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RESEARCH MEMORANDUM

BOUNDARY-LAYER-TRANSITION MEASUREMENTS
IN FULL-SCALE FLIGHT

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and Gilbert Petty, Jr.

High-Speed Flight Station
Edwards, Calif.

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July 28, 1958

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RESEARCH MEMORANDUM

BOUNDARY-LAYER-TRANSITION MEASUREMENTS
IN FULL-SCALE FLIGHT*By Richard D. Banner, John G. McTigue,
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SUMMARY

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Chemical sublimation has been employed for boundary-layer-flow visualization on the wings of a supersonic fighter airplane in level flight at speeds near a Mach number of 2.0. The tests have shown that laminar flow can be obtained over extensive areas of the wing with practical wing-surface conditions.

In addition to the flow visualization tests, a method of continuously monitoring the conditions of the boundary layer has been applied to flight testing, using heated temperature resistance gages installed in a Fiberglas "glove" installation on one wing. Tests were conducted at speeds from a Mach number of 1.2 to a Mach number of 2.0, at altitudes from 35,000 feet to 56,000 feet.

Data obtained at all angles of attack, from near 0° to near 10° , have shown that the maximum transition Reynolds number on the upper surface of the wing varies from about 2.5×10^6 at a Mach number of 1.2 to about 4×10^6 at a Mach number of 2.0. On the lower surface, the maximum transition Reynolds number varies from about 2×10^6 at a Mach number of 1.2 to about 8×10^6 at a Mach number of 2.0.

INTRODUCTION

Because of the greatly increased need for knowledge of full-scale boundary-layer transition and the difficulty of simulating actual flight conditions, a program has been initiated to provide a better understanding of the boundary-layer flow as it exists in supersonic flight. This paper shows the results obtained in the early flight tests which determined the extent of laminar flow that could be obtained with practical wing-surface conditions.

*Title, Unclassified.
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SYMBOLS

R	Reynolds number per foot, V_{∞}/ν , per ft
V_{∞}	free-stream velocity, ft/sec
ν	kinematic viscosity
R_x	nondimensional Reynolds number based on x
x	distance from leading edge
Λ	sweep angle
α	angle of attack
h_p	altitude
M	Mach number
τ	thickness, in.

INSTRUMENTATION AND TECHNIQUES

A fighter airplane was instrumented, as shown in figure 1, for transition investigations on the wings. The basic wing has a modified biconvex airfoil with a thickness ratio of 3.4 percent, a sharp leading edge, and a slight amount of sweep (about 27°). A 1/10-inch-thick Fiberglas glove was installed on the right wing and was instrumented with one row of transition detectors on both the top and bottom surfaces. These detectors provided continuous monitoring of the laminar and turbulent boundary-layer-flow conditions (ref. 1).

Chemical sublimation was employed for boundary-layer-flow visualization on both wings, and cameras (fig. 1) were installed for recording the chemical indications. Many investigators have used the chemical sublimation technique in both wind tunnels and in flight (refs. 2, 3, and others). These tests have extended the use of this technique in flight to speeds near a Mach number of 2.0.

The transition-detector signals (see fig. 2) were multiplexed and recorded on an oscillograph. The sequencing was scheduled to conform to the locations of the detectors on the wing. This arrangement allowed





location of the laminar and turbulent flow areas, within about 5 percent of the chord, by inspection of the records. The reasons for using flow visualization are illustrated in figure 2. Turbulent wedges, originating upstream of the detectors, cause local areas of turbulent flow. As can be seen, the third detector indicates turbulent flow in an area that would otherwise be laminar.

RESULTS AND DISCUSSION

From the 35-millimeter flight film of the chemical indications, photographic enlargements have been made and a typical in-flight photograph of the lower surface of the Fiberglas covered wing is shown in figure 3. The white chemical remaining in the vicinity of the leading edge indicates the extent of laminar flow being experienced on the wing. The field of view of the camera includes the area of the wing from the leading edge rearward to just behind the aileron hinge in the outboard area and some of the inboard area of the wing. In all the tests no laminar flow had been observed in the inboard area, and for that reason this area is omitted in subsequent photographs of this presentation. The area shown is outboard of the 47-percent-exposed-span station.

In some cases the airplane returned from the flight with a chemical indication remaining on the wing. It will be of interest to look at one such indication before proceeding with the main part of the discussion. Figure 4 presents an enlargement of the leading-edge region of the wing. The section seen is about 1 square foot. Note the striations that can be seen in the chemical. Other investigators have also observed these striations in a laminar boundary layer, both in wind-tunnel tests (ref. 4) and in flight (ref. 5) at subsonic speeds. The striations have been attributed to the presence of vortices which are shed from the swept leading edges and contribute to the breakdown of the normally laminar flow. Although it could not be determined when this phenomenon occurred during the flight, it is believed to be worth mentioning since it appears to be a problem that must be considered in determining the extent of laminar flow that could be expected on swept wings.

Turning now to the flight photographs that were taken during the tests, figure 5 shows the effect of the leading-edge-flap "piano type" hinge on producing transition. As can be seen, the hinge tripped the laminar boundary layer producing turbulent wedges which merge rearward of the hinge to form completely turbulent flow over the remainder of the wing. The laminar area is approximately 15 percent of the test area. This condition of the wing is referred to as unfinished. In improving the wing-surface conditions the flap hinge was filled to eliminate any abrupt discontinuities. Also, all rivetheads and screwheads were ground flush with the wing skin and filler material was applied to fill any





pits or small depressions. The whole surface was then sanded. This condition of the wing is referred to as the finished wing. Following the tests with the finished wing, the wing was painted and polished.

The effect of these improvements can be seen by comparing figure 6 with figure 5. Although the Mach number for the test with the painted wing is slightly different, the variation in the altitudes resulted in the same free-stream Reynolds number and the same angles of attack.

In comparing the unfinished and finished wing lower surfaces, it can be seen that considerably more laminar flow was obtained on the finished wing. This is primarily due to smoothing over the leading-edge-flap hinge. Painting the wing surface reduced the average roughness from about 25 to 13 microinches, but the effect on transition was not appreciable on either the top or bottom surface. The extent of laminar flow on the painted wing is about 25 percent of the test area for the upper surface and about 35 percent of the test area for the lower surface.

Realizing that the standards that had been set for roughness were rather arbitrary and that they might differ from those set in the wind tunnel, it was felt, nevertheless, that the maximum in practical improvements to the wing surface had been reached. The extent of laminar flow that was observed on the finished and painted wing is considered to be representative of the maximum that might reasonably be expected for these flight conditions. This conclusion was arrived at because the extreme care that was taken in producing the Fiberglas surface finish had resulted in an average roughness of only 7 microinches.

A comparison of the finished and painted wing and the Fiberglas covered wing is shown in figure 7. For clarity, the leading edges are all shown to the left. Covering the wing with Fiberglas had slightly altered the wing profile, and the leading edge had been rounded to 1/10-inch radius, instead of the sharp leading edge of the basic wing. Also, waviness measurements at 1/2-inch increments indicated an average deviation of about 0.003 inch on the Fiberglas covered wing as compared with 0.006 inch on the basic wing. Exactly what effect these changes produced locally could not be determined; however, as can be seen, no large differences in the overall extent of laminar flow is evidenced. In order to determine the effect of Mach number and altitude on the extent of laminar flow, the transition-detector installation on the Fiberglas covered wing was utilized.

Tests were conducted at speeds from a Mach number of 1.2 to a Mach number of 2.0 at altitudes from 35,000 to 56,000 feet. The free-stream Reynolds number varied from 1.3 to 4.3×10^6 per foot. The maximum transition Reynolds numbers (based on free-stream conditions and the





distance to the point of transition) that were obtained on the Fiberglas test area are shown in figure 8.

Data obtained at all angles of attack, from near 0° to near 10° , have been used to construct the curves. As can be seen, the maximum transition Reynolds number on the top surface of the wing varied from about 2.5×10^6 at a Mach number of 1.2 to about 4×10^6 at a Mach number of 2.0. The trend on the lower surface is generally to more laminar flow, with the maximum transition Reynolds number varying from about 2×10^6 at a Mach number of 1.2 to about 8×10^6 at a Mach number of 2.0.

Although no attempt has yet been made to separate the effects of the variables that contribute to the results presented herein, the results are encouraging in that laminar flow has been obtained over extensive areas of a wing surface at supersonic speeds with practical wing-surface conditions.

Further flight testing should include investigations to determine what effects on the boundary layer are experienced when the leading edge is altered, when the angle of attack is varied, when shock-wave—boundary-layer interaction takes place, and when other factors enter the problem as important variables.

CONCLUSIONS

Chemical sublimation has been employed for boundary-layer-flow visualization on the wings of a supersonic fighter airplane in level flight at speeds near a Mach number of 2.0. The tests have shown that laminar flow can be obtained over extensive areas of the wing with practical wing-surface conditions.

In addition to the flow visualization tests, a method of continuously monitoring the conditions of the boundary layer has been applied to flight testing, using heated temperature resistance gages installed in a Fiberglas "glove" installation on one wing. Tests were conducted at speeds from a Mach number of 1.2 to a Mach number of 2.0, at altitudes from 35,000 feet to 56,000 feet.

Data obtained at all angles of attack, from near 0° to near 10° , have shown that the maximum transition Reynolds number on the upper surface of the wing varies from about 2.5×10^6 at a Mach number of 1.2 to about 4×10^6 at a Mach number of 2.0. On the lower surface, the maximum



transition Reynolds number varies from about 2×10^6 at a Mach number of 1.2 to about 8×10^6 at a Mach number of 2.0.

High-Speed Flight Station,
National Advisory Committee for Aeronautics,
Edwards, Calif., March 20, 1958.

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AIRPLANE-TRANSITION TEST AREAS

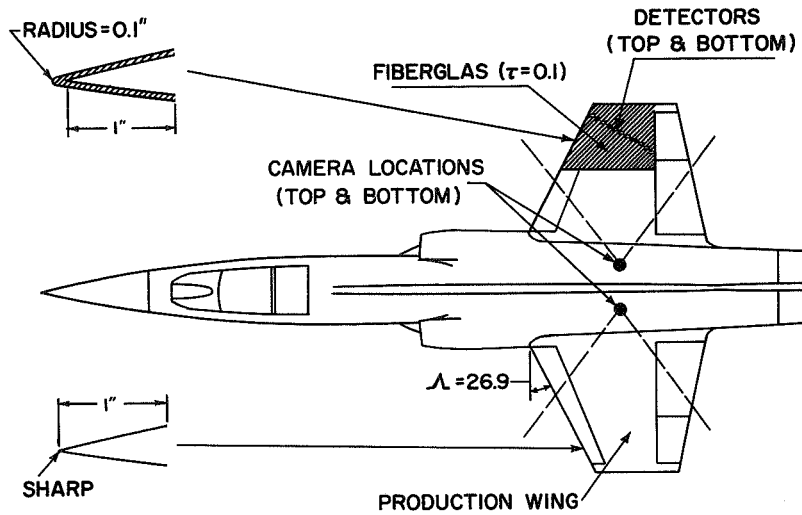


Figure 1

TRANSITION TEST METHODS CHEMICAL SUBLIMATION

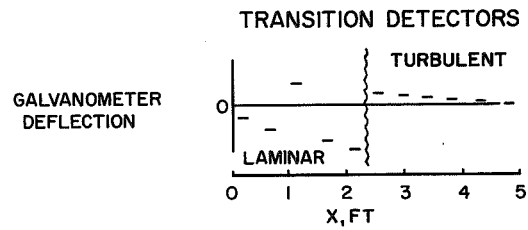
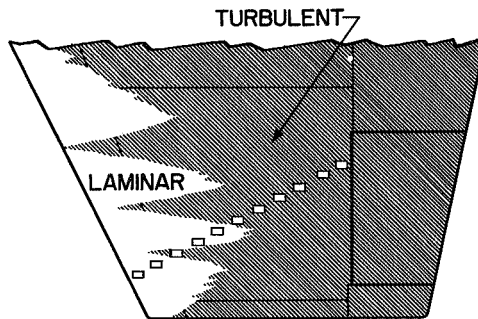
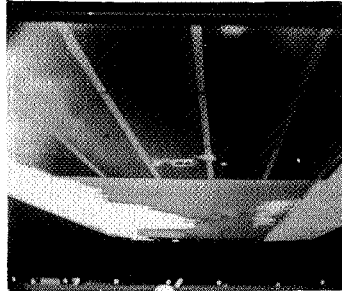


Figure 2

IN-FLIGHT PHOTOGRAPHS

TYPICAL 35-MM ENLARGEMENT



ARTIST'S CLARIFICATION



Figure 3

INDICATION OF THE PRESENCE OF VORTICES

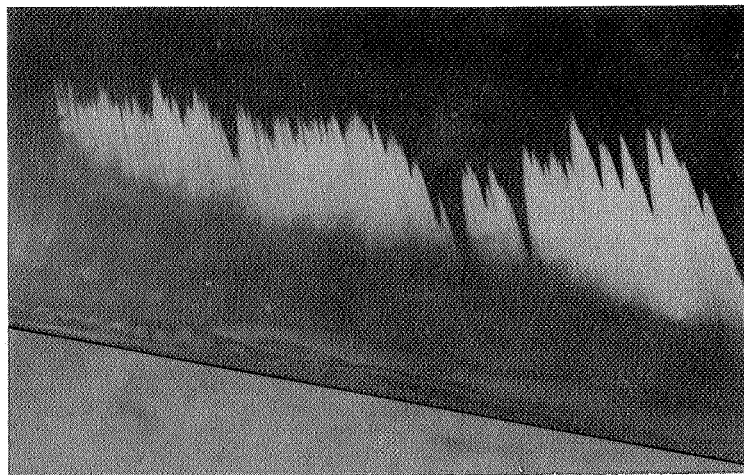


Figure 4

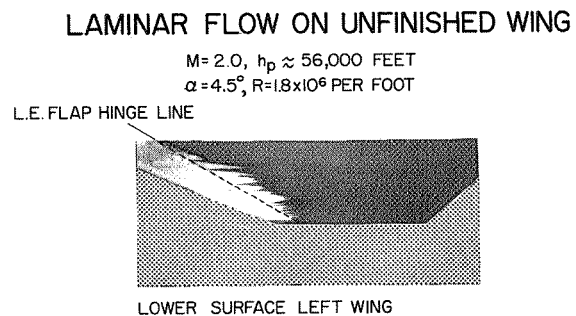


Figure 5

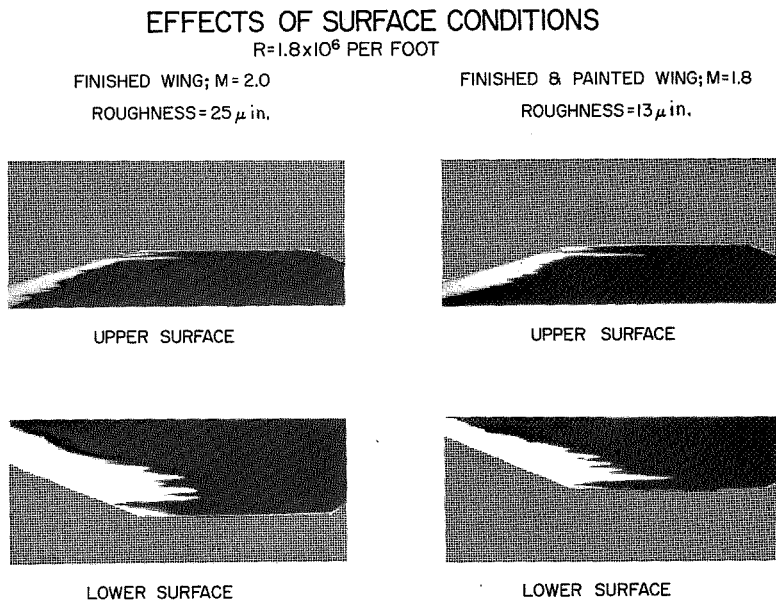


Figure 6

COMPARISON OF LEFT- & RIGHT-WING TRANSITION
 $M=1.8, R=1.8 \times 10^6$ PER FOOT

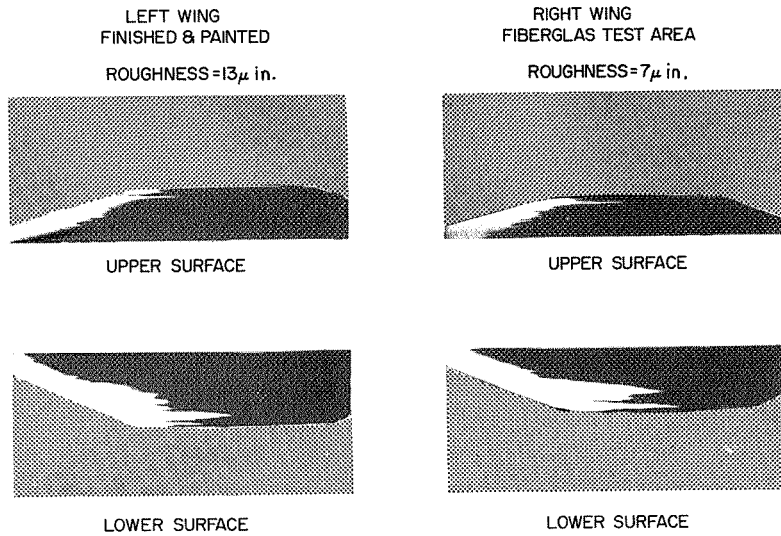


Figure 7

MAXIMUM TRANSITION REYNOLDS NUMBERS
 FIBERGLAS TEST AREA- RIGHT WING
 $10^\circ > \alpha > 0^\circ$

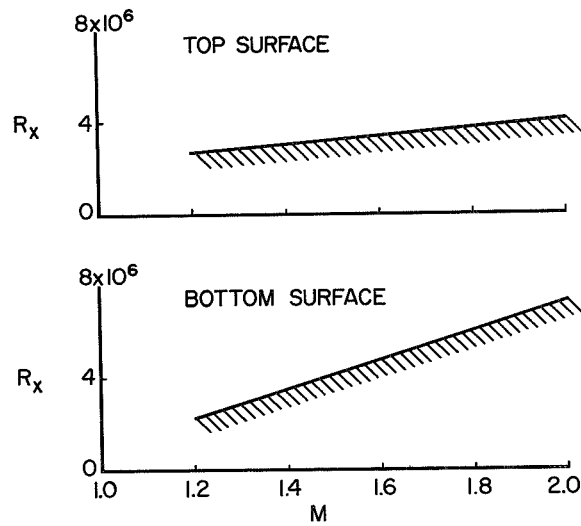


Figure 8