Development of Sonic Inlets for Turbofan Engines

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Results of a current program for sonic inlet technology development are presented. This program includes configuration and mechanical design selection of concepts, aerodynamic design description of the models, and results of test evaluation. In the test program several sonic inlet concepts were tested and compared for aerodynamic and acoustic performance. Results of these comparative evaluations are presented. Near-field measurements were taken inside the sonic inlet on several inlet models. Results of these tests are discussed with respect to the effect of Mach number gradients on noise attenuation and rotor shock wave attenuation, and boundary-layer effects on noise propagation. The test facilities and experimental techniques employed are described briefly.

Nomenclature

D = fan face diameter, in.
L = distance from fan face to highlight plane, in.
M = Mach number
$\overline{M}$ = average Mach number
P = pressure, lb/sq in.
$\bar{P}$ = average pressure, lb/sq in.
PNL = perceived noise level, PNdb
Rec = average total pressure recovery, P_t/P_{t_∞}
SPL = sound pressure level, db
W_a = weight flow, lb/sec
δ = $\bar{P}_t/P_{std}$
δ^* = boundary-layer thickness
θ = T_{amb}/T_{std}

Subscripts

amb = ambient (anechoic chamber)
max = maximum
min = minimum
N = normalized
std = standard
t = total
∞ = ambient condition

1.0 Introduction

SONIC inlet noise and performance potential has been under investigation for the past several years (see Bibliography). These programs covered a large number of investigations and a variety of configurations and experimental techniques. They have demonstrated the basic feasibility of sonic inlets and the potential for obtaining large reductions in inlet noise by inlet choking, thus contributing significantly to inlet noise technology. However, limitations exist in the results of these earlier experiments. Sufficient attention was paid to the aerodynamic design details on full-scale sonic inlet engine tests conducted, but in most cases configuration constraints were imposed in order to use existing hardware, resulting in compromised performance. Acoustic test data from engine tests were usually affected by other engine noise sources. Most of the scale-model tests lacked the aerodynamic design detail and adequate instrumentation for the evaluation of the aerodynamic performance potential of the different configurations.

The task of identifying a sonic inlet for a practical airplane application requires more than the demonstration of feasibility. Configuration concepts have to be compared and evaluated in terms of their aerodynamic and noise performance, mechanical design feasibility, and overall system considerations. The technology data base gained from past program results was inadequate for these types of studies.

This program, conducted under a NASA-Lewis contract,[1,2] had the objective of expanding the technology base for sonic inlets. In the area of aerodynamic performance, pressure recovery of the different inlet concepts had to be established for comparison studies. Flow distortion and inlet/engine compatibility trades required the determination of the distortion characteristics of different sonic inlet concepts. Finally, the noise potential of different inlets had to be demonstrated.

The program was mainly directed toward sonic inlets for STOL propulsion system inlet noise reduction, with emphasis on the augmentor wing application. The inlet noise reduction requirement stems from the low aft arc noise level demonstrated by the augmentor wing propulsion system concept.[3] As a result, the dominant noise source is that radiated forward by the engine fan. Preliminary analyses indicated a required inlet noise suppression of 20 to 30 PNdb at a 500-ft sideline distance. By the demonstration of the objective inlet noise level reduction, the noise of a 150-passenger augmentor wing STOL airplane could be reduced below 95 PNdb at a 500-ft sideline. The program conducted has demonstrated that the noise level objectives can be achieved and exceeded by sonic inlets, with aerodynamic losses substantially lower than previous sonic inlet program results indicated. This performance was demonstrated with a realistic inlet length and without the use of sophisticated flow control devices.

2.0 Concept Selection and Model Design

The program was initiated with a sonic inlet configuration and concept selection study. In this effort, preliminary mechanical design layouts were made of several promising sonic inlet concepts. These concepts were studied in two separate groups, consisting of single-passage inlets (throat undivided) and multipassage inlets (throat subdivided). All concepts were evaluated in terms of performance, mechanical design, and operational characteristics, and potential problem areas were identified. On the

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Index category: Aircraft Noise, Aerodynamics (Including Sonic Boom).

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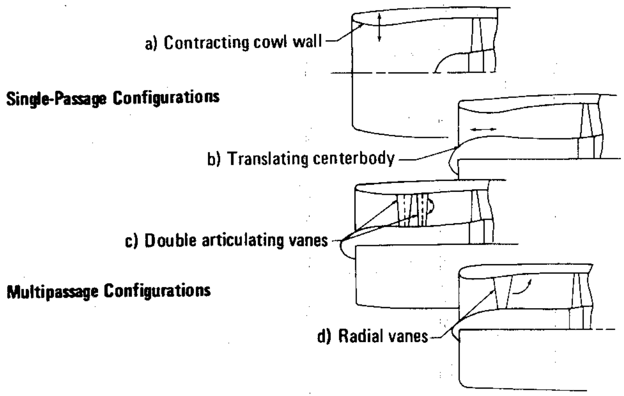

Fig. 1 Sonic inlet test configurations.

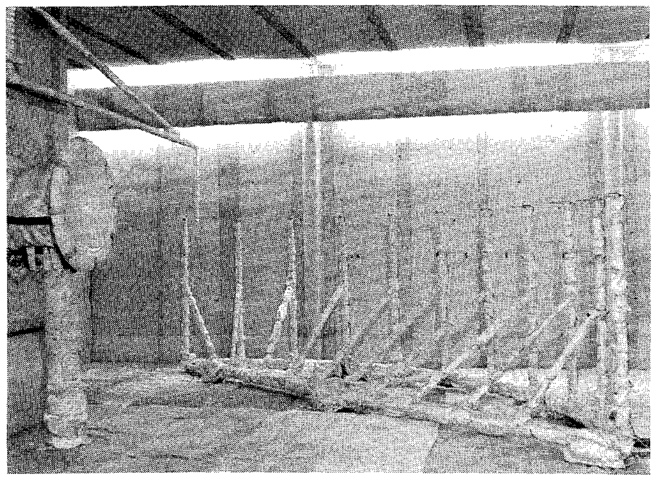

Fig. 3 12-in. rig microphone array.

basis of this concept screening study, two single-passage and two multipassage inlets were selected for preliminary test evaluation. These were translating centerbody, contracting cowl wall, retractable radial vane, and double articulated radial vane. A schematic representation of each of these concepts is shown on Fig. 1.

The four selected inlet models were designed for test evaluation. The basic design weight flows for all configurations at full scale were as follows:

approach = 402 lb/sec
takeoff = 515 lb/sec
maximum cruise = 476 lb/sec

These were based on engine criteria used for system design and evaluation studies of jet STOL aircraft.[3]

The design procedures were similar for both of the single-passage inlets, with throat and diffuser exit areas defined by the engine airflow requirements; the prime variables were length (L/D, inlet length to fan diameter ratio) and diffuser area distribution. These variables were initially selected on a trial-and-error basis and evaluated with the aid of a computerized potential flow program combined with a boundary-layer program. Surface Mach number, boundary-layer shape factors, and boundary-layer thickness were calculated and plotted as a function of diffuser length. The criterion used for inlet optimization was the attainment of minimum length without boundary-layer separation or excessive boundary-layer thickness. Predicted inlet performance parameters for a translating centerbody inlet in the approach configuration are shown in Fig. 2. The Mach number distribution was selected to achieve rapid diffusion in the high Mach number region. In general this diffusion rate distribution resulted in designs with the lowest boundary-layer thickness and compressible shape factors.

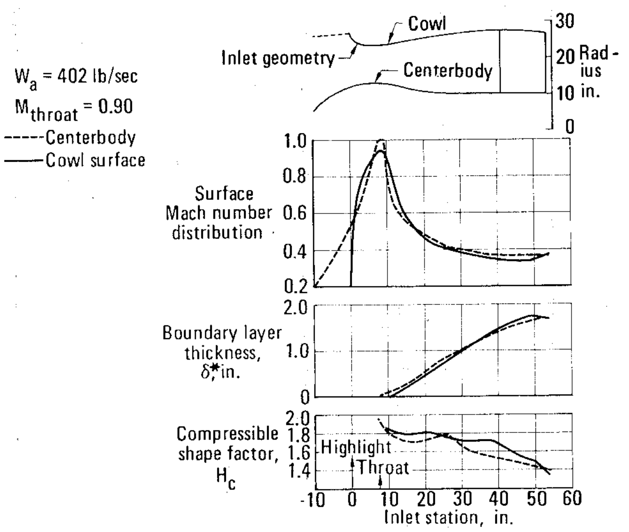

Fig. 2 Design parameters for $L/D = 1$ translating centerbody inlet (approach configuration).

The multipassage inlet models were designed by the use of a freestream axisymmetric radial equilibrium turbomachinery design program. Standard NACA airfoil shapes were used in the airfoil selections. The basic design objective for the multipassage inlets was to attain uniform radial Mach number distribution.

3.0 Test Setup and Instrumentation

The tests were conducted in an anechoic chamber (Fig. 3). The wall between the test chamber and the anechoic room was acoustically impermeable, ensuring that measurements taken in the acoustic chamber were not influenced by fan aft-radiated noise. The test fan was a 12-in. tip diameter, single-stage fan with a design tip speed of 1400 fps and a nominal stage design pressure ratio of 1.5. The rotor had 32 blades with a hub/tip ratio of 0.36, followed by a tandem stator with 27 blades in each vane row. An adjustable valve was used in the exhaust for varying the loading on the fan.

Acoustic measurements were taken by a microphone array on a 10-ft radius at 10° intervals. Aerodynamic data were also acquired during testing for evaluation of inlet recovery, distortion, and throat Mach number. Wall static pressure measurements were made.

A baseline inlet was tested in the program to study the near-field noise attenuation characteristics of sonic inlets.

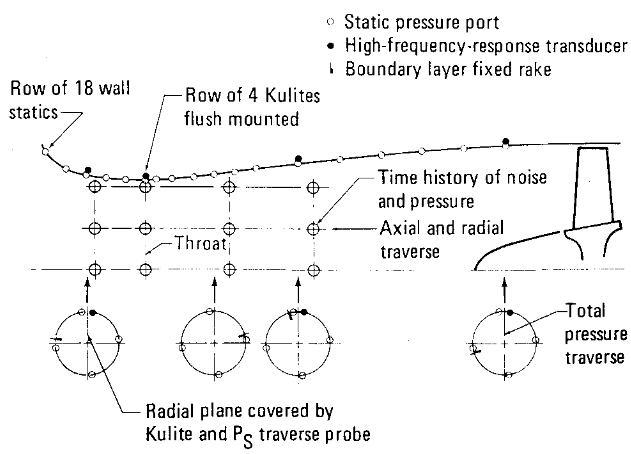

Fig. 4 Sonic inlet noise mechanisms test instrumentation.

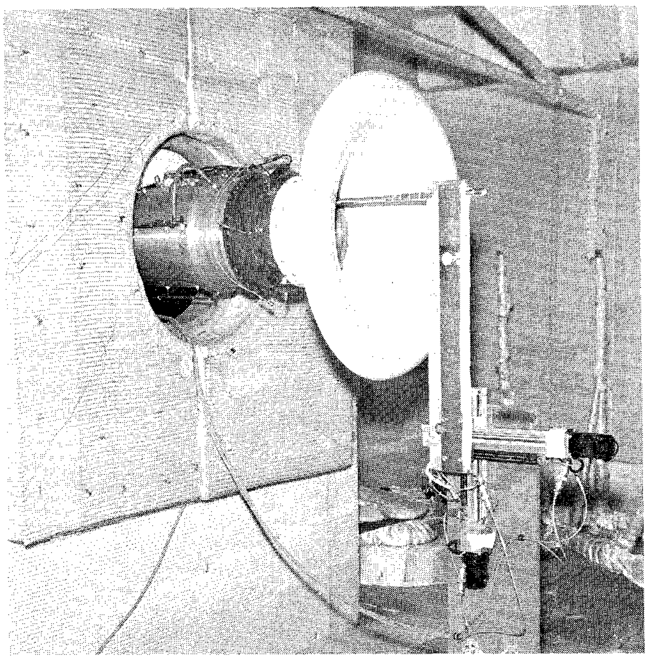

Fig. 5 12-in. fan rig system.

This inlet was more extensively instrumented than the other models. Instrumentation included a line of wall static pressures, boundary-layer rakes, and inlet wall Kulite probes for noise measurement (Fig. 4). Total pressure rakes were traversed at the exit plane of the inlet to obtain recovery and flow distortion data.

The baseline inlet was probed in the near field with a specially constructed probe for continuous measurement of noise and static pressure inside the inlet (Fig. 5). The probe was mounted on an X-Y traverse mechanism to enable continuous recording in both axial and radial planes.

The probe carried on its tip a miniature (1/8-in.-diam) transducer for measurement of noise. The transducer was calibrated prior to usage by imposing pure tone signals on the transducer and data acquisition systems at various frequencies in the range of interest. The resulting spectrum (Fig. 6) showed that the system had a noise floor of 80 db. This calibration was important in distinguishing between aerodynamic noise floor and data acquisition system floor. Two static pressure taps were drilled 1 in. upstream of the probe end to minimize the probe end effect on the static pressure measurements.

4.0 Results and Discussion

Near-Field Noise Surveys

In support of the concept selection effort, a test was conducted with the specific purpose of obtaining near-field noise measurements inside a sonic inlet to gain some understanding of the noise attenuation mechanisms involved with large Mach number gradients such as those existing at or near sonic flow conditions. The experimental approach was chosen for lack of analytical tools in this technology area.

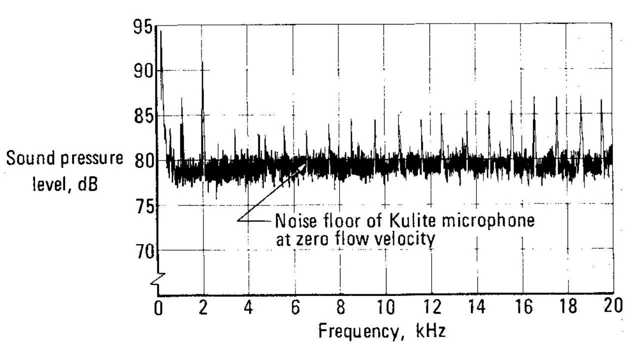

Fig. 6 Nearfield transducer noise floor calibration.

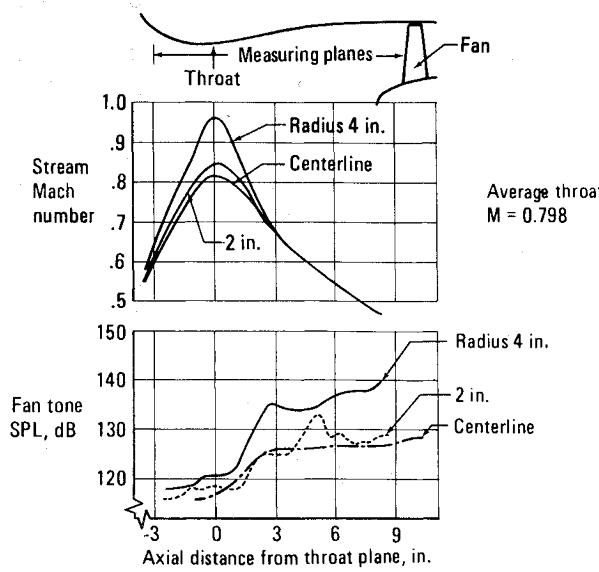

Fig. 7 Effect of radial and axial location of fan tone levels.

The objectives of these experiments were to determine and provide some insights into potential problem areas. Of specific interest were Mach number gradient effects on noise attenuation, shock wave propagation through the sonic throat, and noise leakage through the boundary layer.

Typical Mach number and blade passage frequency noise contours are shown based on the axial traverses at various radial positions (Fig. 7). The data show that the noise level at the fan face was greatly dependent on radial position. The noise levels were lowest at the hub, and highest at the tip. In the throat region, rapid noise reduction took place when the Mach number exceeded 0.7, with the minimum noise measured at the throat. Radial noise gradients disappeared at the throat. The leveling out of the noise trace in the throat region is believed to have been a result of the aerodynamic turbulence in the flow creating a noise floor.

Specific attention was centered on the attenuation of shock waves and multiple pure tones by sonic inlets at supersonic fan tip speeds. Noise measurements were taken near the fan face, downstream of the throat, and in the throat. The spectrum comparisons for these measuring points indicated that all pure tones were effectively attenuated by the sonic inlet (Fig. 8).

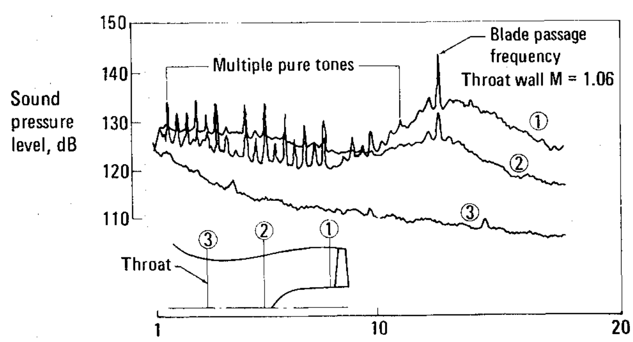

Fig. 8 Noise spectrum attenuation behind a sonic inlet.

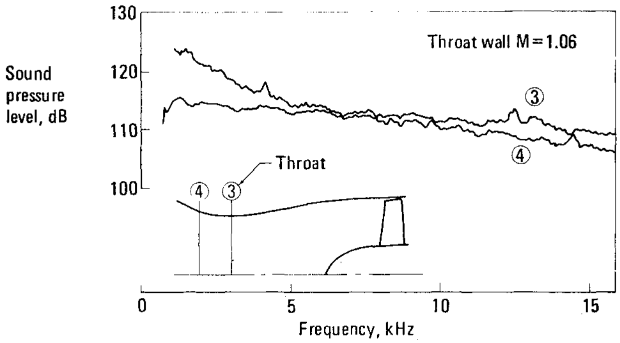

Fig. 9 Noise spectrum comparison ahead of a sonic inlet.

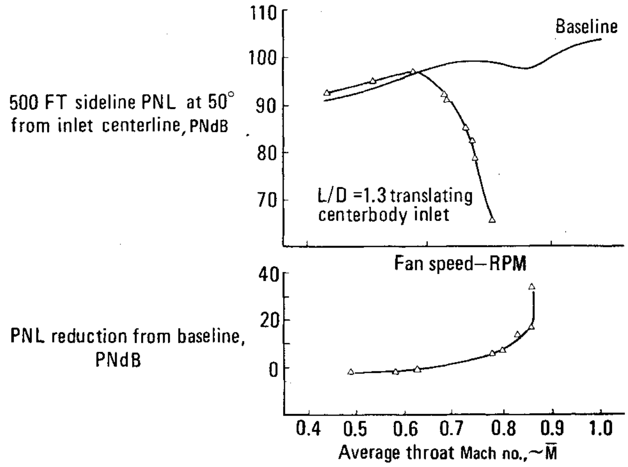

Fig. 11 PNL vs fan rpm and ΔPNL-reduction vs throat Mach number.

Another point of interest was the possible noise generated by the sonic plane itself. Therefore, measurements were also taken ahead of the throat plane (Fig. 9). The comparisons indicate that no significant noise increase was experienced because of noise generation by the sonic plane for the test conditions investigated.

Figure 10 shows the blade passage tones measured by the traversing probe at two different radial positions in comparison to the flush-mounted probes on the inlet wall for an average throat Mach number of 0.98. This data comparison indicates that the noise level in the boundary layer attenuated at rates similar to those of the freestream. Thus, it was concluded that a thin boundary layer was not a significant path for noise propagation or "leakage" in sonic inlets.

Over-All Performance Evaluation

The over-all performance evaluation testing of the four inlet concepts described in Sec.2.0 was conducted by increasing the fan speed in increments to produce weight flow increase, and thus Mach number increase, with constant throat area inlets. The data were compared to the baseline noise levels at the same fan speed. A typical test run with sonic inlet and the baseline configuration is shown in Fig. 11.

To determine the noise reduction in terms of PNdb, the model data were scaled up to a larger fan size that would be representative in size of an augmentor wing transport powerplant. Scaling was accomplished by accepted methods of shifting the frequencies by geometric scale ratio and making an adjustment for sound pressure level by 10 $\log_{10}$ of the weight flow ratios between model and large-scale fan. The scaled-up data were extrapolated to 500-ft sideline by standard methods. All other data are presented based on direct measurements on the model. (No Reynolds number corrections were made.)

Performance parameters are shown in terms of inlet normalized Mach number (M_N), which was introduced to eliminate large deviations at the choked condition in calculated Mach numbers for the different test configurations, due to the effective flow area differences inherent in configurations from the design area values (Fig. 12). For the choked condition the Mach number for all configurations was defined as unity in calculating normalized Mach numbers. This minimized the apparent differences in performance due to effective area differences (Fig. 13).

Single-passage inlet test results show that a 28 to 30 PNdb inlet noise reduction at approach engine power setting could be achieved by sonic inlets with 0.97 inlet recovery within acceptable inlet distortion limits (Fig. 14). This showed that sonic inlets within realistic length limits ($L/D = 1$) can be designed to achieve high noise reduction and acceptable aerodynamic performance on a static performance basis. Takeoff configuration tests have shown similarly good results (Fig. 15).

The measured single-passage inlet fan face total pressure distortion levels are shown for a range of Mach number conditions (Fig. 16). With the exception of the last test condition shown, no significant flow separation was indicated by these measurements. A further investigation of the flow separation phenomenon at a high Mach num-

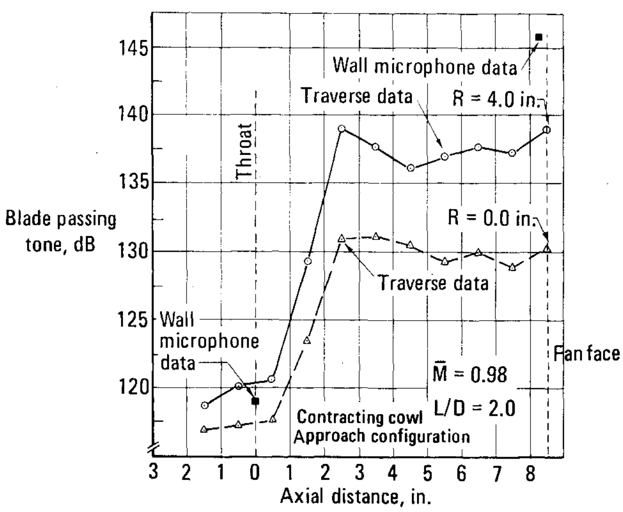

Fig. 10 Blade passing rone vs axial distance.

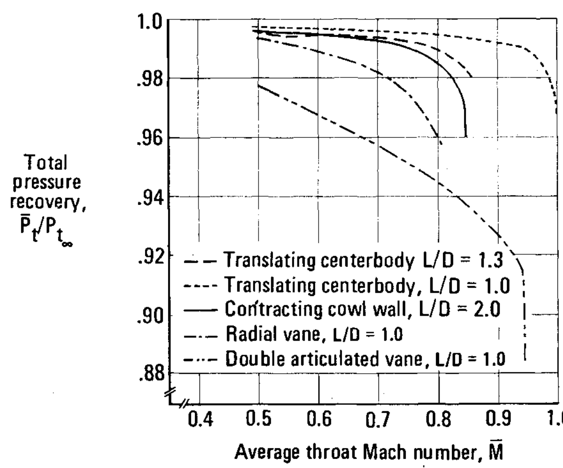

Fig. 12 Comparison of inlet recovery vs average throat Mach number, approach configuration.

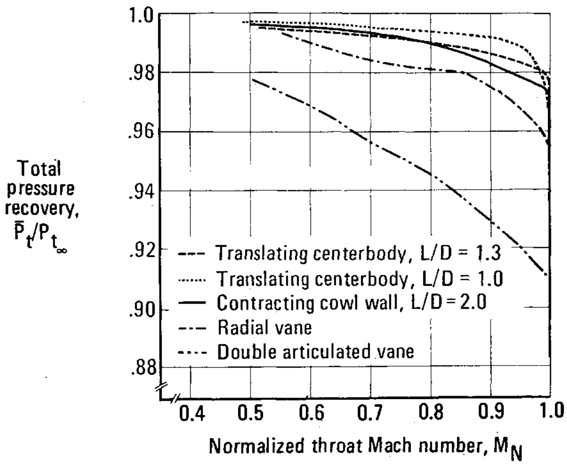

Fig. 13 Comparison of inlet recovery vs normalized throat Mach number, approach configuration.

ber and low recovery operation would be desirable to extend the range of operating envelope of these inlets.

Two multipassage sonic inlets were tested. Acoustic and aerodynamic data from these tests are shown in comparison to the single-passage inlets on Fig. 14. The multipassage inlets produced lower noise reduction with significantly lower pressure recovery. In spite of the poorer static performance of the multipassage inlets, they should be considered candidates until wind tunnel evaluation of the different concepts can be made. Multipassage inlets are believed to be less sensitive to angle of attack and crosswind conditions.

Spectral Attenuation

Spectral comparison of the noise for the baseline and sonic inlet indicates that some dependence of attenuation

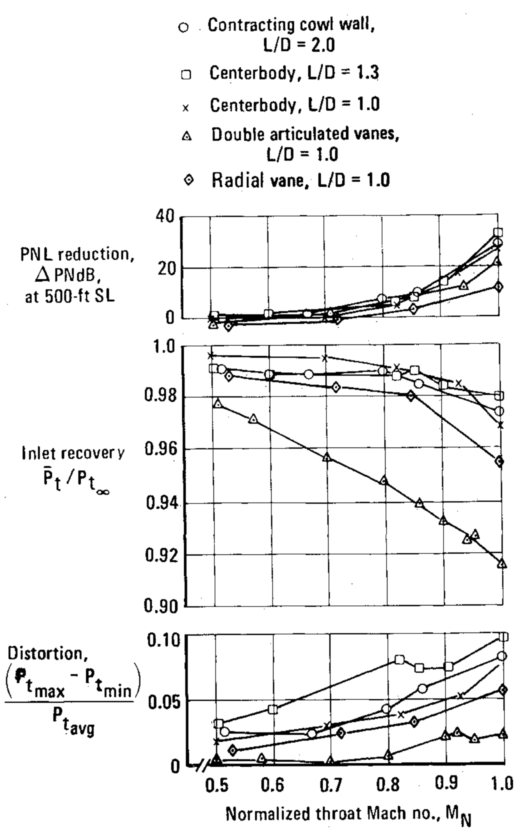

Fig. 14 Sonic inlet performance—approach configuration.

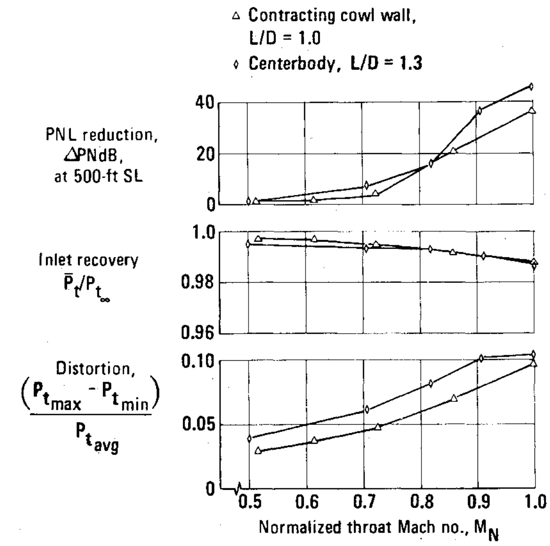

Fig. 15 Sonic inlet performance—takeoff configuration.

effectiveness existed as a function of frequency (Fig. 17). Pure tones of high frequency and high-frequency broadband noise were more effectively attenuated than low-frequency noise. This phenomenon seemed to be Mach number dependent, indicating that flow velocity effects may interact with the direction of noise propagation. Thus, plane waves (low-frequency noise) traveling in the axial direction would propagate from the inlet at high flow Mach numbers, while traversely propagating modes (high frequency) got reflected at lower inlet flow velocities. Further analytical work would be required to fully understand and quantify these relations.

Noise Directivity Effects

Noise directivity measurements taken in the acoustic chamber with adequate inlet sidewall insulation and other noise sources minimized (Fig. 18) indicated that the noise was effectively reduced at all angles in the forward arc. This result helps to clarify some questions that arose from earlier experiments with regard to sideline effectiveness of sonic inlets.

Acoustic Lining Effectiveness with Sonic Inlets

The L/D = 1.3 translating centerbody inlet has been tested with acoustic lining applied to the cowl and center-

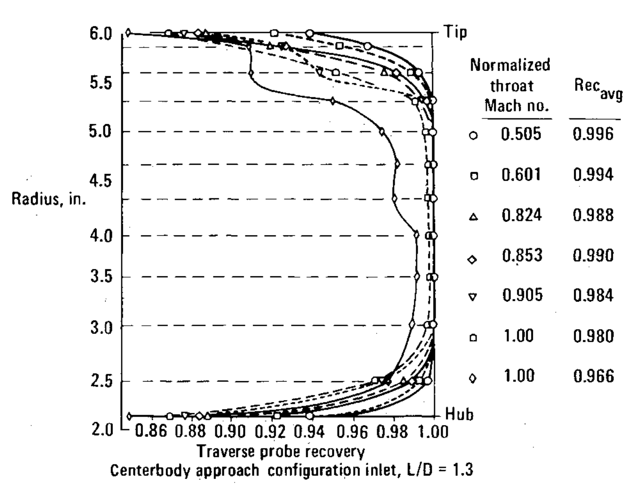

Fig. 16 Sonic inlet fan face radial pressure profile.

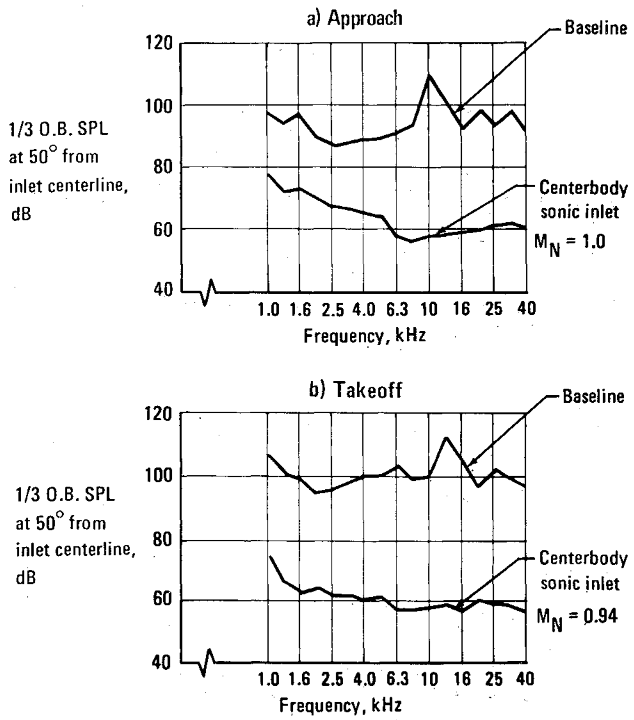

Fig. 17 Noise spectrum comparison—scale model data.

body surfaces. The lining consisted of a tuned resonator, with a polyimide facing sheet and honeycomb backing.

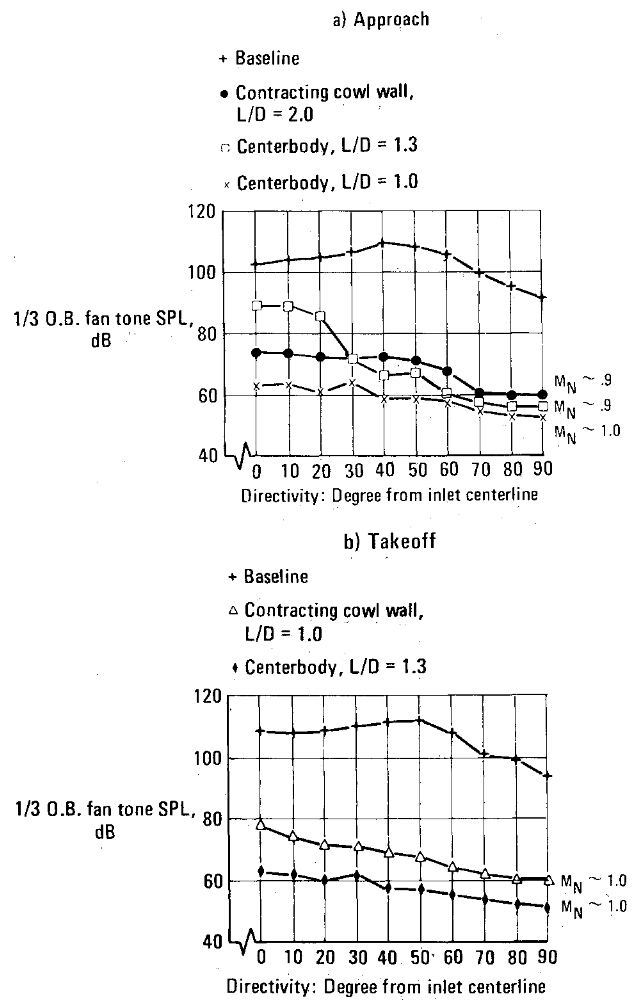

Fig. 18 Fan tone directivity comparison.

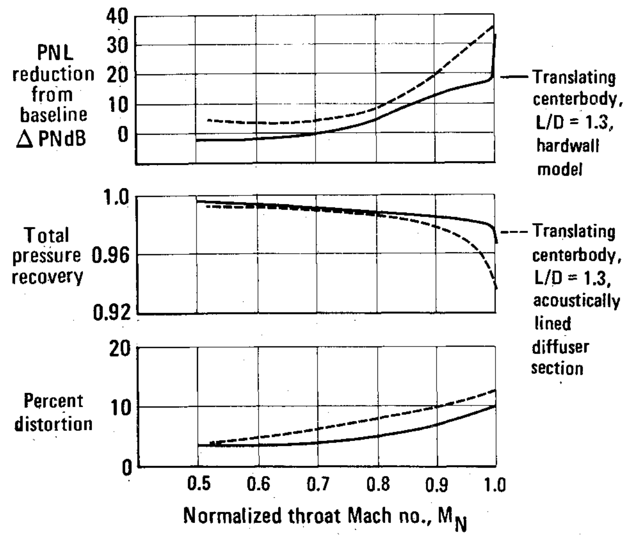

Fig. 19 Effects of acoustic lining on the L/D = 1.3 translating centerbody inlet, approach configuration.

The lining was tuned for the blade passage frequency at high flow Mach numbers. The first configuration tested was lined throughout the inlet. Aerodynamic data obtained on this test indicated severe flow separation in the diffuser section. This configuration also showed very poor noise reduction. Since the hardwall configuration of this inlet geometry showed good performance, it was evident that the surface roughness increase was the cause of the change in performance.

Several successive tests were run masking various amounts of treatment in the throat region. When the lining was masked on the centerbody and the cowl surface to the point where the surface Mach number was 0.70, performance of the inlet was restored to nearly that of the hardwall inlet. The results of this final lining test, presented in Fig. 19, show that some inlet throat Mach number reduction can be achieved for the same noise reduction with acoustic lining. The payoff for using acoustic lining with sonic inlets depends on the particular noise reduction level desired.

Sonic Inlet Stability Observations

During the tests it was noticed that the flow and noise reduction became unstable in the transonic flow region ($M_{\text{wall}} \approx 0.9 - 1.1$). This phenomenon was exhibited by intermittent flashes of short-duration noise spikes in the data and by the appearance of characteristic sounds of turbulent flow separation. Since the data were reduced by 32 sec time averaging of the acoustic signal, in most cases the phenomenon is not evident in the noise data due to the relatively short duration of these sound bursts. Detailed investigation of the instability phenomenon was beyond the scope of this program. It appears, though, that these instabilities resulted from diffuser flow separation initiated by unstable shock-boundary layer interaction in the throat region. Further investigation of this phenomenon should be undertaken in the future.

Fan Operation with Sonic Inlets

Figure 20 shows the operating line for two sonic inlets superimposed on the baseline fan map. These operating lines show that changes in fan operating points were very gradual even after the inlet choke point, where the recovery versus throat Mach number took a vertical drop. The choke points are shown for each test run. The radial vane inlet in the approach configuration showed a gradual as-

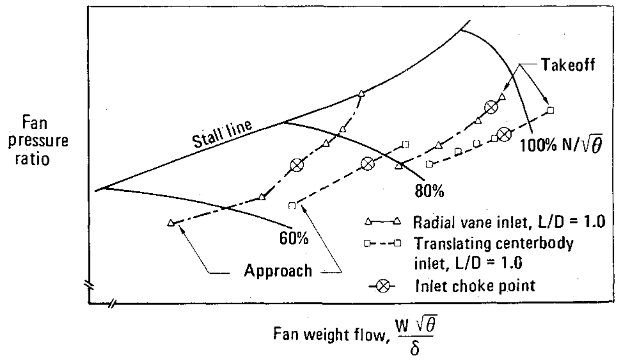

Fig. 20 Fan operating lines with sonic inlets.

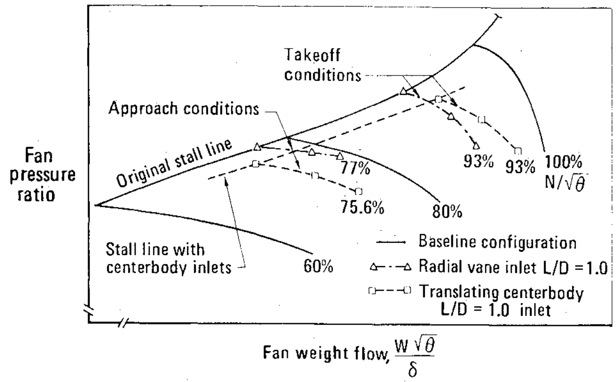

Fig. 21 Effect of sonic inlets on fan stall line.

cendance to the stall line; however, the highest test points were of unrealistically low recovery points for real inlet-engine operation. All other operating lines exhibited a near-normal operating line representative of constant exhaust nozzle area setting with conventional inlets.

Figure 21 shows stall line changes due to flow distortion created by two sonic inlets. The radial vane inlet showed no effect on the fan stall line, but the centerbody inlet reduced the stall margin by a small amount. This difference between the two different inlets could be attributed to the difference in radial flow distortion created by the inlets.

5.0 Conclusions

Results of this program lead to the following conclusions.

1) A sonic inlet can be designed to produce inlet noise reduction in the range of 25 to 30 PNdb with less than 3% inlet recovery loss on approach and less than 1% in takeoff and cruise operation. Inlet flow distortion effects remain within acceptable limits for satisfactory engine/inlet compatibility. It has been demonstrated that, using the latest available design technology, sonic inlets can be designed within an L/D limit of unity consistent with the objective of this program.

2) Inlet noise reduction in excess of 40 PNdb is possible by use of sonic inlets with increased performance loss.

3) Single-passage inlets are superior in terms of inlet recovery to multipassage inlets. The trend in terms of total pressure distortion is reversed.

4) Increasing the inlet throat Mach number from about 0.5 to 1.0 resulted in increasing noise attenuation, with most of the effect occurring above Mach 0.7.

5) The rate of decay of the blade passing tone was found to be less on the centerline than near the outer wall for a contracting cowl wall model. This was attributed to the radial Mach number gradient in the throat region.

6) The multiple pure tone (buzz-saw) noise did not propagate through the inlet throat plane at high throat Mach numbers.

7) No evidence was found of noise leakage through the inlet throat boundary layer.

8) No significant noise generation by the sonic plane was observed.

9) Spectral comparison of the noise for the baseline and sonic inlet indicated that some dependence of attenuation effectiveness existed as a function of frequency. Pure tones of high frequency and high-frequency broadband noise were more effectively attenuated than low-frequency noise.

10) Noise directivity measurements taken in the acoustic chamber with adequate inlet sidewall insulation and other noise sources minimized indicated that the noise was effectively reduced at all angles in the forward arc.

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Static Aeroelasticity and the Flying Wing

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This paper demonstrates, by means of elementary examples, certain features of flying wing static aeroelasticity. Prominent among these are the influence of trimming control surfaces on wing divergence. Models are formulated using elementary beam-rod differential equations and aerodynamic strip theory. Divergence of an unswept wing, rolling freely about a pinned shaft, is discussed. The resulting torsional divergence speed is nearly three times that of a nonrolling wing of half the span, clamped at the root. If the rolling velocity of the full wing is trimmed by elevons, antisymmetrical divergence may occur at a speed lower than the classical torsional divergence speed. The case of a wing trimmed in roll by 30% Fowler flaps is presented. A similar elementary analysis of an oblique or yawed wing free to roll about a pinned shaft parallel to an airstream and trimmed in roll is also presented. Considering only wing bending flexibility, it is found, through the application of Galerkin's method, that the application of ailerons to trim roll results in a divergence *q* five times that of a similar symmetrical sweptforward wing clamped at the center. Finally, a similar freely flying yawed wing is trimmed in pitch and roll by elevons such that the total lift equals the aircraft weight. It is found that the divergence *q* which occurs in this case is nearly twelve times that of the clamped symmetrical sweptforward wing.

Introduction

VEHICLES based on the flying wing arrangement, or something closely approximating it, have engaged in manned, powered flight for at least as far back as 1910. Gibbs-Smith[1] describes the Dunne No. 5 Tailless of that date, and his historical review also encompasses the Hill Pterodactyl of 1926, the Lippisch Tailless Research Monoplane of 1931 and others. More familiar is the series of scaled and experimental aircraft which culminated in the Northrop XB-35 (Fig. 1) and XB-49; details on them can be found, for instance, in Janes' for 1948.[2] At about the same time, in Great Britain, Armstrong Whitworth Aircraft Limited were testing the A.W. 52 series.[3]

Although the objects of great enthusiasm by their protagonists, none of the early flying wings seems to have evolved into a successful operational aircraft. There are several reasons for their failures, but two appear most prominent. The first was the extreme difficulty of achieving satisfactory unaugmented handling qualities, control and dynamic stability, especially in the lateral-directional modes. An excellent summary of the state of knowledge about this problem during World War II is given by Donlan.[4] The second is that they simply were not big enough. As observed also by Donlan[4] flying wings of that era allowed inadequate provision for payload and only "for large airplanes having spans of 150–500 ft, the volume of the wing alone may be sufficient to enclose bulk or weight of an appreciable magnitude even with the thin wing sections required for high speed."

It is worth mentioning that the 1940's generation of tailless vehicles attained what might be described as moderate subsonic performance. Thus Janes[2] cites a speed of 350 mph for the rocket-powered Northrop MX-324, whereas Murray[3] estimates a maximum of 500 mph at sea level (including compressibility effects) for the A. W. 52 (E9/44) with two Nene 1 turbojets. The corresponding range of dynamic pressures, coupled with the fact that their struc-

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