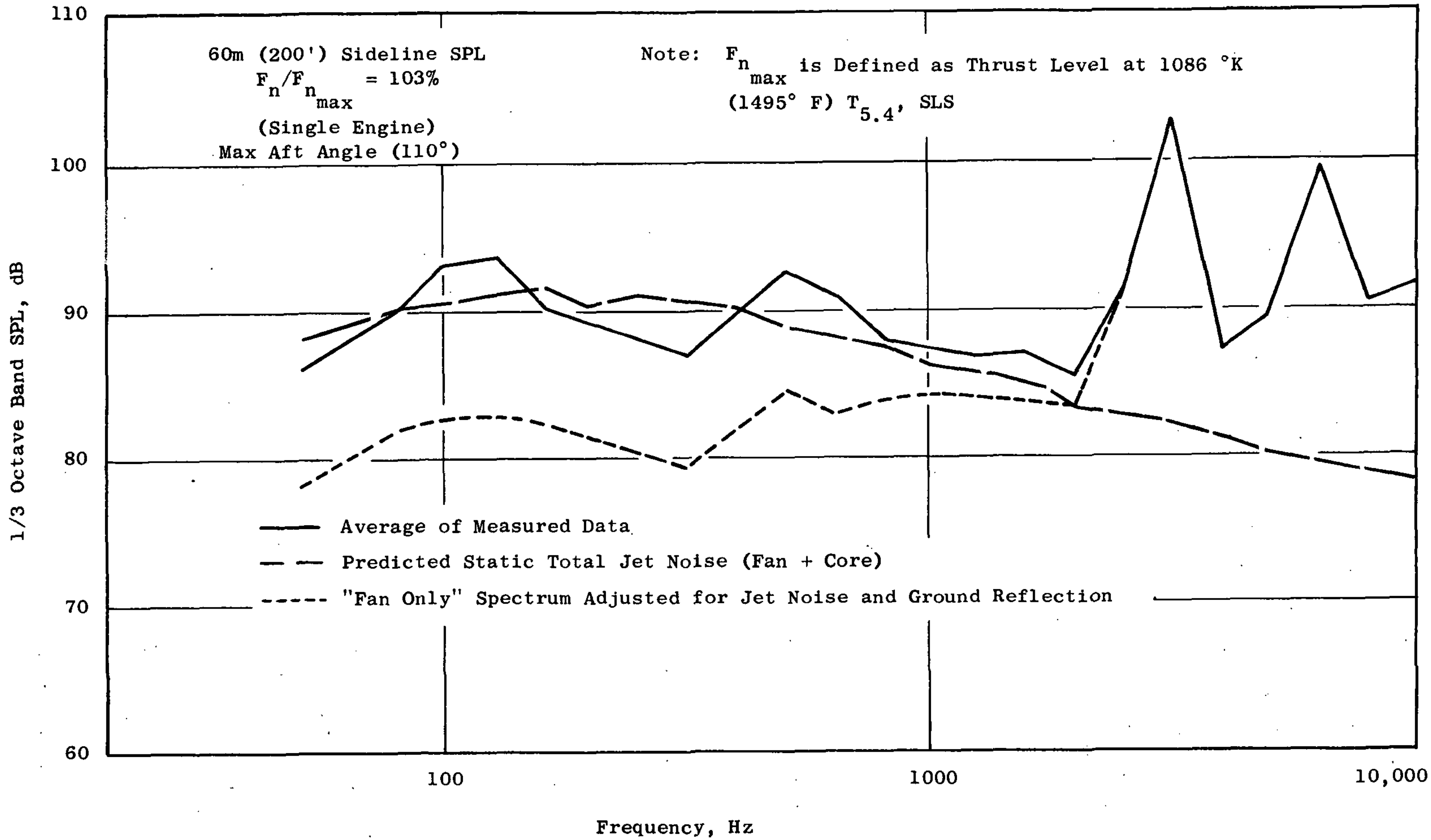

Figure 6. TF34/GE2-005 March, 1971 Edwards Test Data.

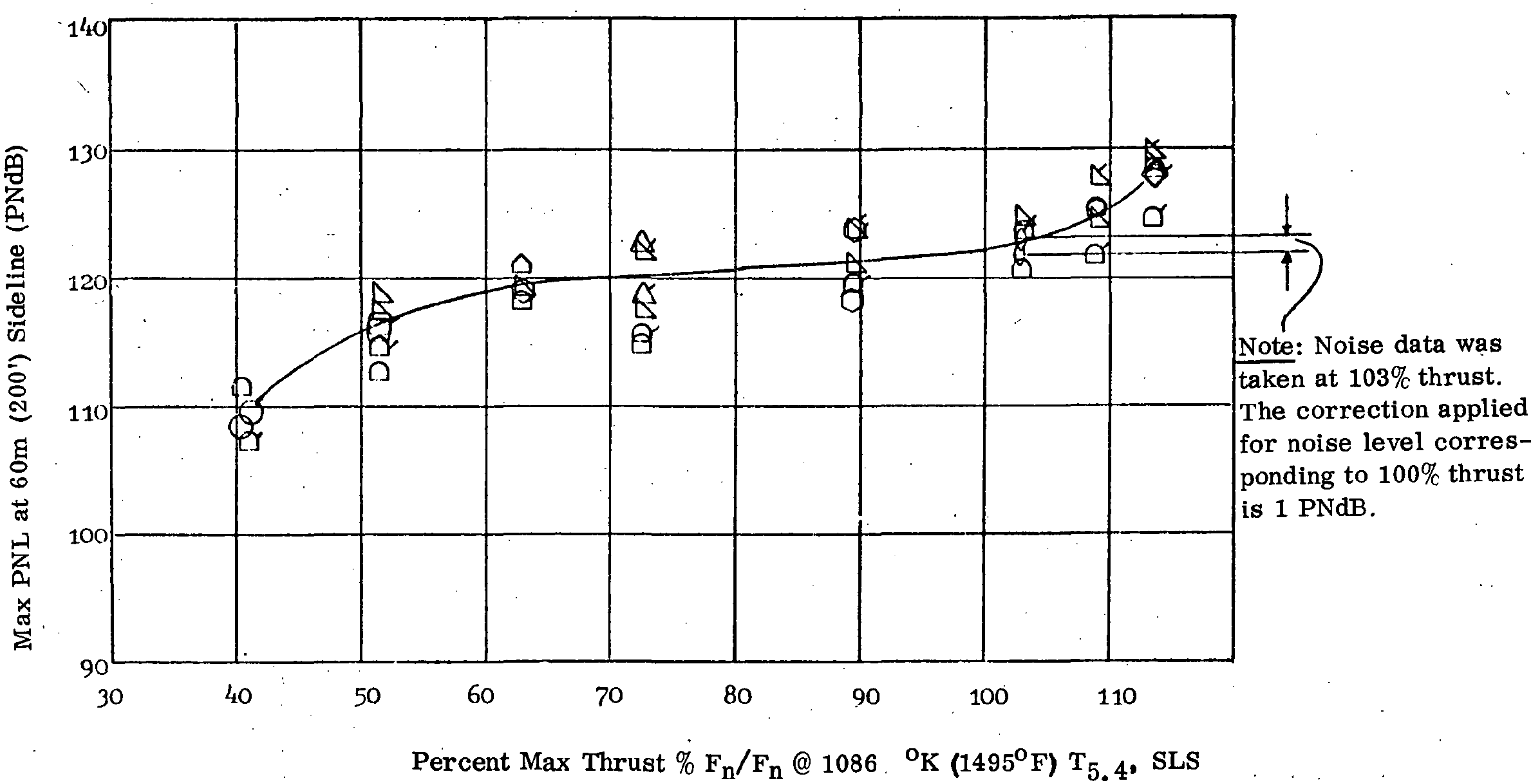

Figure 7. TF34/GE2-005 March, 1971 Edwards Test Data.

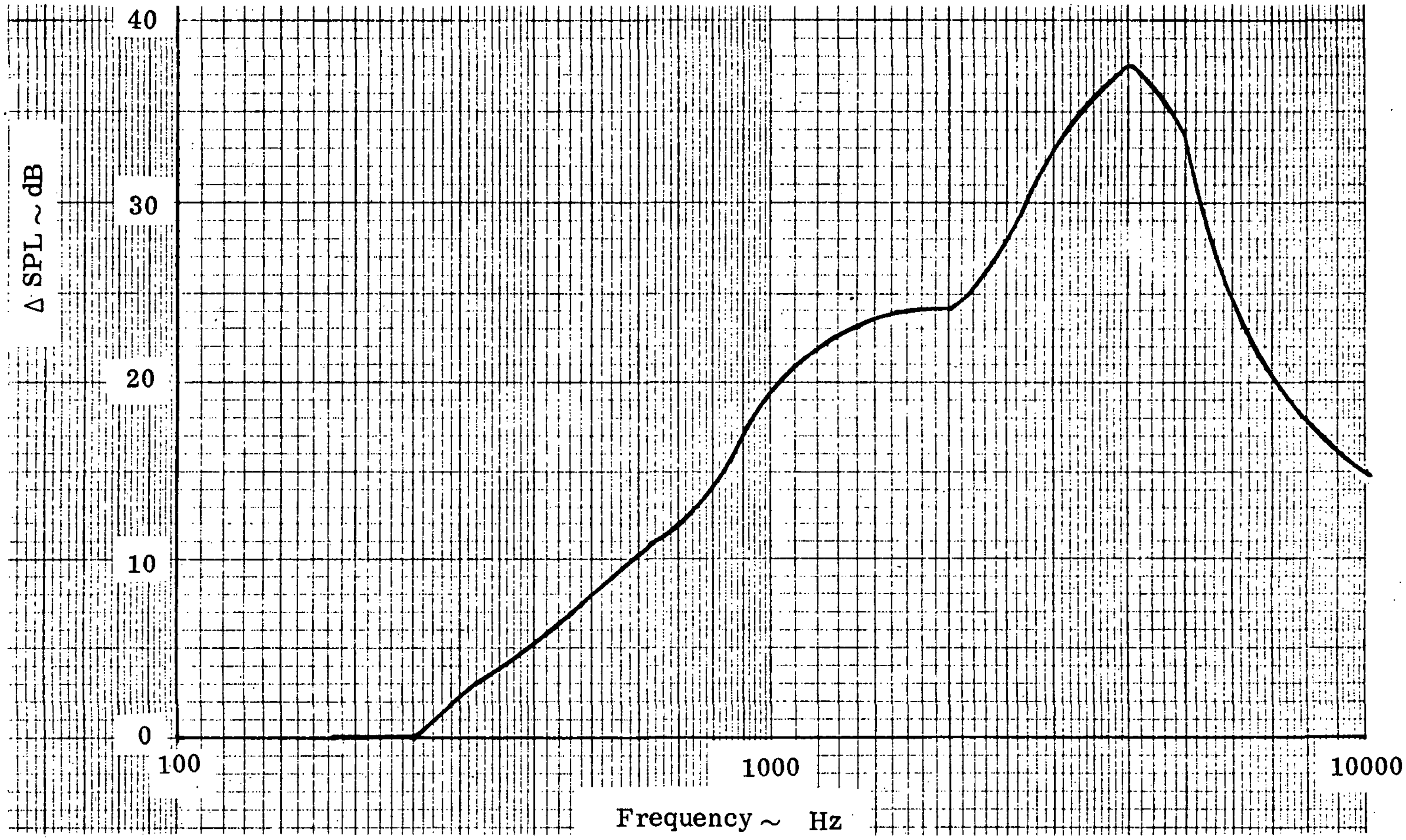

Figure 8. TF34 Exhaust Treatment Suppression Bandwidth SDOF Treatment.

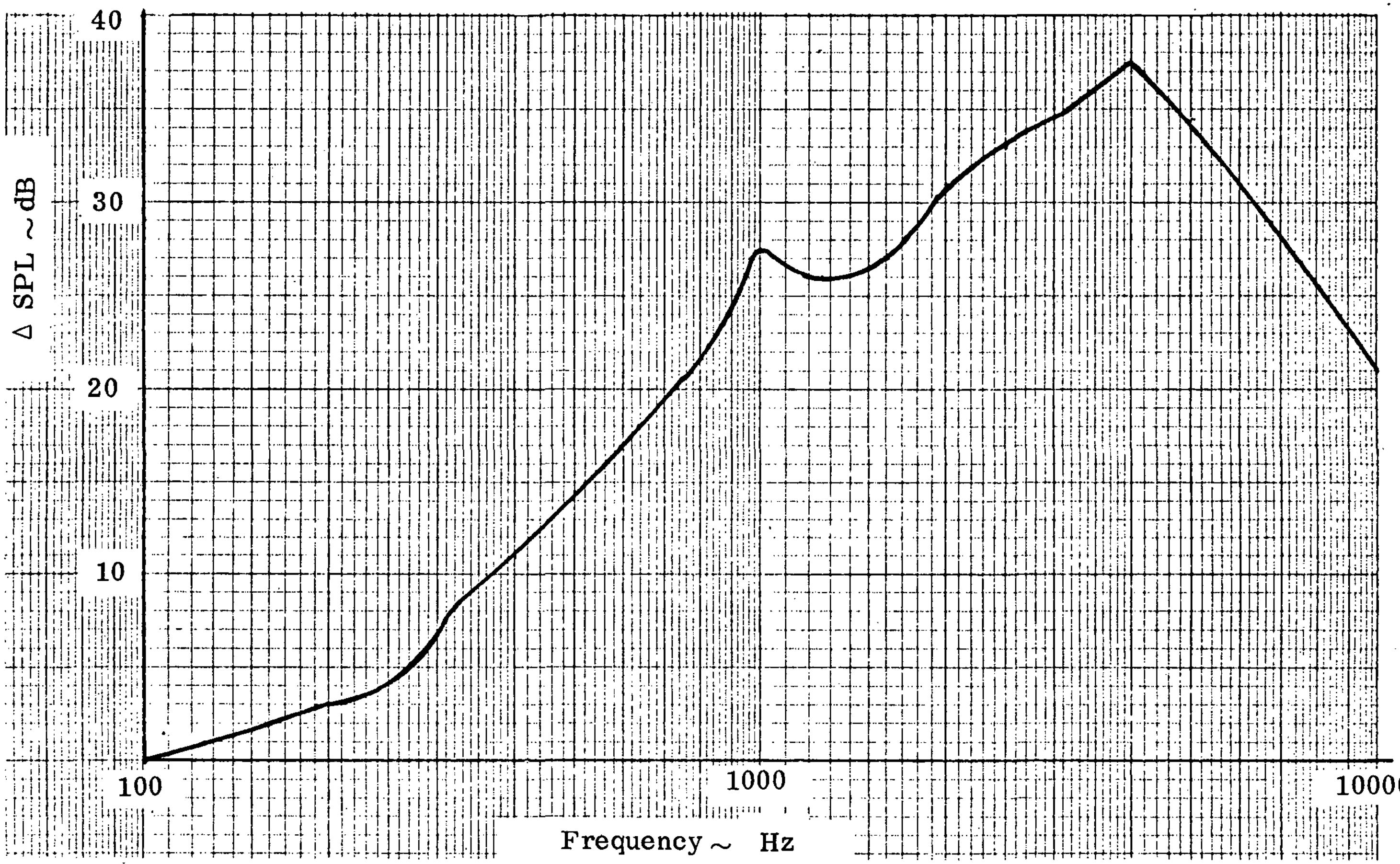

Figure 9. TF34 Fan Exhaust Treatment Suppression Bandwidth Scottfelt Treatment.

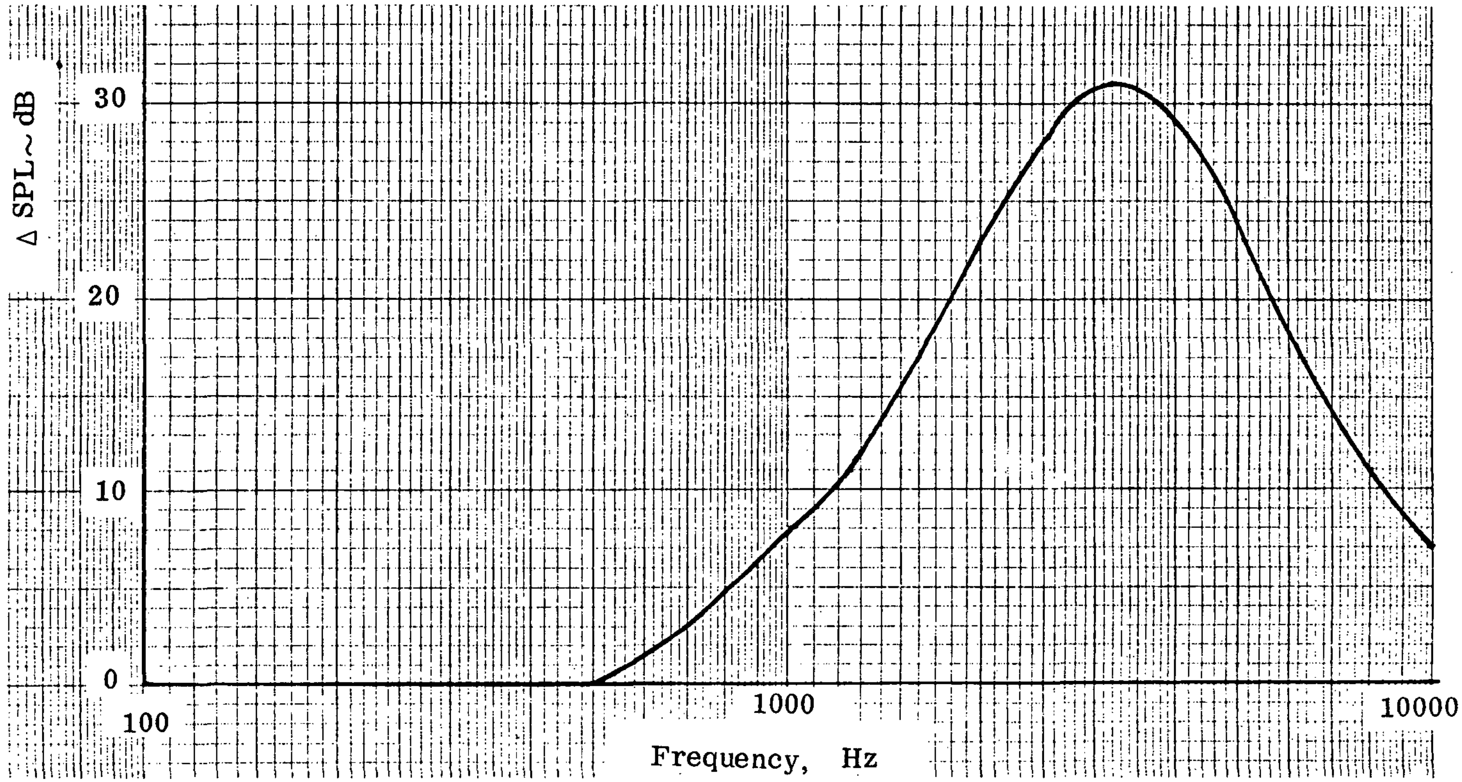

Figure 10. TF34 NASA Inlet Treatment Suppression Bandwidth SDOF Treatment.

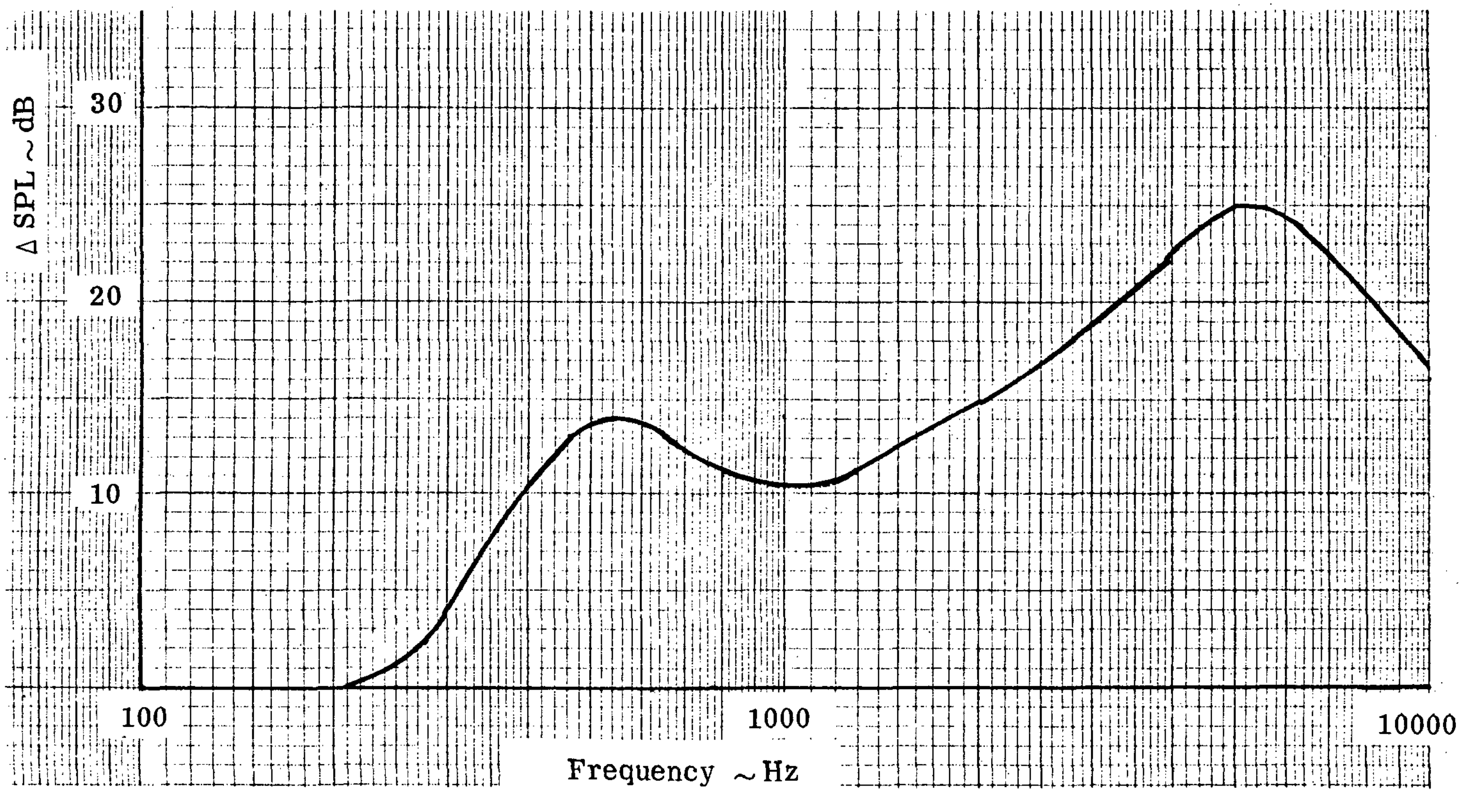

Figure 11. TF34 Core Exhaust Treatment Suppression Bandwidth Cerafelt Treatment.

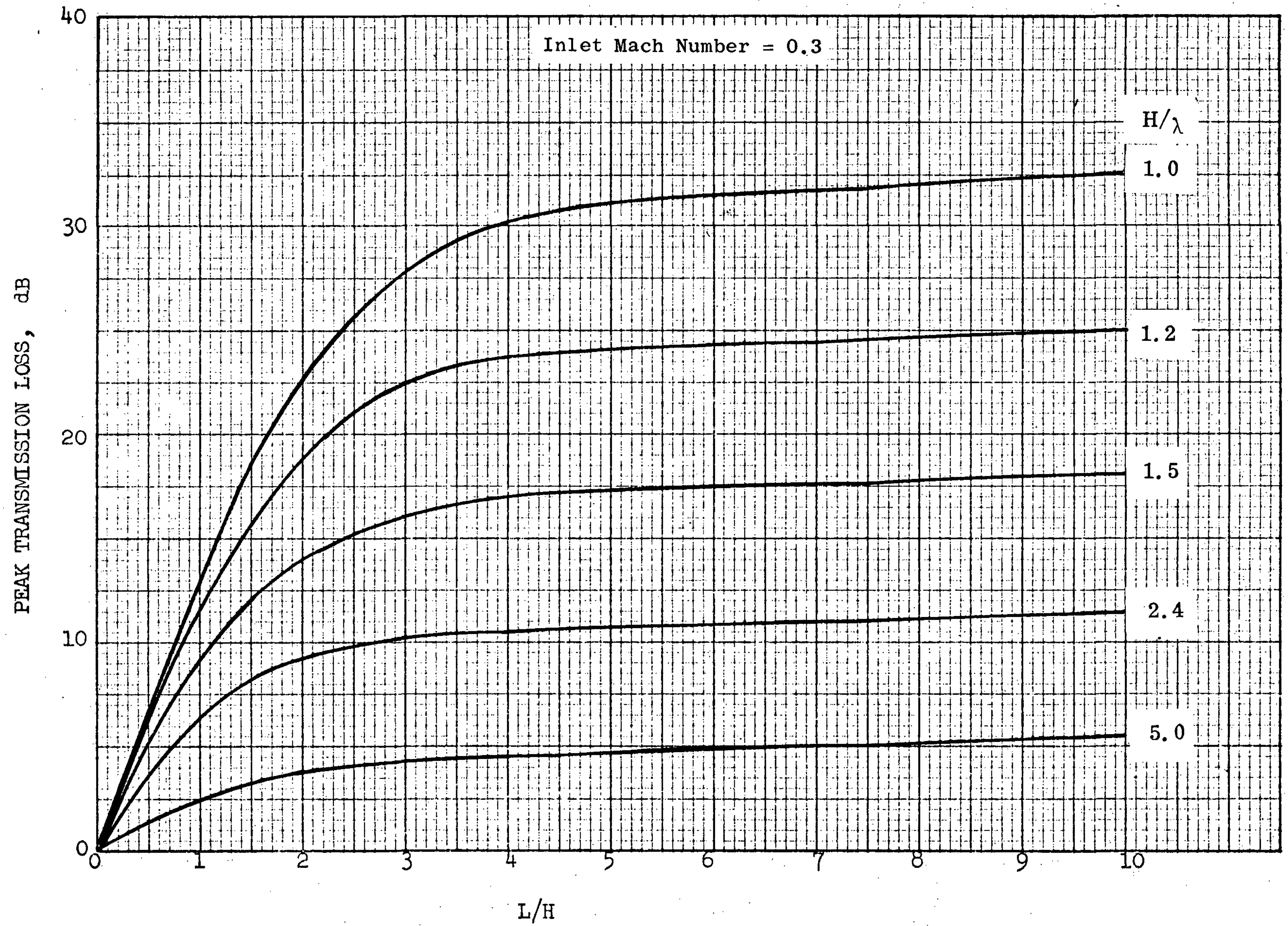

Figure 12. Inlet.

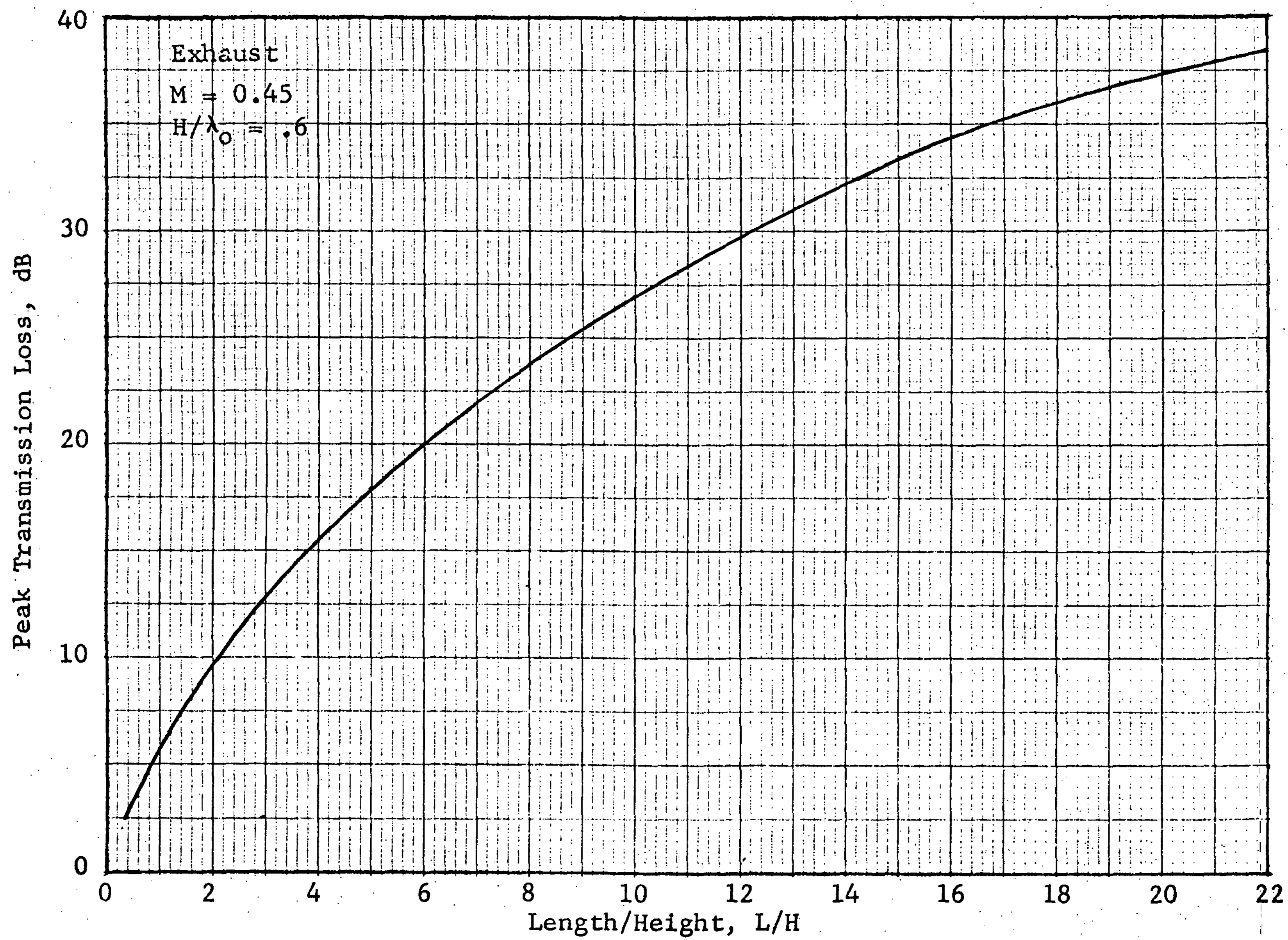

Figure 13. Exhaust.

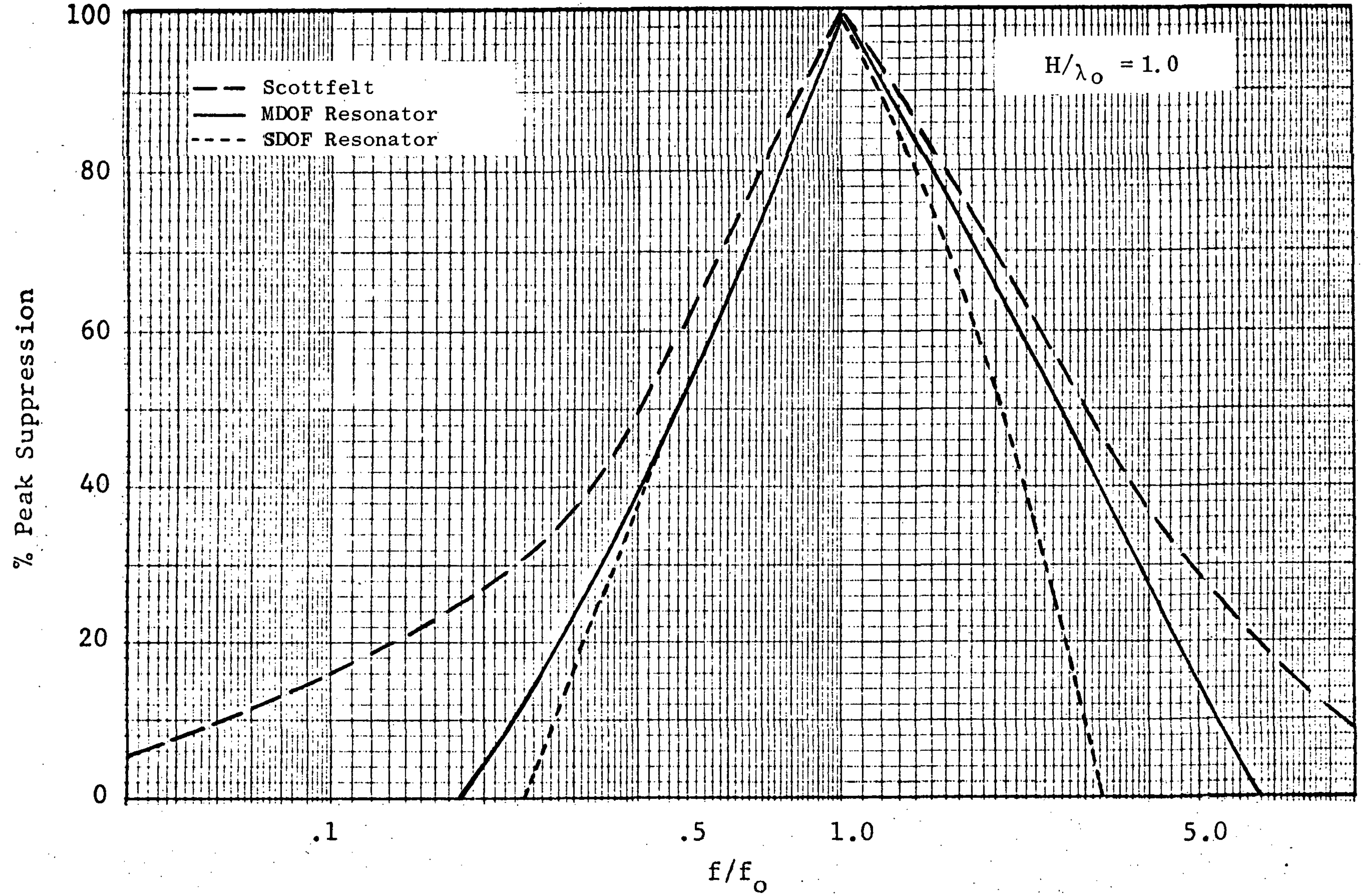

Figure 14. % Bandwidth - Duct Data.

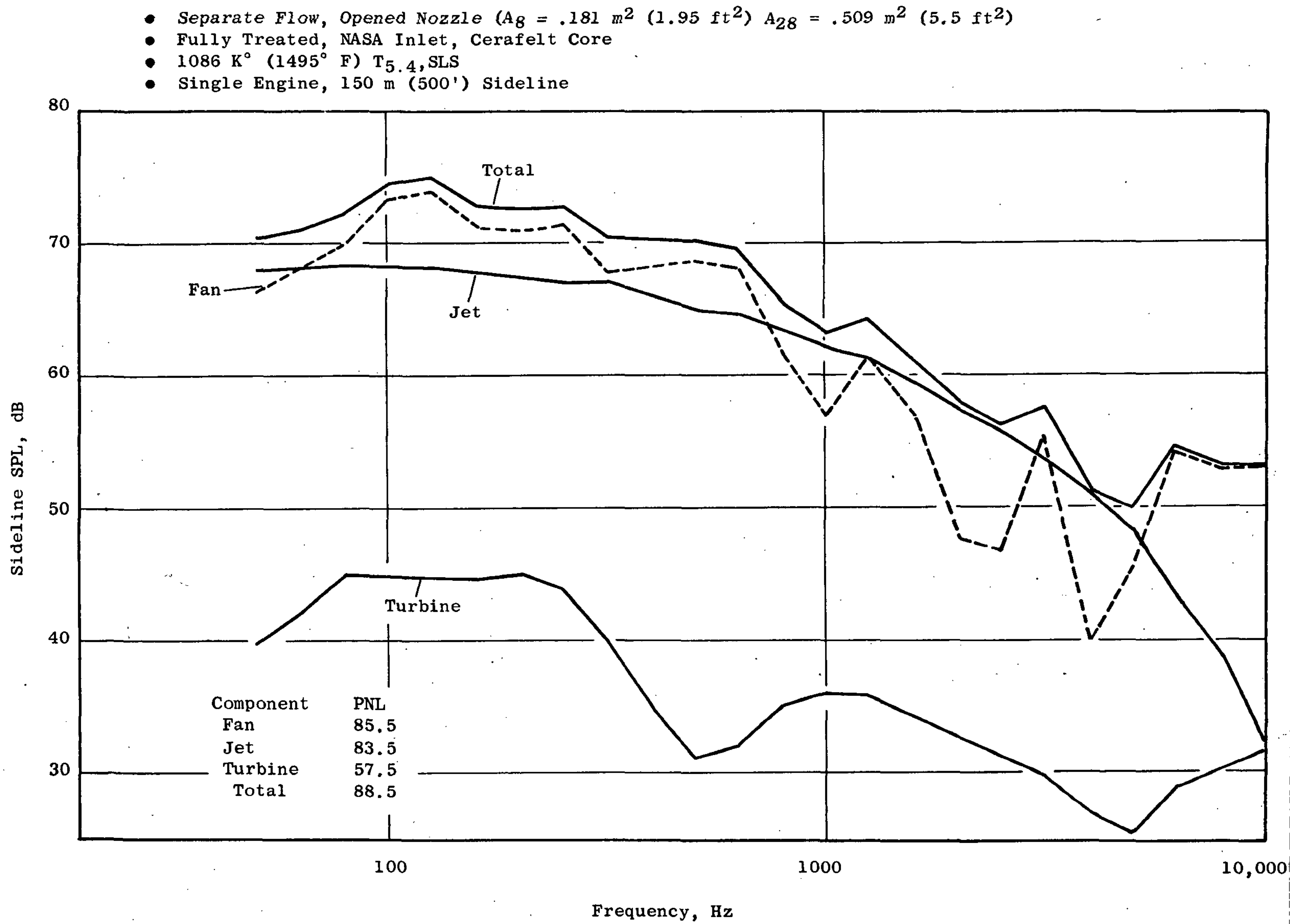

Figure 15. TF34 Quiet Nacelle Estimated Frequency Spectrum at Max. Fwd. Angle (50°) = .87 Rad.

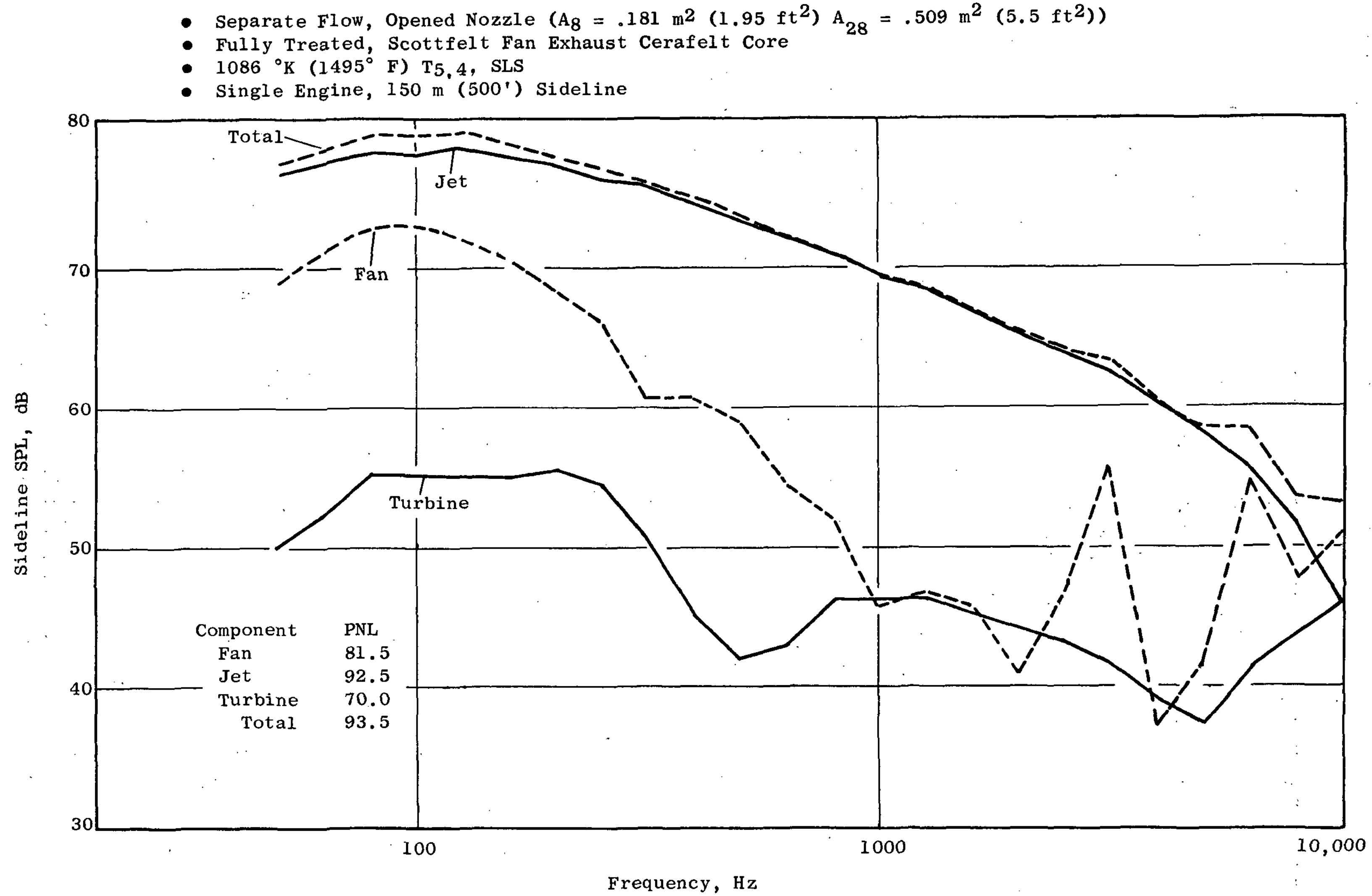

Figure 16. TF34 Quiet Nacelle Estimated Frequency Spectrum at Max. Aft Angle (110°) = 1.9 Rad.

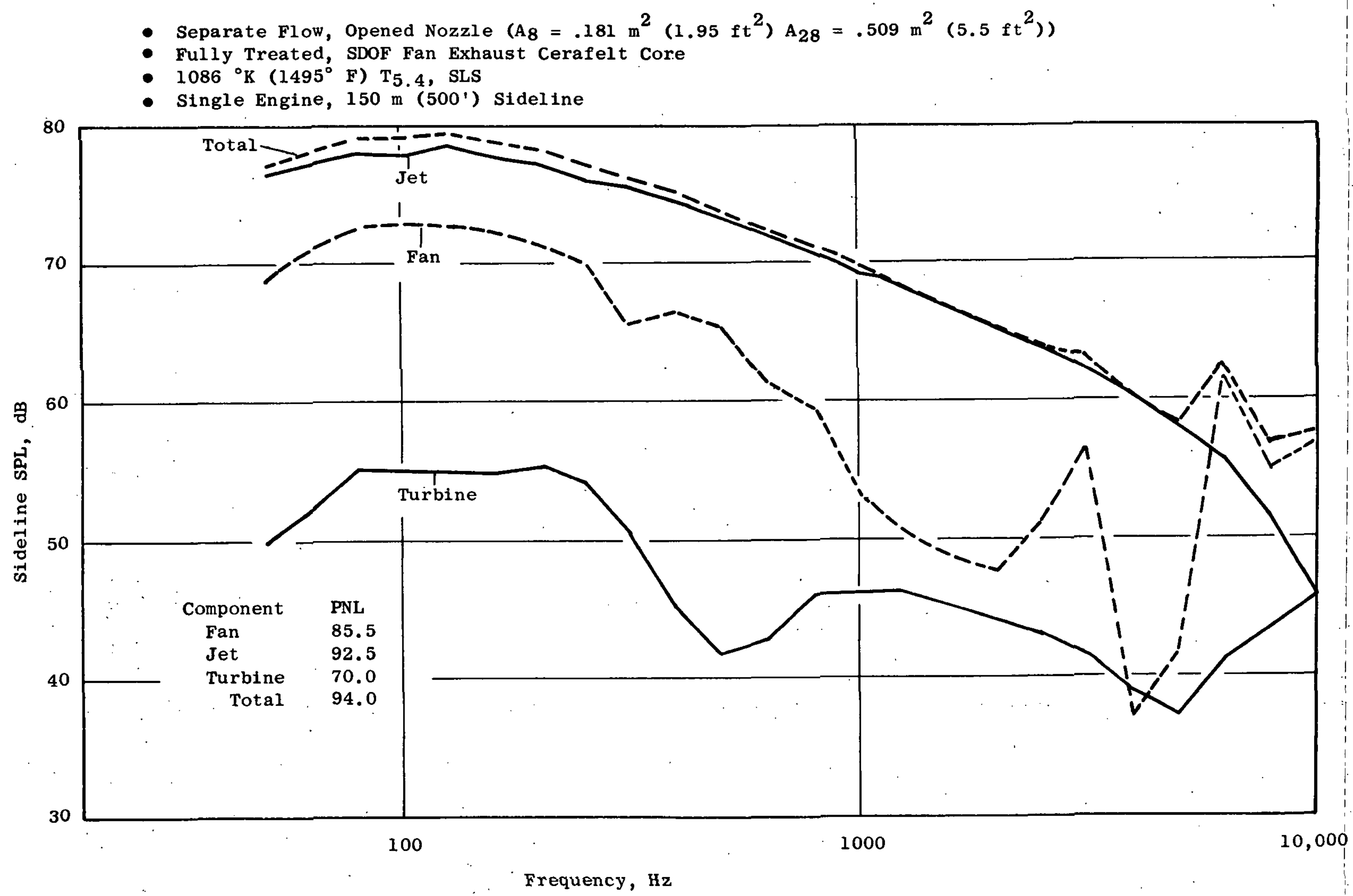

Figure 17. TF34 Quiet Nacelle Estimated Frequency Spectrum at Aft Max. Angle (110°) = 1.9 Rad.

3. AERODYNAMIC DESIGN

Fan Exhaust Duct - The duct is designed for a basic Mach number of 0.45 with the fully thickened splitters in place. These splitters have 12 mm (.45 in.) of extra thickness incorporated. The General Electric Fluxplot computer program for compressible flow was used to design the duct, supplemented by cross-sectional flow area plots versus axial distance, Figure 18 . The original design as determined from area plots resulted in satisfactory flow conditions except for small areas of local high Mach number (~.8) on the outer surface of both splitter leading edges, see Figures 19-20. Incipient separation of the boundary layer was also predicted in this area; Figure 21 shows that the value of the shape factor H exceeded 1.8, the approximate threshold for separation. Modifications to the outer wall and to the splitter leading edge shape (changed to a NACA airfoil contour) resulted in good Mach number and shape factor distributions, as shown also in Figures 19 - 22. The final wall Mach number distribution is shown in Figure 23 . Figure 24 shows the same plot with the splitters completely removed and shows that a Mach number of .26 in the duct can be achieved, with smooth Mach number changes. This Figure indicates that one possible test configuration, i.e., with lowered Mach number for perhaps increased noise attenuation, will have a satisfactory flow situation without any modification to the basic flowpath as defined by the inner and outer walls. The Fluxplot program results can also be used to calculate pressure drops. The splitter pressure drop was based on a skin friction coefficient of 0.005 which is 50% higher than for a flat plate to allow for the roughness of the perforated surface of the acoustic treatment. The wall friction was similarly calculated using a boundary layer computer program.

The net scrubbed areas used for these calculations are:

Outer Splitter	m^2 (ft^2)	11.38 (120)
Inner Splitter	m^2 (ft^2)	8.58 (93)

The resulting pressure drops are as follows:

		$\Delta P/P_T$
6 "A" Frame Struts*		.00130
12 o'clock pylon skin friction		.00083
6 o'clock pylon skin friction		.00067
Outer splitter skin friction		.01216
Inner splitter skin friction		.00912
Duct wall friction		.01584
Pylon splitter interference		.00291
Goose neck diffusion loss		.01013
	Total	.05796

* Standard TF34 cycle deck loss

Inlet - The inlet with bellmouth and sound suppressors per NASA CR 501945, was aerodynamically evaluated for pressure losses. A potential flow solution for an untreated inlet revealed that the outermost splitter as designed experienced a maximum deviation of the trailing edge contour from the flow field of .279 radians (16°). This angularity was determined to present a risk of separation if local conditions varied, and a revised contour as shown in Figure 25 was proposed with a maximum deviation of .185 radians (10.6°). The major contributor to the pressure loss is the large scrubbed surface area of acoustic treatment, for which a skin coefficient of .005 was assumed.

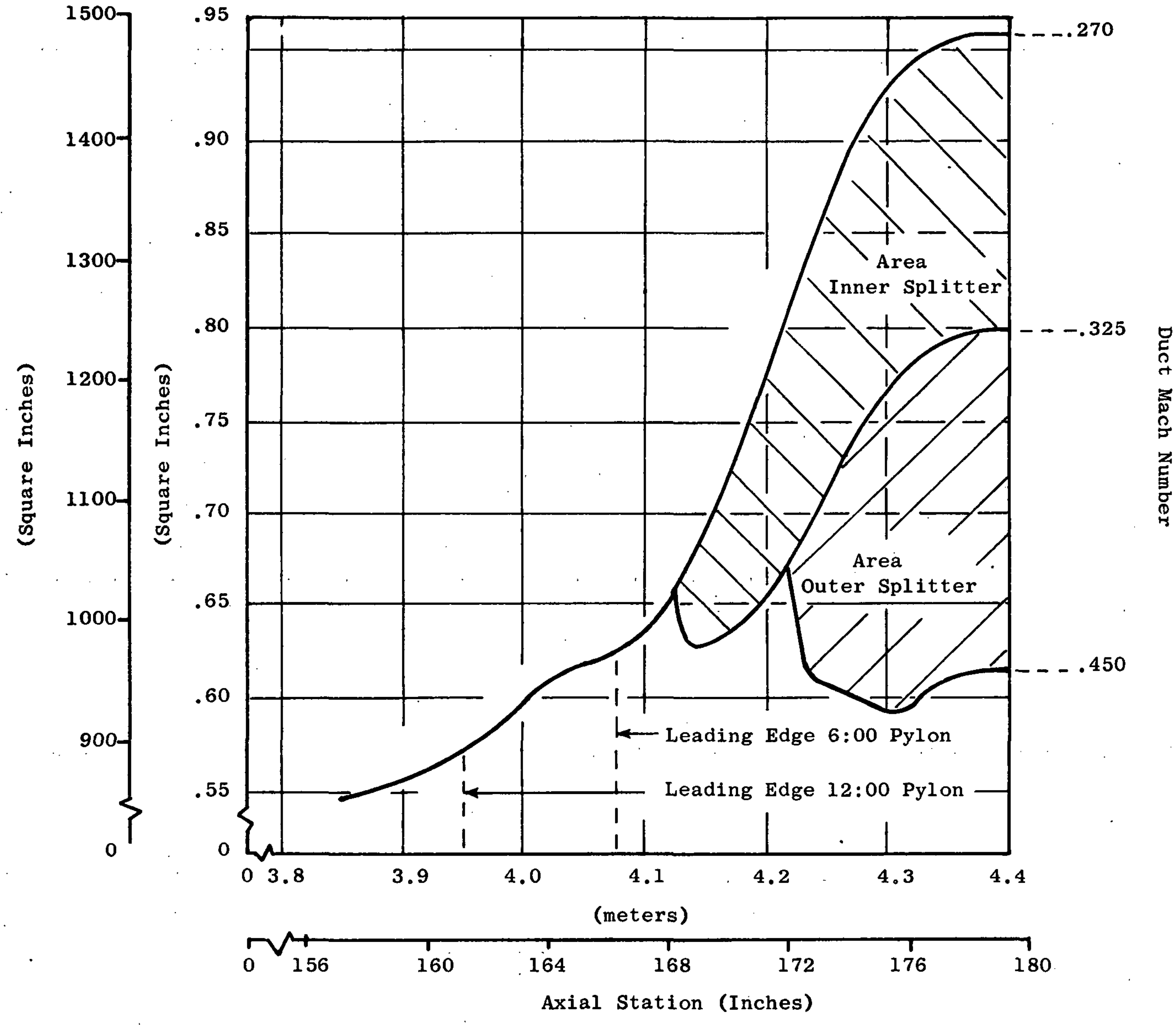

Figure 18. TF34 Quite Engine Fan Duct Area Distribution.

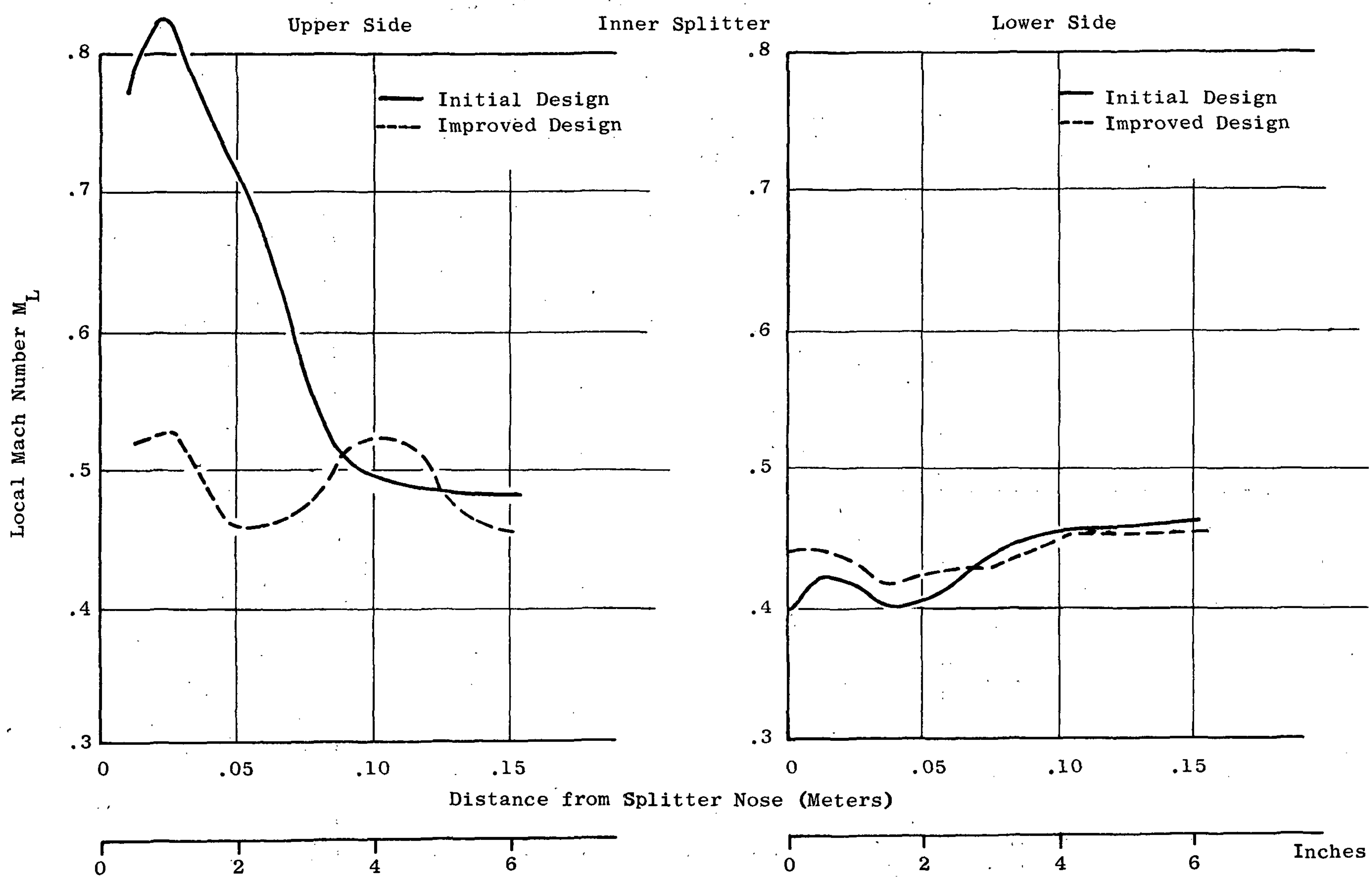

Figure 19. TF34/NASA Quiet Engine Comparison of Improved Splitter Nose.

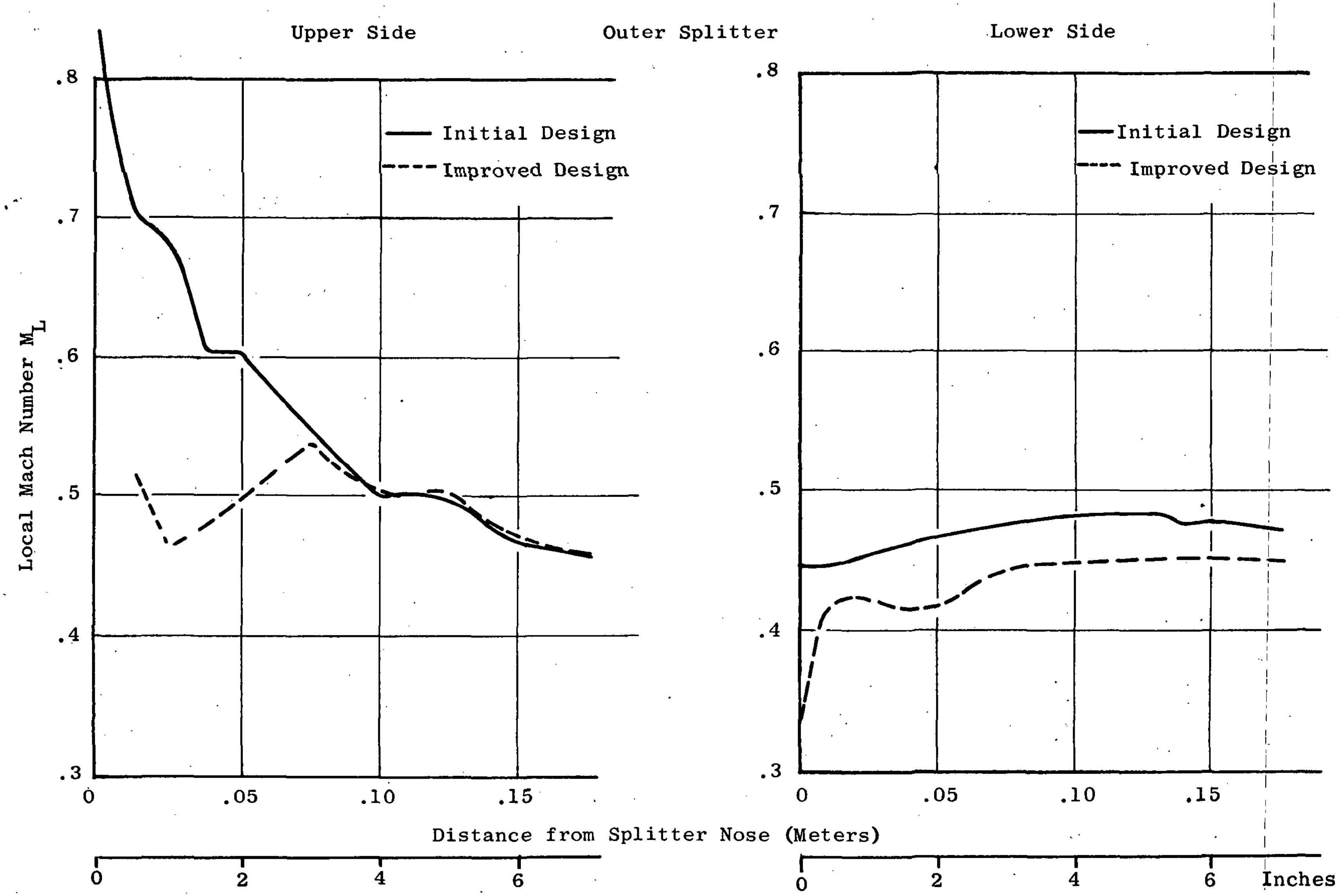

Figure 20. TF34/NASA Quiet Engine Comparison of Improved Splitter Nose.

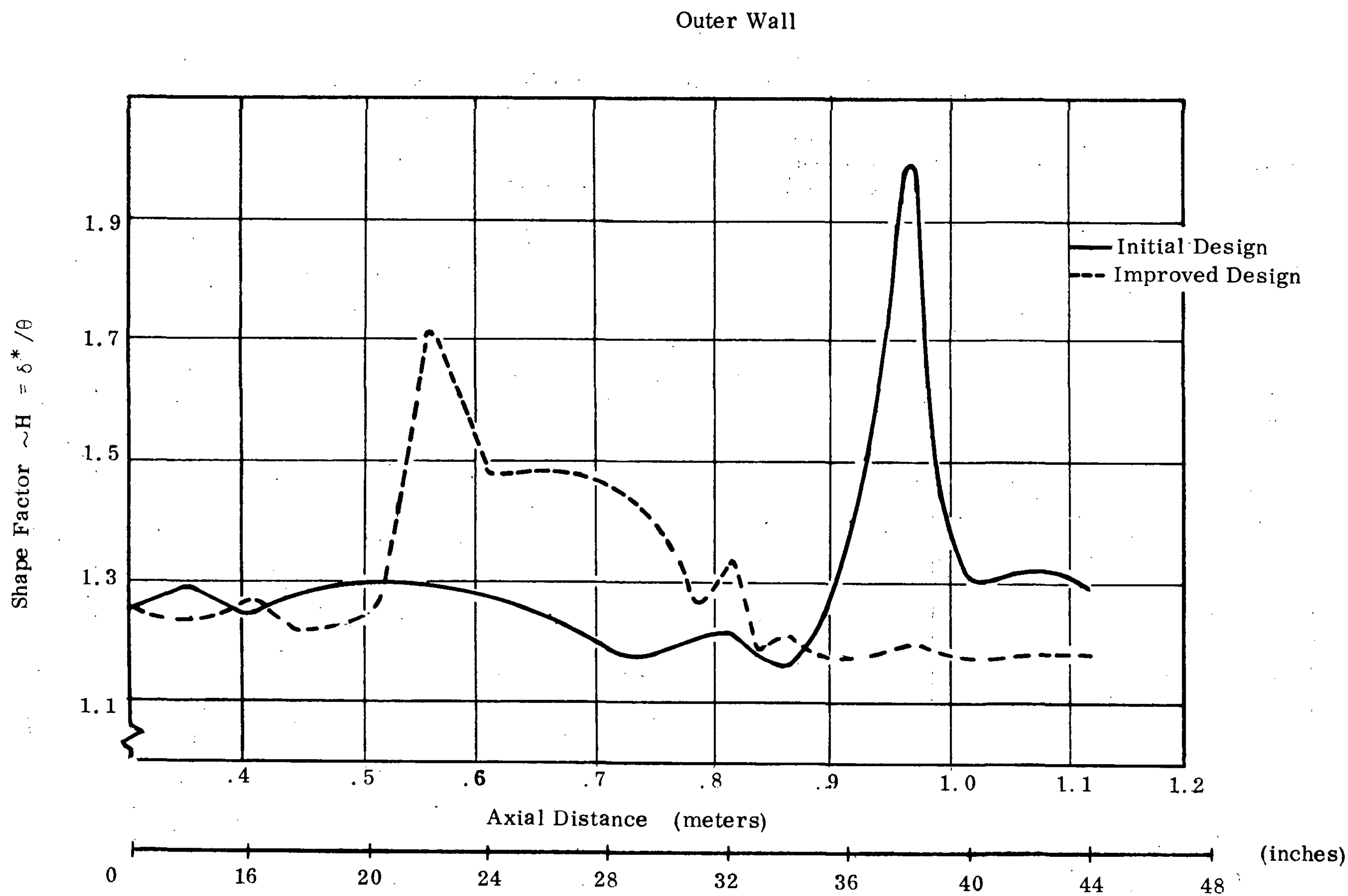

Figure 21. TF34/NASA Quiet Engine Turbulent Boundary Layer Growth with Skin Friction by Rotta Method.

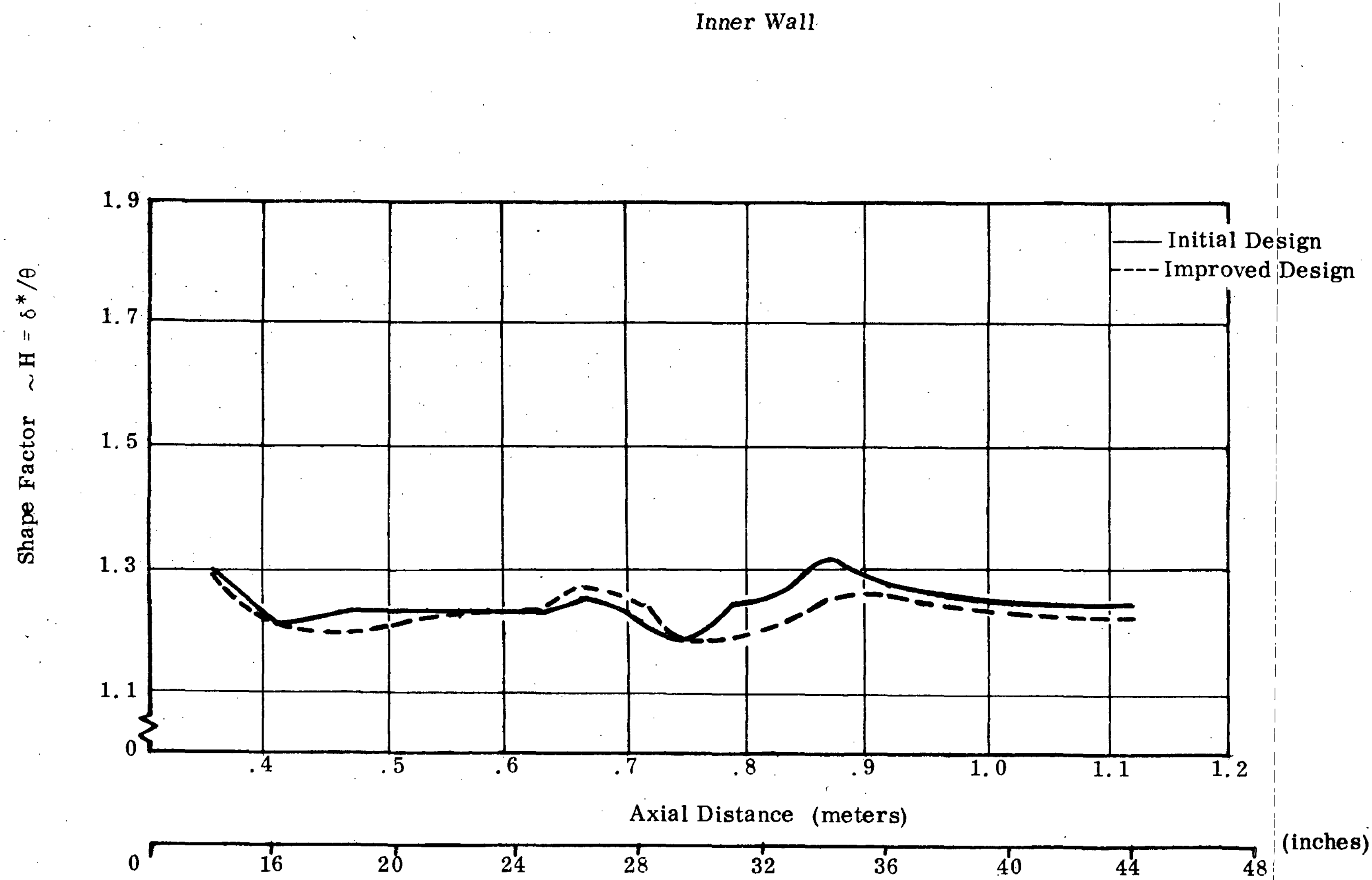

Figure 22. TF34/NASA Quiet Engine Turbulent Boundary Layer Growth with Skin Friction by Rotta Method.

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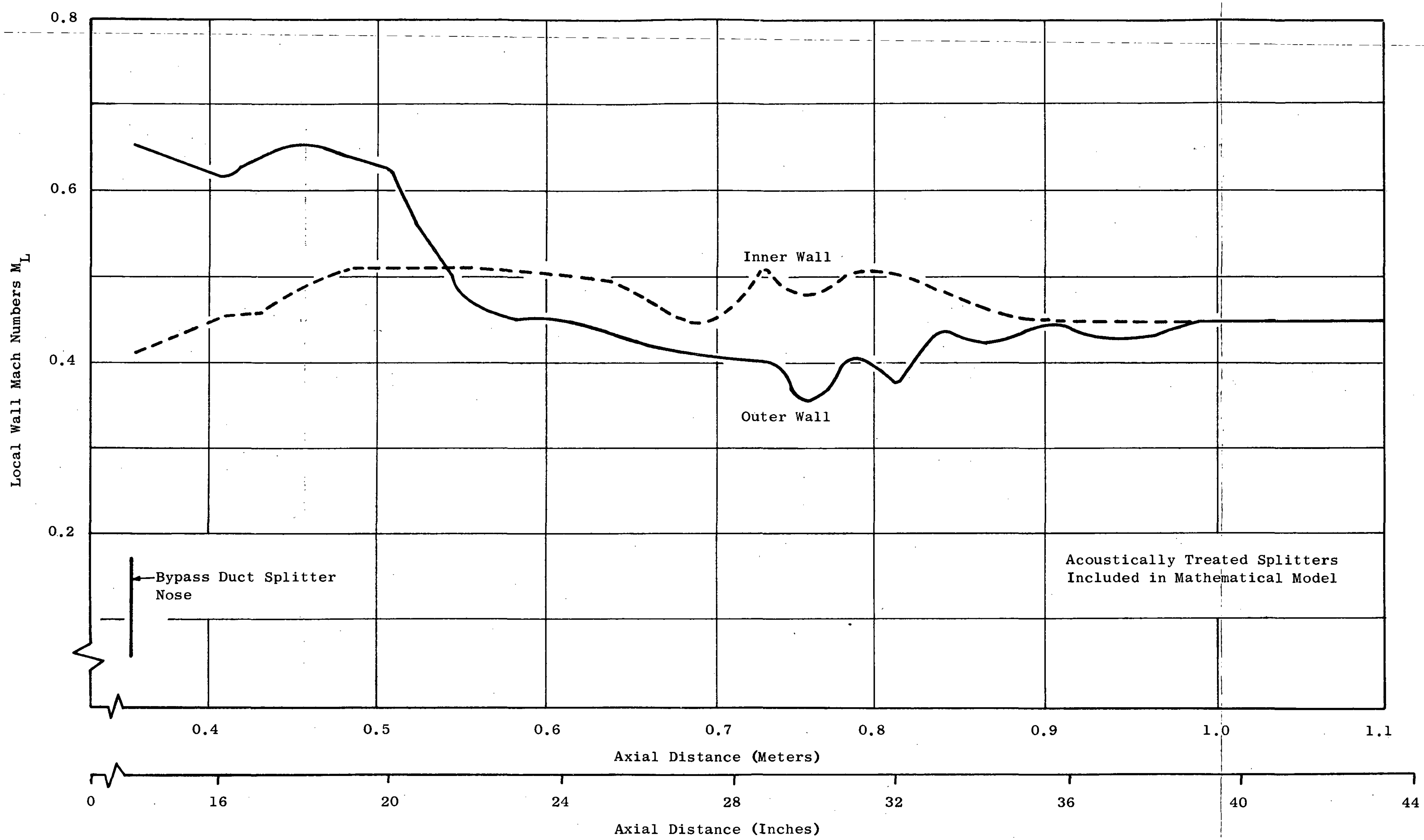

Figure 23. TF34/NASA Quiet Engine Compressible Fluid Flow Fluxplot Solution Local Wall Mach Numbers.

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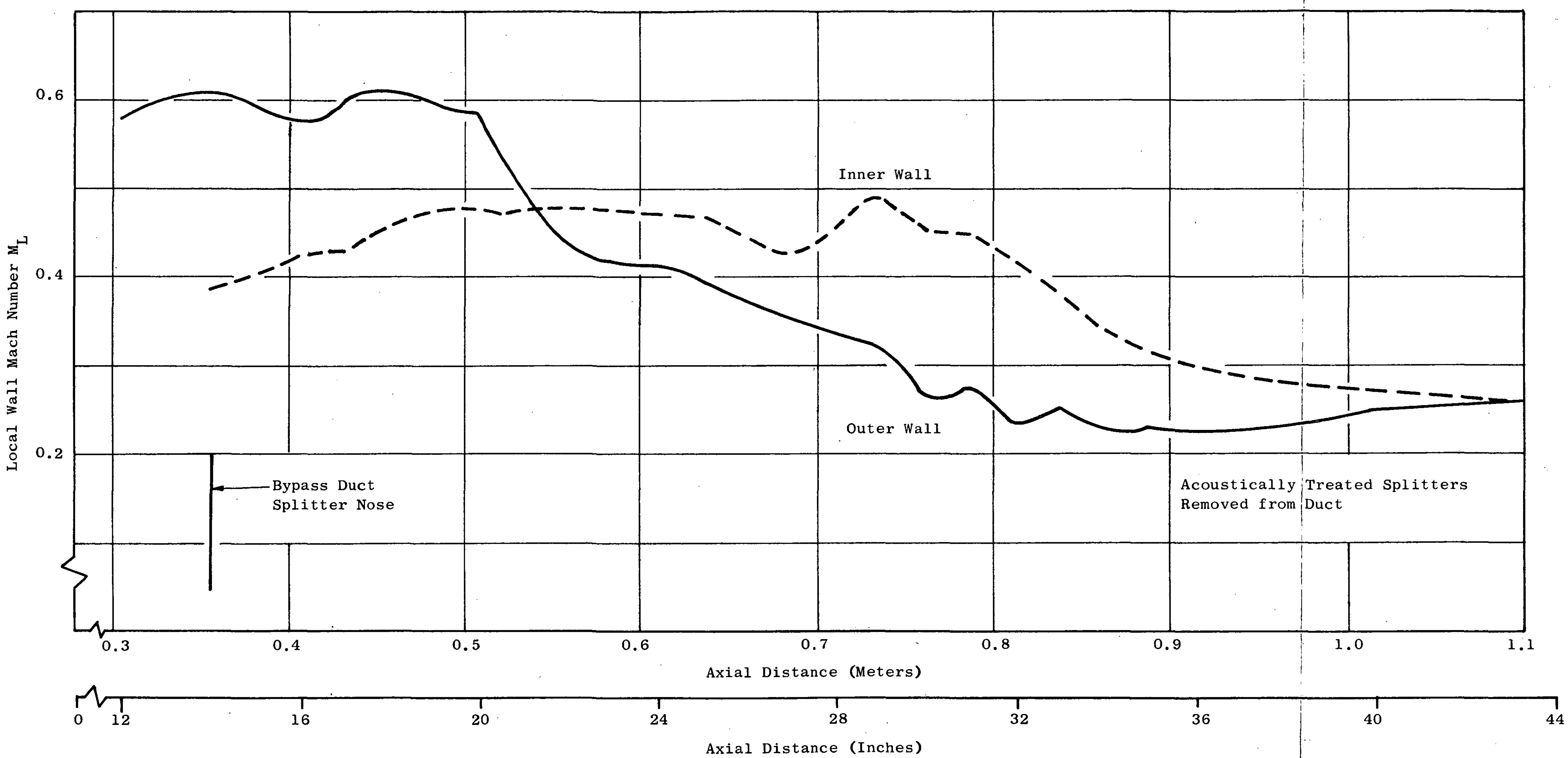

Figure 24. TF34/NASA Quiet Engine Compressible Fluid Flow Fluxplot Solution Local Wall Mach Numbers.

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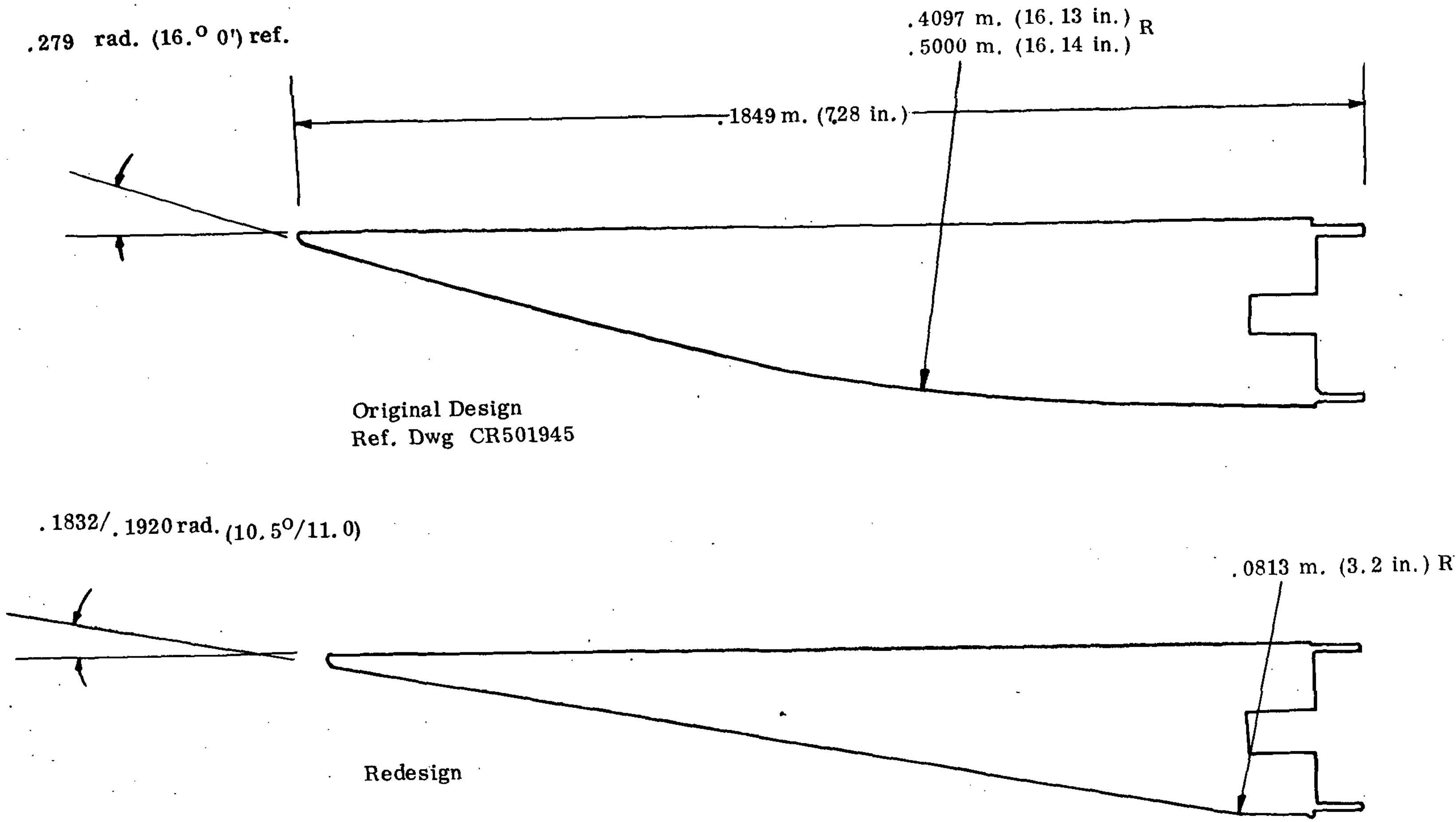


Figure 25. Redesigned Trailing Edge for Outer Splitter of the Fan Inlet Acoustic Suppressor. (CR 501945)

The resulting pressure drops are as follows:

	$\Delta P/P_{TO}$
Bellmouth and Wall Friction	.0038
4 Radial Struts	.00067
Interference Effects of Struts and Splitters	.00019
Outer Splitter Skin Friction	.0053
Middle Splitter Skin Friction	.0035
Inner Splitter Skin Friction	.00145
Diffusion Through Splitters	.0100 (Aft-Most Splitter Position)
Total	.02491

The flow distribution and average passage Mach number, resulting from the inlet splitters is as follows:

	Airflow/Channel / Total Airflow	Area Channel / Total Area	Mach Number
Channel 1	.349	.352	.576
Channel 2	.274	.274	.581
Channel 3	.197	.195	.584
Channel 4	.180	.179	.585

Note: Channel designation assumed that 1 is at outermost radius.

The inlet distortion encountered by the fan due to strut and splitter wakes was calculated to be N_{DI} = 1.083%. This distortion is not considered excessive as a clean TF34 inlet runs at N_{DI} = .5%. (N_{DI} is a distortion index taking account of strength and distribution).

Core Exhaust Duct - The core duct total pressure decrement was obtained by calculating a local velocity ratio on the inner and outer wall, entering these ratios into a boundary layer program and obtaining the boundary layer loss. The predicted core exhaust duct scrubbing drag is 9.12% $\Delta P_T/q5.5$. See Figure 33 for Station designations.

4. PERFORMANCE

In order to assess the penalties imposed on the engine by the noise suppression losses, the $\Delta P/P$ defined in Section 3 have to be converted into loss parameters which can accommodate flow variations. Since these losses are mainly due to friction, the magnitude is proportional to flow velocity, i.e. dynamic pressure, at a given reference station. The losses were calculated assuming flows and Mach numbers which will change as the cycle accommodates the level of loss. Therefore, for the performance calculations, these $\Delta P/P's$ were translated into $\Delta P/q$ losses where q is the dynamic pressure at a reference flow area.

The core exhaust is treated as a lumped system with all losses referenced to dynamic pressure at turbine discharge. Standard TF34 cycles include 12.27% $\Delta P/q_{55}$ between turbine exit and the turbine exhaust frame aft flange. The losses used for performance calculations are as follows:

Inlet	$\Delta P/q1$ = 20.09%	A_1 ref =	.981 m^2 (1520 in^2)
Fan Duct	$\Delta P/q_{24}$ = 46.70%	A_{24} ref =	.613 m^2 (950 in^2)
Core Duct	$\Delta P/q5.5$ = 22.39%	$A5.5$ ref =	.178 m^2 (275.3 in^2)

In addition, friction losses are assumed for the chutes of the internal mixer in the mixed flow systems. This loss for the cold chute (fan flow mixer) is $\Delta P/q_{26} = 6.5\%$. Station 26, cold chute inlet, is defined as that area required to provide a Mach number of .45 at the sizing condition. Station 27, or cold chute exit, is sized to provide a .52 Mach number after $\Delta P/q_{26}$ is applied to the flow. The loss for the hot chute (core flow mixer) is estimated at 6.0% $\Delta P/q5.5$. The total core loss between Station 5.5, turbine exit, and Station 7.7, hot chute exit, is $\Delta P/q5.5 = 28.39\%$. Station 7.7 is sized by reducing the Station 5.5 flow area by 10%. A velocity decayer, when included is debited for a pressure loss of 6.2% $\Delta P/Q7.8$, where Station 7.8 is an area equal to the sum of A7.7 and A27.

- Separated Flow - Performance data for the separated flow engines was run with minimum and maximum fan and core nozzle areas allowed by current hardware design. The core nozzle areas, A_8, range from .116 to .181 m^2 (180 to 281 in^2), the fan nozzle areas, A_{28}, from .406 to .509 m^2 (630 to 790 in^2). The engines were run to the maximum (5 minute rating) fan turbine rotor inlet temperature ($T_{5.4}$) with combinations of above areas and installation losses at sea level static, standard conditions. This temperature, 1086 ^oK (1495^oF), determines the maximum available thrust at any given ambient condition. The fan operating points were checked for successful operation, i.e., no excessive reduction in stall margin, and an engine operating line was run at 670.6m (220 ft) altitude, 266.7^oK (20^oF) temperature representative of test ambient conditions at Edwards Flight Test Center. The fan operating points are shown in Figures 26 and 27. Table IV shows the operating conditions of the four cycles. Table V indicates nominal TF34 operating performance. Figure 28 shows the relationship between exhaust velocities and engine thrust at the altitude condition. As is shown in Figures 26 and 27, use of the smallest fan nozzle area does cause a reduction in fan stall margin, but as these small areas also cause increases in fan jet velocity, their usefulness is diminished. The lowest jet noise configuration would be with maximum A_8 and A_{28}. Figures 26 and 27 show that nominal fan operating line performance is achievable within the range of hardware variation.

- Mixed Flow - Mixed flow cycle performance data was also computed. The mixed flow engines were sized to run at a fan tip pressure ratio, P_{23}/P_2 of 1.4888 at maximum T_5. Since AE77 is geometrically determined, the mechanism for setting fan pressure ratio is by adjustment of cold chute areas. When the proper fan operating line has been set, the cold and hot flows mix, decayer losses if present are applied, and nozzle exit area, A_8, is then determined to satisfy continuity at the exit plane. The fan operating points are shown on Figure 29 and this choice of pressure ratio results in satisfactory fan operation. Table VI indicates the important operating conditions at maximum thrust at sea level static and 670.6m (2200 ft), 266.7 ^oK (20 ^oF) Figure 30 shows the relationship between exhaust velocity and engine thrust for both conical and decayer nozzles.

Since the level of the fan operating line is determined by the cold chute area, and nozzle exhaust area is determined by continuity, flexibility must be designed into the hardware to allow setting of discrete operating conditions. Opening the areas depresses the fan operating line resulting in reduced pressure ratio, increased engine airflow, and reduced jet velocities. The absolute amount of opening must be relatively small under the constraint of fixed geometry aircraft operation due to unsatisfactory fan operation at altitude cruise conditions, where the fan operating line drops so low as to be unpredictable.

The hardware as designed for mixed flow operation has the capability of trim to allow setting of specific operating conditions. The procedures required for proper trimming are described in Table IV. The maximum areas achievable with the current hardware is as follows:

	Conical Nozzle		Decayer Nozzle	
A_{26} max m^2 (in.^{2})	.685	(1061)	.625	(1061)
A_{27} max m^2 (in.^{2})	.621	(962)	.621	(967)
A_8 max m^2 (in.^{2})	.714	(1107)	.721	(1117)

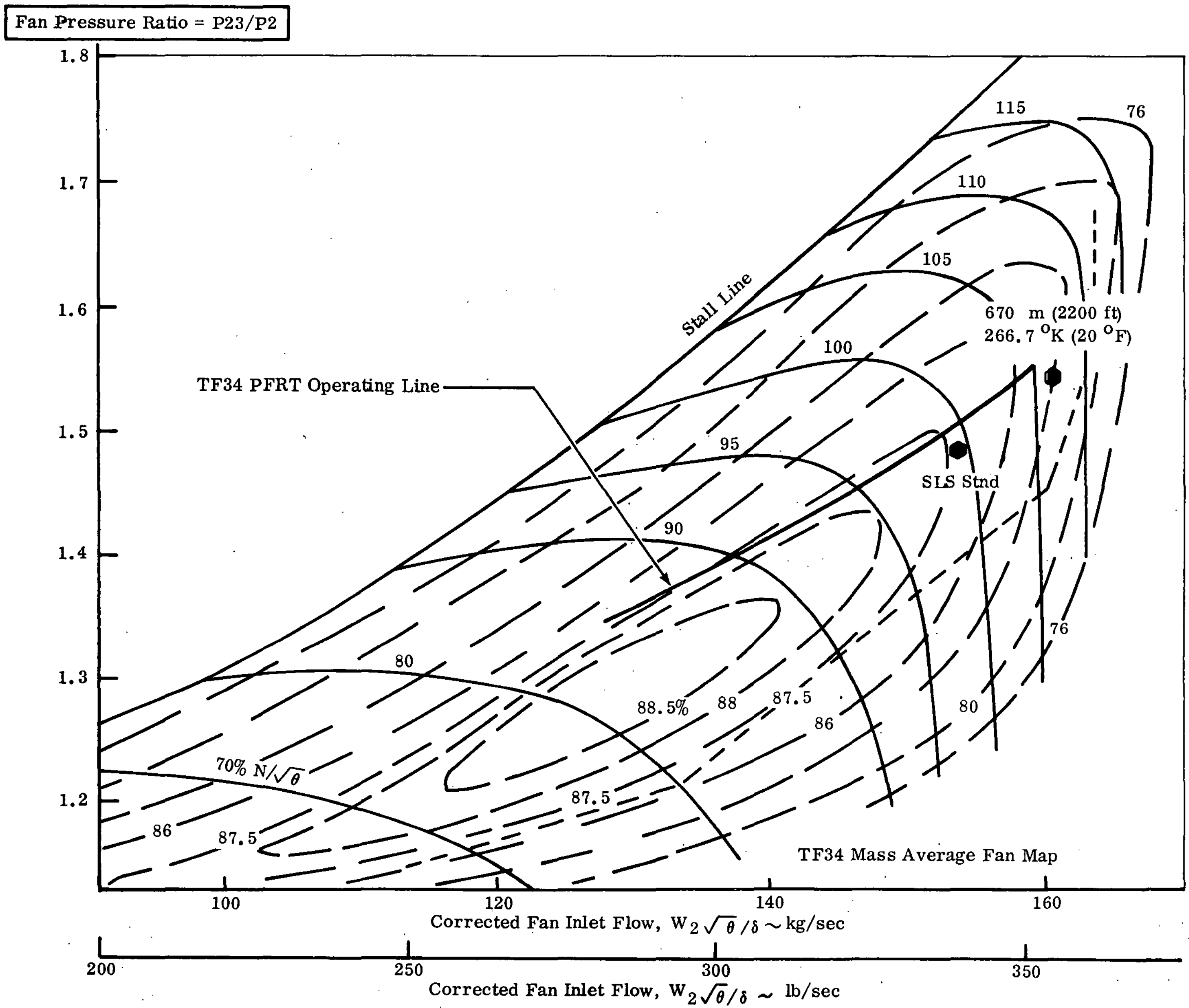


Figure 26. TF34 Quiet Engine Mixed Flow Cycle Fan Operating Points 1086 °K (1495° F) $T_{5.4}$.

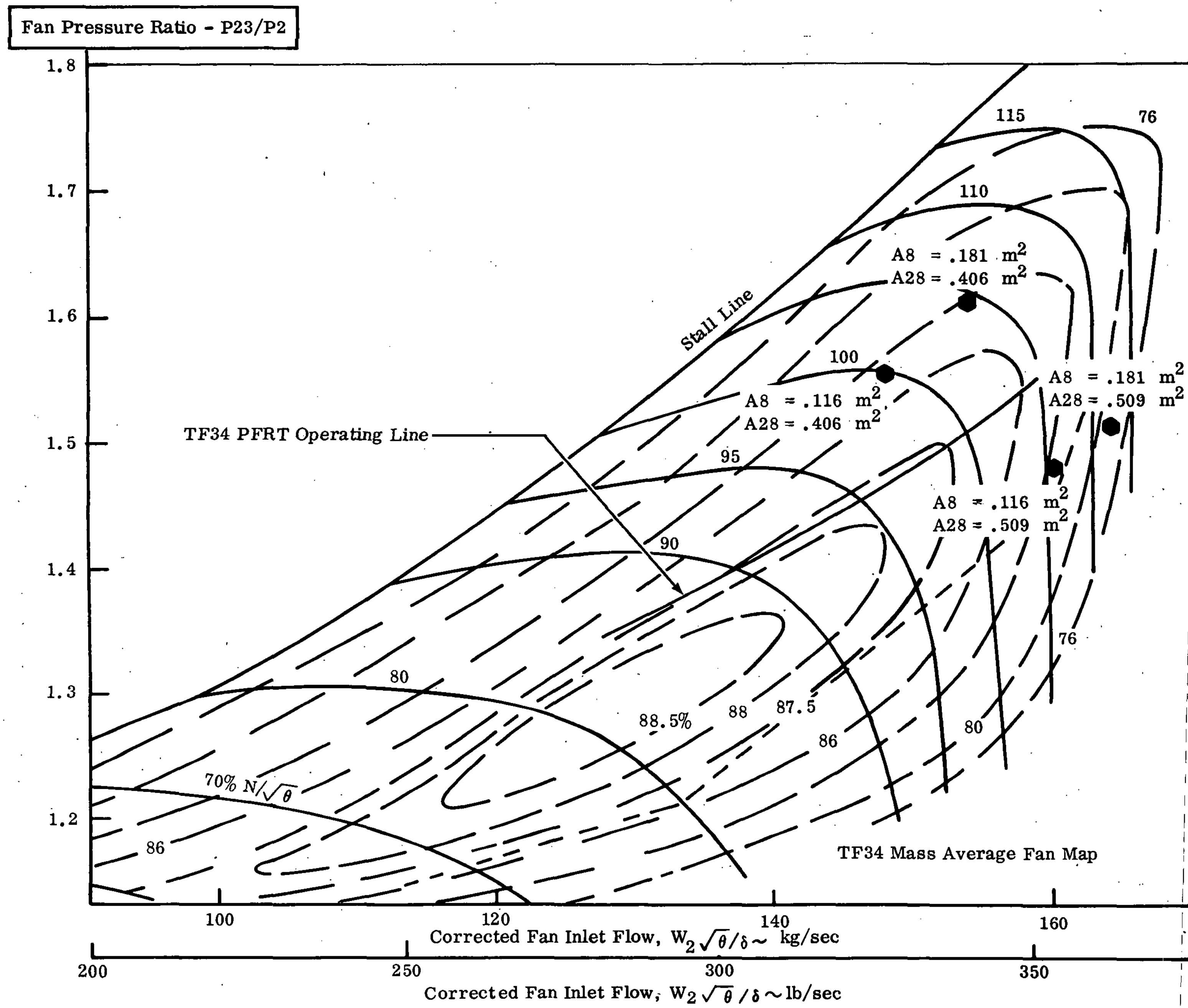
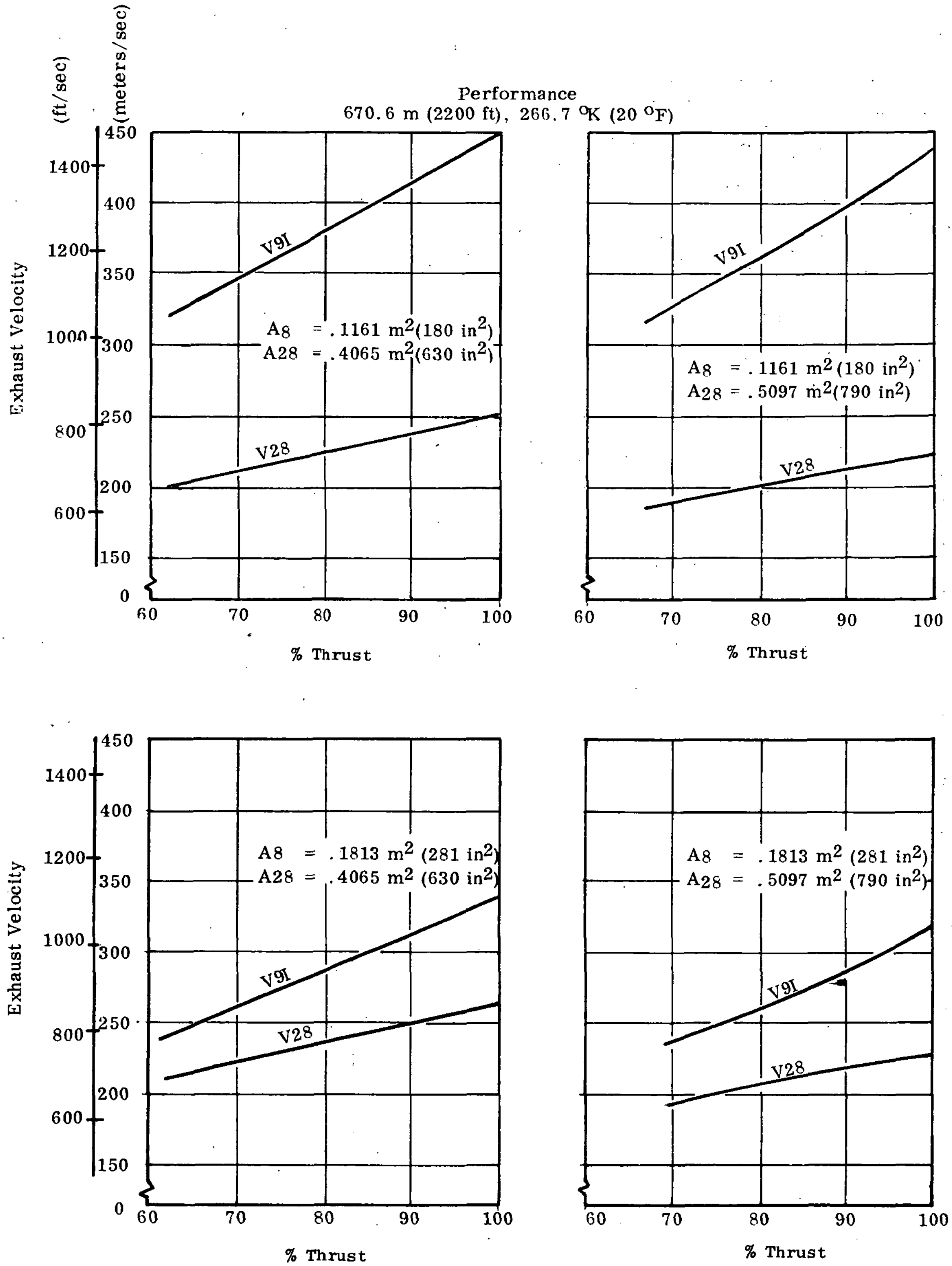


Figure 27. TF34 Quiet Engine Fan Operating Points, Separate Flow 670 m (2200 ft), 266.7°K (20° F) 1086 °K(1495° F) $T_{5.4}$.



NOTE: 100% Thrust Level Determined at 1086 °K (1495°F) $T_{5.4}$

Figure 28. TF34 Quiet Engine - Separated Flow.

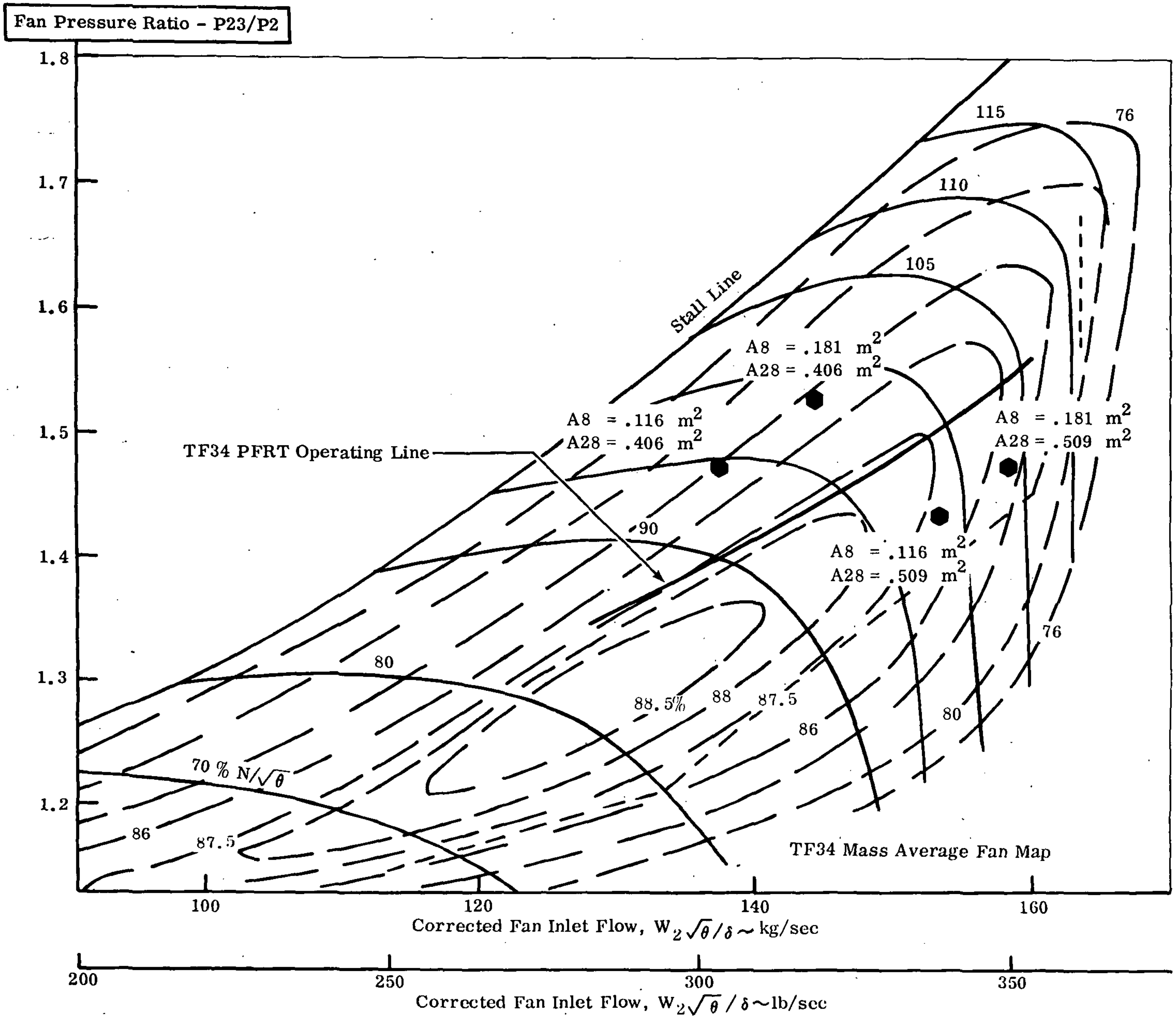


Figure 29. TF34 Quiet Engine Match Points Separate Flow Sea Level Static, Standard Day 1086 °K(1495° F) $T_{5.4}$.

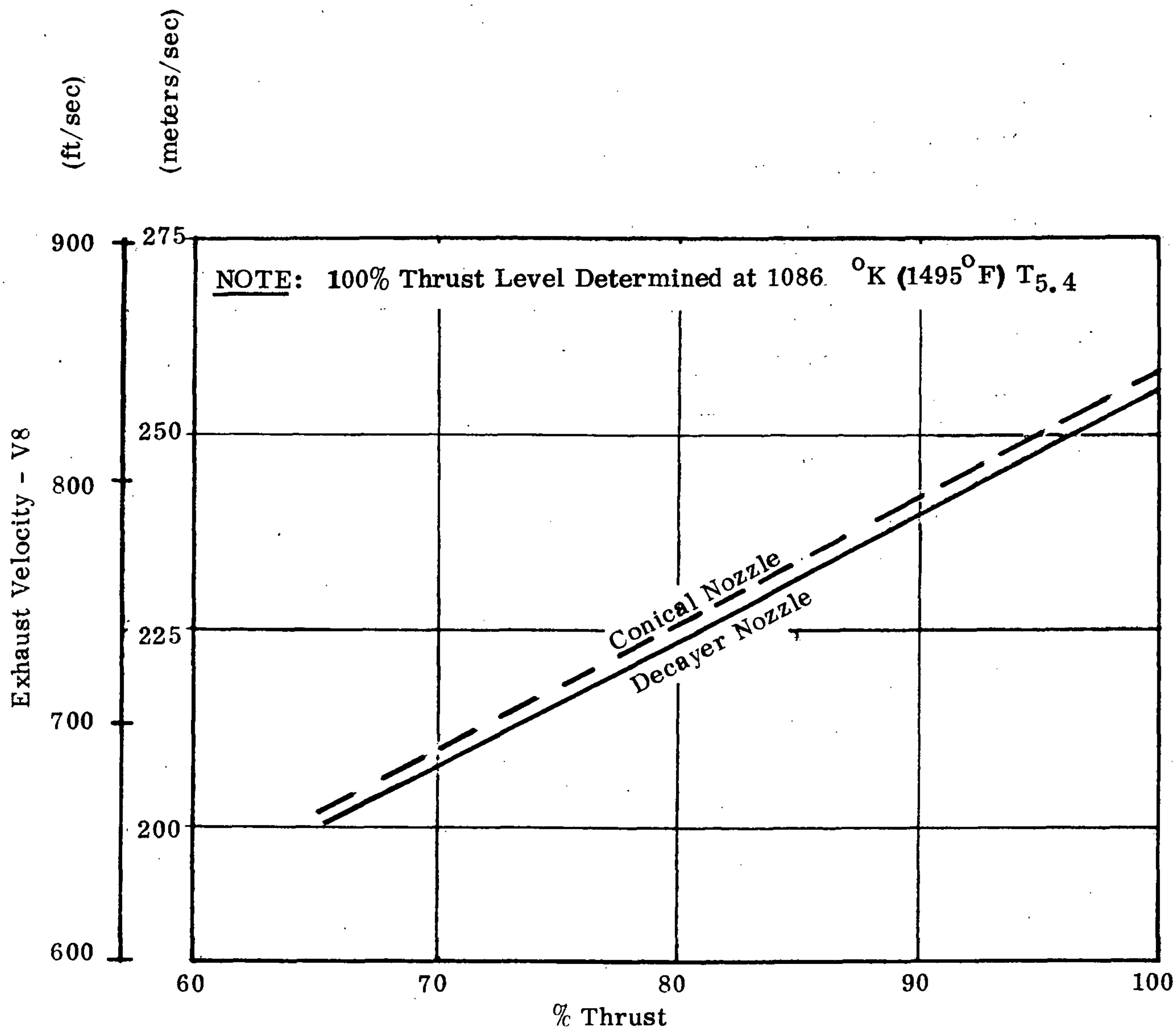


Figure 30.TF34 Quiet Engine - Mixed Flow 670.6 m (2200 ft), 266.7 °K (20° F).

Table IV - TF34 - Quiet Engine Performance - Separated Flow

1086 °K (1495°F) $T_{5.4}$

Sea Level Static - Standard Day

Core Nozzle Exhaust Area	A8	m^2	(in^2)	.116 (180)	.116 (180)	.181 (281)	.18 281
Fan Nozzle Exhaust Area	A28	m^2	(in^2)	.406 (630)	.51 (790)	.406 (630)	.51 (790)
Thrust Fn		N	(lb)	34545 (7766)	34972 (7862)	35768 (8041)	35568 (7996)
Specific Fuel Consumption SFC		Kg/N-sec	(lb/hr-lb)	.0422 (.414)	.0411 (.403)	.0413 (.405)	.0408 (.400)
Inlet Pressure Loss	$\Delta P/p$	%		1.61	2.02	1.78	2.19
Fan Exhaust Pressure Loss	$\Delta P/p$	%		3.80	5.35	3.99	5.60
Core Exhaust Pressure Loss	$\Delta P/p$	%		1.68	1.68	2.35	2.31
Inlet Airflow	W2	Kg/sec	(lb/sec)	134 (295.7)	149 (328.7)	141 (310.1)	155 (341.2)
Fan Rotational Speed	N_F	%		94.8	98.3	98.4	103.5
Fan Pressure Ratio	P23/P2			1.481	1.434	1.540	1.475
Fan Exhaust Velocity	V28	m/sec	(ft/sec)	242 (795.0)	220 (720.8)	255 (838.4)	230 (753.9)
Core Isentropic Exhaust Velocity	V9I	m/sec	(ft/sec)	418 (1372.5)	410 (1346.8)	315 (1034.7)	303 (995.6)
Bypass Ratio BPR				5.78	6.67	6.01	6.86

NOTE: 100% Fan Speed = 733 rad/sec (7000 RPM)

1086 °K (1495°F) $T_{5.4}$

670.56 m (2200 ft) Altitude , 266.67 °K (20 °F) Temperature

Thrust Fn		N	(lb)	37276 (8380	36364 (8175)	38388 (8630)	36057 (8106)
Specific Fuel Consumption SFC		Kg/N -sec	(lb/hr-lb)	.0415 (.407)	.0416 (.408)	.0407 (.399)	.0417 (.409)
Fan Rotational Speed	N_F	%		96.1	101.8	100.8	108.1
Fan Pressure Ratio	P23/P2			1.57	1.49	1.63	1.52
Fan Exhaust Velocity	V28	m/sec	(ft/sec)	251 (822.9)	223 (732.6)	266 (873.3)	231 (759.3)
Core Isentropic Exhaust Velocity	V9I	m/sec	(ft/sec)	451 (1478.8)	439 (1440.5)	340 (1115.9)	322 (1055.3)

Exhaust nozzle areas will be changed during the Quiet Nacelle program to provide for different jet velocities and for the variation in duct losses of the suppressors. This is achieved by cutting back the edges of the conical convergent nozzles to increase the areas. Two nozzles for the core and fan flow are to be built. One set of core and fan nozzles will be sized at a minimum area to provide a standard for jet velocity noise. The other set of nozzles will be trimmed to the maximum possible area trimmed to obtain minimum jet velocities. The range of areas is defined in NASA LeRC Drawing CF 501939.

Table V
Nominal TF34 Operating Condition
Sea Level Static Standard Day
1086 °K (1495°F) $T_{5.4}$

Core Nozzle Exhaust Area	A8	m^2 (in^2)	.1347 (208.8)
Fan Nozzle Exhaust Area	A28	m^2 (in^2)	.4224 (654.7)
Thrust Fn	N	(lb)	40901 (9195)
Specific Fuel Consumption		Kg/N-sec (lb/hr-lb)	.0368 (.361)
Inlet Pressure Loss	$\Delta P/p$	%	0
Fan Exhaust Pressure Loss	$\Delta P/p$	%	0.77
Core Exhaust Pressure Loss	$\Delta P/P$	%	1.78
Inlet Airflow	W_2	kg/sec (lb/hr-lb)	152.6 (336.4)
Fan Rotational Speed	N_F	%	98.6
Fan Pressure Ratio		P23/P2	1.488
Fan Exhaust Velocity V28		m/sec (ft/sec)	262.9 (862.6)
Core Isentropic Exhaust Velocity		m/sec (ft/sec)	385.0 (1263.1)
Bypass Ratio	BPR		6.47

NOTE: 100% Fan Speed = 733 rad/sec (7000 rpm)

TABLE VI - TF34 QUIET ENGINE
Mixed Flow Cycle Performance

1086 °K (1495°F) $T_{5.4}$

			Conical Nozzle SLS	Conical Nozzle EFTC[1]	Decayer Nozzle SLS	Decayer Nozzle EFTC[1]
Cold Chute Inlet Flow Area	AE26	m^2 (in^2)	.6097 (945.0)	→	→	→
Cold Chute Exit Flow Area	AE27	m^2 (in^2)	.5530 (857.1)	→	→	→
Hot Chute Exit Flow Area	AE77	m^2 (in^2)	.1599 (247.8)	→	→	→
Nozzle Exhaust Area	A8	m^2 (in^2)	.6495 (1006.7)	→	.6552 (1015.5)	→
Thrust F_N		N (lb)	35710 (8028)	37525 (8436)	35541 (7990)	37205 (8364)
Specific Fuel Consumption	SFC	Kg/N-sec(lb/hr-lb)	.0409 (.401)	.0410 (.402)	.0411 (.403)	.0413 (.405)
Inlet Pressure Loss	$\Delta P/p$	%	2.04	2.29	2.04	2.29
Fan Duct Pressure Loss	$\Delta P/p$	%	5.00	5.22	5.00	5.22
Cold Chute Pressure Loss	$\Delta P/P$	%	0.80	0.84	0.80	0.84
Hot Chute Pressure Loss	$\Delta P/p$	%	2.62	2.93	2.62	2.93
Velocity Decayer Pressure Loss	$\Delta P/p$	%	-	-	0.99	1.07
Inlet Airflow W2		Kg/sec(lb/sec)	149.6 (329.9)	151.3 (333.6)	149.6 (329.9)	151.3 (333.6)
Fan Rotational Speed N_F		%	99.5	102.7	99.5	102.7
Fan Pressure Ratio P23/P2			1.489	1.555	1.489	1.555
Exhaust Velocity V8		m/sec (ft/sec)	247.9 (813.2)	257.6 (845.2)	246.7 (809.3)	255.6 (838.7)
Bypass Ratio BPR			6.55	6.44	6.55	6.44

NOTE: 100% fan speed = 733 rad/sec (7000 rpm)

[1] EFTC refers to Edwards Flight Test Center; 670.6m (2200 ft), 266.7 °F (20 °F)

5. <u>INSTRUMENTATION AND MOUNTING</u>

a. <u>Instrumentation</u> provided in the quiet nacelle is shown in Figure 31. Broadly speaking, basic engine instrumentation is required to accurately determine the precise engine operating point for correlation with noise level. In addition, the engine which is just in the final stage of development will necessarily be run off-design in the Quiet Nacelle program. Careful monitoring of the engine is required to protect the engine from possible dangerous operating conditions.

The total pressure rakes provide data on possible adverse flow induced by the suppressors. They also allow determination of engine airflows, pressure ratios, flow distribution in suppressor splitter passages and suppressor pressure losses.

The acoustic instrumentation in the nacelle is especially directed at determination of the amount of suppression achieved. The suppression objectives of the Quiet Nacelle program are well beyond the current state-of-the-art, thus simple measurements of the resulting engine noise level are not sufficient. It is necessary to measure noise energy and spectra into and out of the suppressors to fully understand how the suppressor works (or does not). The traversing acoustic probes provided in the suppressors serve this purpose.

Kulites are specified on the walls of the suppressors. A Kulite is essentially a pressure transducer with frequency response of 0 to 50000 cps. They are small (3 mm (1/8 in.) dia) and can be flush mounted. They provide another tool to analyze suppressor performance which is more direct than microphones located outside the engine. They should provide data on boundary layer noise content also.

b. <u>Engine/Nacelle Mounting</u> - The TF34 Quiet Nacelle engine is mounted on the test stand identical to all previous engines in the TF34 program. See Figure 32 The stand to be used is a TF34 development test stand which assures the minimum of surprises and cost in successfully getting the engine on test.

The nacelle, both inlet and fan exhaust, are mounted independent of the engine on the same test rig pylon. The nacelles are heavy ground test hardware which would impose excessive loads and deflections if mounted directly on the engine. The interface between engine and nacelle is through flexible seals designed to allow relative motion and still maintain a smooth flowpath surface. The seals will not transmit excessive loads into the engine.

The nacelle mounting provides for adjustment at the relative position of nacelle and engine to account for tolerances, thermal growth and mount elasticity. Proper relative location assures a smooth quiet flowpath.

c. Instrumentation Justification

Acoustic Instrumentation

Traversing Acoustic Probes are required to determine the noise energy and spectra in the fan ducts as a function of radial position. There currently is no data of this type on the TF34 engine. To optimize a suppressor design the radial energy and spectra noise source data is required. The source data is obtained from the two traversing acoustic probes located close to the fan both forward and aft. The efficacy of the suppressors is determined similarly by traversing acoustic probes located at the forward end of the inlet suppressor and aft end of the fan exhaust suppressor. This allows a direct real time comparison of source noise into the suppressors and the suppressed noise out.

The Core Acoustic Rake is used to collect data on the noise emanating from the low pressure turbine exhaust. The rake which has four radial stations is located in the exhaust stream just aft of the Station 8 plane at the core exhaust. The source data will be obtained by using the rake during the unsuppressed baseline tests. The amount of core suppression achieved is determined by comparing the unsuppressed baseline noise data with core exhaust noise data taken with the core suppressor in place. The contribution to suppression due only to added duct length required by the suppressor is determined by using the rake with an untreated exhaust duct the same length as the core suppressor. There is still considerable uncertainty concerning the noise spectra of the core exhaust over the engine operating range. Here, as in the fan suppressors, the comprehensive data provided by rakes is required to allow future optimized designs for Questol.

Kulite pressure transducers are specified in the fan exhaust duct suppressor principally to determine noise level versus axial length. There are laboratory data that indicate diminishing returns of suppression versus length. See Figure 15 and 16. However, the performance of a suppressor in an engine application will differ from the laboratory experiments. The Kulite data will quantify the difference. It is possible that the noise generated in the boundary layer of the fan exhaust will be quite important since the overall suppression objective is so large. The duct Mach number of .45 is about 161 m/sec (360 mph) air velocity over a perforated surface of considerable area, which certainly has the potential of generating considerable noise.

Combining the data from the traversing acoustic rakes, the Kulites and the other instruments not mounted on the nacelle, it should be possible to gain sufficient insight for future improved acoustic design.

One Kulite is located in the bay between nacelle and engine to measure noise that is generated by the core and carried in the bay cooling air. This instrumentation is necessary because the large suppression objectives will make this flanking noise source potentially important.

Accelerometers are mounted on the aft core cowl to measure noise radiated from the cowl. The conical core cowl which extends aft from the fan nozzle to the core nozzle radiates directly to ambient on the separated flow engine. It can not be covered with the acoustic blanket material used on other parts of the engine because of the high velocity fan jet.

The source of the noise radiated from the cowl is 1) the noise contained in the bay cooling air, 2) the noise in the bay due to various acoustic leak paths (such as the nacelle flexible seals) and 3) noise radiated from the core casing. The cowl radiated noise can be important because of the high level of suppression of the nacelle.

Aero-Thermo Instrumentation

Station 1 — P_T and P_S measurements are used to determine Mach number. T_T measurements with Mach number and known flow area determine engine airflow. P_T in conjunction with Station 2 P_T are used to compute inlet splitter ΔP. T_T is used to reference engine data to standard conditions.

Station 1.5 — T_T measurements are used to reference engine data to standard conditions when Station 1 rakes are removed.

Station 2 — P_T and P_S measurements determine Mach number. Individual P_T data is used to assess inlet distortion. In conjunction with Station 1 P_T is used to compute inlet splitter ΔP. P_T is used to reference engine parameters to standard conditions.

Station 24 — P_T and P_S measurements determine Mach number. P_T in conjunction with Station 2 P_T determine fan pressure ratio. T_T in conjunction with Station 1 T_T determines fan temperature rise, thus fan efficiency. P_T in conjunction with Station 25 P_T determine aft splitter ΔP.

Station 24.5 — P_T and P_S measurements determine Mach number. Mach number data is used to assess proper flow distribution between splitters and to correlate with acoustic suppression achieved.

Station 25 — P_T and P_S measurements determine Mach number. P_T in conjunction with Station 24 P_T determine aft splitter ΔP. P_T reduced by a small loss value determines Station 28 P_T which is used with $P_{ambient}$ and/ or Station 28 P_S to determine fan nozzle pressure ratio, and thus fan exit velocity.

Station 28 — P_S measurements determine actual fan nozzle exhaust pressure. Used with Station 25 P_T to determine fan nozzle velocity.

Station 2C — P_T and P_S measurements determine Mach number which with T_T and flow area determine compressor air flow. T_T used to reference core operation to standard conditions. P_T in conjunction with Station 3 P_S determine compressor pressure rise. T_T in conjunction with Station 3 T_T determine compressor temperature rise, thus compressor efficiency.

Station 3 — P_S and T_T are used together with Station 2 P_T and T_T to determine compressor efficiency. PS and fuel flow rate provide a check on proper fuel scheduling. P_T is derived from P_S and used with T_T, W_T, and a known flow parameter to determine maximum engine cycle temperature (T_{T4}). Combustor enthalpy balance and cooling flow rates provides a check on core airflow (W_{2c}).

Station 5.4 T_T is the main control input for engine operation. It can be used in conjunction with Station 6 T_T to determine fan turbine efficiency. P_T is used to determine engine pressure ratio, and with T_T and the calculated Station 4 P_T is used to check calculation of maximum engine cycle temperature.

Station 5.5 P_T is used in conjunction with Station 5.4 P_T and Stations 5.4 and 6 T_T to determine fan turbine efficiency. It is also used in conjunction with Station 7 P_T to determine tailpipe ΔP.

Station 6 30 thermocouples are used primarily to assess temperature spread. High non-uniformity of temperature is the first indication of deterioration in engine health.

Station 7 P_T is used in conjunction with Station 5.5 P_T to determine tailpipe ΔP. It is used with a small loss factor to determine Station 8 P_T which is used in conjunction with Station 8 P_S to determine core nozzle pressure ratio, thus core exit velocity.

Station 8 P_S is used with Station 7 P_T to determine core nozzle pressure ratio, thus core exit velocity. It is also used in conjunction with $P_{ambient}$ to determine the level of fan nozzle flow interaction with core nozzle.

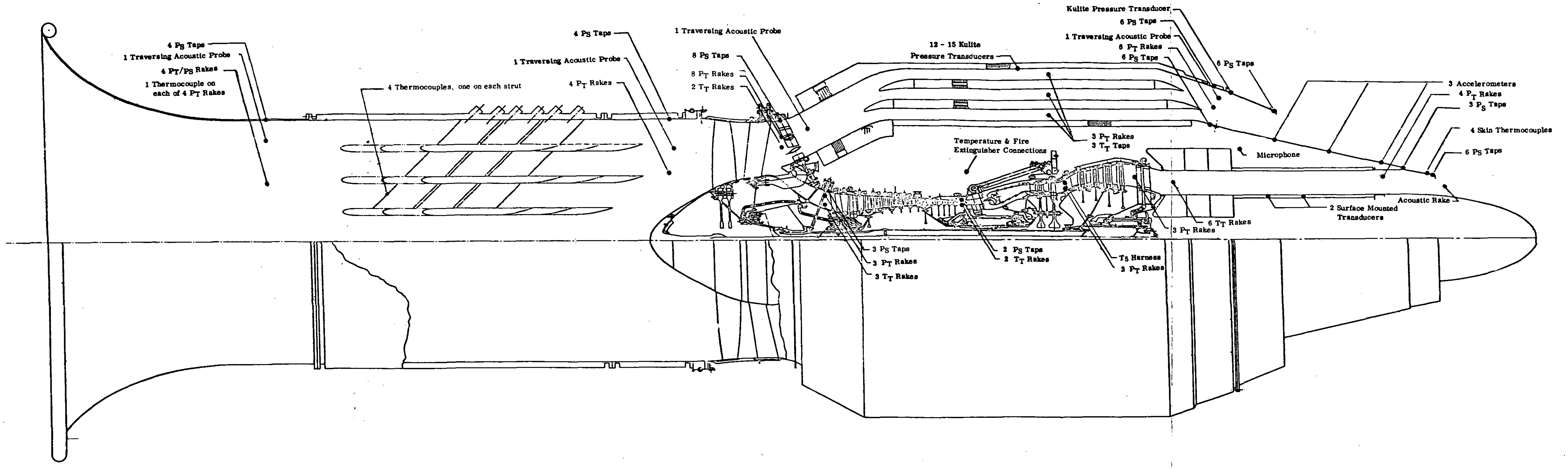


Figure 31. NASA TF34 Quiet Nacelle Instrumentation.

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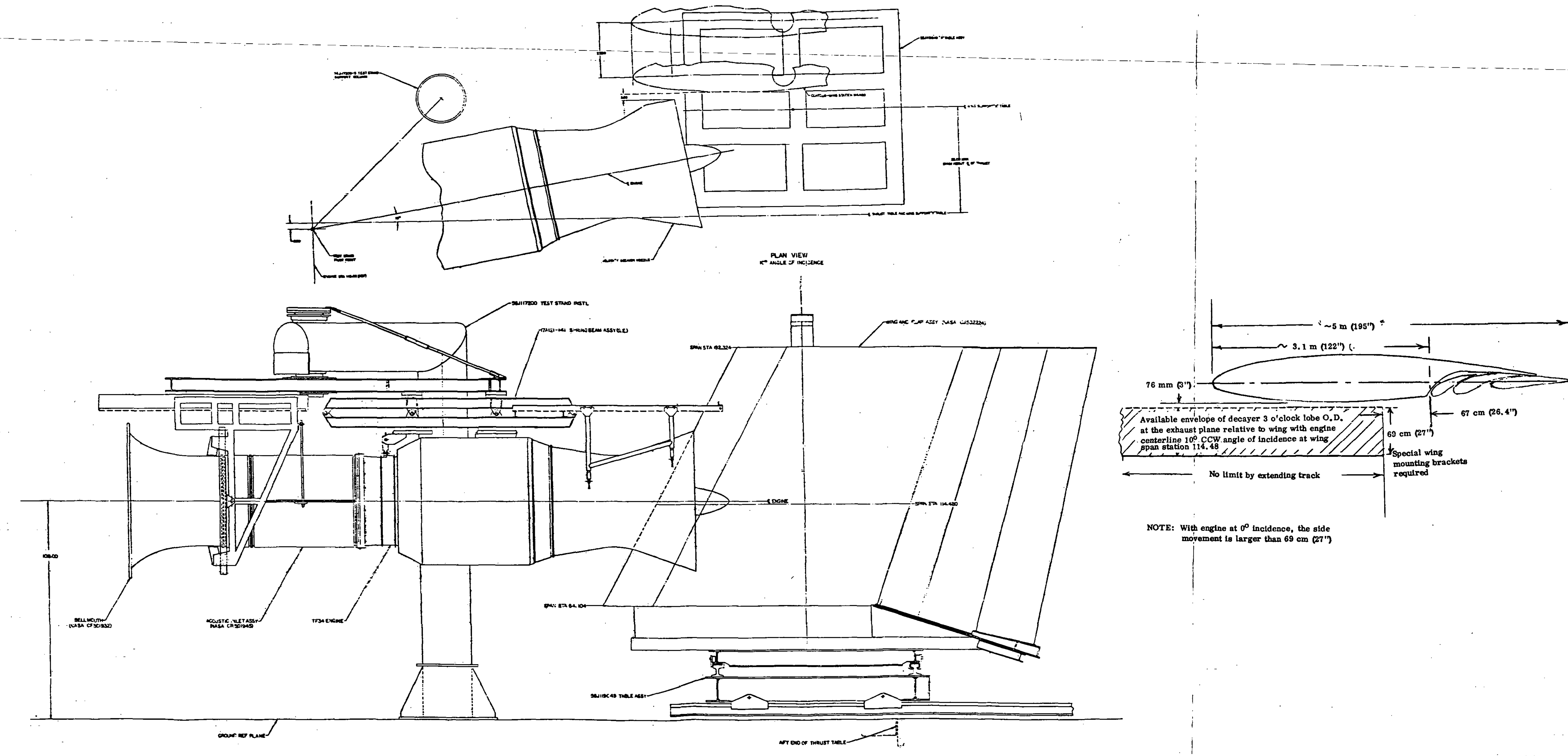

Figure 32. NASA TF34 Quiet Nacelle and EBF Wing Section Installed in the North Site Engine Test Stand at Edwards Flight Test Center.

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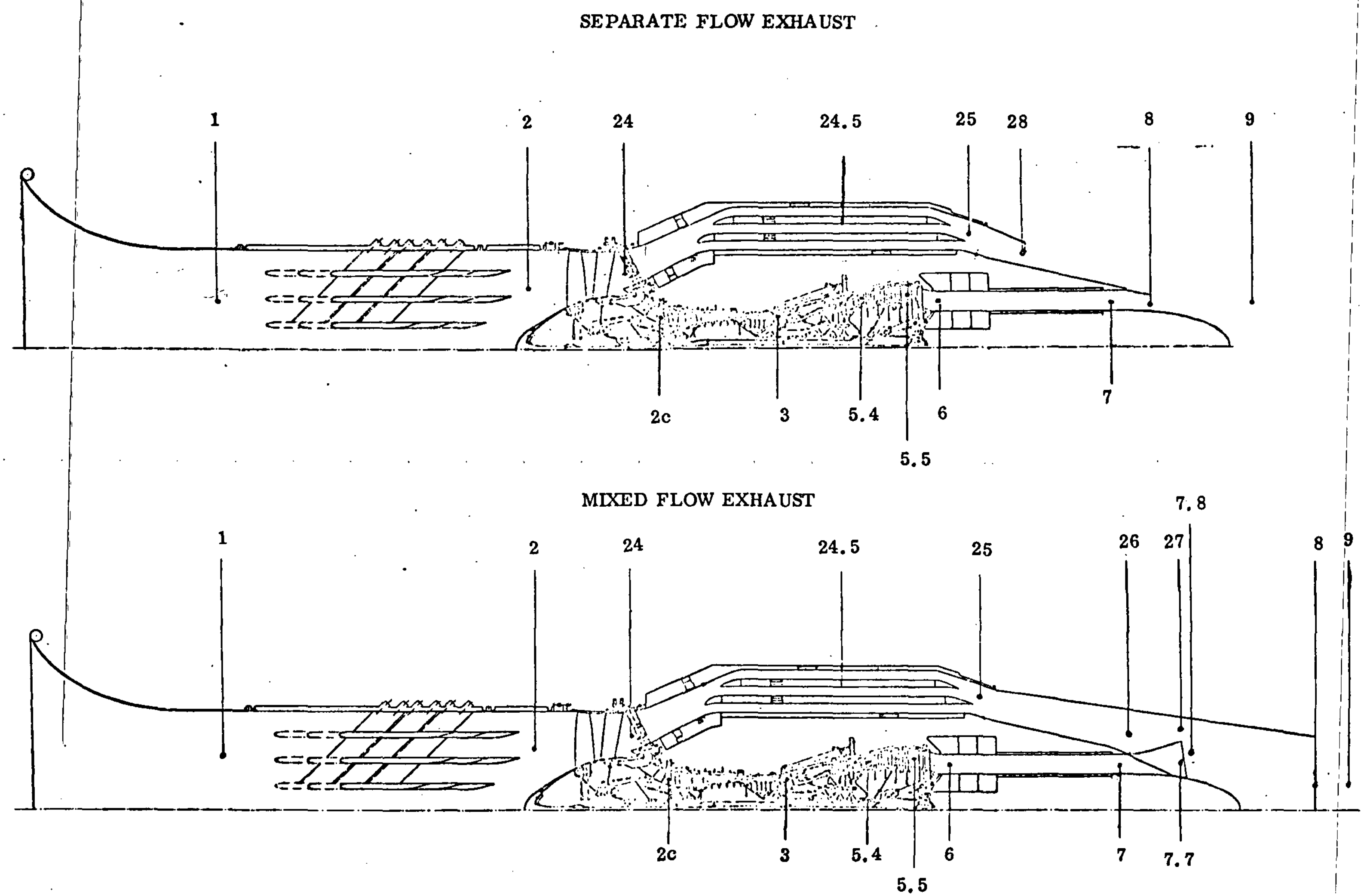

Figure 33. Engine Station Designation.