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AEROCOUSTIC PERFORMANCE COMPARISON OF CORE ENGINE NOISE SUPPRESSORS ON NASA QUIET ENGINE "C"

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Aero-Acoustic Performance Comparison of Core Engine Noise Suppressors on NASA Quiet Engine "C"

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Abstract

The purpose of the experimental program reported herein was to evaluate and compare the relative aero-acoustic effectiveness of two core engine suppressors, a contractor-designed suppressor delivered with the Quiet Engine, and a NASA-designed suppressor, designed and built subsequently. The NASA suppressor was tested with and without a splitter making a total of three configurations being reported in addition to the baseline hardware case. The aero-dynamic results are presented in terms of tailpipe pressure loss, corrected net thrust, and corrected specific fuel consumption as functions of engine power setting. The acoustic results are divided into duct and far-field acoustic data. The NASA-designed core suppressor did the better job of suppressing aft end noise, but the splitter associated with it caused a significant engine performance penalty. The NASA core suppressor without the splitter suppressed most of the core noise without any engine performance penalty.

Introduction

To achieve core engine noise suppression for the approach flight condition of a typical C-17 mission, many types of suppressors have been proposed. The purpose of the experimental program reported herein was to evaluate and compare the relative aero-acoustic effectiveness of two core engine suppressors, a contractor-designed suppressor delivered with the Quiet Engine, and a NASA-designed suppressor designed and built subsequently. The NASA suppressor was tested with and without a splitter making a total of three configurations being reported in addition to the baseline hardware case.

As a part of the NASA Quiet Engine contractor program, a core suppressor was designed and tested on Quiet Engine "C". The design was aimed mainly at suppression of turbine noise at approach power and its attenuation characteristic was centered at a frequency of 5000 Hertz. The mechanical design was selected in consideration of types of acoustic materials available, methods of fabrication, and duty cycles required. This information was presented in Refs. 1 and 2. The performance of this configuration was reported in Refs. 2 and 3 and showed that the totally suppressed engine was aft-end noise dominated over the range of engine power settings.

In an effort to provide a greater suppression of the core noise, a second suppressor was designed having upstream and downstream sections, each designed for different frequencies. Single degree of freedom (DOF) treatment was used in each section. The second suppressor incorporated a tailpipe splitter to improve the suppression at a frequency of 6300 Hz centered around the turbine tones. The second section of treatment was designed for 2500 Hz (much lower than the lowest frequency turbine tones), to suppress other broadband noise emanating from the core. The splitter was designed so that it could be removed, resulting in a third configuration to be evaluated.

The three configurations, the contractor design and the NASA design with and without the splitter, were evaluated aero-acoustically on Quiet Engine "C" at the Engine Noise Test Facility of NASA Lewis Research Center. The core suppressor configurations were tested over a range of engine power settings from approach to takeoff. Fan noise was suppressed forward by a sonic inlet described in Ref. 4 in its takeoff configuration, and aft by a massive aft fan suppressor(5) so that the aft core noise could be measured. A co-axial fan and core nozzle arrangement was installed to minimize the turbulent mixing zone between the fan and core jets resulting in a reduction in the jet exhaust sound scattering(6) and also to minimize "haystacking"(6). This permitted the better discrimination of the various turbine tones and consequently allowed the noise reduction at the core suppressor to be measured more accurately in the far field. Aerodynamic performance tests as well as both far-field and duct acoustic tests were performed and results are compared herein to baseline test results of the engine with no core acoustic treatment. Results from the acoustic suppression prediction method of Ref. 7 are compared with the experimental results.

Facility Description

The test program was performed at the Engine Noise Test Facility located at Lewis Research Center adjacent to, but sufficiently far from the Flight Research Building so that accurate acoustic measurements could be obtained. The facility is shown schematically in Fig. 1.

The 17 far-field microphones were at the same height as the engine centerline, 5.0 m (16.4 ft) on a 45.7-m (150-ft) radius spaced every 10° from the inlet axis to 160°. The reflecting plane was hard pavement. Ground microphones were installed at 110°, 120°, 130°, and 140° from the inlet axis.

A photograph showing the installation of Quiet Engine "C" at the Engine Noise Test Facility is presented in Fig. 2. Engine operation was controlled from the Flight Research Building where the noise instrumentation and analysis equipment were located.

Engine Description

The NASA Quiet Engine "C", a low noise technology turbofan demonstrator, was designed, built, and acoustically evaluated under the NASA/GT Experimental Quiet Engine Program. The 97,300-newton (22,000-lb) thrust class turbofan consisted of a newly developed, high tip speed, single-stage fan. It was designed for the altitude cruise condition with a corrected tip speed of 478 m/sec (1550 ft/sec) at a bypass pressure ratio of 1.6, and with a corrected fan flow of 415 kg/sec (915 lb/sec). The
fan had 26 unshrouded rotor blades and 60 outlet guide vanes. Further details are presented in Ref. 5.

Core Suppressor Design

A schematic of the contractor-designed core suppressor (hereafter referred to as configuration 1) is presented in Fig. 3. The acoustic treatment was of the SDOF type with a 7% porosity face plate and a total thickness of 0.64 cm (0.25 in.). The face plate had a thickness of 0.008 cm (0.003 in.).

A schematic of the NASA designed core suppressor (hereafter referred to as configuration 2) is presented in Fig. 4(a) and a photograph is shown in Fig. 4(b). The instrumentation shown is discussed in the "Experimental Methods" section of this report. Two sections of treatment with two thicknesses were provided with a splitter in the upstream, high frequency section. The splitter reduced the frequency parameter, $fH/c$, and increased the average length to height ratio, $L/H$, both of which increased the effectiveness of the treatment in that section. The splitter was removable so that the performance penalty/noise suppression tradeoff could be measured. It was realized that the splitter used in this design might cause more pressure drop than would be desirable necessitating a new tailpipe design. However, a new tailpipe design was not considered economical for this demonstration test. The configuration with the splitter removed will hereafter be referred to as configuration 3.

The design goal for configuration 2 was determined from the NOY-weighted baseline spectrum data at engine approach power obtained by the contractor in his tests of Engine "C". These data are presented in Fig. 5. The assumption was made that all the NOY was measured at 1200 ft in the fan field above a level of 60 dB was emanating from the core. By suppressing the NOY-weighted spectrum down to a level of about 80 dB, it was hoped that essentially all of the turbine and other core noise above a frequency of 500 Hz could be effectively eliminated. With this goal in mind, many acoustic liner designs were analyzed in an effort to reach the assumption required. The acoustic liner design method described in Ref. 7 along with the design constraints of not extending the tailpipe resulted in a design which incorporated a splitter in the high frequency section. Another constraint imposed by the hardware can be seen in Fig. 4(a). The downstream end of the low frequency section had no treatment on the outside wall because no room existed for the thick treatment. The result was that the last 19 cm (7.5 in.) of treatment on the tailcone had a very large average effective passage height of 58 cm (23 in.), which was twice the average passage height of 29 cm. For the high frequency treatment, an average length of only 21.9 cm (8.6 in.) in the outer passage had like treatment opposite. With the splitter removed, the average $L/H$ value for the high frequency section of treatment decreased from 2.42 to 1.09.

The predicted acoustic suppression of the configuration using the method of Ref. 7, along with the design goal for configuration 2 are presented in Fig. 6. The design center frequency of suppression for configuration 1 was 5000 Hz. The two types of treatment employed in configurations 2 and 3 had design center frequencies of 2500 and 6300 Hz. The effect of removing the splitter causes a drastic reduction in predicted suppression from 20.3 to 5.6 dB at a frequency of 6300 Hz. The predicted reduction at 2500 Hz also drops from 10 to 6 dB. This is because the "skirt" of off-peak attenuation of the high frequency treatment also drops when the peak attenuation, $S/B$ (at 6300 Hz). The off-peak attenuation of "skirt" of configuration 1 has a flatter slope than the other configurations. This characteristic is caused by the prediction program which treats configuration 1 as an "overdamped" liner. Overdamping occurs when the resistance ratio ($R/R_{eq}$) exceeds unity.

The aerodynamic design of the splitter used in configuration 2 was carefully considered to minimize any additional tailpipe pressure loss. A potential flow analysis was done and a co-planar section of the loading edge of the splitter was designed to minimize the possibility of separation over the range of engine flow conditions. Additional design details are presented in Appendix A.

The mechanical design of configuration 2 also entailed some problems in consideration of the effects of the engine environment on the suppressor structure. These are treated in some detail in Appendix B.

Fully Suppressed Engine Configuration

The fully suppressed engine configuration employed in these tests is shown schematically in Fig. 7. Details of the acoustic treatment in the inlet and aft fan ducts are also shown. The acoustic performance of this configuration was reported in Ref. 4. Shown also in Fig. 7 is the co-planar nozzle extension which was used for the majority of the tests.

Experimental Methods

Aerodynamic and acoustic data were obtained over a range of corrected fan speeds from 56 to 93 percent of design for all three core suppressor configurations and the hardware baseline configuration.

The acoustic instrumentation and data recording system had a flat response over the frequency range of interest (20 to 20,000 Hz). The acoustic signals were FM recorded from all channels simultaneously on magnetic tape. Each of the three samples for a given corrected fan speed was reduced separately by using a 1/3-octave-band analyzer. The resulting sound pressure levels were arithmetically averaged, adjusted to standard day atmospheric conditions; and side-line perceived noise levels were calculated using the standardized procedures presented in Ref. 8. The narrow band data reported herein are given as measured without any correction.

Aerodynamic instrumentation was located in planes identified in Fig. 8. The total pressures and temperatures at the turbine outlet and the core nozzle outlet stations were located on centers of equal flow areas. The spacing of the total pressure measured at the splitter station is indicated in Fig. 9. The pressures, temperatures, and other outputs were received as microvolt signals by the facility data system, digitized and transmitted from a local minicomputer to a remote data collector at the laboratory main computer facility.
Aerodynamic performance

Total annulus area measured from the outside to the inside pressure ratios are about 1.16 and the splitter takes off power, the inside and outside package average -

tan of about 1.06 to 1.02 at the centerline. The inside pressure ratio peaks at the center and falls off at the turbine outlet station for both approach and takeoff power settings are presented in Fig. 10. Total-

A comparison of the predicted noise reduction of the core suppressors to the duct results and the far-field results is then discussed. Finally, time histories of tone corrected levels (%F) are shown for the configurations at the approach flight condition.

A comparison of corrected net thrust as a function of corrected fan speed is presented in Fig. 12 for the baseline and the three suppressor configurations. The baseline and configurations 1 and 3 data fall on essentially one line. The penalty for the splitter installation of configuration 2 is readily apparent. At takeoff rating, the penalty amounts to a decrement in corrected net thrust of about 4 percent. A more significant concern, however, is the effect of configuration changes on specific fuel consumption. This effect is illustrated next in Fig. 13. Corrected specific fuel consumption (SFC) is plotted as a function of corrected net thrust. Again the penalty for the splitter installation is apparent. The penalty in increased SFC at takeoff amounts to about 5 percent.

The effects of the splitter installation on Quiet Engine "C" performance is somewhat amplified by the use of the sonic inlet which is used to minimize front radiated fan noise. The somewhat lower inlet recovery produced by the sonic inlet in comparison to a conventional inlet[1] causes an increase in corrected SFC as corrected thrust is increased beyond 75 percent of takeoff power (fig. 13), and also causes the decrease in slope of corrected thrust as a function of corrected fan speed (fig. 12). It is estimated that at takeoff power the penalty in corrected SFC would be reduced from about 5 percent to 4-1/2 percent if a conventional inlet has been utilized in this investigation.

The losses associated with the splitter could be minimized in any new engine development by designing a new tailpipe to accommodate the splitter. The splitter blocks approximately 9 percent of the flow area and raises the surface Mach number around the nose of the splitter at takeoff to about 0.57. A somewhat smaller turbine outlet Mach number and less swirl (both, of course, functions of engine design) would also help reduce the losses reported herein. Placement of the splitter downstream in a larger flow cross-section area would also reduce
the aerodynamic losses. However, this would also result in a lower noise reduction, according to the acoustic design prediction method.

Acoustic Performance

Duct sound power level reductions. - The duct microphone data which was taken with the baseline and all three suppressor configurations is summarized in Fig. 14. Sound power level reduction in dB is shown as a function of 1/3-octave-band frequency for the three configurations for both approach (Fig. 14(a)) and takeoff (Fig. 14(b)) power settings. As expected, at the approach power setting, configuration 2 suppresses core noise over a wide band of frequency. The 2500-Hz 1/3-octave band is suppressed 10 dB and the 6500 Hz band is suppressed 15.5 dB. Even the 10000-Hz band is suppressed by 5 dB and the 20000 Hz band by 4.5 dB. In contrast with this performance, configurations 1 and 3 have peak suppression values of about 7 dB at the 4000 Hz band. As would be expected, of the two configurations, configuration 1 has better suppression in the lower frequency bands and configuration 2 has better suppression in the higher frequency bands. At takeoff power setting, configuration 3 also produces reasonable suppression over the entire bandwidth presented, with peak suppression of 12.5 at 10000 Hz. Configurations 1 and 3 also show peak suppressions at 5000 Hz of 10.5 and 6.5 dB, respectively. The PNL reductions achieved by the three configurations at takeoff are lower than those at approach. The acoustic design point was at approach power so that the flow conditions are non-optimum at takeoff. A second reason for the lower PNL reduction at takeoff, however, is that a jet noise floor has been reached. This will be discussed in some detail later in connection with the far-field acoustic results.

Farfield sound pressure level reductions. - The far-field PNL directivity data on a sideline are presented in Figs. 15 and 16 for approach and takeoff power settings, respectively. The approach power data are shown on a 114-m (375-ft) sideline as a function of angle from the engine inlet. The baseline noise peaks at angles of 50° and 120° from the inlet. Configuration 2 reduces the baseline noise by about 5.0 PNdB at 50° and 7.5 PNdB at 120°. Configuration 3 reduces the noise around the same amount as configuration 2 over the entire range of angles. Configuration 1 reduces aft quadrant noise about half as well as the other two configurations but does little in the front quadrant. The behavior of configurations 2 and 3 makes one suspect that a noise floor has been reached. At any rate, the engine equipped with either configuration 2 or 3 is no longer aft-noise dominated at approach power.

The PNL data at takeoff power are shown in Fig. 16 for a 205-m sideline as a function of angle from the engine inlet. In addition, jet noise calculated by the method of Ref. 9 is shown for comparison. The reductions in noise by all three configurations are small in comparison to the reductions at approach power. The aft peak of the baseline configuration noise at 120° is reduced by about 2.5 dB by configurations 2 and 3. The reductions at the other angles are also small. The reason is obvious when the jet noise floor is taken into account.

Shown in Fig. 17 is a comparison of 120° aft angle 305-m sideline PNL values for the baseline and three suppressor configurations as a function of corrected fan speed. The calculated jet noise is shown for reference. The estimated aft fan noise floor is also shown. The trends at approach and takeoff power settings have already been discussed in connection with Figs. 15 and 16. The trends at approach and takeoff are confirmed by the data points at 55 and 75 percent corrected design fan speed. Configuration 3 provides about the same amount of suppression as configuration 2 over the range of engine power settings tested. It appears that aft fan noise and jet noise are limiting the apparent reduction achieved by configuration 2 at takeoff and that aft fan noise alone constitutes a floor at approach.

In order to appraise the suppressor configurations on the basis of suppression of 1/3-octave-band noise, Fig. 18 is presented. These sound pressure level (SPL) data are for the 120° microphone at approach power setting extrapolated to a 114-m sideline. At frequencies between 1000 and 4000 Hz, configurations 2 and 3 suppress aft core noise equally well. At frequencies above 4000 Hz configuration 2 shows the greatest suppression. Configuration 1 does not accomplish much suppression at all frequencies in comparison to the others.

To further detail the comparison of these suppressors at approach power setting, far-field 120° microphone narrow band SPL data are presented in Fig. 19 over a range of 10,000 Hz. Shown in Fig. 19(a) is the baseline spectrum. The first and second stage low pressure (LP) turbine blade passing frequency (ST7) tones are readily apparent at frequencies of 6500 and 7500 Hz. This spectrum is repeated on Figs. 19(b), (c), and (d) so that the suppression of the configurations over the range of frequencies (10,000 Hz) can be noted. Displayed in Fig. 19(b) is the spectrum from configuration 2 showing the effect of nozzle configuration on farfield noise. As was reported in Ref. 3 and as predicted in Ref. 6, the spectrum of the standard fan nozzle (dashed) shows that "haystacking" did occur between the fan harmonic tones at 5900 Hz and the turbine tones emerging from the core. On the other hand, the coplanar nozzle configuration narrow band spectrum does contain clearly defined tones. The PNL for both configurations added up to about the same level, however. The second stage turbine tone for the coplanar nozzle is of higher amplitude than the tone for the standard fan nozzle, as reported in Ref. 3. It is reasonable that the turbine tones emanating from the core nozzle are modulated by the relatively larger eddies caused by the flow around the core nozzle when the fan nozzle is short compared to very small eddies with a coplanar nozzle.

A comparison of the splitterless configurations is presented in Fig. 19(c). Configuration 3 suppresses the noise better at frequencies between 1500 and 5900 and also suppresses the tones somewhat more than configuration 1. The frequencies between 5900 and 5400 Hz is the only range where configuration 1 is better.

A comparison of configurations 2 and 3 is presented in Fig. 19(d) to show the effect of the splitter. The two turbine tones were reduced by about 5 dB by the splitter. The SRL over the frequency range between 4500 and 7400 shows greater suppression by configuration 2. A close examination of the traces of configuration 2 shows that the fan RPM and first two harmonics are starting to appear in the narrow band data at a level above all other noise.
except for the second stage turbine tone. This would suggest that, since the sound power contained in that tone on a PME basis is very small, the noise floor is caused by the fan. In addition, the reduction in broadband noise at frequencies above 4500 Hz by configuration 2 means little. Referring to Fig. 17 at low power settings, the greater suppression achieved by configuration 2 cannot be measured on a PME basis.

At takeoff power setting, the picture is similar. Presented in Fig. 20 is a broadband plot of SSW. The fan BPF and first harmonic tones are above the LP turbine first stage BPF tone. In addition, the low frequency jet noise, determined according to Ref. 9, is also starting to become dominant. The noise floor has been reached and the good suppression at frequencies beyond 6000 Hz means little in terms of PME as shown in Fig. 17 at takeoff power setting. Since the PME directivity shown in Fig. 15 is not dominant, it is obvious that jet noise and aft fan noise are causing the floor.

Comparison with predictions. — In order to consider the accuracy of the prediction method, the sound pressure level reductions measured in the core duct and in the far field at 200° angle from the inlet are presented as a function of 1/3-octave-band frequency in Fig. 21 for all three suppressor configurations at the approach power design point. Configuration 1 sound pressure level reduction shown in Fig. 21(a) shows relatively good agreement between the far-field and in-duct measurements. The pure tone reductions taken from narrowbands of Fig. 15 are also shown by the bashed symbols. The prediction overestimated the reduction over the frequency bandwidth. The discussion in the "Tone Suppressor Design" section of the report should be recalled at this point. The fact that the configuration 1 liner is treated as "overdamped" according to the prediction method gives the broad "skirt" of the prediction curve. Perhaps, since this prediction is based on semiempiricism, these empirical data should be considered for a possible modification of the design method for the "overdamped" case.

Shown in Fig. 21(b) are the data for configuration 2. In addition, an estimate of the aft fan noise floor is based on Fig. 15(b). Here, relatively good agreement exists between both kinds of measurements and the prediction method. Even the peak suppression predicted at 6500 Hz is confirmed by the far-field data. The predicted peak suppression at 2600 Hz is not confirmed, due to the aforementioned fan noise floor at this frequency (Fig. 15(b)).

Presented in Fig. 21(c) are the data for configuration 3. The same aft fan noise floor from Fig. 21(b) is also shown. The agreement between the measurements and the prediction is not as good as that shown in Fig. 21(b). The far-field suppression data, in fact, exceed the predicted suppression results and the measured in-duct suppression in the frequency range from 1600 to 4000 Hz and in the 1/3-octave bands of 6300 and 8000 Hz. A conclusion from the data presented in Fig. 21 is that the prediction method of Ref. 7 is a useful tool, but that improvements for off-optimum (overdamped) liner cases should be considered.

Effect on flyover noise. — A comparison of the approach time histories of tone corrected perceived noise levels (PME's), of the various configurations are presented in Fig. 22. These data were calculated from the far-field measured noise data assuming a four-engine B707/DC8 type of approach with an altitude of 114 m (375 ft) above the observer's station. Effective Perceived Noise Levels (EPNL's) are also presented in terms of Effective Perceived Noise decibels (EPNLdB's).

Recall that the jet noise does not constitute a noise floor at this power setting, and therefore, no relative velocity corrections were needed in the calculation. Very little reduction in EPNL and therefore, EPNLdB was accomplished by configuration 1. Configurations 2 and 3 both show good reductions in EPNL equivalent to reductions of 2 and 3 EPNLdB. Here, the fan noise floor apparently limits the reduction of EPNLdB. Even though configuration 2 exhibited greater noise reduction than configuration 3 at frequencies above 4000 Hz (Fig. 21), the advantage does not influence PME or EPML values noticeably. Further reduction in fan noise would be required in order to perceive any small advantage in configuration 2.

Summary of Important Conclusions

1. The splitter associated with configuration 2 resulted in a penalty of about 4 percent in thrust at rated takeoff power. The associated penalty in specific fuel consumption was about 6 percent. A conventional inlet could reduce the SFC loss to 4 percent. For application to any new engine development, the flow Mach number in the tailpipe would have to be decreased to minimize these losses.

2. At the maximum noise angle of 120° on a 114-m (375-ft) sideline at approach power, a reduction of 3.0 EPNLdB was shown by configuration 1. A reduction of 7.5 EPNLdB was credited to configuration 2, and 8.0 EPNLdB to configuration 3.

3. The turbine tones were suppressed somewhat by configurations 1 and 3. Configuration 2, as it should have, did the best job in suppressing the turbine tones.

4. The use of a cofigurable nozzle allows more accurate measurements of the 
6. In the comparison of 1/3-octave-band predicted noise reduction with the far-field data at the maximum noise angle of 120°, configuration 3 exceeded the design goal and, in fact, produced almost as much acoustic suppression on a PME basis as did configuration 2, without any aerodynamic performance loss. Configuration 2 did give substantially larger suppression of the higher frequency sound, including turbine tones as it was designed to do; however, the reduction at these frequencies did not contribute appreciably to reducing the PME level due to the encroachment of the fan noise floor.

7. Time histories of approach flyover noise indicate that a reduction of 0.5 EPNLdB was achieved by configuration 1. Configurations 2 and 3 have reductions of about 3 EPNLdB.
Appendix A

Details of Design Method

The aerodynamic design of the tailpipe with the splitter installation was the result of an iterative analysis, using the streamtube computer program (40). The design objectives were to conform to the geometric constraints and to minimize loads and losses associated with the splitter.

The estimated hot no-slip contour without the splitter was first analyzed with "STC" (Streamtube Calculation Program). The program utilized inlet total pressure and temperature profiles. A splitter was then placed in the stream so that it was aligned with the no-splitter streamline pattern. Tentative leading and trailing contours were placed on the center section so that the leading and trailing points were along the same streamline. This was done to avoid having to iterate in STC to find the correct flow split and to minimize axial forces. Several contours were tried in obtaining a smooth Mach number distribution. The NASA-1/ellipse contour was finally chosen.

Using the Mach number distribution from STC, the boundary layer displacement thickness was calculated and added to the hard contour. This new contour was then run through STC to obtain final Mach distribution. These distributions were then used in a boundary layer program to calculate friction drag and separation function (modified Stratford separation function (11-13)). Separation was definitely not indicated whether on the upper leading edge or the trailing contour. The critical value of F(x) was just being reached at the trailing edge.

Total pressure loss due to friction was obtained by adjusting drag values calculated by the boundary layer program in order to account for the increased drag of acoustically treated surfaces. The procedure was to increase drag calculated in the program by the ratio 0.375 x (Atreated/Atotal) for each flow surface. Losses due to support struts were determined by the method of Ref. 14. Eight NASA-0015 struts were assumed. Calculated losses (Kg/psf) at location = 0.60 x 0.840 are summarized in the following table

<table>
<thead>
<tr>
<th>Area at station 8.0, cm²</th>
<th>4194</th>
<th>5484</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wall and splitter friction</td>
<td>0.680</td>
<td>0.840</td>
</tr>
<tr>
<td>Strut profile and friction</td>
<td>0.012</td>
<td>0.017</td>
</tr>
<tr>
<td>Strut interference</td>
<td>0.016</td>
<td>0.020</td>
</tr>
<tr>
<td>Total</td>
<td>0.710</td>
<td>0.862</td>
</tr>
<tr>
<td>Add 10% for turns, gaps, etc.</td>
<td>.76</td>
<td>.97</td>
</tr>
</tbody>
</table>

It should be noted that all of the above results are, for all practical purposes, independent of strut location.

Appendix B

Engine Hardware Design and Fabrication

The engine hardware utilized in this program consisted of three acoustic configurations plus a baseline hardwall configuration. The baseline hardwall configuration and configuration 1 have been previously reported in Ref. 42 and will not be discussed here. Configurations 2 and 3 were essentially the same wall hardware, with and without the acoustic splitter. Early in the design phase it was clear that the acoustic splitter was the most difficult part to be designed. Also, the previous experience in generating configuration 1 hardware could be applied to the new design.

Basically the walls consisted of single layer honeycomb sandwich conical structures furnace brazed into a single conical piece. Each piece was bolted together with sufficient body-bound (interference fit) bolts to ensure structural rigidity. Complete structural, thermal, and vibratory analysis was performed. The analysis considered gaseous loading, maneuver loading, and stress due to constrictions through the honeycomb thickness. Of the three, thermal gradients was the primary stress contributor. Finally, a local buckling failure analysis was performed.

For the acoustic splitter, analysis was performed with special emphasis on the mounting struts attaching the acoustic splitter to the wall. Eight equally spaced radial airfoil shaped struts were selected with the struts tilted forward from the inner circumference to the leading edge of the acoustic splitter as shown in Fig. 4. This strut configuration was selected because it minimized thermal stresses without complicating the fabrication procedure. The aerodynamic leading edge of the splitter was hollowed out to reduce weight and thermal stress. Also a significant weight reduction was achieved by maintaining honeycomb structure to the trailing edge.

The entire structure, including face sheets and the thin-film resistance-welded honeycomb core, was fabricated from Inconel 625 alloy. This nickel-base alloy has very good high temperature strength in the 1000°F range and has excellent formability. The brazing alloy, AG 4777, was selected as the high temperature (1000°F range) brazing alloy because of its compatibility with Inconel 625, its ability to be welded after brazing, and its ability to bridge small gaps as well as create reasonably large fillets.

The technology for forming, assembling, and brazing complex high temperature alloy honeycomb sandwich structures is well established in several companies within the United States. Each company has different techniques, varies its approach, but the outcome is the same — a continuous, homogeneous sandwich structure. The most significant aspect of producing high quality brazed honeycomb sandwich structures was the utilization of manufacturing procedures which paid attention to cleanliness as well as precision process control in the production of each component. This was followed by careful inspection of the brazing operation by X-ray techniques and "coin-tapping" to ensure that there were no defective brazed areas. Finally, with each honeycomb sandwich to be furnace brazed, an identical sandwich test specimen was included for fibrous tensile tests to assure that the brazing cycle produced a quality assembly.

References


Figure 1. - Engine Noise Test Facility plot plan showing thrust stand, microphone array, control and noise instrumentation rooms.

Figure 2. - Engine noise test facility at Lewis Research Center. View of Quiet Engine "C" installation showing the sonic inlet used to quiet inlet fan noise.
Figure 4 - Contractor-designed core suppressor for NASA Quiet Engine "C". (Configuration 1.)

NOTE: ALL DIMENSIONS IN cm UNLESS OTHERWISE NOTED.

Both treatments have face plate thicknesses of 0.08 cm and hole sizes of 0.19 cm in diameter.
(b) PHOTOGRAPH OF CONFIGURATION 2 SHOWING SPLITTER AND AERODYNAMIC INSTRUMENTATION.

Figure 4. Concluded.

Figure 5. NOY-weighted 1/3-octave-band spectrum far-field baseline data showing noise reduction goal. 120P microphone extrapolated to 114-m sideline at approach power.
Figure 6. - Comparison of design suppression predictions for all configurations at peak noise aft angle of 120° and approach power setting using method of reference 7.

Figure 7. - Quiet Engine "C", cross section of fully suppressed configuration showing core suppressor configuration 1 installed.
Figure 8. - Core suppressor instrumentation schematic.

Figure 9. - Total-to-static-pressure ratio at turbine outlet station.
Figure 10. - Total-to-static-pressure ratio behind splitter of configuration 2.

Figure 11. - Comparison of engine parameters for configurations with and without splitter over range of corrected core engine speeds.
Figure 12. - Comparison of variation of corrected net thrust with corrected fan speed for configurations with and without core splitters.

Figure 13. - Comparison of variation of corrected specific fuel consumption with corrected net thrust for configurations with and without splitter.
Figure 14. - Sound power level reduction over treated surface of suppressor configurations compared to baseline (hard surface) configuration. Induct acoustic probe data.

Figure 15. - Perceived noise directivities at approach power for all configurations.
Figure 16. - Perceived noise level directivities at take-off power for all configurations.

Figure 17. - Comparison of perceived noise level at peak noise aft angle of 120° over range of engine power settings.
Figure 18. - Comparison of 1/3-octave spectra for baseline and suppressed configurations. Far-field 120° microphone data. Approach power. Data corrected to standard day of 59° F and 70 percent relative humidity.

Figure 19. - Approach power narrow-band spectra at peak noise aft angle of 120°; 45.7 m; filter bandwidth, 30 Hz.
FIRST STAGE -- BASELINE LP TURBINE CONFIGURATION 3
90 ------- CONFIGURATION 1 SECOND STAGE LP TURBINE

(c) COMPARISON OF CONFIGURATIONS 1 AND 3.

(d) EFFECT OF SPLINTER.
Figure 19. - Concluded.

Pressure level, dB (re 2 x 10^-5 Pa)

FIRST STAGE LP TURBINE
SECOND STAGE LP TURBINE

Pressure level, dB (re 2 x 10^-5 Pa)

(d) EFFECT OF SPLINTER.
Figure 19. - Concluded.

Figure 20. - Take-off power narrow band spectra. Configuration 2 with coplanar fan nozzle. 120° microphone data at 45.7 m. Filter bandwidth, 30 Hz.
Figure 21. - Comparison of predicted, far-field (120°) and in-duct sound level reduction characteristics. Approach power design point.

Figure 22. - Approach time histories (flyover noise) for baseline and suppressor configurations of tone corrected PNL's.