Extensive Supersonic Natural Laminar Flow on the Aerion Business Jet

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Few aircraft are specifically designed to emphasize passive laminar flow at the preliminary design stage. The Aerion natural laminar flow supersonic business jet concept requires just that. This paper discusses some details of the unique technology developed for the design of Aerion’s revolutionary jet. An overview of the laminar flow features of the configuration are presented, along with a summary of the design-oriented transition prediction methodology used for aerodynamic design of the airplane. Results from aerodynamic shape optimizations of the business jet and of a new high Reynolds number supersonic laminar flow experiment are also presented.

I. Introduction

Figure 1. The Aerion Corporation supersonic business jet.

Dr. Richard Tracy has championed the concept of supersonic natural laminar flow aircraft for many years. The idea has slowly gained momentum and is now a central feature of the Aerion Corporation’s supersonic corporate jet shown in Figure 1. Considerable work has been completed in the last several years, including detailed aerodynamic design of the complete aircraft configuration and the design of a full-scale Reynolds number laminar flow experiment.

Unfortunately, supersonic natural laminar flow (SSNLF) aircraft are particularly tricky to design. It is essentially an attempt to use passive laminar flow control by careful design of the aerodynamic shape. The Aerion aircraft, like most supersonic designs, has a low aspect ratio wing, therefore shape design must involve three-dimensional analysis of the entire configuration. Simply incorporating a laminar flow airfoil, as might

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be done for low-speed aircraft, does not take advantage of most of the laminar flow extent possible in the supersonic regime.

Aerion’s basic SSNLF concept involves a low-sweep wing with sharp-nosed sections similar to biconvex airfoils. This permits the stabilization of the laminar boundary layer with favorable streamwise pressure gradients without inducing large spanwise gradients that lead to crossflow transition. The sharp leading edges also reduce wave drag and eliminate attachment-line transition. Figure 2 is a schematic of the laminar extent possible with Aerion’s configuration compared to a conventional delta-wing airplane. The conventional airplane’s blunt and highly swept leading edges induce both a lot of boundary-layer crossflow and significant adverse streamwise pressure gradients aft of the leading edge. This results in very little laminar flow possible on a conventional delta-wing configuration. It should be noted that there are ways to delay transition on delta-wing airplanes, as is currently being pursued at the Japan Aerospace Exploration Agency, but those efforts also involve compromises, and the extent of laminar flow still cannot match what is possible with Aerion’s low-sweep wing.

The low-sweep configuration leads to a string of design issues since, for instance, the wing has to be very thin to reduce the wave drag which in turn affects the structural design and available fuel volume. In short, the payload-range capability of the airplane is in jeopardy unless careful multidisciplinary trades are performed. Low wing sweep can also pose more challenges if designing for quiet sonic boom characteristics. Because low wing sweep has such a dramatic effect on many aspects of the design, there is a strong need to accurately predict transition. Simplistic and overly conservative transition prediction methods will unfairly penalize an SSNLF design.

Although laminar flow is a central feature of Aerion’s concept, and the main focus of this paper, its drag-reducing capability is not its only contribution. While the thinness of the wing poses design problems, the low wing sweep is a genuine aerodynamic advantage. The SSNLF airplane has a higher aspect ratio wing than conventional supersonic aircraft, giving it substantially better low-speed performance. This includes a truly long-range cruise capability at high subsonic and low supersonic speeds and also helps shorten takeoff and landing field lengths. Balanced field length requirements, which are more aggressive for business jets than for airliners, are more easily met with the SSNLF configuration than conventional delta-winged configurations. This is not only due to the larger aspect ratio wing, but more importantly, due to the generous amounts of trailing-edge flaps that can be used. The high-lift advantages of the configuration outweigh the adverse
effects of sharp leading edge airfoils on maximum lift.

The capability to cruise efficiently at transonic and low supersonic speeds is important for meeting the present regulatory restrictions on supersonic flight over certain countries. For instance, over the United States, supersonic flight is prohibited, but the Aerion airplane can cruise efficiently at Mach 0.95 to 0.98. Over many other countries, only the generation of a perceptible sonic boom is prohibited, and the Aerion airplane can then cruise at about Mach 1.1 to 1.2, depending on local winds aloft and other atmospheric properties. This is another significant advantage over conventional supersonic aircraft which is especially important for business jets since a plane change or an extra fuel stop on either end of an over-water leg is exactly the inconvenience that corporate customers are trying to avoid.

The laminar flow concept, in essence, can be viewed as an alternative to variable wing sweep and its inherent weight, cost and reliability penalties. However, just as hinging the wings drives many aspects of variable-sweep aircraft design, the SSNLF design is highly driven by the need for compatibility with extensive laminar flow extent.

Prior to this program, transition prediction was rarely incorporated into the early design of an aircraft other than the occasional use of a laminar flow airfoil. For the Aerion design, a custom design-oriented transition prediction methodology is utilized since existing transition prediction software are typically research-oriented, require expert interaction and cannot be automated. With this custom tool, important configuration design trades as well as more detailed aerodynamic lines can be analyzed and numerically optimized while taking into account the effects on transition location and overall cruise drag.

The following sections summarize the transition prediction methodology and its use in the design of both the Aerion business jet and a supersonic natural laminar flow experimental test article.

II. Design Approach

Conventional aerodynamic design of supersonic aircraft generally involves the minimization of wave drag. As such, the bulk of the work can be done using inviscid flow computations like supersonic panel methods or Euler computations. However if even relatively rudimentary transition prediction is to be accomplished, a detailed boundary-layer computation is necessary.

On the other hand, 3D Navier-Stokes computations are not yet capable of the quick execution times necessary for this type of design problem. A less expensive, yet reasonably accurate method of boundary layer computation compatible with numerical design optimization is used. The method provides a calculation of skin friction, including sufficient modeling to enable boundary layer transition prediction, while demanding little computational resources since the analysis is called thousands of times in the course of a design optimization procedure.

The computational approximation chosen for this work is the combination of a fast and reasonably automatic 3D inviscid flow solver, such as linear panel method or a Cartesian-grid nonlinear Euler solver with a boundary-layer code. Similar viscous-inviscid solutions are known to give good results in subsonic and transonic flight regimes. For supersonic flight, this approach is even more attractive since the boundary
layer is very thin (due to high Reynolds numbers) and the pressure distributions are typically very clean (no suction peaks or standing shocks on the wing).

II.A. Transition Prediction Methodology

Unfortunately, the fidelity of the boundary-layer solution necessary for transition prediction is considerably more demanding than for drag prediction. Where a simple two-dimensional integral boundary-layer solver might be enough to get a very good drag estimate when transition location is fixed, it does not provide the necessary information to determine transition on a three-dimensional wing.

The present development assumes that 3D transition to turbulence on swept wings can be split into streamwise and crossflow criteria that are calculated separately and subsequently recombined. Compressible linear stability theory is heavily used since it successfully predicts Tollmien-Schlichting (TS) dominated transition, and, it is believed, stationary crossflow (CF) transition on swept wings. Since the goal is to provide a transition analysis that can be used for aircraft design optimization, linear stability results are predetermined and stored as parametric fits. These fits are evaluated during the optimization to calculate instability amplification. The $e^n$ criterion is used to predict transition for both the streamwise TS calculation and the crossflow modes (but with different values of $n$).

The first consideration when picking a boundary-layer solver is that it must provide a three-dimensional solution to predict crossflow in the boundary layer. Integral solvers are not good enough since they assume the shape of the crossflow velocity profile (commonly the Mager profile) which does not provide enough degrees of freedom to approximate the actual profile, particularly when the crossflow changes direction.

In the present work, the boundary-layer properties are computed using a custom sweep/taper boundary-layer program. It is based on a combination of Bradshaw’s sweep/taper theory with the method of Kaups and Cebeci. It is a different approach to overcoming the sweep limitations of the Cebeci code than the one taken by Horton and Stock. The present method, however, also introduces three-dimensional corrections.
to the traditional sweep/taper theory (sometimes called swept-wing conical flow theory). These corrections, which have not been implemented in any of the other sweep/taper programs, are of primary importance to accurate resolution of the boundary-layer crossflow. Many more details about the custom boundary-layer code, including very favorable comparisons with three-dimensional Navier-Stokes solutions can be found in references 7, 8 and 9.

Due to the nature of sweep/taper theory, boundary-layer properties are calculated not along streamwise slices as is done with strip theory, but along circular arcs. The arcs lie perpendicular to the leading and trailing edges of the swept and tapered wing (see Figure 3) and are stacked up from root to tip to provide a boundary layer solution over the entire wing surface. These underlying arcs may be apparent in some of the plots in this paper.

The custom boundary-layer program also calculates the laminar instability growth according to a parametric model of linear stability data. The parameters used for the stationary crossflow fit are summarized in Figure 4. These parameters were suggested by Ray Dagenhart\textsuperscript{10} and used by him to successfully model growth rates calculated with an incompressible linear stability code. The present method adds the ratio of wall to boundary-layer edge temperatures to the list, but, unlike Tollmien-Schlichting instabilities, compressibility corrections to the crossflow modes are small. Crossflow n-factors are calculated by modeling the amplification rates and integrating them while simultaneously solving for the boundary layer flow.

Similarly, Tollmien-Schlichting amplification factors are modeled based on streamwise kinematic shape factor, momentum thickness and the wall-to-edge temperature ratio. The idea is borrowed from the works of Drela,\textsuperscript{11} Gleyzes\textsuperscript{12} and, to a lesser extent, Stock and Dagenhart.\textsuperscript{13} All of these previous works involve correlations of incompressible linear stability theory along with some claims that they could be applied at up to transonic speeds. The present method adds the temperature ratio as a new parameter to account for compressibility effects.

Two methods have been used to combine TS and crossflow n-factors into a three-dimensional transition criterion. In one, the crossflow and TS modes are taken as completely separate, with their own critical n-factor. This approach is least conservative and makes most sense when using linear stability theory directly. This corresponds to using the square-shaped transition curve in Figure 5.

The second, and preferred approach in the present method, is to allow for some interference between TS and crossflow modes. Arnal’s\textsuperscript{14} review of recent flight and wind tunnel data on swept wings suggests a weak TS-CF interaction: between the circular “mild interaction” and square “no interaction” curves of Figure 5. The present approach uses the circular arc. This is done not just to model the physical interaction between the instabilities, but also to account for three-dimensional effects that are neglected in the parametric fits. In particular, TS modes will see greater amplification in flows with crossflow, and, similarly, the streamwise velocity profile does influence crossflow vortex growth rates. Neither of these effects are explicitly modeled by the parametric fits because of the choice of parameters, but are qualitatively taken into account in the combined transition criterion.

II.B. Business Jet Design

Most of the aerodynamic shape design work is accomplished using the RAGE\textsuperscript{15} geometry tool with A502,\textsuperscript{16} a high-order linear panel code, or the Cart3D nonlinear Euler solver\textsuperscript{17} to compute the three-dimensional inviscid flow and the custom transition code to compute viscous drag on the wing. Gradient-free optimization algorithms are used to minimize drag. Presently a genetic algorithm and the nonlinear simplex method have been used with success. Most other optimization algorithms cannot handle the complex and sometimes non-smooth design space that occurs with free-transition drag calculations.

It has been found that once the aircraft configuration includes the basic configuration features necessary for natural laminar flow, relatively small geometry modifications can be incorporated to maximize the laminar flow extent. This observation has held constant from early round-fuselage and simple trapezoidal wing configurations\textsuperscript{9} to the current Aerion configuration. This observation, along with the properties of supersonic flow, allows some decoupling between different aspects of the aerodynamic design.

Figure 6 shows typical design variables used for viscous optimization on the Aerion airplane. A wing-body-strake geometry is used. The engine nacelles, boundary-layer diverter and empennage are left off because they have minimal effect on the pressure distribution on the wings. The goal of the optimization is to minimize drag while maintaining a fixed lift coefficient.

Airfoil sections, camber and twist are included in the optimization, as are a few variables defining the wing planform. Since it has been found that reasonably small “waviness” in the fuselage width distribution near
the wing is what it takes to achieve full-chord laminar flow, the aft fuselage is kept fixed, and therefore the lack of engine nacelles and empennage does not have a strong effect on the results of these viscous optimizations. Those items are instead included in off-line inviscid shape optimizations and during propulsion integration work. The resulting geometry updates are then routinely incorporated into the viscous optimizations.

An example of the small “waves” added to the fuselage as a result of viscous optimizations is shown in Figure 7. This is a simple configuration consisting of a round fuselage with a trapezoidal wing flying at Mach 1.5. The radius distribution of three fuselages is shown in the figure, highlighting the differences between the traditional “area-ruled” fuselage shaped for minimum wave drag compared to fuselages designed for maximizing laminar flow and minimizing total drag. The wavy radius distribution serves to modify the pressure distribution on the wing along Mach cones, effectively canceling the Mach wave that emanates from the intersection of the wing leading edge and fuselage. This shaping straightens the isobars on the wing surface, helping to reduce boundary-layer crossflow. Note that both the optimizations that maximize laminar flow and minimize total drag result in similar shape modifications near the leading edge of the wing. The maximum laminar flow optimization is more aggressive towards mid-chord region, but the wave drag penalty is too great, so the total drag optimization results in a compromise and does not attempt to achieve as much laminar flow on the wing.
Figure 8. Early difficulties with extensive laminar flow in presence of strake. Large white areas on upper and lower wing surface \((N^* > 1)\) denote turbulent flow.

Figure 9. Airfoils at the wing-strake intersection with leading-edge droop at the leading-edge crank.

It has since been found that even more laminar flow can be obtained by using non-circular fuselages, and in particular, allowing the fuselage width distributions above and below the wing to be designed independently. Since the airplane flies at an angle of attack in cruise, the effects of the wing-fuselage intersection on the wing pressure distribution is different on the upper and lower surfaces leading to different optimum fuselage shapes above and below the wing.

A couple of years ago, Aerion decided to add a strake to the business jet for reasons of high-lift, fuel volume and engine integration, among others. Figure 8 shows an early strake design along with the pressure distribution and predicted laminar instability N-factor. The N-factor used here is scaled between zero and one, where one and above (white in the figure) denotes transition and turbulent flow. The previous fuselage shaping for laminar flow proved to be insufficient on wing-body-strake configurations because the crank in the leading edge basically throws a pressure disturbance on the outboard wing, and fuselage shaping cannot cancel that disturbance. The wave is particularly visible on the lower surface pressure contours in the figure.

Numerous, and initially unsuccessful attempts were made to recover the laminar flow by shaping airfoils, drooping leading edges and smoothing the crank in the planform. Figure 9 shows some of that work, particularly the leading-edge droop present right at the leading-edge crank. The droop has been found to be beneficial to increased laminar flow extent, and can be qualitatively understood as aligning the sharp leading edge with the upwash induced by the highly swept and blunt-nosed airfoil sections of the strake.

The major breakthrough, however, came not by smoothing the crank in the leading edge, but by exaggerating it. Michael Henderson of Aerion first suggested adding a “notch” in the planform as a result of his
Figure 10. Typical result of design optimization including transition prediction.

Figure 11. Typical convergence history of a wing-body-strake design using a genetic algorithm.
experiments with A502. Subsequently, numerical optimization of the planform shape in the vicinity of the planform crank clearly supported his suggestion: a nearly 90 degree angle between the sharp leading edge of the outer wing and the blunt leading edge of the strake, along with local airfoil shape optimization cleans up the outer wing pressure distribution very nicely.

Figure 10 shows a recent wing-body-strake design, including the notch in the wing leading edge. This optimization is performed with a simple real-valued genetic algorithm due to Kroo.9 A sample convergence plot is provided in Figure 11. Due to the poor scaling of genetic algorithms with increased number of design variables, a series of optimizations is usually performed by freeing up only a subset of the design variables at a time. As the design improves and approaches the optimum, more design variables are used at once. The optimization of Figure 11 is with 24 design variables and a population size of 2304, all of which are plotted, leading to the thick band in the figure.

In addition to cycling through design variables as the sequence of optimizations progress, an N-factor margin is added and gradually increased. Without enforcing a margin, the optimizer will generally just lower the N-factors until transition is no longer predicted, and go no further. Frequently, the nearly critical N-factors occur well forward of the predicted transition location. Obviously this situation can lead to undesirable designs that are too sensitive to numerical inaccuracies, manufacturing tolerances and various operational issues. To implement a margin, optimizations are performed with a temporarily lowered critical N-factor value. Typically a 20% to 30% margin is achievable without a noticeable wave drag increase. Since the transition front normally occurs where there is a rapid rise in N-factor, adding a moderate margin has only a minimal effect on transition location, but is very effective at lowering the N-factors ahead of the
Figure 14. Result of slight tweaking of geometry using Cart3D to recover the laminar flow extent originally designed with A502.

transition front.

Note that in Figure 10, the laminar flow extent is greater on the lower surface than on the upper. The basic viscous-inviscid drag trade results in optimum designs with less than 100% laminar flow. The relative amounts of laminar flow, however, can be traded between the upper and lower surfaces without an overall drag penalty.

It was deliberately chosen to favor more laminar flow on the lower surface because the bottom of the wing is planned to be kept more aerodynamically clean. Engine nacelles overlap with the upper wing, causing shock waves and other pressure disturbances that would not allow full-chord laminar flow on the inboard upper surface. Similarly, upper surface spoilers (if used in the final design) are likely to trip the boundary layer on top of the wing. The lower surface, on the other hand, will have no protuberances. Flap and aileron hinges are planned to be faired with flexible membranes. And perhaps most importantly, the lower surface is considerably easier to clean between flights (the wing is approximately seven feet above the ramp).

Getting the optimization to favor more laminar flow on one surface is easily performed by running through early iterations with a modified objective function that weighs lower surface laminar flow more heavily than upper surface laminar flow. The optimization then doesn’t strictly reduce total drag, but the objective closely follows the total drag. After the optimization achieves more laminar flow on the lower surface than the upper, the objective function is changed back to total drag. Subsequent optimizations generally maintain the bias in lower surface laminar flow even though it is not explicitly enforced in the optimization problem. Without a partially optimized starting point biasing laminar flow to the desired wing surface, the viscous optimizations do not seem to favor one surface over the other and the relative amount of upper to lower surface laminar extent may be random.

After the gross design features are optimized using A502, the aerodynamic analysis fidelity is increased by switching the inviscid solver to Cart3D and using Euler solutions to drive the boundary-layer and transition prediction solvers. Since Euler solutions, although fast with today’s computers, are still orders of magnitude slower than A502, fewer design variables are possible with Euler-based optimization. So far, slight tweaks have been found to be necessary in order to recover the same laminar flow extent predicted by the A502-based optimal solutions after switching to Euler analysis.

Figure 12 shows a comparison between A502-based and Cart3D-based transition predictions on the same geometry. Figure 13 shows the detailed pressure distributions. They match very well, but there is a subtle increase in isobar sweep in the Cart3D solution. This increases boundary-layer crossflow enough to cause early transition. Interestingly, this difference seems to be pronounced by the presence of the strake. Comparisons between A502 and Euler-based transition analyses on previous wing-body configurations are in much better
agreement. Increasing panel density near the leading edge notch in A502 solutions does not fully resolve this discrepancy.

Subtle adjustments to the airfoil sections and the lower fuselage width distribution, using Cart3D-based analysis, recovered the laminar flow extent from the A502-based optimizations. Since Cart3D is relatively computationally intensive, the genetic algorithm could not be used. Instead, the well-known Nelder-Mead nonlinear simplex method\textsuperscript{19} is utilized.

It has been found, however, that laminar flow optimizations progress much more quickly when the extent of laminar flow is maximized instead of minimizing drag. The viscous-inviscid trade results in a much more complicated design space and the simplex method can have difficulty with it. Starting with the A502-based optimization result, then switching to Cart3D-based optimizations, but instead maximizing laminar flow until the original A502-based laminar flow extent is reached has proven to be a successful approach. To prevent an uncontrolled increase wave drag, an inviscid drag constraint can be added to the optimization. The result of such an optimization is shown in Figure 14. The N-factor margin is not as great on the outboard portions of the lower surface, where nearly white areas can be seen in the figure stretching to about 50% chord. But that is because less much margin was specified during the Cart3D-based optimizations—further optimization can undoubtedly recover the margin as well. Drag minimization using Cart3D is also achievable, and no different than using A502 from an optimization standpoint except that it is much more computationally intensive.

Future work will also include the use of Cart3D-IBL, a viscous-inviscid Euler plus integrated boundary-layer solver. At the flight Reynolds number of the business jet, the wing boundary layer is very thin, particularly the laminar portion. But the long fuselage does lead to much thicker boundary layers. The Cart3D-IBL solver will take into account displacement effects of the wing and fuselage boundary layers, and allow more accurate fuselage and strake shape design for laminar flow.

Of course, Navier-Stokes solutions can also be used to drive the transition prediction code which allows for minor fine-tuning of the high-speed lines as well as higher-fidelity linear stability and parabolized stability calculations using industry-standard solvers such as COSAL\textsuperscript{20} or LASTRAC.\textsuperscript{21}

III. Supersonic Laminar Flow Experiments

III.A. Overview

Supersonic natural laminar flow has been demonstrated in the past, although experiments are few and difficult to reproduce numerically. One very early test was performed with a German V-2 rocket shortly after the war. Another was performed by NASA in 1959 with a Lockheed F-104.\textsuperscript{22} One of the pictures is reproduced in Figure 15. That experiment achieved an impressive amount of laminar flow on an airplane not originally designed for it. In particular the wing-fuselage junction is not shaped to minimize crossflow, leading to the triangular area of laminar flow in the picture.

NASA also partnered with Aeron’s predecessor (then called ASSET) on a test using an F-15B in 1999.\textsuperscript{23} A small, approximately 3-foot span wing was mounted vertically on the F-15 center hard point, and imaged
Figure 16. Successful previous SSNLF experiment. The black sub-scale wing slung underneath the F-15 (at left) achieved full-chord laminar flow, as seen by an infrared image (at right) which compares well with numerical calculations (center).

Figure 17. Two mile test track at Sandia National Laboratories.

Figure 18. Small test article used to investigate possibility of using test track for laminar flow experiments, moving from left to right. Bright spot on leading edge is boundary-layer trip, darker triangular area is laminar flow.

with an infrared camera. Laminar flow calculations using Cart3D and the custom sweep/taper code agree favorably with the test (see Figure 16).

Both the F-104 and F-15 tests demonstrated significant amounts of natural laminar flow. Unfortunately, the chord Reynolds numbers on both of those tests is considerably smaller than will be experienced by the Aerion corporate jet. Additionally, the complex flow field around the F-15 makes numerical studies difficult, so only some of the data is useful for comparison with transition prediction algorithms.

A future high Reynolds number test is currently in planning. Since it is difficult to achieve the required Reynolds number at Mach 1.5 or 1.6 in a quiet tunnel, alternatives have been studied. One promising technique is a rocket-powered sled test. The Mach and Reynolds number requirements are not a problem, neither is the quiet free-stream flow. On the other hand, vibration levels are high due to the intermittent contact between the sliding shoes and rails. Structural vibration can excite laminar instabilities and lead to early transition just like the free-stream turbulence and noise that plague wind tunnel tests.

Due to the difficulty in dealing with the structural vibrations and remote infrared camera equipment,
a series of build-up tests have been planned. It turns out that the vibrations are strong enough that the structural integrity of the sled is in question, not just its effect on laminar flow. A picture from one of the recently completed tests is shown in Figure 18. Laminar flow was obtained on a small wedge-shaped blade at Mach 1.7 and proves that at least some laminar flow is possible in the high-vibration environment. The dark triangle on the blade, near the upper-right in the figure, is laminar flow. Transition is caused by the upper tip of the blade and by a large trip dot on the leading edge about one-quarter of the span from the bottom.

III.B. Sled Test Aerodynamic Design

After experiencing the difficulty in accurately modeling the flow under the F-15, considerable effort was expended to design a sled which is aerodynamically simple and keeps extraneous shock and expansions away from the wing. The test wing is a scaled-down version of the business jet wing mounted vertically on the sled. The body of the sled houses structural mounts and vibration isolators. All of the compact designs considered suffered from sled-body generated pressure disturbances impinging on the test wing. Particularly
difficult to avoid are reflections of the pressure disturbances on the rails and track bed. The track is shown in Figure 17.

In considering many candidate designs and computing the flow around most of them with Cart3D, a monorail design with a long forebody was identified as the most promising configuration (see Figure 19). The forebody is analogous to a fuselage, permitting the same local shaping for laminar flow that is used in the airplane design. It also houses the structure and holds onto the rail with steel “shoes.” The forebody is long so that reflections of the forebody shock miss the test wing completely. Aerodynamically the flow on the wing is very close to that with an infinitely-long fairing and a simple ground plane replacing the track. Although a more accurate representation with the finite fairing and rails and track bed also possible to compute when necessary. One such computation is shown in Figure 20.

In order to be able to reuse the test wing for other purposes, for instance in a wind-tunnel test using a flat ground plane instead of the fairing, it was decided to keep simple biconvex airfoils on the wing and to restrict all of the laminar flow specific shaping to the forebody. This prevents the situation where the wing is specifically tuned to work only with the sled fairing attached.

Due to the large number of cells required to model the full sled configuration, a simplified geometry was used for the optimizations. Early optimizations were performed on an infinitely long fairing, wing and ground plane. Parts of the track bed and rails were later added to the optimization geometry since they contribute to slight but noticeable effects on the boundary-layer instability N-factors.

Cart3D flow solutions and the custom sweep/taper code were used just as in the airplane optimization. Since drag is not a consideration for this test, all optimizations maximized laminar flow extent with a generous margin on N-factor. Conversely, N-factor could have been minimized, which is a reasonable objective, however the real goal is to push larger N-factors towards the trailing edge. The approach chosen for the present work is to maximize the extent of laminar flow with increasingly conservative transition criterion applied during a series of optimizations.

Figures 21 and 22 show the predicted laminar flow on the baseline, unshaped fairing and a series of optimization results leading to the final fairing shape. One consequence of the rocket-sled platform is that the test Mach number is not precisely known ahead of time. A design that works for a range of Mach numbers is therefore desired. The simple technique of using an objective function which is a linear combination of the laminar flow extent at two different Mach numbers does the trick.

It was found that the optimization at two Mach numbers actually allows more design variables to be used since there is less of a concern that the optimizer will create a point design by taking advantage of some fortuitous wave cancellation that only occurs at one Mach number. The final result at the bottom of Figure 22 shows very small N-factors between Mach 1.5 to 1.6 on the entire surface of the test wing, with full-chord laminar flow predicted between Mach 1.4 and 1.7.

Although a similar two-point optimization has not been performed on the airplane design, it is desirable in the future. Thus far, the airplane has exhibited adequate off-design laminar flow extent, however that cannot be guaranteed without a multi-point optimization approach. The discovery that it actually permits the use of more design variables and therefore finer control over the shape design makes this approach very attractive. The only drawback is that it is very computationally expensive.

Other aerodynamic work in support of the sled test has been performed and is not discussed here in detail. The list includes linear stability calculations using LASTRAC to provide criteria for the vibration isolation system (Figure 23), the computation of laminar instabilities with cold-wall boundary conditions (since the test is very quick making adiabatic conditions difficult to achieve), aerodynamic loads for structural design and drag prediction of the entire sled train including the “utility” sled that houses the rockets (Figure 24).

IV. Conclusion

Over the last three years considerable advances have been made in the design for supersonic natural laminar flow at Aerion and at Desktop Aeronautics. As the Aerion airplane configuration has become increasingly detailed and refined, the design for laminar flow has also evolved. Shaping for laminar flow has proved to be robust to basic design trades such as high-lift considerations and stability and control issues while still providing a strong performance benefit. The process has uncovered basic shaping characteristics and important design variables that encourage extensive laminar flow with minimal impact to wave drag.

The design of a high Reynolds number laminar flow experiment has also contributed to improving the airplane design process, particularly with the use of high-fidelity Euler solutions and multi-point optimiza-
Figure 21. Laminar flow extent on unoptimized monorail sled.

Figure 22. Laminar flow instability predictions on the rocket sled at various Mach numbers and for various levels of design optimization.
Figure 23. LASTRAC was used to compute both T-S and crossflow instabilities on simplified geometries.

Figure 24. Pusher sled position evaluated for effect of bow-shock structure on test wing and total sled train drag.
tion techniques. The test will hopefully also contribute by confirming the fundamental supersonic natural laminar flow technology and provide data to help refine the transition prediction algorithms used in design optimization.

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