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Arild Bertelrud
Analytical Services & Materials/NASA Langley Research Center
Hampton, VA

Paul Kolodziej
NASA Ames Research Center, Moffett Field, Mountain View, CA

Greg K. Noffz
NASA Ames/Dryden Flight Research Facility, Edwards, CA

Afzal Godil
High Technology Corporation/NASA Langley Research Center
Hampton, VA

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PLANS FOR IN-FLIGHT MEASUREMENT OF HYPERSONIC CROSSFLOW TRANSITION ON THE PEGASUS® LAUNCH VEHICLE

Arild Bertelrud*, Paul Kolodziej**,
Greg K. Noffz† and Afzal Godil‡

Abstract

A flight measurements program is underway to obtain flight data for validating stability theory-based transition estimation techniques for crossflow-induced boundary-layer transition at Mach 6-8. A smooth wing glove, designed to enhance crossflow transition and suppress Tollmien-Schlichting (T-S) instabilities, will be mounted on the delta wing of the first stage of the Pegasus® booster. Transition behavior will be documented during the latter part of the first stage burn, using recently developed high-frequency response instrumentation in addition to traditional techniques. The instrumentation package will include "intelligent" software capable of yielding space and time-correlations obtained through extensive use of fast Fourier transforms (FFTs), "windowing" and multi-stage compression as well as a variety of techniques for quality assurance. The flight experiment covers a Mach number range (6 to 8) where low background disturbance data cannot currently be obtained in ground facilities. These flight results will support transition estimation code development for application to supersonic laminar flow control (LFC) and National Aero-Space Plane (NASP) class vehicles in areas where crossflow, rather than T-S waves, is the dominant instability mode.

* Analytical Services & Materials/NASA Langley, Hampton, VA 23665

** NASA Ames/Moffett, Mountain View, CA 94035

† NASA Ames/Dryden, Edwards, CA 93523

‡ High Technology Corporation/NASA Langley, Hampton, VA 23665

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Introduction

Renewed interest in airbreathing flight at hypersonic speeds, (e.g., NASP), has prompted research to improve the understanding of aerodynamic and aerothermodynamic flow phenomena over hypersonic vehicles[1]. Among the central issues in this work is the fundamental understanding of and ability to predict transition for high-speed flows[2]. The area of transition from laminar to turbulent flow is of both fundamental and practical importance, since the onset of transition increases both viscous drag and surface heating, and dictates the forebody boundary-layer thickness entering the inlet of NASP-class vehicles.

A flight experiment is in preparation to characterize crossflow transition under hypersonic Mach numbers and conditions of interest for the development of NASP single-stage-to-orbit launch as well as hypersonic cruise vehicles. The experiment will be used to support the code validation efforts currently underway regarding stability codes at high Mach numbers. The current experiment is to be performed in flight rather than in a wind tunnel to avoid the effects of sound radiation from the turbulent nozzle-wall boundary layers and facility-dependent free-stream turbulence fields[3,4].

The flight experiment will be performed on the delta wing of the first stage of the Pegasus launch vehicle in a Mach number range from 6 to 8 and at altitudes ranging from 130,000 to 200,000 ft, as shown in Figure 1. A B-52 is used to carry the launch vehicle to 40,000 ft, altitude where it is detached from the airplane. The first stage booster propels the vehicle to approximately Mach 8 at 200,000 ft, where

approximately Mach 8 at 200,000 ft, where staging occurs. The experiment which is performed as a piggyback to the payload launch, is designed to have a negligible influence on the primary launch goal. This paper describes the general problems associated with the flight transition experiment, the approach to handling several of the issues and some of the peculiarities of this particular experiment.

In a code validation experiment, it is necessary to have a clear definition of the amount and type of information required. In particular, transition may be instigated by a variety of disturbances and developed through several different instability modes. A fundamental issue is whether it is sufficient to know the location of transition or if supplemental information is required to allow a more direct comparison with parameters modeled in the codes. As illustrated in Figure 2 (from Ref. 5), regarding only the beginning and the end of transition, various types of instrumentation may yield different results, simply because of the nature of the physical parameters to which the sensor responds, and whether it is used to obtain surface or flow-field quantities. There may be a practical limit to the frequency response and/or operational temperature of the instrumentation, and there is often a limitation on the data transfer bandwidth. The issue then becomes whether data of a certain amount and quality are sufficient for validation purposes.

The total data gathering and transmission time in each flight is 20 seconds, since the first stage is non-recoverable. This constraint poses unique problems in dealing with the sensors and their data handling. Separate solutions have not been made to all the instrumentation issues, but the various data obtained has been split into several different categories. To limit the changes to the vehicle, it was decided to conduct the experiment utilizing the original wing. A part-span glove will be mounted in the forward mid-span region and faired onto this wing.

The project is being carried out in close cooperation among three NASA centers:

Langley:	Aerodynamic experiment design, instrumentation definition and prototype development of transition-particular sensors and post-flight data reduction and comparison with theory.
Moffett:	Glove design and manufacture, ground testing of materials, structural integrity and thermal protection.
Dryden:	Overall project responsibility and operations/scheduling, aerodynamic instrumentation and development of flightworthy equipment.

Three tests are to be performed piggy back on three separate flights with the first launch planned for the Spring of 1993. The primary purposes of the various flights are:

Flight 1 (FX-0)	Proof-of-concept regarding the glove to answer the question, "Can transition data be obtained and documented?"
Flight 2 (FX-1)	Advanced instrumentation verification
Flight 3 (FX-2)	Transition physics measurements

As will be discussed in detail, it is necessary to adopt a flexible approach in carrying out this type of experiment in a piggyback fashion. This approach which has several advantages over flight research programs consisting of a single flight, which is analogous to the progressive development of a research program in a wind tunnel. By proceeding in a stepwise manner, initial experiments can be used to provide the experience and the basis for more complex experiments. Measurements can also be repeated on several flights to establish a high degree of confidence in the results. The risk and cost of developing new instrumentation can be reduced by utilizing early flights to test prototypes of sensors required for later flights.

Although the current launch schedule for the Pegasus vehicle contains more than half a dozen flights per year, (the development of

design, hardware, instrumentation and analysis tools has to be aimed at only one or two realistic opportunities per year.) A go/no-go condition will exist depending on the following:

- Available payload margin. As scheduling unfolds, the margin may exist, disappear or become available on particular launches. The stage 1 - payload ratio is on the order of 1:20; i.e., a 300 pound experiment (glove and instrumentation) on the first stage would require a 15 pound (orbital) payload margin.
- Suitable trajectory. This is often well defined at an early stage, since it mainly depends on the desired payload injection altitude. It is obtained (most often) through variations in the magnitude and duration of angle of attack during the first portion of stage 1 burn, and it is, therefore, possible to work with a band of allowable altitude-Mach-time variations.
- Carrier airplane and launch location. To date a B-52 flying from Edwards, California, has been used, but future plans include flights from Cape Canaveral, Florida, or Wallops, Virginia, and the use of an L-1011 with in-flight refueling capability. These changes may cause problems since the Pegasus vehicle is assembled and mated with the carrier aircraft at NASA Dryden in California. The transcontinental flight requires transportable checkout capability and perhaps some replacement of sensors or modules.
- Since the experiment is performed in a piggyback mode (i.e., it will not be possible to postpone or change a launch due to problems experienced in connection with the crossflow transition experiment), a sufficient degree of autonomy and redundancy must be present to ensure a successful experiment. Delays may also occur due to payload rescheduling or changes; therefore it is necessary to have options for required checks and changes up to launch.

One feature that makes the experiment feasible, is the mating of the different stages, payload and wings occurs within a couple of weeks before launch. Thus, the experiment can be prepared for one set of wings, which will be used only on a launch with suitable launch/trajectory characteristics.

Geometry and Trajectory

The geometry of the baseline Pegasus flight vehicle is shown in Figure 3. The vehicle consists of a blunt-nosed cylindrical body 1.27 m (50") in diameter and 15.24 m (50 feet) in length, with a delta wing mounted on top. The vehicle is made from graphite composites and has an ablative thermal protection system.

An account of the vehicle, its design and the results from the first flight can be found in Refs. 6-8. Because both local Reynolds number and temperature history influence the transition location, the design of the glove shape for the transition experiment is very dependent on trajectory. Figure 4 shows the ranges of altitude, Mach number and angle of attack for typical trajectories that have been considered. Data obtained above Mach 5 are of primary interest. The nominal design for the transition experiment includes an angle of attack as close to zero as possible. The innocent-appearing trajectory variations, however, have a profound effect on the local flow conditions and the tendency to transition since the Reynolds number varies almost an order of magnitude between the highest and lowest trajectories.

Glove Shape

This paper will give only a brief account of the aerodynamic design process. Further details are available in Ref. 9. The wing glove was designed utilizing a Navier-Stokes code[10] in conjunction with a linear stability code. Since the use of Navier-Stokes codes and linear stability analysis is time consuming, a preliminary design used simplified flow analysis and the MARIA code[11] for crossflow stability estimates. Details of transition predictions in general and of supersonic and hypersonic flow, can be found in Refs. 12–14.

The glove was designed to obtain crossflow-dominated transition on the gloved portion of the wing at 20–40% of the chord at a flight Mach of 6. The glove shape was designed to meet the following objectives:

1) Flow at the attachment line must be laminar (See Ref. 15 for criteria).

2) The crossflow instability must be maximized via increases in the chordwise pressure gradient along the chord.
3) Second-mode and T-S instability must be attenuated via favorable chordwise stream-wise pressure gradient over the entire glove.

During the design process, the sensitivity of crossflow transition to a large number of parameters was investigated, including trajectory variations, allowable wing thickness, leading-edge radius and surface temperature. In particular, the trajectory variations described in the previous section posed a major problem. Since the Reynolds number is strongly dependent upon the local altitude, the measurement window at Mach=6 may vary almost an order of magnitude in Reynolds number. In addition, the trajectory influences the wall temperature level and distribution. For example, the attachment line transition will cause turbulent heating on the entire glove up to a Mach number which is trajectory dependent. From there on there will be laminar heating in the leading edge region but turbulent further back. The transition front, which in itself is a function of the temperature, will gradually move back during the measurement window, until it is off the glove at or before the vehicle reaches Mach 8. Therefore, the occurrence of transition over a particular sensor location will be highly dependent on the trajectory flown. To evaluate and bound these effects, the mean flow computations and the associated stability computations have been performed for a window of M=6 to 8, wall temperature ratios T_w/T_{aw}=0.4 to 1 and angle of attack 0 to 1 degree.

Several geometries were examined to determine a shape which yielded the highest probability of success. Ellipses and sub- and super-ellipses were examined, along with different combinations of maximum glove thickness and leading edge radius. The leading edge radius had a little influence on the final amplification; it was more important to retain the acceleration over a long distance. Thus, it was determined that biconvex shapes yielded the best compromise. The maximum thickness chosen represents a compromise between the aerodynamic stability, the need for space for instrumentation between the glove surface and the basic wing, and a reasonable rear fairing.

Since the glove skin is shaped through bending around the leading edge, a coordinate system normal to the leading edge was used for glove definition. XN is the coordinate in the wing plane normal to the leading edge of the basic vehicle (with zero at the leading edge), and Z is the ordinate normal to the wing plane. The biconvex glove shape is described by two circles: a 1" leading edge radius with the center at XN= -1 in., Z=0, and of 163.184 in. radius, with the center located at XN=33 in., Z=-157.934 in. The two circles are joined by a Bézier curve from 60 degrees angle on the leading edge circle to XN=6 in., with continuous ordinates and first derivatives at both ends and continuous second derivative at the front junction point. To provide a smooth transition, or ramp, from the basic wing to the glove measurement surface a fairing is attached along the inboard, outboard and trailing edges. The most critical fairing, on the inboard edge, is in the shape of a hyperbolic tangent with the maximum slope at the intersection with the basic wing. This intersection line follows the inviscid streamline. On the outboard and rear edges, where the shape is not as critical, there is a simple linear blend with the wing.

Due to time and space constraints upon the design process, the shape was carried through the entire span and faired to the rear in the computational analysis. Figure 5 shows the current glove shape, and Figure 6 provides estimates of the flow characteristics around the vehicle, including the wing. The (hypersonic) flow is characterized by two strong shocks: a bow shock, wrapping around the vehicle in an axisymmetric fashion and a wing shock. The wing tips are always inside the bow shock, an issue that has been studied extensively in both computations and wind-tunnel tests of the vehicle, since tip impingement of the bow shock would have a tremendous influence on a sensitive experiment such as transition. The

computations shown here have been verified in wind tunnel tests[16] on a 10 inch model in a Mach 6 wind tunnel which yielded Schlieren photographs and surface oil flow patterns. The same model was used in another Mach 6 facility at NASA LaRC to obtain surface pressure distributions using pressure-sensitive paint[17,18]. The second shock is a wing shock wrapping around the leading edge of the wing. This can be seen clearly in Figure 7, along with its interaction with the bow shock. The free-stream disturbance fields reaching the outer part of the boundary layer will have been processed through two shocks and probably will have changed character from the undisturbed free-stream conditions[19].

Much weaker shocks are caused by the irregularities, fillets and protuberances already existing on the Pegasus vehicle. Installing the glove on the wing, in fact, adds another protuberance into the flow. A primary goal of the design is to minimize the impingement of any shocks on the glove test surface. In cases where the existing protuberances cannot be removed (e.g., the bolt heads at the stage separation ring), an analysis was made to determine the shock envelope and whether or not it intersects the glove test surface.

Transition experiments differ from other flow measurements on vehicles due to the large number of possible causes and paths leading to transition. A proper validation experiment should ensure that the desired transition behavior is the one which actually occurs. For example, the fillet area (Figure 8) will produce a shock. Where this shock intersects the wing shock, a free shear layer might apear that may cause transition either within itself or upon impingement/interaction with the test surface. If turbulent flow occurs on the surface, it will spread at an angle which depends mainly on the local free-stream Mach number[20]. Fortunately this angle decreases as the Mach number increases, but this flow is another serious concern.

After completion of the aerodynamic design of a glove with the presumed transitional characteristics, the shape was used by Nielsen Engineering and Research to determine whether the glove design was "benign" in terms of vehicle aerodynamics[21]. The results indicate that the basic design is sound, but that the added weight, in combination with possible lift reversals through the transonic/supersonic regime, causes some concern. If necessary, this problem can be handled either through minor glove modifications, the addition of a second glove on the left wing or increased control power.

Glove Manufacture

The basic Pegasus wing is a composite structure that is built by laminating a graphite fiber/epoxy skin over a foam core. Instrumentation is difficult to install directly in the basic wing because penetrations in the load-bearing graphite skin weaken the structure. A cost effective alternative to redesigning the basic wing for the crossflow transition experiment is to build an instrumented platform or glove and install it over the existing wing. The glove must provide a smooth, stable, non-ablating measurement surface capable of withstanding aerothermal heating to Mach 8.

The biconvex glove shape described earlier is built from conventional fiberglass/foam-core technology that is frequently used at Ames to modify existing aircraft structures. Figure 9 illustrates the general construction of the glove. First, a fiberglass cloth is laid up over the basic wing, impregnated with epoxy resin and cured to form a wing splash. Then, blocks of polyurethane foam are bonded in place on the splash and shaped to the required geometry. At the leading edge where the aerothermal heating is high, a fibrous silica ceramic known as FRCI-20 is used instead of the foam. The structure is finished by adding a second skin of fiberglass/epoxy to the top of the foam, closing the edges out, and cutting channels for instrumentation. Finally, a 0.0625 in. thick copper (101) skin containing a variety of sensors is bonded over the structure. The high thermal conductivity of the Copper minimizes the temperature gradient at the leading edge

and, consequently, the skin deformation due to thermal stresses.

After the glove is fabricated, a second set of foam blocks are bonded in place on the outboard and rear edges of the splash. These blocks are shaped to a simple linear ramp or fairing down to the basic wing. A second fiberglass skin is added to the top of the foam, which is then coated with a low temperature ablator. On the inboard edge of the splash a second set of FRCI-20 blocks is bonded in place. These blocks are shaped to the more complex hyperbolic geometry described earlier. After shaping, the blocks are removed and impregnated with a shatter-resistant coating known as TUFI. The coated blocks are reinstalled to form a smooth, non-ablating ceramic fairing along the inboard edge of the glove.

A small section of the composite wing glove was tested in the Ames Aerodynamic Heating Facility (AHF)[22]. This facility is a constricted arcjet capable of generating the high enthalpy flow and aerothermal heating experienced during hypersonic flight. For this test, argon was used instead of air to duplicate the relatively low heat fluxes of first stage ascent. In flight, the first stage of Pegasus and the glove will experience a parabolically shaped aerothermal heat pulse lasting 80 seconds, with a maximum laminar heat flux of 180 kW/m^2. This heat pulse is difficult to duplicate in an arcjet. Instead, a square pulse is typically used, one which has the same integrated heat load as the flight aerothermal heating, but a lower maximum. At the operating conditions set for this test, stagnation conditions on a 0.0102m (4 in) diameter hemisphere were a heat flux of 238 kW/m^2 and a pressure of 670 Pa. These conditions correspond to a stagnation enthalpy of 4.2 J/Kg. Free-stream conditions included a Mach number of 7.8, and a unit Reynolds number of 38,000/m.

Although the arcjet free-stream conditions do not duplicate flight conditions, the surface heat flux distribution on the glove skin is duplicated. Maximum heating occurs on the stagnation line of the glove leading edge at zero degrees angle of attack. Behind the stagnation line, the heat flux rapidly decreases as the glove skin bends around the wing and becomes almost parallel to the free-stream. Glove skin temperatures measured in the arcjet flow reflect this behavior. For example, after 80 seconds of exposure, the stagnation temperature of the copper skin was 885 K. At a streamwise distance of 2 in, or about 90 degrees from the stagnation point, the copper skin temperature was at 818 K. Further back, at a streamwise distance of 14 in., the copper skin was at 433 K and the temperature gradient was almost zero. At the same conditions, a stainless steel skin had much higher leading-edge temperatures and gradients.

In order to gain experience, refine fabrication techniques and develop assembly procedures, a 12 in. wide prototype of the FX-0 glove is currently under construction. This prototype incorporates several improvements that focus on reducing the leading-edge temperature, improving the biconvex shape definition, demonstrating instrumentation installation and minimizing vibration of the copper skin. By increasing the copper skin thickness (adding more mass), the leading-edge temperature and gradient can be decreased. For the prototype glove currently under construction, a 0.125 in thick copper skin was chemically milled down to 0.050 in. everywhere except for the leading-edge region. Chemically milling the copper skin in this manner can reduce the leading-edge temperature without increasing the weight of the skin excessively. Thermocouples and static pressure taps have been mechanically attached to this skin because it is difficult to solder or weld without localized annealing and distortion. A shaker test of the prototype is planned to evaluate the vibrational characteristics of the integrated structure.

The experiment imposes several glove requirements :

- The surface must be smooth, with a local Roughness Reynolds number $Rk<10$ (Ref. 23). At the altitudes in question, the Reynolds numbers are fairly moderate, so the standard smoothness of a rolled sheet of metal meets this criterion.
- The waviness must be very small, in the order 0.0002 (Ref. 24). This requirement calls for a material that is fairly easy to bend. For

the present approach, copper was chosen over stainless steel, although the margin to annealing is very small at the M=8 trajectory point. This margin necessitates an arcjet test, to be described below.

- The glove must have the strength to take the loads as the vehicle accelerates to supersonic speeds at 20 degrees angle of attack after launch. This requirement also necessitates both arcjet testing for bonding at elevated temperatures and a bending test of a mockup to establish a valid manufacturing procedure for the estimated stresses. As the glove heats up during the flight, its tension and internal stresses will change. To estimate these changes, shaker experiments will be combined with analytical and computational modes, as exemplified in Ref. 25. The rocket engine will burn during the entire measurement period and will cause vibrations transmitted through the structure of the vehicle. Therefore it is necessary to design the glove so that the vibrations are properly damped by the glove structure and attachment technique.
- Techniques had to be found to allow the mounting of pressure taps and other sensors to the skin without causing local disturbances. The delay time in the pressure system becomes an increasingly critical issue as the altitude increases, and the standard techniques in use at NASA Dryden to compute this have been utilized in the present experiment[26].
- The fairing will have turbulent flow early in the trajectory. Additional requirements include ease of fabrication into a three-dimensional shape and flexibility to allow for wing movement (obtained through segmentation).

The glove material chosen was copper (101) without any protective layer. (In previous tests, it appears that transition may have been caused by static discharge through charge buildup on non-conducting protective surfaces). To minimize the risk of attachment line contamination from the fuselage boundary layer during the measurement period, it was necessary to minimize the leading edge radius. While the actual analysis included separate studies of the local flow properties (Ref.9), a rule of thumb indicates that, when D=leading edge diameter, ReD<80,000 should ensure laminar flow. Therefore it was decided to use a 1" leading edge radius for the glove. An initial torch test with 1/16 in. thick copper, which simulated the heat fluxes to the glove caused annealing of the copper in the leading-edge region, since temperatures went above 1100 degrees F. In subsequent tests, the copper thickness was increased to 1/8 in. in the leading-edge region and gradually reduced to 1/16 in. further back through chemical etching. Up to this point in the testing, stainless steel had been considered an alternative for the skin material, but the increased leading edge thickness lowered the temperature sufficiently to avoid the annealing problems.

Thus far, engineering methods have been used for the aerothermal heating calculations[27,28]. Assuming laminar heat transfer in the leading-edge region, the maximum heat flux should be approximately 180 kW/m^2, occurring roughly 60 seconds into the flight, i.e. roughly the same time the measurement window is reached. If the leading edge is turbulent, the heating will be greater. In the case of this event, an existing formula has been used to ensure glove survivability up to first stage separation.

Instrumentation

A variety of sensors exists for the measurement of flow properties and transition in flight, including basic low-frequency and high-frequency techniques[29,30]. A main guideline for the project is to use proven sensors that require only a moderate element of verification and adaptation. This experiment does not call for novel techniques, although integrating the techniques into a transition characterization package may be new. FX-0 will be conducted without adaptation, this is the only flight for which firm decisions have been made on sensor choice and configuration. In previous flight tests, a moderate amount of instrumentation has been sufficient to determine the transition location itself[31], but when extraneous parameters or effects such as angle of attack or rocket burn influences are present, traditional methods do not provide enough data for drawing

conclusions about transition behavior[32].

The First Flight

In the first flight, denoted FX-0, only a limited subset of the sensor types under consideration will be included. This section presents the sensor issues for this flight.

Instrumentation Layout

When laying out the instrumentation on the surface of the glove, it is necessary to recognize that any piece of instrumentation, be it a static pressure tap, a hot film sensor or any other sensor, may cause transition. Therefore instrumentation sites are positioned as shown in Figure 10, using the turbulent spreading angles given by the local free-stream Mach number. The boundary-layer rake, Preston tubes and hot films are located far back on the glove to yield information on the real boundary-layer thickness at the 'least disturbed' spanwise station. The hot films which are used for flow direction measurements, are located far back due to their difficulties in sustaining the high leading -edge temperatures. All data in the FX-0 are routed through the PCM system of the vehicle and transmitted in real time to ground stations.

Instrumentation Installation

The primary technique for measuring transition passage over each particular sensor is to look at the time history of temperature. This was done in the Reentry F experiment (Refs. 31 and 33), based on a change in the dT/dt slope. Measurements of local temperature through the use of thermocouples very close underneath the surface, plus knowledge of the heat conduction characteristics of copper, allow an estimate of both true surface temperature and heat transfer. From this data, the transition location can be determined as it passes across the thermocouple. The thermocouples will be spot-welded, chromel-alumel thermocouple junctions. Each wire is insulated with shrink tubing, a hole is drilled at the skin location and the thermocouple is peened in place. The static pressure taps are manufactured by machining off the head from 4–40 stainless steel screws with a 1/16" hole placed in the center of each screw shank. The 1/16" outer diameter stainless steel tubing is then annealed to the screw. A suitable hole is drilled and tapped at the skin location, and the tubing is threaded through the hole.

Because the purpose of the first flight is to verify the glove concept, it is essential to verify the glove integrity, i.e., structural response to vibrations, wing deformations, thermal stresses etc., and their effect upon the transition. Accelerometers will be mounted on the wing on and under the glove at several locations to allow a determination of glove surface vibration frequencies and wavelengths. From the boundary-layer stability computations, it is predicted that frequencies in the order of 3.5 up to 10 kHz are most amplified, depending on the point in trajectory and location along the glove. Also, while traveling waves are predicted to have maximum amplification, the boundary layer is unstable all the way down to stationary waves. At 1/4 the maximum amplified frequency, the N-factor is still of the order 5 and might under some circumstances, cause premature transition given sufficient initial disturbance amplitude.

The "probeless" rake design, based on work by Keener and Hopkins[34], consists of a pair of wedges with total pressure ports drilled in the face of the blunt leading edges. The lack of protruding probes reduces the risk of damage due to aerodynamic heating[35]. It is especially important to have confidence in the integrity of the rake since it cannot be examined post-flight. Although the measurement window of the transition experiment encompasses the last 20 seconds of the flight, the rake will attempt to characterize the boundary layer (i.e., determine the momentum thickness, etc.) for the entire first stage flight. This requires a close grouping of total pressure ports near the surface to define the "knee" in the integrand of momentum thickness. Attaining a close grouping while keeping the port sides from becoming unreasonably small (due to lag and fabrication concerns) requires one of two

alternatives. The rake can be canted at some angle less than 90 degrees to the surface so that there is more space between the ports for the same vertical spacing or the rake can be designed with two vertical sections and the ports distributed to avoid interference.The latter approach was chosen to ensure a symmetrical flow field about all the ports. The concept of positioning the leading edge of the base plate at the trailing edge of the vertical wedges was retained from the Keener/Hopkins design to minimize baseplate interference on the lower ports.

The Preston tubes[36,37], which traditionally are used to determine the local skin friction coefficient , are utilized here in a "semi-quantitative" fashion. The total pressure is assumed to change drastically as the transition moves past the Preston tube in the chordwise direction. However, it is important to look at the sensor output as a function of both time and spanwise position. The spanwise data should yield information on turbulent wedges occurring along the span of the model, as shown in Figure 10. If most of the Preston tubes indicate a laminar flow, they will show a fairly low total pressure. Increasing pressure would indicate that the flow is starting to undergo transition since the turbulent layer will have a larger time-averaged slope of the velocity profile. However, if the Preston tubes are far back from where transition starts, the boundary layer thickness will have grown so that the local shear is reduced. The boundary-layer rake is useful as a "sanity check".

An additional important piece of information is the streamwise extent of the transition region. As a rule of thumb, the end of transition is roughly 1.6 times the streamwise distance of the start[38]. This measurement roughly corresponds to the Preston tube indication of increased wall shear/total pressure. Different transition detection techniques yield different indications of the length of the transition region as well as its onset[5]. Another important consideration is the wall temperature conditions expected during the trajectory. As mentioned previously, the initial assumption has been to determine the gross effects on transition through computations ranging from $T_w/T_{aw}=0.4-1$. The stability calculations indicate that the main effect of the temperature ratio to change the effective boundary-layer thickness; the boundary-layer thickness is proportional to the square-root of the temperature ratio. Since the existing temperature at any location will affect the stability characteristics of the profile in terms of local amplification, it is necessary to couple the pressure information with the wall temperature measurements.

Another "sanity check" of the pressure and heat transfer measurements for transition detection and predicted surface flow directions, is the use of dual hot film sensors. The technique consists of heating a thin film to a constant temperature above the wall temperature[39] and monitoring the forced heat loss as a function of wall shear. McCroskey[40] developed a dual gauge capable of yielding flow direction as well as magnitude. The analysis technique developed in Ref. 41 is used in this case. Traditionally, single hot films are used to yield qualitative transition information through AC-coupling of the output. It is not uncommon for experimenters to use the frequency spectra to assign,T-S frequencies for example, to the results. We attempt to avoid this, since the hot film sensors traditionally have a highly non-linear sensitivity with absolute level of the heat transfer. Also, the transfer function is highly non-linear in terms of the frequency output. The use of two films perpendicular to each other generally provides accurate information on flow direction. In this case, the information is used to determine whether the Preston tube output is due to flow essentially in the direction of the sensor or if a large crossflow is present.

In addition to the traditional heat transfer gauges which have a fairly low frequency response, miniature ones, made using vacuum sputtering techniques[42], are under consideration. In the latter case, the frequency response is very high, and the output requires high amplification. The miniature gauge may be a candidate for flights FX-1 and FX-2 after proper ground evaluation and a possible feasibility test during FX-0. As mentioned

earlier, it is essential to document both the time-averaged properties of the atmosphere[43] and the disturbance field around the vehicle[44]. This may be done through measurements on the vehicle (the use of high-temperature pressure probes) or off-vehicle (the use of sounding rockets). Some measurements of this type are required during the first flight, but the bulk of the data is to be obtained during the later flights.

Ground Testing for Later Flights

A large number of limited tests are being performed to allow proper interpretation of the flight data. As mentioned previously several experiments were conducted just to develop the proper techniques for manufacturing the glove.

Another set of experiments is required to ensure the proper handling of the signals from the high-frequency instrumentation planned for FX-1 and FX-2. These include the development of a transition characterization package (TCR) consisting of sensors for measuring the physical features of the flow that characterizes the transition process and a digital signal processing system for processing the raw data, compressing it and selecting portions of it to be transmitted through the antenna systems of the Pegasus®.

Sensors

Boundary Layer Acoustic Monitors (BLAM) are active piezoelectric sensors commonly used by the Air Force for transition monitoring on reentry vehicles. Although they are capable of extremely high frequency response, the present application requires adaptations of the sensors in at least two respects:

- Traditionally, BLAMs are used with a non-metallic surface. Because the proposed copper has a different acoustic response it is necessary to establish the proper transfer functions.
- Normally the gauges are mounted underneath ablative surfaces; i.e., they are thermally protected from the temperature environment on the skin. In the present experiment, it is absolutely necessary to avoid ablative products and to work with internal thermal protection[45,46].

The current project calls for the following steps. The salient characteristics of the current gauges must be determined from initial tests in a turbulent boundary layer. The pressure fluctuation spectra are known in the literature and from separate measurements with pressure transducers. The heat and skin problems are then addressed, and the resulting gauge is subjected to the same turbulent boundary layer. When this information and the DSP system are available, the gauges will be put into a hypersonic, transitional flow to determine the response in an intermittently, transitional flow. As a last test a gauge will be mounted in the Ames Arcjet facility and tested under flight conditions.

Fast Pressure Transducers

To document the physics of the transition process as completely as possible, it is necessary to obtain space-time correlations of the fluctuations as they appear. A description of the type of signals one may obtain in the nonlinear, intermittent stages of transition can be found in Ref. 47, where the time-signatures of sensors in the presence of turbulent spots have been illustrated. As Figure 11 shows, an array of three pressure taps will be used to determine the local fluctuations and their characteristics at one particular location. Through cross-correlations between the sensors, it is possible to determine both the frequency and the wavelength and direction of the waves. Since the maximum amplified frequencies are expected to be of the order 5 kHz, it is possible to use ported fast pressure transducers. In the ground testing and for the interface with the DSP system, such an array will be included in the turbulent experiment, both in cold tunnels and in the Arcjet.

The types of fast pressure transducers that are available do not generally accept the high temperatures on the Pegasus' wing skin. Therefore it is necessary to reduce the temperature at the diaphragm. This might be done using an ablative coating. Since an ablative coating certainly will affect the frequency response of the sensor, it is necessary to do a thermal analysis of the required ablative thickness, then compare the transfer function for one traditional, uncoated

sensor and one coated with the ablative. If a fairly uniform damping can be obtained, a sensor must be put through the heating environment to determine response while the gauge coating ablates away. In flight FX-1 (at least), it will be necessary to use 4 sensors, 3 having an ablative coating and the fourth unprotected. The fourth will probably not survive up through Mach 6, but it can be used to check the performance of the other three as long as it survives.

Transition Characterization Package

A host of separate issues concerning the ruggedization of equipment and adaptation to the PCM stream used to handle all aero data on the Pegasus® are not yet well defined. They are highly dependent on the type and the rate of data the DSP system will transfer real-time. In the initial stages, each sensor is taken separately and characterized as to its output, both in DC and AC form, as it is subjected to a turbulent and later laminar/transitional boundary layer. From this type of information, it is possible to set up simulation models for the sensors and sensor packages. The models can be used as guideline to develop the DSP system and to enable a fast check of the implications of changing trajectories. Since the experiment is of the piggyback type, it must be anticipated that a go/no-go condition has to be considered for a variety of options.

During the FX-1 flight, A Sensor Evaluation Module (SEM) will be flown to evaluate the performance of different types of sensors during actual flight conditions and to gather valuable data for the physical interpretation of the transition process. Figure 12 shows a block-diagram of the sensor package up to the interface with the DSP system. In ground development of the sensor package together with the DSP, a flat copper surface having the appropriate thermal protection system will be investigated with the prototype of the DSP system.

Initially, artificial transitional signals are used, since the intermittent nature of the outer part of turbulent boundary layers exhibits similar characteristics with regard to signal characteristics. A large portion of the DSP analysis concerns the handling of the raw data in blocks of 1024 samples and performing FFTs. As a sanity check, there will be analog output in terms of the rms-level, which should be the same as the integral of the obtained spectra. However, it is necessary to determine when the data are "corrupted." Corruption may occur if the data are obtained across a turbulent/laminar interface in an intermittent signal. To this end, a wavelet representation[48,49] of the data blocks is being attempted. If the representation is suitable, it will be incorporated in the detection part of the DSP system. Another approach being pursued in parallel is the use of a pattern recognition chip developed by JPL for DNA molecule characterization[50]. This chip handles the collected data and compares it to an existing query regarding patterns. An analog approach to signal verification is to use true root-mean-square circuits, which allow integrated spectra confirmation. If the signal is not intermittent or does not have large variations as in the case of wave packets, a miniaturized analog spectral analyzer[51] may be used. In the next series of tests, the turbulent/transitional signals from the sensors will be used to develop the necessary algorithms for a realtime system. The progress and sophistication of the DSP system will occur as a series of upgrades to the system software. The level of sophistication going into each flight will depend on the software version available when the flight is scheduled. The focus of the entire development effort is to ensure maximal quality control and information throughput relative to the data transmission throughput that is possible with the latest version of the PCM system; i.e. the DSP will receive the basic PCM stream of the aero PCM system and fill in its appropriate portion as it sends the PCM further on to be transmitted to the ground. Above all, when data compression is done, the emphasis is on not losing any data due to inadequate event characterization schemes.

Conclusions

The characteristics of a crossflow transition

experiment for code validation in flight at hypersonic speeds have been outlined. The effort thus far has resulted in a promising glove shape. It appears that technical difficulties can be overcome through extensive preparations via ground testing and simulations. The experiment must be built up through a stepwise approach, since several existing techniques have to be modified or adapted. Although some of the systems must be built from scratch, no extensive new development or exploration is needed. The implications of piggybacking the experiments to available flights put severe conditions on the flexibility of the approach and make it extremely difficult to reach specific milestones, however.

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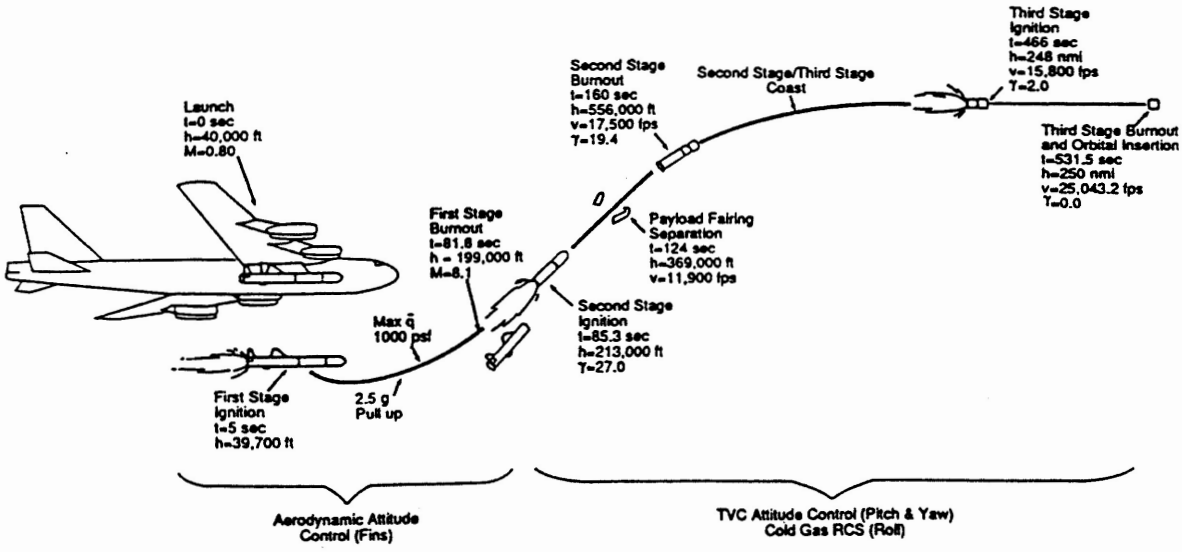

Figure 1. Sketch of the Pegasus® launch vehicle in captive position on B-52 and a typical launch. Fron Mendenhall et.al. (Ref. 7)

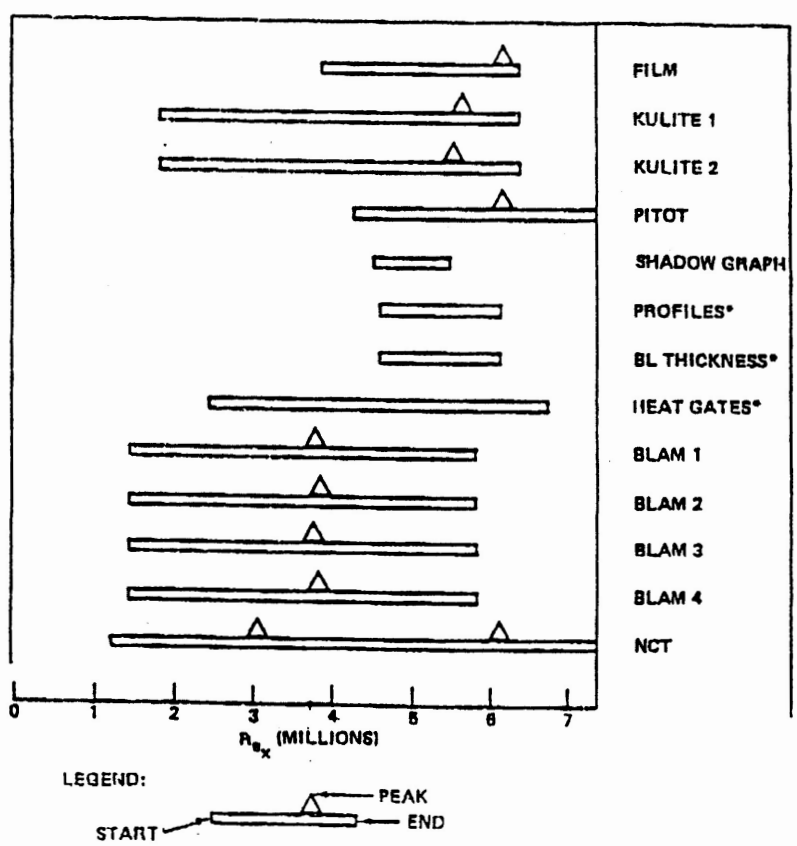

Figure 2. Summary of transition detection criteria: beginning, end and peak. From Demetriates (Ref. 5)

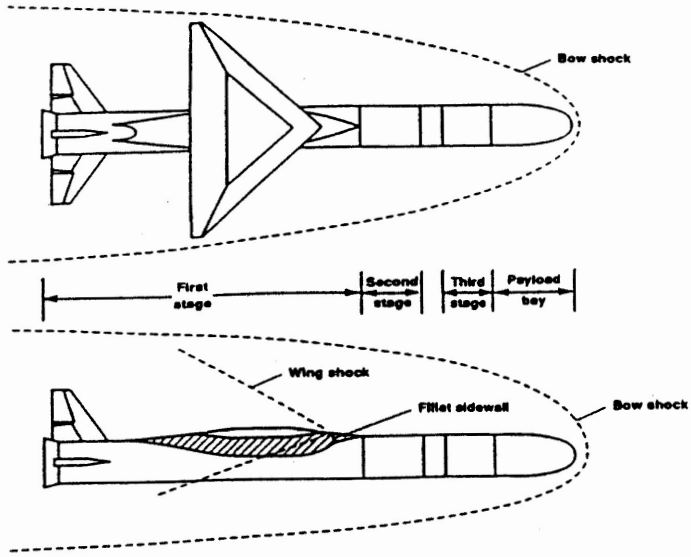

Figure 3. Geometry of the baseline Peagasus Vehicle

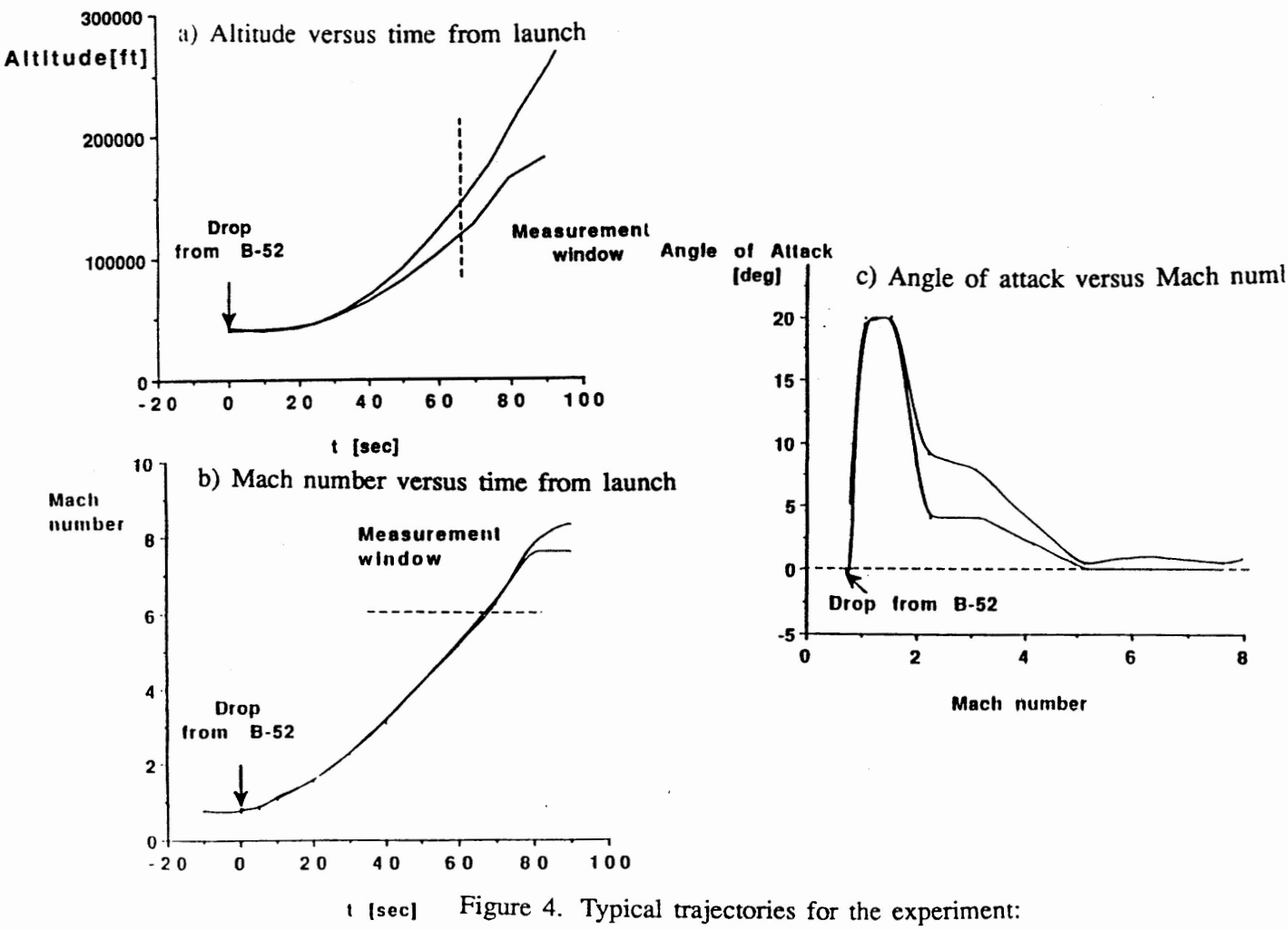

Figure 4. Typical trajectories for the experiment:

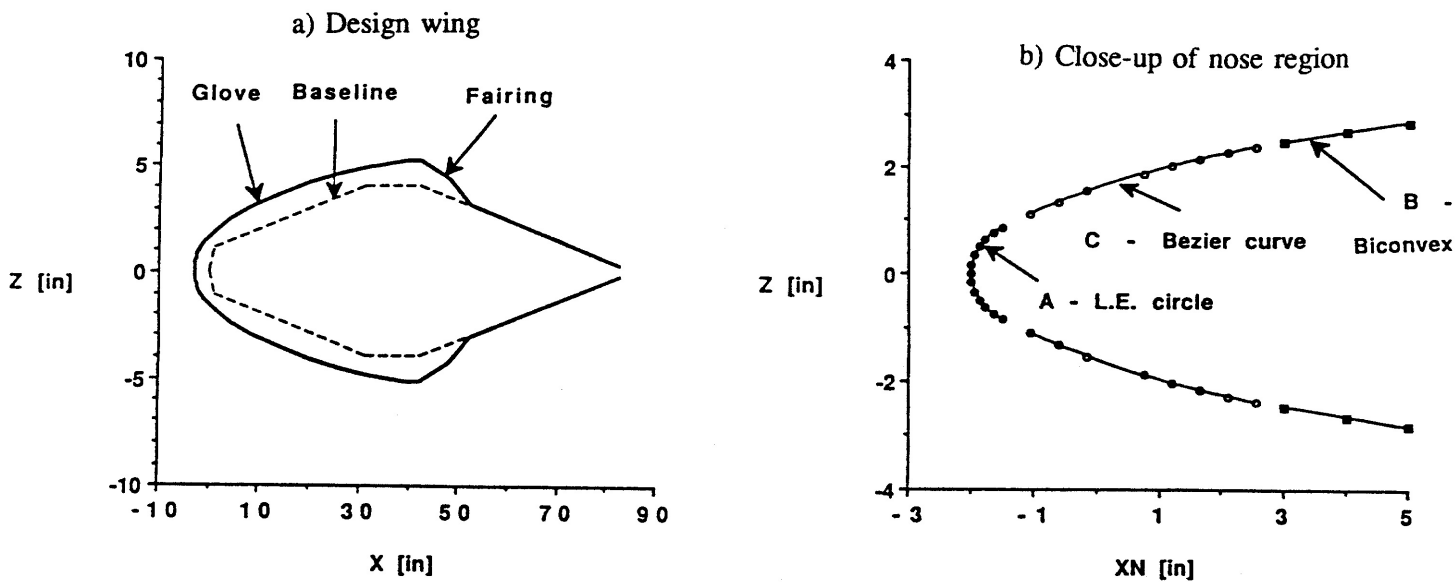

Figure 5. Glove airfoil shape —Note: vertical scale exaggerated.

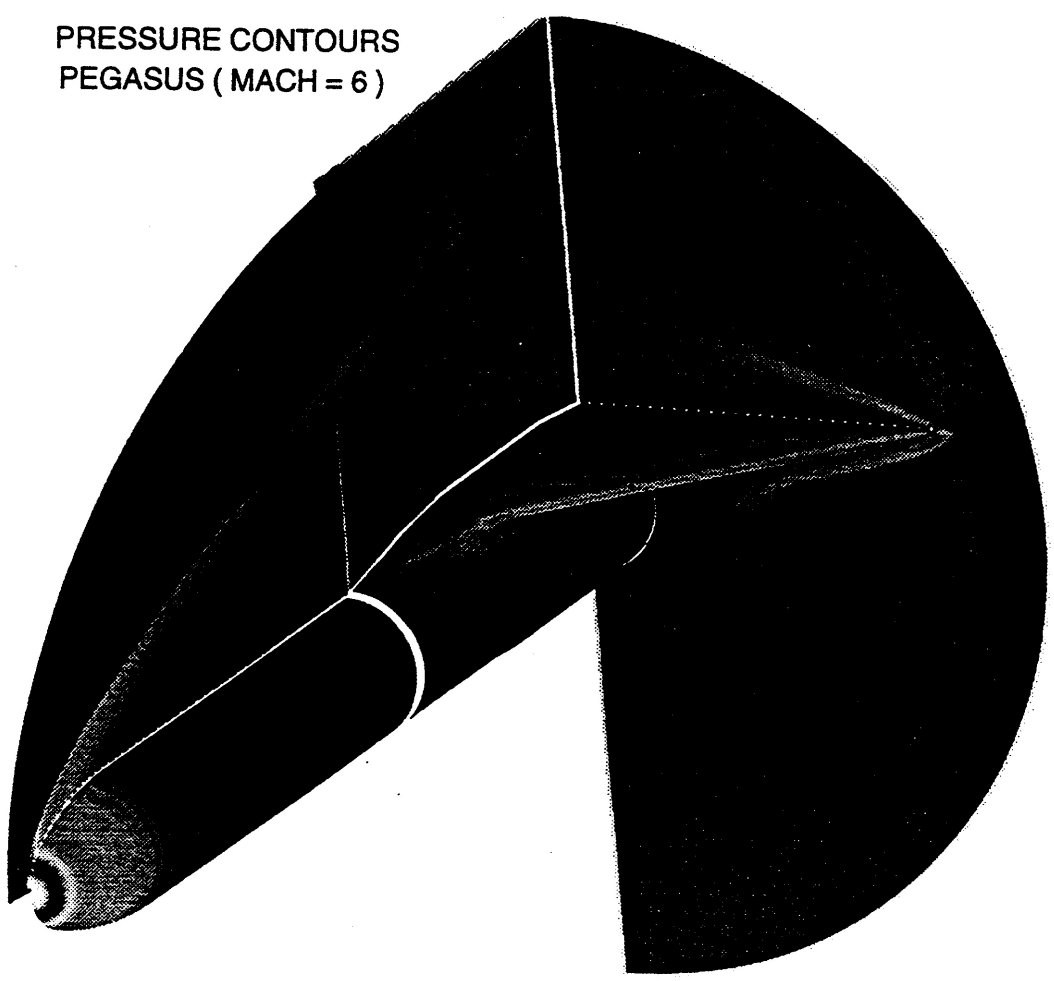

Figure 6. Flow patterns around the vehicle — bow shock

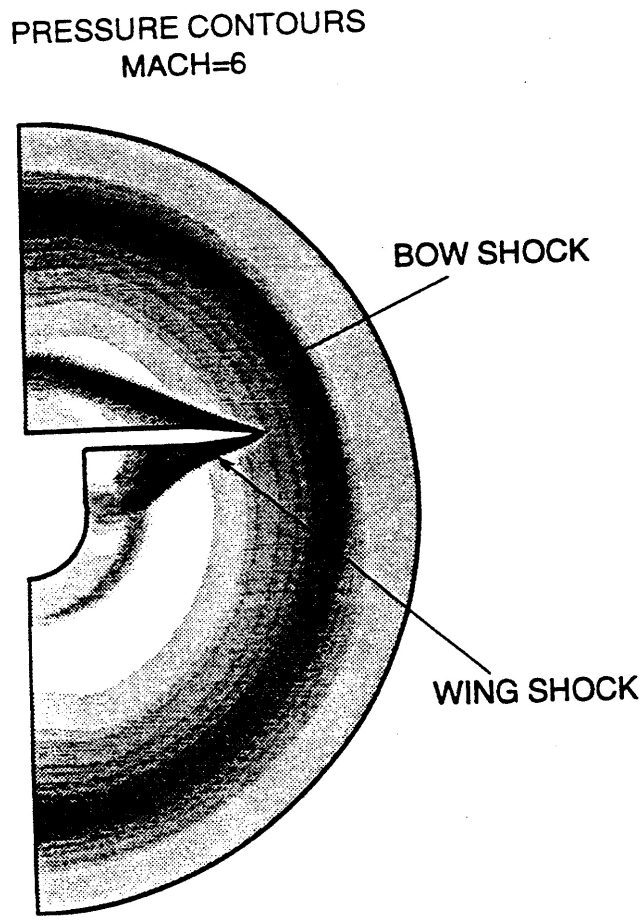

Figure 7. Flow patterns around the vehicle — wing shock

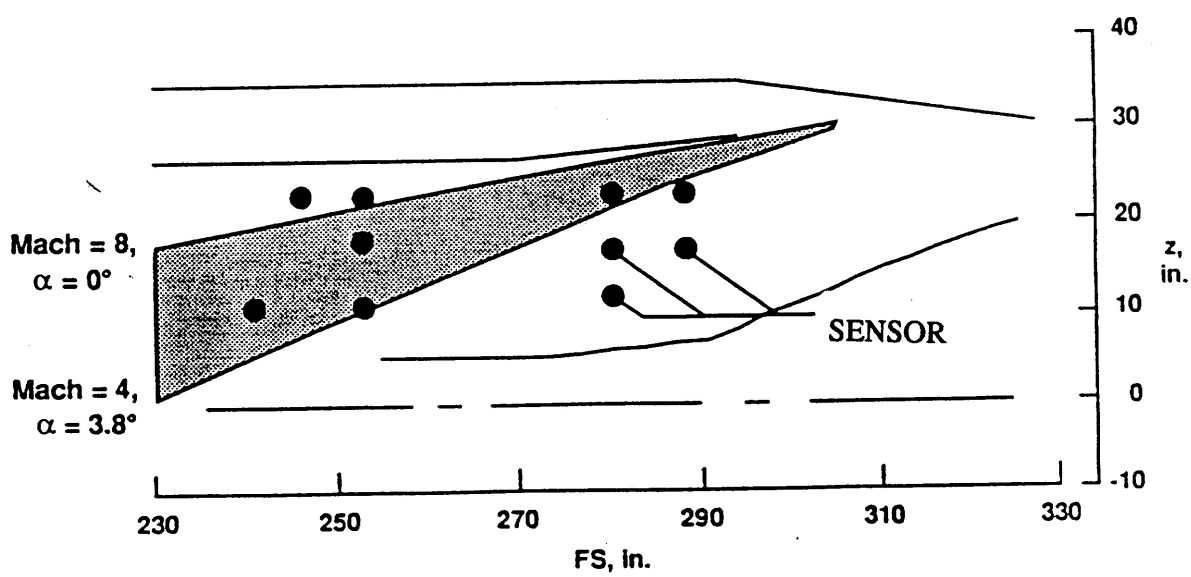

Figure 8. Fillet area with possible wing shock pattern impingement. The symbols mark locations of the sensors during the first flight. From Noffz et.at. (Ref. 8)

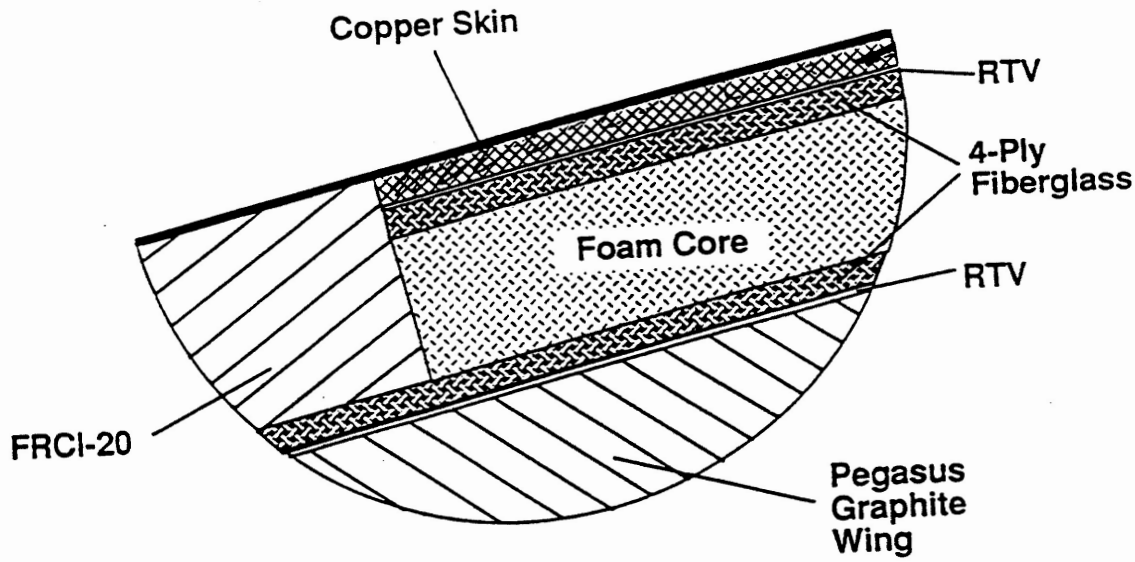

Figure 9. Glove/fairing buildup

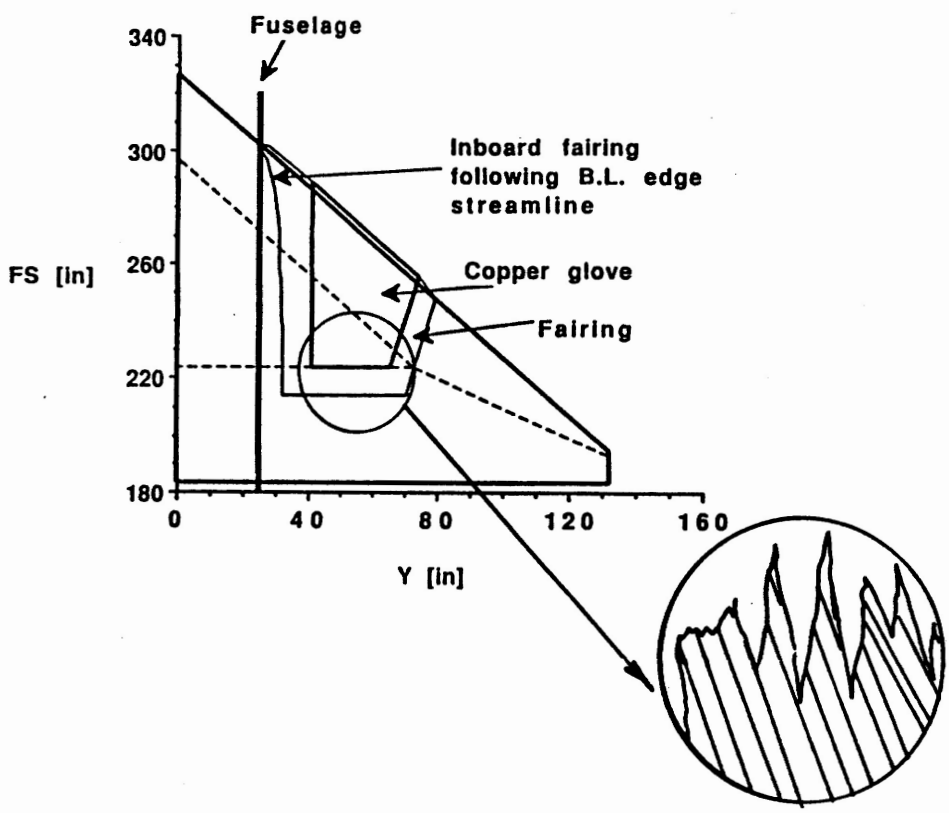

Figure 10. Glove planform and sketch of possible transition pattern

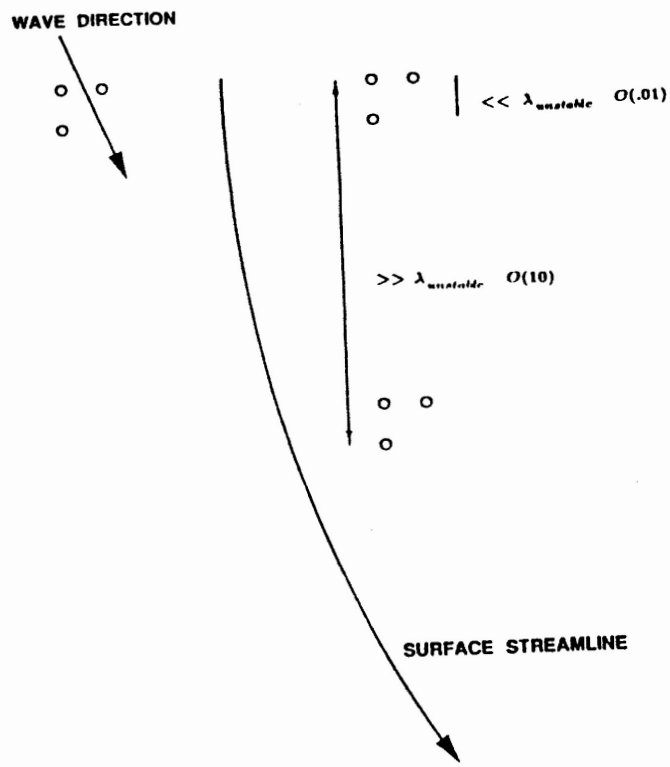

Figure 11. Array of fast pressure transducers

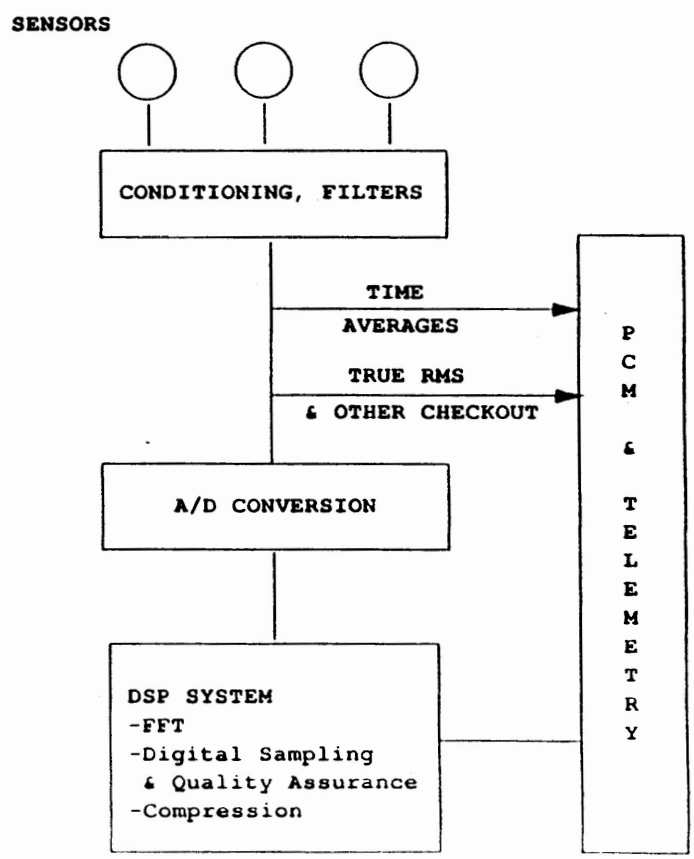

Figure 12. Transition characterization package