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RESEARCH MEMORANDUM

EFFECT OF MACH NUMBER ON OVER-ALL PERFORMANCE OF SINGLE-STAGE AXIAL-FLOW COMPRESSOR DESIGNED FOR HIGH PRESSURE RATIO

By Charles H. Voit, Donald C. Guentert
and James F. Dugan

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON
July 10, 1950

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

EFFECT OF MACH NUMBER ON OVER-ALL PERFORMANCE OF SINGLE-STAGE AXIAL-FLOW COMPRESSOR DESIGNED FOR HIGH PRESSURE RATIO

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SUMMARY

A complete stage of an axial-flow compressor designed to produce a high pressure ratio by use of the optimum combination of high blade loading and high relative inlet Mach number was investigated at speeds from 110 to 130 percent of design speed (836 ft/sec); the data obtained and the results of a previous investigation of the same compressor at speeds from 50 to 100 percent of design speed were used to determine the effect of Mach number on over-all performance. The blading of this compressor was designed on the basis of low-speed cascade data and did not account for any variation of blade performance with Mach number. This investigation covered a range of relative inlet Mach numbers at the mean radius of the rotor blades from 0.34 to 0.91.

At the highest equivalent tip speed investigated (1088 ft/sec), a maximum total-pressure ratio of 1.635 was obtained at an adiabatic efficiency of 0.74; at the peak-efficiency point for this speed, a total-pressure ratio of 1.628 was obtained at an adiabatic efficiency of 0.75.

The peak adiabatic efficiency gradually decreased from 0.93 to 0.89 as the relative inlet Mach number at the mean radius of the rotor blades increased from 0.34 to 0.77 and the total-pressure ratio increased from 1.095 to 1.515. Above a relative Mach number of 0.77, the efficiency rapidly decreased. At the highest speed investigated, the peak efficiency decreased to 0.75 at a relative inlet Mach number at the rotor inlet of 0.91.

The angle of attack at the mean radius of the rotor inlet for peak efficiency was nearly constant up to a Mach number of 0.68 and was approximately 5.5° above the design angle of attack obtained from cascade data. At Mach numbers above 0.68, the angle of attack for maximum efficiency decreased and approached a minimum value approximately 1.4° above the design value.

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INTRODUCTION

In order to obtain a maximum pressure ratio from an axial-flow compressor stage with a given stagger angle, it is necessary to use the optimum combination of relative inlet Mach number and blade loading. The performance of such an axial-flow compressor stage using NACA 65-series blower-blade sections and designed on the basis of the combination of blade camber and the resulting critical Mach number that would produce the maximum pressure ratio is reported in reference 1. (The critical Mach number is that inlet Mach number at which sonic velocity is obtained on the blades.) According to cascade data obtained at the NACA Langley laboratory, however, the critical Mach number may be exceeded up to the force-break Mach number without a sharp increase in losses. Above the force-break Mach number, a sharp increase in drag and decrease in lift occur. The range of angle of attack for low drag, however, decreases sharply above the critical Mach number and limits the efficient flow range at high Mach numbers.

The single-stage compressor described in reference 1 was therefore investigated at the NACA Lewis laboratory at speeds up to 130 percent of design speed (836 ft/sec) in order to determine the limiting Mach number and pressure ratio and to determine the effect of Mach number on efficiency, flow range, and optimum operating flow conditions.

Investigations were made at equivalent tip speeds of 920, 962, 1004, 1046, and 1088 feet per second corresponding to 110, 115, 120, 125, and 130 percent of design speed, respectively. From these data and the results of reference 1, the variation of total-pressure ratio, efficiency, and relative Mach number at the inlet of the rotor blades at the mean radius was determined over a range of equivalent tip speeds, and the limiting practical value of relative inlet Mach number at the rotor inlet for this blade design was ascertained. The range of practical operation of the compressor stage was also evaluated.

SYMBOLS

The following symbols are used in this report:

P total pressure, pounds per square foot absolute

U blade velocity at any radius, feet per second

1308

$U_t/\sqrt{\theta}$ equivalent tip speed corrected to standard NACA sea-level conditions, feet per second

W weight flow, pounds per second

$W\sqrt{\theta}/\delta$ weight flow corrected to standard NACA sea-level conditions, pounds per second

δ ratio of absolute pressure to standard NACA sea-level absolute pressure

η_{ad} mass-flow weighted average adiabatic efficiency

θ ratio of absolute temperature to standard NACA sea-level absolute temperature

Subscripts:

0 inlet measuring station

3 outlet measuring station

t tip radius

APPARATUS AND PROCEDURE

Blade design. - The blading of this compressor was the same as that used in reference 1. The rotor and stator blades were designed to produce the maximum pressure ratio by use of the optimum combination of blade loading and critical Mach number. The variation of critical Mach number with camber was obtained from two-dimensional high-speed cascade data of the NACA Langley laboratory.

The blade sections and the angle-of-attack settings were obtained from low-speed cascade data (references 2 and 3) to give the flattest pressure distribution for the desired turning angle. Because of the flat pressure distribution of these design values, they represent the approximate angles of attack for maximum critical Mach number.

The following assumptions were used in the compressor design:

(1) Velocity diagram based on a wheel-type rotation added by the inlet guide vanes, a vortex addition by the rotor, and a symmetrical velocity diagram at the mean radius; rotation added by the rotor removed by the stator

(2) Ratio of axial velocity at the mean radius to rotor tip speed of 0.6, corresponding approximately to maximum power-input conditions

(3) Simple radial equilibrium of pressure from hub to tip at the entrance to each blade row 1308

(4) Relative Mach number at the mean radius entering the rotor equal to the absolute Mach number at the same radius entering the stator

The stage consisted of a row of 40 guide vanes, 44 rotor blades, and 46 stator blades. The guide vanes were variable-chord, circular-arc sheet-metal vanes with a radius of curvature of 4.25 inches. The rotor blades were variable-camber, 6-percent thick NACA 65-series blower-blade section with a camber that varied from 13.1 at the tip to 20.8 at the hub. The stator blades were a constant-camber NACA 65-(13.8)06 section. The variation of stator-blade turning angle from hub to tip was very small and a constant-camber section, with the camber and the angle of attack for the flattest pressure distribution at the mean radius, was used.

A summary of the blade design is presented in the following table:

Guide vane			
	Hub	Mean	Tip
Radius ratio, leading edge	0.747	0.906	1.000
Chord, in.	1.43	1.76	1.97
Included angle, deg	19.3	23.9	26.8
Solidity	1.740	1.765	1.790
Incident angle, deg	0	0	0
Blade thickness, in.	0.06	0.06	0.06
Blade section	Circular arc, 4.25-in. radius		
Number of blades 40			

Rotor blade			
	Hub	Mean	Tip
Radius ratio, leading edge	0.800	0.906	1.000
Turning angle, deg	39.7	31.2	23.4
Angle of attack, deg	25.6	18.7	15.7
Relative inlet Mach number	0.668	0.704	0.740
Stagger angle, deg	45.0	49.1	52.8
Chord, in.	1.35	1.35	1.35
Solidity	1.69	1.50	1.35
Blade section	65-(20.8)06	65-(16.0)06	65-(13.1)06
Number of blades 44			

Stator blade			
	Hub	Mean	Tip
Radius ratio, leading edge	0.832	0.906	1.000
Turning angle, deg	31.2	30.7	29.5
Angle of attack, deg	17.4	16.75	15.8
Stagger angle, deg	47.2	47.7	49.1
Absolute inlet Mach number	0.734	0.702	0.664
Chord, in.	1.35	1.35	1.35
Solidity	1.69	1.56	1.41
Blade section	65-(13.8)06	65-(13.8)06	65-(13.8)06
Number of blades 46			

These blades were installed in a variable-component axial-flow compressor that had a constant tip diameter of 14 inches and a hub diameter that varied from 10.464 inches at the guide-vane inlet to 12.100 inches at the outlet measuring station, approximately 0.9 chord-length downstream of the stator blades.

Installation. - A schematic diagram of the compressor installation is shown in figure 1. Air was drawn from the test cell through a thin-plate orifice mounted on the end of an orifice tank and flowed through a butterfly valve into a depression tank 4 feet in diameter and 6 feet long. A series of screens and a 3- by 3-inch honeycomb were used in the depression tank to obtain smooth uniform flow into the compressor bellmouth inlet. The air was discharged through a collector into dual outlets connected to the laboratory exhaust system. The air flow and the inlet-tank pressure were controlled by butterfly valves in the outlet and inlet ducting. The compressor was driven by a 400-horsepower, 20,000-rpm dynamometer.

Instrumentation. - The instrumentation used in this investigation was the same as that described in reference 1. Instrumentation was provided in the depression tank, station 0; after the guide vanes, station 1; and at the compressor outlet, station 3. No measurements were made after the rotor, station 2.

Preliminary circumferential surveys were made in order to locate the survey instruments so that they were removed from the wakes of upstream stationary blades and instruments.

At station 0, the pressure and temperature measurements were assumed to be stagnation values because of the very low velocities in the depression tank. At station 1, approximately 1/4 chord-length upstream of the rotor blades, measurements were made of total pressure, static pressure, and flow angle for six radial positions.

Compressor-outlet measurements of total pressure, static pressure, flow angle, and total temperature were made at station 3. Total-pressure measurements were made with 15-tube circumferential rakes, 1308 which covered a complete blade passage, at each of four radial positions. Static-pressure measurements were made with a single radial survey. Total-temperature measurements were made with four radial thermocouple rakes, spaced around the periphery, that were connected differentially with those at station 0 in such a manner as to measure an average circumferential value of the temperature rise across the compressor at each of four radii. Flow-angle measurements were made with a single radial survey.

Air flow through the compressor was measured with a thin-plate orifice. Compressor speed was measured by an electronic-type tachometer.

A more complete description of the instrumentation is given in reference 1.

Procedure. - The equivalent tip speed $U_t/\sqrt{\theta}$ was varied from 920 to 1088 feet per second, corresponding to a range of 110 to 130 percent of design speed (836 ft/sec). At each speed the air flow was varied from the maximum obtainable to a point near surge, but the two highest speeds were not extended to surge. For this investigation, it was necessary to maintain the pressure in the inlet depression tank at 15 inches of mercury absolute because of the power limitations of the drive equipment. The investigation reported in reference 1 was made at a constant inlet-tank pressure of 25 inches of mercury absolute. The range of Reynolds number, based on blade chord, for this investigation was approximately 258,000 to 290,000 and that for reference 1 was 210,000 to 396,000. Check points were made at 80 percent of design speed at inlet pressures of 15 and 25 inches of mercury absolute, which agreed well and indicated that the effect of the reduced inlet pressure on the performance of this single-stage unit at the higher speeds was negligible.

Methods of calculation. - The total-pressure ratio used in this investigation was obtained from a mass-flow weighted average of the isentropic energy input integrated across the flow passage (reference 4).

The adiabatic efficiency used in evaluating the compressor performance was calculated from a mass-flow weighted average of the total-temperature rise across the compressor and a mass-flow weighted average of the isentropic energy addition (reference 4).

RESULTS AND DISCUSSION

Over-All Total-Pressure Ratio and Adiabatic Efficiency

Over-all total-pressure ratio P_3/P_0 is plotted against corrected weight flow $W\sqrt{\theta}/\delta$ in figure 2. The data presented in reference 1 are included in this figure for comparison. The dashed curve indicates the flow condition for design angle of attack of the air entering the rotor at the mean radius; this condition may be considered to be the operating point at which the main portion of the blade most nearly approaches design conditions because the radial distribution of angle of attack approached design closely except near the hub and the tip. This design angle of attack corresponds to the design point of references 2 and 3, which is the angle of attack for the flattest pressure distribution for this blade section. The maximum-flow limit approaches the design flow condition as the speed is increased and, at the highest speed (1088 ft/sec), design flow conditions did not occur within the obtainable flow range. The total flow range also decreased as the speed increased. The flow range at the two highest speeds was not established because the flow was not decreased to surge in order to avoid the possibility of blade failure due to the combination of vibrations and the high stresses resulting from the high tip speeds.

The total-pressure ratios and the weight flows for peak efficiency are shown by the dot-dash line crossing the speed curves in figure 2. The peak efficiency and the peak pressure ratio occur at approximately the same weight flow at all speeds up to design speed. The flow range from peak efficiency to surge decreases as the speed increases up to design speed (836 ft/sec). Above design speed, the peak-efficiency point shifts toward the high-flow region and approaches the design flow condition at an equivalent tip speed of 1004 feet per second.

The mass-flow weighted average adiabatic efficiency for all speeds investigated for this compressor is plotted against corrected weight flow in figure 3. The design flow condition at the mean radius entering the rotor is indicated by a bar on each efficiency curve. At equivalent tip speeds from 669 to 962 feet per second, corresponding to 80 to 115 percent of design speed, respectively, the efficiency is nearly constant at each speed over an appreciable range of corrected weight flow corresponding to an angle-of-attack range of approximately 9°. Above design speed (836 ft/sec), the point of maximum efficiency shifts toward design flow condition indicating that the angle of attack for maximum lift-drag ratio decreases with increasing Mach number.

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Constant-efficiency contour lines are superimposed on the plot of total-pressure ratio against corrected weight flow in figure 4. A maximum pressure ratio of 1.635 was obtained at an equivalent tip speed of 1088 feet per second with an adiabatic efficiency of 0.74 (see fig. 3), and a pressure ratio of 1.628 was obtained with an adiabatic efficiency of 0.75 at the maximum-efficiency point for this speed.

Variation of Over-All Performance

with Tip Speed and Mach Number

The variation of total-pressure ratio, adiabatic efficiency, and relative Mach number at the mean radius of the rotor inlet with equivalent tip speed is shown in figure 5 for two flow conditions. The solid curves represent the flow condition for design angle of attack, corresponding to the design point of references 3 and 4 for this blade section, at the rotor-inlet mean radius. The dashed curves represent the flow condition for maximum efficiency at each speed. By use of the Mach number curves, the compressor performance at the two flow conditions can be compared on the basis of the same Mach number at the mean radius at the rotor inlet as shown by the vertical lines. The mean-radius Mach number is used to represent the operating Mach number of the entire blade. The Mach number at the tip ranged from 0.02 at the low speeds to 0.05 at the higher speeds above those at the mean radius, and at the hub ranged from 0.08 at low speeds to 0.10 at high speeds below those at the mean radius.

The efficiency curves in figure 5 are fairly flat over a range of tip speeds from 418 to 920 feet per second, corresponding to a Mach number range of approximately 0.34 to 0.77, with the maximum-efficiency curve decreasing from 0.93 to 0.89. Above a tip speed of 920 feet per second, the efficiency decreases rapidly.

The total-pressure ratio for design angle of attack at the rotor-blade mean radius increases steadily with tip speed up to 1010 feet per second, corresponding to a relative Mach number of 0.87, beyond which a decrease in pressure ratio occurs. At the highest speed investigated (1088 ft/sec), design angle of attack at the mean radius did not occur within the obtainable flow range. The pressure ratio obtained at the maximum-efficiency operating point increases from 1.095 to 1.635 as the relative inlet Mach number increases from 0.34 to 0.91 and is higher than that obtained at the design angle of attack at all speeds. This pressure-ratio curve tends to flatten out in the high Mach number region, but does not drop off as does the pressure-ratio curve for design angle of attack.

The operating point giving the highest pressure ratio without a sharp decrease in efficiency occurs at an equivalent tip speed of 920 feet per second, at which point a total-pressure ratio of 1.515 was obtained at an adiabatic efficiency of 0.89. This point corresponds to 110 percent of design speed and a relative inlet Mach number at the mean radius of the rotor blade of 0.77.

The inflection in the pressure-ratio curve for maximum efficiency at an equivalent tip speed of 836 feet per second, corresponding to a relative Mach number entering the rotor of 0.68, is caused by a decrease in the angle of attack for maximum efficiency, as shown in figure 6.

The variation in angle of attack at the rotor mean radius with relative Mach number at the rotor inlet is shown in figure 6 for three efficiency conditions: (1) maximum efficiency, (2) an efficiency of 0.85, and (3) an efficiency 0.02 below the maximum efficiency at any Mach number. The design angle of attack is also shown for comparison. In this figure, the flow conditions at the inlet to the rotor at the mean radius are used to indicate the operating point of the compressor. Because the radial distribution of angle of attack and Mach number approached design closely except near the hub and the tip, and because the angle of attack for each blade section was designed for the flattest pressure distribution by the use of a variable-camber blade, the operating point of the mean section is probably representative of the entire passage.

An appreciable range of weight flow over which the change in efficiency is small exists at speeds from 80 to 110 percent of design speed, making it somewhat difficult to select the maximum-efficiency point. Consideration of the trend of the efficiency curves with speed, however, indicates that the angle of attack for maximum efficiency is nearly constant up to a Mach number of 0.68 and is approximately 5.5° above design angle of attack.

Above a Mach number of 0.68, the angle of attack for maximum efficiency decreases and approaches a minimum value approximately 1.4° above the design angle of attack. This trend would be expected from a consideration of the effect of Mach number on the low-drag range of an airfoil section. At low Mach numbers, this range is defined by a minimum and a maximum angle of attack beyond which stall and flow separation occur with resulting high losses. If compressibility effects did not exist, these limits of angle of attack would remain approximately constant at all Mach numbers. For the blading of this compressor, the trend of these limits is represented by the approximately constant values of angle of attack for an efficiency of 0.85 at the low Mach numbers, equal to approximately $8\frac{1}{2}^{\circ}$ above and 5° below design angle of attack.

Because of compressibility effects, however, shock losses occur at the extremities of the low-drag range as the force-break Mach number is reached, resulting in a decrease in the low-drag operating range at the higher Mach numbers, as well as an increase in drag over the entire operating range. The decrease in low-drag operating range with increasing Mach number is shown in figure 6. For an efficiency of 0.85 or higher (fig. 6), the range of possible angle of attack decreases rapidly above a Mach number of about 0.70 and 0.84 was the highest Mach number at which an efficiency of 0.85 could be obtained. The decrease in pressure ratio obtained at design angle of attack at the high Mach numbers (fig. 5) is probably due to this decrease in low-drag range because the design angle of attack is apparently too low to fall within the low-drag region at these Mach numbers. Because the lift coefficient of a typical airfoil section increases appreciably beyond the angle of attack for minimum drag, the angle of attack for maximum lift-drag ratio, or maximum efficiency, lies near the high-angle-of-attack end of the low-drag range. The angle of attack for maximum efficiency would therefore be expected to decrease with the decrease in the low-drag range encountered at the high Mach numbers, as shown in figure 6, and to approach the angle of attack for maximum force-break Mach number. Results of a British cascade investigation show a similar trend in the performance characteristics at high Mach number for the airfoil sections studied in that investigation.

Because the design angle of attack for the blading in this compressor was based on the angle of attack giving the flattest pressure distribution and because the flattest pressure distribution would be expected to permit a maximum force-break Mach number, the angle of attack for maximum efficiency would be expected to coincide with the design angle of attack at the highest Mach number. That they do not coincide may be caused by several possible reasons: (1) Difficulty may be encountered in selecting the exact angle of attack giving the flattest pressure distribution at low Mach numbers; (2) the angle of attack giving the flattest pressure distribution may be different at high Mach numbers from that at low Mach numbers at which the design

angle of attack was determined; (3) mismatching of the rotor and the stator could cause the maximum efficiency of the rotor-stator combination to occur at a different angle of attack from that for either row taken separately; and (4) the angle of attack plotted in figure 6
1308 does not account for any change in axial velocity across the blade row; the effect of this change in axial velocity, however, would probably be to increase the discrepancy.

Application to Multistage Compressors

A stage of an axial-flow compressor can be designed to produce a high pressure ratio with an acceptable efficiency by use of a proper combination of high blade loading and high relative Mach number (reference 1). The present investigation shows that even higher pressure ratios could be obtained by a further increase in Mach number if some sacrifice in flow range and efficiency were acceptable at these high Mach numbers.

In the application of such a high-pressure-ratio stage to a multistage compressor, however, the reduction in the stage operating range may be undesirable because of stage-matching problems. For this reason, limiting the operating Mach number and stage pressure ratio is perhaps advantageous. In addition, because the rotational speed of a compressor is established by the limiting Mach number in the inlet stages where the temperature is low, it may be impossible to operate the later stages at high equivalent speeds. Operation of the later stages near the force-break value of relative Mach number may be accomplished by increasing the axial component of velocity, but this method is ineffective in obtaining high stage pressure ratios.

In the design of an axial-flow compressor stage for multistage application in a jet engine, an adequate range with good efficiency under part-speed operating conditions is necessary for good starting and accelerating characteristics. The operating line of an axial-flow compressor in a jet engine is usually such that the angle of attack of the inlet stages will increase with a decrease in engine speed. Conversely, the lower compression at part-speed operation will cause the angle of attack of the later stages to decrease from that at rated speed. Designing the inlet stages for an angle of attack near the lower limit of the good-efficiency range and the later stages near the upper limit is therefore desirable in order to remain in the good-efficiency range at part-speed operation. In compressors designed to operate at extremely high Mach numbers, however, matching of the blade rows is very critical because of the reduced low-drag range, and the angle of attack must be that for maximum force-break Mach number. 1308

SUMMARY OF RESULTS

A single-stage axial-flow compressor designed for a high pressure ratio was operated over a range of relative inlet Mach numbers at the mean radius of the rotor blades from 0.34 to 0.91 at speeds from 110 to 130 percent of design speed (836 ft/sec). The data obtained and the results of a previous investigation of the same compressor at speeds from 50 to 100 percent of design speed were combined and are summarized as follows:

1. A maximum total-pressure ratio of 1.635 with an adiabatic efficiency of 0.74 was obtained at an equivalent tip speed of 1088 feet per second and, at the peak-efficiency point for this speed, a pressure ratio of 1.628 was obtained at an efficiency of 0.75.

2. At equivalent tip speeds corresponding to 80 to 115 percent of design speed, the efficiency was nearly constant at a given speed over an appreciable range of weight flow. This flow range corresponded to a rotor angle-of-attack range of approximately 9°.

3. The peak efficiency gradually decreased from 0.93 at a relative Mach number of 0.34 at the rotor inlet to 0.89 at a relative Mach number of 0.77, and above a Mach number of 0.77 the efficiency rapidly decreased.

4. The operating point giving the highest pressure ratio without a sharp decrease in efficiency occurred at 110 percent of design speed, at which a total-pressure ratio of 1.515 with an adiabatic efficiency of 0.89 was obtained. This corresponded to an equivalent tip speed of 920 feet per second and a relative inlet Mach number at the mean radius of the rotor of 0.77.

5. The angle of attack at the mean radius entering the rotor for maximum efficiency was nearly constant up to a relative Mach number of approximately 0.68 and was approximately 5.5° above the design value obtained from low-speed cascade data. Above a Mach number of 0.68, the angle of attack for maximum efficiency decreased and

approached a minimum value approximately 1.4° above the design value. At the higher Mach numbers, the angle-of-attack range for good efficiency rapidly decreased.

1308

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National Advisory Committee for Aeronautics,
Cleveland, Ohio.

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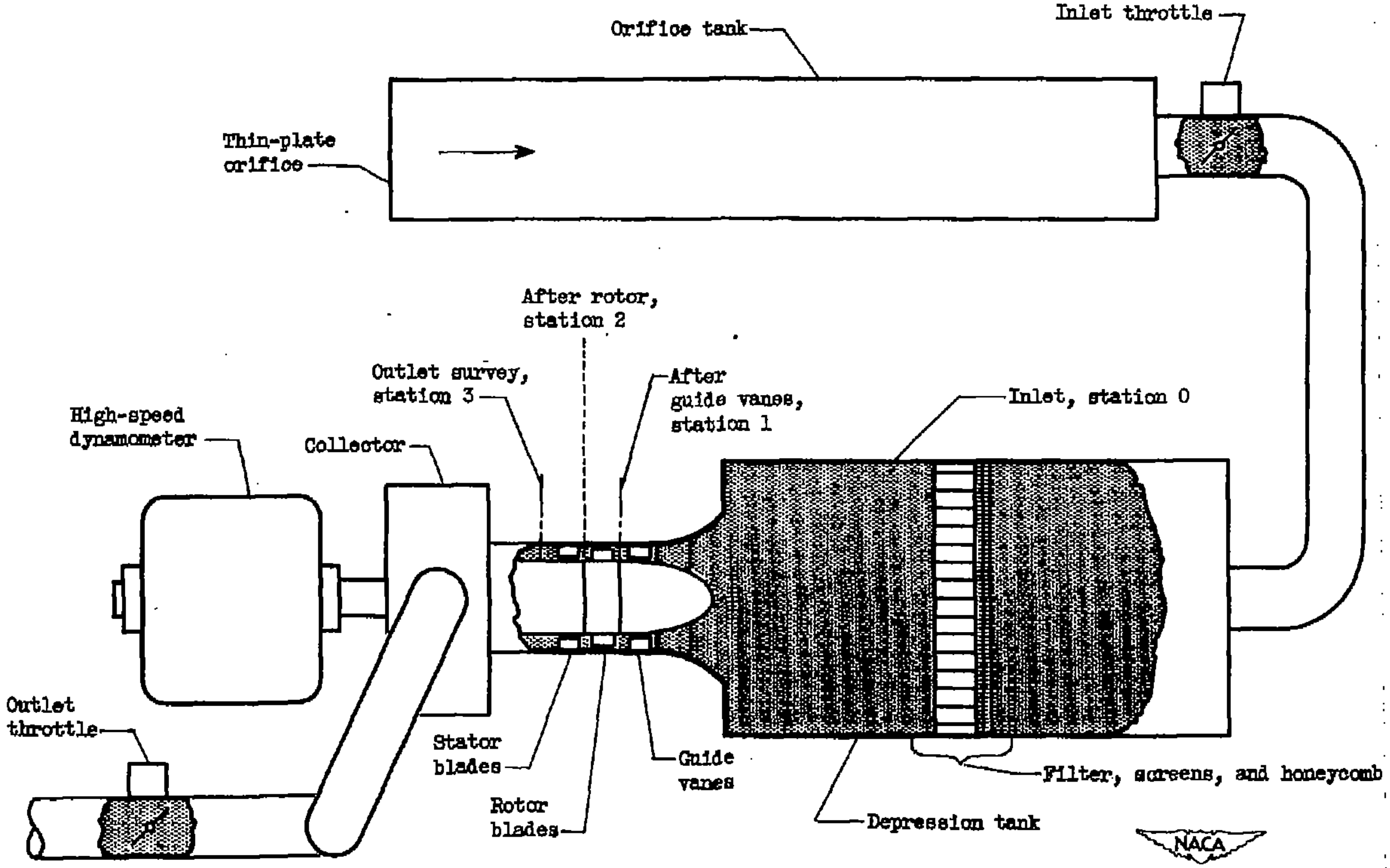

Figure 1. - Schematic diagram of compressor installation.

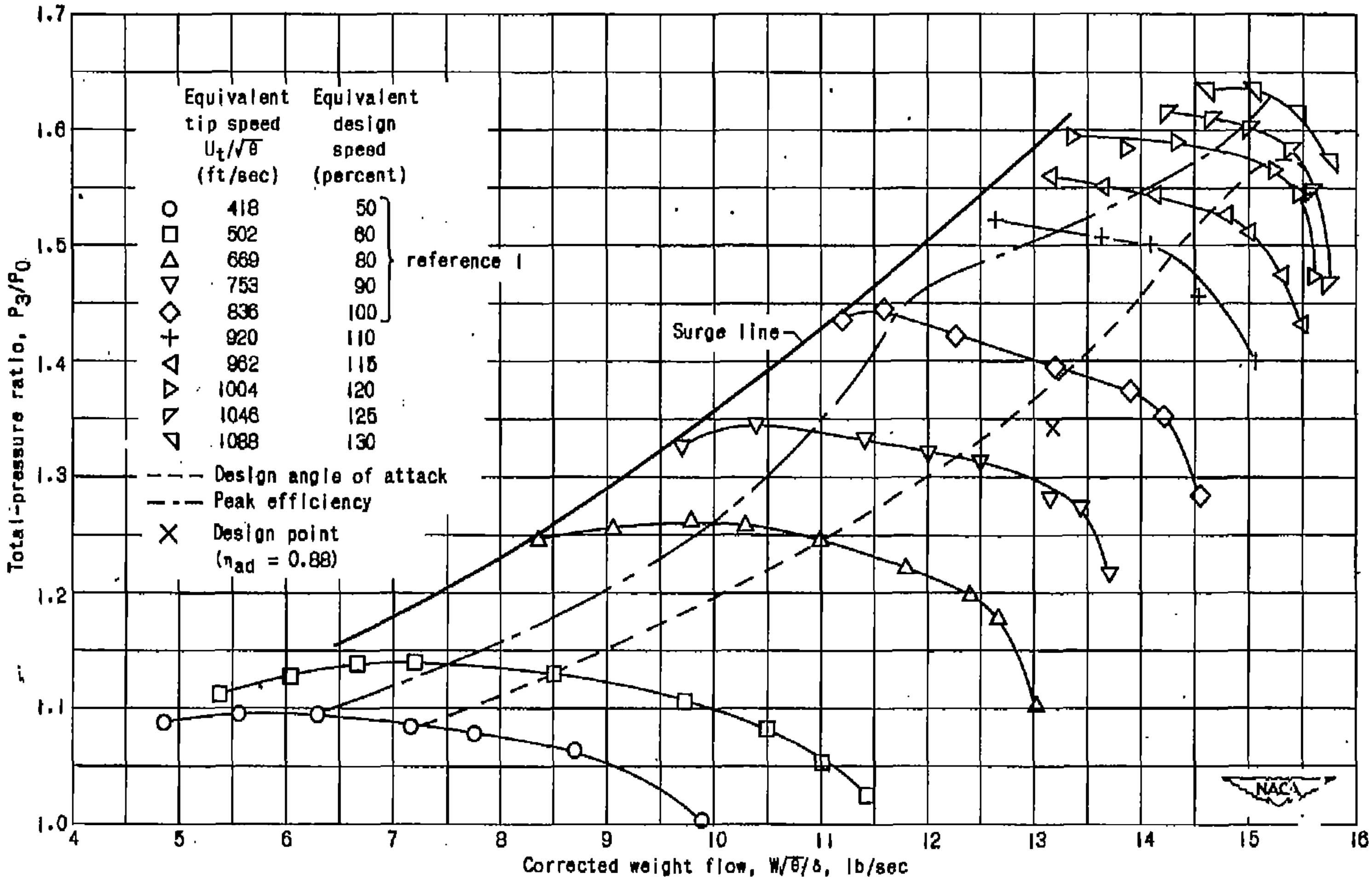

Figure 2. - Variation of total-pressure ratio with corrected weight flow for equivalent tip speeds from 418 to 1088 feet per second.

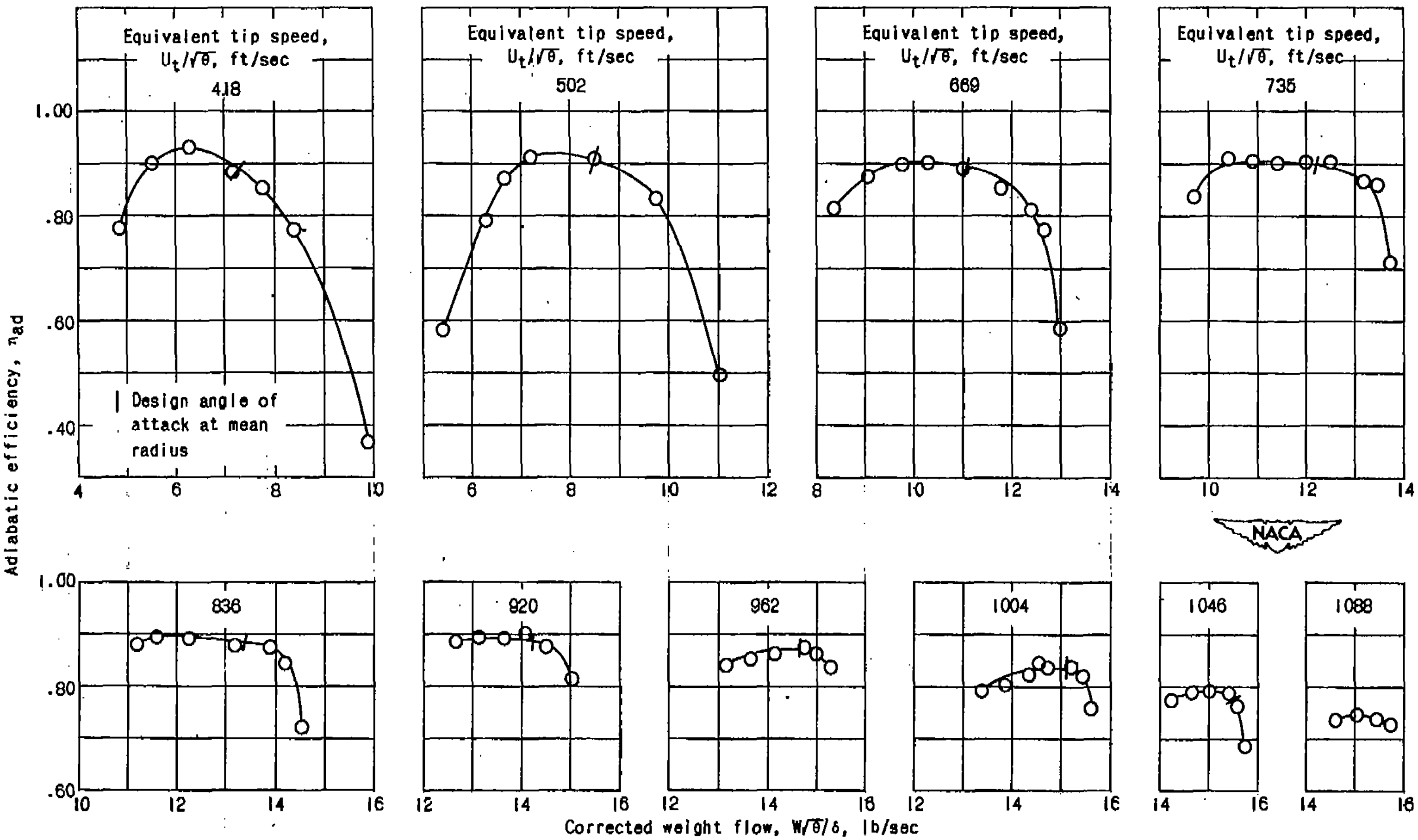

Figure 3. - Variation of adiabatic efficiency with corrected weight flow for equivalent tip speeds from 418 to 1088 feet per second.

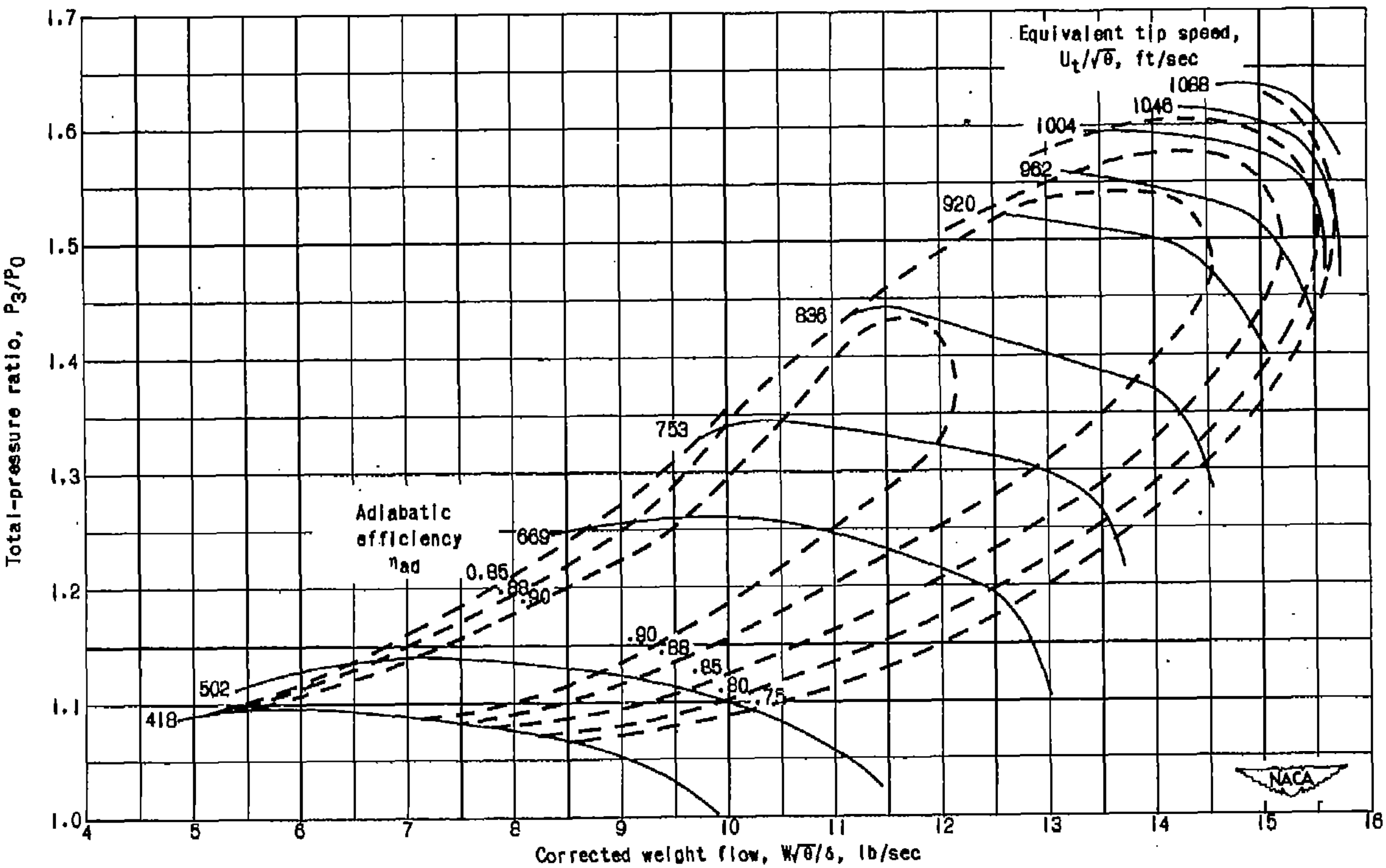

Figure 4. - Over-all performance characteristics for equivalent tip speeds from 418 to 1088 feet per second.

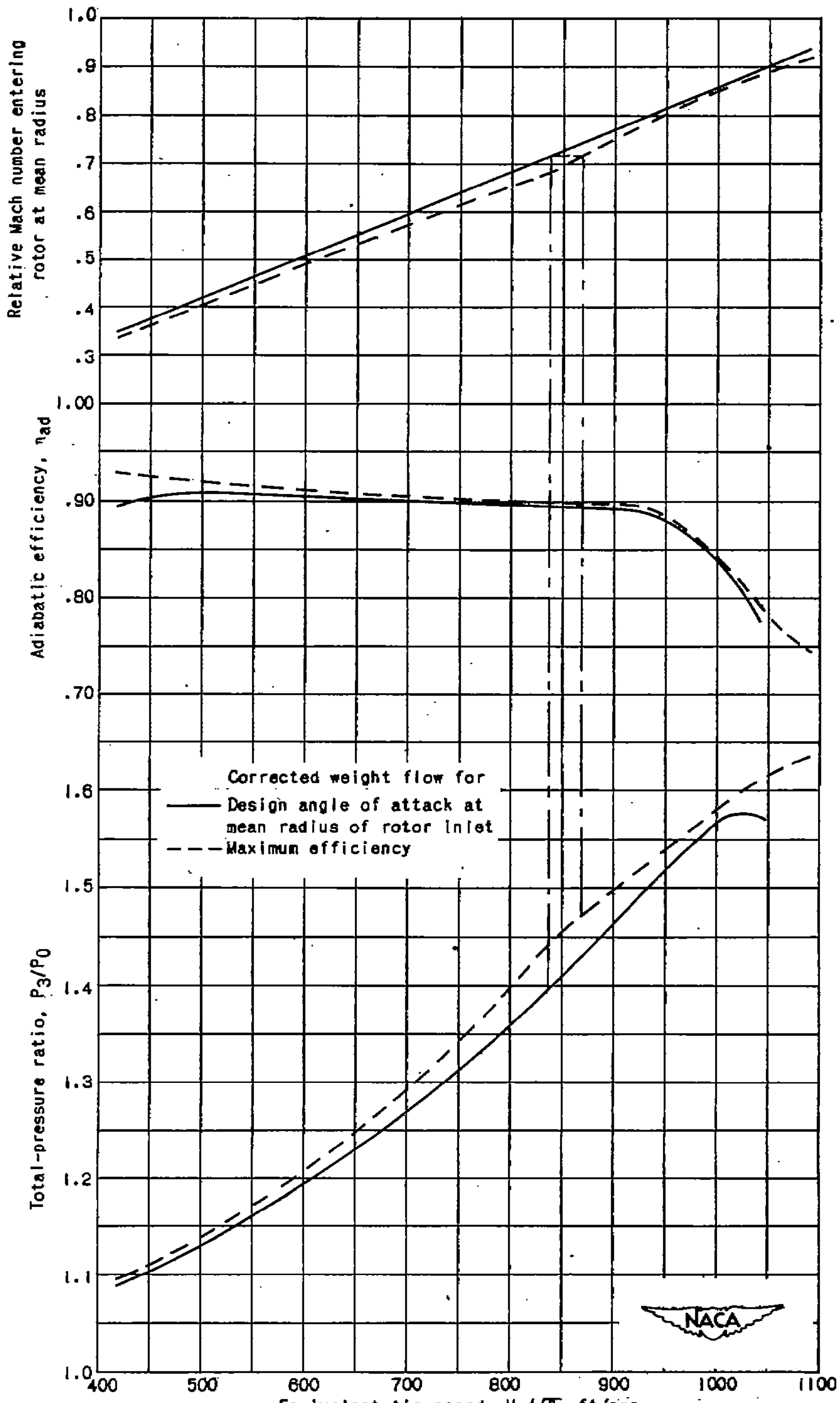

Figure 5. - Variation of total-pressure ratio, adiabatic efficiency, and relative Mach number with equivalent tip speed.

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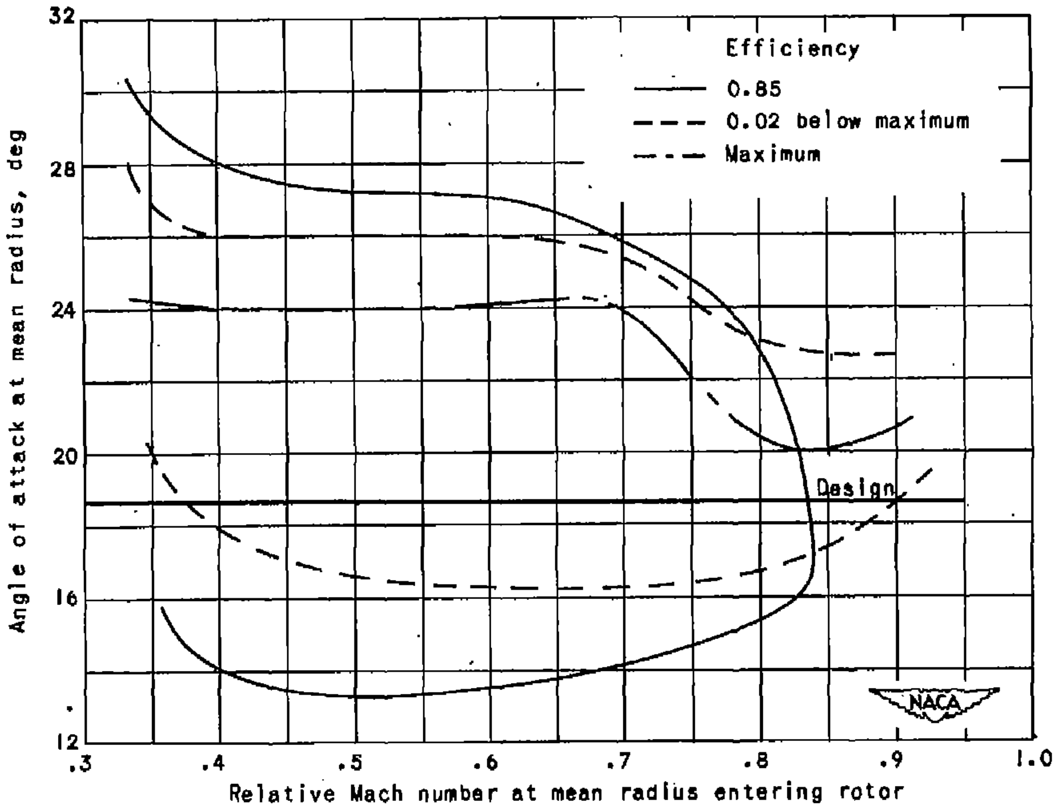

Figure 6. - Variation of rotor angle of attack at mean radius with relative rotor-inlet Mach number for three efficiency conditions.