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An Evaluation of Aerodynamic Prediction Methods Applied to the XB-70 for Use in High Speed Aircraft Stability and Control System Design

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Abstract

A key consideration in the development of flight control systems early in the design stage is the availability of aerodynamic information for different candidate configurations. Aerodynamic estimation methods must be available to provide the connection between the configuration geometry and its stability and control characteristics. The accuracy of current estimation methods is studied in this paper through application of DATCOM and APAS to available XB-70 wind tunnel and flight test data. The study was carried out for the subsonic approach condition and three supersonic conditions. Tables and charts are presented to provide a quantitative assessment of the accuracy of the predictions, and areas requiring improvement are identified. Results show that APAS and DATCOM predictions are good for most lateral/directional stability and control derivatives. Estimations for the pitching moment slope, yaw damping and yawing moment due to flap deflection derivatives are only fair. The most difficult derivative to predict is the rolling moment due to sideslip.

Introduction

To understand the connection between the airplane shape and its dynamics, an aerodynamic model of the vehicle must be available. In the early design stages the designer must concentrate on the key parameters and conditions which play a role in developing the overall vehicle concept. Normally this is static stability level and control for trim at key flight conditions.

The general problem of aerodynamic modeling can be formidable. Details are available in the AGARD Lecture Series volume on Dynamic Stability Parameters[1]. However, when considering the development of an approximate aerodynamic model for use in control system design in the conceptual design environment, a more basic approach should be used.

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Aerodynamic modeling for control system design has traditionally relied heavily on wind tunnel testing. However, the detailed testing used to create aerodynamic math models for simulation studies occurs well after the configuration geometry is thought to be defined with a high degree of certainty. Wind tunnel model design and fabrication takes six months *at best*, and subsequent wind tunnel availability, entry and data analysis require several more months at a minimum. Thus, it takes nearly a year to go from concept to stability and control data. Even then, the experimental data is subject to uncertainty due to Reynolds number effects and support interference.

Thus, current designers typically use DATCOM[2] level estimates as a starting point. Although valuable, a more thorough evaluation of DATCOM estimation capability is required for HSCT class vehicles, to build the foundation for improved predictions. It appears timely to re-examine the estimation of aerodynamic math models for control system design integration in the initial conceptual design phase. An approximate model focused directly toward the vehicle synthesis issues is required. With the current ever-more-powerful workstations available, better experimental diagnostics available from flight and wind tunnel data, and even some CFD results, the environment for improving the situation exists.

Numerous codes are available which can provide inviscid estimations (NASA Langley has generated several, including codes by Carlson, AERO2S[3], *etc*., and Lamar, VLM4.997[4]). They can be used to obtain results for vehicles with established flight and wind tunnel data bases. By examining the agreement with the data, adjustments can be developed to reflect effects not modeled. Similar issues, but for hypersonic vehicles only (North American X-15, Space Shuttle Orbiter, etc.), were addressed in papers by Maughmer, *et al*.[5] and Boyden, *et al*.[6] They examined the prediction accuracy of the Aerodynamic Preliminary Analysis System (APAS)[7, 8] integrated with the Unified Distributed Panel (UDP) program[9] and an enhanced version of the Hypersonic Arbitrary Body Program Mark III (HABP)[10]. For HSCT class aircraft such a basis for adjustment does not exist. This paper addresses this problem by comparing the results of predictions from several methods with flight and wind tunnel test data available for the XB-70.

Using these comparisons, semi-theoretical estimates can be developed to reflect the best possible current estimating methodology available, and identify explicitly methodology development needs. This is an area that seems to have been overlooked in technology development programs in recent years because these programs were not directed toward design and multidisciplinary design optimization (MDO) technology development. However, the situation is now changing. Emerging MDO methods are leading to renewed interest in the full range of aerodynamic prediction methods.[11] With this combination of aerodynamic methodology and experimental data, the foundation is established for a conceptual design level control system oriented aero math model code.

Aerodynamic Methods

The key methods selected to compare with flight test data are the standard benchmark from the US Air Force: DATCOM,[2] a general subsonic vortex lattice method due to Jacob Kay,[12] a standard method developed at Rockwell International and used at NASA: the Aerodynamic Preliminary Analysis System (APAS).[7, 8, 9, 10] The computational methods assume linear, inviscid aerodynamics with compressibility effects included using the Prandtl-Glauert rule. This means the flow is attached with low angles of attack, sideslip and control surface deflection. The nonlinear problem of pitch-up has been addressed by Benoliel[13] and will be used as appropriate.

APAS is an aerodynamic analysis system developed at Rockwell International and is capable of analyzing arbitrary three-dimensional configurations from low subsonic to high hypersonic Mach numbers with short computation times. It combines three programs: an interactive program usually called APAS,[7, 8] in which geometry definition and analysis run initialization and evaluation are made; a version of the Woodward[9] subsonic/supersonic panel program called UDP (Unified Distributed Panel), which performs the aerodynamic analysis based on linear potential flow theory, slender body theory and source and vortex first-order panel methods; and an enhanced version of the Hypersonic Arbitrary Body Program (HABP)[10] based on impact theory for hypersonic speeds. The maximum total number of panels, which can be used to represent configuration geometry, is limited to 500.

XB-70 Geometry Modeling Issues

The geometry of the XB-70 was obtained from numerous NASA reports describing the program. The best are by Arnaiz, *et al*.[14] and by Daugherty.[15] We have previously established the geometry using these reports in work validating the ACSYNT program through comparison with a range of aircraft, including the XB-70[16].

DATCOM

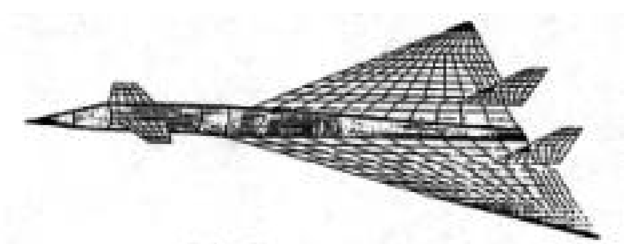
Since DATCOM was designed to estimate aerodynamic derivatives of conventional configurations, some elements of the XB-70 geometry cannot be represented exactly, or sometimes cannot even be modeled at all. For example, during flight at supersonic speeds the XB-70 airplane deflects the wing tips down, first, to catch the shock wave coming from the nacelle inlets to gain extra lift, second, to increase directional stability and, third, to decrease pitching moment. The benefit deflecting the wing tips to reduce the aerodynamic center shift is not well documented in the open literature. For Mach numbers M=1.6, 2.2 and 2.5 the deflection angle is 65 degrees. DATCOM provides no method for estimating aerodynamic derivatives of such configurations. Therefore, no lateral-directional derivatives were computed for supersonic speeds. However, we tried to estimate longitudinal derivatives at supersonic speeds by first computing derivatives for the configuration with undeflected outboard wings and then subtracting the part contributed by undeflected tips and adding the part due to tips deflection. A DATCOM list of "unconventional" features of the XB-70 also includes nacelles (no method for handling nacelles of such shape and size is available), canard (no explicit method for wing-canard configurations, only for configurations "with the ratio of the forward- to the aft-surface span is less than 1.5"), twin vertical tails (the tail can be either single and mounted on the body or twin, but mounted on the wing tips), flaps on the wing and deflected parts of the vertical tail (only control devices with constant flap-chord-to-airfoil-chord ratio are allowed). Therefore, nacelles are not included in the DATCOM model. Assumptions (possibly crude), that downwash due to canard only influences the inboard wing and wing tip deflection doesn’t cause large changes in pressure distribution on both inboard and outboard wings, are made. For twin vertical tails mounted on the fuselage Blake[17] recommends use of an "equivalent" single vertical tail with the same total area, aspect ratio, taper ratio and sweep. However the XB-70 vertical panels are mounted on the wing far from fuselage and, therefore, a formula for the twin tail mounted on the wing tips is employed for subsonic speeds. Wing flaps and moving parts of the vertical tail are replaced by surfaces with the same area, but having constant flap-chord-to-airfoil-chord ratio. Then, a correction for the fixed triangular part of the rudder is applied. For calculations of the rolling and yawing moments due to flap deflection DATCOM imposes restrictions on the aspect and taper ratios for which the XB-70 does not fit. Therefore, the closest allowable values in DATCOM are used instead of actual aspect and taper ratios.

JKay VLM

The subsonic VLM code by Jacob Kay[12] uses the projection of the airplane onto X-Y plane (wing plane) and X-Z plane (plane of symmetry) to estimate longitudinal and lateral-directional derivatives respectively. Such a simplified representation results in, first, poor fuselage modeling and, second, in neglecting the wing and horizontal tail contribution to some lateral-directional derivatives. The rolling moment due to sideslip is probably the best example. Geometry of control surfaces in this code is subject to the same restrictions as in DATCOM and they were treated in a similar manner.

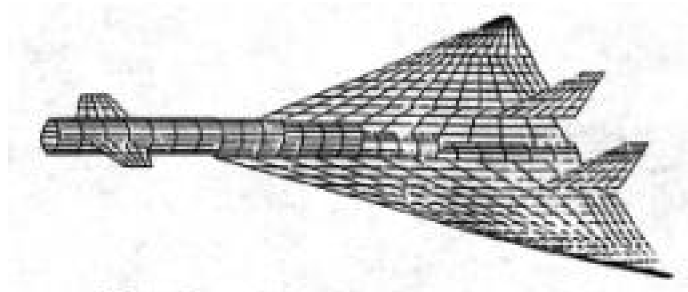
APAS

To account for carryover loads induced on the body by adjacent components, an interference shell is constructed. Since the XB-70 fuselage is not a straight cylinder, but highly bended, and the nacelles expand spanwise toward the end, a concatenating of two or more shells is needed to accommodate longitudinal variations in body cross sections. This process may require some experience, especially if the configuration cross sections have a complex shape (circle cross-sections merge to rectangular cross-sections for the XB-70). Therefore, to simplify the analysis, we excluded nacelles from our original APAS model of the XB-70. We will refer to this model as APAS 1. The simplified slender body, interference shell and paneled surface components for the APAS 1 configuration are shown in Figs. 1a and 1b for subsonic (wing tips extended) flight conditions. Later we constructed a more elaborate second model, APAS 2, which is shown in Fig. 2 for supersonic (wing tips deflected 65 deg. down) flight conditions. The slender body component for this configuration, Fig. 2a, is more complex and includes both fuselage and nacelles. Shells and paneled surfaces for this configuration are shown in Fig. 2b. The APAS 1 and APAS 2 models have different paneling of the inboard part of the wing, but the total number of panels used to represent the planform is the same. Since the flap hinge line separates panels placed on and off the flap, the spanwise alignment of the leading edges of all panels on the wing is not possible. However, numerical experiments have shown that this alignment does not affect the result significantly. To preserve the actual geometry of the twin vertical tail oblique hinge line a special paneling, Fig. 1a, was used to compute derivatives due to rudder deflection. Panels were clustered near the intersection of the hinge line and leading edge.



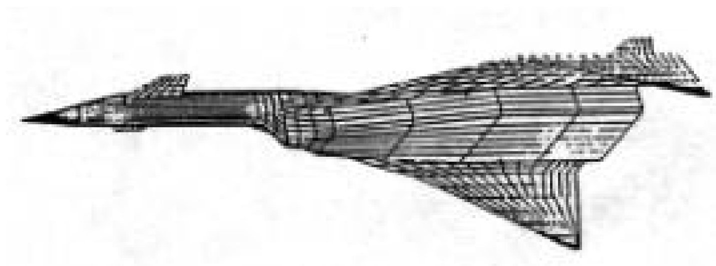
a) simplified slender body component and paneled surfaces

Figure 1. APAS 1 model of the XB-70.

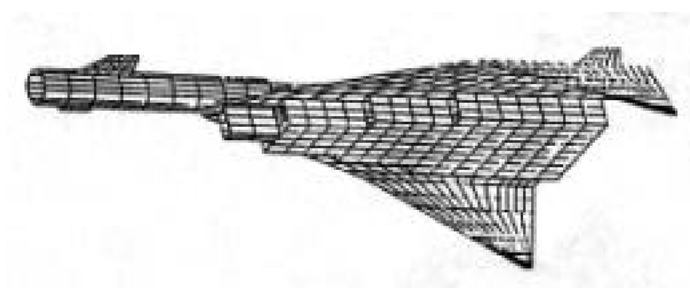


b) interference shells and paneled surfaces

Figure 1. APAS 1 model of the XB-70 (concluded).



a) elaborate slender body component and paneled surfaces

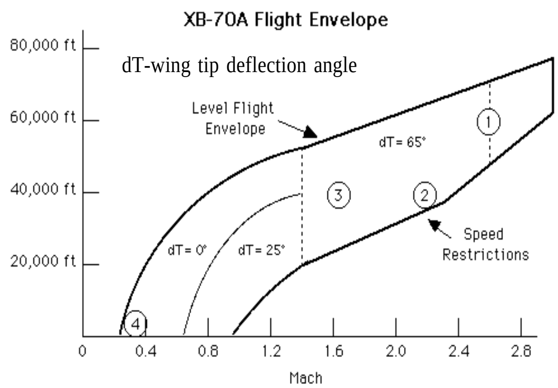


b) interference shells and paneled surfaces

Figure 2. APAS 2 model of the XB-70.

XB-70 Flight Conditions

To evaluate aerodynamic prediction techniques we have selected four flight conditions from the data report by R. K. Heffley and W.F. Jewell,[18] and which includes both low speed and supersonic flight conditions. These flight conditions are shown in Fig. 3 and were chosen as those likely to define the boundaries of the aircraft control requirements. For example, the powered approach flight condition defines the smallest dynamic pressure (142 lb/ft^2) encountered by the XB-70 while the Mach 2.2, 40,000 ft altitude case defines the highest dynamic pressure (1,335 lb/ft^2) flight condition. The first and third flight conditions were chosen as high and low speed cruise flight conditions, respectively, where a high speed transport aircraft is likely to fly the majority of the time. The key conditions are the low speed power approach configuration and the supersonic cruise at Mach 2.5.



1) Cruise at 60,000 ft and Mach 2.5
2) Cruise at 40,000 ft and Mach 2.2
3) Cruise at 40,000 ft and Mach 1.6
4) Powered Approach, sea level and Mach 0.31

Figure 3. Critical flight conditions for the XB-70.

Data Discussion

The data values used in our comparison were obtained from the NASA report by Heffley and Jewell[18], which is a composite of many sources. For the power approach case this reference contains a table of derivatives as well as plots of all derivatives versus Mach number. It is interesting to note that the table values are not consistent with the plots for four derivatives, the pitching moment slope, the yawing moment due to flap deflection, and the rolling and yawing moments due to rudder deflection. Instead of the table derivatives, values interpolated from the plots were used as data for comparisons.

All estimation methods considered in this paper exclude aeroelastic effects, while the data values are given for a flexible airplane. The NASA report[19] by Wolowicz, *et al.*, demonstrates that aeroelastic corrections are significant. For example, according to this report, the difference between rigid and flexible airplane predictions for subsonic speeds is no less than 50% for the pitching moment slope ($\Delta C_{m\alpha}$=0.001 to 0.002 per deg.), 15% and 10% for the rolling moment due to sideslip and flap deflection respectively. For supersonic flight conditions the flexibility corrections for $C_{m\alpha}$, $C_{l\beta}$ and $C_{l\delta flp}$ derivatives are as high as 50%, 100% and 50% respectively. Their report also contains flight test data uncertainties shown here in Table 1, which are useful in understanding the results discussed in the next section.

Table 1

Flight test data uncertainties (from Ref. 19)

Longitudinal Derivatives	Maximum Uncertainty Subsonic	Maximum Uncertainty Supersonic
$C_{L\alpha}$	20%	10%
$C_{m\alpha}$	10%	5%
C_{mq}	40%	30%
$C_{L\delta e}$	100%	30%
$C_{m\delta e}$	20%	10%

Lateral-Directional Derivatives	Maximum Uncertainty
C_{Yb}	20%
$C_{l\beta}$	15%
$C_{n\beta}$	5%
C_{np}	20-100%
C_{nr}	30%
C_{lp}	30%
C_{lr}	50%
$C_{Y\delta a}$	100%
$C_{n\delta a}$	30%
$C_{l\delta a}$	15%
$C_{Y\delta r}$	50%
$C_{n\delta r}$	5%
$C_{l\delta r}$	30%

Aerodynamic Prediction Evaluation

This section contains an analysis of the results obtained in this evaluative study. The work is establishing a benchmark for future improvements in aerodynamic prediction methodology.

The comparisons of the aerodynamic derivative estimations provided by DATCOM, VLM and the two APAS models, APAS 1 and APAS 2, to the data values were made at Mach 0.31, 1.6, 2.2 and 2.5 with the center of gravity located at 23.5% and 21.8% of the MAC for subsonic and supersonic speeds respectively. All aerodynamic derivatives are given per radian. The rolling and yawing moment derivatives due to flap deflection and all derivatives due to rudder deflection correspond to asymmetric flap and vertical stabilizer deflections respectively.

Assuming that corrections due to flexibility and flight test data uncertainties for our rigid airplane calculations are of approximately the same order as those mentioned in the previous section and Ref. 19, we can better understand and explain errors made in our estimations.

According to linear aerodynamics all derivatives considered in this paper except for $C_{l\beta}$ are independent of the angle of attack and all of them were computed at zero angle of attack. APAS allows the user to account for the leading edge vortex suction effects in analysis, but this option was never used. The evidence of wind tunnel testing reported in Ref. 15 is that for all flight conditions except for cruise at Mach 2.2 the vortex suction effect is not negligible and therefore results in additional error in our estimations.

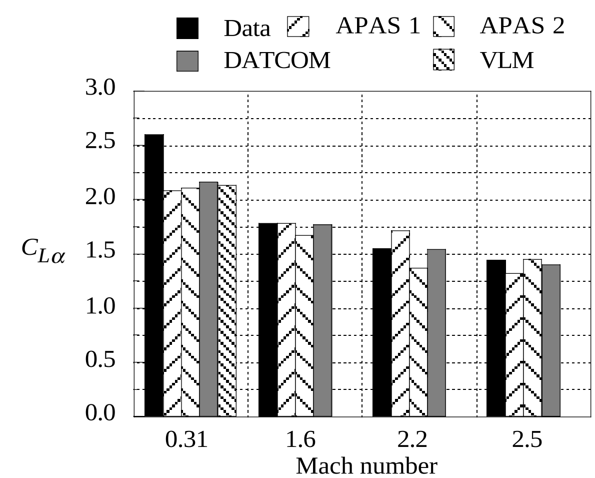
Numerical experiment has shown that the difference between results obtained using a uniform panel distribution on the twin vertical tail and the special clustered paneling described earlier is less than 2%. Thus, uniform and clustered paneling schemes are equivalent and both can be used for all cases.

Stability Derivatives..

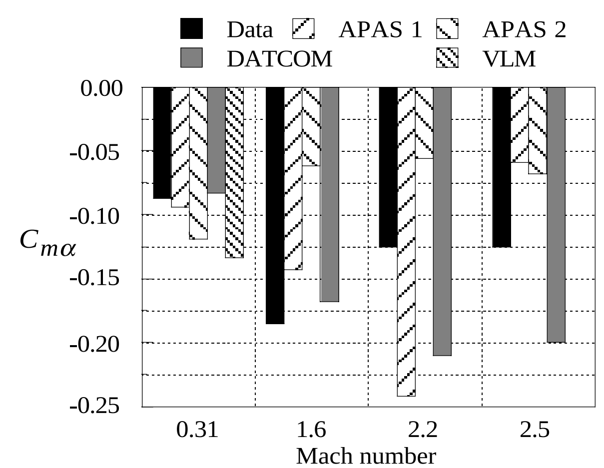
The angle of attack derivatives are shown in Fig. 4. For the lift-curve slope, Fig. 4a, the VLM, DATCOM and both APAS predictions are very close to each other and to the data values for all Mach numbers, except for Mach 0.31. The difference for powered approach can be partially explained by the fact that we neglected leading edge vortex suction effects and gear influence along with possible flow separation. Note that lift does not change much in the presence of nacelles (APAS 2 has nacelles, APAS 1 does not). Poor correlation of the pitching moment coefficient, Fig. 4b, with data for all Mach numbers probably indicates that the pressure distribution is strongly affected by nacelles, especially for Mach numbers 1.6 and 2.2, aeroelastic deformations and by tips deflection at supersonic speeds. Errors in $C_{L\alpha}$ prediction also result in errors in the pitching moment coefficient. Since for subsonic speeds the C.G. is located very close to the aerodynamic center of pressure, the pitching moment is very sensitive to the C.G location. For example, in the case of powered approach, a 1 % MAC C.G. shift results in 10 % change in the pitching moment slope***.

Derivatives due to sideslip are shown in Fig. 5. Note that for supersonic speeds only APAS is able to predict these derivatives. The side force derivative, Fig. 5a, is well predicted for all speeds. Surprisingly, the simplified APAS model gives a more accurate result, especially for the powered approach case. The yawing moment derivative is shown in Fig. 5b. According to this figure nacelles have a big influence on this derivative and, therefore, a detailed description of the aircraft geometry is vital for accurate estimations of $C_{n\beta}$. Estimations of the rolling moment derivative, Fig. 5c, are far from the data values. Rolling moment is not constant with the angle of attack even in linear theory and depends on the lift coefficient. Note that APAS fails to predict the rolling moment variation with alpha and gives the same result for all values of the angle of attack. Therefore, the rolling moment due to sideslip seems to be the most difficult derivative to estimate especially for the powered approach case when the angle of attack is not small. Comparison between VLM and DATCOM values for the rolling moment shows that the VLM code (in J.Kay’s formulation) is able to predict only the contribution of the vertical tail or tails (the value of the rolling moment due to sideslip provided by the JKay's VLM code is almost the same as the vertical tails contribution according to DATCOM), while the wing is the main contributor to $C_{l\beta}$.

Fig. 6 contains derivatives due to pitch, roll and yaw rate. Pitch damping estimations, Fig. 6a, are close to the data values only for Mach numbers 0.31 and 1.6. For other flight conditions DATCOM estimations are more accurate than both APAS models. The roll damping predictions, Fig. 6b, are good for all speeds and nacelles modeling has minimal effect on prediction accuracy. Values of yaw damping, Fig. 6c, are about 30-50% less than the data, and nacelle modeling again is not very important for this derivative. According to DATCOM,



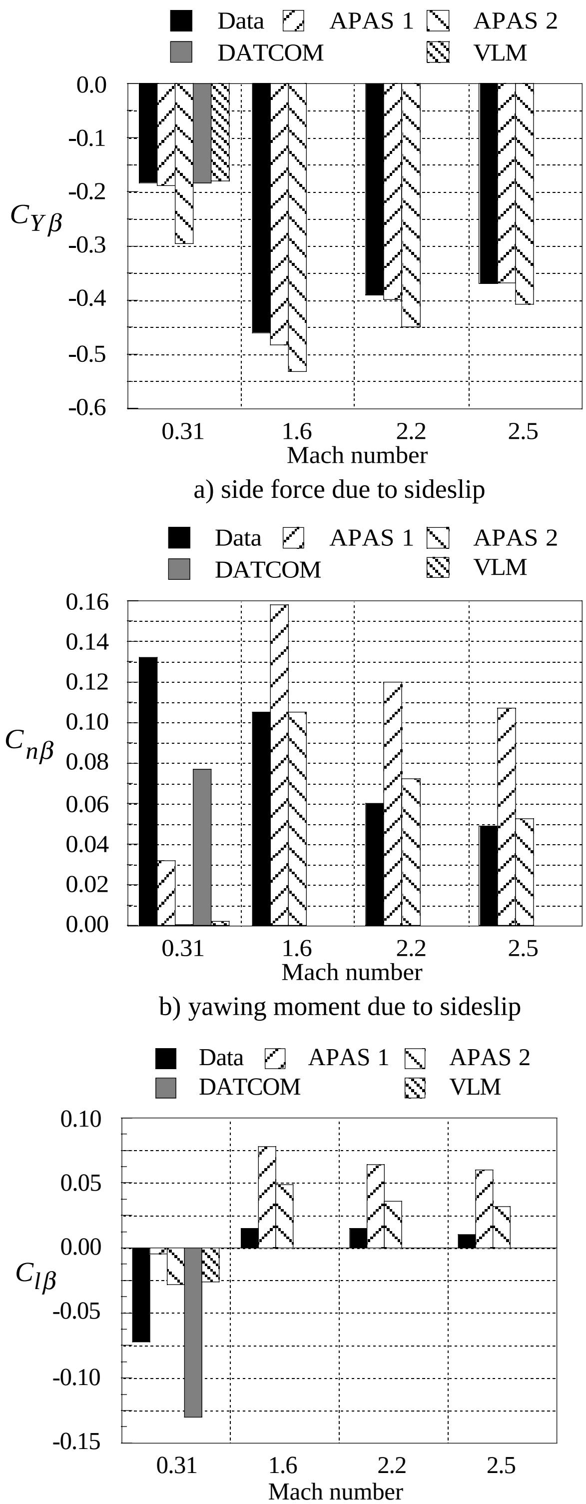
a) lift curve slope



b) pitching moment slope

Figure 4. Angle of attack derivatives.

*** All data referred in this section are the flight test data. Having presented the paper we have found APAS2 prediction of the AOA derivatives to be very close to the wind tunnel test data. This means that APAS estimations require extrapolation to full-scale conditions in the same manner as the wind tunnel data.



a) side force due to sideslip

b) yawing moment due to sideslip

c) rolling moment due to sideslip

Figure 5. Sideslip angle derivatives.

the effects of the wing and body on this derivative are assumed to be negligible at subsonic speeds, and yaw damping is simply $2(C_{n\beta})^2/C_{Y\beta}$. Substitution of the DATCOM predictions into this formula for Mach 0.31gives a result close to APAS estimation, but not to the data value, since the $C_{n\beta}$prediction is poor. However, using data values of the side force and yawing moment instead of computed would produce an estimation with an error of less than 20% value. This value is shown in Fig. 6c as the DATCOM prediction.

Control Derivatives.

Flap effectiveness plots are shown in Fig. 7. Lift due to flap deflection, Fig. 7a, is slightly overestimated due to the inviscid solution, but still well predicted by all methods. Overestimation of the pitching moment due to flap deflection, Fig. 7b, is bigger than for the lift, but tends to decrease as Mach number increases. For the rolling moment due to flap deflection, Fig. 7c, only the subsonic predictions by all methods are close to the flight test value. For supersonic speeds APAS and DATCOM estimations are in good agreement with each other, but not with data values. Yawing moment due to flap deflection, Fig. 7d, is closely predicted by APAS 1 only at subsonic speeds. Accurate estimation of this derivative is difficult, because it requires the drag consideration. DATCOM and VLM fail to take into account vertical-tail-flap interference and, therefore, underestimate the induced drag. The real flap is divided into six segments to avoid binding due to wing bending, but in some wind tunnel models, as well as in all methods discussed here, the flap was not segmented, which according to Ref. 15 results in "significant differences in longitudinal characteristics associated with segmentation of the elevon". Flexibility effects are also especially significant for the rolling moment.

Canard effectiveness is presented in Fig. 8. Data values for subsonic speed are not available. During powered approach the canard flap is deflected 20 deg. down and the canard itself is fixed. The net lift increment due to canard deflection is usually small (approximately 1/4 of the lift due to flap deflection) and can be decomposed into two major contributors mutually compensating each other: lift change experienced by the canard alone (as if there is no wing) and lift change due to downwash experienced by the main wing. Since canard-wing interference effects can not be accurately estimated by DATCOM (at least for the XB-70 configuration), only the first contributor was computed using DATCOM. The second effect was neglected. Originally the net change in lift due to canard deflection was estimated using APAS and VLM, but the values obtained were close to each other and much smaller than data. To find the reason for this, the contribution of the canard alone was estimated using APAS (our VLM code needs modifications to do this) and plotted in Figs. 8a and 8b instead of the net values of force and moment. For supersonic speeds both DATCOM and APAS predictions of the lift, Fig. 8a, and pitching moment, Fig. 8b, are very close to the data, meaning that the data do not show the downwash effects predicted theoretically.

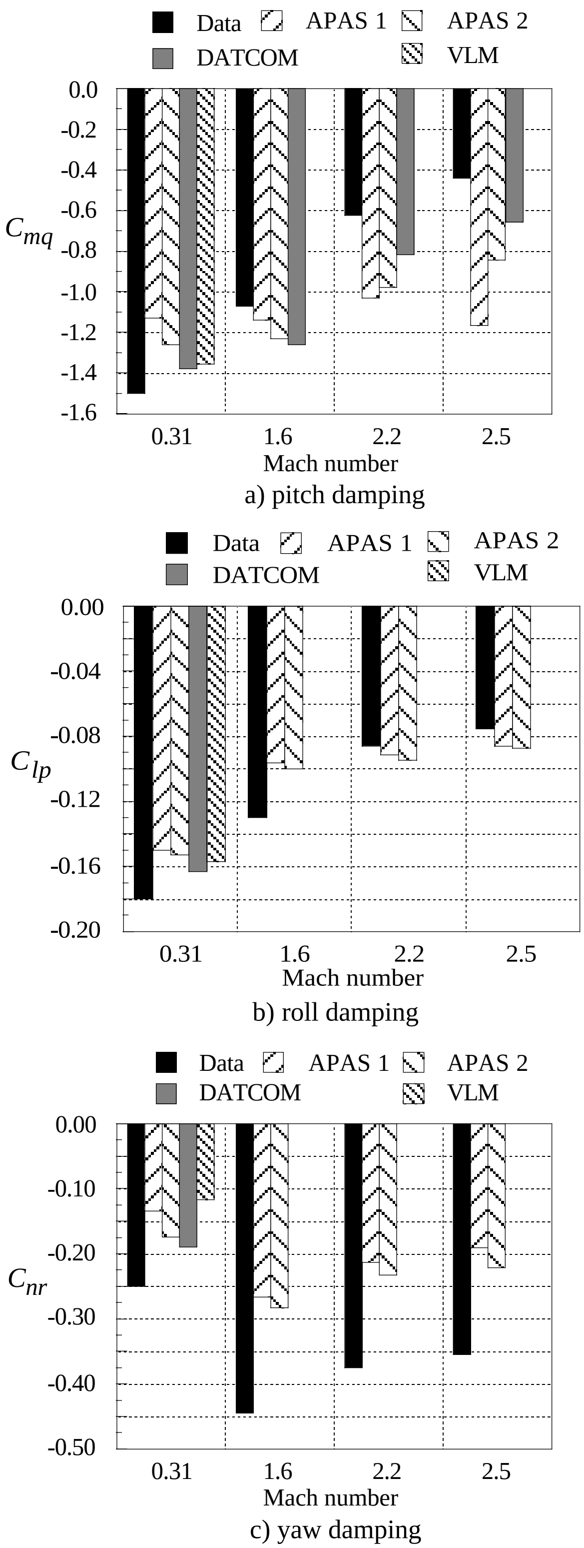


Figure 6. Derivatives due to pitch, roll and yaw rate.

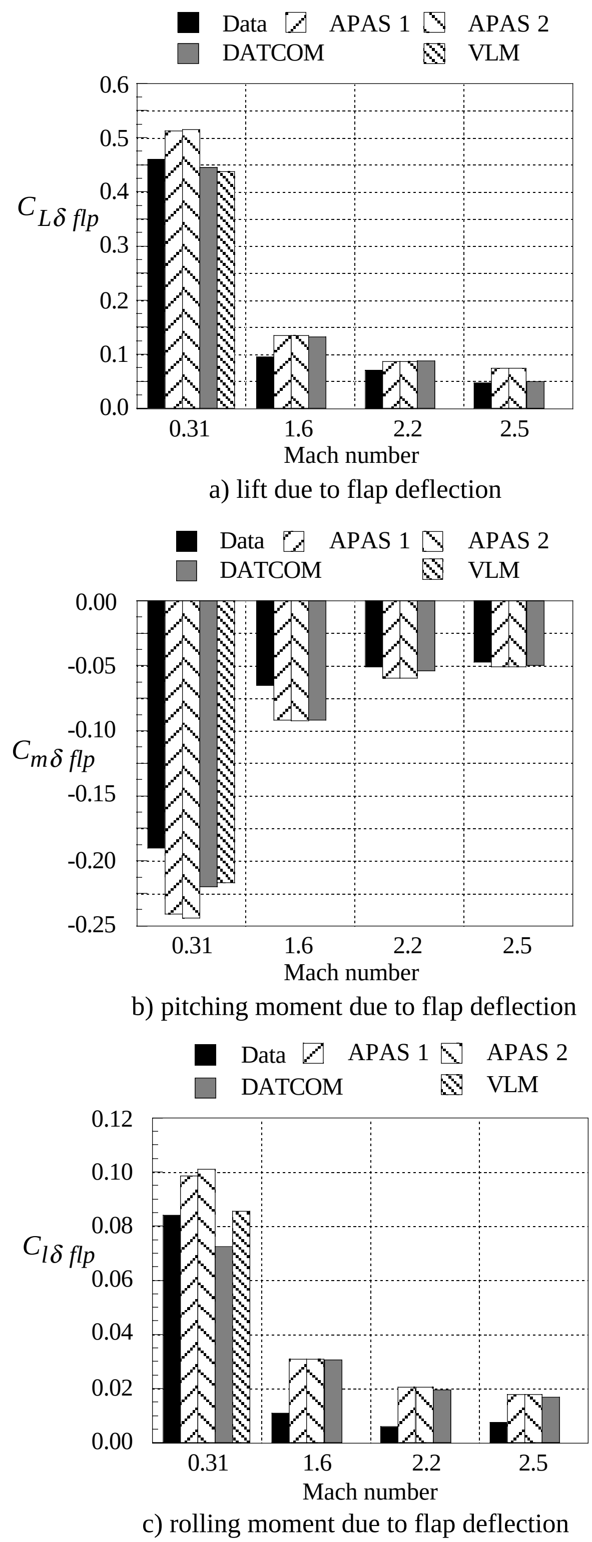
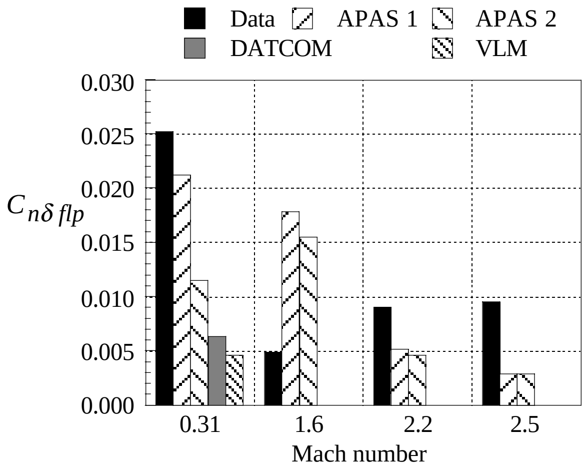
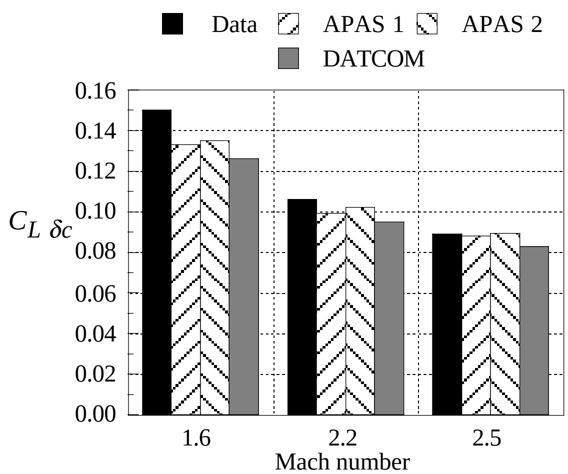


Figure 7. Flap effectiveness.

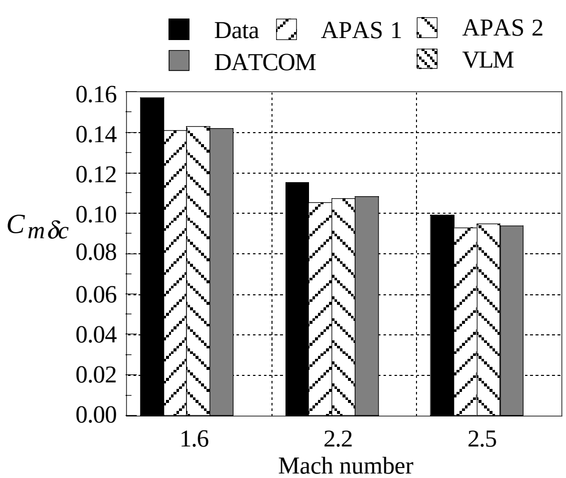


d) yawing moment due to flap deflection.

Figure 7. Flap effectiveness (concluded).



a) lift due to canard deflection



b) pitching moment due to canard deflection

Figure 8. Canard effectiveness.

Fig. 9 contains plots for rudder effectiveness. Side force and yawing moment due to rudder deflection are predicted fairly well by all methods for subsonic speed and by APAS for supersonic speeds. Rolling moment estimation is unacceptable only for Mach 2.5. Reduced accuracy for yawing and rolling moment estimations may result from the problems associated with modeling of the oblique hinge line and the estimated 30% uncertainty in data values (see Table 1).

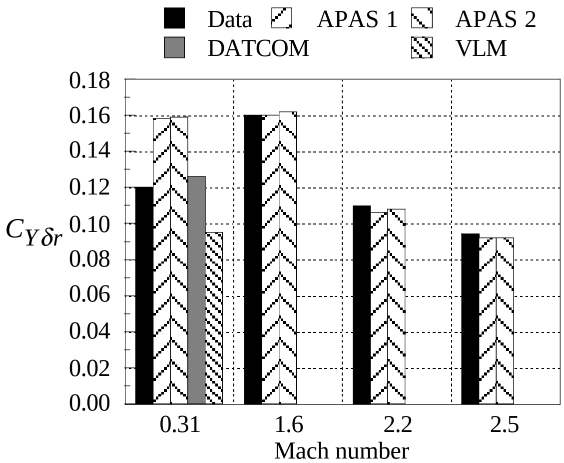
Comparing Figures 7, 8 and 9 we can conclude that nacelles modeling is not important for control derivative estimation, since both simplified and detailed models produce very close results.

Summary and Conclusions

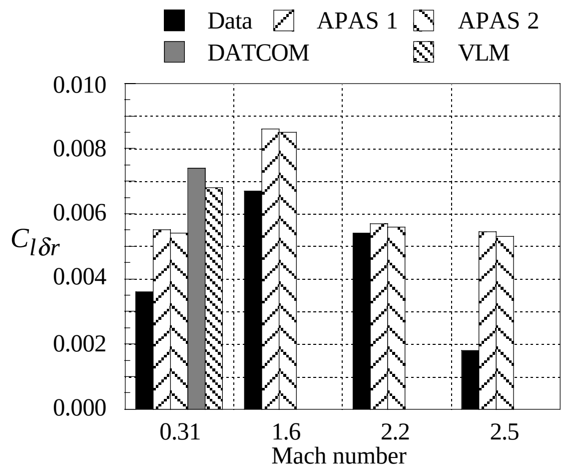
Tables 2 and 3 contain a summary of errors, in percent, made for different aerodynamic derivatives. Two values, Min and Max, are shown for the XB-70 derivatives. They are the errors associated with the most and the least accurate predictions respectively. Clearly, if the gap between these two errors is small, then all methods provide the same level of accuracy, and the cheapest and/or the fastest method can be used for the estimation of this particular derivative. Note again that the XB-70 configuration is not conventional, and, therefore, we likely underestimate the level of accuracy achievable using DATCOM for a more typical configuration.

Along with the approximate flow model the biggest error sources are data uncertainties and flexibility effects. The worst accuracy was obtained for the derivatives of rolling and yawing moments, however, the pitching moment slope also was not predicted well enough. Thus, these parameters are "critical" from the aerodynamic point of view. Some of them might have a big influence on aircraft performance and control system design, i.e. they are "critical" from the control point of view, and, therefore, need to be estimated more accurately using more sophisticated methods.

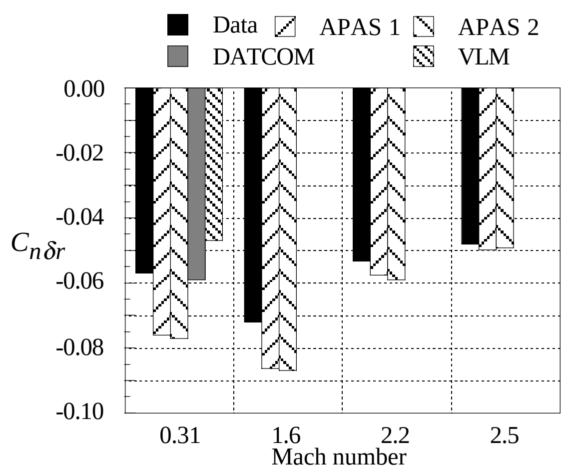
The VLM code implements a simplified version of the Vortex-Lattice method and, therefore, cannot give more accurate results than APAS. Overall APAS does not exhibit significant improvement of accuracy versus DATCOM, but it imposes few, if any, restrictions, provides an excellent interface, and gives results much faster than the other methods, provided the geometry file is given or have been previously created. Since DATCOM is restricted to conventional configurations, in some cases APAS is the only method currently available for estimations. However, for many derivatives DATCOM provides analytical expressions, which can be used in MDO design to compute sensitivities analytically rather than numerically. Clearly, even if those analytical formulas are less accurate than APAS predictions, the amount of CPU time they can save still makes them valuable.



a) side force due to rudder deflection



b) rolling moment due to rudder deflection



c) yawing moment due to rudder

Figure 9. Rudder effectiveness.

We have identified the effect of the XB-70 nacelles on the results obtained from APAS. They are important for $C_{m\alpha}$ and $C_{n\beta}$. We have also identified derivatives that appear to require improved estimates for use in control system design.

Acknowledgment

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Table 2. Accuracy of stability derivative estimations for the XB-70. Prediction Error, %

	M=0.31		M=1.6		M=2.2		M=2.5	
Derivative	Min	Max	Min	Max	Min	Max	Min	Max
$C_{L\alpha}$	-16.9	-19.8	0.0	6.2	-0.6-	11.6	-0.7	-8.3
$C_{m\alpha}$	-4.6	54.0	-9.2	-66.9	-55.0	93.6	-45.4	59.2
$C_{Y\beta}$	0.5	61.2	5.0	15.6	2.3	15.3	-0.5	10.3
$C_{n\beta}$	-41.8	-99.6	0.0	55.3	2.0	100.5	7.5	118.0
$C_{l\beta}$	-61.3	-93.4	224.7	419.3	140.4	327.4	215.2	502.0
C_{mq}	-8.0	-24.7	6.5	17.6	30.2	64.8.	49.5	92.0
C_{lp}	-9.4	-16.7	-23.2	-25.2	6.5	10.3	14.0	16.2
C_{nr}	-24 0	-52.7	-36.5	-40.5	-37.2	-43.9	37.5	46.2

Table 3 Accuracy of control derivative estimations for the XB-70, Prediction Error, %

	M=0.31		M=1.6		M=2.2		M=2.5	
Derivative	Min	Max	Min	Max	Min	Max	Min	Max
$C_{L\delta fl}$	-3.3	12.0	38.6	42.7	21.4	23.4	5.5	58.3
$C_{m\delta fl}$	14.3	28.4	41.0	41.6	5.9	16.9	5.5	8.3
$C_{n\delta fl}$	-15.9	-81.7	218.0	266.3	-42.7	-49.0	-69.9	-69.9
$C_{l\delta fl}$	1.8	20.3	178.6	180.8	226.7	243.5	121.5	134.2
$C_{L\delta can}$			-10.6	-16.7	-3.8	-10.5	-0.4	-0.9
$C_{m\delta can}$			-9.0	-10.4	-6.3	-8.5	-4.5	-6.3
$C_{Y\delta rud}$	4.5	31.0	0.0	0.0	-3.8	-3.8	-2.2	-2.2
$C_{n\delta rud}$	3.6	32.4	20.0	20.0	7.7	7.7	4.3	4.3
$C_{l\delta rud}$	3.2	17.8	27.8	28.4	3.4	5.4	190.0	202.0