Treatment of Supersonic Configurations by an Updated Low-Order Panel Method

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A recently updated version of the Woodward linearized subsonic/supersonic panel method (USSAERO) has been applied to the calculation of the supersonic characteristics of wing body combinations The use of a new singularity with directional properties (triplet) for representing body effects and the newly developed extension to the supersonic case of a nonplanar boundary condition over the wing(s) overcomes many of the shortcomings exhibited by former USSAERO versions when analyzing complex configurations, such as the fighter type air plane at supersonic speeds Examples of applications to significant test cases are presented and discussed Comparisons of the results with other theoretical and/or experimental data demonstrate capabilities and limitations of the present method

Nomenclature

b	= span length
c	= chord length
C_D	= drag coefficient
C_L	= lift coefficient
C_M	= pitching moment coefficient
C_p	= pressure coefficient
d	= body diameter length
L	= body length
M	= freestream Mach number
p	= pressure
p_∞	= freestream pressure
Re	= Reynolds number
$u\ v\ w$	= perturbation velocity components
V	= total velocity magnitude
V_∞	= freestream total velocity magnitude
$x\ y\ z$	= Cartesian coordinates
α	= angle of attack
β	= Prandtl Glauert number, $=\sqrt{1-M^2}$
ΔC_p	= load coefficient
γ	= specific heat ratio
ϕ	= perturbation potential

I Introduction

CURRENT trends in the design of future advanced air planes emphasize supersonic performance and the renewed interest of industry in computational methods capable of supporting aerodynamic design at supersonic speeds

Within existing techniques in this regime, finite difference methods solving either the full potential or Euler equations have made the most significant advances in recent years and successful solutions over simpler configurations have been reported [1 4] For applications to complete aircraft con figurations however the lack of efficient numerical procedures to generate the computational grid around ar bitrary three dimensional configurations is still a major problem area for these methods

In subsonic flow the surface singularity technique has been demonstrated to be the most efficient approach to the solution of inviscid subcritical flows around arbitrarily complex three dimensional configurations; a variety of these methods has been developed The extension of this technique to supersonic flow, however has led to some unique stability problems and only a few supersonic panel methods have been presented [5 16]

Basically, the numerical instabilities have been associated with the propagation of spurious Mach waves inside bodies and thick wings and with discontinuities in the perturbation velocity flowfield

In the panel method of Ehlers et al [11 13] continuity of higher order source and doublet strengths across the panel edges eliminates any infinite singularity in the velocity field and mixed internal external boundary conditions cancel flow perturbations inside any closed surface Successful ap plications of the method have been presented in Refs 14 and 15 An alternate approach based on the use of a low order singularity with directional properties called a triplet has been developed by Woodward and has been demonstrated successfully in the analysis of isolated bodies [9 10]

The goal of the present investigation is the evaluation of a numerical method mainly based on the panel method of Woodward including the triplet singularity, for the analysis of realistic airplane configurations

II General Description of the Method

The numerical method used in this paper is based mainly on version B of Ref 8 with inclusion of the triplet formulation described in Refs 9 and 10 representing body effects in supersonic flow. A detailed description of the basic theory can be found in Ref 7

Basic Formulation

As originally developed by Woodward the USSAERO code is intended to be a unified approach to solve both subsonic and supersonic steady inviscid flows about arbitrary three dimensional configurations

The solutions are governed by the linearized potential equation

$$\beta^2\phi_{xx}+\phi_{yy}+\phi_{zz}=0 \tag{1}$$

where $\beta^2=1-M_\infty^2$ is the compressibility parameter and ϕ is the potential function of the perturbation velocity

The boundary conditions associated with Eq (1) are as follows

1) Vanishing of the normal velocity at the external surface of the configuration

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2) Fulfillment of the Kutta condition at the subsonic trailing edges of the lifting surfaces (if any)

Except for a very limited number of cases solutions to Eq (1) and associated boundary conditions cannot be found analytically In the surface singularity techniques, such as the present one, the actual configuration is replaced by a discrete number of distributions of singularities which satisfy Eq (1) and whose strengths are adjusted to satisfy the boundary conditions required by the geometry of the particular con figuration at a discrete number of control points

Paneling Schemes

Although this may result in somewhat arbitrary paneling schemes in certain cases the method discriminates between body like and wing like components of the total con figuration, for which different paneling schemes and different types of singularity distributions are used

Surface panels used to describe the bodies must have leading and trailing edges lying in planes perpendicular to the body axis Thus, the body must be split up into a certain number of cross sections which can be of arbitrary shape and quadrilateral panels are formed connecting points lying on adjacent sections Adjacent sections having the same number of panels are assembled together to form a body segment Presently the code can compute a total number of body panels equal to 1800 (on half configuration when it is symmetrical to the $y=0$ plane), distributed over 12 body segments In supersonic flow however, the use of the triplet singularity, which requires the grouping of six panels implies that only one segment can be used to describe a single body This can be a severe restriction for the paneling of wing body con figurations (possible solutions to overcome the problem will be discussed in Secs VI and VII)

Wing panels must have side edges parallel to the freestream velocity Again wings must be divided into segments, which have straight leading and trailing edges which may be swept and can be further subdivided in streamwise strips having the same number of panels A maximum number of 1800 wing panels distributed over 60 streamwise strips is allowed by the present version of the code

Singularity Distributions

Originally released USSAERO versions used constant source panels on the external surface of the body To prevent the propagation and reflection of spurious Mach waves in the interior of bodies at supersonic speeds Woodward developed a new singularity, called a triplet, which is obtained by superimposing a vortex distribution on a constant source panel For panels having supersonic leading edges, this combination of elementary singularities cancels the induced perturbation velocity in the flowfield below the panel, giving the desired directional property Figure 1 simplifies the basic concept for a two dimensional flow

For arbitrarily shaped configurations or even for isolated axisymmetric bodies at angle of attack which as it is well known, can be treated by the much simpler line singularity methods a special arrangement of six panels is required to make up a triplet singularity This triplet singularity forms closed vortex rings preventing violation of the Helmholtz vortex conservation law without requiring a trailing vortex wake (Fig 2)

On wing panels, the original USSAERO version uses source and vortex singularity distributions which are linearly varying in chordwise direction and constant over the span For wings having supersonic leading edges however, Cenko[17] suggested that better results are obtained when the linearly varying vortices are replaced by constant vortex distributions Fur thermore, to stabilize the numerical solution, the location of the control points has to be moved to 95% of the local chord of all constant vortex panels except for those lying ahead of the Mach lines originating from the wing apices Pressure distributions obtained using constant vortices are in closer agreement with linear theory than former results as shown by the results of a test case carried out in Ref 18 (see Fig 3)

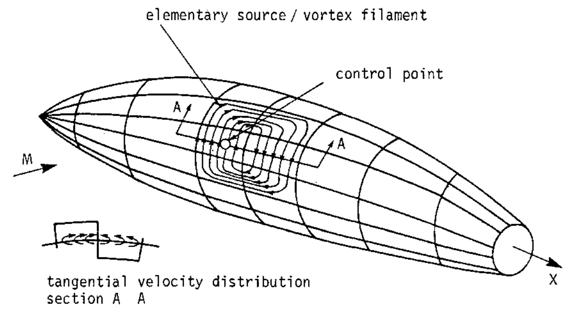

Fig 2 Triplet arrangement on arbitrary body [10]

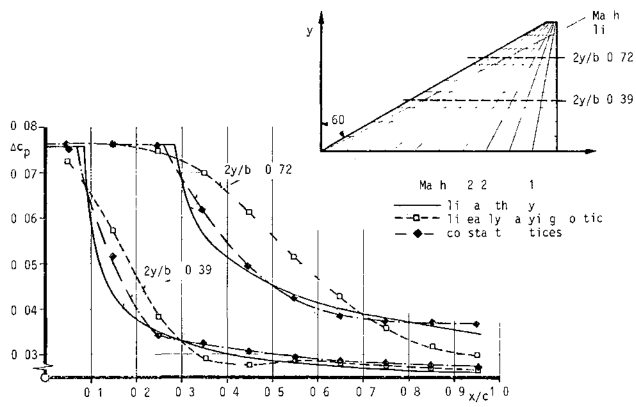

Fig 3 Loading distribution comparisons on a thin delta wing

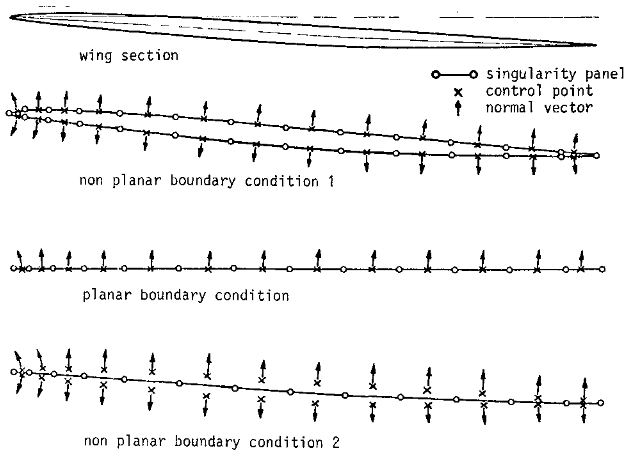

Fig 4 Wing boundary condition comparisons

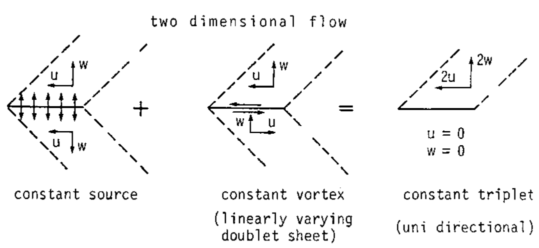

Fig 1 Supersonic triplet concept [10]

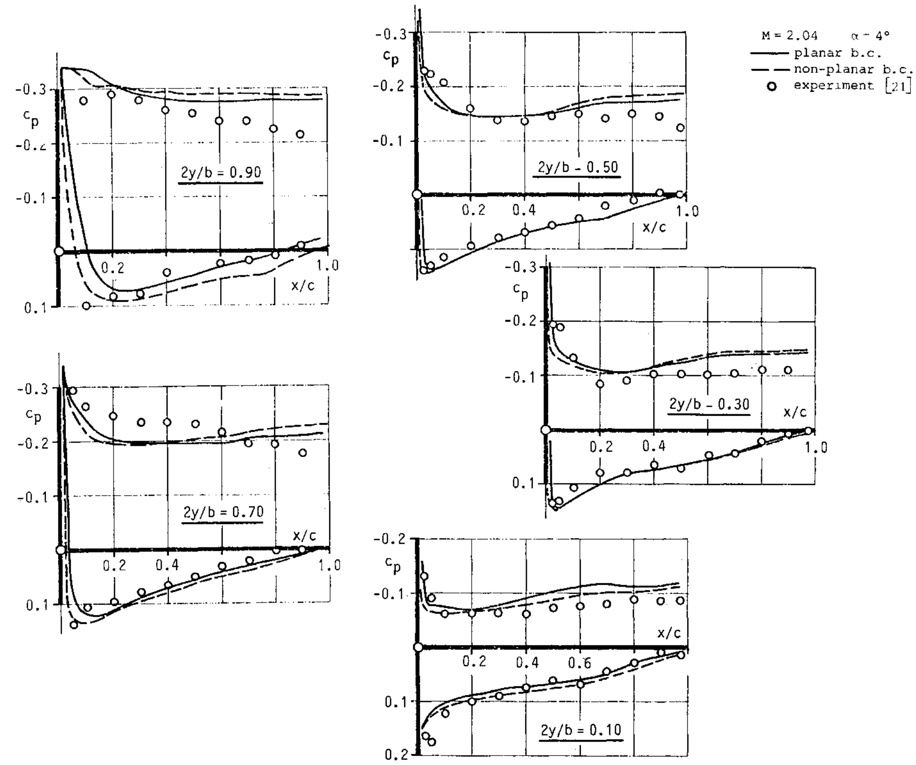

Fig. 5 Pressure distribution comparisons on Carlson wing 2.

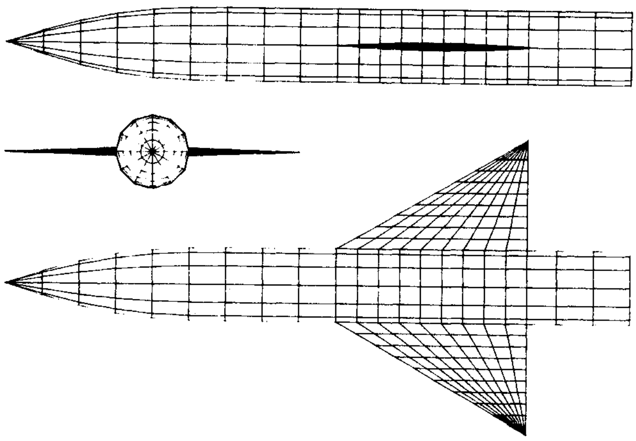

Fig. 6 AGARD model B configuration paneling.

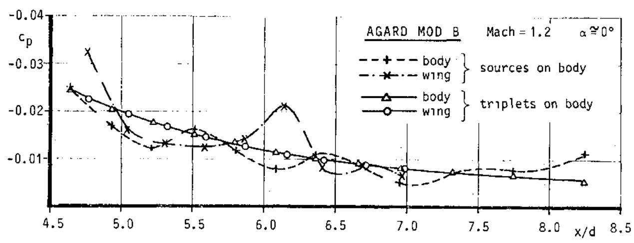

Fig. 7 Pressure distribution comparisons along wing root of AGARD model B.

For wing-body combinations, the method calculates the interference effects directly, i.e., wing singularities influence body control points and vice versa. In the original version, the wing lift is carried over onto the fuselage by extending the bound vorticity of the wing to the symmetry plane. Using triplets on the body at supersonic speeds, however, this approach proved inadequate, resulting in surface pressure fluctuations in the wing-body intersection region. The reasons for this behavior, while not fully understood, can be related to the presence of singularity distributions inside the body It was found, indeed, that replacing this internal vortex sheet by an equivalent line vortex which follows the wing root and continues downstream to form the inboard edge of the wake, improves the smoothness of the pressures. In principle, concentrated line vortices introduce too high induced velocities at nearby control points, but numerical studies have shown this effect to be very local and to have no significant influence on computed total forces and moments.

All of the aforementioned modifications, i.e, the triplet singularity distributions for bodies, the constant vortex arrangement for wing segments featuring supersonic leading edges, and the replacement of the internal vortex sheet for wing-body combinations, have been introduced into the numerical method presented here as user's default options at supersonic speeds.

Boundary Conditions

Tangential boundary conditions are applied on wings and bodies, i.e., the normal component of the total velocity (freestream plus perturbation velocities) is set equal to zero at each control point, except when a different value is entered by the user. This condition is referred to as velocity boundary condition or Goethert's rule 2 in Ref. 8. For bodies the control points are always located at the centroid of the panels which form the external surface of the configuration; for the wings *three* optional methods are available. Originally released USSAERO codes applied a nonplanar boundary condition option, whose use was restricted, however, to subsonic Mach numbers and a planar one valid for both subsonic and supersonic speeds.

For the original nonplanar boundary condition, the singularity panels are wrapped around the actual surface of the wing and the control points are located at the centroid of each panel. From all of the possible and ideally equivalent arrangements of sources and vortices, the present version of the method chooses the principle of symmetrical sing-

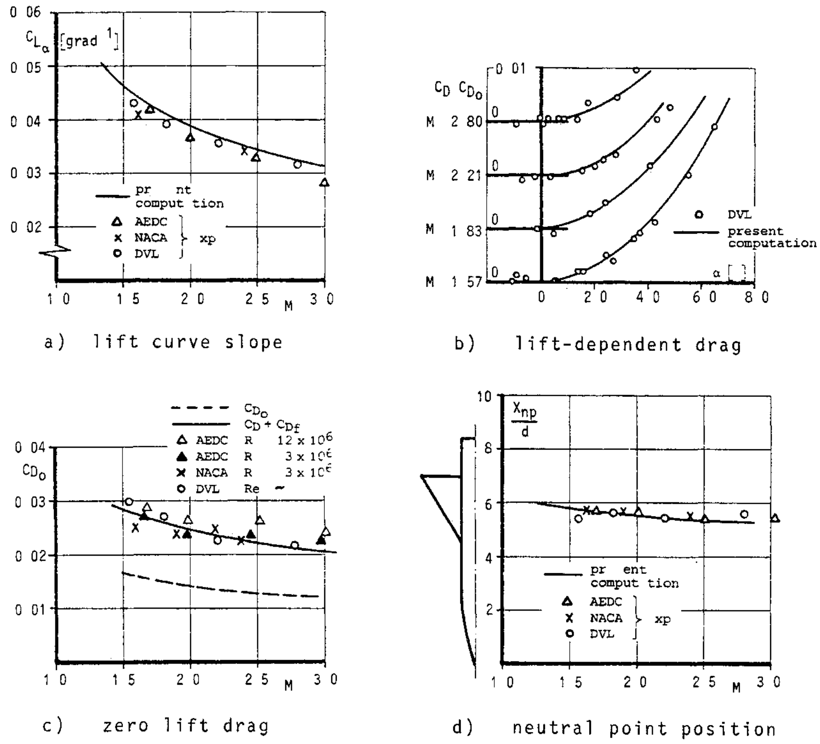

Fig 8 Longitudinal characteristic comparisons on AGARD model B

ularities [8,19] Here the constant source and the linearly varying vortex distributions have the same strengths on corresponding panels on the upper and lower surfaces of each wing strip This combination automatically satisfies the Kutta condition since the vorticity is zero at the trailing edge At supersonic freestream velocities however, this type of paneling tends to generate numerical instabilities resulting from the propagation and reflection of Mach waves in the interior of the wing

The planar boundary condition overcomes this problem because in this option the actual wing is replaced by an "equivalent" thin one This "equivalent" wing is obtained by projecting each wing segment onto a plane which is parallel to the x axis and passes through the leading edge According to linearized boundary conditions, wing thickness is represented by linearly varying source distributions whose strengths are proportional to the thickness slope while vortex distributions are used to take into account the combined effects of camber twist, and angle of attack For linearly varying vortex distributions the Kutta condition is satisfied automatically while wings having supersonic trailing edges require an ad ditional vortex distribution and an additional control point on the last panel of each streamwise strip When the constant vortex option is used no extra equation is required for supersonic trailing edges, while subsonic trailing edges are solved using a linearly varying vortex distribution on the last panel

Supersonic wings usually can be characterized as thin and mildly cambered when the values of thickness and camber are compared with the local chord, so that the use of planar panels with linearized boundary conditions yields acceptable results for most isolated wings Evolution of current design into blended wing body configurations stresses the need of a method capable of applying the exact boundary conditions at the actual wing surface even at supersonic speeds

Although a supersonic nonplanar boundary condition using triplet singularities has been developed already [10] the numerical complexity involved (the triplets can be used only for panels with supersonic leading edges) and a certain waviness in the solution makes a different approach ad visable This led to the formulation of a new boundary condition called nonplanar boundary condition 2 In this formulation the singularity panels are located on the *mean* surface of the wing but the control points are located on the *external* surface of the wing At high supersonic Mach numbers some correction is necessary to prevent the control points from falling upstream of the Mach lines emanating from the edges of associated singularity panel As a simple fix, the edges of the singularity panels are shifted upstream by a distance equal to the product of the Mach angle times the local thickness

Since each ' mean" panel has both source and vortex distributions the total number of unknown singularity strengths equals the number of panel control points Figure 4 presents a schematic comparison of the three options available for the present method For a thin wing of arbitrary twist and camber the two nonplanar boundary condition options become exactly equivalent but for a thick wing the distribution of singularity panels on a single surface featured by the new nonplanar option prevents any undesirable propagation and reflection of interior Mach waves

Figure 5 gives the pressure distribution calculated for a mildly cambered and twisted arrow wing having 70 deg sweepback and a 3% biconvex section (wing 2 of Ref 20) The results obtained using both the planar and nonplanar boundary conditions at $M_\infty = 2\ 05$ and $\alpha = 4$ deg are com pared with experimental data from Ref 21

Results are smooth and generally in acceptable agreement with experimental data As could be expected from the moderate thickness (3%) and the type of configuration (isolated wing) computed pressure distributions show no significant sensitivity to the particular type of boundary condition employed

Calculation of Pressures, Forces, and Moments

The pressure coefficient at each control point of the body is calculated from the resultant velocity using the exact isen tropic formula

$$C_p = \frac{2}{\gamma M^2}\left\{\left[1+\frac{\gamma-1}{2}M^2\left(\frac{1-V^2}{V_\infty^2}\right)\right]^{\frac{\gamma}{\gamma-1}}-1\right\} \qquad (2)$$

Although the linear formula for the pressure coefficient is more consistent on the wing when the planar boundary condition is used, a closer agreement between computed pressures with experimental data especially on the pressure side, has been observed using Eq (2) For this reason the isentropic formula is used in the present method also for wings regardless of the type of boundary condition

The forces and moments acting on the configuration are then computed by numerical integration assuming a constant value of the C_p for each panel When the planar boundary condition or the nonplanar boundary condition 2 is used the integration is carried out on the external panels

Computational Considerations

The amount of computing time required to solve a single combination of Mach number and angle of attack for a configuration basically depends on the type of the con figuration, the total number of panels, and the particular value of the Mach number In calculations of multiple angles of attack at the same Mach number, significant time savings are achieved because the matrix influence is computed only once For the applications presented here computer CPU times range from about 3 min for the wing body of Sec IV to about 95 min for the complete canard wing body combination of Sec VII

The numerical program presented herein is written in ANS FORTRAN 77 for the MBB VAX 11/780 computer system

A postprocessor code is used to provide plots of the paneled configuration and the computed pressure via Tektronix 4014 1 terminal with hardcopy or a Tektronix 4663 plotter

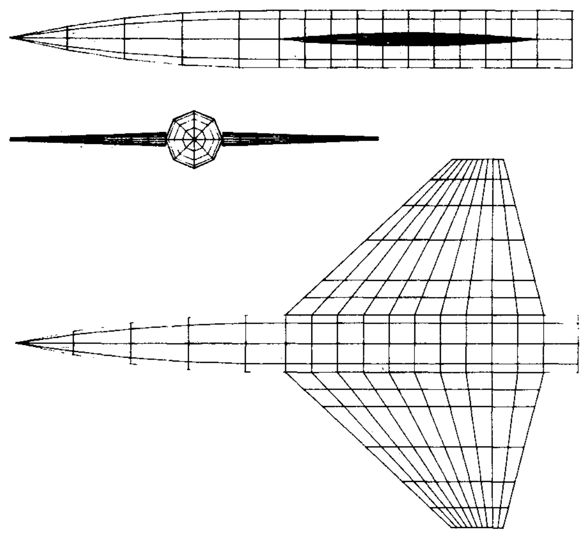

Fig 9 Hedman wing body configuration paneling

III AGARD Model B Analysis

A preliminary evaluation of the present method was carried out on a delta wing combination, the AGARD model B Figure 6 shows the scheme used for paneling the axisymmetric body and the wing having a 2% thick parabolic section

Results from a former version of USSAERO[22] proved to be disappointing at supersonic speeds due to the appearance of wiggles on wing surface pressures while calculations with the present method showed fairly smooth pressures Much of the improvement is due to the replacement of body sources by triplets as it is evident from Fig 7, where comparisons of pressures for a simplified configuration with a thin wing are presented at Mach 1 2 and nearly zero angle of attack.

In order to provide comparisons with the experimental data published in Ref 23 calculations were carried out for a wide range of supersonic Mach numbers and angles of attack using the nonplanar boundary condition 2

Results presented in Fig 8 show an excellent overall agreement with experimental longitudinal characteristics Zero drag values (Fig 8c) have been compared using the experimental data not including base pressure and correcting the computed values C_D for skin friction drag C_{Df} The estimate of the friction drag has been done assuming fully turbulent flow on a polished model

IV Hedman Wing-Body Configuration Analysis

The improvements found in the analysis of the AGARD model B suggest reanalyzing another wing body con figuration, for which previous results[22] showed unsatisfactory correlation with experimental pressure measurements The Hedman configuration consists of a trapezoidal wing with a

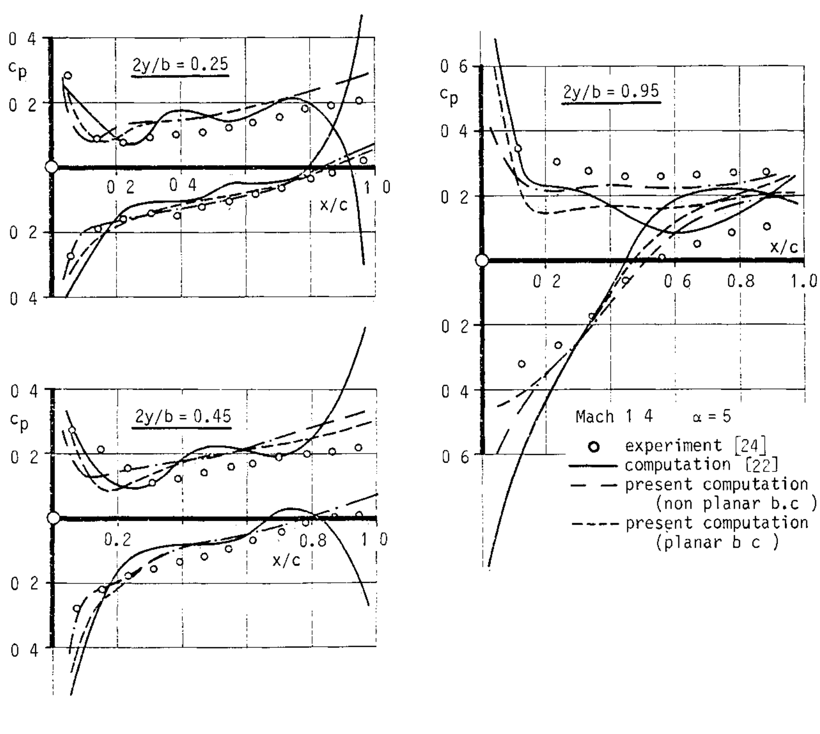

Fig 10 Wing pressure comparisons on Hedman wing body

5% thick uncambered circular arc section mounted on a paraboloid cylinder body In order to retain consistency with previous calculations this analysis used the same early paneling scheme, Fig 9

Here again significant improvements were obtained, as shown in Fig 10 where wing surface pressure comparisons between the former and the updated version of the code and the experimental data of Ref 24 are presented for three spanwise sections These comparisons show results from the present method using both boundary conditions to be in better agreement with experiments At the tip section where all theoretical data underestimate the experimental load, the nonlinear boundary condition option gives significantly better results that the planar one

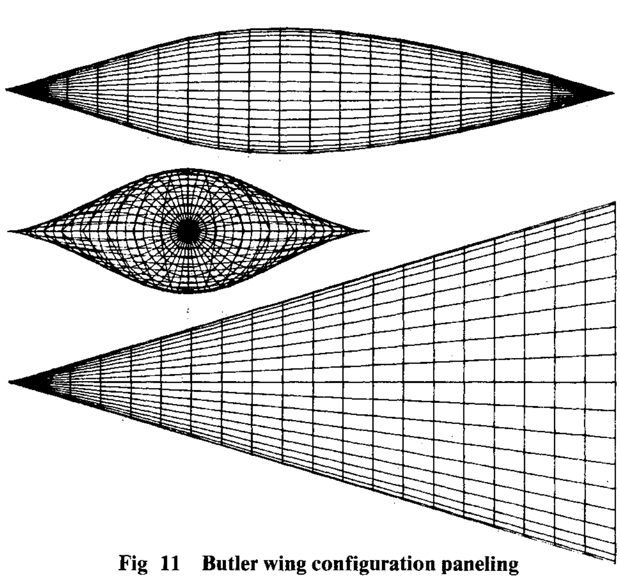

Fig 11 Butler wing configuration paneling

V Butler Wing Configuration Analysis

Another application was the analysis of a configuration originally proposed by Butler[25] to test hypersonic slender body solutions More recently, the same configuration was included by the AGARD FDP Working Group 07 in a series of test cases for steady inviscid transonic/supersonic flow

The planform of this configuration is an isosceles triangle For the first 20% of its length the configuration is a circular cone, while the remainder has elliptic sections which become more eccentric as the sharp trailing edge is approached A body like scheme was used for the paneling, see Fig 11

Availability of results from a finite difference method (FDM) solving Euler's equations by a marching procedure[26] suggested running the same flow conditions, zero angle of attack at Mach number 2 5 and evaluating the usefulness of simulating nonlinear effects in the panel code by using a compressibility factor which depends on local Mach number

Comparisons of pressure data between results from the FDM and the present method, using compressibility factors depending on local or freestream Mach number are presented in Figs 12 and 13 On the body surface Fig 12 longitudinal pressure distributions show a good correlation at the sym metry plane section ($\phi = 90$ deg) but the evolution of the pressure along the leading edge ($\phi = 0$ deg) predicted by the FDM code is not repeated in the panel solutions Analysis of the circumferential pressures, however shows the existence of a high pressure gradient in the leading edge region, so that the major discrepancies are concentrated in a very small region The local Mach solution shows better agreement with FDM results in the conical part of the configuration but tends to repeat the constant Mach solution results as the trailing edge is approached A more significant improvement is produced in the external flowfield characteristics The comparisons of Fig 13 indicate that although a correct representation of shock waves is clearly out of the realm of the linearized panel methods a better simulation of the shock contour is obtained

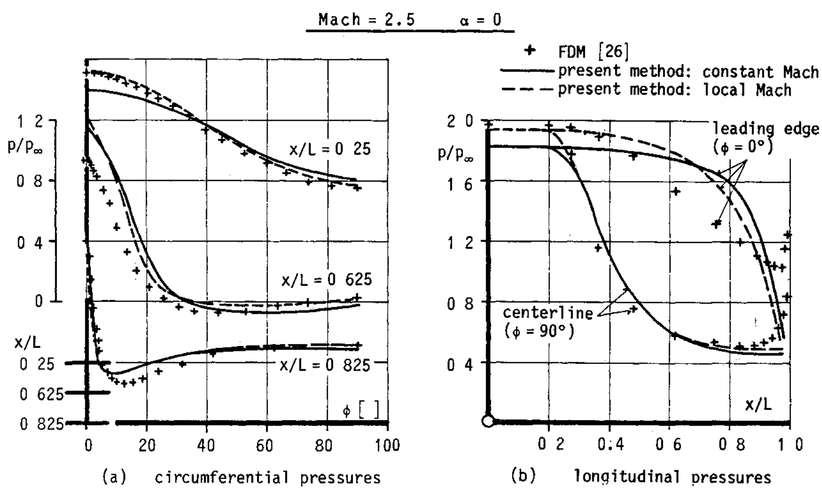

Fig 12 Surface pressure comparisons on Butler wing

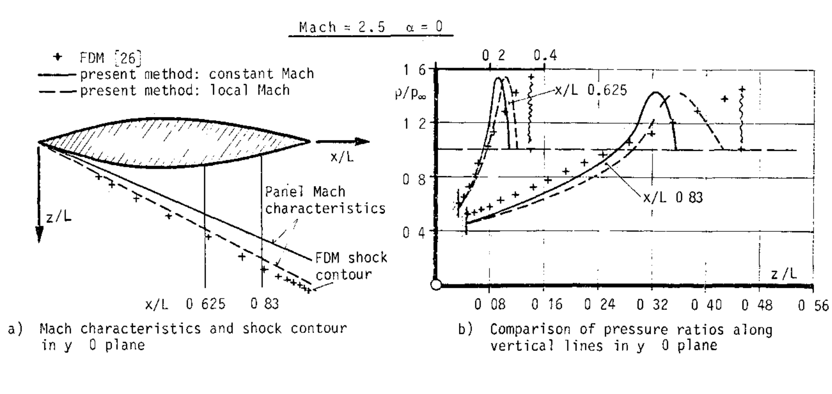

Fig 13 Flowfield pressure comparisons on Butler wing

when local Mach number propagation effects are taken into account

The importance of simulating nonlinear effects in the supersonic analysis of weapon separation characteristics has been discussed in detail in Ref 15, where a simple method for improving correlation between experimental data and off body pressures predicted by a constant Mach number solution has been presented There computed off body pressures were shifted upstream by the difference between the location in the off body survey plane of Mach lines determined by the freestream and local Mach number coming from one single body point

The present local Mach number solution, by which each panel of the configuration propagates disturbances along the local Mach lines must be regarded as a better approximation of nonlinear effects

VI Supersonic Cruise Airplane Configuration Analysis

The present method also was applied to a supersonic configuration designed and tested by NASA [27,28] The con figuration features a long slender fuselage of triangular cross section and a highly swept modified arrow wing whose camber twist and thickness distributions were designed for good cruise performance at the design Mach number 3 0 The original configuration also featured two engine nacelles and wing mounted vertical fins which were not included in this analysis

Figure 14 presents the paneling scheme used for this configuration The geometry of the panels was extracted from Ref 28 Fuselage data had to be manipulated to obtain cross sections which all have the same number of points, as required by the use of triplets Moreover, the original paneling modeled the wing body intersection defining a root plane parallel to the body axis and treating the geometry inboard of this plane as part of the fuselage definition see Fig 15 This also implies that the interior of the root section must be paneled to close the vortex rings of the fuselage triplets

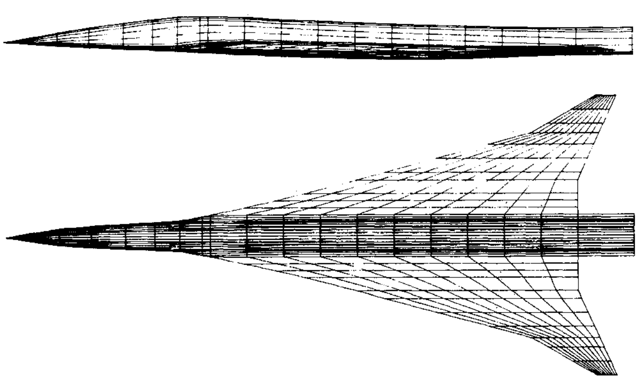

Fig 14 Supersonic cruise airplane configuration paneling

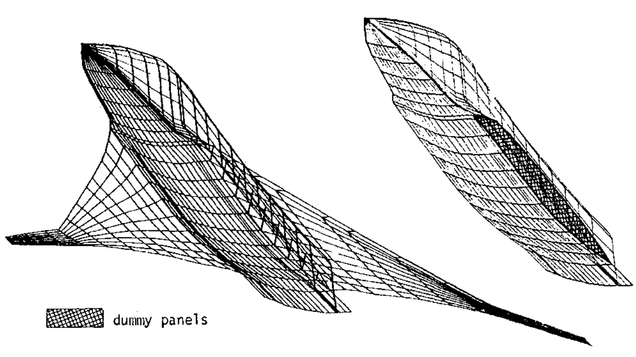

Fig 15 Application of the dummy paneling option to the supersonic cruise airplane

Enforcement of boundary conditions on these panels resulted in unrealistic surface pressures To overcome the problem a so called dummy paneling option was developed, by which panels marked as ' dummy'' are used only to close vortex rings of triplet singularities associated with neighboring panels but for which no boundary condition is prescribed

Figure 16 presents comparisons of longitudinal charac teristics at the design Mach number between results from the present method and experimental data published in Ref 27 The overall agreement is very good, except for the minimum drag value For the drag polar comparison, theoretical data were corrected for skin friction and transition strip effects taking the estimate used in Ref 14 where a pilot code of the higher order panel method was applied to this configuration

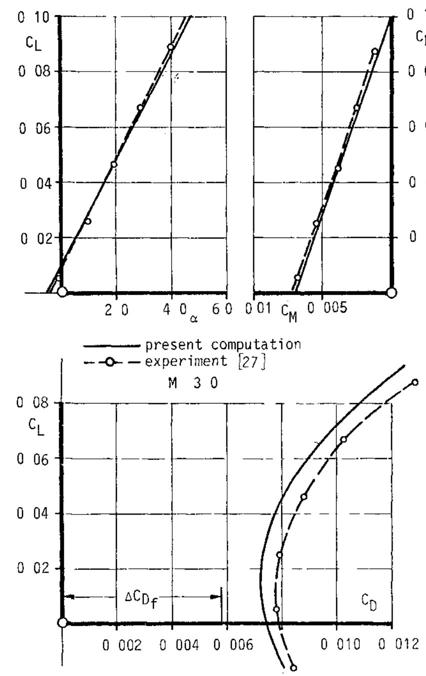

Fig 16 Longitudinal characteristic comparisons on the supersonic cruise airplane

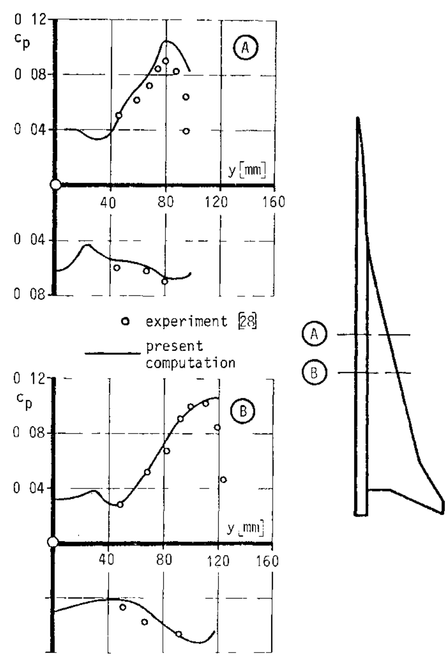

Fig 17 Wing pressure comparisons on the supersonic cruise air plane

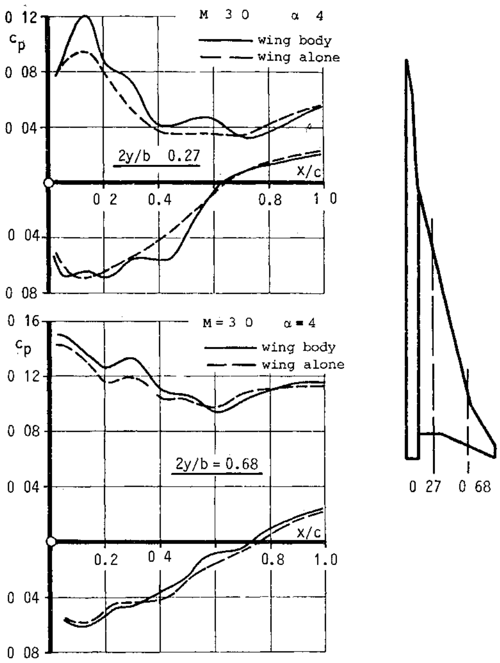

Fig 18 Effect of body on wing pressure distributions for the supersonic cruise airplane

In Fig. 17 surface pressure results are compared with wing pressure measurements from Ref 28 along two sections normal to the body axis at Mach = 3 0 $\alpha = 4$ deg Computed pressures are in good agreement with measurements except at the leading edge where the local load is overestimated An analogous discrepancy also affected the PAN AIR results of Ref 14, where the failure of the theoretical predictions were referred to the existence of nonlinear disturbances along the leading edge

Application of the new nonlinear boundary condition resulted in surface pressures on fuselage and wing which fair smoothly

Looking at the wing surface pressures however some waviness is apparent along the streamwise direction Fig 18 Calculations of the isolated wing revealed an adverse body effect, which will be discussed in Sec VII

VII Wing-Body-Canard Configuration Analysis

Another example of applying the present method was the analysis of a wing body canard configuration representative of typical advanced tactical fighter airplane (Fig 19) This configuration features a cranked wing, a fuselage with a canopy and a two dimensional belly intake, and a trapezoidal canard The two side fuselage mounted vertical fins of the original configuration were not considered in the present analysis

Configuration Modeling

In order to study the paneling effects for this complex configuration three different paneling schemes were defined Fig 20

The first scheme, Fig 20a is the closest representation of the configuration Only minor geometrical simplifications were used to comply with the paneling restrictions outlined in Sec II These simplifications include some corrections in the afterbody shape and the removal of the boundary layer diverter between the intake and the fuselage Both wing and canard have a "thick" representation allowed by use of the nonplanar boundary condition 2 The "dummy" paneling option was used for the fuselage panels whose control points lie in the interior of the wing and the canard root A major problem was found in the constraint to model the intake without splitting up the body into two segments A simple solution was to close the intake with a fairing so that all cross sections of the body have the same number of panels Preventing the flow from entering the intake however in duces a locally large distortion of the flow due to the strong compression on the fairing followed by a high suction on the external cowl of the actual intake as shown in Fig 21 As a

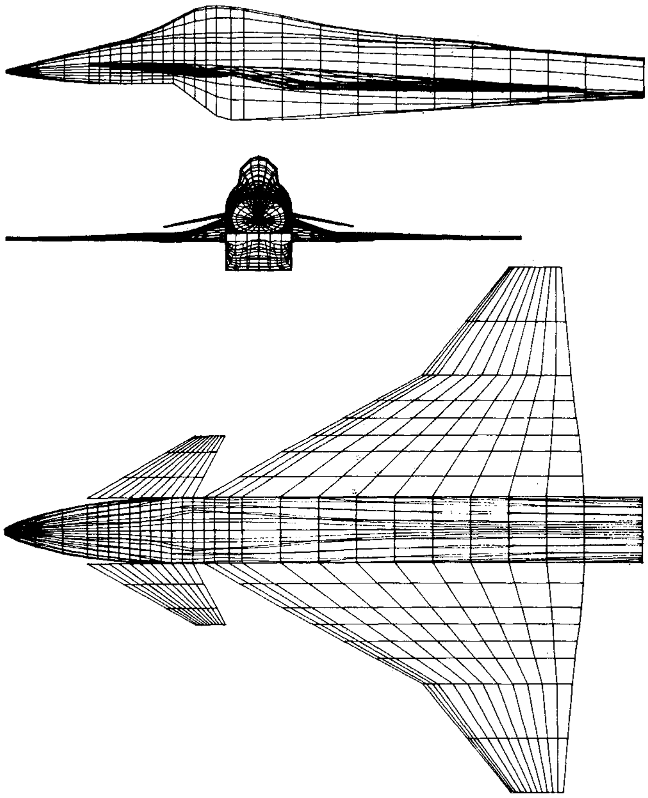

Fig 19 Wing body canard configuration basic paneling scheme

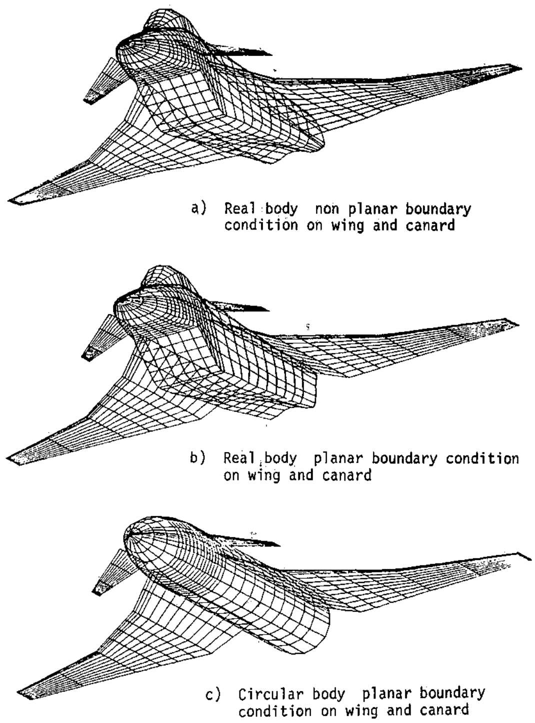

Fig 20 Comparison of different paneling schemes of the wing body canard configuration

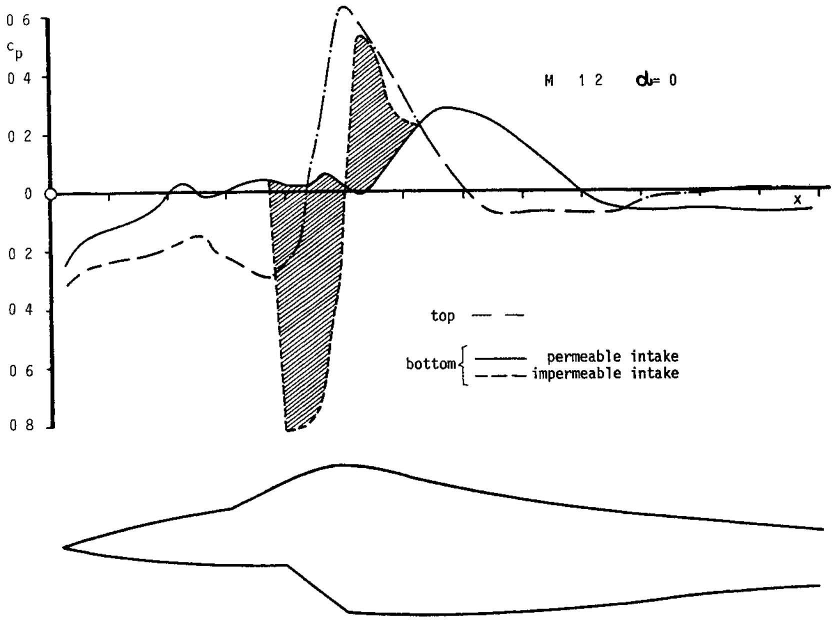

Fig 21 Isolated body pressures on the wing body canard configuration

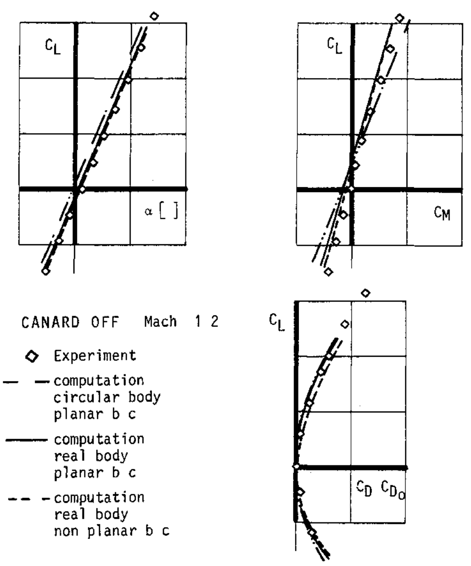

Fig 22 Longitudinal characteristic comparisons on the wing body canard configuration

compromise, a "permeable" fairing was used to control the inlet flow by specifying a nonzero value for the total normal velocity at the fairing panels, thus obtaining a more realistic simulation of the intake flow (Fig 21)

Since the presence of the canopy and the intake results in a complicated evolution of body cross sections in the crucial area of interference between the canard and the wing a simpler paneling was used to evaluate the effects due to configuration modeling In this simplified representation of the configuration, an equivalent body of revolution replaces the "real" fuselage and planar boundary conditions are used for the wing and canard (Fig 20c)

Finally, to separate effects induced by the body shapes and the boundary condition type a third paneling was defined characterized by a real fuselage and planarized wing canard representations (Fig 20b) Due to the rather high incidence of the wing root section the paneling of the fuselage side had to be modified in order to avoid close proximity between wing edge vortices and fuselage control points

The vortex wakes shed from the lifting surfaces are automatically modeled in a plane extending downstream of the trailing edge along the relevant camber surface For all of the present configurations this implies that the canard wake is located just above the wing a position appropriate for angles of attack close to 0 deg

Computed Results

Initial calculations were carried out at Mach 1 2 for the canard off configuration Comparisons of the computed longitudinal characteristics and experimental data obtained by a MBB force model show a reasonable agreement overall Fig 22

A correct representation of the fuselage gives better predictions for the lift value at zero angle of attack and stability characteristics than the axisymmetric body and use of the nonplanar boundary condition seems to give the best estimate for the lift dependent drag polar

Unfortunately further calculations carried out for the complete configuration revealed that the quality of these comparisons was fortuitous In addition there were some indications that wing pressures were too sensitive to the paneling of the configuration, Fig 23 As should be expected, rather large body effects are predicted on wing pressures but the pressure spikes produced at isolated points of the two inboard sections are symptomatic of numerical troubles The analysis of the flowfield characteristics induced by the isolated "real" fuselage revealed the same type of numerical instabilities, which, furthermore, are very sensitive to small changes in the freestream Mach number (Fig 24)

The problem has been referred to the jump of the per turbation velocity across the Mach lines originating along the panel edges Normally contributions from two adjacent panels tend to cancel each other but for adjacent panels which have a large variation in geometrical slope or do not have contiguous edges this jump is not attenuated sufficiently to prevent induction of spurious perturbations on control points close to the characteristic lines

In retrospect, the paneling of the real body of Fig 21 showed the existence of these undesirable features, especially in the canopy intake region, where the use of quadrilateral flat panels implies geometrical discontinuities on the edges The directional property of the triplets (Fig 1) does not allow the propagation of these jumps inside the fuselage, but cannot prevent their propagation in the external flowfield Indeed strong numerical instabilities were produced on the canard surface in the analysis of the complete configuration

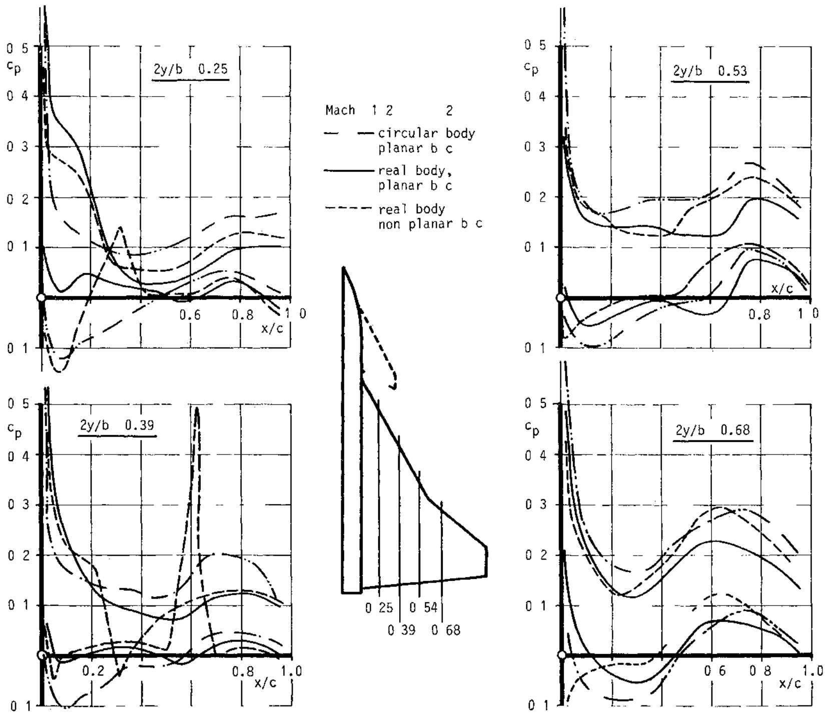

Fig 23 Effect of paneling type on wing pressure distributions

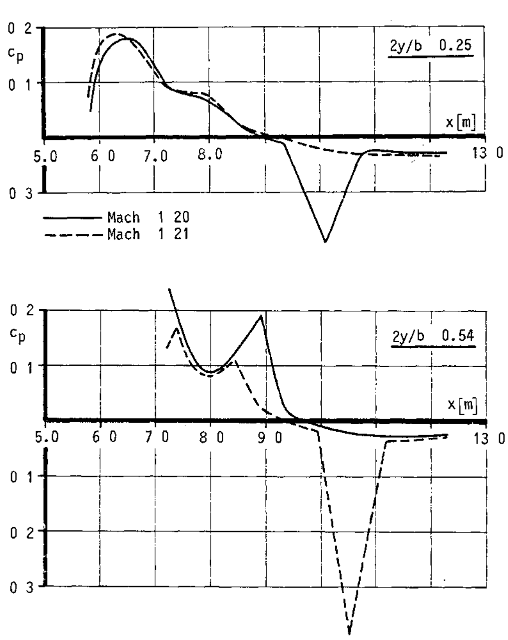

Fig 24 Effect of Mach number on off body pressure

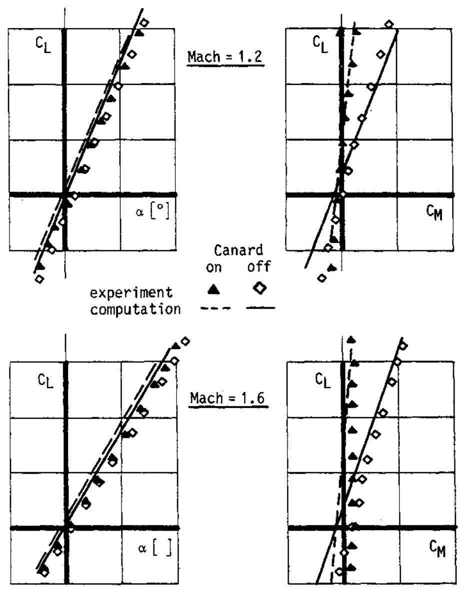

Fig 25 Longitudinal characteristics comparisons on the wing body canard configuration

Calculations carried out with the simplified configuration on the contrary produced good results The effects induced by the presence of the canard on the longitudinal charac teristics are in good agreement with the experimental data (Fig 25) Wing pressures are smooth for both canard off and canard on configurations (Fig 26)

Although local refinements to the paneling scheme could be attempted it does not seem possible to find panel layouts avoiding numerical instabilities while adequately representing this type of geometry

A similar problem already has been overcome in the higher order panel method by subdividing each panel into eight triangular subpanels in such a way that all panel edges are contiguous with adjacent panels [12,13] If feasible, the same approach will be pursued in the present method

VIII Conclusions

A modified version of the subsonic supersonic panel method of Woodward has been developed and applied to the analysis of a variety of sample configurations in supersonic flow Problems associated with propagation of Mach waves in the interior of bodies and thick wings appear to be eliminated by use of the triplet singularity on the bodies and the development of a new type of nonplanar boundary condition for the wings As a result accurate prediction of longitudinal characteristics and surface pressures was obtained for most of the test cases Further, the low computing cost associated with the use of lower order singularity distributions makes the present method attractive even for applications where in clusion of nonlinear effects requiring iterative solutions is necessary (e g flowfield calculations for analysis of weapon

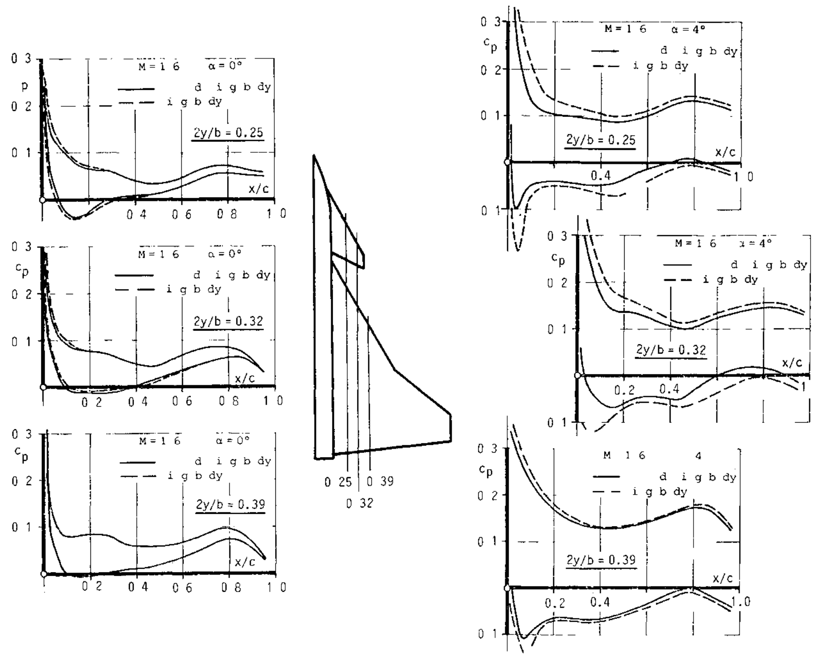

Fig 26 Effect of canard on wing pressure dis tributions

separation characteristics) However lack of geometrical continuity on the surface of a very complicated fuselage shape has been observed to generate spurious Mach waves in the external flowfield which affect the numerical solution Continued efforts are expected to overcome this problem in the near future

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