SUPERSONIC BOUNDARY-LAYER TRANSITION ON THE LaRC F-106 AND THE DFRF F-15 AIRCRAFT

Part I: Transition Measurements and Stability Analysis
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Part II: Aerodynamic Predictions
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PART I: TRANSITION MEASUREMENTS AND STABILITY ANALYSIS

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NOTATION

SYMBOLS

F	-	frequency (Hertz)
H,h	-	altitude (feet)
M	-	Mach number
C_p	-	pressure coefficient = $P-P_\infty/q_\infty$
C_q	-	suction coefficient = $(\rho V)_w/(\rho U)_\infty$
N	-	N-factor
x/c	-	chord fraction
c	-	chord (feet)
U,V	-	velocity (ft/sec)
q	-	dynamic pressure (lb_f/ft^2)
p	-	pressure (lb/ft^2)
Re	-	Reynolds number = Ul/ν
l	-	characteristic length (feet)
α	-	angle of attack (degrees)
Λ	-	sweep angle (degrees)
ρ	-	density ($slugs/ft^3$)
ν	-	kinematic viscosity (ft^2/sec)

SUBSCRIPTS

∞	-	free-stream condition
w	-	wall condition
t	-	transition
cf	-	crossflow

INTRODUCTION

Because of renewed interest in the supersonic flight regime and the recent success with laminar flow techniques at subsonic speeds, there is a developing interest in examining laminar flow control for increased fuel efficiency at high speeds. However, flight data in the area of supersonic boundary-layer transition phenomena is scarce. In late 1985, a limited window of opportunity of about 6 months existed where the Langley F-106 and the Dryden F-15 aircraft were available for flight tests. Two exploratory supersonic flight tests were conducted to expand upon the supersonic boundary-layer transition data base. The objectives of these cursory flight tests were to explore boundary-layer transition measurement techniques and to identify future experiments. In addition, for future design considerations, there was a need to obtain transition data to evaluate existing stability theory criteria at supersonic speeds and to use measured aerodynamic pressure data to evaluate full potential flow codes in the supersonic regime. This latter point will be discussed in Part II of this paper.

Primary

- Explore boundary layer transition at supersonic cruise
- Develop experience in supersonic transition measurement techniques
- Identify future experiments

Secondary

- Obtain transition data to evaluate existing stability theory criteria at supersonic speeds
- Use measured aerodynamic pressure data to evaluate full potential flow codes in supersonic regime

APPROACH

These exploratory supersonic flight tests were conducted utilizing surface cleanup gloves on the right wing and vertical tail of the Langley F-106 and the right wing of the Dryden F-15 test aircraft. Each glove was instrumented with surface pressure orifices and hot films to obtain measured pressure and transition data during flight. The measured transition data was correlated with compressible, linear, boundary-layer stability theory which is considered by many as the state-of-the-art transition prediction method. The measured pressure data was used to evaluate a non-linear, inviscid, full potential code which is described in more detail in Part II of this report.

- Perform exploratory supersonic flight tests using surface cleanup gloves
  - F-15
  - F-106
- Correlate results with computations
  - COSAL; compressible, linear boundary layer stability theory
  - NCOREL; non-linear, full potential, inviscid code

DRYDEN F-15 TEST AIRCRAFT

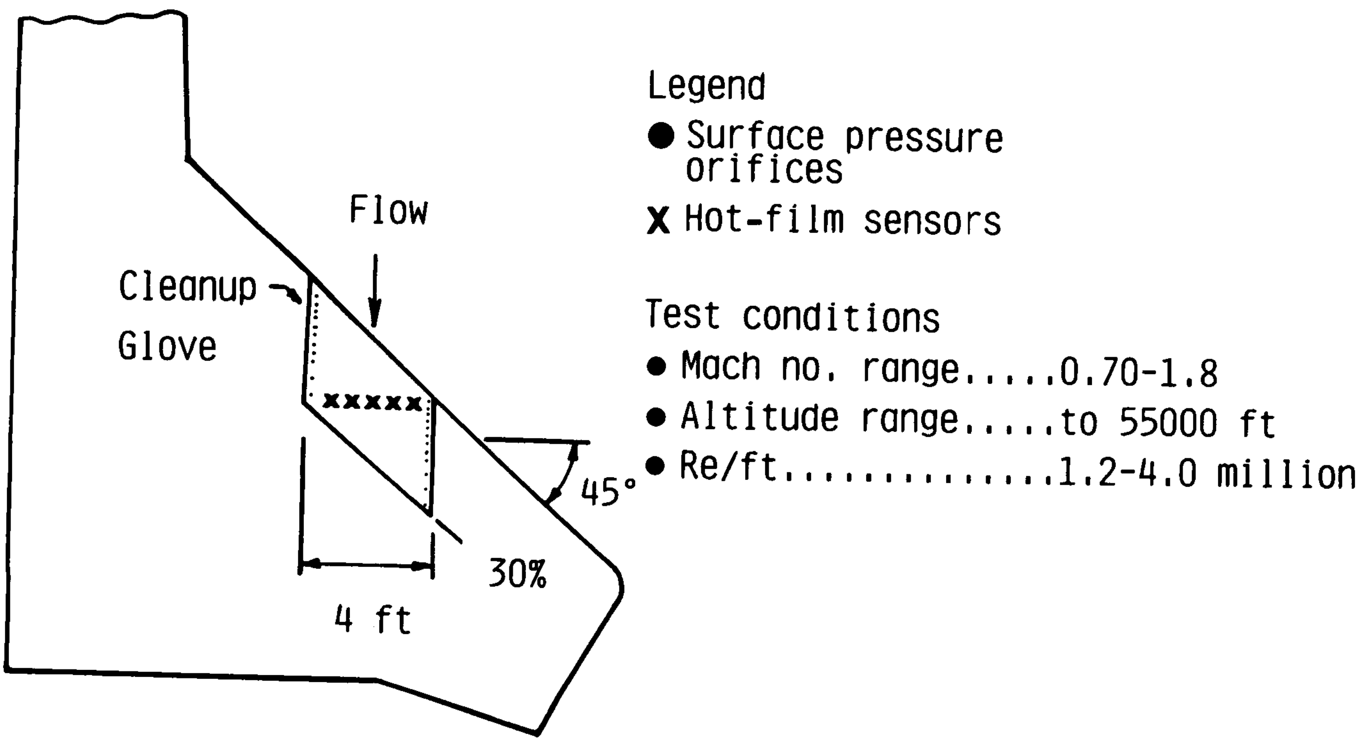
The flight experiment was conducted using an F-15 twin engine fighter type aircraft with a wing leading-edge sweep of 45 degrees. The F-15, normally used for propulsion tests at NASA Ames-Dryden Flight Research Facility, has the ability to reach speeds in excess of Mach 2. Data from previous flight experiments showed that the F-15 wing produces a pressure distribution which under ideal conditions may yield small amounts of laminar flow. A photograph of the F-15 with the foam and fiberglass test section on the right wing is shown in the figure below.

INSTRUMENTATION LAYOUT AND TEST CONDITIONS

Two separate instrumentation systems were used for the flight experiment. Quantities such as Mach number, altitude, and angle of attack were obtained from the aircraft's main instrumentation system. The main system employed two absolute pressure transducers to measure total and static pressure from the flight test Pitot static probe mounted on a noseboom.

The instrumentation system for the test section consisted of a 32 port electronic scanivalve, five temperature compensated hot-film anemometers, and an absolute pressure transducer. The electronic scanivalve was used to measure pressures from the two rows of 15 flush static pressure orifices and the absolute pressure transducer measured the pressure on the backside of the scanivalve. The temperature compensated hot-film anemometers, similar to the system described in reference 1, were used to measure transition location. Temperature compensated hot-film anemometers were used because previous experience at supersonic speeds showed that uncompensated anemometers were sensitive to the local total temperature, thereby affecting the overall sensitivity of the anemometers between flight conditions. The temperature compensated anemometers eliminate this problem. The locations of the hot-film sensors for the first phase of the flight tests were 5%, 10%, 15%, 20%, and 15% chord. The hot-film sensors were at 1%, 2%, 4%, 10%, and 15% chord for the second phase of flight tests.

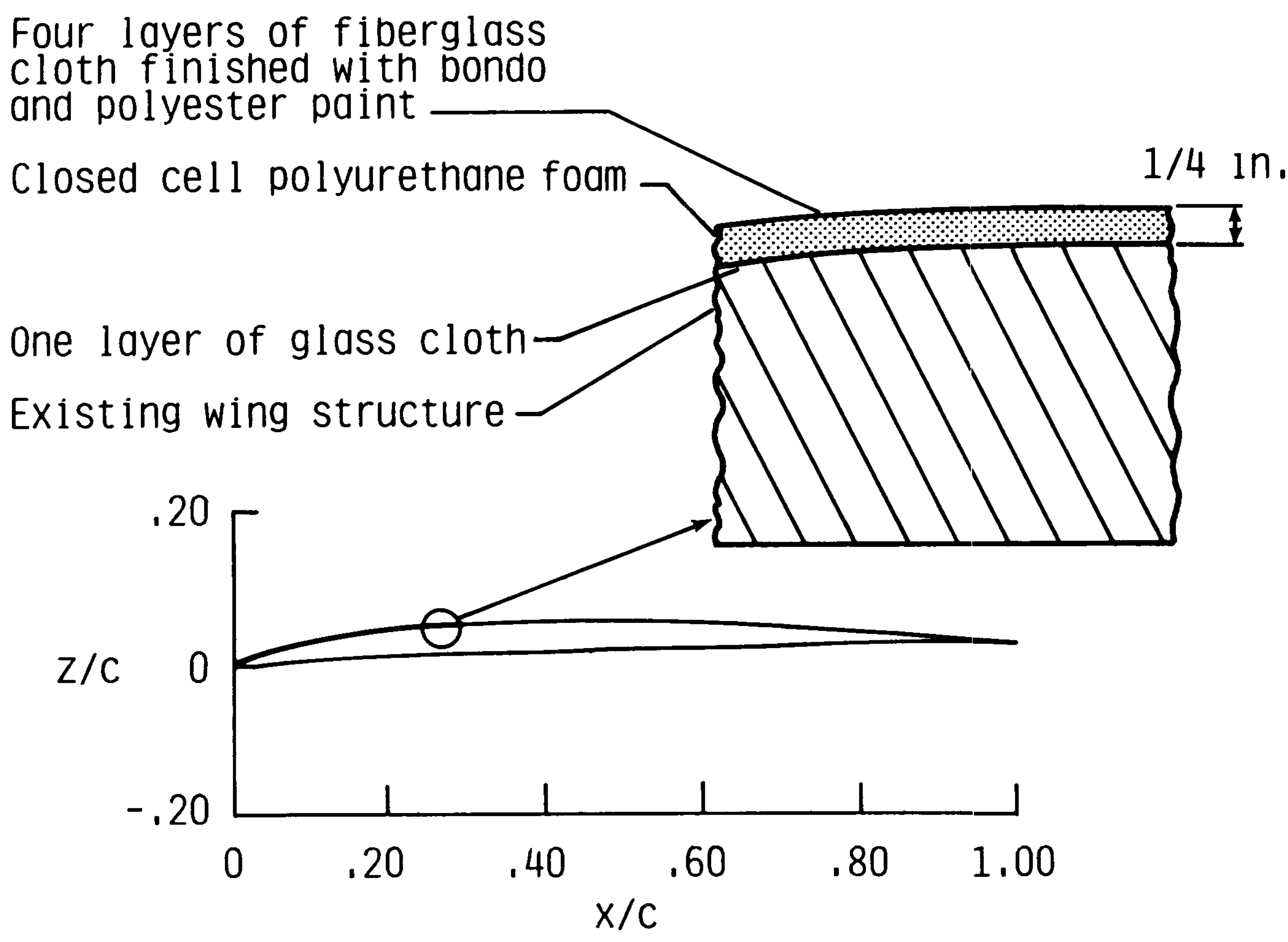
Test points were flown at Mach numbers ranging from 0.7 to 1.8 and altitudes between 20,000 feet and 55,000 feet. Angle of attack ranged from -1 to 10 degrees. The unit Reynolds number ranged from 1.2 to 4.0 million/foot. In order to vary angle of attack and hold Mach number and altitude constant, constant G-loading turns were flown.



FOAM AND FIBERGLASS TEST SECTION

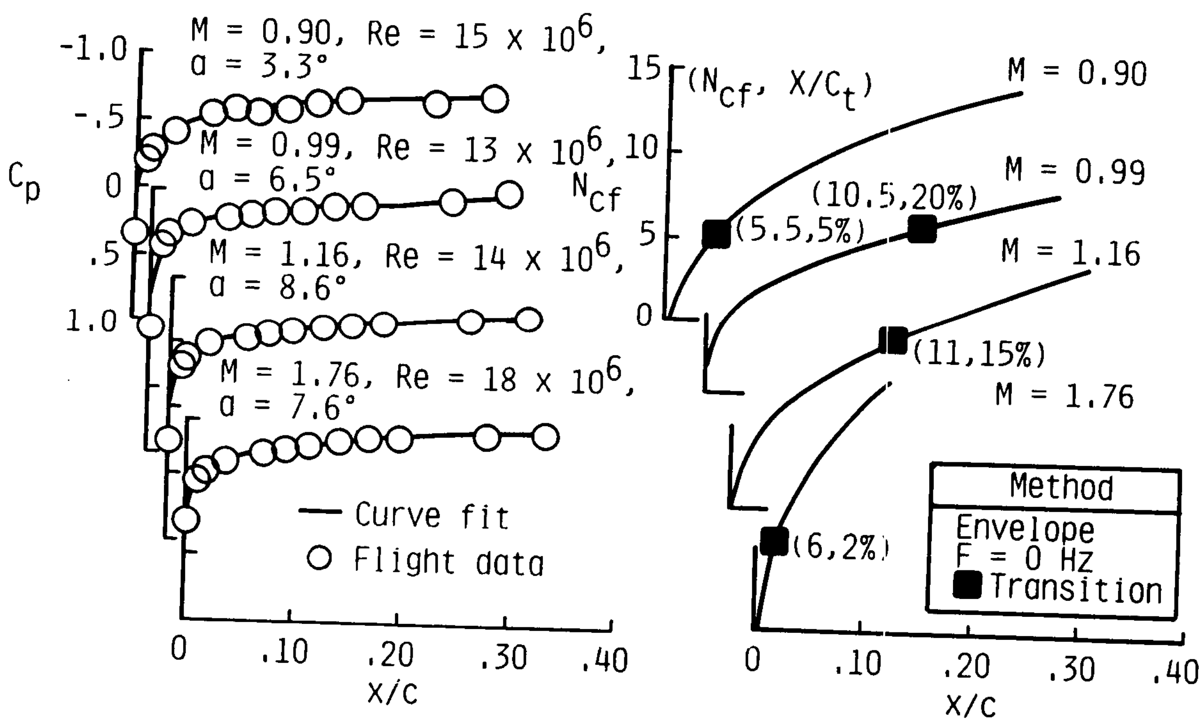
The foam and fiberglass test section was placed on the right wing of the F-15 to eliminate the possible effects of surface imperfections. The test section retained the existing airfoil shape but added approximately 1/4 inch thickness. The glove was 4 feet wide and extended past 30% chord. It was constructed using one layer of unidirectional fiberglass under 1/8 inch thick polyethylene foam covered with four layers of bidirectional fiberglass. The surface consisted of body filler and polyester paint. The waviness did not exceed 0.00075 inch/inch.

The test section configuration was changed during the latter part of the flight experiment. A notch/bump was added to the inboard side of the leading edge of the test section to eliminate the possible effects of leading-edge contamination.



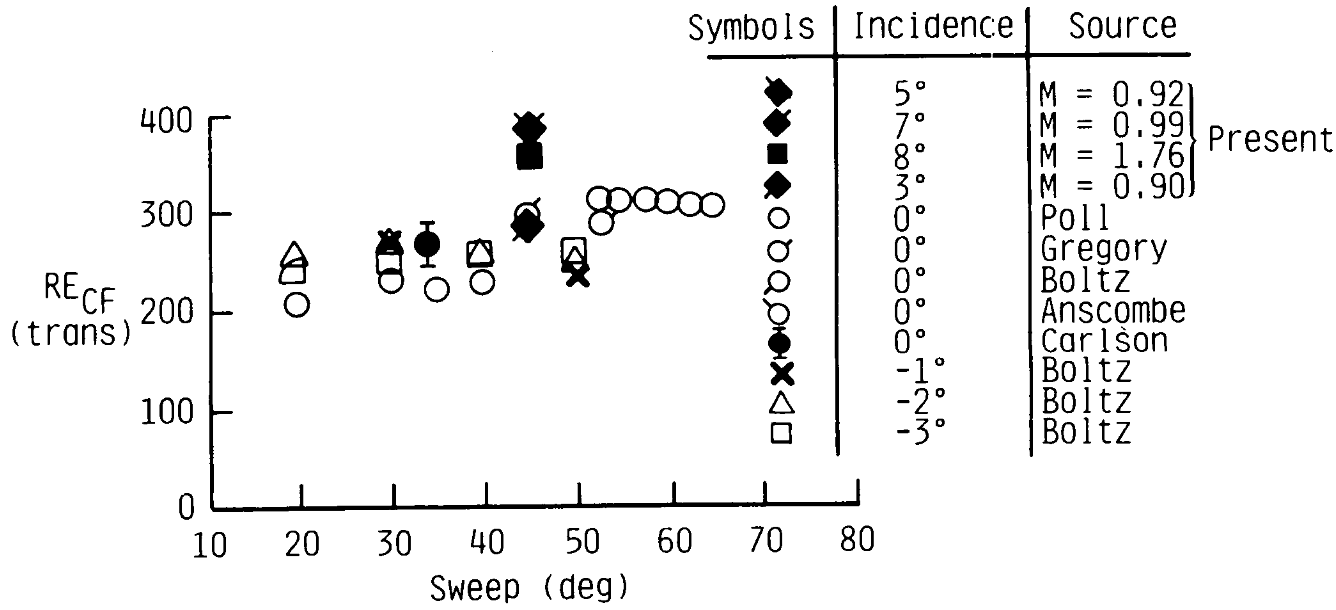
STABILITY ANALYSIS OF THE F-15 SUPERSONIC BOUNDARY-LAYER TRANSITION FLIGHT EXPERIMENT RESULTS

Shown in the figure below are four test points from the flight test. On the left, the pressure distributions as a function of chord fraction are presented for Mach numbers ranging from 0.90 to 1.76, Reynolds numbers ranging from 13 to 18 million, and angles of attack from about 3 to 9 degrees. The differences in the levels of the pressure distributions are mainly due to an angle of attack effect. Test points away from the 1-G loading condition were acquired in constant G windup turns and were considered very nearly steady state. On the right, the corresponding cross-flow N-factor results are presented as calculated from the compressible linear stability theory code COSAL. The envelope method was used and the cross-flow disturbances were assumed to be stationary (zero frequency). As can be seen from the figure, at M=0.98 and M=1.16, the cross-flow N-factor at transition was 10.5 and 11, respectively. These results are in agreement with previous subsonic transition correlations that transition should occur where the N-factor is in the range of 9-12. On the other hand, for M=0.90 and M=1.76 the cross-flow N-factors were about 6 at transition. There may be several explanations for this result. The above analysis was conducted for zero frequency disturbances and without the effects of surface curvature included. It is known that travelling disturbances (non-zero frequencies) can be more highly amplified, which could result in higher N-factors at transition. Conversely, convex surface curvature has a stabilizing effect on the disturbances. It is possible that, with further analysis which takes into account the aforementioned effects, the N-factors at transition might fall into the expected range of 9-12.



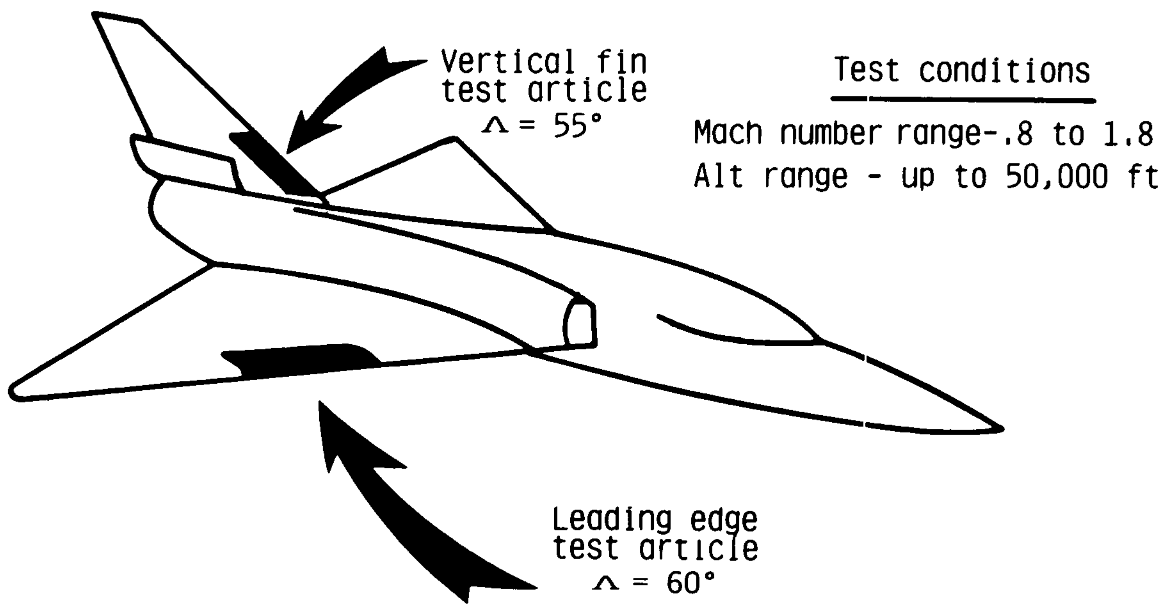
CROSS-FLOW REYNOLDS NUMBER AT TRANSITION

Another method that has been used in the past for correlating boundary-layer transition data is that of comparing the computed value of the cross-flow Reynolds number at the point of transition with previous data. Presented in the figure below is the cross-flow Reynolds number at transition as a function of sweep for a series of previous incompressible investigations as compiled by Poll (Ref. 2). The results of the present investigation show good agreement with past results.



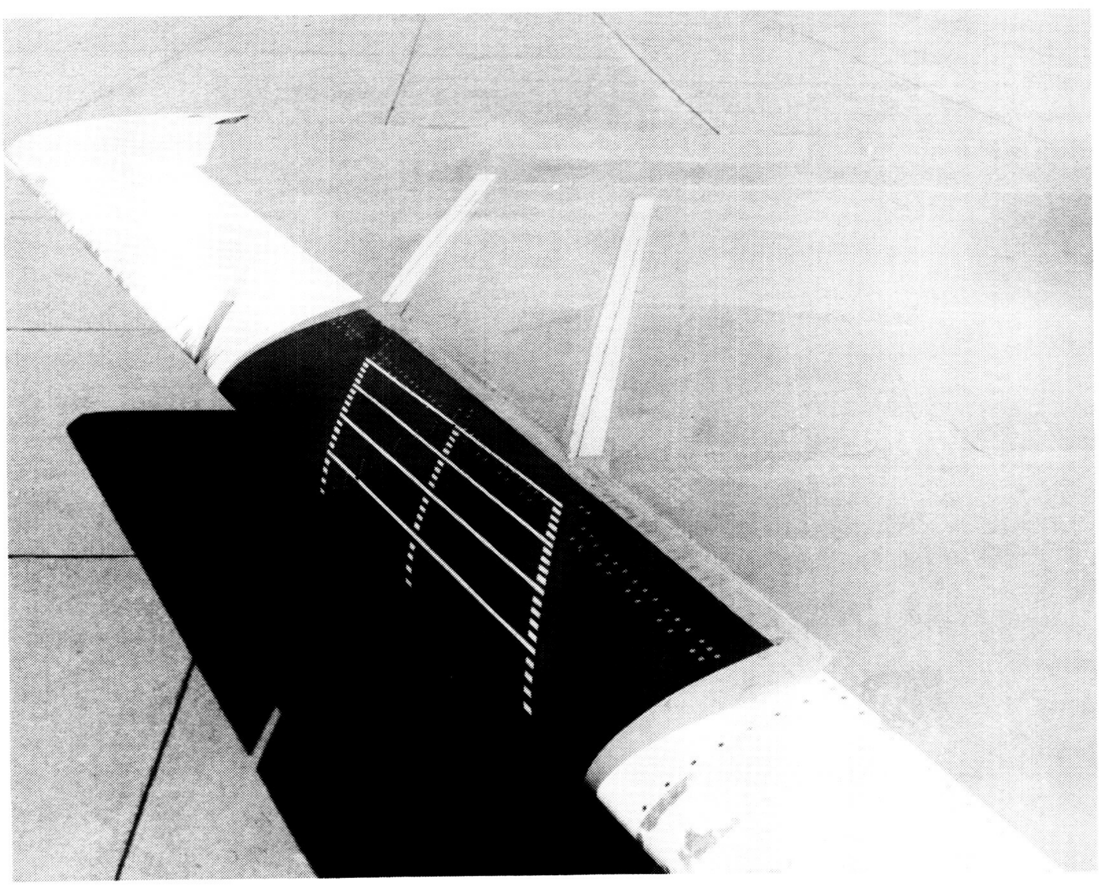
F-106 SUPERSONIC BOUNDARY-LAYER TRANSITION FLIGHT TEST

At Langley Research Center, the F-106 aircraft was utilized to conduct an exploratory supersonic boundary-layer transition flight experiment. A schematic of the F-106 is shown in the figure below. Surface clean-up gloves were mounted on the right wing which has a sweep of 60 degrees and the vertical tail which has a sweep of 55 degrees. The gloves were constructed by laminating alternate layers of fiberglass and epoxy over the existing wing. The final layer was sanded, primed, and sprayed with lacquer to achieve a very smooth finish. The gloves added approximately 0.10 inch thickness to the existing surfaces. During the flight tests, the Mach number ranged from 0.80 to 1.80, and the altitude varied from 30,000 to 50,000 feet resulting in unit Reynolds numbers in the range of 1.6 to 5.2 million per foot. The angle of attack varied from about 3 degrees up to 14 degrees. Because of fuel and supersonic air space limitations, data had to be collected in either a slowly varying acceleration or deceleration mode.



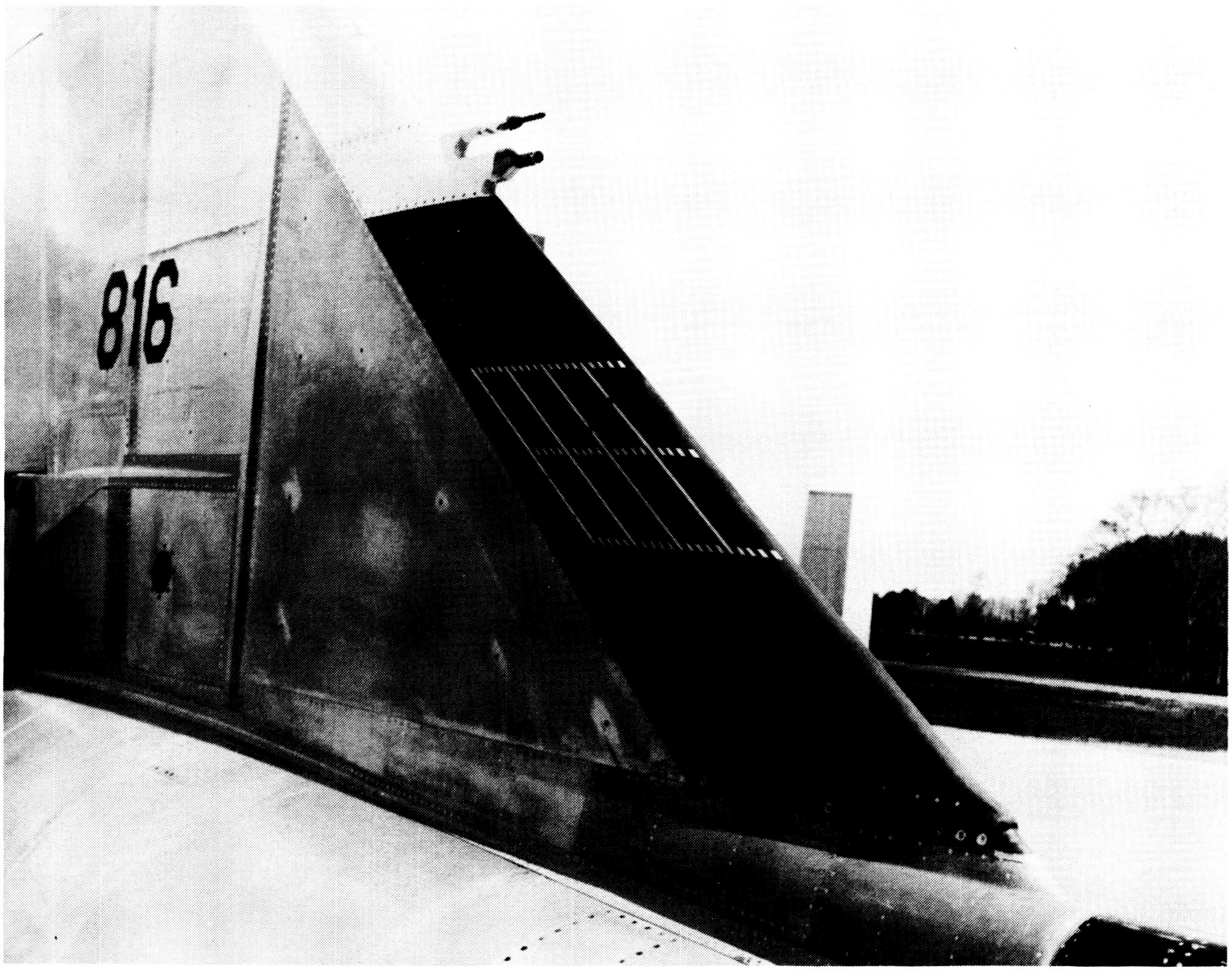
F-106 WING GLOVE

The surface clean-up glove on the right wing of the F-106 is shown in the figure below. The glove has a span of about 3 feet and extends beyond 20% chord. Surface pressure measurements were made in two rows on the wing. The inboard row consists of 18 flush taps on the glove followed by a Strip-A-Tube pressure belt with 12 taps. The outboard row consists of 20 flush orifices on the glove followed by a 10 tap Strip-A-Tube belt. The inboard and outboard pressure rows were oriented in the streamwise direction and were located at fractional semispan locations of about 0.5 and 0.6, respectively. A set of eight hot-film sensors are mounted on the glove for transition detection purposes. The hot-film sensors were mounted so that they were normal to the leading edge. The first hot film was located approximately 0.5 inch from the leading edge with others following at intervals of 1.5 inches. Since attachment line contamination was anticipated for this flight test, a Gaster bump was built onto the leading edge near the inboard edge of the glove to alleviate this problem. The best location of the bump was uncertain because of the unknown migration of the attachment line throughout the flight as flight conditions change. Its effectiveness was thus somewhat uncertain.



F-106 VERTICAL TAIL GLOVE

The surface clean-up glove for the vertical tail is shown in the figure below. There are two sets of pressure orifices as on the wing. The inboard row consists of 20 flush surface taps, and the outboard row has 16 flush surface taps. Each row is oriented in the streamwise direction. The inboard and outboard rows of pressure orifices are located approximately 15 and 31 inches from the fuselage, respectively. In addition, a set of eight hot-films were mounted on the glove for transition detection. The spacing of the hot films on the vertical tail glove was the same as that of the hot films on the wing glove. A Gaster bump was built on the leading edge near the inboard location of the glove to control attachment line contamination. On the vertical tail, the bump could be expected to be more effective than on the wing because the attachment line location (and hence the best position for the bump) was known.



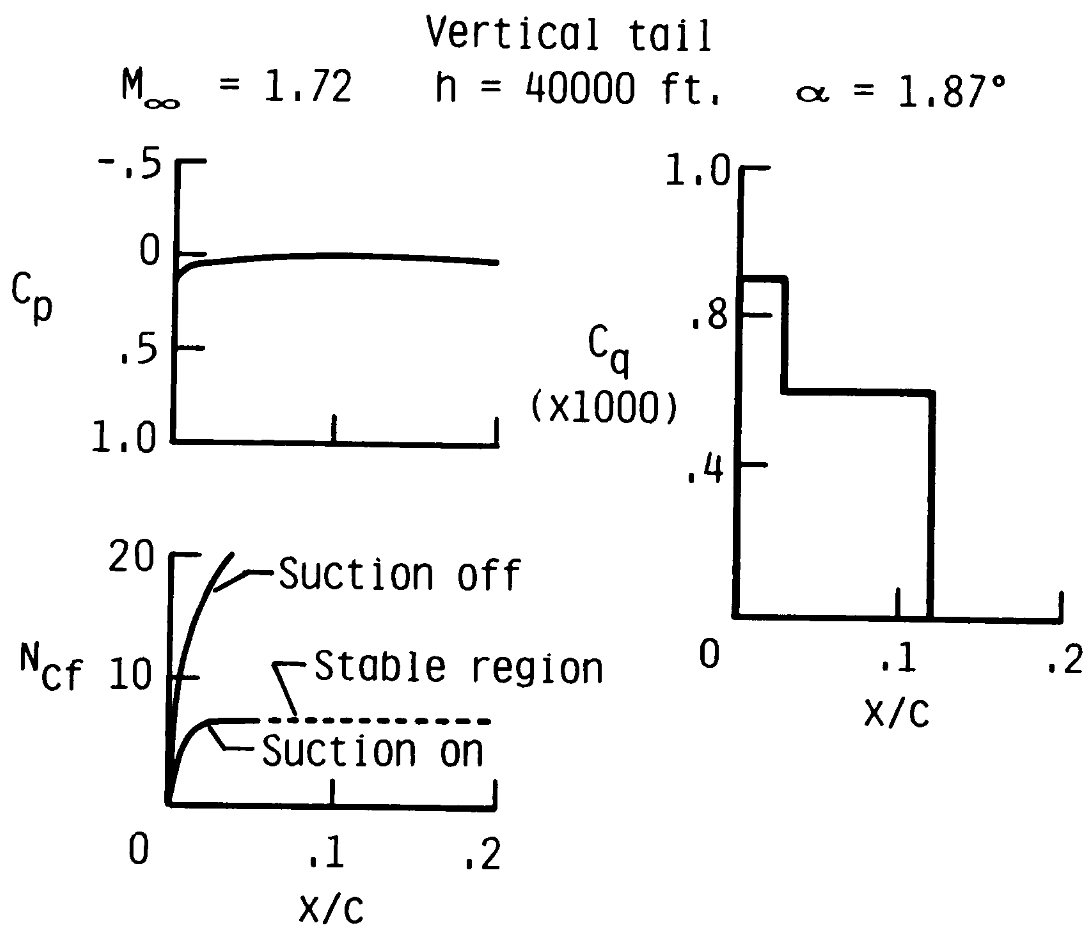
F-106 SUPERSONIC TRANSITION
FLIGHT TEST RESULTS

The table below shows four test conditions where transition was observed downstream of the attachment line. For all other test points, turbulent flow existed at the first hot-film gage (0.5% chord). Transition was dominated by attachment line contamination and rapid cross-flow disturbance growth because of the high sweep angles. For many of the flight conditions, it seems that the Gaster bump, utilized to control attachment line contamination, may not have worked; but, it has been effective in other flight tests for controlling attachment line problems and may have just needed to be "fine tuned". It has been shown in previous studies that leading-edge suction is very effective in controlling cross-flow disturbance growth which is present in highly swept applications such as those considered here.

Surface	M_∞	Hp (feet)	α (degrees)	Transition location
Wing $\Lambda = 60°$	0.86 1.50	44000 52000	7.5 5.8	1.5% c 0.5% c
Vertical $\Lambda = 55°$	0.91 1.10	48500 51000	10.5 7.7	5.0% c 1.5% c

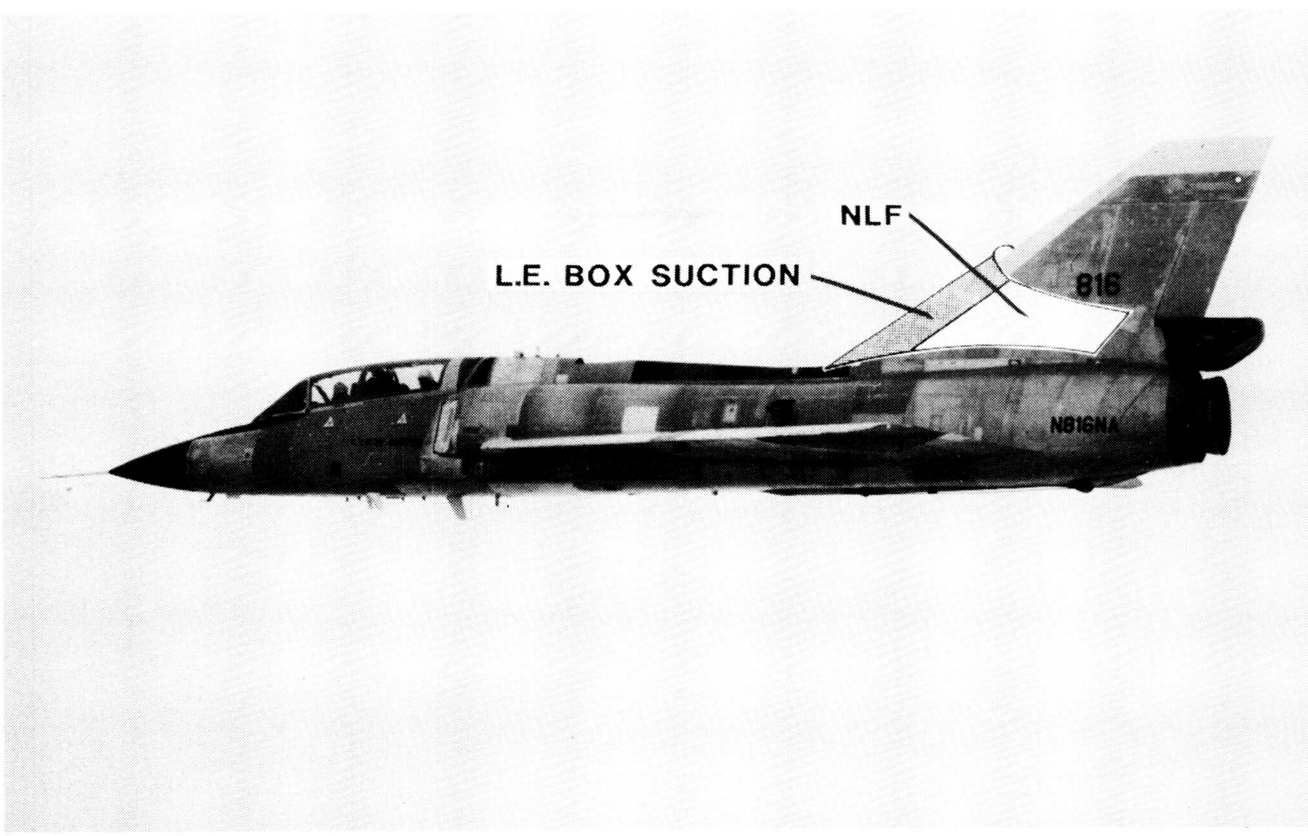
F-106 SUPERSONIC BOUNDARY-LAYER FLIGHT TEST RESULTS

Presented in the figure below is the effect of suction on the cross-flow N-factor as predicted by compressible, linear stability theory. The C_p distribution on the vertical tail is shown for M=1.72, altitude of 40,000 feet, and angle of attack of about 2 degrees. For the present flights with no suction on the glove, the results show that the cross-flow N-factor increases very rapidly in the region of the attachment line. As shown below, N_{cf} has a value of 10 at 2% chord. With the suction distribution shown on the right, the cross-flow N-factor is reduced significantly. The N_{cf} has a maximum of about 6 at 5 percent chord. The disturbances are stable or damped out to about 20 percent chord. This level of growth is considered to be below the critical value for transition.



F-106 SUPERSONIC LAMINAR-FLOW FLIGHT EXPERIMENT

The results of these two exploratory supersonic flight tests seem to indicate that to achieve laminar flow past the leading-edge wing box consistently, suction must be utilized in the leading-edge region to control cross-flow disturbance growth. Also, the problem of attachment line contamination must be addressed. By properly designing the leading edge, contamination can be eliminated. Presently, there is growing interest in conducting a supersonic flight test with suction in the leading-edge region. Shown in the figure below, is an artist's rendition of how a test article including leading-edge box suction followed by a natural laminar-flow glove would appear on the LaRC F-106. The feasibility of such a flight test is currently being studied at the LaRC.



SUMMARY

For the case of the F-15 flight tests, boundary-layer transition was observed up to Mach numbers of 1.2. For very limited and specific flight conditions, laminar flow existed back to about 20% chord on the surface clean-up glove. The hot-film instrumentation was effective for locating the region of transition.

For the F-106 flight tests, transition on the wing or vertical tail generally occurred very near the attachment line. Transition was believed to be caused by either attachment line contamination or strong cross-flow development due to the high sweep angles of the test articles.

The compressible stability analysis showed that cross-flow N-factors were in the range of 5-12 at transition. Future plans are to analyze the results including the effects of surface curvature and non-stationary cross-flow disturbances. It was shown that cross-flow Reynolds numbers at transition correlated well with previous incompressible results.

F-15 Flight Tests

- Boundary layer transition was observed up to Mach numbers of 1.2
- Hot film instrumentation worked well for locating the region of transition

F-106 Flight Tests

- Transition on the wing or vertical tail generally occurred very near the attachment line
- Transition was believed to be due to either attachment line contamination or strong crossflow development due to the high sweep

Stability analysis

- N-factors at transition were in the range of 5-12
- Crossflow Reynolds numbers at transition correlated well with previous incompressible results

REFERENCES

1. Chiles, Harry R.; and Johnson, J. Blair: Development of a Temperature-Compensated Hot-Film Anemometer System for Boundary-Layer Transition Detection on High Performance Aircraft. NASA TM 86732, 1985.

2. Poll, D. I. A.: Some Aspects of the Flow Near a Swept Attachment Line with Particular Reference to Boundary Layer Transition. Cranfield Institute of Technology, CoA Report No. 7805, 1978.