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# Use of Boundary Layer Transition Detection to Validate Full-Scale Flight Performance Predictions


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Full-scale flight performance predictions can be made using CFD or a combination of CFD and analytical skin-friction predictions. However, no matter what method is used to obtain full-scale flight performance predictions knowledge of the boundary layer state is critical. The implementation of CFD codes solving the Navier-Stokes equations to obtain these predictions is still a time consuming, expensive process. In addition, to ultimately obtain accurate performance predictions the transition location must be fixed in the CFD model. An example, using the M2.4-7A geometry, of the change in Navier-Stokes solution with changes in transition and in turbulence model will be shown. Oil flow visualization using the M2.4-7A 4.0% scale model in the 14'x22' wind tunnel shows that fixing transition at 10% x/c in the CFD model best captures the flow physics of the wing flow field.

A less costly method of obtaining full-scale performance predictions is the use of non-linear Euler codes or linear CFD codes, such as panel methods, combined with analytical skin-friction predictions. Again, knowledge of the boundary layer state is critical to the accurate determination of full-scale flight performance. Boundary layer transition detection has been performed at 0.3 and 0.9 Mach numbers over an extensive Reynolds number range using the 2.2% scale Reference H model in the NTF. A temperature sensitive paint system was used to determine the boundary layer state for these conditions. Data was obtained for three configurations: the baseline, undeflected flaps configuration; the transonic cruise configuration; and, the high-lift configuration. It was determined that at low Reynolds number conditions, in the 8 to 10 million Reynolds number range, the baseline configuration has extensive regions of laminar flow, in fact significantly more than analytical skin-friction methods predict. This configuration is fully turbulent at about 30 million Reynolds number for both 0.3 and 0.9 Mach numbers. Both the transonic cruise and the high-lift configurations were fully turbulent aft of the leading-edge flap hingeline at all Reynolds numbers.

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This presentation is again the successful result of the collaboration of NASA, McDonnell Douglas, and Boeing researchers in planning and testing an HSCT-class configuration under a wide variety of conditions. It focuses on the affect the boundary-layer state has on our ability to predict full-scale flight performance.

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# Acknowledgments

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  - **Team Leader:** T.G. Popernack, Jr.
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  - **Purdue University:** J. Sullivan
  - **University of Florida:** K. Schanze
  - **Old Dominion University:** D.M. Oglesby

This paper represents more than just a wind tunnel test or a CFD study. It is the combined effort of a huge number of researchers. Of particular interest is the LaRC Temperature Sensitive Paint Team who put together the transition detection technique that will ultimately enhance full-scale flight performance predictions from low Reynolds number wind tunnel data. This team brings an immense body of knowledge to bear on the problem of transition detection. It is composed of engineers and scientists from NASA, industry, and academia.

# Acknowledgments

- **NTF TSP Staff:**
  - Thomas E. Deans, Jr., Jerry B. Adcock, Ronald Shaiken, Joshua L. Hester, W. Allen Kilgore, Robert J. Done, P. Anthony Guthrie, Pat Quander Jones, Bruce Raiford, John F. Bledsoe, Greg F. Walkup, Arbria L. Wright, Dave Butler, John L. Stobierski, J. Russel McHatton, William G. Johnson, Jr., W. Chris Alexander, David Coleman, John W. Meador
- **McDonnell Douglas Staff:**
  - Marty Morris, Mike Benne, Tom Kihlken, John Donovan

More than just the TSP Team, the people who made the initial tests using this transition detection technique work should be recognized. Without the extraordinary effort and dedication of the NTF TSP Staff we would not have the success story told in this presentation. The NTF TSP Staff put in the extra effort, often repeating conditions, and pushing tunnel operations beyond perceived limits that allowed us to acquire the wide range of transition data we now have.

# Presentation Outline

- Effect of Fixing Transition on CFD Solutions
- Transition Effects in Analytical Skin-friction Predictions
- Transition Detection using Temperature Sensitive Paints
- Summary
- The Next Steps

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## Overall Wind Tunnel Test Objectives

- **Obtain free and fixed transition data on three configurations: baseline, high-lift, and transonic**
  - Use data to design/validate a transition fixing methodology for HSCT-class configurations
  - Incorporate data in full-scale flight performance prediction methodology
- **Obtain data for CFD validation**

## Goal

- **Be able to predict full-scale flight performance using low Reynolds number wind tunnel test data *with confidence***

Last year I said that, "To develop full-scale performance predictions an understanding of Reynolds number effects on HSCT-class configurations is essential." Today we still have the same overriding premise in our wind tunnel test objectives. Our ultimate goal is to be able to predict full-scale flight performance using the data we acquire during configurations development, at low Reynolds number. When I say, "with confidence," I mean that we should be able to say what the level of confidence is in our predictions.

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# Model/Configurations Definition Used for Transition Tests

**2.2% Model used for testing in the NTF**

**Wing: Reference H wing**

**Flap deflections available:**

| | LE | TE |
|---|---|---|
| Baseline | 0/0 | 0/0 |
| Transonic | 0/10 | 0/3 |
| High Lift | 30/30 | 10/10 |

**Body: Fuselage truncated at station 60.8150**

**Nacelles: Axisymmetric**

The 2.2% scale Reference H model used for transition testing at the NTF includes:

- wing - with various flap deflections representing high-speed and high-lift configurations
- fuselage
- axisymmetric nacelles

The truncated fuselage is run on the straight sting. Trips normally applied include the forebody ring and nacelle internals. A "conventional" wing tripping scheme based on Braslow criteria was used to obtain the fixed transition data.

# Model/Configurations Definition

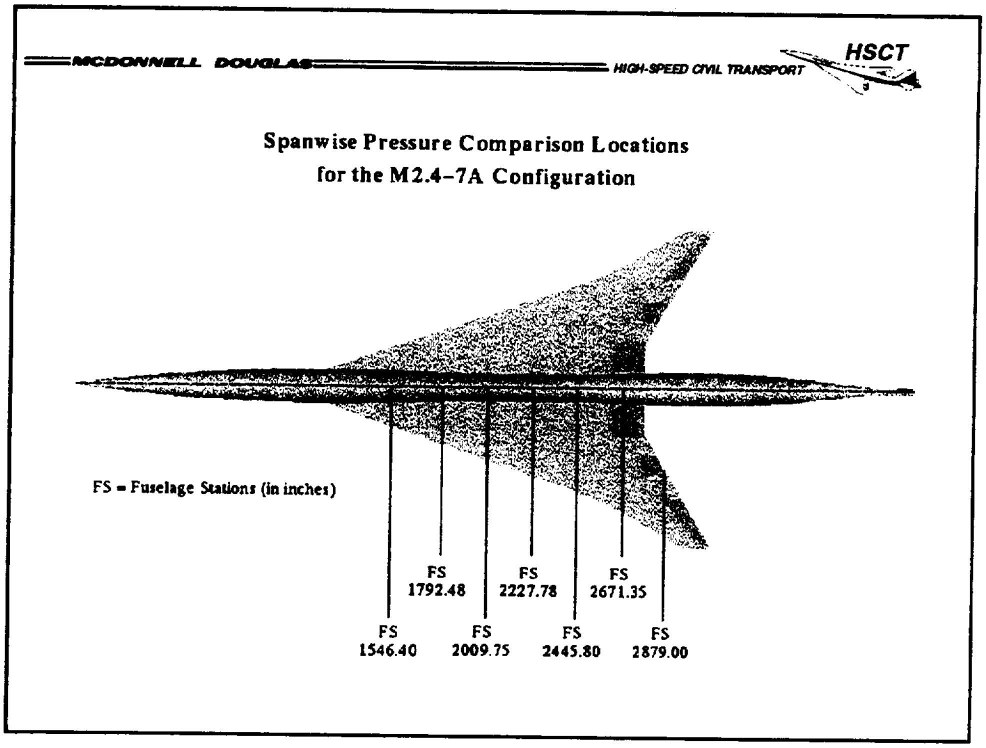
**4.0% model used for testing in the 14'x22'**

| | |
|---|---|
| **Wing:** | **M2.4-7A Arrow wing**<br>**Flap deflections available: various** |
| **Body:** | **Complete fuselage** |
| **Nacelles:** | **Axisymmetric** |

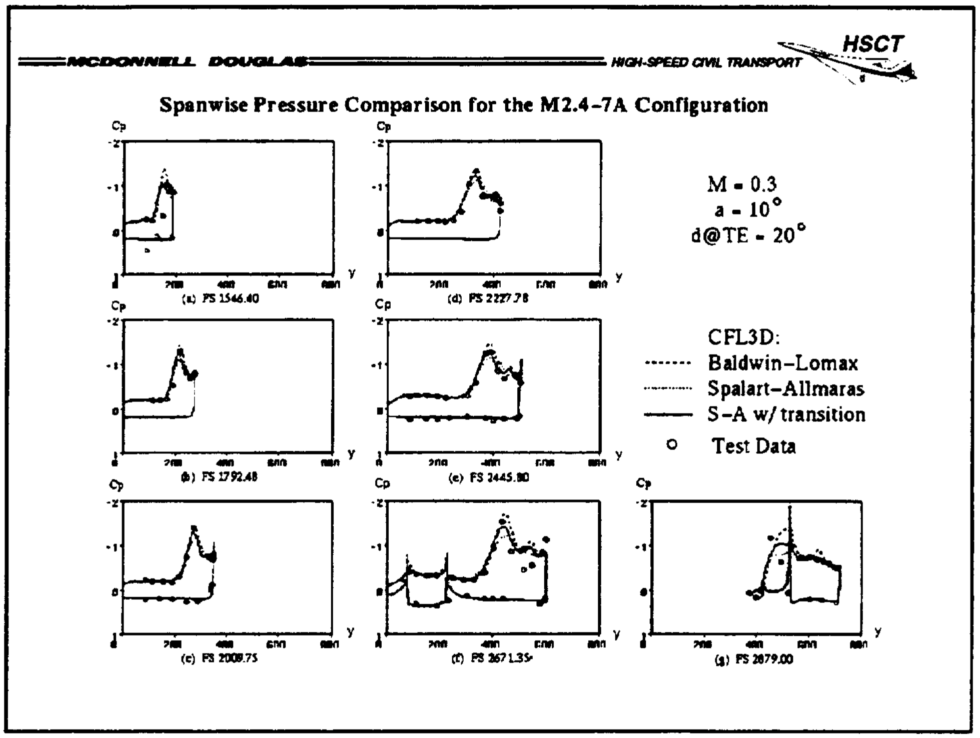
The 4.0% scale M2.4-7A Arrow wing model used at the 14'x22' includes:

- wing with various leading- and trailing-edge flap deflections
- complete fuselage and tails
- axisymmetric nacelles

The model is run on a post mount.




The spanwise pressure distribution at various stations can be used to illustrate the effect of describing the boundary layer state on the CFD solution. This figure shows which stations are used.




CFD solutions using two different turbulence models were obtained as well as the solution fixing transition at 10% x/c. This slide illustrates the difference in the solutions obtained for these cases. In addition to determining which turbulence model to use, describing the boundary-layer state plays an important role in obtaining CFD solutions that best model wing flow field.

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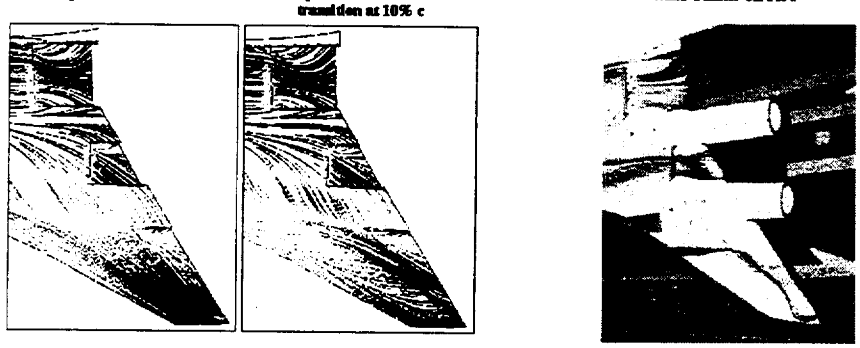
## Comparison of Wind Tunnel Oil Flow and CFD Solutions Near the Trailing-Edge for the M2.4-7A Configuration

Trailing-Edge Flap Deflection = 20°

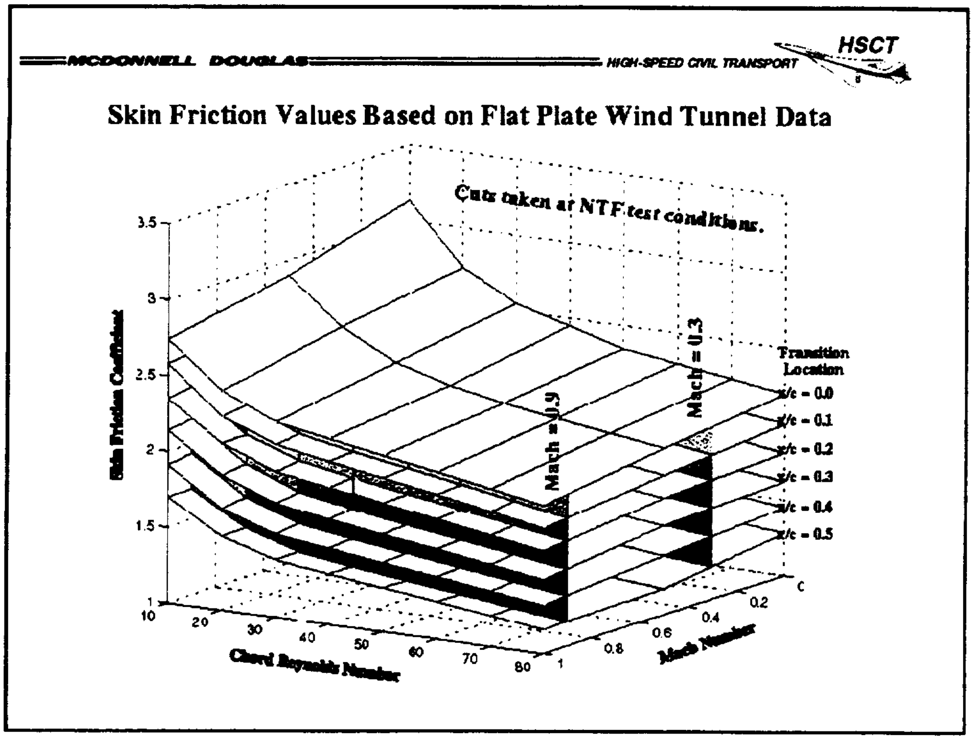
Alpha = 10 Degrees

Effect of Fixing Transition on Computed 'Oil Flow'

- Spalart-Allmaras
- Spalart-Allmaras transition at 10% c
- Wind Tunnel Oil Flow



As shown in this figure, fixing transition at 10% x/c in obtaining the CFD solution better models vortex formation.




This slide illustrates the variation of flat plate skin friction coefficient with Reynolds number and Mach number for various transition locations. A cut taken at an NTF test condition yields a family of curves representing the flat plate skin friction coefficient as a function of Reynolds number at various transition locations. This cut represents a linar interpolation between original data at Mach 0.0, 0.5, and 1.0. This data was obtained from the "Clutter charts," Douglas Aircraft Company, Inc., Report Number ES 29074. They represent a smooth, insulated flat plate.

# Analytical Skin-friction Calculations

**Skin-friction drag = Contribution from wing + Contribution from fuselage**

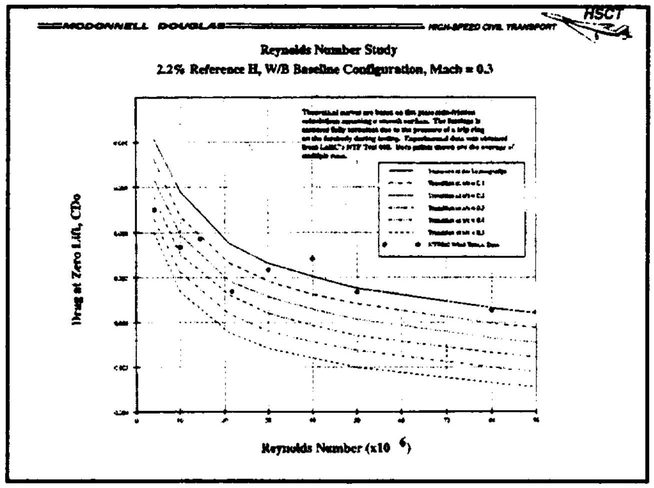
$$= \Delta C_{D_{Wing}} + \Delta C_{D_{Fuselage}}$$

**Each contribution = Form Factor $*$ ($S_{wet}/S_{ref}$) $*$ $C_f$**

**Form Factor, $S_{wet}$ and $S_{ref}$ based on physical geometry**
**$C_f$ based on previous analytical and experimental work**

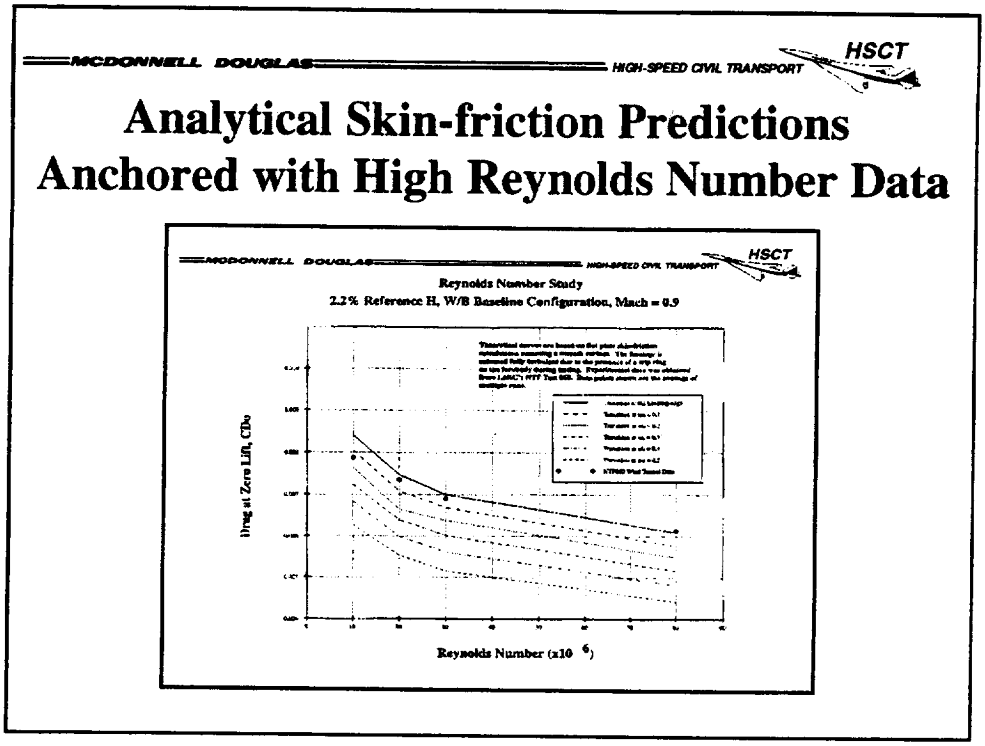
The flat plate skin friction coefficient is scaled by the form factor, the wetted area, and the reference area. These factors are based on physical geometry. The TI group provided the values of these factors. Because of the presence of the forebody trip ring, the fuselage can be considered fully turbulent. Thus the contribution from the fuselage becomes constant based on Reynolds number while overall skin-friction drag varies as a function of transition location on the wing.

# Analytical Skin-friction Predictions Anchored with High Reynolds Number Data



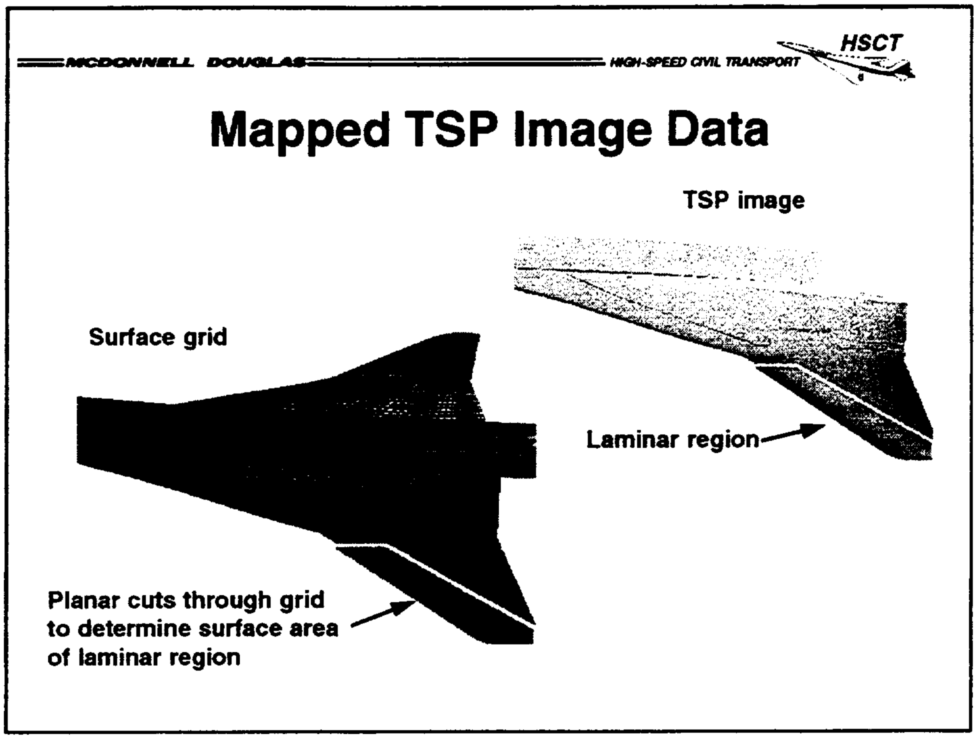

The chart in this slide was presented last year. It represents the scaled flat plate skin friction coefficient for various transition locations, anchored at the minimum drag level for the high Reynolds number condition. This particular slide includes data for the Mach 0.3 case. The 40 million Reynolds number data shown in this chart has since been determined to be bad.

This chart illustrates the variation in transition location as a function of Reynolds number.




The chart in this slide was also presented last year. It represents the scaled flat plate skin friction coefficient for various transition locations, anchored at the minimum drag level for the high Reynolds number condition. This particular slide includes data for the Mach 0.9 case.

This chart also illustrates the variation in transition location as a function of Reynolds number.




The capability of directly determining the boundary layer state, that is laminar versus turbulent, allows us to reconsider the analytical skin friction predictions. To determine the square inches of laminar boundary layer present the 2-D TSP image acquired during two NTF tests last year was mapped to a 3-D grid. Because the extent of the laminar boundary layer is not symmetric, this grid was split into upper and lower surfaces. These surfaces were cut with planes determined by two points at the edge of the laminar boundary layer nearest the side-of-body and two points at the edge of the laminar boundary layer nearest the trailing-edge. This technique disallows turbulent wedges issuing from areas of damaged paint. However, for conditions where a larger transitional region occurs it may overstate the extent of the laminar boundary layer. The surface area representing this "laminar region" was then computed.

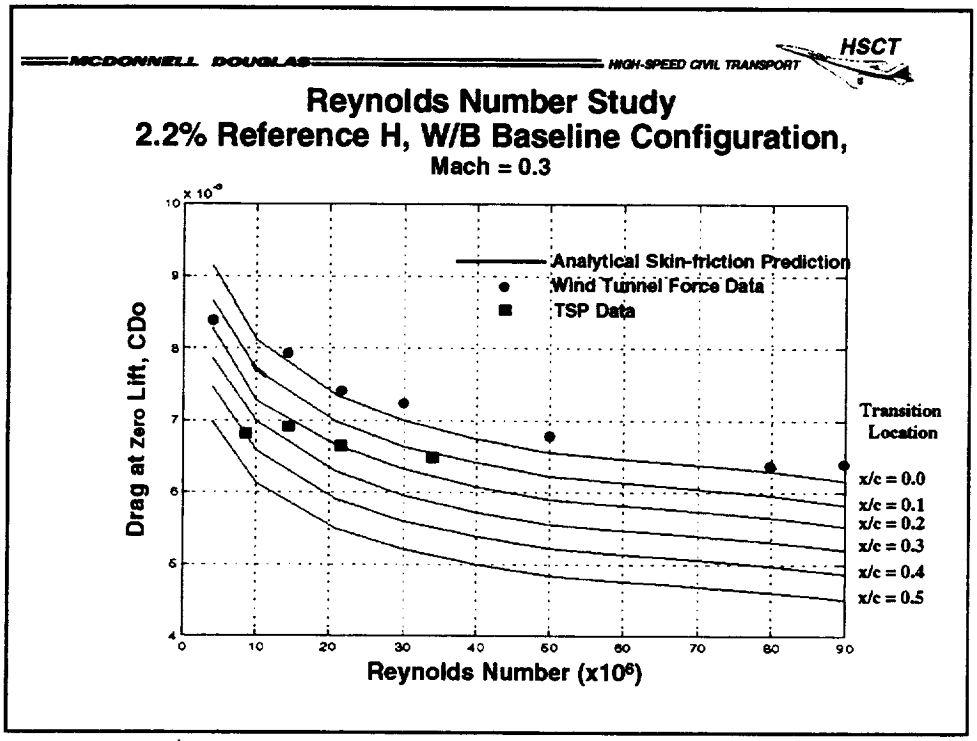
## Percent Laminar Surface Area

| Mach Number | Chord Reynolds Number | Laminar Area ($in^2$) | Percent Laminar Area Based on Wimpress Area | Percent Laminar Area Based on Gross Area |
|---|---|---|---|---|
| 0.3 | 8.5 | 109.9 | 44.5 | 39.6 |
| | 14.4 | 65.4 | 26.4 | 23.5 |
| | 21.6 | 54.6 | 22.1 | 19.7 |
| | 34.0 | ~32.8 | 13.3 | 11.8 |
| 0.9 | 10.2 | 93.8 | 37.9 | 33.7 |
| | 20.0 | 50.9 | 20.6 | 18.3 |
| | 30.0 | 35.3 | 14.3 | 12.7 |

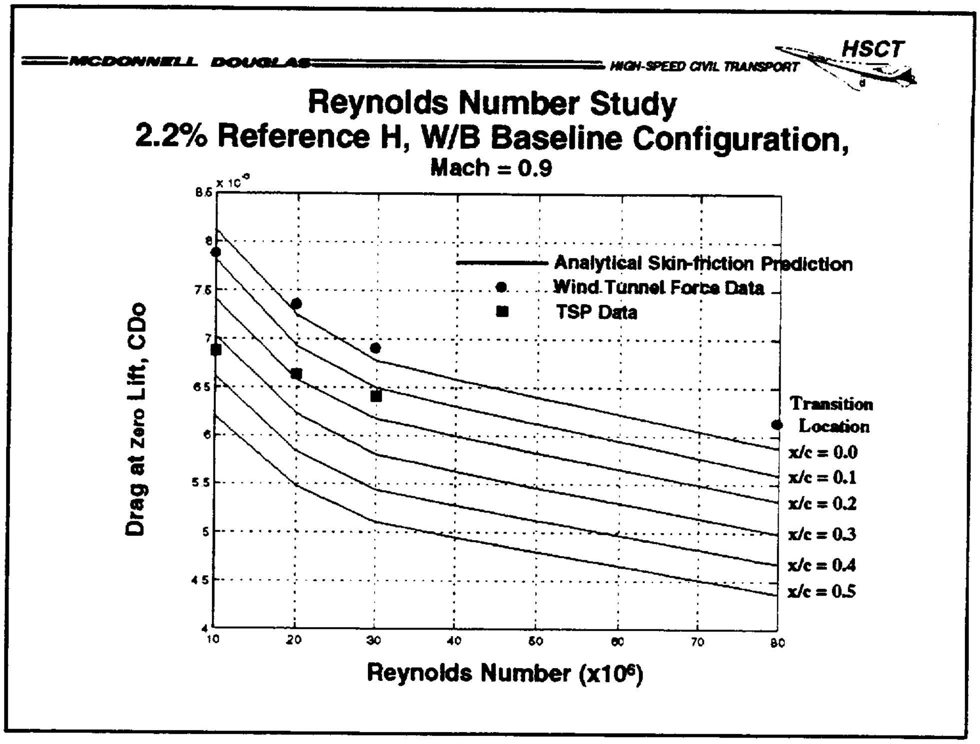
This table in this slide shows the computed "laminar region" for various test conditions. Because no lower surface data was obtained for the Mach 0.3, 34.0 million Reynolds number condition, the upper surface laminar area was doubled to obtain the value shown. Specific values for upper and lower surface areas are available on request.

To obtain this table the following assumption was made.

1) It was assumed that the flat plate skin coefficient data was obtained at zero degrees angle of attack. Since the twist on the outboard panel (where most of the laminar boundary layer exists) is about one and one-half degrees, this table was computed for data obtained at one degree angle of attack.

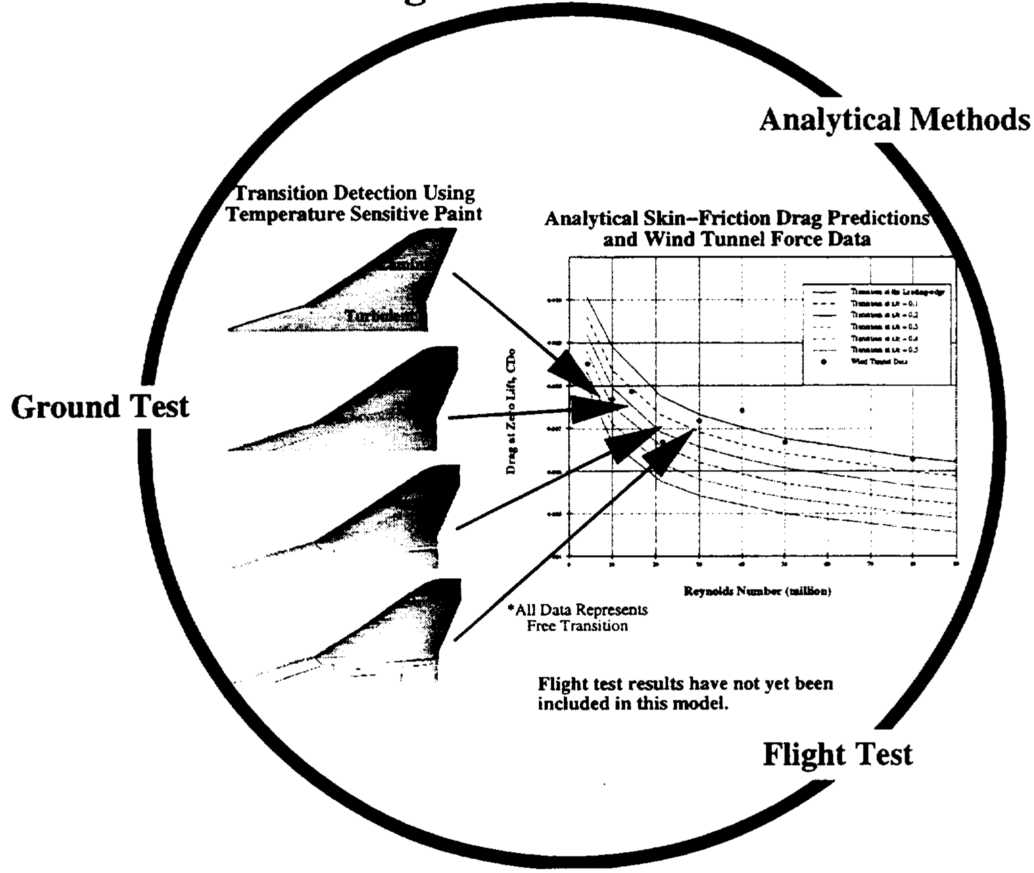



This slide illustrates the analytical skin friction predictions for various transition locations, the wind tunnel force data previously acquired, and the computed laminar surface areas at Mach 0.3. Previously, anchoring the analytical skin friction curves using high Reynolds number data moved the curves such that there appeared to be more laminar flow at low Reynolds numbers than analysis alone predicts. Direct determination of the laminar surface area bears this out. However, there still appears to be a discrepancy at low Reynolds numbers. This may be due to the presence of other phenomena such as separation. It may be also be due to data quality.




This slide illustrates the analytical skin friction predictions for various transition locations, the wind tunnel force data previously acquired, and the computed laminar surface areas at Mach 0.9. As in the previous slide, when the analytical skin friction curves were anchored using high Reynolds number data the curves moved such that there appeared to be more laminar flow at low Reynolds numbers than analysis alone predicted. Again, direct determination of the laminar surface area bears this out. And again, there still appears to be a discrepancy at low Reynolds numbers. This may be due to the presence of other phenomena such as separation. It may be also be due to data quality. However, because the Mach 0.9 data is acquired at higher dynamic pressures data quality issues in coefficients generally become less observable.

# Boundary Layer Transition Detection Reduces Risk in Full–Scale Flight Performance Predictions




**Summary illustrating scaled skin–friction curves anchored using high Reynolds Number data, NTF wind tunnel data, and NTF TSP data. This figure illustrates the consistency in trends and levels between the three data sources. It also depicts the interdependency between over all design techniques, that is, between ground test, flight test, and analytical methods.**

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## Summary

- Fixing transition has a significant effect on CFD solutions.
- Measured drag levels at low Reynolds numbers are somewhat different than changes in boundary-layer state show indicating the presence of other phenomena, for example separation.
- Knowledge of the boundary-layer state allows anchoring analytical skin friction predictions with low Reynolds number data.
- An assessment of the confidence level can be made.

As shown in this presentation, fixing transition has a significant effect on CFD solutions. This can be seen in both the resulting pressure distributions and in surface streamlines illustrating vortex formation.

Measure drag levels indicate the presence of phenomena other than boundary layer transition. Trends across Mach numbers between force data and transition data are consistent.

Once the boundary layer state has been determined analytical skin friction predictions can be anchored and full-scale flight performance predictions completed. An assessment of the confidence level of the full-scale flight performance prediction can be made by determining upper and lower bounds on the extent of laminar surface area and force data quality.

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## The Next Steps

- Acquisition of low Reynolds number transition data on additional models to support methodology development.
- Continued transition detection development.
- Adaptation of methodology to configurations and conditions of interest.
  - The high-lift configuration appears to be fully turbulent at angles of attack of interest.
  - The outboard leading-edge flap is laminar at angles of attack of interest for the transonic configuration.
  - The cruise configuration maintains some laminar flow outboard at the cruise angle of attack.
- Incorporation of stability code predictions.

Continued effort to determine the extent of the laminar boundary layer including acquisition of data on models at low Reynolds numbers will be key in fully developing a methodology for full-scale flight performance predictions. This includes continuing to develop transition detection techniques and understanding how to apply this methodology to additional configurations at a variety of conditions. Incorporation of stability code results will play a major role in developing computational techniques that completely model the flow physics present.