SHOCK LOSS ATTENUATION IN TRANSONIC ROTORS
USING AREA RULING CONCEPTS FOR PART-SPAN DAMPERS

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ABSTRACT

The transonic area rule as proposed by R. T. Whitcomb (1952) has been successfully applied to external flow since that time. This paper reports a preliminary semi-empirical study done in the early 1980’s to determine if a specific area ruling concept can be applied to internal flows, in particular to transonic fans using part-span dampers (i.e., part-span shrouds). Cascade and rotor correlations reported herein indicate that the area rule may be applied to internal flows. A simple precursory channel flow experiment has been done that shows an attenuation of shock losses using area ruling in supercritical internal flow.

LIST OF SYMBOLS

- aspect ratio
- cross-sectional area of the blade
- cross-sectional area of the part-span damper
- total annulus area
- blockage parameter, \( \tau \times \sigma \)
- drag coefficient
- momentum loss coefficient based on cascade inlet
- blade or damper chord
- drag
- blade span or height
- Mach number
- cascade inlet Mach number
- average rotor inlet Mach number

\( P \) - pressure
\( PR \) - pressure ratio
\( PSD \) - part span damper
\( t \) - maximum thickness
\( X \) - axial distance along axisymmetric body
\( Y \) - distance perpendicular to blade chord
\( Z \) - rotor axial distance
\( ZOC \) - blade chord length projected into the axial direction
\( Z \) - \( Z/ZOC \)
\( \alpha_1 \) - angle of attack
\( \beta_1 \) - cascade inlet air angle
\( \gamma \) - cascade stagger angle
\( \sigma \) - cascade and rotor blade solidity, chord/spacing
\( \tau \) - maximum thickness ratio
\( \overline{\omega} \) - average total pressure loss coefficient

Subscripts

max - maximum
0 - initial value or total value
1 - inlet
2 - outlet
arw - area ruled wall
sw - straight wall

Superscripts

* - critical value
' - relative value
INTRODUCTION

A series of correlations (Roberts 1979a, 1979b) has been formulated and proposed to predict the design and off-design aerodynamic performance of part-span dampers (PSD) on transonic rotors (see Figure 1). These correlations are based on rotor performance data where the damper was located near the mid-chord of the blade where the blade shape and thickness were not modified to take into account the increase in blockage due to the damper.

These correlations indicate that to minimize damper loss several things should be taken into consideration.

1. Damper leading and trailing edges should be as sharp as possible.
2. The damper should be located as near to the hub as possible so as to minimize shock loss.
3. Work input should be minimized at the PSD location.
4. The damper should be as thin as is possible so as to minimize the area influenced.

However, after all of these design rules have been followed, there remains a damper of some size in some location on the rotor that gives rise to additional profile and shock losses. How then should we attempt to further optimize the shape of the damper/rotor combination to minimize losses? A fruitful area of past research that has allowed a drag reduction for wing/body combinations has been the area rule proposed by Whitcomb (1952). The question is, can the area rule which has been developed for external flows be applied to internal flow through rotating channels? If this application is possible, it could be a method of further reducing losses due to part-span dampers. The purpose of this study then is to determine the possibility of applying the transonic area rule to internal flows.

THE AREA RULE

Whitcomb (1952) has shown how the drag of wing/body combinations at transonic speeds may be reduced to a surprising extent by simply cutting out a portion of the fuselage to compensate for the area blocked by the wing. Whitcomb's deduction of the area rule was based on considerations of stream tube and the phenomena of choking which follow from one-dimensional flow theory. Each individual stream tube of a three-dimensional flow field must obey the law of one-dimensional flow. Although the three-dimensional field cannot be calculated on this basis alone, nevertheless, it provides a good starting point. Figures 2 and 3 show an example of Whitcomb's experiments that demonstrate the concept of the area rule. The results demonstrate again the effectiveness of basic and simple physical thinking.

It can be seen that the drag rise for wing and cylindrical body combination with that for a comparable body of revolution and a cylindrical body alone are compared. It can be seen that the wing/body combination and the equivalent body of revolution exhibit the same drag rise characteristics. Then, in Figure 3, a logical conclusion of this reasoning is presented where the effects of body indentation on the transonic drag rise is shown. In this figure, an indentation is cut out of the fuselage to compensate for the cross-sectional area of the wing. It can be seen from this figure that the amount of wave drag incurred by the wing/body combination is greatly reduced from that where the indentation is not present. This simple concept of blockage or area allowance for additional wing blockage is seen to be a powerful method for the reduction of drag due to shock waves in external flow. The question now is whether the area rule will apply to the flow through rotating compressor channels. One might try to determine the range of applicability of the area rule as was done by Spreiter (1956). In that paper, insight into the range of applicability of the transonic area rule was gained by comparison with the loss rise curve of transonic flow and with available experimental data for a large family of three-dimensional rectangular wings having NACA 63 series profiles.

TRANSONIC SIMILARITY RULES FOR CASCADES AND ROTORS

Figure 4 shows the application of the area rule to rectangular wings of identical area distributions. There are two rectangular airfoils of the same aspect ratio-thickness parameter, $A \times T$, that have the same wave drag rise characteristics. What this demonstrates is that airfoils of different thickness and aspect ratio have the same transonic drag rise characteristics because their effective area distribution is the same. A similar comparison to make for channel flow would be the transonic loss rise characteristics of cascades. This was done for a series of cascade experiments using NASA 65 airfoils. Figure 5 shows that for the same value of cascade blockage parameter, $B_c = \tau \sigma$, the transonic loss characteristics are similar for different cascades composed of NACA 65 series airfoils (cascades in Figure 5). The aspect ratio of the cascades was not involved in this correlation as it was in Spreiter's correlation due to the fact that the flow was two-dimensional. However, thickness varied from 6% to 10%, solidity from a value of .6 to 1.5, and camber from 15° to 35°. What Figure 5 indicates is that for a similar area distribution, the amount of blockage dictates at what Mach number the passage experiences the transonic drag rise. This is shown in Figure 6 where the loss rise Mach number $M_f$ is plotted against blockage parameter. It is seen there that for the higher blockages, the critical Mach number decreases, whereas for lower blockage it appears insensitive.

Figures 5 and 6 demonstrate that transonic similarity principles can be applied to flow through cascade channels. In a manner similar to the correlations done for blockage parameters of cascades, the NASA rotor data that was used in the previous damper correlations (Roberts 1979a, 1979b) could be used to correlate loss rise Mach number characteristics with blockage. With this in mind, 9 transonic NASA rotors were selected as shown in Table 2, with varying pressure ratios and geometries. From the shock loss model of Schwenk, Lewis and Hartmann, and the part-speed data on these rotors, the loss rise curve and loss rise Mach numbers were estimated. Details of the shock loss model can be found in Schwenk et al 1957.

From the part speed data on the nine rotors, the drag rise curve and drag rise Mach numbers were estimated both from plotting overall average Mach number as a function of total average loss coefficients for the whole rotor, and by looking at the calculations made with the shock loss model to see when there was a difference between total and profile loss coefficients. For the rotors, although there were different thickness distributions, there would be some basis of similarity in comparing the maximum area ratios between the rotors and drag rise Mach number. The area distributions along streamlines through the fans under study were estimated using the rotor geometry and experimental pressure probe survey data taken during
testing near design point. In this way, the area distributions were taken approximately normal to the relative flow and then projected onto the axial direction.

Blade thickness was summed up on lines of constant percent chord running from the hub to the tip. The thickness of the damper was taken into account by calculating the area increment due to a bi-convex airfoil located at the appropriate nondimensional axial distance. A typical distribution of rotor area projected in the axial direction is shown in Figure 7a, 7b and 7c. It can be seen that the contribution to the overall area distribution due to the damper can be significant. This method of determining area distribution was computerized and the distributions were estimated for all the rotors of Table 2. Figure 8 shows the correlation between estimated drag rise Mach number and maximum rotor-damper area ratio (Ag + App) /Ag, which is a blockage parameter similar to Be for cascades. It can be seen that there is a trend toward lower drag rise Mach numbers as the area ratio increases. This is consistent with the correlation of Figure 6.

Several of the rotors of Table 2 were chosen to further test out the applicability of area ruling to rotating channel flows. It was reasoned that rotors with similar area distributions should have similar drag rise characteristics as shown by Whitcomb. Figures 9 and 10 support this supposition.

The preceding correlations and calculations have indicated that shaping and area ruling concepts might be successfully applied to transonic rotor flow to reduce wave drag. Several application techniques and a simple preliminary verification experiment are proposed in the next section.

DAMPER SHAPING AND AREA RULING TECHNIQUES TO REDUCE ROTOR WAVE DRAG

A possible method of decreasing the wave drag due to dampers would be to increase their chord while maintaining the same thickness (i.e., increasing the fineness ratio and adding area to smooth the area distribution). Two wings of the same thickness as but with one having a larger chord than the other will exhibit a decrease in drag. For the same maximum thickness, the transonic theory tells us that the drag coefficient will decrease inversely with the length squared (Sears 1947). Therefore, there is a potential wave drag reduction by a factor of 25 going from a damper of fixed thickness covering 20% of the blade chord to one of the same thickness covering 100% of the blade chord. To get an indication that this was reasonable, a two-dimensional channel calculation was made for a damper of constant thickness covering 20% of a channel and one covering 100% of a channel at a transonic Mach number of 0.92 using the method of Murman and Cole (1971). The results are shown in Figure 12 where the channel flow is shown to scale. For the first situation, that where the damper profile covers 20% of the channel chord, we see a relatively large region of supersonic flow. For the damper with the same thickness but covering the total channel length, the supersonic flow has decreased significantly, and so has the wave drag coefficient in this case, as predicted by the method of Sears. This difference is due to the nonlinear nature of the wave drag in the vicinity of the critical Mach number.

Figure 1 shows a flow path schematic for transonic fan rotors showing the hub contour, rotor blade and the part-span damper. There are several methods that could be used to area rule in the vicinity of the damper.

1) Blade area ruling: in the vicinity of the damper, the blade itself is relieved as if the blade is the fuselage and the damper is the wing for a wing/body combination. There are two difficulties with this approach. The first is that the blades are very thin as presently designed, in fact as thin as they can be, and any further removal of blade material in the vicinity of the damper could cause structural problems. Secondly, the work input in the damper location could be affected by area ruling a blade, because of the interference with the flow turning and camber distribution. However, the wiggles and odd blade twisting in the damper region of many modern fan blades indicate that some form of area ruling is being used—although none of this has been reported in detail in the open literature.

2) Area rule the damper or shape it and increase the chord. A damper could have its maximum thickness toward the leading and trailing edge of the blade with less thickness at the damper mid-chord where the blade is thickest. This could help alleviate some of the wave drag; however, there could be some manufacturing and mechanical difficulties. A more promising concept might be to increase the chord and also to shape the damper so that the maximum damper thickness is toward the trailing edge of the blade and behind the shock region associated with the blade leading edge.

3) Perhaps the least difficult method of area ruling the channel is to contour the hub and tip casing to allow for the blockage of the damper as shown in Figure 12. This could be done with a minimum effect upon the blading itself and on the shape of the damper. Furthermore, contouring the end-walls could be done either at the hub or the tip depending upon which was the most aerodynamically effective and mechanically desirable. Finally, the simplest and potentially optimum method of reducing the wave drag might be to contour the tip and/or hub casing and to increase the chord of the damper to that of the full blade chord length as is shown in Figure 13. This would require no exotic blade or damper shapes and could be easily designed into any new transonic fan.

Before the last of these concepts is applied to the actual design of the transonic fan, it would be desirable to do a basic test in a simple low cost situation, such as the channel flow experiment shown in Figure 14, which could be tested in a small transonic tunnel. This channel is the simple analog of an uncambered cascade at zero stagger angle. The simplified channel experiment can verify the flow physics of the third internal area ruling concept above.

PRELIMINARY TESTING

The simple channel flow experiment described below was to be the first in a proposed series of tests culminating in a transonic rotor test of the "casing" area ruling concept (number three above). However, research budget reductions forced the cancellation of subsequent experiments.

A small 6 x 6 inch (152.4 x 152.4 mm) transonic/supersonic blow down wind tunnel was used to test the model shown in Figure 14. This tunnel is located at San Jose State University, California, is described by Middagh (1971). To keep cost as low as possible, the blade/damper combination was adapted from an available double-wedge airfoil model with a smaller, double-wedge damper added. Two configurations were tested, see Figures 15 and 16.
Figure 15 is a photograph of the blade/damper model with straight walls and Figure 16 is the same model with the walls area ruled for the damper. The double wedge airfoil chord was six inches (152.4mm). The blade chord/thickness ratio was 10 percent and the damper chord/thickness ratio was 30 percent. The damper chord was 23 percent of the blade chord.

Figure 17 shows an open view of the wind tunnel test section. After the wind tunnel was calibrated, an upstream total pressure probe and a downstream nine-prong total pressure rake were used to measure the mass averaged total pressure drop across the blade/damper model. A series of static pressure taps through the test section allowed the Mach number distribution to be calculated.

Testing was conducted at a subsonic Mach number of M = 0.43 and a blade chord Reynolds number of $R_c = 2.5 \times 10^6$ to determine the magnitude of the viscous total pressure losses. Viscous losses were found to be $\approx 2\%$ for both straight wall and area ruled models.

Due to the relatively large blockage of the model $(\approx 18\%)$, the inlet Mach number for the supercritical test was only M = 0.63 with expansion through the test section to transonic Mach numbers*. An example Mach number distribution is shown in Figure 18. Reynolds number for these tests were $R_c = 3 \times 10^6$.

If the total pressure loss coefficient for the channel is defined as
\[
\bar{\omega}_C = \frac{(P_{o1} - P_{o2})}{P_{o1}}
\]
then the drop in total pressures across the model for straight walls (sw) and area ruled walls (arw) was:

\[
\bar{\omega}_C(\text{viscous}) = \frac{27.70-26.46}{27} = 0.046 (M = 0.4)
\]
\[
\bar{\omega}_C(\text{supercritical, sw}) = \frac{27.00-25.96}{27} = 0.032 (M = 0.63)
\]
\[
\bar{\omega}_C(\text{supercritical, arw}) = \frac{27.00-24.16}{27} = 0.105 (M = 0.63)
\]
where values of total pressure are in psia**.

The viscous loss coefficient at M = 0.43 is approximately the same as that at M = 0.63. Therefore, an approximation of channel shock loss coefficient can be made by subtracting out the measured viscous loss coefficient:

\[
\bar{\omega}_{\text{shock}} = \bar{\omega}_C - (\bar{\omega}_C) \text{viscous},
\]
\[
\bar{\omega}_{\text{shock, sw}} = 0.107,
\]
\[
\bar{\omega}_{\text{shock, arw}} = 0.085.
\]

This preliminary testing indicates a decrease in shock pressure losses of approximately 20% for the area ruled channel when compared to the straight wall model.

CONCLUSION

Correlations and preliminary testing reported herein indicate that internal area ruling of transonic rotors with part-span dampers could result in a worthwhile reduction in shock losses. The simplest form of area ruling is at the rotor tip where the casing treatment could be contoured to relieve the blockage due to the damper.

ACKNOWLEDGEMENT

The authors wish to thank Mr. H. L. van Rintel and Mr. G. Rizvi for their extensive work on the wind tunnel measurements. This work has been sponsored by NASA-Lewis Research Center under Grants NAS3 - 21994 and NAG 3-196. The authors are appreciative of this support. Finally, the authors are grateful to "Mel" Hartmann, who encouraged this work and to "Cal" Ball who was very helpful in securing grant support.

REFERENCES


Middagh, R. T., 1971, "Optimization and Calibration of the 6 x 6 Inch Supersonic Wind Tunnel," ME-180 Special Projects, San Jose State College Mechanical Engineering Dept., San Jose, California.


TABLE 1.- CASCADE CONFIGURATIONS USED FOR BLOCKAGE PARAMETER CORRELATION

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### TABLE 2. - NASA TRANSONIC ROTORS
(For Symbol Code See Roberts 1979a)

<table>
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**Figure 1. Rotor-Blade Row with Part-Span Dampers**

Downloaded from http://nanoengineeringmedical.sagepub.com/content/78903/V03AT15A041/2403360/v03at15a041-93-gt-190.pdf by guest on 15 December 2021
Figure 2. Comparisons of the drag rise for the delta-wing and cylindrical-body combination with that for the comparable body of revolution and the cylindrical body alone (Whitcomb 1952).

Figure 3. The effects on transonic drag obtained by indenting the body of a delta wing-body combination (Whitcomb 1952).
\[ A = \text{Aspect ratio} = \frac{h}{c} \]

\[ \tau = \text{Thickness ratio} = \frac{t}{c} \]

\[ A \cdot \tau = 0.16 \]

Figure 4. - Comparison of the drag rise characteristics of two rectangular wings (after Spreiter 1956).
Figure 5. - Correlation of momentum-loss coefficient rise with Mach number for cascades with identical blockage parameters, $B_p = \pi \sigma$ (Symbol code in Table 1).

Figure 6. - Variation of loss rise Mach number with blockage parameter for cascades of 65-series airfoils.
Figure 7a - Meridional view of rotor 3.

Figure 7b - Span-wise thickness distribution for rotor 3.
Figure 7c - Normalized rotor area distribution for NASA Rotor 3.

Figure 8. - Correlation of rotor loss rise Mach number with rotor-annulus area ratio estimated from experimental data. (Symbol code in Table 2)
Figure 9. - Comparison of loss coefficient rise and normalized area distributions for NASA rotors 3, 12 and 18.
Figure 10. - Comparison of loss coefficient rise and normalized area distributions for NASA rotors 4 and 6.
Figure 11. - Comparison of calculated wave drag of long and short chord damper profiles in 2-D transonic channel flow using the method of Murman and Cole (1971).
Casing contour area ruling

Flow

Hub contour area ruling

Figure 12. - Flow path schematic for transonic fan rotors using contour area ruling.

Casing contour area ruling

Flow

Part span damper chord increase

Hub contour area ruling

Figure 13. - Flow path schematic for transonic fan rotors using contour area ruling and damper chord increase.
FIGURE 14. - Model Configuration

FIGURE 15. Test Model - Straight Configuration

FIGURE 16. Test Model - Area Ruled for Damper
Figure 17. San Jose State University 6 in. x 6 in. High Speed Blow Down Wind Tunnel.

Figure 18. Average Mach Numbers At Locations Relative to Model for Supercritical Runs with Area Ruled Model Configuration.