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INVITED LECTURE:

Theoretical Knowledge Base for Accelerated Transonic Design

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Abstract:
Based on the classical gasdynamic models for transonic flow, airfoil and 3D configuration design tools are developed for fast predesign studies. Singularities, sonic and shock surfaces are seen as part of the initial geometries in inverse design studies. For direct approach geometry preprocessors are developed and visualization postprocessors are adapted to control the designed and expected mechanisms. Creating a knowledge base for future aerodynamic design expert systems is seen as a goal of this effort combining theoretical aerodynamics and modern software technology.

1. Introduction

This paper is intended to illustrate a fragment of developments toward systematic high speed design, that is here aerodynamics in the regime of transonic and supersonic Mach numbers. The purpose is to show the role of a combination of gasdynamics and geometry in the development of modern software for aerodynamic design in the virtual environment of personal and workstation computers. Here it is not intended to once more derive the basics for faster numerical simulation (CFD): rather some simplified models of the basic equations are briefly mentioned because they paved the way to a better understanding of local flow phenomena, or as a consequence, of the requirements for a local aerodynamic contour shaping in order to control local inviscid flow phenomena. In the transonic regime, these phenomena are dominated by the interaction of surface geometry and surfaces within the flow field, for instance the boundary between locally subsonic and locally supersonic flow. These sonic surfaces, but also shock wave surfaces may be seen as part of the complete geometry set consisting of configuration and important flow features under design conditions. Motivation of this contribution is therefore to explain the gasdynamic background for some practical geometry tools for aerodynamic design, which take into account sonic and shock surfaces as part of the boundary conditions. Building on the pioneering basics of Guderley [1] and Oswatitsch [2], many of these ideas underlying the outlined concepts have been developed within the author's past theoretical work in transonics at DLR in Göttingen. Collaboration projects with Universities of Arizona and Colorado helped to focus this work on practical airframe and turbomachinery design problems.

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2. Gasdynamic modelling

In a time long before the arrival of the digital computer, the model equations for compressible flow were derived. Since then, we know the Reynolds-averaged Navier-Stokes for the full problem, the Euler equations for their inviscid simplification and the Potential equation for a further simplification to isoenergetic flows. The latter extended the classical knowledge base of hydrodynamics into the compressible flow regime. A necessity to find solutions to these equations then led to several attempts to transform them, for instance to reduce the formidable difficulties stemming from the nonlinearity of the potential equation. In 2D flow, the hodograph transformation leads to an inversion of the problem, trading linear equations for nonlinear boundary conditions. Several mathematical methods were developed to create the first transonic airfoils. Elegant problem formulations could not hide the fact that solving mapped counterparts of real world problems never became very popular with the aerospace design engineer. Nevertheless, in a time when usually only numerical discretization of complex problems is seen as the way to get deeper insight into flow problems, some of these mapped model equations still have some value. One form of the "near sonic" model equations was found particularly useful, because it not only gave a number of flow models for transonic phenomena in closed analytical form but also led the way toward design principles for practical airfoils and wings.

2.1 Local flow quality and singularity models

Near sonic Beltrami equations

The potential equation for a near-sonic plane or axisymmetric flow has a particularly elegant formulation in variables of state or in characteristic variables (for details see Ref [3]), illustrating the formal relationship between in-

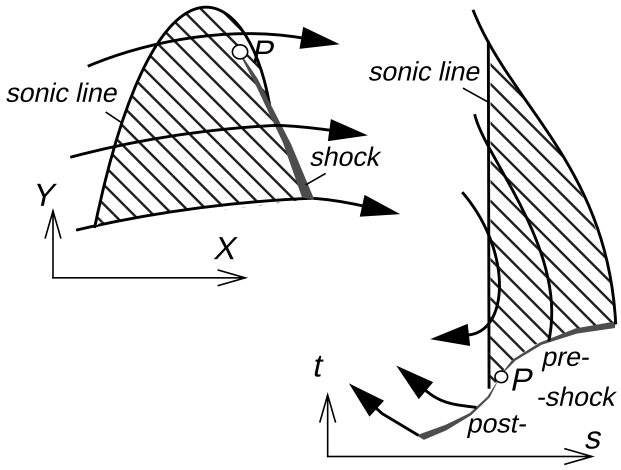

Fig. 1: A 2D flow element in (X, Y) physical space and its mapping to an analog flow in hodograph variables, (s,t Rheograph plane); Example: local supersonic flow field with shock formation (P) in the flow.

compressible, transonic, plane and axially symmetric potential problems: characteristic equation and compatibility relation define a system of quasi-linear first order differential equations (Beltrami equations):

$$\begin{aligned} V_t &= Y^{p_1} U_s \\ V_s &= j Y^{p_1} U_t \end{aligned} \qquad (1a)$$

$$\begin{aligned} X_s &= U^{p_2} Y_t \\ X_t &= j U^{p_2} Y_s \end{aligned} \qquad (1b)$$

Here the variables (X, Y) denote the physical space and (U, V) a normalized pair of velocity variables, namely the Prandtl-Meyer function and the flow angle. Both pairs are dependent variables in a workspace (s, t) "Rheograph plane" which is identical with characteristic variables in supersonic flow (j=1, U>0) and their analytical continuation beyond the sonic line where j=-1, U < 0. Exponents p_1 and p_2 have a switch function: p_1 =0 denotes plane 2D flow, p_1 = 1 indicates axially symmetric flow; p_2 = 0 results in a simple mapping of linear subsonic or supersonic flow while p_2 = 1/3 switches to transonic flow. These equations include most of the flow models described by the pioneers in theoretical transonics, in closed analytical form or - for transonic axisymmetric flow, where a weak nonlinearity persists - in a numerically very suitable form.

Equations (1) describe a large number of educational solutions which should be kept 'alive' as part of the knowledge base for transonic design and phenomena analysis. Because of (1) representing a system for quasi-conformal or characteristic mapping, solutions found may be interpreted as transformations of geometries consisting of both the boundary conditions and all details of the flow, appearing as an analog flow in the rheograph plane.

Analog flow example: Singular shock wave behavior

Shock waves occur in transonic and supersonic flows. They pose interesting mapping problems, like the one resulting from a recompression shock terminating a local supersonic flow pattern, illustrated here (Fig. 1) qualitatively. Onother example is the singular behavior of incipient shock attachment to a wedge in low supersonic Mach number. It may illustrate the value of these equations without any computation and solely by applying the very common knowledge base of local conformal mapping functions in the subsonic part (U<0, j=-1) of the (s,t) plane:

The question is asked, how does a shockwave, detached from a slender wedge tip in near sonic flow $M_\infty > 1$ change its shape when it attaches to the wedge tip because M_∞ is increased? In hodograph formulation and its special working plane "Rheograph" (s,t) the problem is posed by the mapped shock shape (the shock polar) and prescribed directions of the post-shock streamlines emanating from it (Fig. 2). This shock polar near the maximum deflection location is represented by a curved arc, while the wedge is mapped by a simple horizontal line. The gasdynamic problem of investigating the flow field near a shock wave detachment is transformed here into a conformal mapping problem: the curved shock wave crest ("maximum deflection point" *A*) meets a horizontal bar as mapped flow boundary, while the stream lines there have to leave the shock polar under 45 degrees. This geometric conflict will yield a singularity when the shock polar touches the bar, i. e. when the flow attaches to the wedge. A simple series of conformal mappings of the vicinity of this curved angle sector space to a regularized domain results in a weak logarithmic singularity which also yields the pressure distribution on the wedge near the tip:

$$c_p = c_{p,\,Att} - \frac{const}{ln(x)} + \ldots.$$

Similar purely analytical results are obtained for the attached shock wave for further increased M_∞ before the post - shock flow is completely supersonic, for a more detailed description of this application of classical mapping techniques see [4]. These new solutions complement well-known singularities describing other features of shocks in the transonic regime, like the Oswatitsch-Zierep singularity of a normal shock on a curved wall [5] and the present author's analysis of a detached shock wave vanishing in an airfoil flow with M_∞ approaching unity, [6]. All these singular points in 2D flows with recompression shocks can be found by local conformal mapping in the rheograph plane.

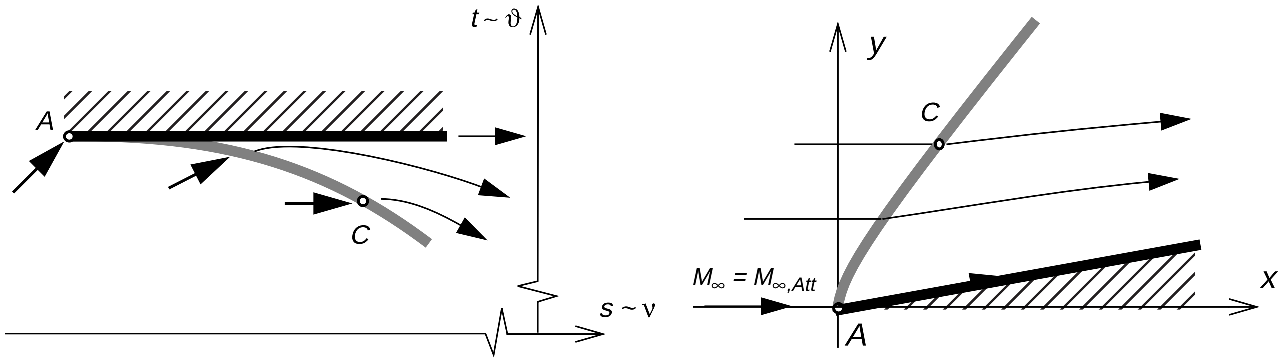

Figure 2: Understanding shock attachment / detachment to a wedge: Hodograph mapping and shock relations transform problem to an analog incompressible flow detail or simple conformal mapping case: Wedge tip and shock wave interval between maximum deflection point A and Crocco's point C.

2.2 Flow model construction from given sonic locus or shock

In the rheograph plane (Fig. 1) subsonic flow is separated from supersonic flow by the sonic locus, where U(s,t) = 0. Solving the model equations (1) for a transonic problem, where both types of flow (U < 0, U > 0) occur, therefore requires applying separate solution methods for both parts, with the need to have common data at the line U(s,t) = 0. For plane ($p_1 = 0$) and transonic flow ($p_2 = 1/3$), solution to (1a) is the first step, because it is decoupled from (1b). Choosing U = s, V = t is just the simplest solution, without loosing generality of subsequently finding solutions X = X(s,t), Y(s,t) to the linear system (1b) in a second step. Creating a solution to both the subsonic part and the supersonic part now requires prescribing data X*(t), Y*(t) along s = 0 and using them as part of boundary conditions to an elliptic problem in the subsonic domain (s<0) and as initial data for a hyperbolic problem in the supersonic domain (s > 0). Two decades ago, techniques were developed to design transonic airfoils using this concept [7]. A rheoelectric 'analog computer' provided a very educational tool to understand also the background of Garabedian's method of complex characteristics to design shock-free airfoils [8], which was a mathematically elegant method but did not provide a lasting engineering knowledge base because of its complexity.

From 'Fictitious Gas' to a flow control design concept

The next step was to extend the above-mentioned concept of analytical continuation across the sonic line into 3D space [9] which led the way toward an airfoil and wing design concept called "Fictitious Gas" (FG) method [10, 11, 12, 13, 14]. Having learned from the hodograph boundary and initial value problems, CFD numerical simulation invited to stay in physical space, provide a suitable subsonic solution including a sonic line or surface and subsequently perform a numerical marching derived from the method of characteristics discussed below. Any existing potential flow code could be used for applying the FG concept, any input airfoil could be made shock-free within certain limits of given free-stream Mach number and lift. Recently the concept was applied also to the Euler [15] and Navier Stokes [16] equations, which opened "realistic" thermodynamic interpretations of the involved "fictitious" mathematical manipulations: In potential flow, a fictitious relation between density and velocity has been set up to replace the isentropic relation. Tracing back this manipulation to the Euler equations resulted in a modified equation of state with temperature controlled by local velocity:

$$p/\rho = R(T+\Delta T(q/a^*))$$

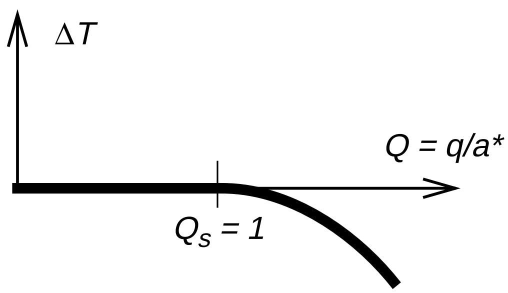

Figure 3: Modified equation of state with temperature controlled by local velocity beyond (sonic) threshold velocity

Application of this concept to supercritical flows ensures confinement of thermal control to local supercritical domains if threshold velocity Q_s is equal to critical sound speed. Global entropy production balance is equal to zero, ensuring global (wave) drag to vanish. A flow controlled this way is smooth and shock-free and it can be shown that the controlled domain is subsonic but with flow velocities greater than the critical speed of sound, a*. This results from a change to mathematically elliptic type while the original supersonic equations, like (1b), is hyperbolic. However, bringing such a flow to an experiment or even to an aircraft wing, this would require a technical solution for the modelled internal cooling and subsequent reheating within the flow domain.

Not excluding such a flow control concept for the future, we presently try to make only theoretical use by combining this concept with the previously mentioned method to compute a local supersonic flow domain from given sonic surface conditions: Only the subcritical flow domain (without flow control) bounded by most of the configuration is already computed but data along the sonic locus are used to march toward the contour (Fig.4, above) and replace the controlled flow domain by a ideal gas supersonic flow. Of practical interest is now the resulting new surface contour compared to the initial surface: Delicate changes of local curvature, rather than airfoil thickness changes, are the secret of gaining a new contour accomodating a shock-free flow. This will be outlined in an example further below.

Prescribing shock waves in supersonic flow

For supersonic applications, the same concept of integrating flow field solutions by starting at an initial surface has been successfully used [17], (Fig. 4, below): The sonic surface in 3D space is replaced here by a given shock bow wave geometry in unperturbed supersonic free-stream. Shock relations then prescribe the post-shock flow conditions which are the initial conditions for a numerical cross flow marching resulting in a domain of the computed flow field. Suitable selection of integrated streamlines restrict the used part of this result to the flow field between a body and its bow wave. In the most general approach, an arbitrary shock surface with local inclination variation toward free stream flow within the boundaries between M_∞ - characteristics and maximum oblique shock deflection is the initial surface for integrating the 3D Euler equations in an inverse mode: the body surface physically generating the pre-defined shock wave is the result of this inverse approach. A numerical method has been developed by Jones [18] and used for waverider configuration design. A restriction to arbitrary strength but axisymmetricc flows resulted in a simplification to a 2D non-isentropic method of characteristics developed by Qian [3] and a different restriction to constant 3D shock strength but arbitrary supersonic wing leading edges yields a quasi-conical flow concept ("Osculating Cones", [17]), which gave rise to a rapid 3D supersonic design method developed by Center [19].

2.3 Characteristics in 2D and in 3D flow

In supersonic flow, Mach waves or characteristics are appearing as straight orthogonal lines ξ = t+s, η = t-s in the s,t rheograph plane. Basic system (1), or it's full compressible potential flow extension, see [13], maps these families of lines to 2D curves in the physical plane, within the domain of supersonic local Mach numbers. Fig. 5 shows a 2D airfoil flow element with a chosen surface point C and both characteristics AC and CB intersecting it inclined to the flow direction with an angle α = arcsin(1/M): regions of influence from upstream and dependence to downstream are defined this way. For given flow data within the sonic line segment AB the solution for supersonic flow is completely determined within the triangle ABC. This is the basis for the 2D inverse method of characteristics allowing a replacement of the domain with local flow control by an isentropic supersonic domain.

In 3D flow, both families of characteristics here form Mach - conoids, as illustrated in Fig. 5 for a 3D wing ele-

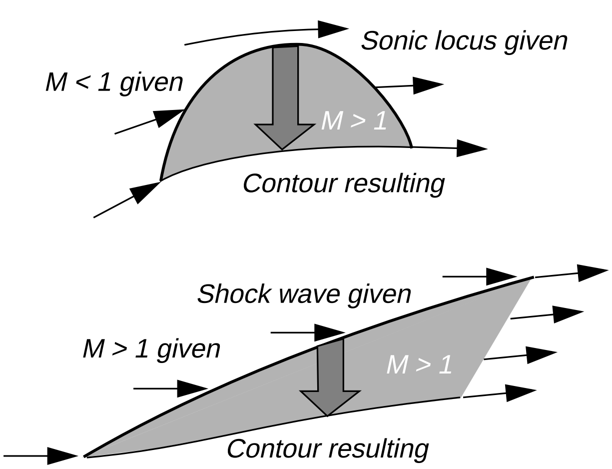

Figure 4: Inverse design concepts for local shape modifications in transonic (above) and in supersonic flow (below). Vertical arrows illustrate marching direction.

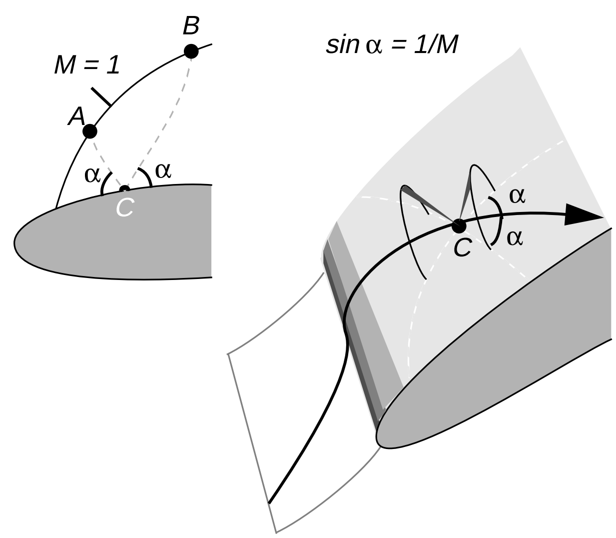

Figure 5: Characteristics (Mach waves) in 2D airfoil flow and on a 3D swept wing element.

ment. They may intersect the sonic surface far away from surface location C and will not form closed domains comparable to the 2D sonic line interval AB. Mathematically an ill-posed problem, the inverse integration of 3D local supersonic flow fields with the cross marching technique nevertheless gave very satisfactory results for transonic wings [11, 12] as well as for waverider wings [18, 19].

3. Geometric modelling

Besides trying to get the aerodynamicist familiar with special flow phenomena in the high speed regime and enabling him/her even using these for achieving desirable aerodynamic properties of configurations to be designed, it must of course also be the goal to develop software tools which contain this knowledge base as a black box.
It is therefore a major task to introduce gasdynamic modelling like the examples shown above suitably into the preprocessing software within a computerized design system. Such preprocessors are mainly geometry surface generators for the configuration to be designed, but also for discretizing the surrounding flow space and clustering numerical grid points near expected singularities, at expected shock surface locations and other immaterial boundaries of the free stream.

3.1 Airfoil Geometry

Airfoils always have been the basic elements for aircraft wings or turbomachinery blades. Examples like the following case study motivate the development of a flexible mathematical function describing 2D airfoils with a number of parameters as low as possible but selected and calibrated by the gasdynamic knowledge base, and as high as necessary to duplicate a variety of aerodynamically efficient known airfoils.

Example: Shock-free airfoil design parameters

Design requirements for a 17% thick supercritical airfoil in transonic flow $M_\infty = 0.71$ and a lift coefficient of $c_l = 0.6$ [20] gave rise for this case study using the outlined concept of a 'fictitious gas' or 'flow control' computation, starting with a baseline airfoil A (Fig. 6) given by a set of support points and splined in a blown-up scale. This has the advantage of reducing the curvature peak at the leading edge and at the same time emphasizes local curvature details at the airfoil upper surface where shock waves may be minimized by inverse design or via an optimization strategy. Here the method of characteristics is used to arrive at shock-free flow using numerical simulation results from an Euler solver with a modified equation of state as mentioned above. The resulting airfoil B is somewhat flattened, but it is the local curvature changes which count for transonic flow quality. Curvature distribution along airfoil

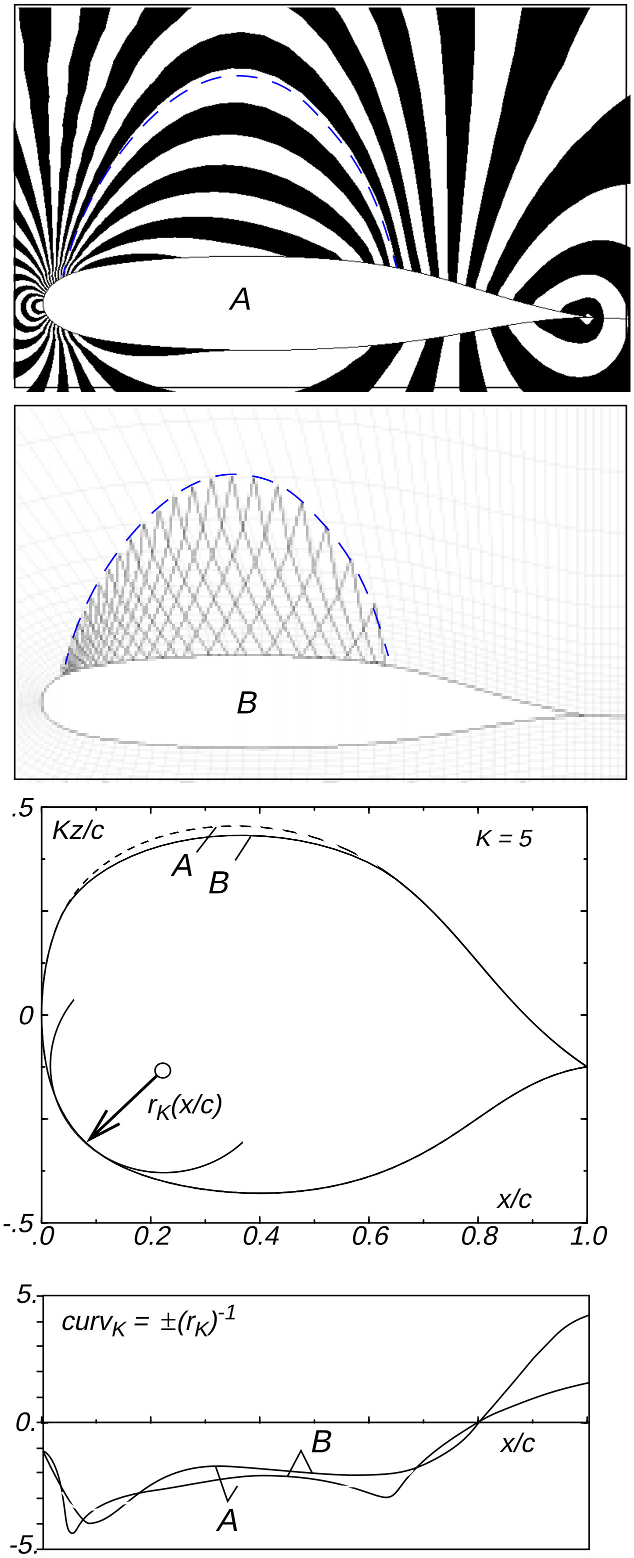

Figure 6: Baseline airfoil A and isobars in transonic flow $M_\infty = 0.707$ with fictitious gas or flow control within supercritical domain (above).
Redesigned airfoil B with shock-free flow, characteristics pattern; blown up (factor K) airfoils A and B.
Curvature distribution along chord x/c (below)

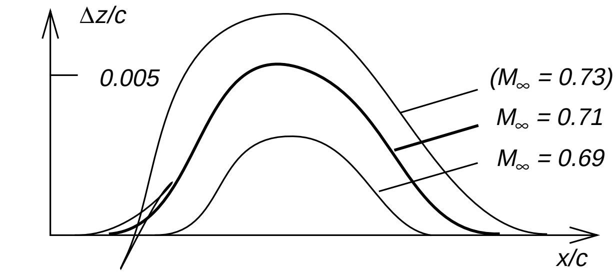

Figure 7: Surface modification bumps resulting from F. G. design method: smooth data for $M_\infty < 0.73$ indicating limit line occurrence at higher Mach numbers.

chord resulting from airfoil support points may not be very smooth if direct geometry cubic spline generation is used. A smoothing by two orders and at the same time desirable local curvature control may be obtained if the curvature itself is splined from support points and a dense data distribution is integrated using the natural equation of the curve

$$\sin\left(\operatorname{atan}\left(K \cdot \frac{dz}{dx}\right)\right) = 1 + \int_0 curv_K(x)dx$$

giving the slope angle of the airfoil curve which then can easily be composed of circle arc segments with local radii given.

3.2 Adaptive devices on airfoils and wings

Designing the airframe and its structure to mechanically adapt the flow boundary is a practical consequence of applying the theoretical aerodynamic knowledge base to numerically optimize an objective function like the lift-over-drag ratio. Shock-free design with its resulting subtracted surface bump (Fig. 6) sparked this concept early [21] when bringing it to reality was still very unlikely, today such technology is being studied seriously. Simple mathematical models for numerically found geometry changes, but also to model known control devices are of great help for the developments of adaptation technologies:

Surface bumps

Local geometry modifications like the one changing the illustrated airfoils A to B can be discribed by a smooth bump function

$$\Delta z = \Delta z_{max}(\sin(f(\xi))^{g(\xi)}$$

with ξ the suitably scaled interval of needed modifications along chord, the sine function ensuring closure, function f allowing for a non-symmetry correction and function g starting and ending with the value 3 ensuring a smooth ramp and values within the interval controlling bump crest curvature. Parameters for these functions are adjusted easily for a surface result from the F. G. method. These also allow an approximative treatment to arrive at airfoils for extreme design conditions where the theoretical surface modification is already found to be discontinuous with limit lines (see Fig. 7).

Airfoils with local, small bumps have been investigated experimentally [22] to control viscous interaction at the contour very locally at the footpoint of a recompression shock: wave drag is reduced with nearly no increase of viscous drag. Further refinement of this technology includes the use of smart materials to practically create bumps on wind tunnel models and flight test wings, controlled by software using mathematical models like the one given above.

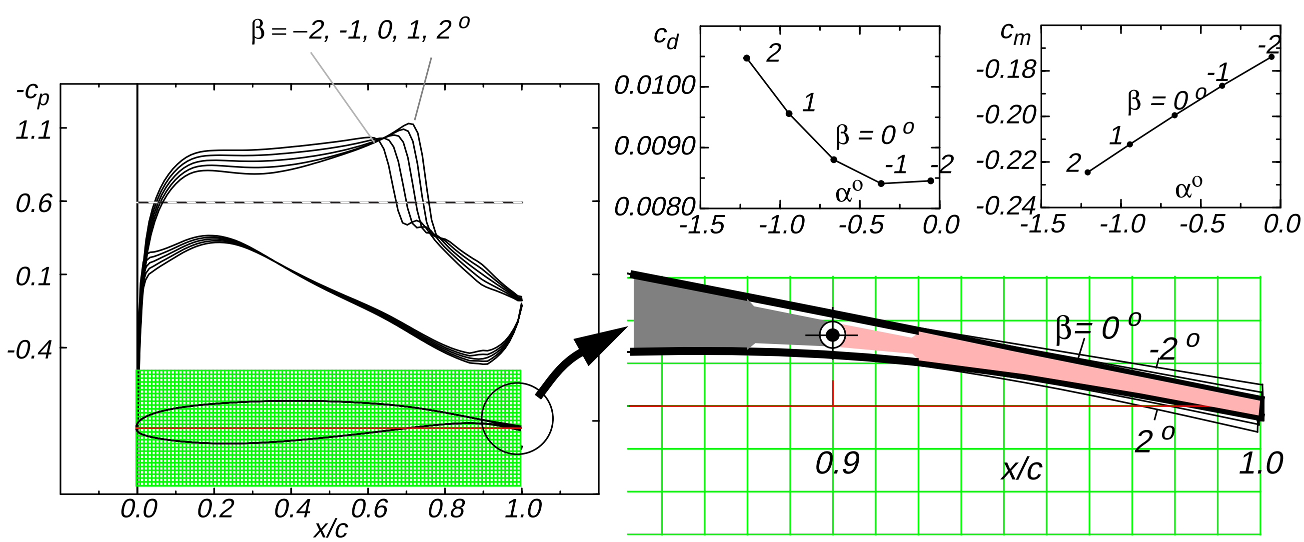

Figure 8: Variable camber sealed flap airfoil numerical simulation with MSES code, $M_\infty = 0.75$, Re = 40 Mill., $c_l = 0.7$;
Flap deflection β, drag and moment coefficients for constant lift at variable angle of attack α;
Geometric modelling of elastic flap sealing interval between rigid airfoil and rigid flap.

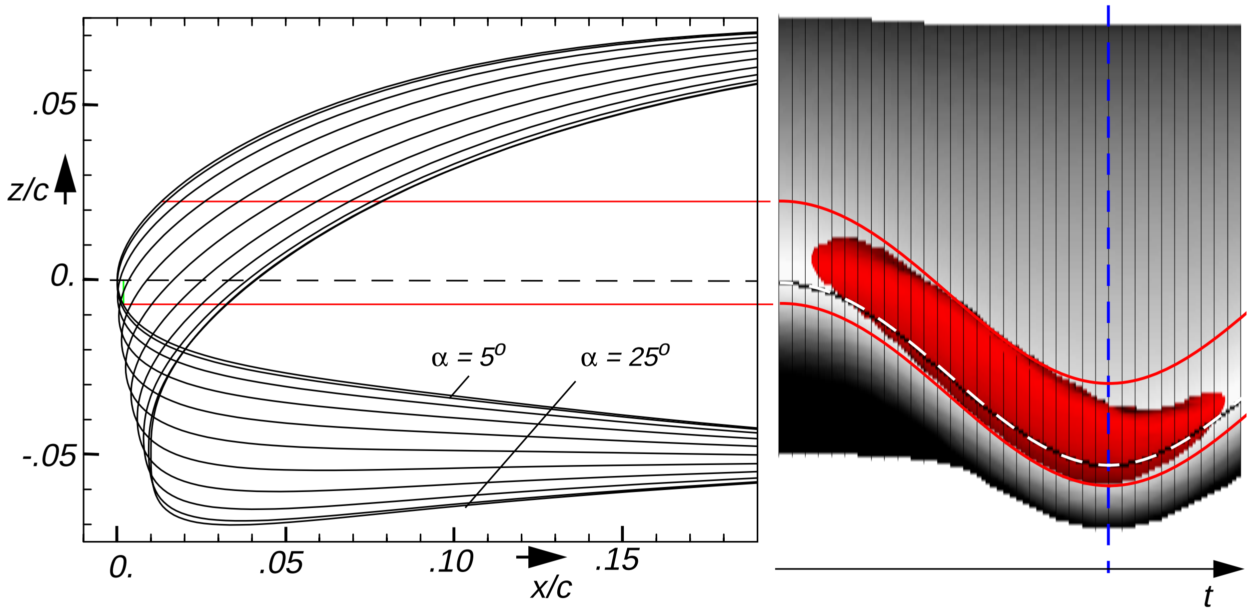

Figure 9: Cyclic nose droop motion of an airfoil during pitch α(t) in M_∞ = 0.3: Appearance of an unsteady supersonic bubble at the leading edge. Proposing additional local cyclic shape flattening at the leading edge should control shock-boundary layer interaction and hence downstream separation leading to dynamic stall.

Variable camber concept: Sealed flaps and slats

In contrast to the new concept of smart bumps on the airfoil, trailing edge flaps and leading edge slats are already known aerodynamic devices for low speed and to obtain high lift. In high speed flow these wing components are useful for maintaining smooth cruise flight by rapidly responding to gusts and clear air turbulence. A geometry preprocessor for Drela's airfoil analysis and design computer program MSES, included in a new aerodynamic expert system by Zores [23], allows a rapid variation of boundary conditions modelling flap or slat deflection to compute aerodynamic performance of whole series of airfoil shapes. Fig. 8 illustrates an example for a transonic airfoil maintaining lift at varying angles of attack through flap deflections. Such results this way can be obtained within minutes on a workstation, graphic visualization completes an efficient aerodynamic design and analysis system with components for "pre-, fast- and postprocessing".

Nose droop airfoils

Adaptation of the airfoil nose geometry is another concept to favorably influence aerodynamic performance: The example of unsteady airfoil flow derived from the cyclic motion of a helicopter rotor was investigated numerically solving the Reynolds averaged Navier Stokes equations and a periodic nose droop reducing dynamic stall was modelled [24]. Though the use of the NS equations for design still cannot be termed "fastprocessing", again the systematic geometric preprocessing greatly helps the acceleration of carrying out numerous computer runs and understanding the resulting modelled phenomena. A transonic phenomenon occurs at even very low Mach numbers. Fig. 9 illustrates unsteady airfoil flow as a 3D problem with time as the third coordinate. At high angles of attack separation might occur at the leading edge (dynamic stall). Separation may be avoided by a nose droop and flow quality further improved by applying the knowledge base of shock-free design to a very local area at the leading edge.

<u>3.3 Selection of 3D design parameters</u>

Airfoils, sometimes designed with parametric geometry modifications as outlined above, are used to define 3D wing sections. Airfoils designed for transonic flow may suitably be used for swept wing design, where 2D airfoil flow quality is preserved in the plane normal to the leading edge even when the transonic or supersonic wing flow Mach number M_∞ is much higher than the airfoil Mach number $M_{\infty N}$. This wellknown fact may be combined with a visualization of propagated characteristics on the wing surface (see Fig. 5) determined solely by the local (supersonic) Mach number. Fig. 10 shows an example using the airfoil "B" flow of Fig. 6 for definition of a shock-free supersonic infinite wing flow with $\Lambda = 60^o$ sweep angle. Given a numerical simulation result of a 3D flow, the postprocessing part of the above-mentioned software system allows an integration of streamlines as well as of the propagation of characteristics along surfaces in the flow. A grid composed of both characteristics families is drawn and defines regions of dependence downstream of any

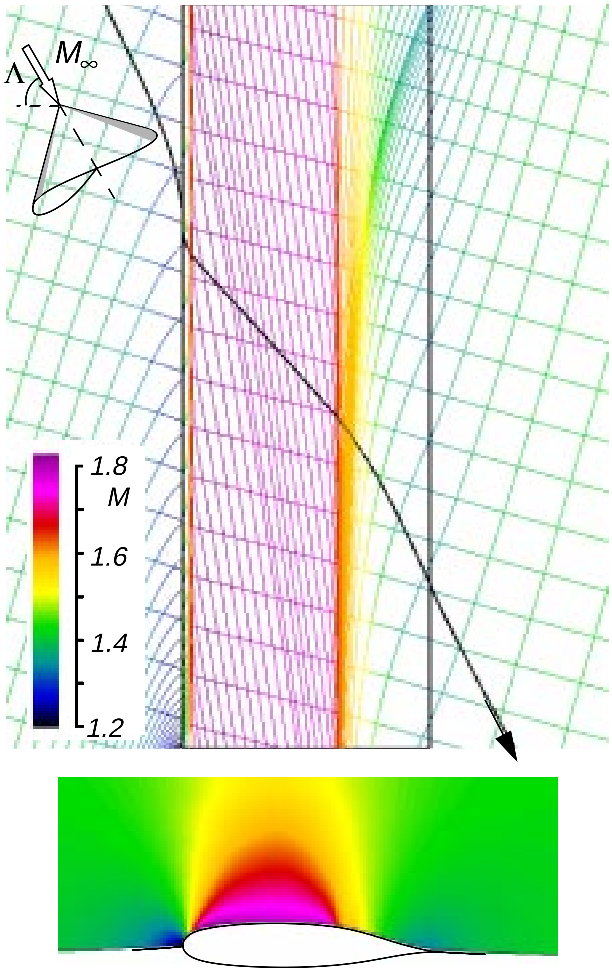

Figure 10: Infinite swept wing flow (M_∞ = 1.414, Λ = 60^o), based on supercritical airfoil design $M_{\infty N}$ = 0.707: Characteristics in wing plane, upper wing surface. Mach iso- fringes in plane normal to subsonic leading edge.

surface point on the wing. This may be used for local spanwise and downstream geometry variations without changing certain upstream parts which may be aerodynamically acceptable. The number of parameters in an optimization procedure is reduced and thus the design process accelerated.

Configuration generator and 4D extensions

Besides of a series of generated airfoils, the planform shape, twist and dihedral distribution along span are the basic wing parameters modelled by a set of analytical functions. Fuselage, propulsive devices and empennage are modelled in a similar way. These functions are input to a basic “geometry generator” [25] which became a widely used practical tool for many applications [26]. Presently, it is extended for more flexibility and more complex shapes, as an interactive preprocessor software for aerodynamic expert systems. Examples of configura-

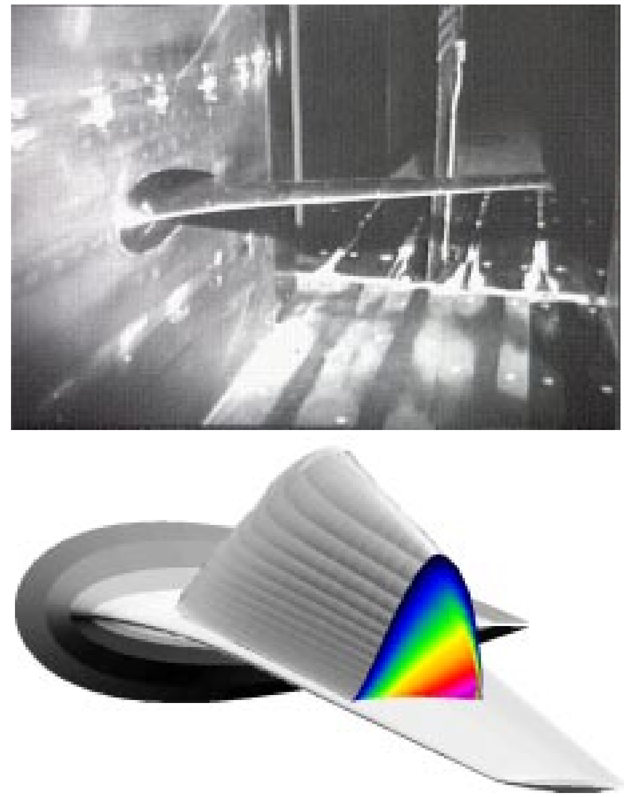

Figure 11: DLR-F5 Test wing configuration: Half model in transonic wind tunnel (above); numerical simulation with RANS code, M_∞ = 0.82. Resulting local supersonic domain visualization (below).

tion details which are difficult to shape are junctures between objects, like wing fillets and fairings; the knowledge about high speed flow quality in these regions is still poor and standard CAD systems provide surface elements which are not tailored to include suitable parameters for flow quality optimization there. Past wind tunnel experimental results for manually shaped models is a guideline to mathematical modelling of these geometry details. Parametric variations and numerical simulation using the resulting boundary conditions give an idea of the efficiency of selected shape models.

Aeroelastic numerical simulation requires unsteady variations of boundary conditions. Similar to the 3D extension of airfoil flows with time (Fig. 9), 3D airframes need to be generated in 4 dimensions for unsteady phenomena modelling. Such capacity can equally be used for running numerical optimization procedures between predefined shape variations, as well as modelling the real time variation of its mechanical counterparts: adaptive devices on 3D configurations.

Visualization of unsteady modelling requires video animation. Some of our first case studies using software for geometric modelling and CFD simulation were used to develop a video capacity of the postprocessing tools [27]. Among these are experimentally investigated test cases outlined below.

4. Test cases for extending the knowledge base

It is mainly three-dimensional phenomena of both viscous and inviscid high speed flow, which still need better understanding so that simplified modelling may accelerate the design concepts. Geometry definition and preliminary CFD simulation was therefore used within the past decade at DLR Göttingen to design two wind tunnel models:

DLR-F5

A wing half model mounted to a wall in a closed walls test section of the Göttingen transonic wind tunnel served as a test case for Navier Stokes code validation [28]. Exactly defined boundary conditions, geometry of wing plus tunnel as well as measured inlet and exit flow parameters are made available to the CFD community. There was a poor agreement of numerical results with each other and with experiment in 1987, it would be interesting to repeat the comparison with CFD methods today: The database is still available for validation of numerical methods [29].
Another a-posteriori purpose of this case study was the successful development of the HIGHEND postprocessing software [30], visualizing experimental and numerical data. Fig. 11 shows a cut through the sonic bubble on the wing. The large fillet at the wing root results in a smooth expansion near the wall, avoiding the leading edge oblique shock usually emanating from a root corner of a swept wing.

DLR - F9

Since definition of DLR-F5 and testing the preprocessor software by duplicating a number of existing aircraft wings, the requirements to define more complex details of realistic airframe components led to extensions of the geometry tools. A challenging example is the definition of a generic high wing aircraft with a parametric variation of the wing root area. Fuselage shape of military transport aircraft is known to have a poor areodynamic performance because of the practical requirements of the large wheel gear box and upward curved afterbody, (Fig. 12), so that wing circulation near the wing root is low. A design goal therefore is an improvement of the wing root and fuselage roof area near their juncture to increase total lift. For the development of software and information technology several other tasks are obvious if such a test case database would exist:

CAD: Preprocessor software should prepare geometry data so that the aerodynamicist is able to prescribe very

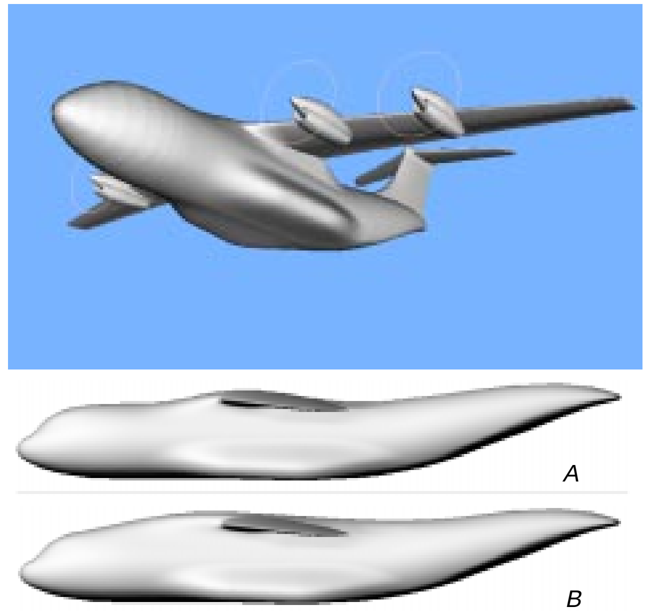

Figure 12: Generic transport aircraft modelled with geometry preprocessor software; parametric shape variation proposals.

precisely the shape details found by his optimizing efforts, as input for trained CAD personnel preparing data for model production and subsequent airframe component definition. Standard CAD software should read the output of aerodynamic design activity using geometry preprocessor data.

CFD: These same data should be used for CFD, ‘fastprocessing’ codes as well as high performance computing activity should have realistic, precisely defined and widely available data to encourage interaction between algorithm developers as well as engineers interested in applied aerodynamics code validation.

MDO, (Multidisciplinary Design Optimization): Aerodynamic layout is just one aspect of global aircraft design. Structural and operational requirements interfere strongly with aerodynamically desirable results. A multidisciplinary approach is therefore necessary in aircraft industry. A generic high wing aircraft will be an excellent testbed for MDO goals with various degrees of complexity. Generating a series of shapes (Fig. 12) controlled by one global variation parameter which controls a multiplicity of geometry detail parameters provides a welcome data base for much more than just aerodynamic research and development.

Configuration DLR-F9 is a recently defined case study and an attempt to create a database for just these different goals in aerodynamics as well as in information technology. Wind tunnel experiments (Fig. 13) with cropped wings will be devoted to test adaptive components. New

CFD code [31] development at DLR using unstructured grids uses this example as a challenging test case, a first result of the Euler code version is shown in Fig. 14. Again the sonic bubble is evaluated by the postprocessor, showing that the supersonic domain is reaching across the fuselage but there might still be an option to increase lift by reshaping this part of the configuration.

5. Conclusions

A review of some transonic design work has been presented which is basically building on mathematical mapping resulting from the classical near sonic model equations, its mixed type analytical flow solutions and their geometric shapes, including boundary conditions, sonic and shock surfaces. The importance of geometry tools with parameters designed from flow phenomena knowledge stems from the possibilities to use such tools on modern interactive workstations for rapid preliminary design, but also to create consistent product-oriented data for CAD, CFD and MDO working groups in the industrial environment.
Some of the old gasdynamic basics along with educational elements of classical mathematics and geometry can be preserved in new media created by information technology, which might be attractive for student education in the aeronautical sciences and thus help sparking the creativity of a future generation of engineers.

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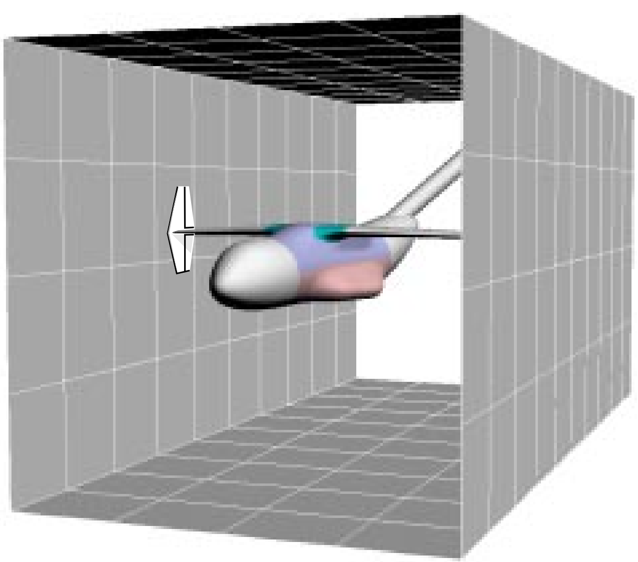

Figure 13: DLR-F9 wind tunnel arrangement of a large configuration for detailed investigations in the wing root area: cropped wings with end plates for circulation distribution control, fuselage with modular and adaptive parts.

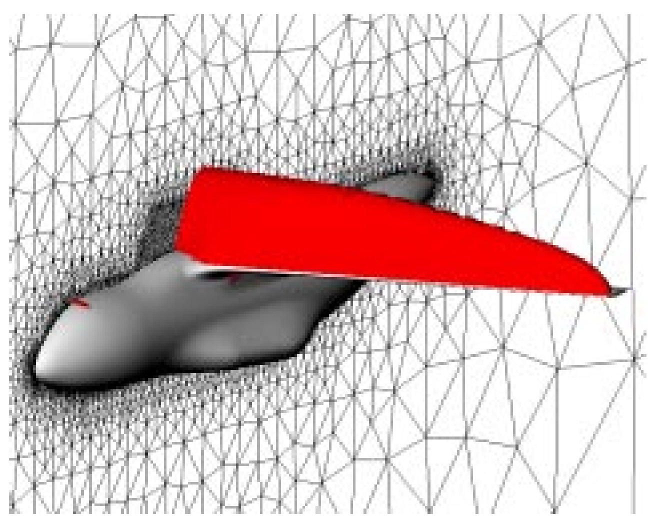

Figure 14: Generic high wing transport wing-body configuration (Juncture variation A); Euler simulation (DLR - τ code) using an unstructured grid with 0.5 Mill. points of 2.5 Mill. tetraedric cells. Visualization of the sonic bubble on the upper wing and fuselage surface.(M_∞ = 0.76, C_L = 0.4)

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