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AN ATTACHED FLOW DESIGN OF A NONINTERFERRING LEADING EDGE EXTENSION TO A THICK DELTA WING

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Abstract

An analytical procedure for the determination of the shape of a Leading-Edge Extension (LEE) which satisfies design criteria, including especially noninterference at the wing design point, has been developed for thick delta wings. The LEE device best satisfying all criteria is designed to be mounted on a wing along a dividing stream surface associated with an attached flow design lift coefficient \( C_{L, d} \) of greater than zero. This device is intended to improve the aerodynamic performance of transonic aircraft at \( C_L > C_{L, d} \) by controlling the wing flow field with the vortex system emanating from the LEE leading edge. In order to quantify this process a twisted and cambered thick delta wing was chosen for the initial application of this design procedure. Appropriate computer codes representing potential and vortex flows were employed to determine the dividing stream surface at \( C_L > C_{L, d} \) and an optimized LEE planform shape at \( C_L > C_{L, d} \), respectively. To aid in the LEE selection, the aerodynamic effectiveness of 36 planforms was investigated at \( C_L > C_{L, d} \). This study showed that reducing the span of the candidate LEES has the most detrimental effect on overall aerodynamic efficiency, regardless of the shape or area. Furthermore, for a fixed area, constant-chord LEE candidates are relatively more efficient than those with sweep less than the wing. At \( C_L > C_{L, d} \), the presence of the LEE planform best satisfying the design criteria was found to have no effect on the wing alone aerodynamic performance.

Nomenclature

- \( A \): aspect ratio of wing
- \( b \): span
- \( C_D \): drag coefficient, \( \text{drag}/q_m \)
- \( C_L \): lift coefficient, \( \text{lift}/q_m \)
- \( C_m \): pitching moment coefficient, \( \text{pitching moment}/q_m \cdot S \)
- \( C_P \): pressure coefficient, \( p - p_\infty \)/\( q_m \)
- \( C_{P, u} \): lifting pressure coefficient, \( C_{n, u} - C_{p, e} \)
- \( c \): chord
- \( \bar{c} \): reference chord
- \( FVS \): free vortex sheet
- \( L/D \): lift-to-drag ratio
- \( LEE \): leading-edge extension
- \( M \): free-stream Mach number
- \( p \): static pressure
- \( p_w \): free-stream static pressure
- \( PAN AIR \): panel aerodynamics computer code
- \( PSS \): pseudo-stagnation streamline
- \( PSSS \): pseudo-stagnation stream surface
- \( q_m \): free-stream dynamic pressure

Subscripts

- \( S \): wing reference area
- \( V_x \): \( X \)-component of the total velocity vector
- \( V_y \): \( Y \)-component of the total velocity vector
- \( V_z \): \( Z \)-component of the total velocity vector
- \( VL-M-SA \): vortex lattice method coupled with suction analogy
- \( x, y, z \): coordinate axes centered at the leading-edge apex
- \( x/c \): fractional distance along the local chord of the called out surface
- \( \alpha \): angle of attack, degrees
- \( n \): fraction of wing theoretical semispan (\( b/2 = 16.77 \) in.)
- \( \eta \): \( \text{LEE}/b_w \)
- \( \lambda \): leading-edge sweep angle, degrees

Introduction

Future swept-wing aircraft capable of cruising at high subsonic or supersonic speeds are likely to be required to operate efficiently over an extended portion of their flight envelope. There are two basic approaches to designing such aircraft. The first is a conventional approach and seeks to maintain fully attached flow at each point of the envelope, whereas, the second approach attempts to use the organized separated flow at off-design and attached flow at design conditions. The design criterion of the conventional approach is more desirable, because an aerodynamically efficient aircraft always achieves its best performance with attached flow unless the wing is extremely slender. The primary cause of this high efficiency is the production of aerodynamic thrust associated with attached flow at the leading edge. In order to maintain attached flow on such swept wings, techniques such as variable camber at the leading edge, a leading-edge flap, and large leading-edge radii have been developed. These techniques, illustrated in figure la via streamwise wing cuts, have been known for their potential to delay the onset of the leading-edge flow separation on moderate swept wings. However, the natural tendency of flow toward separation for highly swept wings, especially at off-design conditions such as takeoff, landing, and maneuvering, appears inevitable. At off-design performance, the flow characteristics of such aircraft are changed dramatically by the formation of a generally stable and coherent leading-edge vortex system. The schematic flow
representation at off-design condition is shown by a streamwise cut (fig. 1b) for blunt leading-edge swept wings. The resultant vortex system generates additional lift, caused by low pressure regions under the stable vortex system, and produces the well-known nonlinear aerodynamic behavior called "vortex lift." Accompanying the additional lift is the increased drag which results from the loss of the leading-edge suction associated with attached flow around the leading edge. This drag increase restricts the subsonic and transonic sustained maneuver, because of the excess engine thrust required. Furthermore, with increasing angle of attack, the shed vortex system has an inboard movement of its center and may fail to reattach on the wing and/or experience breakdown. The latter two phenomena result in a pitch-up pitching moment.

As technology in aircraft design has developed, methods for improving multimission capability have been explored. The subject of this study, to design the wing to achieve fully attached flow at the cruise design condition, and controlled leading-edge separation at takeoff, landing, and maneuvering. This method is an alternative approach to the conventional attached flow approach for designing a high subsonic or supersonic cruise swept-wing aircraft. The basic concept of this alternative approach is to let the flow separate and roll up into an organized leading-edge vortex system, which is located appropriately. For this purpose, a family of vortex control devices, such as fixed (i.e., sharp leading-edge extensions) and movable leading-edge extensions (i.e., leading-edge vortex flap) have been developed through extensive parametric studies on different experimental wing models. It has been shown that such devices, when properly designed and positioned, can confine the entire leading-edge vortices to the device upper surface and provide flow reattachment on the wing along the knee or hinge line. As a result, the wing not only has additional lift but also generates a thrust force component, as the low pressure associated with the confined vortices acts on the neighboring surfaces.

The objective of the present study is to develop a leading-edge device which would improve the aerodynamic performance and pitching moment characteristics of a thick swept-wing, cambered and twisted, high-subsonic, and low-supersonic aircraft at off-design conditions. This leading-edge device, designated as a Leading-Edge Extension (LEE), is to be mounted to a wing along the dividing stream surface, called herein, the "Pseudo-Stagnation Stream Surface (PSSS)," associated with the attached flow lift coefficient ($C_{L\theta}$). (Note that $C_{L\theta}$ = 0 work is reported in reference 11.) The Surface is called "pseudo" stagnation because, at its intersection with the wing, the velocity components are not completely zero, except at the center line. In fact, except at the center line of a three-dimensional swept wing, there exists no other point on the wing surface, from a potential flow viewpoint, where zero sideline ($W_y$) will occur. The PSSS is a dividing stream surface which separates the incoming flow into two regimes, in general, over the upper and under the lower wing surfaces. Two streamwise cuts through the PSSS are shown schematically in figure 2 to illustrate the surface curvature.

The LEE is a portion of the PSSS and, if properly determined, should not affect the main wing aerodynamic performance at the attached flow design angle of attack ($\alpha_d$). (Note that the angle of attack associated with $C_{L\theta}$ is defined as $\alpha_d$). This is illustrated by a streamwise cut through the LEE in figure 3a. However, at higher angles of incidence, vortices would be generated as a result of forced flow separation by the sharp leading edges of the LEE device. These vortices can be controlled through LEE planform shape optimization by varying parameters such as the chordwise extension, spanwise extension, and leading-edge sweep angle. A properly designed LEE planform can capture the entire leading edge vortex on its upper surface and provide flow reattachment at, or near, the wing upper surface leading edge (fig. 3b). The confined leading-edge vortex system induces suction pressure which acts on the LEE upper surface and the forward-facing area of the wing leading edges, providing an additional lift and an effective leading-edge thrust recovery. As a result, the aerodynamic thrust force generated in the flight direction yields a reduction in drag, relative to a planar configuration, and the added lift permits the aircraft to operate at lower angles of attack which may delay pitch up, due to the improved trailing-edge flow. (Note: Skin-friction drag is ignored throughout this study.)

**Design Procedure**

In order to accomplish the task of designing an aerodynamically efficient LEE planform shape, an analytical procedure had to be developed. This design procedure, which forms the basis of the present study, can be outlined in two major steps:

a) Analytical determination of PSSS at the attached flow design condition for the wing.

b) Analytical optimization of the chordwise extent and the planform shape of the PSSS at separated flow conditions. This step would, in fact, determine the optimum LEE size for the given wing.

The final LEE is considered to be optimum in this study when the following criteria, which are referred to as the design requirements, are satisfied:

- Its presence on the wing does not change the pressures and, therefore, the aerodynamic performance of the wing alone at the design lift coefficient.
- The net lifting pressure across it approaches zero (targeted value) at the design lift coefficient.
- It maintains a minimum planform area and chord length especially in the tip region where the wing local chord becomes shorter.

**Analytical Tools**

To demonstrate the design procedure outlined earlier, computer codes (i.e., analytical tools) and a candidate wing had to be selected. As a result, a thick, round-edged, twisted and cambered
wing of approximately triangular planform having a sweep of 58° and an aspect ratio of 2.3, was chosen to provide the first application of this technique. At the outset, four computer codes were considered for analytical execution of the present study at a high subsonic Mach number. These codes were Free Vortex Sheet code (FVS)\textsuperscript{12}, Panel Aerodynamics code (PAN AIR)\textsuperscript{15}, Vortex Lattice Method with Suction Analogy code (VLM-SA)\textsuperscript{14-16}, and a transonic computer code. Although efforts were made to obtain and employ a nonlinear transonic computer code in this study, due to the high subsonic Mach numbers of interest, none was available to the authors when this study began which could reliably estimate the pressures on thick-delta wings. The FVS was not employed because of the first author’s unsuccessful past experience, which included efforts to obtain a converged solution for the DM-1 with a leading-edge extension of reference 11\textsuperscript{17}. Further valid flow-field results could not be obtained by the VLM-SA because of the lack of thickness modeling by the code. The PAN AIR code was evaluated by modeling the candidate wing geometry using the flow conditions of interest and pressure distribution obtained from the code is compared with experimental results and discussed in appendix A. Hence, after the preliminary examination of the remaining code options, the PAN AIR was assigned to determine the PSSS, and the VLM-SA code to establish the proper extent for the LEE.

**PSSS Determination**

**Assumptions**

- Part 1 of the two-part present study seeks to determine a representation of the PSSS based on the following assumptions:
  - There exists a PSSS associated with a swept-wing aircraft at the attached flow design condition.
  - The intersection of the PSSS with a number of parallel \( XZ \) planes spanning the wing stations, is the graph of the pressure distribution representative of the Pseudo-Stagnation Streamline (PSS) leading to the pseudo-stagnation point (i.e., \( \|V_x\| = \text{minimum}, \|V_z\| = 0 \)); note that \( |V_x| \) is not assumed to be small nor zero, it is not treated in part 1 of this study).
  - The PSS shapes are derived from the local slopes of the resultant velocities \( \sqrt{V_x^2 + V_z^2} \) at appropriate points in the \( XZ \) plane.
  - A spanwise surface fitted linearly through the resulting intersections is an approximation of the PSSS described in the first assumption.

Part 2 of the present study shows how improvements can be made in the resulting LEE by including the influence of \( V_y \).

**Survey Networks**

The survey networks adopted in the present study were vertical \( XZ \) planes located at 16 different stations along the semispan of the wing model. These survey networks were generated such that each would enclose the nose portion of its corresponding stationary wing section a distance of approximately 0.08\% of the wing \( c_r \). The networks began at the upper surface just behind the leading edge and extended around the nose to the lower surface mid-chord. Due to the similarity of the survey network geometries and the involved process of their generation, only one typical survey network (located at the fourth station) shown in figure 4, will be discussed.

This figure also shows the planform distribution of the other survey network locations over the semispan of the wing model. Further, the enlarged cross-sectional view of the survey network and the nose portion of its corresponding wing section at the fourth station is shown in figure 5a. Since the PAN AIR code velocity field solutions were assigned to be calculated at the center point of each panel in a particular survey network, it was essential to provide the survey networks with enough panels so that, once the resultant velocity vectors associated with \( V_x \) and \( V_z \) were plotted, the pseudo-stagnation streamlines could be depicted graphically. For this purpose, a geometrical computer code, called GEOMABS\textsuperscript{18}, was employed to intensify the paneling on the survey networks. Figure 5b, shows the rerepaneled survey network. It can be seen from the figure that the panel density is concentrated primarily around the portion of the survey network which faces the nose of the associated wing section. This would allow more velocity vector solutions, which are needed to determine graphically the accurate location of the resulting pseudo-stagnation streamlines as they meet their corresponding wing section. Similar survey networks were generated for all 16 semispan stations of the wing model. Each individual survey network was positioned on the wingspan, and separate PAN AIR code execution was performed.

**PAN AIR Analysis**

The PAN AIR task was successfully accomplished, and the velocity field solutions for each wing section was analyzed. A velocity network was generated at the attached flow design lift coefficient of 0.25 and Mach number of 0.8. (Note that the angle of attack associated with \( C_{Ld} \) was 6.0°). Neglecting the sidewash (\( V_y \)) effect, the resultant velocity vectors obtained from vectorial addition of the axial (\( V_x \)) and the upwash (\( V_z \)) velocity components associated with each wing section were plotted. Further, the streamlines associated with minimum velocity magnitude (i.e., \( |V_x| \) = \text{minimum}, \( |V_z| = 0 \), pseudo-stagnation point) was drawn tangent to the plotted velocity vectors. Figure 6 shows the nose portion of a typical airfoil section with its corresponding velocity field and the graphical PSS solution. These graphical streamline solutions yielded their coordinate point relative to the corresponding wing section. Each of these solutions was equally extended out a distance of 4.8 in. (i.e., 19-percent of wing \( c_r \)) ahead of the wing leading edge. This distance was thought to be sufficient to bracket the useful design space of a LEE device from an aerodynamic and structural viewpoint. Uneven velocity field solutions obtained at the tip region prevented the graphical generation of the PSS outboard of the wing 89-percent semispan.
The warped PSSS was represented by fitting straight line segments through the available PSS solutions that ran roughly parallel to the wing leading edge. The three-view computer drawing of the determined PSSS solution is shown in figure 7. Further, five cross-sectional cuts through the wing-PSSS combination and the enlarged cross-sectional view of the same cuts are shown in figure 8. The resulting PSSS has approximately a 14.33 in. semispan (i.e., 89-percent of the wing semispan) and 4.8 in. constant-chord extent.

It is essential to examine the degree of accuracy of the determined PSSS solution. For this purpose, the PAN AIR code was employed once again to model the wing-PSSS combination at the design condition (i.e., $\alpha_0 = 6^\circ$, $M_0 = 0.8$) by specifying the PSSS as a lifting surface. Figure 9a shows the effect of the PSSS presence on the wing pressure distribution at a typical wing section to be insignificant. Also, as shown in figure 9b, the net lifting pressure across the PSSS appears except at the local leading edge for the same typical section. From these results, it is evident that the addition of the PSSS surface does not cause much change in the performance of the wing model at the design condition. Therefore, it is concluded that the determined PSSS solution is close to the actual dividing stream surface (i.e., PSSS) and hence, it completes part (a) of the design procedure outlined earlier.

**LEE Planform Optimization**

Part (b) of the design procedure is performed by employing the VLM-SA code, which attempts to optimize the PSSS planform shape. This optimum shape would then be designated as the shape of the LEE device. The aerodynamic effectiveness of 36 different LEE planform shapes was examined for the given wing by considering the influence of geometrical parameter choices such as constant chord (CLEE), constant sweep (ANGLE), and span extent (SPAN). The optimality of these planforms relative to the basic wing geometry is illustrated schematically in figure 10. Although the twist and camber of the basic wing is represented by its mean camber surface, the thickness effect is ignored by the VLM-SA code. As discussed in Appendix B, the analytical solution for the basic wing model was first used to provide a base line for comparative assessments of the LEE device, appears to be inadequate. As a result, throughout this study the aerodynamic effectiveness of different wing-LEE combinations was emphasized more relative to one another rather than to the basic wing.

The VLM-SA drag polar solutions for the selected constant-chord and constant-sweep LEEs with 89-percent span extent are presented in figure 11. It is evident from this figure that considerable improvement can be achieved in the lift and drag characteristics of the wing-LEE combination by employing a longer LEE chord extension. However, as it was noted, to meet design criteria for the final LEE planform was to satisfy a minimum chord length, especially in the tip region. A smaller chord LEE not only benefits from the reduced structural weight, but it also minimizes the effect of bending moment about the wing-LEE junction. This bending moment occurs at off-design conditions where the low pressure associated with the leading-edge vortices act on the upper surface of the LEE device.

It appears instructive to consider the aerodynamic effectiveness of different LEEs relative to their planform area by considering the effect of other geometrical parameters. For this purpose figure 12 was prepared. In general, this figure shows that the LEE planform area does not have much effect on lift-to-drag ratio over the entire range of angle of attack. Further, with regard to the comparison of the aerodynamic effectiveness of LEEs with different constant chords and constant sweep angles, the following conclusions are drawn based on equal LEE planform area:

- At moderate angles of attack (6° to 10°), it appears that constant-chord LEEs produce a better lift-to-drag ratio.

- At 12° angle of attack, LEEs with constant sweep-angles of 57° to 55° generate better L/D; however, outside this range constant-chord LEEs achieve either the same or better improvements.

- At 14° to 16° angle of attack, only low sweep-angle LEEs appear to be more effective. However, at higher angles of attack (18° to 20°), the figure shows a very slight change in L/D ratio, regardless of the LEE's planform shape or area.

In general, this investigation revealed that the outboard reduction of the LEE-span extent minimized the lift-to-drag ratio, regardless of the LEE's planform shape and area. Also, with the same planform area, it was found that constant chord is relatively more effective than LEEs having sweep angles less than that of the wing. Therefore, two LEE planforms, each with 89-percent span extent relative to the wing span, one with 1 in. and the other with 0.9 in. constant chord, were selected as being the best candidates for the final LEE design planform. These results have been reported in reference 19 and conclude the first part of the present study.

The second part was undertaken to improve the pressure distributions of the wing-LEE combination at $\alpha_0$. For this purpose, the PAN AIR code was employed to model the basic wing with 1.2 in. constant-chord LEE having 89-percent span extent. As shown in figure 13a, the LEE presence appears to disturb the pressure distribution slightly at the leading edge of the typical wing section. This effect indicates that perhaps the LEE surface is not completely aligned with the flow. In fact, the same effect is more obvious from the net lifting pressure across the LEE at the same typical section, as shown in figure 13b. Apparently, the LEE surface is generating some negative pressure on its upper surface, especially around its leading edge. This misalignment of the LEE surface with the incoming flow needs to be understood when the aerodynamic effects of the actual geometry being modeled is reconsidered. This examination points out a deficiency in one of the assumptions made in the PSSS determination (i.e., neglecting the effect of sidewash velocity component). In particular, the spanwise connection of the graphically determined PSS with straight lines
to represent the PSSS produces a different three-
dimensional potential flow problem. Further, it
is important to note that all the pressure distri-
butions presented in this study are based on the
second order solutions where the sidewash effects
are included.

Aerodynamic principles suggest that, in order
to reduce the negative lifting pressure and to
improve the flow characteristics at the LEE leading
edge, the LEE surface must be lowered. These
principles were employed and the LEE surface was
lowered by rotating it slightly (7°) while holding the
pseudo-stagnation points fixed. Figure 14
shows the PANS AIR solutions for the modified LEE
(MOD2)-LEE geometry. This figure shows that by
lowering the original LEE surface, the chordwise
pressure distributions on the basic wing and
across the LEE surface were improved.

Subsequently, the original wing geometry was
determined to be in need of revision so as to
better represent the actual geometry. As a
result, the PANS AIR code was employed once again
to model the modified wing (MOD2-WING) geometry.
Furthermore, the MOD2-LEE was slightly modified
for the new wing geometry at \( \alpha_d \), and is called
MOD2-LEE. As shown in figure 15, the chordwise
pressure distribution at the same typical wing
section appears to be insensitive to the LEE
presence. This figure also shows the net lifting
pressure across the LEE surface to be approxi-
mately zero (i.e., the targeted design value).
The summary of the force and moment results
obtained from PANS AIR code are presented in figure
16. This figure shows how well each different wing
LEE combination duplicates the wing-alone
aerodynamic results at \( \alpha_d \). Thus, the previously
provided method has been shown to theoretically
duplicate the effect of the LEE at the
design angle of attack.

Concluding Remarks

The present study demonstrated the applica-
bility of a newly developed analytical design
procedure for the determination of a noninter-
ferring Leading-Edge Extension (LEE) which satis-
fies certain design criteria for thick delta
wings. This procedure led to a successful estima-
tion of the shape of the LEE device, which is a
portion of the pseudo-stagnation flow surface, that
had essentially no effect on the wing-alone
aerodynamic results at its design angle of attack.
Through an examination of the available analytical
tools, the PANS AIR and VLM-SA computer codes were
employed to carry out the first application of the
developed design procedure for the given wing of
the present study.

The results obtained for 36 different LEE
platforms suggest that 1.2 in. and 0.8 in.
constant chord with 89-percent span extent satisfy
the constraints (i.e., design criteria) and are
considered good candidates for the final LEE
platform design. Although efforts have been made
to validate the results obtained in the present
study whenever possible, these results need to be
verified experimentally.

Appendix A.- PAN AIR Evaluation

The intended purpose of this appendix is to
evaluate the analytical capability of the PAN AIR
code for thick wing configurations at high Mach
numbers of interest. As a result, the thick wing
geometry of the present study was modeled with PANS
AIR code at \( \alpha_d = \alpha_p \) and \( M_w = 0.8 \). The pressure
distribution obtained by the code is compared with
unpublished experimental data at \( \eta = 0.30 \), in
figure A1. (Note that the experimental data were obtained at the NASA-Langley 7- by 10-Foot High-
Speed Tunnel.) The comparison shows good agree-
ment between the theoretical and experimental data
except on the forward part of the upper surface.
There the flow must be supercritical, which the
theory cannot estimate correctly, even with the
isentropic pressure rule.

Appendix B.- VLM-SA Evaluation

As part of the present study, it was impor-
tant to examine the analytical capability of the
VLM-SA code for a thick wing configuration with a
leading-edge extension. For this purpose, the
experimental data obtained by Wilson and
Lovell\(^1\) on the thick DM-1 with and without
the LEE was selected for validating the results
obtained from the VLM-SA code. The DM-1 is a
symmetrical wing configuration with an airfoil
section like the NACA 0015-64 and no twist, so the
LEE design lift coefficient \( (C_l)_{\alpha_d} \) was zero.
Although the effect of leading-edge radii is
included in the resulting VLM-SA solutions, the
thick DM-1 is approximated by its projected
planoform (flat DM-1) in this study.

Experimental values for the lift and drag
polars obtained by Wilson and Lovell for the DM-
with and without the LEE as well as the resulting
VLM-SA solutions for the same configurations are
presented in figure B1. Obviously, the code over-
estimates the lift for both the DM-1 and DM-1+LEE
combination throughout the angle-of-attack range.
However, the drag polar comparison shows that,
for the basic DM-1, the VLM-SA solutions have the same
variation as the experimental data up to \( C_l = 0.6 \).
Beyond this lift coefficient, the curves differ
due to the disorganized flow over the basic DM-1,
which causes both a drag increase and lift
decrease\(^4\). As a result, the experimental drag
polar is higher than the theoretical solution.
For the DM-1+LEE combination, the VLM-SA over-
estimates the drag in the lift coefficient range of
about 0.05 to 0.80. This difference was rather
expected, because the resulting VLM-SA solutions
do not include the effect of the low pressures
acting between the LEE and upper surface maximum
thicknes line of the wing sections to produce a
thrust. Hence the computed \( C_l \) values are higher
than the experimental data. Therefore, by analogy
it is expected that the VLM-SA solution for the
drag would be higher in the wing+LEE analysis of
the present study than the experimental data.

REFERENCES

1Hentz, W. M., Jr.: Effects of Leading-Edge
Camber on Low-Speed Characteristics of Slender

2Rao, D. M.: Exploratory Subsonic Investigation
of Vortex Flap Concept on Arrow Wing Configura-
tion. NASA CP-2108 (Part I), November 1979.


Figure 1.- Typical flow types occurring at the leading edges of an aircraft.

Figure 2.- Schematic representation of the PSSS corresponding to two wing sections.

Figure 3.- Leading-edge flow of a wing + LEE combination. (Streamwise cut.)
Figure 4. - Semispan planform view of the wing model with survey network located at fourth station.

Figure 7. - Three views of the determined PSSS solution.

Figure 8. - Typical streamwise cuts through the wing + PSSS combination.

Figure 5. - Enlarged cross-sectional view at fourth station.

Figure 6. - Graphically determined PSS from typical section velocity field.

Figure 9. - Effect on wing and PSSS pressures from combination ($n = 0.34$, $\alpha_d = 6.0^\circ$, $M_m = 0.8$).
Figure 10. LEE design parameters and ranges.

Figure 11. Drag polar for constant chord and sweep variation ($\eta_{LEE} = 89\%, M_\infty = 0.8$).

Figure 12. Effect of LEE-planform geometrical-parameters on lift-to-drag ratio ($\eta_{LEE} = 89\%, M_\infty = 0.8$).

Figure 13. Effect on wing and LEE pressures from combination ($\eta = 0.34, \alpha_d = 6.0^\circ, M_\infty = 0.8$).

Figure 14. Effect on wing and MOD$_1$-LEE pressures from combination ($\eta = 0.34, \alpha_d = 6.0^\circ, M_\infty = 0.8$).

Figure 15. Effect on MOD-wing and MOD$_2$-LEE pressures from combination ($\eta = 0.34, \alpha_d = 6.0^\circ, M_\infty = 0.8$).
Figure 16.- Wing and LEE solutions from PAN AIR at $\alpha_d = 6.0^\circ$.

Figure A1.- Experimental and theoretical pressures for the MOD-wing ($\eta = 0.30$, $\alpha_d = 6.0^\circ$, $M_\infty = 0.8$).

Figure B1.- Experimental and theoretical lift and drag polar ($A = 1.8$, $M_\infty = 0.0$).