NASA Technical Memorandum 82752

A Shock Wave Approach to the Noise of Supersonic Propellers

James H. Dittmar and Edward J. Rice
Lewis Research Center
Cleveland, Ohio

December 1981

NASA

A SHOCK WAVE APPROACH TO THE NOISE OF SUPERSONIC PROPELLERS

James H. Dittmar and Edward J. Rice

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

SUMMARY

One of the possible propulsive systems for a future energy efficient airplane is a high tip speed turboprop. When the turboprop airplane is at cruise, the combination of the airplane forward speed and the propeller rotational speed results in supersonic helical velocities over the outer portions of the propeller blades. As a result of these supersonic blade sections and their associated shock waves, these propellers may create a cabin noise problem for the airplane. To model this propeller noise, the pressure ratio across the shock at the propeller tip was calculated and compared with noise data from three propellers. At helical tip Mach numbers over 1.0, using only the tip shock wave, the model gave a fairly good prediction of the noise for a straight bladed propeller and for a propeller swept for aerodynamic purposes. However for another propeller, which was highly swept and designed to have noise cancellations from the inboard propeller sections, the shock strength from the tip overpredicted the noise. In general the good agreement indicates that shock theory is a viable method for predicting the noise from these supersonic propellers but that the shock strengths from all the blade sections need to be properly included.

E-1068

INTRODUCTION

One of the possible propulsive systems for a future energy efficient airplane is a high tip speed turboprop. When the turboprop airplane is at cruise, the combination of the airplane forward speed and the propeller rotational speed results in supersonic helical velocities over the outer portions of the propeller blades. As a result of these supersonic blade sections and their associated shock waves, these propellers may create a cabin noise problem for the airplane.

The noise from three model propellers of this type was obtained by testing in the NASA Lewis 8-by-6 Foot Wind Tunnel and was reported in references 1 and 2. A photograph of the three individual blades is shown in figure 1(a) and a photograph of one of the eight-bladed propellers is shown in figure 1(b). The three propellers have been designated SR-2, SR-1M and SR-3. The SR-2 blade is similar to a conventional straight propeller blade but with a long chord and a relatively low thickness-to-chord ratio at the tip. The SR-1M blade has some sweep built into the outboard area. This sweep was primarily aerodynamic for the purpose of reducing losses. The mid-chord tip sweep, when measured on the helix formed by the advancing blade, is 23°. The SR-3 blade was an attempt to incorporate sweep both for aerodynamics and noise control. The mid-chord tip sweep of this blade was 34°. A comparative listing of the three propellers is found in table I.

A number of theoretical noise prediction models have been developed for these types of propellers. Some of these can be found in references 3 to 7. These theories are based upon linear acoustic theory and utilize the Ffowcs Williams-Hawkings equation of reference 8. One of these theories, that of Farassat, was used to predict the noise of the three tested propellers in reference 9 and a simplified model prediction based on a fly-over shock wave correlation was presented in reference 10. The trends of the theory with helical tip Mach number, Mh_t, are not the same as the data for Mach numbers over 1.0. This is shown in figure 2, taken from reference 9, which has the predictions and data for the three propellers. As can be observed from this figure, the predictions for the three propellers increase for helical tip Mach numbers over 1.0 while the data level off. The leveling off of the noise of supersonic helical tip speed propellers was also shown in reference 11. The difference between data and theory has also been observed for helicopter noise as indicated in figure 14 of reference 12, where the rotor data levels off but the linear theory noise predictions continue to increase with Mach number. These noise prediction models, based upon linear acoustic theory, apparently do not accurately model the noise in the region where shock waves are formed. Therefore an evaluation of shock wave behavior is undertaken in this report. The intent is to investigate the shock wave behavior with increasing Mach number and to relate this to the measured propeller noise.

SHOCK WAVE

Shock Wave Formation

In order to investigate the shock related noise from this type of propeller blade, it is necessary to have a qualitative understanding of the formation of the shocks. A discussion of the process can be found in reference 13, pp. 289-291, and the following is primarily a paraphrasing of this discussion. The discussion concerns the flow around a thin airfoil, at a small angle of attack, in a transonic stream. The figures of reference 13 are redrawn here as figure 3. Figure 3, part (a) is for a free-stream Mach number, M_0, of 0.75, part (b) is for $M_0 = 0.81$, (c) for 0.89, (d) for 0.98 and (e) for 1.4.

In part (a), $M_0 = 0.75$, where the flow around this thin airfoil is subsonic, no shock waves are formed. However, as the free-stream Mach number is increased to 0.81, figure b, local velocities over the blade surface become supersonic. As the flow past the upper surface of the blade decelerates from the high local flow velocities a shock is formed. Initially this is a normal shock and is slightly detached from the surface. As the free-stream Mach number is increased, as in part (c), the upper shock moves back on the surface, becoming oblique, and a shock wave is formed on the bottom surface. In part (d), $M_0 = 0.98$, the oblique shocks have both moved to the trailing edge of the blade. Finally, when the free-stream Mach number is at 1.4, part (e), a bow shock is formed in front of the airfoil. The free-stream Mach number at which a shock appears on the surface of the blade, as in part (b), is a function of the airfoil shape and the angle of attack (high angles of attack and thicker airfoils with large camber cause the shocks to form at lower free-stream Mach numbers).

The intent in the following sections is to use this formation model as a guide in determining the behavior of the pressure ratio across the shock as a function of Mach number. Then the shock wave behavior is used as a model for comparison with the noise behavior of supersonic helical tip speed propellers.

Shock Wave Pressure Ratio

In order to provide insight into the noise of supersonic helical tip speed propellers, the pressure ratio across a two-dimensional blade shock wave will be investigated. The intent here is to investigate the variation of the pressure rise across the shock as Mach number is increased. At the lower Mach numbers the initially formed shocks are assumed to abide by the normal shock relations. The normal shock pressure ratios as a function of incoming Mach number, M_1, have been previously tabulated in reference 14 and table II of this report repeats a portion of reference 14. As the Mach number over the blade increases, the shocks become oblique. The oblique shock pressure ratio will then be used at the lowest Mach number where an oblique solution is available and at all higher Mach numbers. The following two equations for a two-dimensional oblique shock will be used to solve for the pressure ratio across the shock. These two equations are from reference 13 and are equations 7.84 and 7.85 in that reference.

$$\cot \delta = \tan \theta \left[\frac{(\gamma + 1)M_1^2}{2\left(M_1^2 \sin^2 \theta - 1\right)} - 1 \right] \tag{1}$$

$$\frac{P_2}{P_1} = \frac{2\gamma M_1^2 \sin^2 \theta - (\gamma - 1)}{\gamma + 1} \tag{2}$$

In these equations γ is the ratio of specific heats, M_1 is the Mach number upstream of the shock, P_1 is the pressure upstream of the shock, P_2 is the pressure dowstream of the shock, δ is the deflection angle and θ is the shock wave angle. These can be seen in the sketch of figure 4.

The method of solution of these equations is to choose the incoming Mach number of interest, M_1, and using the deflection angle δ, solve equation (1) for the shock wave angle θ. Two solutions for θ are possible, the weak and strong shock wave angles. For our solutions, the weak shock angle, the lesser of the two angles, will be used. This angle is then substituted into equation (2) to obtain the pressure ratio across the shock, P_2/P_1. This oblique shock solution would then be used for all of the Mach numbers where an oblique shock is possible and the normal shock pressure ratios would be used at the lower supersonic Mach numbers.

For an example, the case with a one-half of a degree deflection angle, δ, is presented. An oblique solution exists, for $\delta = 1/2^\circ$, down to a Mach number of approximately 1.04. Below this Mach number the normal shock pressure ratios from table II are used. This results in the curve of figure 5.

Figure 5 is interesting in that it shows some of the general trends of the data of figure 2. The level increases with Mach number initially and then after peaking, approaches a constant value. This is similar in general to the data curve shape. It is interesting to note that some peaking does occur in the data of figure 2 (particularly for SR-3) even though the curves through the data were not drawn with this peak. The generally favorable shape comparison between the data and the shock wave pressure ratio seems to reinforce the position that the noise can be modeled by shock wave theory. This leads to an actual theory-data comparison for the propeller models which will be undertaken in the next section.

PROPELLER COMPARISONS

In order to more accurately compare the shock wave trends with the propeller data a more detailed prediction is undertaken. The majority of the noise from the propellers is assumed to come from the outermost blade section and since the shock equations are two dimensional, only this outermost airfoil section is modeled.

SR-2 Comparison

The SR-2 propeller is a straight bladed propeller with approximately a 2 percent thick airfoil at the tip. This leads to a deflection angle of approximately 1° and this is chosen for δ. It should be noted here that the thicker the blade is for the same chord, the larger the deflection angle would be and hence the stronger the shocks, i.e., blades with larger percent thicknesses make more noise. When the calculations were performed, the curves shown in figure 6 were the result. Part (a) is a plot of P_2/P_1 versus M_1.

The overall sound pressure level was calculated as follows,

$$\mathrm{OASPL}_{\substack{\text{shock at}\\ \text{blade}}} = 20 \log_{10}\left(\frac{P_2 - P_1}{P_{ref}}\right) = 20 \log_{10}\left(\frac{\Delta P}{P_{ref}}\right) \qquad (3)$$

where $P_{ref} = 2 \times 10^{-5}\ N/M^2$ and P_1 was taken as atmospheric pressure.

This results in the curve shown in figure 6(b). As can be observed, this curve is different in both the ordinate and abscissa from the SR-2 data of figure 2. The shift in the Mach number is the result of M_1, the local Mach number, being greater than the free-stream Mach number, M_0, as was previously indicated in the Shock Wave Formation Section.

As was shown in reference 12, the "delocalization" of the shock appears to occur at a free-stream Mach number of about 0.9 for the helicopter blades tested and which is assumed to apply to the propeller blades under study, i.e., a local Mach number, M_1, of 1.0 occurs at a helical tip Mach number of 0.9. Applying this assumption allows the curve to be shifted as shown in figure 7. As can be seen in this figure, the level predicted for the shock wave pressure rise at the surface of the blade is still higher than the blade passage tone measured on the wind tunnel wall but the Mach number variations of the curves now have a number of similarities. The data and theory curves both seem to rise in the same helical tip Mach number region (0.9 to 0.95) and they both reach an asymptotic value at helical tip Mach numbers over about 1.0.

This comparison of the shock strength from a single blade with the measured fundamental tone noise of the propeller is sufficiently interesting to merit further corrections to the theoretical result to make it more nearly apply to the experimental situation. The corrections involve accounting for the time duration between shocks as it affects the average signal strength, allowing for attenuations of the signal with distance, and accounting for the presence of the tunnel walls.

The correction for the time duration was used previously in references 10 and 15 and is

$$\underset{\substack{\text{rms at}\\\text{blade}}}{\text{OASPL}} = \underset{\substack{\text{shock at}\\\text{blade}}}{\text{OASPL}} + 20 \log_{10} \left(\frac{T^*}{T}\right)^{1/2} \qquad (4)$$

where

$$T = \frac{\pi D}{B V_T}, \qquad (5)$$

$$T^* = \frac{C}{M_{ht} a_o}, \qquad (6)$$

D is the propeller diameter, B is the number of blades, V_t is the propeller tip speed, C is the blade chord, M_{ht} is the helical tip Mach number and a_o is the speed of sound.

From reference 15, the distance correction for a shock was based on 20 times the log of the distance to the three quarters power. It will then be assumed here that for the near field translation of the shock to the tunnel walls, that the correction should be

$$\frac{P_{wall}}{P_{blade}} = 15 \log_{10} \frac{X}{h} \qquad (7)$$

where h is assumed to be the length of a line drawn from the microphone to its point of tangency on the propeller circumference and X is a normalizing dimension for the blade section. For the purposes of our two dimensional model X is chosen as the airfoil chord, C.

In addition, the tunnel walls have the effect of doubling the incident pressure, which amounts to a 6-dB noise addition.

The final corrected sound pressure on the tunnel walls is given by

$$\text{OASPL}_{\substack{\text{rms at}\\ \text{tunnel}\\ \text{wall}}} = 20 \log_{10}\left(\frac{\Delta P}{P_{ref}}\right) + 20 \log\left(\frac{T^*}{T}\right)^{1/2} + 15 \log \frac{C}{h} + 6. \qquad (8)$$

In reference 10, it was noted that the data are almost completely dominated by the blade passage tone so this overall sound pressure level is compared to the maximum blade passage tone noise on the tunnel wall in figure 8.

The two-dimensional shock theory shows the same trends as the data, as was mentioned before. At helical tip Mach numbers over about 1.0 the comparison in both trend and level is very good with the shock theory slightly under estimating the data. The underprediction is what would be expected since only the outer portion of the blade was included in the calculation. In the region from $M_{ht} = 0.92$ to $M_{ht} = 1.0$, the shock theory overpredicts the data. This is the transition range between the normal and oblique shock regimes and it may be that this range is not handled correctly in this shock wave formulation.

Swept Blade Comparisons

The previous comparison was for a blade with a straight leading edge, SR-2. The following compares the two dimensional shock theory predictions with data for two blades with swept leading edges. The first blade, SR-1M, was swept for aerodynamic purposes to reduce losses. The second blade, SR-3, was swept both for aerodynamics and acoustics. The intent of the acoustic sweep was to provide some cancellation of noise by proper phasing of the noise coming from various sections of the blade.

Figure 9 compares the predictions of the two-dimensional shock wave theory with the noise data from the two propellers. Figure 9(a) is for SR-1M and 9(b) for SR-3. (The theory predictions include the same Mach number shift as was used for SR-2.) As can be observed from the data curves on these figures, and as was noted previously in references 2 and 10, the sweep built into SR-1M and SR-3 serves to delay the formation of shocks to a higher helical tip Mach number. It was observed in reference 10 that these shifts amounted to 0.06 M for SR-1M and 0.11 M for SR-3. If the predicted curves for the two propellers are shifted by these amounts, the curves of figure 10 result. Figure 10(a) is for SR-1M and here the shift has brought the data and theory more closely into alignment with the final noise levels beyond $M = 1.1$ being in fairly close agreement. In figure 10(b), SR-3, the shift has brought the Mach number dependence of the curves more closely into agreement. It even appears that the slight peak in the data occurs at the same Mach number, 1.07, as the large theory peak. However, the levels predicted by the shock theory are higher than the data at the higher Mach numbers.

The difference between measured and predicted levels for SR-3 at the higher Mach numbers is probably the difference between the two dimensional theory representation and the expected three dimensional design performance of SR-3, which was designed to have the sound from different spanwise sections cancel each other. It appears that this concept was successful since SR-3 is some 6-dB quieter than SR-2 at the design point. However, the two-dimensional shock representation used here was only for the outermost section of the blade and shocks from the inboard sections and any possible cancellations therefrom were not included.

It appears that, in order to accurately predict the noise from propellers like SR-3 using shock theory, the contributions from all of the sections of the blade will have to be considered. In other words a three dimensional model will have to be constructed. This would hopefully provide a more accurate prediction for the SR-3 propeller and possibly decrease the theory at the lower Mach numbers to be more in line with the small peaks observed in the data. The inclusion of the shocks from all of the supersonic blade sections is a more difficult task than this initial undertaking since it requires determining both the shock pressures and the timing of the shock arrivals. It may be possible to do this by sectioning the blade hub to tip and properly adding the contributions from each section. In any case some three dimensional shock wave representation is the recommended direction for future work in this area.

CONCLUDING REMARKS

Since it appears that linear acoustic theory does not accurately predict the noise of propellers operating at supersonic helical tip speeds, a two-dimensional model based on shock wave theory was investigated. This model, which was based on the pressure difference across a shock wave, showed many of the same trends with increasing Mach number as did the existing propeller data, particularly the leveling off of the noise with increasing propeller tip Mach number.

As a result of the general trend agreement a more detailed comparison was undertaken for the straight bladed SR-2 propeller where the two-dimensional theory modeled only the outboard blade section as the noise source. With an estimated Mach number adjustment to account for the fact that the shock starts on the blade before the free-stream Mach number is one and with the proper conversion and translation of the pressure rise across the shock on the blade into an estimate of the noise measured on the tunnel wall, the model gave a good prediction for the measured blade passage noise of the propeller at helical tip Mach numbers over 1.0. Below a helical tip Mach number of 1.0, the shock theory had the same general trends as the data but the peak in the theoretical noise curve was higher than the measured data. One possible reason for this is that the contributions to the noise from other sections of the blade, not included in the theory, may be acting to cancel some of the noise calculated from the outboard section.

A comparison between the two-dimensional shock pressure rise and the noise from two swept bladed propellers was also undertaken. The sweep of the blades results in the delay of the start of the shock on the blades and when this was included, the shock model fairly well predicted the supersonic helical tip speed noise of one of the swept blades, SR-1M. The trends with Mach number were fairly well predicted for the other swept blade, SR-3, but the shock model overpredicted the noise of this propeller at supersonic helical tip speeds. The SR-3 propeller was designed to have noise cancellation from some of the inboard sections. Since the shocks from the inboard sections were not included in this two-dimensional model, this may be the explanation for the overprediction.

The generally good agreement between the two-dimensional shock theory and the propeller data may be partially fortuitous but it indicates that shock theory is a viable candidate to be used in place of the linearized acoustic theory to more accurately predict the levels and trends of the noise from propellers operating at supersonic helical tip speeds. This model might also be applicable for both helicopter blade passage tone noise and turbofan multiple pure tone noise which appear to have the same dependence on Mach number (over 1.0) as does the propeller noise.

In this paper only the outboard section of the blade was modeled in the two-dimensional shock noise theory. In order to more correctly model the propeller noise, the recommended next step is to have some form of three-dimensional model which includes the shocks from all the blade sections and accounts for both their strengths and the timing of their arrival at the measurement points.

REFERENCES

1. Dittmar, J. H.; Blaha, B. J.; and Jeracki, R. J.: Tone Noise of Three Supersonic Helical Tip Speed Propellers in a Wind Tunnel at 0.8 Mach Number. NASA TM-79046, 1978.

2. Dittmar, J. H.; Jeracki, R. J.; and Blaha, B. J.: Tone Noise of Three Supersonic Helical Tip Speed Propellers in a Wind Tunnel. NASA TM-79167, 1979.

3. Farassat, F.: Theory of Noise Generation from Moving Bodies with an Application to Helicopter Rotors. NASA TR R-451, 1975.

4. Farassat, F.: Discontinuities in Aerodynamics and Aeroacoustics - The Concept and Applications of Generalized Derivatives. J. Sound Vib., vol. 55, no. 2, Nov. 22, 1977, pp. 165-193.

5. Farassat, F.; Morris, C. E. K.; and Nystrom, P. A.: A Comparison of Linear Acoustic Theory with Experimental Noise Data for a Small Scale Hovering Rotor. AIAA Paper 79-0608, Mar. 1979.

6. Hanson, D. B.: Near Field Noise of High Tip Speed Propellers in Forward Flight. AIAA Paper 76-565, July 1976.

7. Hanson, D. B.; and Fink, M. R.: The Importance of Quadrupole Sources in Prediction of Transonic Tip Speed Propeller Noise. J. Sound Vib., vol. 62, no. 1, Jan. 8, 1979, pp. 19-38.

8. Ffowcs Williams, J. E.; and Hawkings, D. L.: Sound Generation by Turbulence and Surfaces in Arbitrary Motion. Philos. Trans. R. Soc. (London), Ser. A, vol. 264, May 8, 1969, pp. 321-342.

9. Dittmar, J. H.: A Comparison Between an Existing Propeller Noise Theory and Wind Tunnel Data. NASA TM-81519, 1980.

10. Dittmar, J. H.: An Evaluation of a Simplified Near Field Noise Model for Supersonic Helical Tip Speed Propellers. NASA TM-81727, 1981.

11. Hubbard, H. H.; and Lassiter, L. W.: Sound from a Two-Blade Propeller at Supersonic Tip Speeds. NACA Rep. 1079, 1952.

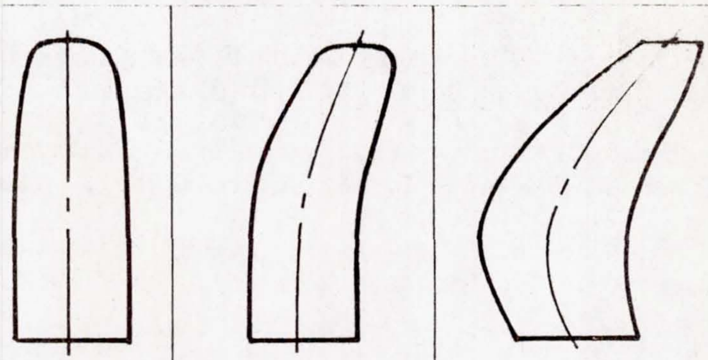
12. Schmitz, F. H.; and Yu, Y. H.: Transonic Rotor Noise: Theoretical and Experimental Comparisons, NASA TM-81236, 1980.

13. Bertin, John J.; and Smith, Michael L.: Aerodynamics for Engineers. Prentice Hall, 1979.

14. Shapiro, Ascher H.: The Dynamics and Thermodynamics of Compressible Fluid Flow. The Ronald Press Company, 1953.

15. Energy Consumption Characteristics of Transports Using the Prop-Fan Concept. (D6-75780, Boeing Commercial Airplane Co.; NASA Contract NAS 2-9104). NASA CR-137937, 1976.

TABLE I. - COMPARISON OF PROPELLERS

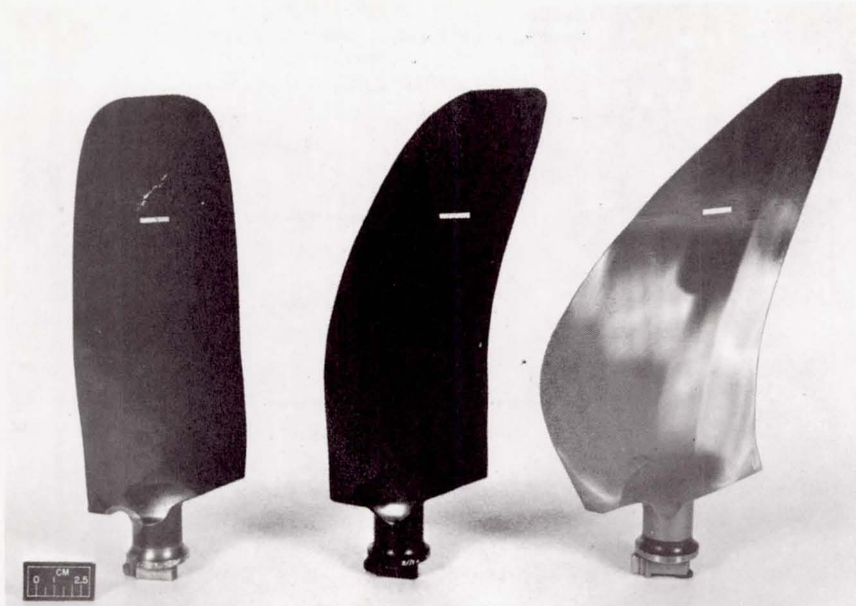


	SR-2	SR-1M	SR-3
Design cruise tip speed, m/sec (ft/sec)	244 (800)	244 (800)	244 (800)
Design cruise power loading kW/m^2 (shp/ft^2)	301 (37.5)	301 (37.5)	301 (37.5)
Number of blades	8	8	8
Tip sweep angle, mid chord, deg	0	23	34
Design efficiency, percent	77	79	81
Nominal diameter, D, cm (in.)	62.2 (24.5)	62.2 (24.5)	62.2 (24.5)

TABLE II. - NORMAL SHOCK PRESSURE RATIO

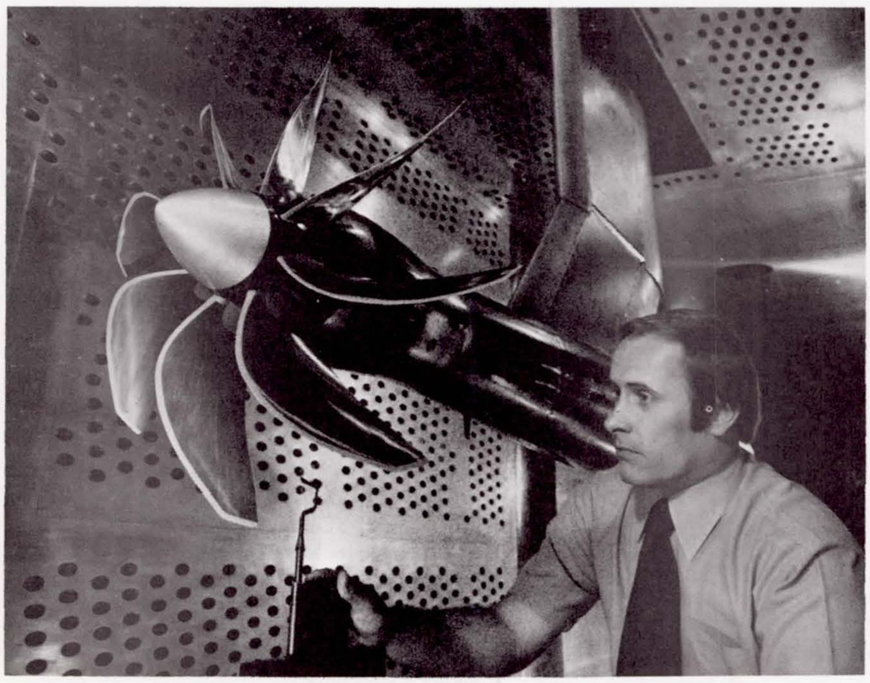
[γ = 1.4 (from ref. 14).]

M_1	P_2/P_1
1.00	1.0000
1.01	1.0234
1.02	1.0471
1.03	1.0710
1.04	1.0952
1.05	1.1196
1.06	1.1442
1.07	1.1690
1.08	1.1941
1.09	1.2194
1.10	1.2450



SR-2 SR-1M SR-3

(a) INDIVIDUAL BLADES.



(b) PROPELLER SR-3 IN TUNNEL.

Figure 1. - Propeller blades.

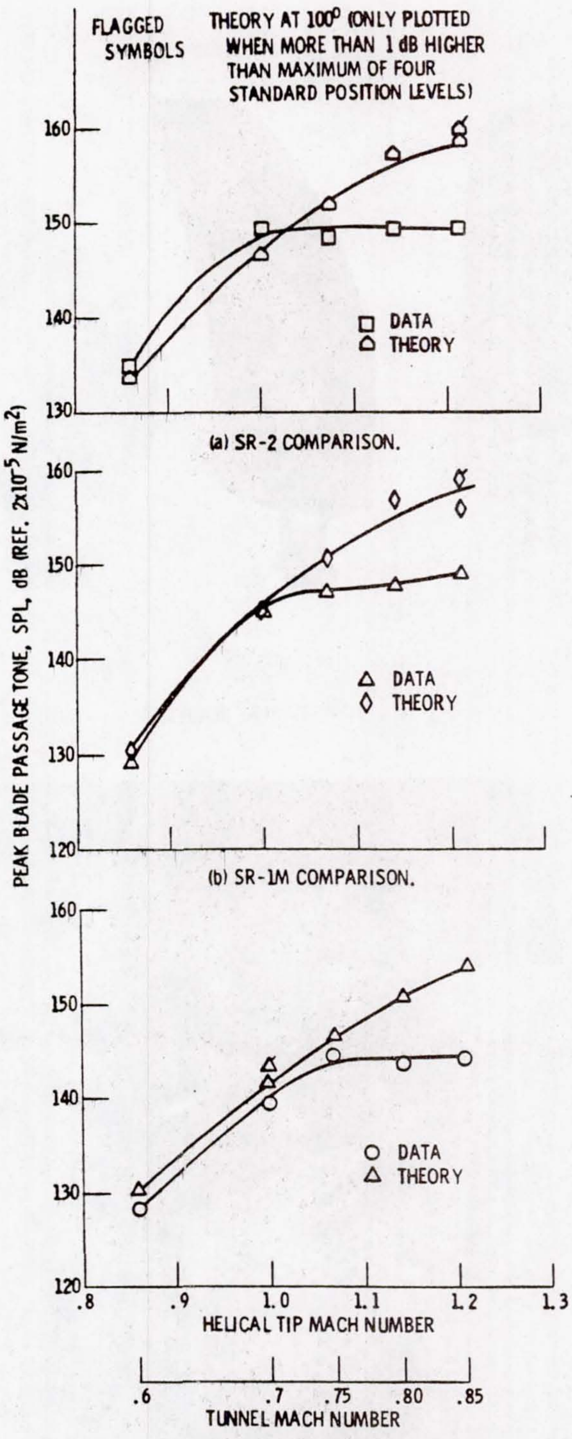


Figure 2. - Peak blade passage tone variation with helical tip Mach number.

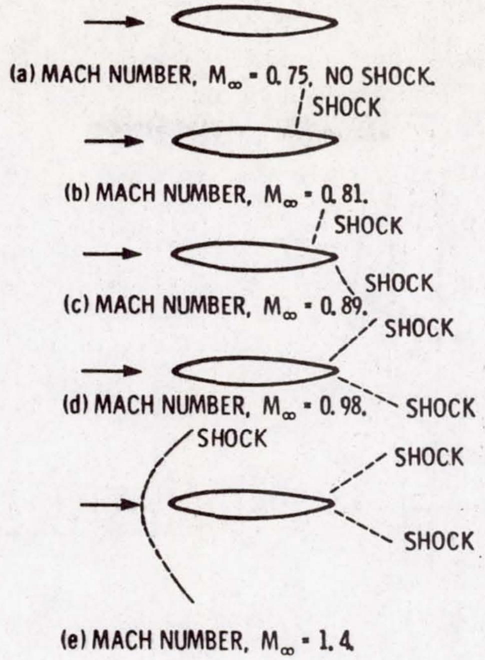


Figure 3. - Flow around an airfoil (from ref. 13).

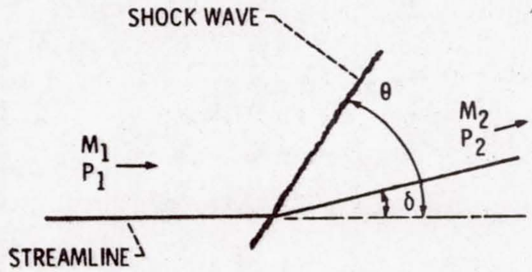


Figure 4. - Oblique shock wave.

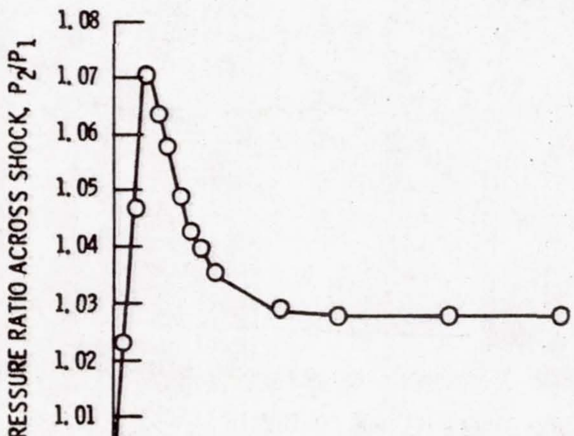
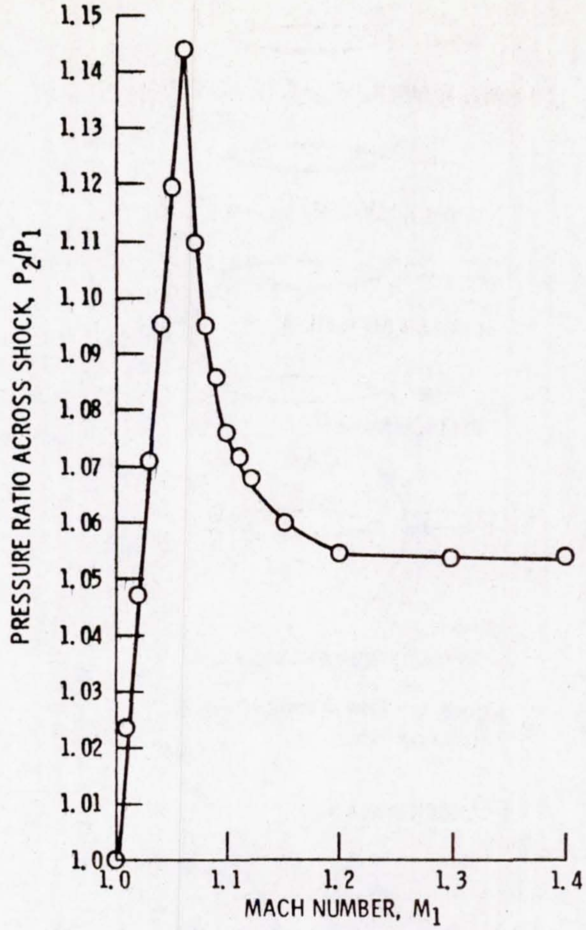
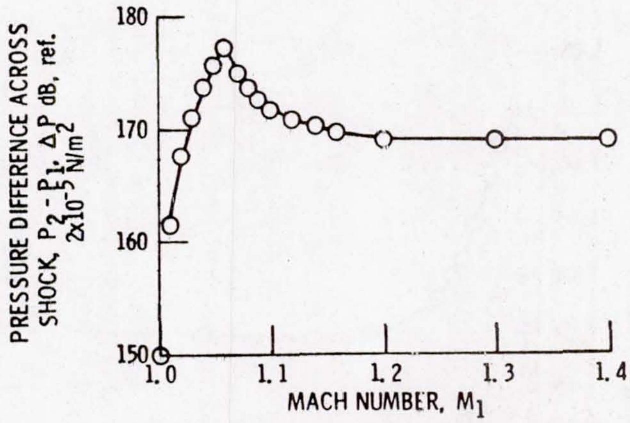


Figure 5. - Shock pressure ratio for a 0.5° deflection angle.



(a) SHOCK PRESSURE RATIO.

Figure 6. - 1.0^0 deflection angle shock.



(b) PRESSURE DIFFERENCE ACROSS SHOCK IN DECIBELS, P_1 ATMOSPHERIC PRESSURE.

Figure 6. - Concluded.

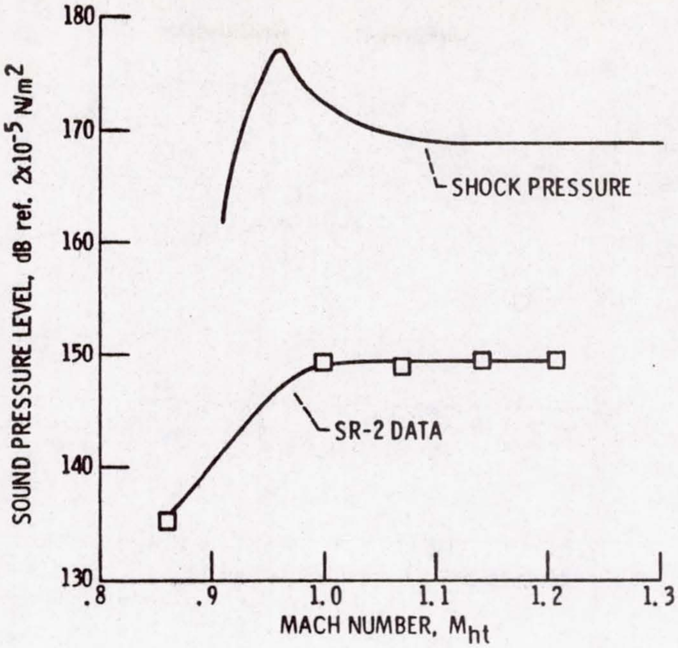


Figure 7. - Shock pressure rise at blade compared with maximum blade passage tone for SR-2 propeller on tunnel wall.

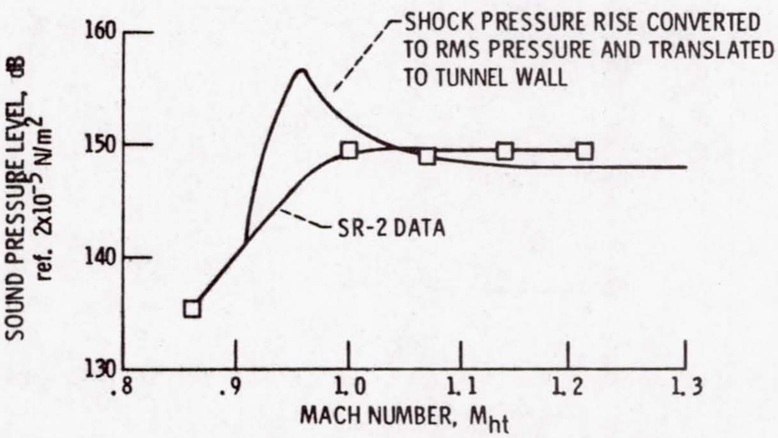
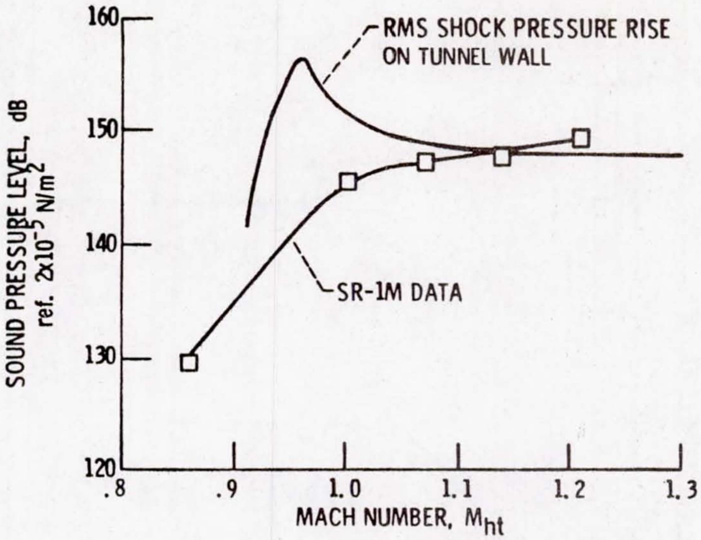
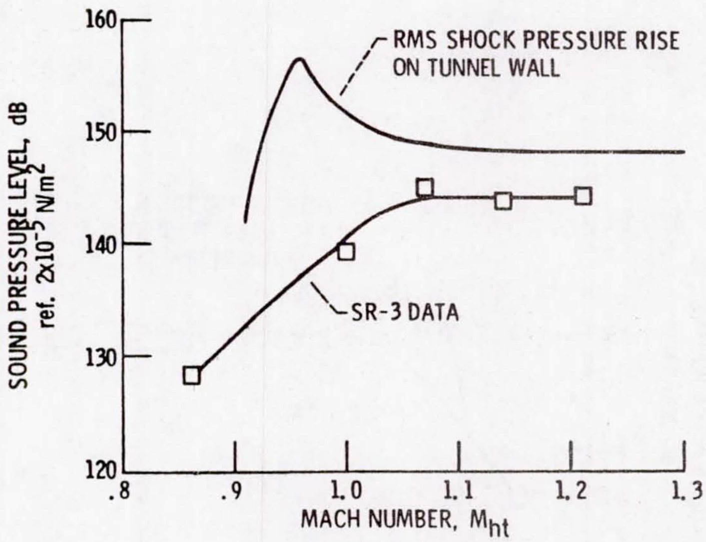


Figure 8. - RMS shock pressure compared with maximum blade passage tone for SR-2, both on tunnel wall.



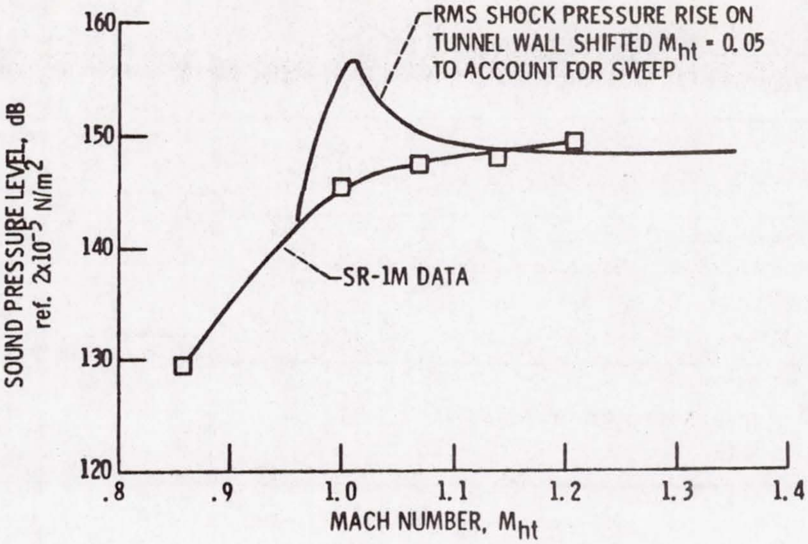
(a) SR-1M.

Figure 9. - RMS shock pressure compared with maximum blade passage tone, both on tunnel wall.



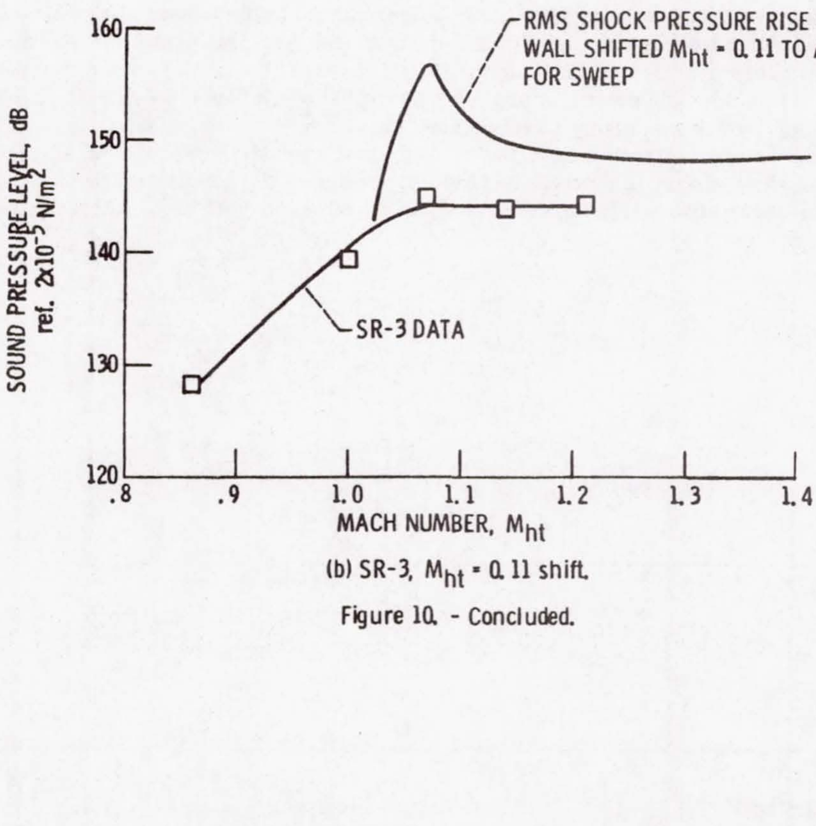
(b) SR-3.

Figure 9. - Concluded.



(a) SR-1M, M_{ht} = 0.05 shift.

Figure 10. - RMS shock pressure, shifted to account for sweep, compared with maximum blade passage tone, both on tunnel wall.



(b) SR-3, M_{ht} = 0.11 shift.

Figure 10. - Concluded.

1. Report No. NASA TM-82752	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle A SHOCK WAVE APPROACH TO THE NOISE OF SUPERSONIC PROPELLERS		5. Report Date December 1981
		6. Performing Organization Code 535-03-12
7. Author(s) James H. Dittmar and Edward J. Rice		8. Performing Organization Report No. E-1068
9. Performing Organization Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135		10. Work Unit No.
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		13. Type of Report and Period Covered Technical Memorandum
		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

One of the possible propulsive systems for a future energy efficient airplane is a high tip speed turboprop. When the turboprop airplane is at cruise, the combination of the airplane forward speed and the propeller rotational speed results in supersonic helical velocities over the outer portions of the propeller blades. As a result of these supersonic blade sections and their associated shock waves, these propellers may create a cabin noise problem for the airplane. To model this propeller noise, the pressure ratio across the shock at the propeller tip was calculated and compared with noise data from three propellers. At helical tip Mach numbers over 1.0, using only the tip shock wave, the model gave a fairly good prediction of the noise for a straight bladed propeller and for a propeller swept for aerodynamic purposes. However for another propeller, which was highly swept and designed to have noise cancellations from the inboard propeller sections, the shock strength from the tip over-predicted the noise. In general the good agreement indicates that shock theory is a viable method for predicting the noise from these supersonic propellers but that the shock strengths from all of the blade sections need to be properly included.

17. Key Words (Suggested by Author(s)) Propeller noise; Noise; Supersonic tip speed; Shock noise		18. Distribution Statement Unclassified - unlimited; STAR Category 71	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages	22. Price*

* For sale by the National Technical Information Service, Springfield, Virginia 22161

National Aeronautics and
Space Administration

Washington, D.C.
20546

Official Business

Penalty for Private Use, $300

SPECIAL FOURTH CLASS MAIL
BOOK

Postage and Fees Paid
National Aeronautics and
Space Administration
NASA-451

U.S. MAIL

NASA

POSTMASTER: If Undeliverable (Section 158 Postal Manual) Do Not Return