1949 BI-ANNUAL INSPECTION

DESCRIPTION OF THE LANGLEY 4- by 4-FOOT SUPERSONIC TUNNEL

References: (1) Tunnel pictorial drawing

Introduction

Just one year ago, the Langley Laboratory of the NACA placed in operation this 4- by 4-foot supersonic tunnel. During the first year, our major effort has been concerned with tunnel calibration and pressure distribution studies of supersonic aircraft configurations. This pictorial drawing shows the general arrangement of the Langley 4-foot supersonic tunnel. You are now in the test chamber of this facility - here. The major construction work going on outside is associated with the increase of power from the present 6,000 horsepower to 45,000 horsepower - as provided by this external drive system.

General Description

The 4-foot supersonic tunnel is a closed throat, single return tunnel driven by an axial-flow compressor. The design Mach number range is from 1.2 to 2.2. With the present power, the operating stagnation pressure is 1/4 atmosphere. When the tunnel power is increased, the test Reynolds number will correspond to full scale for many flight conditions. Let us follow the tunnel circuit around, starting here with the drive unit.

Drive System

This is a seven-stage axial-flow compressor designed for a compression ratio of 2. The flow capacity is 870,000 cubic feet per minute at 1300 RPM. There are over 1100 blades in the compressor with a tip diameter of 11 feet. The rotor blades are fixed in pitch, stator blades - adjustable. The compressor is temporarily driven by a 6000 horsepower water-cooled motor housed within the tunnel. The final drive will consist of external electric motors totaling 45,000 horsepower continuous - 60,000 horsepower for 1/2 hour.

Cooling Coils and Vanes

Downstream of the compressor is the pressure tight expansion joint - the transition section - and the cooling coils, located diagonally in the corner to minimize the power loss. These coils reduce the tunnel air temperature from about 250°F to 110°F.

Drying and Valving

This is the settling chamber. You will notice that this section contains a large seal-off door with a companion valve downstream to isolate the test section. This valve system permits the test section to be vented to atmosphere for access to the model without losing the dry air in the remainder of the tunnel. We have found, from tests in this tunnel at M = 1.6, that the dewpoint must be held to $-35^{\circ}F$ or below to prevent significant condensation effects.

Test Section

And now we reach the test section. As you know, at supersonic speeds the nozzle contour must physically be varied to change the speed. In the 4-foot tunnel, the side walls are fixed and parallel. The top and bottom walls are flexible. The supersonic nozzle and test section are formed by deflecting these walls against interchangeable templates which have been designed to produce uniform flow in the test section. This wall-template arrangement minimizes local wall irregularities and assures duplication of a given nozzle contour. The flexible wall extends from the subsonic region in the entrance cone to the downstream end of the test section - a distance of 25 feet. To match the nozzle variations, there is a variable area entrance cone, an adjustable second-minimum section, and a hinged diffuser section. Thus, the whole test section region is variable from a station in the upstream end of the entrance cone to the downstream end of the adjustable diffuser - a distance of 60 feet.

Model Support System

The model support system shown here provides for sting support from the rear. Forces will be measured by means of a 6-component internal balance. Provisions are also made for bringing out, through the sting, a total of 350 pressure tubes to outside manometers. Test section windows are provided for flow visualization by various optical methods. This completes the description of the tunnel.

1949 BI-ANNUAL INSPECTION

LANGLEY 4- by 4-FOOT SUPERSONIC TUNNEL RESULTS

References:
(1) Large model (1/15 scale)
(2) Small model (1/90 scale)
(3) Chart 1 - Fuselage pressures
(4) Chart 2 - Wing section pressures
(5) Chart 3 - $C_{n\alpha}$ plot
(6) Chart 4 - Span load

At this point I would like to present some representative experimental data obtained in the 4-foot tunnel during the first year of operation. The 4-foot tunnel, together with new companion facilities at the Ames and Lewis Laboratories of the NACA, now provide means for testing large scale models at supersonic speeds with a completeness of detail never before attainable. This is the size of model used in the 4-foot tunnel. Contrast this with the size of model for the Langley 9-inch supersonic tunnel - a facility which many of you visited two years ago at our last Langley Inspection.

This first chart (chart no. 1) shows the pressure distribution over the upper surface of a body of revolution for angles of attack of 0 and 10 degrees at M = 1.59. The symbols are test points; the lines are theoretical results for corresponding angles. The breaks in the curves result from the fact that the body was composed of cylinders, cones, and regions of arbitrary fairing. Excellent agreement is noted between the theoretical and experimental results.

This chart (chart no. 2) shows a typical pressure distribution over a section of a 40° swept wing in the presence of a body - similar to this model (1/15 scale). The data are shown for a symmetrical 8-percent thick circular-arc section at the 44 percent semispan station. The solid lines represent the pressures over the upper surface, and the dashed lines - the lower surface. This is the distribution for 0, - 6, - and 13 degrees angle of attack. You will note a difference between the upper and lower surface experimental pressures at 0 degrees. Since this section is symmetrical, the upper and lower surface pressures should correspond. The difference is a result of wing-fuselage interference effects. A theoretical pressure distribution determined from linear theory for 0 degrees is shown by this colored line for reference. The available theory is not able to account for the presence of the fuselage nor the fact that the shock is detached for Mach number of these tests. This phenomenon of shock detachment will be demonstrated later.

This chart (chart no. 3) shows the variation of pitching moment coefficient, drag coefficient, and angle of attack with normal force coefficient for the 44 percent span station. These data were determined from integration of the wing pressure distributions shown on the previous chart. The symbols and solid lines represent the experimental results. The theoretical results, indicated by the dashed lines, show fair agreement with experiment, but here again the true correlation is masked by fuselage interference and detachment of the shock.

The spanwise distribution normal force coefficient shown on this chart (chart no. 4) was determined from pressures measured at four stations along the span for angles of attack from minus 2 to plus 13 degrees. This is the 44 percent span station shown on the previous chart. A theoretical curve for 13 degrees is also shown. Note that the experimental spanwise distribution is similar to that obtained at subsonic speeds.

In general, the agreement between the experimental and theoretical span load distribution cannot be considered adequate. This discrepancy arises from the fundamental inability of the theory to predict the section characteristics in the presence of a fuselage and when the shock is detached. This holds true even for the most elementary of shapes. It appears, therefore, that one of the main fields of research of the 4-foot tunnel will consist of experimental investigation under conditions where theory is inadequate for predicting the flow phenomena. Such investigations supply the necessary research information for the design of supersonic aircraft and missiles.

1949 BI-ANNUAL INSPECTION

LANGLEY 4- by 4-FOOT SUPERSONIC TUNNEL DEMONSTRATION

References: (1) Variable incidence airfoil model
(2) Schlieren image
(3) Manometer with flow diagram

We have set up, in the tunnel, a demonstration wing - similar to this model - to illustrate the phenomenon of shock detachment. The wing consists of an 8 degree wedge-shaped airfoil with angle of attack controllable from outside the tunnel. It is located in the test section as shown here. The flow will be visualized by means of a schlieren system. The image will be shown on the overhead screen. In addition, we have connected to this manometer a row of orifices along the supersonic nozzle - as shown. The relative speed of the tunnel can be judged by the height of the tubes in relation to the Mach number scale shown. For this demonstration, the nozzle is set for a Mach number of 1.4.

The tunnel is now being brought up to speed for a demonstration of the detached shock. Because of the increasing noise level in the test chamber as the compressor speed is increased, the remainder of the program will be presented over the loud speaker system from a remote station.

TUNNEL DEMONSTRATION

Control From Remote Station

Air On

Schlieren Light On

Test Chamber Lights off.

At supersonic speeds, the character of the flow depends on the type of shock at the leaking edge. If the shock is ATtached, the flow is generally supersonic behind the shock. For the case of a DEtached shock, however, there are always regions of subsonic and supersonic flow. It is this mixed subsonic and supersonic flow that presents the main difficulties in theoretical investigations. Since all supersonic aircraft must operate under these conditions during some phase of flight, a detailed understanding of the detached shock phenomenon is of immediate importance.

The entire flow at this speed is subsonic. The speed is a maximum at the throat and decreases in the expanding area section of the nozzle.

The maximum Mach number is about ______. Some of the variation in tube height is due to a difference in lag in the individual orifices.

Schlieren on screen

The schlieren image of the model will be projected on the overhead screen. The schlieren system is merely an optical method for visualization of the flow. Density gradients are shown as light and dark regions on the screen. Increasing density gradients occur as bright regions.

Watch the manometer. The Mach number in the throat is approaching one.

The flow at the start of the nozzle is now supersonic. The normal shock is moving downstream as indicated by the sharply falling tube heights.

The shock is now approaching the test section.

Observe the schlieren image.

Note the local shock formation on the model. The flow in the test section ahead of the model is still subsonic.

Watch for the normal shock moving through the test section.

Normal shock moves through

The flow in the test section is now completely supersonic. The uniformity of flow is indicated by the uniform intensity of the schlieren image ahead of the model. The manometer tubes indicate a constant Mach number of 1.4 through the test section.

Turn manometer lights off

The airfoil is at zero angle of attack. Observe the symmetrical shock formation attached to the leading edge of the airfoil. The flow is completely supersonic behind the shock. The pressure of the airfoil is not transmitted ahead of the leading edge.

We will now increase the angle of attack.

The angle of attack is 4 degrees. Note the rotation of the shock pattern with the wing.

The angle will now be increased to 7 degrees.

The wave on the lower surface is beginning to curve at the leading edge. This is indicative of the start of shock detachment.

The angle will now be increased to 11 degrees. The shock is moving progressively ahead of the airfoil.

The shock is clearly detached. A region of subsonic flow exists between the shock and the leading edge of the airfoil. It is this subsonic region which enabled the presence of the airfoil to be felt ahead of the leading edge.

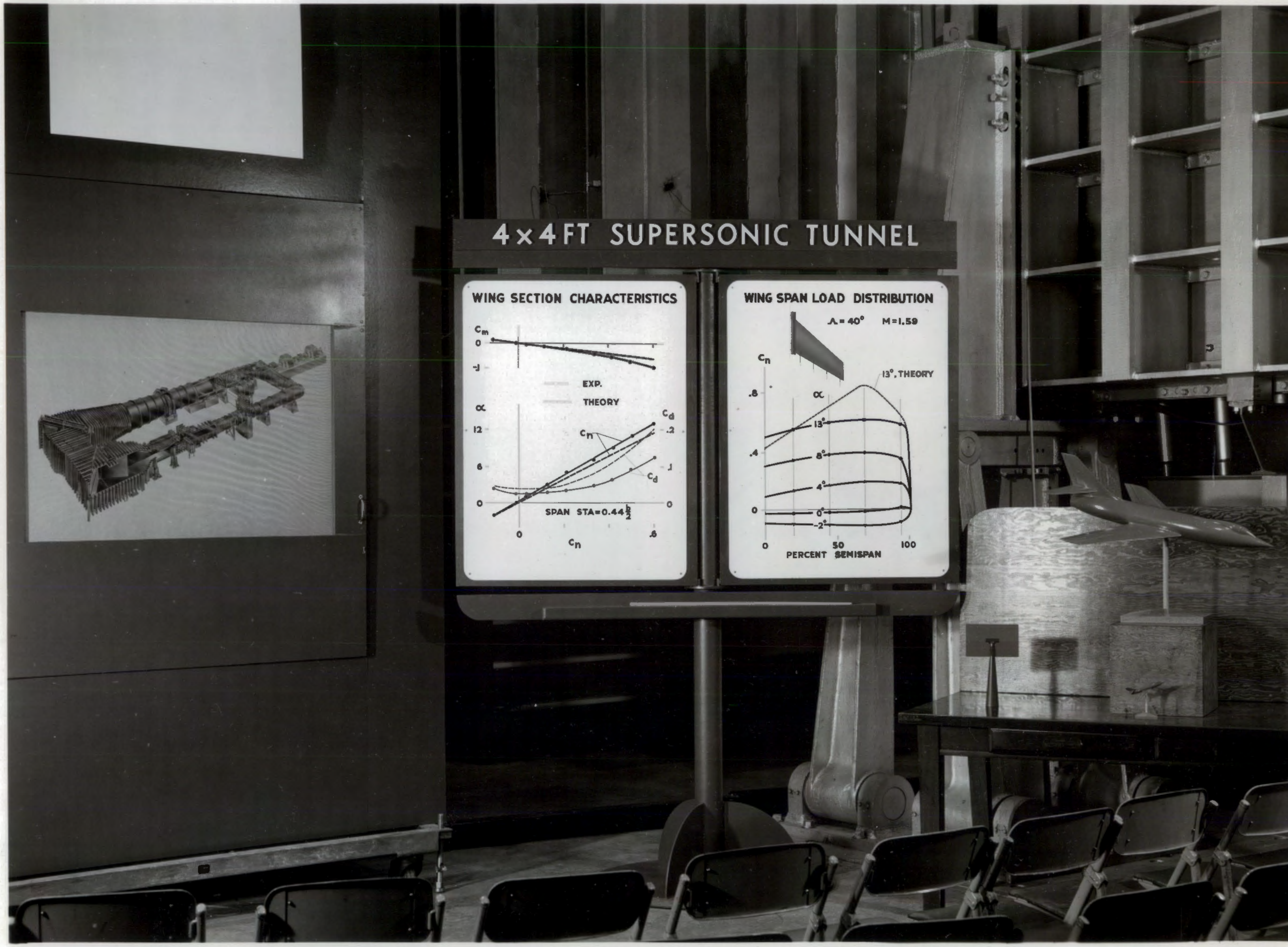
Due to a local separated region at the leading edge of the upper surface, there is a second shock now formed.

The angle of attack will now be reduced to zero. Note that ATtachment as well as DEtachment of the shock does not involve any sudden changes in flow phenomenon.

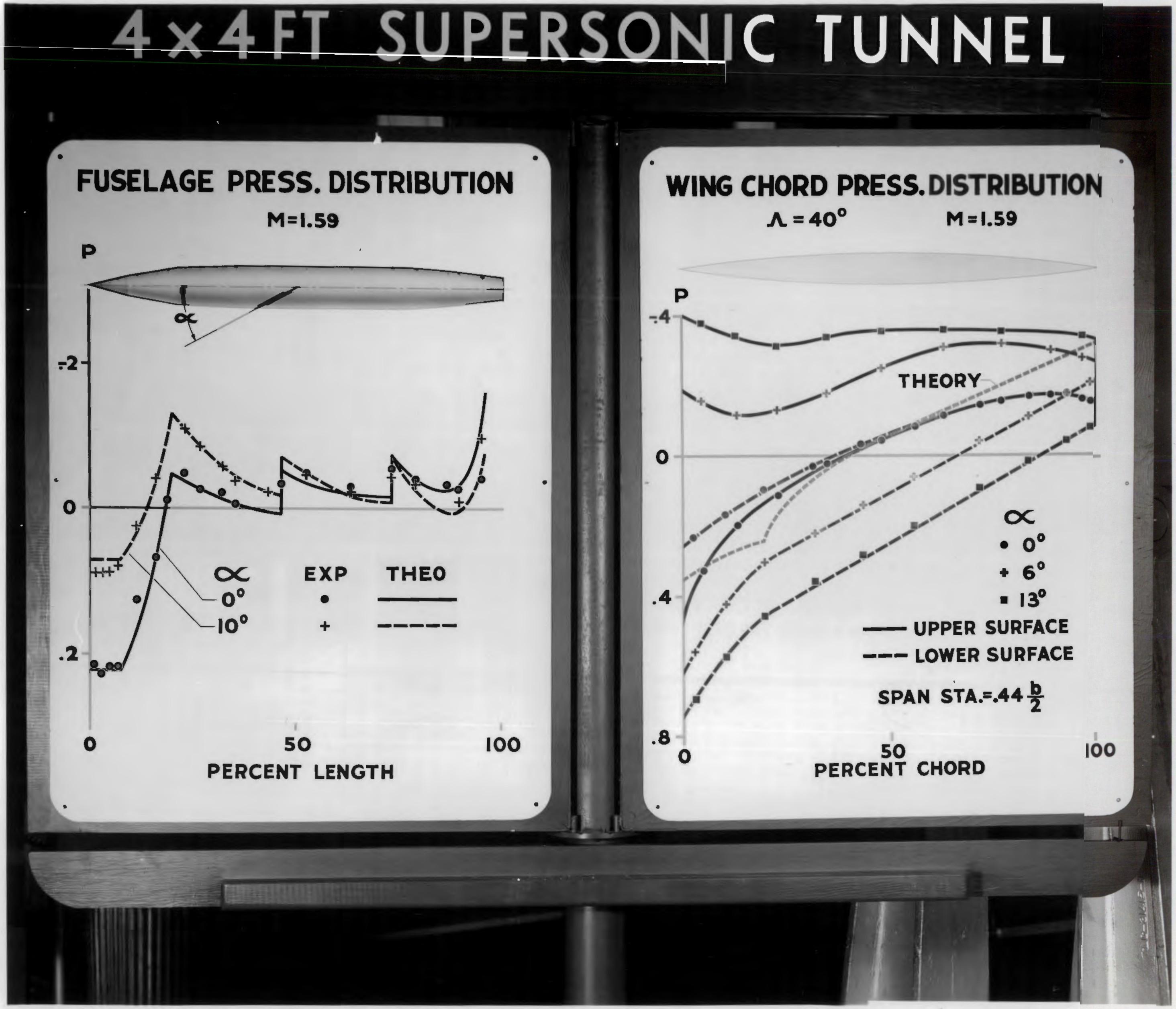
The tunnel will now be reduced in speed. Watch for the normal shock as the test section becomes subsonic.

Test Chamber lights on.

Gentlemen - that completes our demonstration at the 4-foot supersonic tunnel. Please follow you group leader out of the main door.


4×4FT SUPERSONIC TUNNEL
WING SECTION CHARACTERISTICS
EXP.
THEORY
SPAN STA=0.44
WING SPAN LOAD DISTRIBUTION
Λ = 40° M=1.59
13°, THEORY
PERCENT SEMISPAN

LAL 61139



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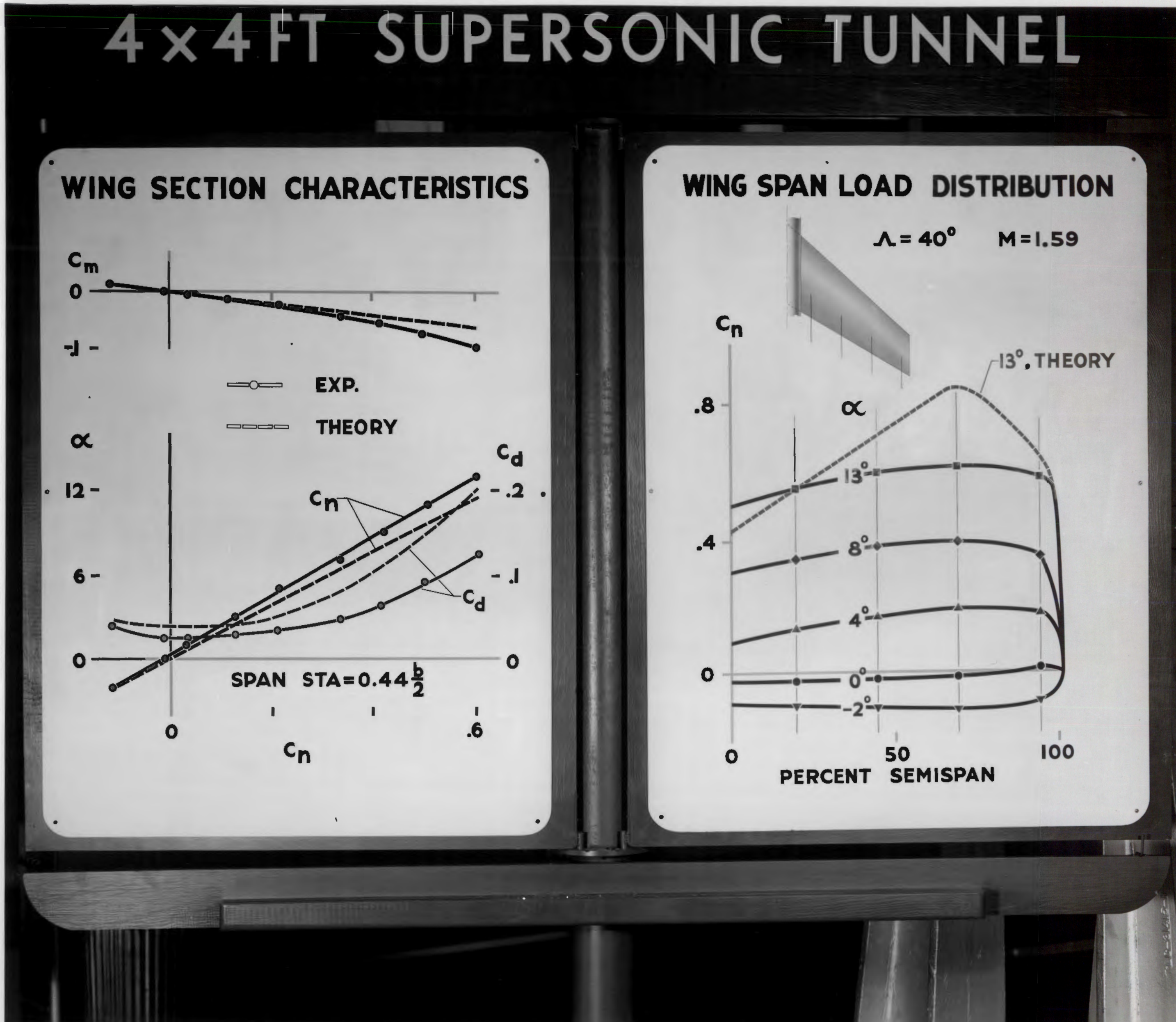

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