Improved Compressors Through Cascade Research

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The range and load-carrying capacity of high speed, transonic and supersonic aircraft are related directly to the fuel economy and thrust of gas turbine engines. For this reason a considerable effort is being devoted to the development of efficient, high-flow, high-pressure ratio, axial-flow compressors. During previous inspections of this laboratory, we have reported the status of research on supersonic compressors. These compressors have great potentialities, and their development is being continued at this laboratory and at the NACA Lewis Laboratory. At present, however, the efficiency of supersonic compressors is not high enough to warrant general application to aircraft gas turbines. Therefore, an equal effort has been devoted to the study of the more efficient subsonic compressor. This program has produced information which has been used in the design of efficient aircraft and industrial axial-flow compressors. Today we would like to discuss recent improvements in axial-flow-compressor blading which have resulted from this program.

From the standpoint of developing high performance power plants, the gas turbine enjoys an advantage over the piston engine in that the compression, combustion, and expansion phases of the thermal cycle are steady-flow processes that occur in component parts which can be separated for individual study. The compressor and turbine components can be further separated into single-stage units and single blade rows for more thorough instrumentation and study over a wider range than is feasible with complete engines. The rotor from such a stage is shown here, and here is a mockup which represents a portion of a typical compressor blade row, either rotor or stator. As a further simplification compressor blade sections can be studied two-dimensionally if the annular blade rows are unwrapped into linear cascades. The blade rows can then be mounted stationary in a cascade tunnel and air blown past them. Two-dimensional cascade testing permits rapid detailed study of blade performance under controlled conditions. Surface pressures and velocities, flow directions, and losses in the wake can be measured with comparative ease. Information can also be obtained over a wide range of Mach numbers.

The maximum Mach number at which the blades can operate efficiently strongly affects the weight flow that can be accepted and the pressure ratio produced by a compressor. As for aircraft wings, optimum high-speed performance occurs if there are no localized high velocity regions on the blade surface. Because of the pressure rise, however, shapes which are best for wing airfoils have undesirable velocity distributions when used as compressor blades. This is illustrated in the first chart which shows the local Mach number distribution on the blade surface plotted along the chord line. A typical wing section commonly used as a compressor blade, the 65-(12) 10, has the local Mach number distribution shown by the black curve. Because of localized high velocities over the forward part of the convex surface, supersonic velocities and shock losses occur at relatively low entering Mach numbers. A compressor blade suited for higher operating Mach number would have constant velocity over the forward part of the convex surface with no localized high velocity regions similar to the red curve. An improved compressor blade section, the 65-(12A) 10, intended to approach this ideal is compared with the conventional section in the upper figure on this chart. The conventional blade shape is shown in gray while the improved blade is outlined in red. Note the rapid deceleration or diffusion of the flow required in the region of the trailing edge of the improved blade corresponding to the sharp curvature in the blade shape.

Cascade Comparison

In order to determine how closely the improved section actually approaches the ideal, and whether the rapid diffusion can be accomplished efficiently, compressor blades of this type have been mounted in the new Langley 7-inch high-speed cascade tunnel shown in the background. The test blades are mounted in the center of the large circular plates. These plates can be rotated to simulate the entering flow over the compressor operating range. As the tunnel is operated, the flow through the blade passages will be shown visually using the schlieren technique. The schlieren image will be projected on this screen. The flow will be from right to left on the screen. A Mach meter will also be projected on the screen to show the entering Mach number. The tunnel will be run at a maximum Mach number of approximately 0.83. As the tunnel is brought to speed notice the small region of separated flow off the trailing edge which appears as a gray cloudy region on the schlieren image. As the entering Mach number reaches 0.77 the first indications of supersonic velocity will appear as waves on the convex surface at about mid-chord.

The speed will be held at 0.77 for a few seconds. Then as the speed is increased to 0.83 the waves become weak shock waves. Note, however, that the amount of separation increases very little after the shocks appear indicating that the performance is still efficient. We will now operate the tunnel. The first sound you will hear is the supercharger which is used for tunnel wall boundary layer removal ahead of the cascade. The main tunnel drive flow is now being started.

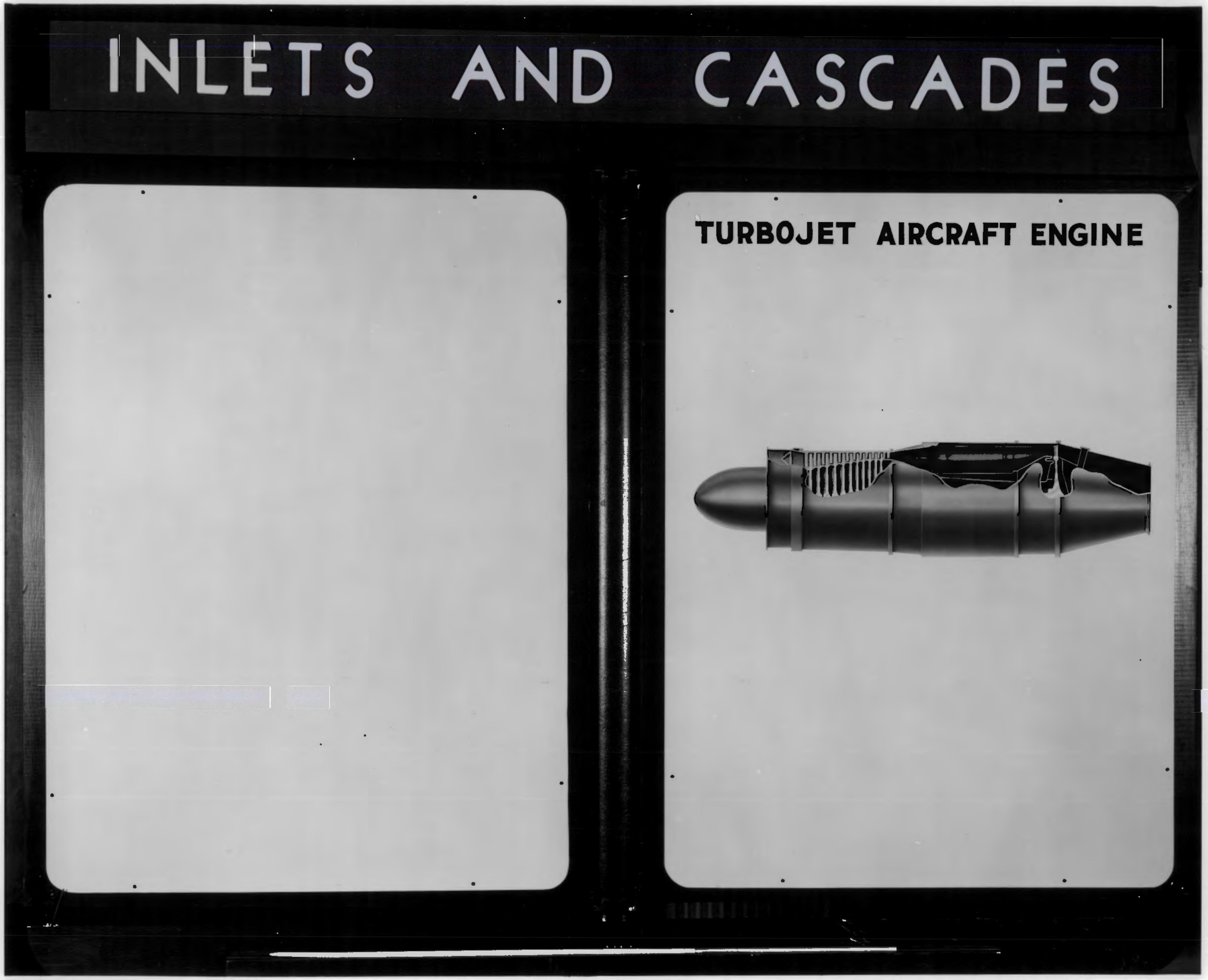
You've seen visual evidence of efficient performance for the improved blades. This chart gives a quantitative comparison of performance between the conventional blade and the improved blade. Shown are the drags of the two blades through the Mach number range with the improved blade shown by the solid line and the conventional blade shown by the dashed line. It can be noted that at Mach numbers from 0.6 to 0.8 the drag of the improved section is significantly less than that of conventional blades.

Rotor Comparison

To verify the results obtained from theory and from cascade tests, an axial-flow compressor rotor has been designed and tested at Langley using the improved blade sections. This is the rotor mounted on the table. The performance of this rotor through the Mach number range is compared to that of a similar rotor having conventional blades on the next chart. The Mach number given here is measured relative to the blades at the mean blade diameter. The performance of the improved rotor is again shown by solid lines and the conventional rotor by dashed lines. It can be seen that the improved rotor retains its efficiency to higher operating Mach numbers. From the stage pressure rise curves, calculated from rotor tests in each case, it can be seen that the improved rotor gives appreciably higher pressure rise primarily because of the increase in permissible operating Mach number.

From these results, it is apparent that increases in operating Mach number indicated by theory and investigated in cascade are realized and even exceeded in actual rotor tests. The next step in the program will be tests of complete stages and systematic cascade tests to provide design data for the entire operating range of axial-flow compressors.

This concludes the presentation at this facility.



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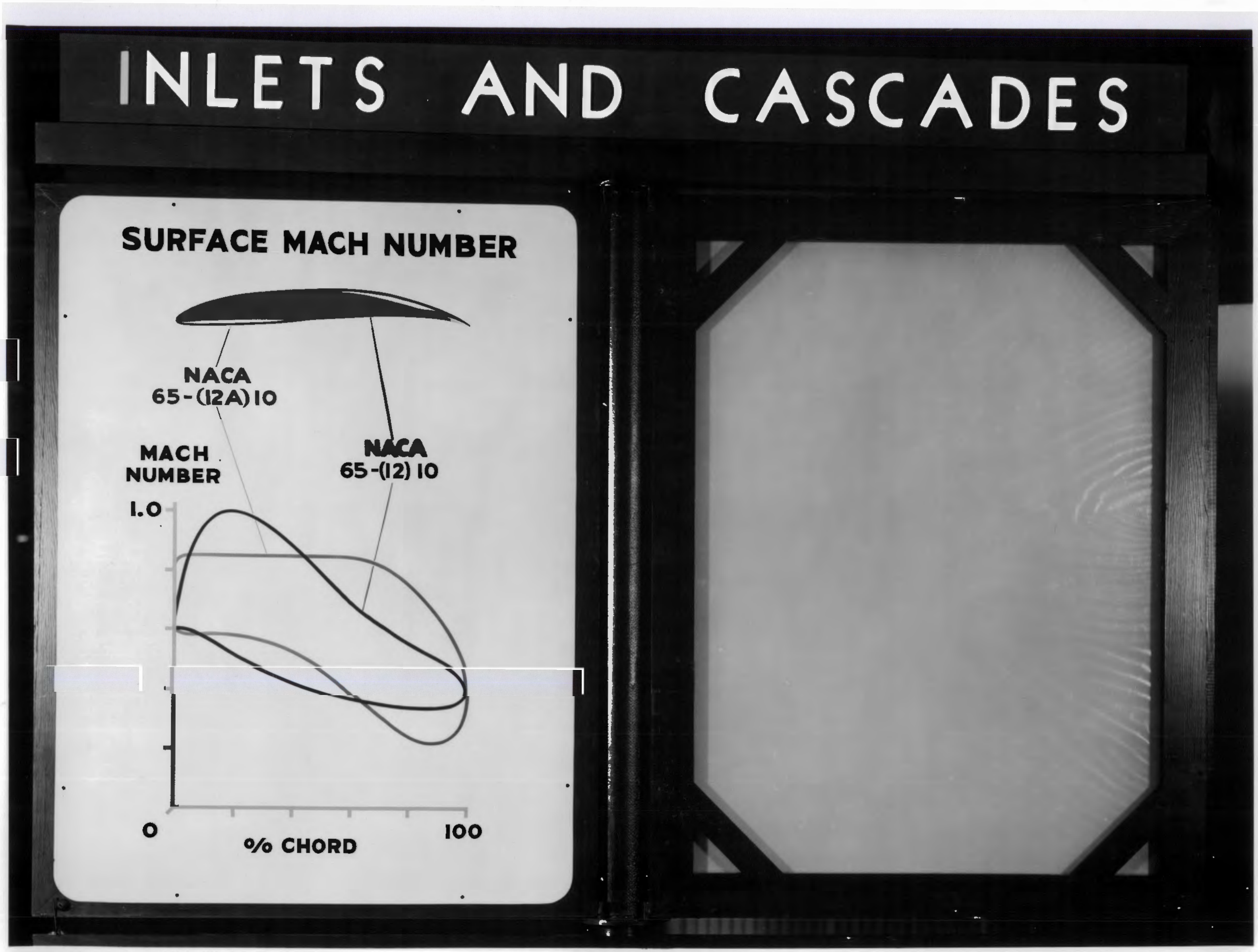

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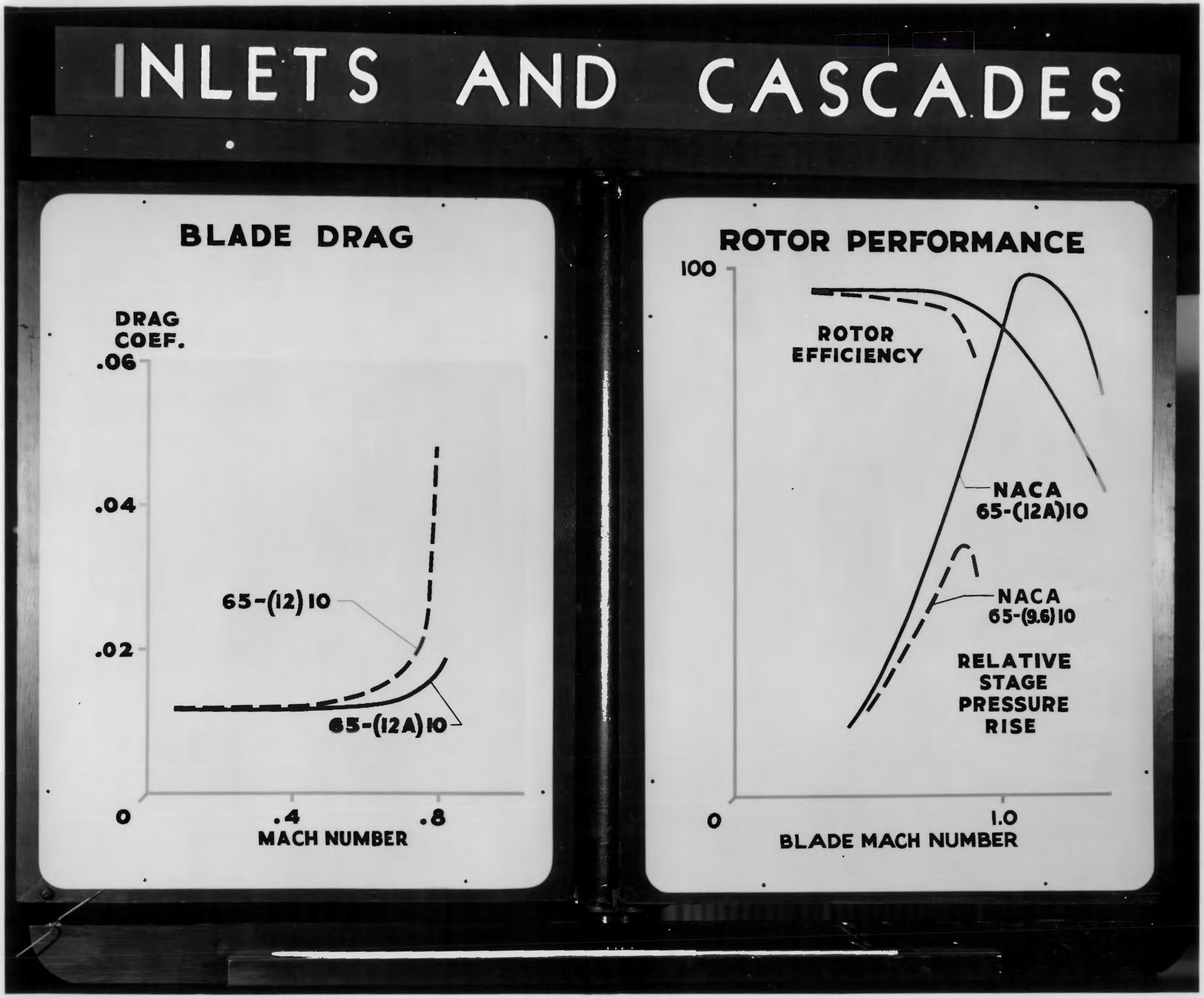

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