Transonic Compressor Technology Advancements

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This paper discusses the highlights of the NASA program on transonic compressors. Effects of blade shape and throat area on losses and flow range are discussed. Some effects of casing treatment on stall margin are presented. Results of tests with varying solidity are also presented. High Mach number, highly loaded stators are discussed and some results of stator hub slit suction are presented.

For turbojet and turbofan aircraft propulsion applications, lightweight, compact, efficient fans and compressors are essential. These requirements have led to the selection of axial-flow type components for most aircraft propulsion systems. At the present time, axial-flow fans and compressors have been developed to a point where improvements in efficiency are not easily achieved. Therefore, the goal of much of the research and development effort is to reduce size and weight without sacrificing efficiency. Present-day supersonic aircraft propulsion systems must also be capable of operating with rather severe distortions of total pressure at the fan or compressor inlet. This stresses the requirement of broad stall-free range of operation. Therefore, reductions in fan or compressor size or weight must be compatible with operating range requirements of specific applications.

The objective of this paper is to review some of the approaches to reduced size and weight and to discuss some of the recent advances in fan and compressor technology that may contribute to improved designs. This paper is not intended to be an exhaustive report on new technology, but will consider some of the advances considered to be significant.

APPROACHES TO REDUCED SIZE AND WEIGHT

Fan or compressor size is significant from the standpoint of compactness, as related to installation problems, and is also significant because of the direct relationship of size on weight. Reductions in size can be achieved by reduced diameter and/or reduced length. Reduced diameter requires increased flow per unit of annulus area or more effective flow area per unit of diameter. Current practice already is quite advanced with regard to flow capacity, so that only limited gains can be made in this area. Reduced length can be achieved by reducing the length of each stage or by reducing the number of stages. Reduced length per stage or increased blade aspect ratio is an effective way of achieving reduced weight. Unfortunately, this approach leads to severe mechanical and aeroelastic problems as well as to severe limitations in attainable flow range. Both of these problem areas are being studied, but to date no major breakthrough has been achieved. This leaves increased pressure ratio per stage as the most promising technique of achieving more compact compressors.

Reductions in compressor weight can, of course, be achieved by reductions in weight per unit size as well as by reductions in compressor size. Reductions in weight per unit size, however, are achieved through improvements in materials or in mechanical design concepts rather than through aerodynamic advancements and will not be considered in this paper.

Increases in stage pressure ratio can be achieved by increasing the tip speed of the rotor or by increasing blade loading. The effectiveness of these two approaches to increasing stage pressure is shown in figure 1. In this figure, pressure ratio is plotted versus rotor tip speed with rotor tip loading as a parameter. Tip loading is expressed in terms of tip diffusion factor, D_{rt}. This plot is based on the assumption of zero pre-whirl at the rotor inlet.

Figure 1 shows that pressure ratio increases can be achieved either by increased loading or by increased tip speed. Maximum gains obviously come from increasing both loading and tip speed. An increase in loading at a given tip speed does not increase the relative inlet Mach number, but increased losses and reduced range of operation may result from the increased diffusion. Increases in tip speed at a given level of loading, of course, result in increased Mach numbers relative to the rotor blading. This may also adversely affect efficiency and range of operation. With zero inlet swirl, the tip relative Mach number at 1000 feet per second tip speed will be on the order of 1.1 and at 1600 feet per second it will be on the order of 1.6. Pre-swirl can, of course, be used to reduce the rotor tip relative Mach number, but this will reduce the attainable pressure ratio for a given tip diffusion factor and will also increase the stator Mach numbers.

In addition to rotor Mach numbers and loading, consideration must be given to stator parameters which are usually critical at the hub. Figure 2 shows variations in stator hub Mach number and stator hub diffusion factor as functions of rotor tip speed with rotor tip diffusion factor as a

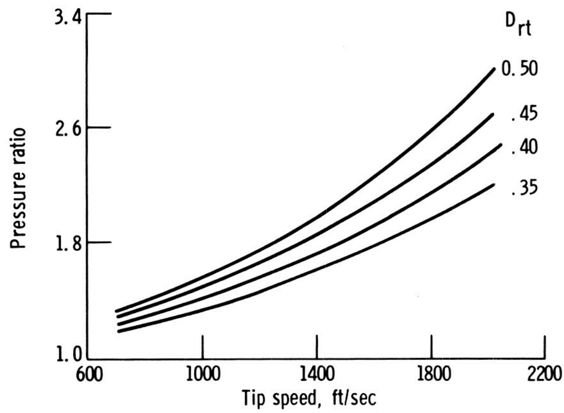

FIGURE 1.—*Effect of tip speed and blade loading on rotor pressure ratio.*

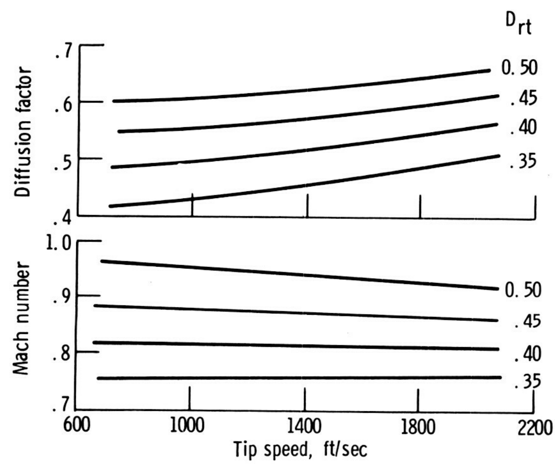

FIGURE 2.—*Effect of rotor tip speed and loading on stator hub loading and Mach number.*

parameter. Stator hub Mach number increases appreciably with increased rotor tip loading at a given tip speed, but is nearly constant with increased tip speed at a fixed level of rotor loading. The stator hub diffusion factor increases at about the same rate as the rotor tip diffusion factor when speed is held constant. Only small increases in stator hub diffusion factor are obtained when tip speed is increased at a constant value of rotor tip loading.

From figures 1 and 2 it is evident that increasing tip speed at a given level of rotor loading requires only the solution of the problem of increased rotor relative Mach numbers, provided, of course, that limiting values of rotor loading are not adversely affected by increased Mach number. Increases in rotor loading, however, result in stator Mach number and loading problems as well as in increased rotor problems. Simply counting problems would indicate that the simplest approach would be to just increase tip speed to attain higher stage pressure ratio. The relative effort required to achieve acceptable performance as well as other design requirements, however, may make the route of high loading, low speed more profitable. Maximum gains are made by increasing both rotor tip speed and loading. Therefore, studies of both increased loading and increased Mach numbers for rotors and stators are required. Studies of the effects of increased loading and/or increased Mach numbers must include evaluation of attainable flow range or stall margin as well as attainable efficiencies. If flow range limitations are severe, it may be necessary to operate the compressor at reduced pressure ratio and efficiency to provide sufficient stall margin. Such restrictions could limit the gains attainable from increased loading or tip speed.

NASA ROTOR PROGRAM

As a part of the overall NASA compressor research program, a number of axial-flow stages have been designed, built, and tested either in-house or by contract to investigate the effects of high loading and high tip speed on performance. Figure 3 shows the range of loadings and wheel speeds studied to date. In this figure, individual stages are plotted at design values of tip speed and tip diffusion factor. The pressure ratios are not exact because of three-dimensional effects and loss gradient effects, but this plot does indicate the range of conditions covered to date. The solid symbols represent subsonic stages. The two lower speed subsonic stages had no inlet guide vanes and the remaining three utilized inlet guide vanes.

These subsonic stages were aimed at investigation of boundary layer control schemes such as slotted blading and are included here just to define the complete scope of the NASA program. Transonic stages cover a range of tip speeds from 1000 to 1600 feet per second and rotor tip D-factors from 0.35 to slightly over 0.5. Most of these transonic stages were designed to investigate blade shapes for high relative Mach numbers.

Design speed maximum efficiencies and the corresponding pressure ratios for these stages are given in figure 4. This figure shows that, in general, maximum efficiency decreases only slightly with speed. The peak efficiency for the 1600-feet-per-second rotor is 0.89. The pressure ratio at that operating point is slightly over 2.0. At the other extreme of the curve is a very highly loaded rotor designed for a tip speed of 1000 feet per second. This rotor has a maximum efficiency of about 0.93 at a pressure ratio of 1.5.

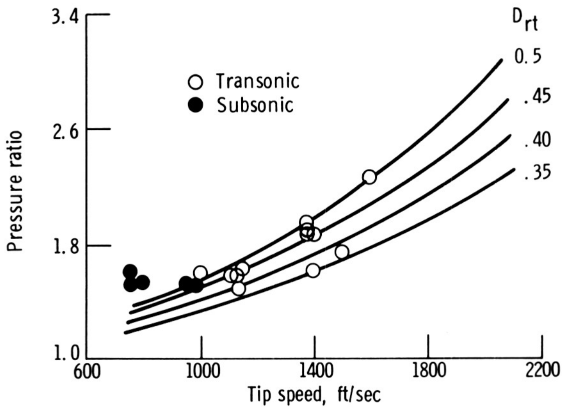

FIGURE 3.—*Rotors tested in NASA program.*

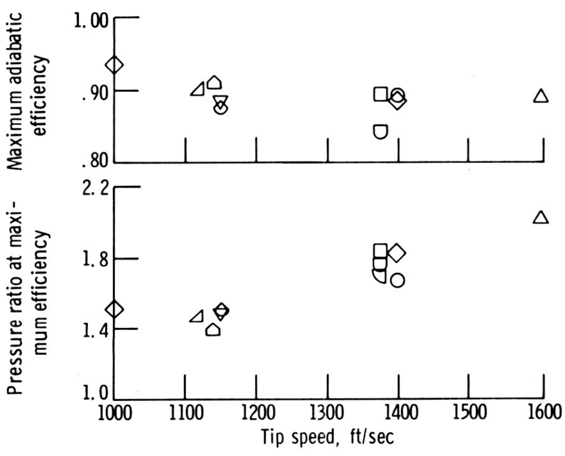

FIGURE 4.—*Experimental rotor performance at design tip speed.*

TRANSONIC BLADE SHAPES

The early transonic axial-flow compressors used double circular arc blade sections. For relative inlet Mach numbers above 1.2, however, this type of blade section was relatively inefficient. An analysis of passage shock losses for this type of blade (ref. 1) indicated that a passage shock could be assumed which was normal to the mean flow path and extended from the leading edge of one blade to the suction surface of the adjacent blade (fig. 5). Shock loss could be approximated by the loss across a normal shock at a Mach number equal to the average of the upstream

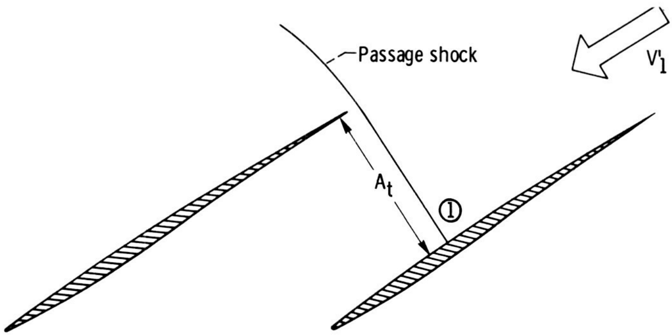

FIGURE 5.—*Typical transonic rotor blade section. Design criteria: minimum suction surface Mach number,* M_1'; *minimum throat area,* A_t.

Mach number and the Mach number at the intersection of the passage shock and the blade suction surface. This passage shock loss in some cases was over half of the total measured loss. For a given inlet relative Mach number, it is obvious that the suction surface Mach number must be reduced if shock losses are to be minimized.

The suction surface Mach number at point 1 in figure 5 is a function of the inlet relative Mach number and the amount of turning from the inlet free stream direction to the suction surface flow direction at point 1. Therefore, low shock losses require minimum suction surface turning upstream of the shock intersection point. As the suction surface turning is reduced, however, the passage throat area indicated as A_t in figure 5 decreases. Sufficient throat area must exist so as to pass the required mass flow. Otherwise, the passage will choke and the blade will be forced to operate at a positive incidence angle. This will increase the suction surface turning and the magnitude of passage shock loss. Therefore, the minimum suction surface turning must be consistent with a ratio of throat area to critical area, A_t/A^*, which is slightly greater than 1. The loss across the passage shock must be included in the determination of the critical area, A^*.

Based on the concept of minimum suction surface turning consistent with adequate throat area, most of the blade profiles for the rotors shown in figure 4 were composed of multiple circular arcs; that is, the pressure and suction surfaces of the forward or supersonic section of the blade are circular arcs with relatively small curvatures. The suction and pressure surfaces of the rear or subsonic portion of the blade are also circular arcs with relatively larger curvatures.

The preceding is a simplified discussion of the critical factors of supersonic blade shape and throat area requirements. In the actual case, blade blockage, streamline convergence in the radial direction, upstream extended compression or expansion wave effects, and boundary layer blockage must be evaluated. These factors are discussed in some detail in reference 2.

EFFECT OF THROAT AREA RATIO

As a part of an investigation of the effect of blade shape on performance, three rotors designed for a tip speed of 1400 feet per second were built and tested. The design speed performance and critical design parameters for these rotors are shown in figure 6. Rotor *1B* was designed for a tip diffusion factor of 0.35 and a small supersonic turning on the suction surface. The estimated suction surface Mach number at the shock point was 1.49 and the ratio of throat area to critical area was 1.08. This critical area ratio accounted for radial streamline convergence and for shock losses but did not account for boundary layer growth on the flow surfaces or any other losses. Rotor *2B* was designed for a tip diffusion factor of 0.45, an estimated maximum suction surface Mach number of 1.51, and

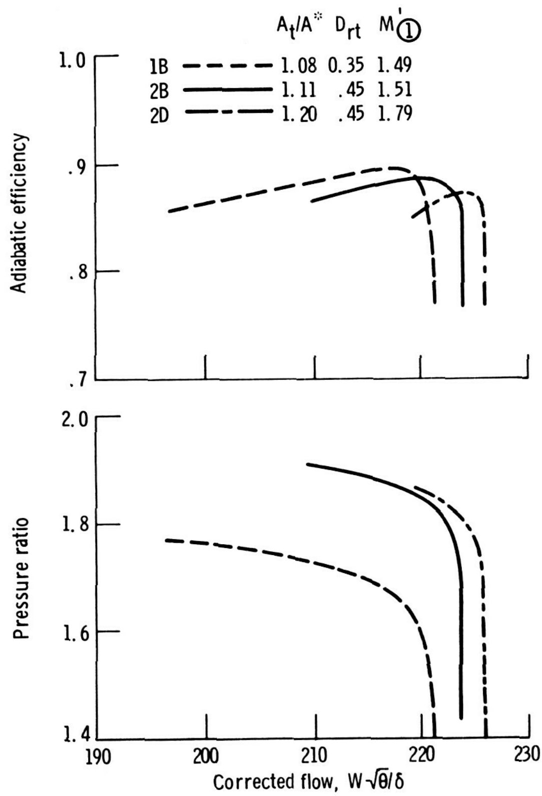

FIGURE 6.—*Effect of throat area ratio on rotor performance. Design speed.*

a critical area ratio of 1.11. Rotor *2D* was also designed for a tip diffusion factor of 0.45, but had an estimated maximum suction surface Mach number of 1.79 and a critical area ratio of 1.2. Additional details on the design and performance of these rotors are given in reference 3.

Comparison of the maximum efficiencies for these three rotors (fig. 6) shows that rotor *1B* is about one point better than rotor *2B* and two points better than rotor *2D*. The design inlet relative Mach number is the same for all three rotors; the suction surface Mach numbers for rotors *1B* and *2B* are comparable, whereas the suction surface Mach number for rotor *2D* is appreciably higher. It may be surmised that the differences in efficiency between rotors *1B* and *2B* may be due to the decrease in subsonic diffusion as a result of lower blade loadings and that the differences between rotors *2B* and *2D* may be due to differences in shock losses as a result of a lower average shock Mach numbers.

A rather surprising difference in performance of these three rotors is the difference in stable operating range. The minimum flow for each rotor represents the point of initiation of rotating stall. As can be seen from figure 6, rotor *1B* has the largest flow range, rotor *2B* the next largest, and rotor *2D* the smallest range. There appears to be a direct relationship between range and critical area ratio. A possible explanation for this range effect is based on the theory that the shock-boundary layer interaction at the existing values of suction surface Mach number are sufficiently high that flow separation must occur downstream of the shock. The passage velocity in the region of the shock probably accelerates to sonic levels. The larger the throat area ratio, the larger the extent of the separated region, and the more critical the flow reattachment and subsonic diffusion process. Therefore, a critical area ratio just sufficient to pass the mass flow not only results in a lower shock loss, but also improves the subsonic diffusion process. Therefore, gains in both efficiency and flow range are obtained.

It should be noted that the degree of supersonic suction surface turning consistent with flow choking is a function of relative inlet Mach number. For low Mach numbers, a fairly large turning is required; for a relative inlet Mach number on the order of 1.4, the suction surface turning can be practically zero, and, for still higher relative inlet Mach numbers, reverse turning can be used. This critical area variation with inlet Mach number, of course, means that the throat area ratio can only be optimized for design speed. The variation of efficiency with speed for these three rotors is shown in figure 7. At design speed and at 110 percent of design speed, the rotor with the smaller critical area ratio is most efficient. At 90 percent of design speed all three rotors have about the same maximum efficiency and at lower speeds the rotors with the larger throat areas have the best efficiency. Therefore, selection of throat area ratio will be dependent on the particular application. If maximum efficiency at design

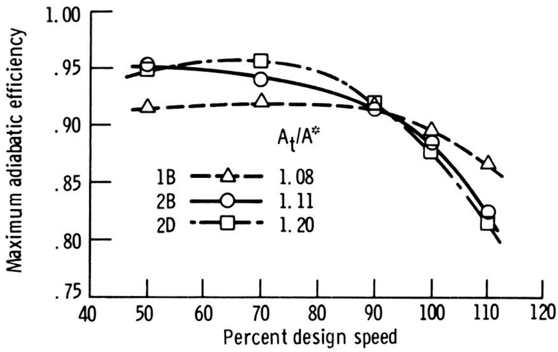

FIGURE 7.—*Effect of rotative speed on maximum efficiency.*

speed is of prime importance, the throat area ratio should be near 1.0. If, however, high efficiency at part speed is more significant, larger throat area ratios are desirable. The lowest area ratio for these studies was 1.08, but other rotors have been designed for area ratios of 1.03 to 1.05. (The deviation from a value of 1.00 is an allowance for boundary layer blockage.)

EFFECT OF CASING TREATMENT

In order to be useful for most turbojet engine applications, compressor stages must have appreciable stall-free-flow range and must be tolerant of severe distortions of inlet pressure. For example, the *2B* stage of figure 6 could be marginal from a range standpoint. Furthermore, most high aspect ratio stages have exhibited insufficient flow range for most applications. During tests of tip boundary layer control on a high aspect ratio rotor blade row (ref. 4), it was discovered that appreciable gains in flow range could be achieved by use of porous material over the rotor blade tips. Figure 8 shows the effect of this treatment on a highly loaded, 1375-feet-per-second rotor tested by NASA. The solid curves show the performance and stall limit for tests with a solid casing; the dashed curves are for the same rotor with casing treatment. In this case, the treatment consisted of 310 slots which were 0.070 inches wide, 0.30 inches deep, about a half blade chord long, and cut at approximately the angle of the blade tip as shown in the sketch in figure 8. As can be seen, this casing treatment had an extremely large effect on the rotor stall margin and did not seriously affect the rotor pressure ratio or maximum efficiency. Other configurations of porous casing also gave appreciable gains in flow range, but some had more adverse effects on pressure ratio and efficiency. The mechanism by which these range improvements are obtained is not understood, but they offer great potential as a means of attaining broad

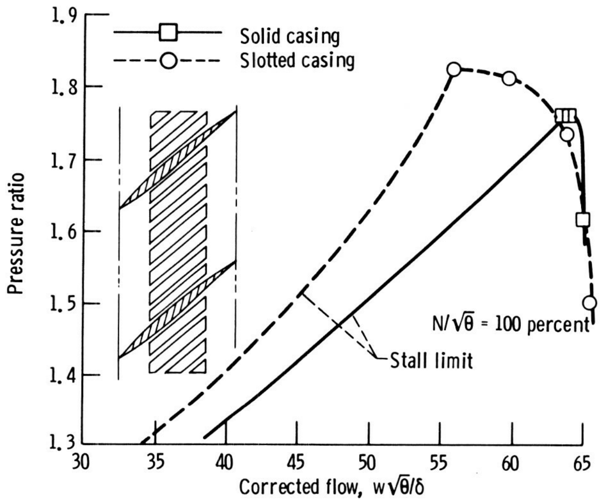

FIGURE 8.—*Effect of rotor casing treatment on stall limit.*

range for highly loaded transonic stages. Such treatments have been applied to complete stages, but have not been tried on multistage fans or compressors yet.

EFFECT OF SOLIDITY ON ROTOR PERFORMANCE

The preceding discussion has considered the losses associated with the supersonic portion of the transonic blade. In another phase of the NASA program, the effects of solidity on subsonic diffusion were considered. For this study, blade rows with tip solidities of 1.1, 1.3, and 1.5, were designed, built, and tested. All three rotors were designed for a tip speed of 1375 feet per second. The variation of basic blade shape is shown in figure 9 along with the design speed performance. As shown in the sketch, the forward or supersonic portions of the blade were the same and solidity was increased by extending the rearward or subsonic portion of the blade. All three rotors were designed for an overall pressure ratio of 1.65, and the camber of the rearward portion of the blade was selected accordingly. T'p section diffusion factors varied from 0.47 to 0.50.

Examination of the performance plots of figure 9 shows that the peak pressure ratio increased with increased solidity. The variation of adiabatic efficiency shows the high solidity rotor with a maximum efficiency of 0.89 at a pressure ratio of 1.82 to be the most efficient.

These data show the maximum efficiency of the 1.3 solidity rotor to be slightly lower than that for the 1.1 solidity rotor. This is believed to be a result of differences in midspan vibration dampers. The dampers on the

1.3 solidity rotor were very thick and poorly streamlined compared to those for the other two rotors. With comparable dampers, it is believed that the 1.3 solidity rotor maximum efficiency would fall between those for the other two rotors. These data clearly show the advantages of high solidity for highly loaded transonic rotors.

Based on conventional correlations of loss parameter versus diffusion factor for rotor tip sections, a lower value solidity would be expected to be optimum. To study this in more detail, existing data were reviewed to determine the influence of solidity on rotor blade tip losses. Conventional plots of loss parameter versus diffusion factor were made in which ranges of solidity were isolated. Figure 10 is such a plot for tip section data. Nominal values of solidity on the order of 0.6, 1.0, and 1.4 are indicated by different symbols. The loss parameter values plotted are

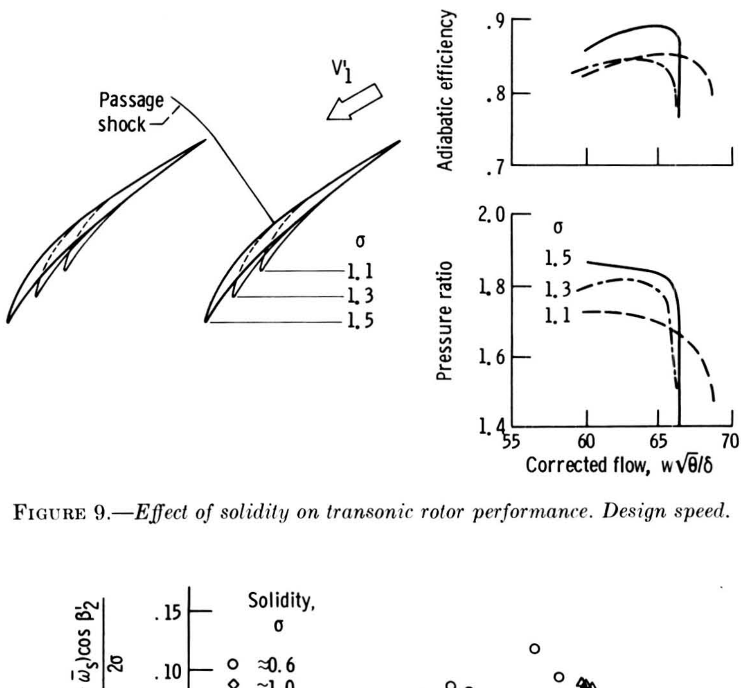

FIGURE 9.—*Effect of solidity on transonic rotor performance. Design speed.*

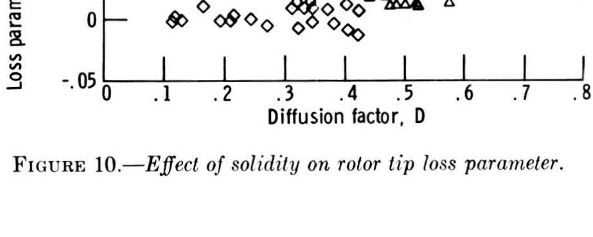

FIGURE 10.—*Effect of solidity on rotor tip loss parameter.*

the profile loss parameters obtained by subtracting estimated shock losses from the total measured losses. As usual for this type of data, there is an appreciable scatter. There is a definite trend, however, for the low solidity data to concentrate near the top of the band and the high solidity data near the bottom of the band for diffusion factors above 0.4.

Based on this data review, one might conclude that the correlation of loss parameter against solidity is not valid and that either solidity should not be included in the loss parameter or the exponent should be much larger than −1. Other forms of loss correlations have been tried with only limited success. Perhaps one source of difficulty is the lack of sufficient data from designs which are truly comparable. Most of the very low solidity data of figure 10 were obtained by simply removing vanes from rotors designed for higher solidities. Therefore, current concepts of optimized blade shape were not incorporated in the low solidity configurations. Secondary flow losses are a significant part of the losses in the rotor tip region, and the influence of these losses on the effects of solidity are probably not adequately accounted for in the existing correlations. It is obvious that much more study is required before solidity can be optimized for a given application. Rotor data as presented in figure 9, however, strongly indicate performance advantages for high solidities for highly loaded transonic rotors.

HIGH MACH NUMBER STATORS

As shown in figure 2, highly loaded rotor stages will result in highly loaded, high Mach number stators. NASA has an active program on high Mach number stators which applies the same principles of minimum shock loss and controlled passage throat area discussed for rotors. Initially this program consisted of a study of a family of transonic stators (ref. 5). Current effort consists of design and test of stators to match research rotors. A typical radial variation of loss parameter for a transonic stator from reference 5 is shown in figure 11. For this particular point, the Mach number varied from 0.8 at the tip to 1.0 at the hub and the hub diffusion factor was on the order of 0.65. From this figure it is evident that hub and tip losses are extremely high for this level of loading and inlet Mach number.

Figure 12 indicates the relative magnitude of the effects of Mach number and diffusion factor. This is a plot of loss parameter and diffusion factor versus inlet Mach number for the stator hub section. These data are also from reference 5. The solid curve represents total loss parameter and the dotted curve represents profile loss parameter. This profile loss parameter is obtained by substracting an estimated shock loss from the measured total losses. Stator diffusion factor is plotted as the dot-dashed

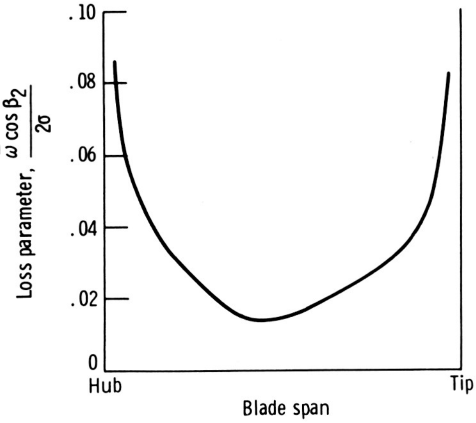

FIGURE 11.—*Radial distribution of stator loss. Inlet Mach number, 0.8 to 1.0; hub diffusion factor, 0.65.*

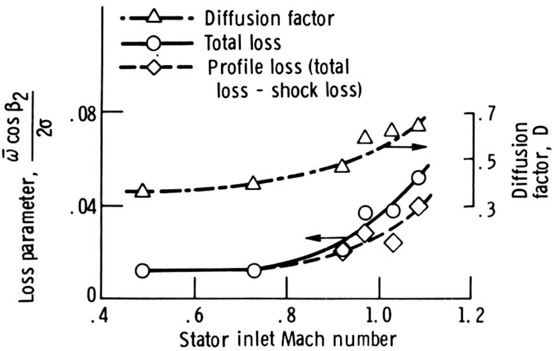

FIGURE 12.—*Effect of Mach number on stator loss.*

curve. As can be seen, an increase in stator Mach number is accompanied by an increase in diffusion factor. A comparison of the profile loss and total loss shows that at an inlet Mach number of 1.00, about 20 percent of the loss is estimated to be due to shock loss; the rest must be attributed to loading. At higher Mach numbers a larger proportion of the loss can be attributed to the effect of Mach number.

Several methods of reducing stator end losses through boundary layer control may prove profitable. One method is that of corner slit suction as reported in reference 6. This is shown schematically in figure 13. The suction slit is located at the intersection of the suction surface and the annulus wall. For this test, the slit was located at the hub end of the stator, extended from 15 to 85 percent of blade chord, and was 0.017 inches wide. The objective of this corner suction was to reduce corner flow separation and thus reduce losses. Results in terms of loss parameter versus span are also shown in figure 13. The circles show losses with the slits plugged, and the diamonds show losses with a slit flow of 0.2 percent of the total flow. Test conditions were a stator hub Mach number of 0.9 and a hub diffusion factor of 0.64. As can be seen, the application of this limited amount of slit suction appreciably reduced the losses in the hub region.

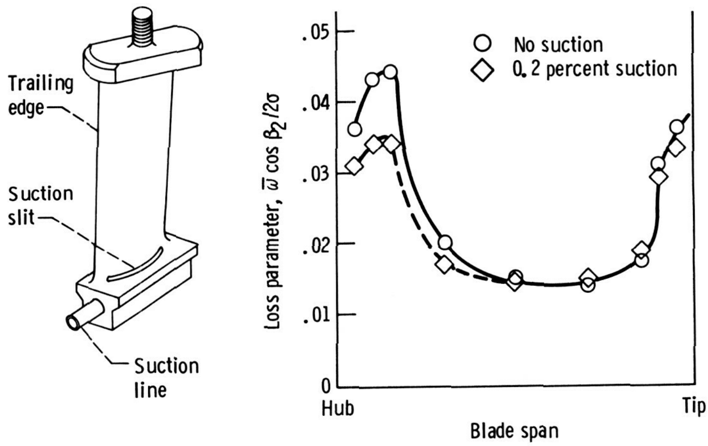

FIGURE 13.—*Effect of slit suction on stator losses. Stator hub Mach number, 0.9; hub diffusion factor, 0.64.*

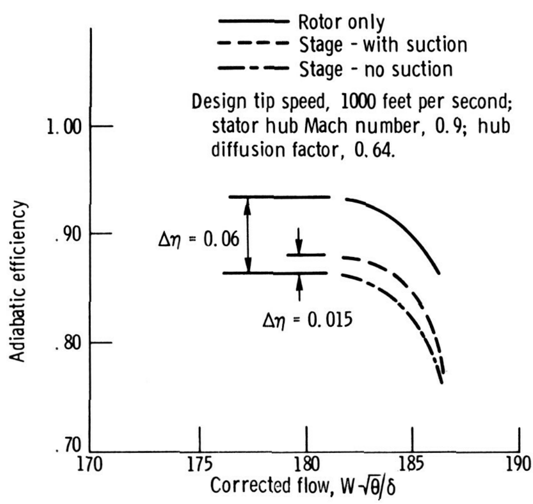

FIGURE 14.—*Effect of stator slit suction on stage performance. Design tip speed, 1000 feet per second; stator hub Mach number, 0.9; hub diffusion factor, 0.64.*

The effect of this stator loss reduction on stage efficiency is shown in figure 14. This particular stage is a highly loaded, low tip speed stage such as might be applied to a low noise, high bypass ratio fan engine. Design tip speed was 1000 feet per second, pressure ratio was 1.50, stator hub inlet Mach number was 0.9 and stator hub diffusion factor was 0.64. The solid curve is rotor efficiency, the dot-dashed curve is stage efficiency with no slit suction, and the dashed curve is stage efficiency with slit suction. This figure shows that the loss across the stator with no slit suction results in a 6-point drop in stage efficiency. Application of stator hub slit suction of 0.2 percent of total flow results in a gain of 1 to $1\frac{1}{2}$ points in stage efficiency. These data clearly show the need for efficient stators for highly loaded compressor stages.

SUMMARY REMARKS

To achieve light-weight, compact fans and compressors for advanced aircraft propulsion systems, it is desirable to obtain axial-flow compressor stages with increased pressure ratio per stage. Stage pressure ratio can be increased by increasing rotor tip speed, by increasing blade loading, or by some combination of these increases. Increased rotor tip speed at a given loading level increases the rotor relative Mach number, but has little effect on stator Mach number or loading. Increased rotor loading, however, results in increases in both stator Mach number and loading.

Very good efficiencies have been achieved for high tip speed transonic rotors by utilizing proper control of suction surface curvature and blade section throat area. Minimum curvature on the forward portion of the suction surface has been used to minimize shock losses. The minimum permissible curvature in this region is limited by choking requirements downstream of the passage shock. Choking limits change with tip speed, so that designs with throat areas near the limit at design speed will show maximum efficiency at design speed but be less efficient at lower speeds than designs with a larger margin from choke. Therefore, selection of the optimum throat area ratio will depend on the requirements of any particular application. Designs with minimum throat area margins also seem to exhibit maximum stall-free operating ranges.

Large gains in flow range without severe penalties in performance have been achieved by use of porous casing treatment over single rotor blade rows.

Increased solidity appears to have a beneficial effect for highly loaded transonic rotors. Highly loaded transonic stators have shown extremely high losses in the blade end regions. Limited tests with slit suction at the junction of the stator suction surface and the hub annulus wall have shown appreciable performance gains for relatively small suction flows.

LIST OF SYMBOLS

A_t Blade passage throat area
A^* Critical area
C Blade chord
D Diffusion factor
M Mach number
S Blade spacing
U_t Rotor tip speed
β_2 Exit flow angle
η Adiabatic efficiency
σ Blade row solidity
$\bar{\omega}$ Loss coefficient

Subscripts

rt Rotor tip
sh Stator hub

Superscripts

$'$ Relative to the rotor blade

REFERENCES

1. Schwenk, Francis C., George W. Lewis, and Melvin J. Hartmann, *A Preliminary Analysis of the Magnitude of Shock Losses in Transonic Compressors.* NACA RM E57A30, 1957.
2. Reddy, J. L., and J. F. Klapproth, *Advanced Fan Development for High Bypass Engines.* AIAA, Paper 68–563, June 1968.
3. Gostelow, J. P., K. W. Krabacher, and L. H. Smith, Jr., *Performance Comparisons of High Mach Number Compressor Rotor Blading.* NASA CR–1256, 1968.
4. Koch, C. C., *Experimental Evaluation of Outer Case Blowing or Bleeding of Single Stage Axial Flow Compressor*, Part VI, Final Report. NASA CR–54592, April 1970.
5. Keenan, M. J., and J. A. Bartok, *Experimental Evaluation of Transonic Stators.* Report PWA-3470, Pratt & Whitney Aircraft, NASA CR–72298, 1969.
6. Peacock, R. E., *Flow Control in Corners of Cascades.* Report ARC–27291, Aeronautical Research Council, Great Britain, October 8, 1965.

DISCUSSION

C. C. KOCH (General Electric Co.): As Mr. Benser discusses in his paper, significant improvements in compressor stall margin have been obtained by the use of porous casing treatments. Although these effects have been observed for some time (e.g., a casing treatment configuration that increased the stall margin of a centrifugal compressor was demonstrated and patented by the General Electric Company in 1945; ref. D–1), our understanding of them is based primarily on empirical results. Several hypotheses do exist concerning the physical processes by which casing treatment increases stall margin, but one must agree with Mr. Benser's conclusion that at present the mechanism is not adequately understood. However, it is likely that more insight into the casing treatment mechanism may result from greater understanding of the growth of casing boundary layers and the stability of these boundary layer flows as the stall limit is approached.

The photograph in figure D–1 is presented in the hope that it may contribute to this understanding. Lamp black and oil patterns on plexiglas rotor casing sections of a four-stage, low-speed research compressor are shown. In this figure the rotor moves from left to right and the through-flow is downward. On each window there are two crayon lines that indicate the leading-edge and trailing-edge planes of the rotor blades. At certain places crayon marks also indicate the trailing edges of upstream stator vanes. It can be seen that the flow direction in the rotor tip clearance region is primarily circumferential, but near the leading edge there is a through-flow component, while in the rear portions of the blades the flow frequently has an upstream component. These flows tend to coalesce at a location in the front half of each blade row. The pattern shown occurred when the compressor was near stall; with more open throttle positions the line of coalescence was further aft. This indicates that the extent of axial back flow increases in the casing boundary layer as a machine is throttled toward stall. Casing boundary layer measurements taken upstream and downstream of the third rotor confirm this and indicate that the displacement thickness of the axial-velocity casing boundary layer increases rapidly as stall is approached.

If, in fact, the growth and ultimate breakdown of the region of coalesced boundary layer flow were the cause of stall, then the above observations suggest that casing treatment may delay stall by reducing back flows or by preventing the forward movement of the coalesced region.

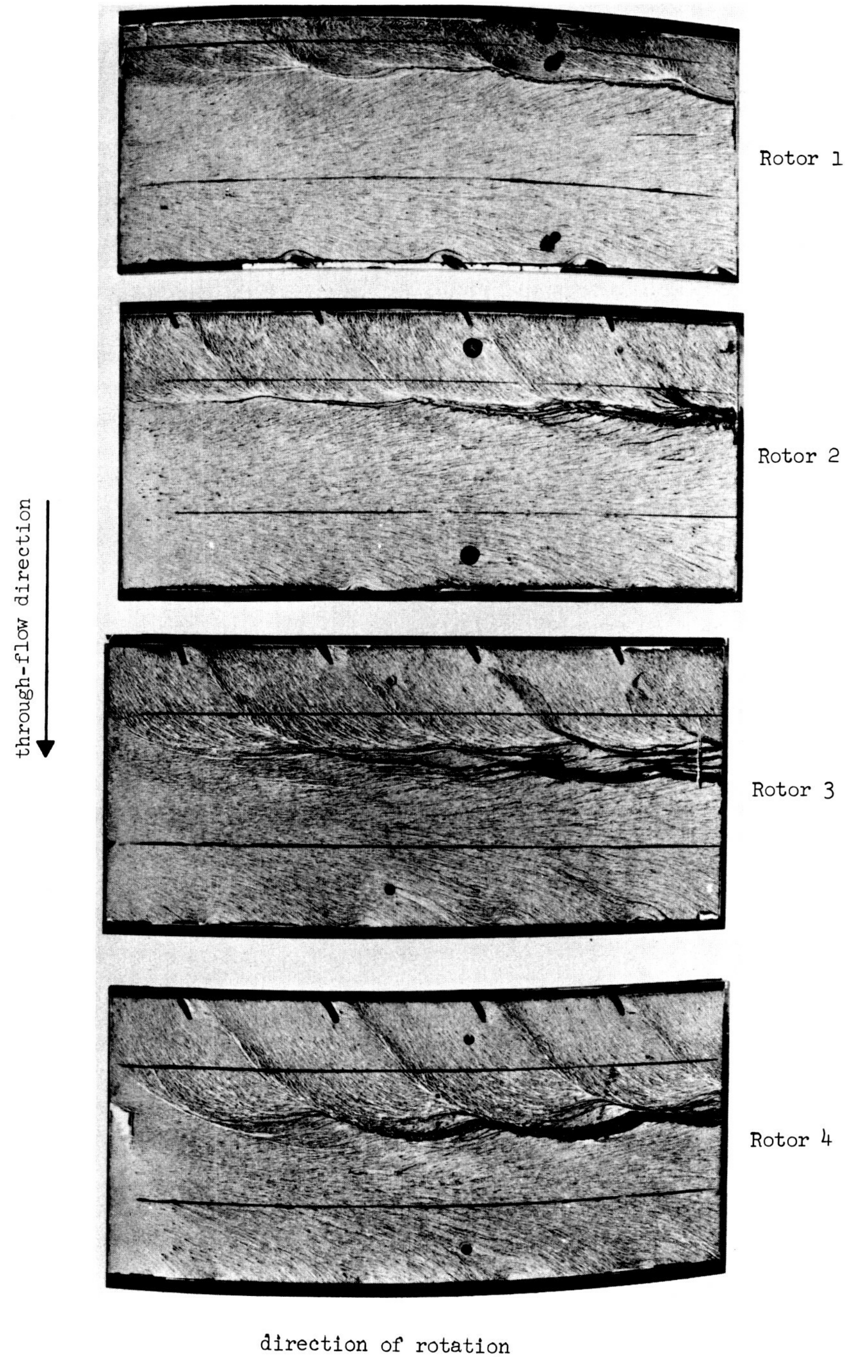

FIGURE D–1.—*Boundary layer flow visualization in rotor tip region using lamp black and oil.*

M. J. KEENAN (Pratt & Whitney): Mr. Benser presents a concise summary of the most significant results obtained in recent NASA investigation of high loading and high tip speed on compressor performance. The relationship between rotative speed (or Mach number) and critical area ratio is particularly interesting. The trend toward higher efficiency at high speeds and lower efficiency at part speed with decreasing area ratios is very strong in the 1600-ft/sec rotor described in reference D–2. This rotor was designed with $A_t/A^*=1.04$ for sections having supersonic relative Mach numbers. The small throat areas, obtained with negative front camber near the tip, forced the rotor to operate at high incidence and high loss at part speed. At a tip speed equivalent to 114-percent design speed in figure 7, this rotor reached its maximum efficiency, 0.890. Its peak efficiency at 800 ft/sec tip speed was 0.863, 2.7 percent lower than at design speed, and 9 percent lower than that of the $A_t/A^*=1.2$ rotor ($2D$) at the same tip speed.

High Mach number stator tests (ref. 5) showed similar results. Multiple circular arc and double circular arc stators were designed with the same incidence angles, but reduced front camber gave the multiple circular arc designs smaller throat areas. These multiple circular arc stators had a clear performance advantage at high Mach numbers, but averaged slightly higher losses at lower Mach numbers.

These data, in themselves, are not conclusive but appear to be an extension of Mr. Benser's observations on the effect of throat area on performance. They indicate a need for further analysis along the lines presented in reference D–3, in which bow wave losses are related to off-design incidence angles, leading to improved loss models.

E. BOXER (NASA Langley Research Center): The Lewis Research Center's compressor research program you have outlined is both comprehensive and well ordered. I have just one question to ask and several comments to make. The question concerns the curves presented in the first two slides. For the ordinate scale to be meaningful, it is necessary to know the assumed value of hub-tip radius ratio, tip solidity, radial distribution of both loss coefficient and work, and the maximum value of tip turning angle or annulus contraction through the rotor. Can they be provided?

The advantages of pursuing a compressor research program based essentially upon tests of rotors or complete stages are self-evident, in that the performance maps obtained are valid, containing as they do the three-dimensional effects of off-design operation, secondary and boundary layer flows, and shock-boundary layer interactions which are only qualitatively predictable. Unfortunately, with this approach, even with peripheral sensors, there is a lack of detailed knowledge of the fluid behavior within the blade passage. Without such knowledge, extrapolation of compressor performance into regions beyond current experience is subject to wide deviations. In addition, for a series of research rotors designed for a given task, it is almost impossible to vary one significant design parameter without subtle changes in other variables which may have an appreciable effect upon the reported trends. For example, the stated influence of throat area ratio upon maximum efficiency at off-design speeds, based upon the data of figure 7, ignores the variation in relative turning angle and overall annular area contraction between rotors *2B* and *2D* (which are identical except for meanline shape) with that of rotor *1B*. It appears likely that throat area ratio is only one of several geometric variables responsible for the observed trend and may not be the principal responsible variable. In fact, rotors *2B* and *2D*, which have the greatest divergence in design throat area ratio, are remarkably similar in their speed-efficiency relationship, whereas rotor *1B* exhibits a distinctly different slope with only a third of the throat area change from that of rotor *2B* as compared to rotors *2B* and *2D*.

To make the NASA-Lewis program more meaningful to compressor designers and to permit extrapolation for conditions beyond the scope of rotor tests, I feel that more fundamental experimental studies should complement the program. A recent paper (ref. D–4) indicates that with careful attention to small details supersonic cascade data can correlate with rotor data. I would suggest that a useful adjunct to a well-rounded program would be a continuation of supersonic cascade testing.

F. GILMAN (Worthington): It may be of interest to this audience to know that Dr. Wislicenus was one of the first to suggest that transonic and supersonic axial-flow compressors could have practical value and were worthy of investigation. The wheel which Dr. Wislicenus designed for transonic relative flow had a 10-inch tip diameter and a constant $7\frac{1}{8}$-inch hub diameter. The five blades of this impeller were milled with straight helical surfaces on their pressure faces and single-circular-arc profiles for suction faces. An overall compressor efficiency of 78 percent was obtained with a tip velocity of 1050 feet per second, and the pressure ratio at best efficiency was 1.30. Stable operation extended between 60 and 90 CFS. Discovery of the workability of this transonic range in 1946 was new and interesting and formed the basis for a later three-stage transonic axial-flow test program at Worthington.

BENSER (author): The assumptions used in the calculations of figures 1 and 2 were as follows:

(1) Hub-tip ratio at the rotor inlet equal 0.5
(2) Constant work input from root to tip
(3) Constant polyhopic efficiency of 0.9 root to tip
(4) Axial velocity ratio of 1 across the rotor blade row
(5) Simple radial equilibrium

(6) Rotor tip solidity equal 1.5
(7) Axial velocity equal 600 ft/sec
(8) Zero inlet whirl

The analysis based on these assumptions is extremely simplified and is used to indicate trends rather than to define exact values of design or performance parameters. A more complete analysis of the effects of variations of design variables is given in reference D–5.

As stated by Mr. Boxer, it is extremely difficult to isolate the effects of variations of a single parameter by comparison of test results from single rotors or single stages, because variations cannot be restricted to a single parameter. By such comparisons, however, it is hoped that some empirical design rules can be derived which will lead to improved design control. The throat area ratio concept is such an empirical rule which has currently been applied to designs with considerable success. The data of figures 6 and 7 show the effect of throat area ratio on efficiency and range. Rotor and stator data discussed by Mr. Keenan as well as other NASA data substantiate the trends shown in these figures.

The flow processes within transonic axial-flow compressor blade rows are extremely complex and our design concepts, which are essentially axisymmetric approaches, are certainly limited where strong shocks exist in the blade-to-blade planes. Supersonic cascades may be useful in studying details of flow in the blade-to-blade plane. More detailed flow measurements within the rotating blade row would also be helpful. To date, the NASA Lewis research has been concentrated on single rotor and single stage tests. The NASA rotor and stage program has been supplemented by theoretical studies, but no supersonic cascade tests have been run.

In our attempts to determine the mechanism by which casing treatment improves stall margin, we have postulated many models, as Mr. Koch has stated. To date, no single model has been found which explains all of the results obtained. This may be due to the fact that we have not as yet set up the proper model, or it may be due to the fact that there is more than one mechanism involved. Many empirical rules for casing treatment have been evolved, but more understanding of the phenomenon is necessary in order to optimize designs for any given application.

REFERENCES

D–1. McMahan, K. D., and N. P. Bailey, *Centrifugal Compressor.* U.S. Patent 2,453,524, application date May 31, 1945.

D–2. Sulam, D. H., M. J. Keenan, and J. T. Flynn, Data and Performance, Multiple Circular Arc Rotor. *Single Stage Evaluation of Highly-Loaded High-Mach-Number Compressor Stages, Part II,* NACA CR 72694, 1970.

D-3. Miller, G. R., and M. J. Hartmann, *Experimental Shock Configurations and Shock Losses in a Transonic-Compressor Rotor at Design Speed.* NACA RM E58A14b.

D-4. Mikolajczak, et al., ASME paper 70-GT-79. Presented in Brussels in the spring of 1970.

D-5. Miller, M. L., and A. C. Bryans, *Parametric Study of Advanced Multistage Axial-Flow Compressors.* NASA CR 797.