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An Improved Process for the Generation of Drag Polars for use in Conceptual/Preliminary Design

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ABSTRACT

One of the most essential contributors in the aircraft sizing and synthesis process is the creation and utilization of accurate drag polars. An improved general procedure to generate drag polars for conceptual and preliminary design purposes in the form of Response Surface Equations is outlined and discussed in this paper. This approach facilitates and supports aerospace system design studies as well as Multi-disciplinary Analysis and Optimization. The analytically created Response Surface Equations replace the empirical aerodynamic relations or historical data found in sizing and synthesis codes, such as the Flight Optimization System (FLOPS). These equations are commonly incorporated into system level studies when a configuration falls beyond the conventional realm. The approach described here is a statistics-based methodology, which combines the use of Design of Experiments and Response Surface Method (RSM). Computational aerodynamic codes based on linearized potential flow and boundary layer theory are employed to generate the needed parametric relationships. The process is facilitated through the use of an automated computational architecture that is capable of handling massive exchanges of data and information. The aforementioned process is demonstrated through an implementation of the procedure for a High Speed Civil Transport concept. The accuracy of these Response Surface Equations is finally tested to demonstrate the fidelity and accuracy of their predictive capability.

INTRODUCTION

System level configuration tradeoffs are needed in the early design phases of any new system. These trade-offs can be greatly facilitated by recent advances in the areas of Multi-Disciplinary Analysis/ Optimization (MDA/MDO) and Robust Design Simulation (RDS), which enable the determination of optimal or robust solutions. By definition, these studies call for the ability to perform rapid design space exploration. More specifically, numerous candidate configurations are investigated and assessed at this stage for a specified mission profile. In the early phases of the design process, the definition of the loft, along with its corresponding aerodynamic characteristics, become perhaps the most important design drivers. This statement holds true even under the Integrated Product and Process Development (IPPD) paradigm that brings forth constraints in considering all product and process disciplines. Therefore, accurate and rapid determination of the corresponding drag polars for each of the configurations present in the design space are needed. To enable system level tradeoffs, these aerodynamic drag polars must be then incorporated into a sizing and synthesis code, which may act as an MDA/MDO environment. Furthermore, information from other disciplines, such as weights from structures, engine data from propulsion, flutter constraints from aeroelasticity, etc. must also be created and integrated into this environment to form a vehicle specific preliminary design analysis (Figure 1). The sizing and synthesis codes used in system level studies, such as the Flight Optimization System (FLOPS) are multi-disciplinary in nature, and include the necessary empirically-based data. Since the data in these empirical models are not valid for concepts outside the historical database, the models or data must be replaced by physics-based analyses if novel configurations are to be investigated. Physics-based aerodynamic analysis codes are concatenated to the sizing and synthesis programs for this purpose.

There are two ways to make the information obtained by physics-based analysis available to the sizing and synthesis process (Figure 2).

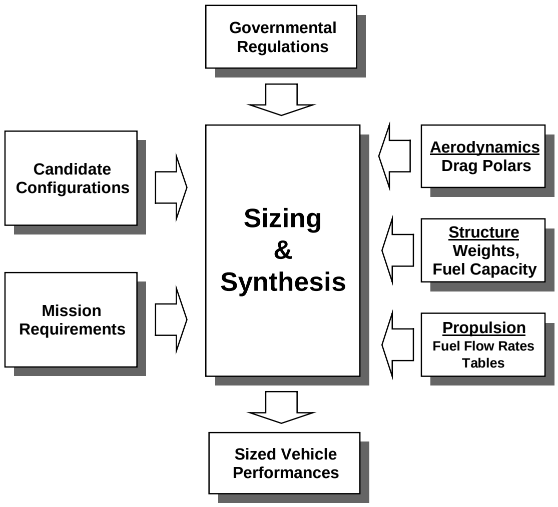


Figure 1. A Typical Sizing and Synthesis

The first way is to link the analysis codes directly to the sizing and synthesis program, which means that the various disciplinary analyses codes are run in real time, while the sizing and synthesis code is waiting for the results for the configuration being analyzed. This method proves to be cumbersome and in most cases impractical, particularly when high-fidelity domain tools are used. The second way is to use a formulation based on the concept of creating Response Surface Equations (RSEs), meta-models which are produced by running off-line actual analysis tools according to a pattern pre-specified by a technique called Design of Experiments (DOE), and subsequently incorporating them in the sizing and synthesis program. Since these disciplinary RSEs are generated before the sizing and synthesis begins, concurrency in disciplinary analyses is achieved by coordinating among the various disciplines the baseline and the variable ranges selected. The use of disciplinary RSEs, instead of actual analysis codes for sizing and synthesis, also enhances to a great extent the efficiency of system level study since the algebraic format of RSEs enables analysis results to be provided instantly.

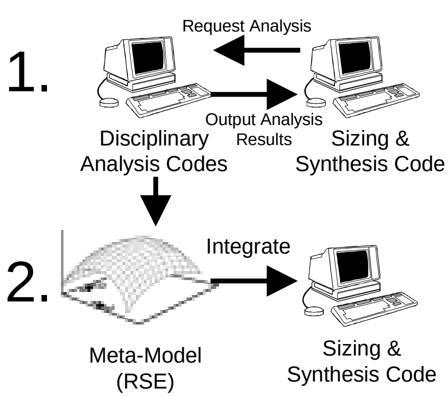


Figure 2. Two Ways of Linking Disciplinary Analyses to create a Physics-based Sizing and Synthesis Capability

The generation of the drag polars used to be inefficient due to lack of an integrated environment and the need for human intervention and monitoring. Since the creation of this approach, the computational modeling and the utilization of the DOE techniques have been improved significantly and the process has become routine. During this time period, different linearized aerodynamic codes have been tested and used in the RSE generation. Their idiosyncrasies have been identified and solutions have been suggested, examined and found.

METHODOLOGY FORMULATION

RESPONSE SURFACE META-MODELS

An RSE is a type of meta-model which could be used for the approximation of cumbersome, time consuming computer analysis programs. It is a relationship created based on the outcomes obtained at various tested conditions. Although the approach is not limited to polynomials, a quadratic polynomial representation is commonly assumed and used since it has been proved again and again to yield good results. A typical quadratic RSE can thus be represented as:

$$R = b_0 + \sum_{i=1}^{n} b_i \cdot x_i + \sum_{i=1}^{n} b_{ii} \cdot x_{ii}^2 + \sum_{i=1}^{n-1} \sum_{j=i+1}^{n} b_{ij} \cdot x_i \cdot x_j \qquad (1)$$

where R is the response, i.e. the function value, b_0 is the intercept term, b_i are the coefficients of the linear terms, better known as the “main effects”, b_{ii} are the coefficients of the pure second order terms, better known as “quadratic effects” and b_{ij} are the coefficients for the cross-product terms, better known as “second-order interactions”. It has been assumed that the quadratic polynomial can be used to approximate the analysis codes by assuming that the higher terms are negligible. The validity of this assumption is obviously dependent on the problem studied and the ranges examined for each variable. It has also been observed that certain variables such as altitude and Mach number tend to have such an overwhelming impact on the responses that the effect of the design variables are overshadowed. This difficulty was overcome by generating RSEs for grids of flying altitude and Mach number. The numerical difficulties in running actual codes, such as divergence at certain design points, can be overcome once the RSEs are generated. The RSEs can also apply the quadratic model to “filter” out some numeric noises[1] caused by model discretization if the high order effects are known to be small.

DESIGN OF EXPERIMENTS

A Design of Experiments (DOE) are orthogonal arrays (matrices) used to produce a set of combinations of tested cases. DOE is a statistical technique to generate the RSEs by determining the "best" combinations of input variables and their levels, based on the number of cases that can be affordable for a desired response resolution. According to this method, each variable is assigned a range with minimum and maximum values defined and normalized, usually as –1 and +1 in the actual DOE table. Each range is then divided into several levels. Full Factorial designs account for the combinations of all desired levels for each variable while Fractional Factorial designs involve a subset of those cases. If certain high order interactions are negligible, Fractional Factorial designs are well suited and can be used to obtain the information about the main effects and low-order interactions. Examples (Table 1) of commonly used Fractional Factorial DOEs for a variety of seven and twelve design variable problems are compared to a 3-level Full Factorial DOE to show the significant reduction in the number of cases needed. The D-optimal experiment represents the limiting case where the number of equations matches the number of unknowns and is thus not a least square regression as the rest of them. This is due to the fact that no degrees of freedom are available for the assessment of fit error.

Table 1. Cases Required for Different DOEs[2]

DOE	7 Variables	12 Variables	Equation
3-Level, Full Factorial	2,187	531,441	3^n
Central Composite	143	4,121	2^n+2n+1
Box-Behnken	62	2,187	-
D-Optimal	36	91	$(n+1)(n+2)/2$

When higher order interactions cannot be estimated independently, the DOE is said to be confounded, and the concept of Resolution is used to categorize the Fractional Factorial designs. Resolution III is the design in which none of the main effects are confounded with any other main effect. On the other hand, main effects are confounded with two-factor (second order) interactions and some two-factor interactions may be confounded with each other. Resolution IV is the design in which no main effect is confounded with any other main effect or two-factor interactions. Two-factor interactions may still be confounded with each other. Resolution V is the design in which no main effect or two-factor interactions are confounded with any other main effect or two-factor interaction. Two-factor interactions are confounded with three-factor (third order) interactions[3]. DOE setups of Resolution V were selected and used in the creation of the RSEs in this paper. In particular, the Central Composite Design (CCD) was selected, which is similar to what is illustrated in Figure 3 for 3 levels and 15 points for 3 factors.[4]

A DOE table can be generated by applying Fractional Factorial Design techniques[7]. First, the number of factors (variables) has to be determined. Then each of the factors is assigned a range to vary, although the actual value is not the concern at present. The maximum value, as stated previously, is represented by “+1”, the minimum by “-1” while a value of “0” is usually the average of the minimum and maximum values. A DOE using a 3 factor CCD is illustrated in Table 2. The input files for analyses codes, commonly called cases, should be prepared according to the variable level settings specified by the DOE matrix. Once the required cases have been run and the responses are obtained, a statistical package, JMP®, is utilized to perform an Analysis of Variance (ANOVA), and through a least squares approach create the needed RSEs. The generation of RSEs using DOE techniques is usually referred as Response Surface Method (RSM).

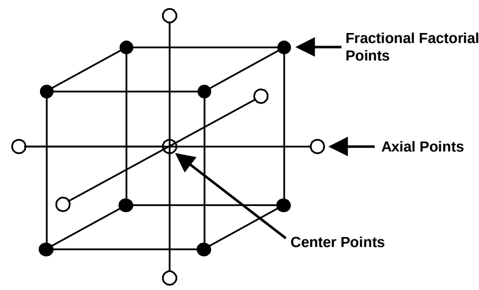


Figure 3. 3-factor Central Composite Design Illustration

Table 2. A DOE Table of 3-factor CCD

Case No.	Factor 1	Factor 2	Factor 3
1	-1	-1	-1
2	-1	-1	+1
3	-1	+1	-1
4	-1	+1	+1
5	+1	-1	-1
6	+1	-1	+1
7	+1	+1	-1
8	+1	+1	+1
9	-1	0	0
10	+1	0	0
11	0	-1	0
12	0	+1	0
13	0	0	-1
14	0	0	+1
15	0	0	0

COMPUTATIONAL AERODYNAMICS

GENERAL APPROACH

Most analyses methods for complex aerospace systems cannot be derived as precise explicit analytical functions from governing equations. These analytical methods can be classified into two categories. The first category is referred to as the empirical approach, which employs analytical expressions consisting of major aerodynamic variables (aspect ratio, Mach number, Reynolds number, etc.), obtained from regressing wind tunnel or flight test data. These results are accurate for models similar to those used for the regression. The resulting empirical expressions can be evaluated very rapidly, although accuracy is compromised when the design configuration deviates from the database used in regression.

The second category is founded on a physics-based approach, which primarily uses numerical techniques to solve (using first principles) the aerodynamic governing equations based on either potential flow theory, Euler or Navier-Stokes formulations. Although these approaches can accommodate different types of vehicles, they require delicate discretization to build an appropriate model, and take a relatively long time to complete the numerical computation. Computational Fluid Dynamics (CFD) by solving Euler or Navier-Stokes equations are a representation of the state-of-the-art physics-based techniques. However, CFD becomes unaffordable when hundreds of configurations need to be analyzed in order to obtain a satisfactory representation of the design space using existing computer technology. Instead, CFD analysis is usually utilized in the preliminary phases of the design process to provide an in-depth analysis of one or, at least, a handful of selected configurations.

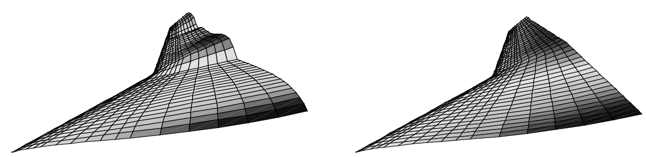
A compromise has been made in this research by using computational aerodynamic tools, mainly based on the Vortex Lattice Method (VLM) and Box methods within the scope of linearized potential flow and boundary layer theory. Both of these methods have discretization capability needed to enable the physics-based analysis of any arbitrary shape of vehicle. Furthermore, since the main concern of system studies at the conceptual/preliminary design stages is to perform a mission analysis, transients like those found in the transonic regime and high angle of attack (AOA) maneuvers are generally beyond the scope of these codes. The linearized aerodynamics obtained are thus capable of covering most of the mission profile, including take-off/landing, which can be treated with theories like the Pohalmaus[5] analogy applicable to the medium high AOA.

ELEMENTS OF THE AERODYNAMIC PREDICTION ENVIRONMENT

Several widely used, public domain aerodynamic panel codes were selected to create the aerodynamic analysis environment needed for the generation of suitable parametric drag polar relationships. This environment is comprised of a wing design program WINGDES, a wave drag minimization program AWAVE, a profile and wave drag code BDAP, a geometric modeler RAM, a Graphic User Interface (GUI) modeler VORVIEW, and an induced drag calculation code VORLAX. The tools existed in isolation, requiring separate definitions of the vehicle geometry. Based on model parameter exchanges, these tools were integrated together to reduce cycle time and facilitate the generation of total drag parametric equations needed for design space exploration. The tools selection was influenced by the fact that these are the tools used by NASA Langley's Systems Analysis Branch, the sponsoring organization for this research. In order to familiarize the reader with the approach, as well as the limitations of the tools selected, a brief description is presented next.

WINGDES

WINGDES[6] is a program developed by NASA Langley for the calculation of the optimum wing camber distribution for a given design C_L. The program seeks a mean wing camber distribution out of all possible combinations of camber surface candidates by matching the angle of attack (AOA) upper limit for full leading edge thrust of the wing with the design AOA. WINGDES is capable of analyzing both subsonic and supersonic flight regimes using a Lifting Line model for the subsonic stage, and Box method for supersonic stage. However, WINGDES has at times produced discontinuous camber distributions, and the program had to be run twice in the process, with the output from the first run feeding the second run of WINGDES, in order to obtain a smoother camber surface[7]. Despite the fact that this approach remedied most of the irregularities observed, there are still combinations encountered (Figure 4), which cannot be realized from the manufacturer's stand point of view. The problem in these cases was handled by reducing the number of control sections in the output so as to enforce linear interpolation between these sections. Currently in this proof of concept and for the simplicity of modeling, only four sections are used: root, inner kink, outer kink and tip section.



a. Original output b. Linear interpolation

Figure 4. Comparison of Wing Camber Distribution from WINGDES before and after Linear Interpolation

AWAVE

AWAVE is a program developed by Boeing and NASA Langley for the determination of fuselage cross sections which yield minimum wave drag. This is achieved by enforcing the supersonic area rule which employs Von Kármán's slender body formula. Three-dimensional aircraft component definitions, such as wing, fuselage, nacelle and empennage are input into the program. Then, the far-field wave drag[8] of the configuration is analyzed based on the equivalent bodies of those components, producing an optimum fuselage cross section distribution along its axis for design Mach numbers larger than 1.0. Sometimes, the code produces very narrow fuselages, usually under the circumstance of large wing area and airfoil thickness. These over area-ruled fuselages are obviously inappropriate for civil transport type aircraft, and a minimum diameter constraint is usually implemented as a remedy in order to bring the fuselage to the acceptable section size (Figure 5). This results obviously in fuselages with increased wave drag when compared to one that has been fully area ruled.

BDAP - The Boeing Design and Analysis Program

BDAP is a collection of several aerodynamic design and analysis programs based on linearized aerodynamics[9,10]. In this formulation, only the skin friction drag and near field wave drag calculation components are of interest.

The skin friction drag analysis is based on the assumption of flat plate under adiabatic wall, turbulent boundary layer conditions, with transition assumed to occur at the leading edge of each of the configuration components, such as wing, fuselage, nacelle, etc. Therefore, it may overestimate the friction drag. However, since the configuration is assumed to be aerodynamically smooth at this point, the overestimation might compensate for the lack of consideration of the interference drags caused by imperfections on the actual aircraft, such as doors, windows and control surface rifts.

The wave drag analysis uses the near field method by applying the Mach Box[11] technique to calculate pressure distributions on the components, in contrast to AWAVE, which applies the far field method based on supersonic area rule.

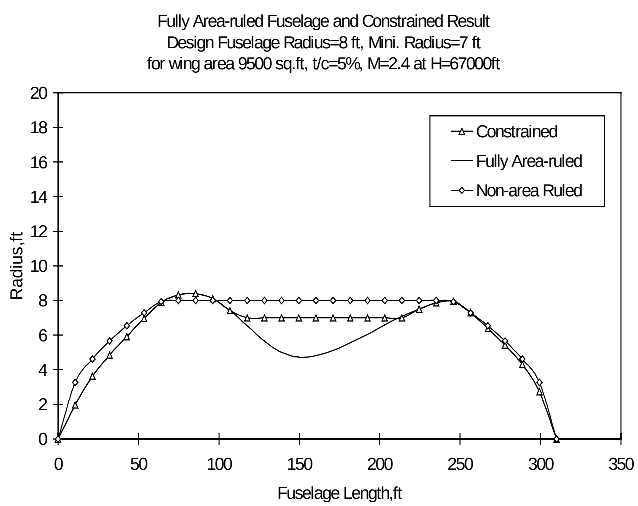


Figure 5. Fuselage Cross Section Distribution Comparison

RAM – Rapid Aircraft Modeler

RAM is a CAD (Computer Aided Design) package developed by NASA Ames Research Center to facilitate the geometric model building for computational aerodynamics. It allows parametric modeling and adjustment of a variety of aircraft components. The software is able to export a hermite file for use by the other aerodynamic codes. Vehicle renderings shown in the figures of this paper were obtained using RAM. The vehicle geometry is defined based on wing parameters optimized by WINGDES, and fuselage parameters area-ruled by AWAVE. Only wing body combinations are modeled for the purpose of generating vortex lattices, since the lift contributions of the rest of the components are neglected in the proposed approach.

VORVIEW

VORVIEW is a GUI for VORLAX, developed by Sterling Federal Systems. The program reads a geometry input file, such as the hermite file created by RAM, and translates the geometric model into a planar panel model with discretization parameters specified by the user.

VORLAX – A Generalized Vortex Lattice Method Code

VORLAX is an induced drag analysis program capable of handling both the supersonic and subsonic flight regimes. The code was developed by Lockheed in the 1970's, based on the Vortex Lattice Method (VLM)[12] of linearized potential flow theory[13]. The tool has been found to be very sensitive to the discretization parameters in the supersonic regime, leading to divergence problems for certain input combinations. For example, discrepancies in the computation of induced drag were documented as a result of small perturbations in planform discretization. For one configuration at the same flight condition (43 slices versus 45 slices taken along the wing span at M=2.4) yielded a discrepancy as high as 5 percent of the induced drag obtained. Thus, for the supersonic conditions, a balance of the spanwise slicing and chordwise slicing has to be considered since the total number of panels that VORLAX is capable of analyzing is fixed at 2000. Discrepancies and overestimations were also detected due to the presence of a gap between the wing and fuselage panel (Figure 6). This problem can usually be avoided by increasing the number of slices along the wing span.

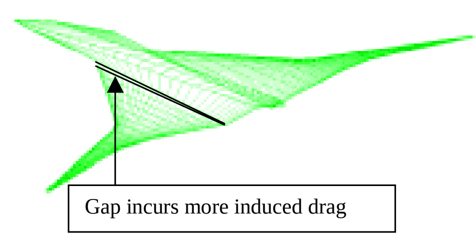


Figure 6. Panel Modeling Sensitivity Issues

CODE VALIDATION

The codes just described have been used by various organizations, and have been validated using experimental data. It would be beneficial to validate some of these codes for the purpose of understanding the their limitations and tendencies for this study. The BDAP and VORLAX programs were validated against an HSCT-like aircraft model (Figure 7), which was tested in a wind tunnel at NASA Langley Research Center[14]. The results of the validation are described next.

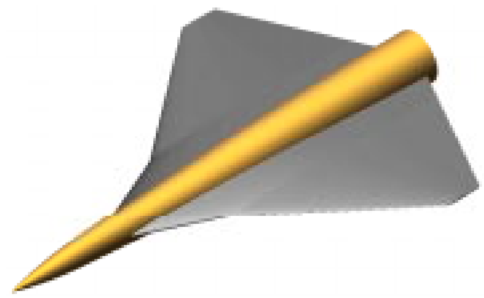


Figure 7. HSCT-like Model Tested by NASA Langley

Data obtained from this wind tunnel test and output from the codes considered were compared. It was found that VORLAX overestimates lift characteristics due to its potential flow theoretical basis, and BDAP overestimates friction drag due to the all-turbulent flat plate assumption. As the Mach number approaches the transonic regime from the supersonic side, BDAP's accuracy appears to degrade. At increased Mach number, around M=2.0, the codes are in good agreement with experimental data (Figure 8). Since most of the mission used for sizing purposes occurs at cruise conditions with small AOA, the analysis codes can be viewed as capable of reflecting reality with great confidence.

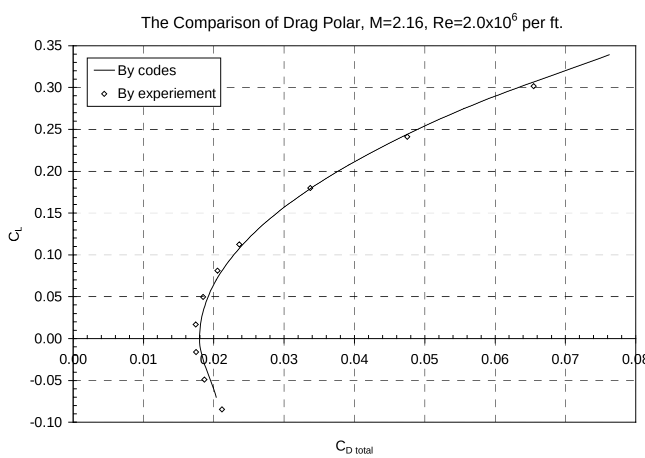


Figure 8. The Drag Polar Comparison between Experimental and Computed Data

IMPLEMENTATION PROCEDURE

The improved procedure for aerodynamic RSE generation is exemplified with an HSCT configuration as described in the following step by step process. The HSCT is a next generation supersonic civil transport proposed by the United States. It caters to the demand for shortening the travel times of current subsonic jets for international routes, especially for oceanic routes to the Pacific rim and European destinations. The design cruise Mach number used in this research is 2.4, which was selected by High Speed Research Committee[15]. With few similar existing design cases, such as the Concorde and the Russian Tu-144, the HSCT aerodynamic characteristics cannot be based strictly on empirical data. A physics-based approach appears to be the preferred course of action here since reasonably accurate data for a variety of design permutations are needed.

PROBLEM IDENTIFICATION

Before employment of the RSM technique begins, the specific problem examined must be defined and formulated. The Ishikawa diagram helps the design team to identify all potential contributors to a problem, and groups them according to a cause and effect pattern. This brainstorming tool facilitates the understanding of the relationships of all the contributory elements towards a desired response, in this case the contributing factors to the drag polar. Figure 9 is an illustration of one such diagram.

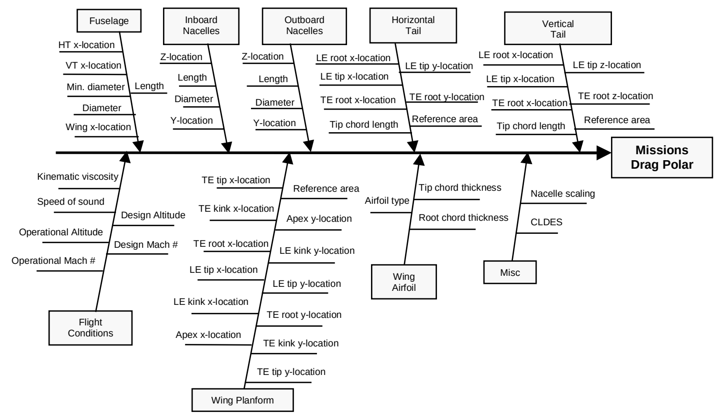


Figure 9. Cause Effect Diagram of Input Variables Used by Codes Selected

The drag polar is expressed in the form in Equation 2,

$$C_D = C_{D_0} + K_1 \cdot C_L + K_2 \cdot C_L^2 \qquad (2)$$

where C_{D0} represents a combination of the friction drag coefficient, C_{Df} and the wave drag coefficient, C_{Dw} (for supersonic only). $K_1 \cdot C_L$ is an approximation of interference drag, while $K_2 \cdot C_L^2$ is the induced drag coefficient. C_{Df} and C_{Dw} are obtained from BDAP while $K_1 \cdot C_L$ and $K_2 \cdot C_L^2$ are computed in VORLAX. These codes need to be run for a variety of combinations to yield the needed coefficients. C_{D0}, K_1 and K_2 are a function of all key design parameters and flight conditions. A total of three equations were thus generated, one for each variable. The resulting approximations are valid only for the chosen baseline configuration and the design space around it as defined by the ranges selected for each of the independent variables.

COMPUTATIONAL MODELING

A typical mission profile (Figure 10) encompassing both a supersonic and subsonic cruise segment was selected to provide a valid range of flight conditions for the aerodynamic computations. It was determined that the character of the drag polar depended primarily on Mach number and operational altitude which dominated the variability of the responses. Therefore, the mission profile was approximated by grids of Mach number and altitude combinations, and RSEs were generated at each grid point. The required mission points were then obtained by linear interpolations between these predefined grids. In this investigation, a mission grid comprised of seven Mach number and ten altitude combinations was used.

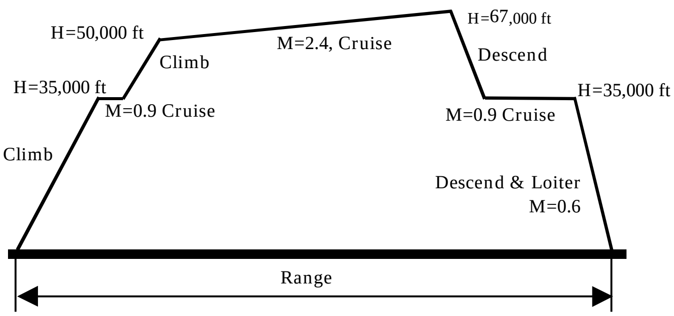


Figure 10. Representation Mission Profile Used to Size a Notional HSCT

The list of variables of interest was identified based on the Ishikawa diagram. A geometric planform for a notional HSCT is shown in Figure 11, and it depicts the most important variables examined. Altogether, 26 variables were identified as needed to model the HSCT at this level (Table 3). Non-dimensional geometric parameters were used for the purpose of sizing the vehicle up and down with the wing reference area. Wing parameters were non-dimensionalized by the semi-span, the horizontal tail ones by its semi-span, the vertical tail by its length, while the relative relocations of the various components with respect to fuselage by the fuselage length.

The variables were subsequently bounded by the maximum and minimum values which are either attainable or realizable. These ranges also define the range of applicability of the obtained RSEs. Generally speaking, the lower and upper limits should be maintained within a reasonable range. Since quadratic representations were assumed for these RSEs, widening the ranges too much may render the assumption of the form of the RSE invalid. For example, wing area is an important factor, and its range has to be maintained within reasonable limits for the baseline configuration. Furthermore, disproportional widening of ranges may dwarf the effects of the rest of factors. Some sample wing planforms investigated are shown in Figure 12.

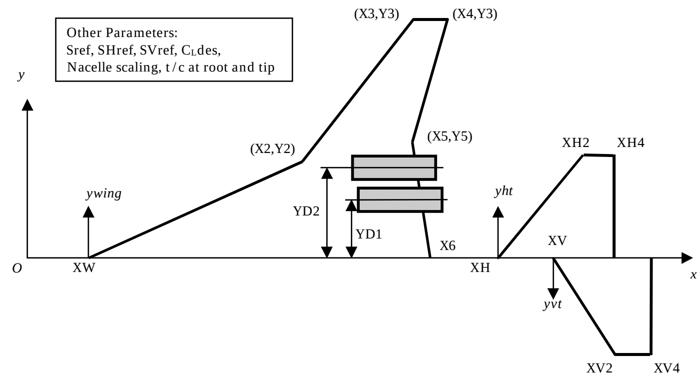


Figure 11. Geometric Variables Selected for the HSCT Stduy

Table 3. Pertinent Aerodynamic Variables Considered

Group	Name	Definition	Min	Maxi	Unit
Mission	CLDES	Design lift coefficient	0.08	0.12	nondimen
	MOPER	Operational Mach Number	0.3	2.4	
	HOPER	Operational Altitude	0	67000	ft
Wing	X2	Kink LE x-location	1.54	1.69	nondimen
	X3	Tip LE x-location	2.1	2.36	nondimen
	X4	Tip LE x-location	2.4	2.58	nondimen
	X5	Kink TE x-location	2.19	2.37	nondimen
	X6	Root TE x-location	2.18	2.5	nondimen
	Y2	LE Kink y-location	0.44	0.58	nondimen
	Y5	TE Kink y-location	0.43	0.6	nondimen
	TCR	Thickness to chord ratio at Root	3	5	% Chord
	TCT	Thickness to chord ratio at Tip	2	4	% Chord
	SREF	Wing reference area	7000	9000	sq. ft
Horizon. Tail	XH2	HT Tip LE x-location	0.95	1.73	nondimen
	XH4	HT Root TE x-location	1.31	2.08	nondimen
	CTHTND	HT tip chord length	0.29	0.51	nondimen
	SHREF	HT reference area	400	700	sq. ft
Vertical Tail	XV2	VT Tip LE x-location	0.84	1.73	nondimen
	XV4	VT Root TE x-location	1	1.92	nondimen
	CTVTND	VT tip chord length	0.38	0.43	nondimen
	SVREF	VT reference area	350	550	sq. ft
Nacelle	NACSCAL	Factor for scaling the nacelles	0.9	1.1	nondimen
	YD1	Inboard nacelle y-location	0.24	0.3	nondimen
	YD2	Outboard nacelle y-location	0.49	0.55	nondimen
Rellocation	XW	Wing apex loc. in % relat. to fuse	0.22	0.28	nondimen
	XH	HT apex loc. in % relat. to fuse	0.82	0.874	nondimen
	XV	VT apex loc. in % relat. to fuse	0.81	0.864	nondimen
	XCG	X-location center of gravity	179.8	201.5	ft

The airfoils used in this research are NACA 000X for the subsonic leading edge and biconvex for the supersonic leading edge. The wing itself will have a twist and camber distribution based on the results obtained by WINGDES, which attempts to find the optimal distribution that minimizes the overall wing induced drag. WINGDES analyzes only the wing planform, while AWAVE and BDAP analyze the entire aircraft model to optimize the fuselage cross section and calculate of friction drag and wave drag respectively. The wing fuselage combination is used in the VORLAX, which analyzes the induced drag for the aircraft.

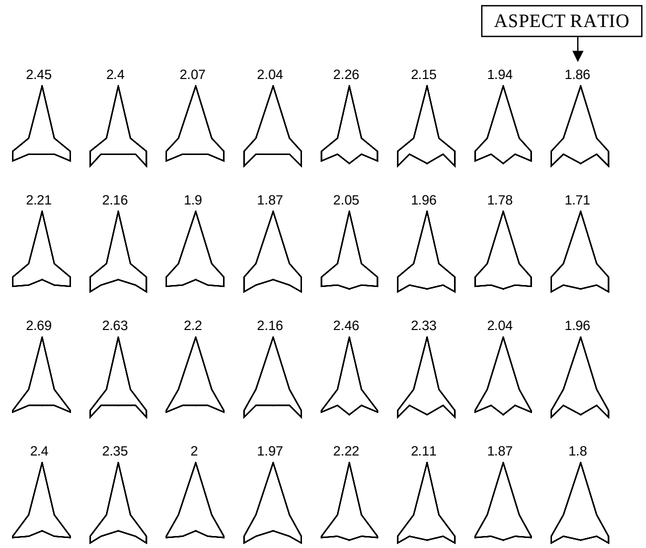
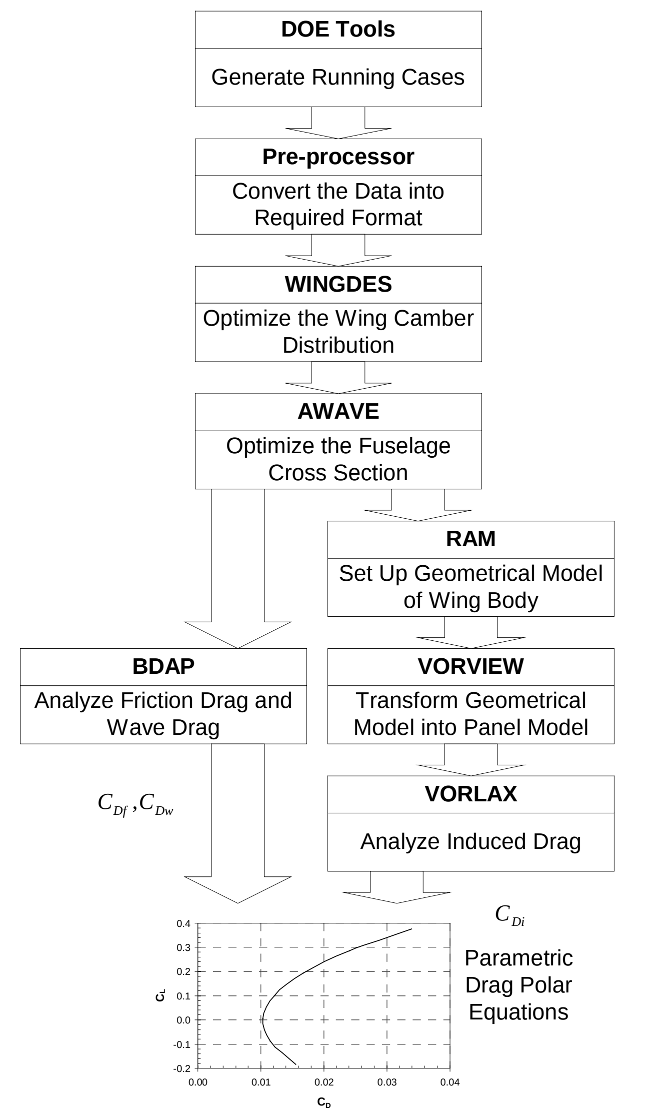


Figure 12. Sample HSCT Wing Planforms Investigated for the Creation of Parametric Drag Polar Equations



$$C_D = C_{D0} + k_1 C_L + k_2 C_L^2$$

Figure 13. Environment for the Generation of Parametric Drag Polar Equations

COMPUTATIONAL ARCHITECTURE

Based on our experience, most of the legacy tools selected to perform the aerodynamic analyses are not very robust. This implies that there is a good chance that the codes will often fail for some of the examined combinations. As a remedy a few exploratory cases should be tried in order to identify the most critical conditions in the design space which lead to program failures, convergence errors, singularities, etc. Once the boundaries are determined the DOE approach calls for the execution of hundreds of cases which could take several days on workstations, such as the IBM RS6000 3CT or the SGI Indigo 2. Therefore, establishment of a fully automated computational architecture is essential to achieve efficiency in RSE generation. Human intervention should only be required when needed, such as in DOE type selection, launching batch processes, and results verification.

The computational architecture created to account for most of the issues described above is shown in Figure 13. In the process, WINGDES is run first to obtain an optimized camber surface, followed by AWAVE to obtain the shape of the area-ruled fuselage. The BDAP and VORLAX packages are called next to determine the parasite and induced drag respectively. Several in-house linking programs and scripts have been developed to connect these aerodynamic analysis codes and facilitate the transfer of data between them. Pre- and post-processors have been written to translate data inputs to the correct formats required by the various codes. This architecture is designed for modular capability. For example, the pre-processor for RAM can be changed to that for AERO2S[10], which is capable of low speed aerodynamic analysis, if take-off and landing drag polars are needed. Once the architecture was established and tested, the whole process was considered automated for generating RSEs according to any DOE setup desired.

SCREENING TEST

Since 26 variables were identified as potential contributors to the drag calculations, it was imperative that a screening test be performed to identify the most significant contributors. This becomes evident when one considers that for a quadratic RSE with 26 variables, 3^{26} combinations must be examined for a Full Factorial design. The technique used to identify the variables that contribute the most is based on a well-known statistical method called ANOVA. According to this method, appropriate ranges are selected for each variable, and a 2-level DOE is run to determine their impact to the variability of each response. In this way, generation of a handful of cases in conjunction with the employment of the Pareto Principle which ranks each variable according to its relative contribution to the variability of a given response. In an abstract definition, the Pareto Principle simply states that usually a small subset (~20%) of factors among all factors in a system control the majority (~80%) of the variability of the outcome. The results of this ANOVA and the corresponding Pareto charts, illustrated in Figure 14 for C_{D0}, were generated and analyzed using the statistical analysis package JMP®[8]. In interpreting the Pareto chart in Figure 14, please note that each of the normalized bars indicates the magnitude of the relative contribution to the variability of the response for each variable while the solid line denotes their cumulative contributions.

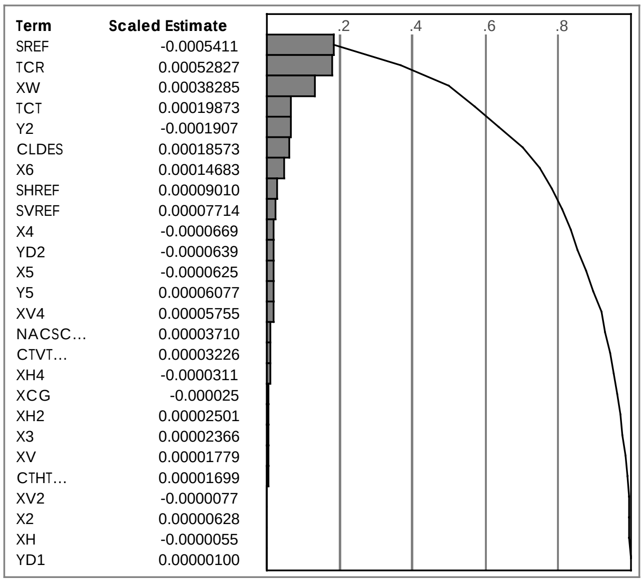


Figure 14. Identification of Most Significant Contributors to the Variability of C_{D0} at M=2.4

Inspection of the results yielded by this screening test performed on all three responses K_2, K_1 and C_{D0} indicated that 16 common variables (Table 4) were significant enough to retain in the formulation of these RSEs.

Table 4. Most Influential Contributor Variables Selected for RSE Generation

No.	Name	Group	Min.	Max.	Unit
1	Y2	WING	0.44	0.58	
2	X6	WING	2.18	2.5	
3	X2	WING	1.54	1.69	
4	XW	RELLOC	0.22	0.28	
5	X4	WING	2.4	2.58	
6	X3	WING	2.1	2.36	
7	X5	WING	2.19	2.37	
8	Y5	WING	0.43	0.6	
9	SREF	WING	7000	9000	sq.ft
10	CLDES	MISS	0.08	0.12	
11	TCR	WING	3	5	%
12	TCT	WING	2	4	%
13	SHREF	HT	400	700	sq.ft
14	NACSCAL	FUSNAC	0.9	1.1	
15	SVREF	VT	350	550	sq.ft
16	YD2	FUSNAC	0.49	0.55	

RSE GENERATION

After the screening test was completed, a second order quadratic equation was assumed to be an adequate model for these three responses. Since this is just a hypothesis, a least squares method was used to provide the best fit. Allowing for an adequate number of degrees of freedom, a 16 variable face-centered Central Composite Design of Experiment (CCD) was selected. This specific setup is created by merging together a resolution IV Fractional Factorial design with the center point of the envisioned hyper-cube and a set of face-centered axial points. This setup requires 289 cases to be run in order to create the desired RSEs. Each of the 289 cases for seven Mach numbers and ten altitude combinations takes about 15 minutes on the RS6000 and the SGI workstations. The computation requires several days for the entire matrix of possible Mach number and altitude combinations to be created. But it does represent a one time investment in the sense that these equations are created once, and if done correctly, they hold true for all studies of similar vehicles within the selected ranges.

Since some of the legacy tools used often have their own optimizers embedded inside, convergence problems were encountered during the running of the codes for some of the input combinations. The problems were fixed manually by adjusting the discretization parameters, and repeating the analysis for the cases where problems were detected. A very few occurrences were also observed where for no apparent physical reason the codes exhibited inexplicable sensitivities and the authors were forced to exclude them from the analysis.

All the responses of K_2, K_1 and C_{D0} for the 289 cases along with the DOE experimental table setup were input back into JMP® to generate the RSEs. A sample prediction profile at M=2.4, altitude 58,500 ft of these RSEs, generated by JMP is shown in Figure 15. The prediction profile is a graphical way to view and compare their magnitude and sensitivities. It also provides an interactive method allowing the user to observe the change of the responses immediately. The “-1” and “1”s represent the lower and upper bounds of the variables. An interface subroutine has been created to convert these non-dimensionalized settings to their actual dimensional values for the creation of the aerodynamic tables for use in sizing and synthesis.

To assess the accuracy of these equations with respect to the codes that created them, which are now considered to be the true models, verification tests were performed. The first one is referred to as the R^2 fit and it measures the variation of the fit with respect to the 289 measured points. An R^2 of 1 in this case indicates a perfect fit. R^2 can be computed using Equation 3:

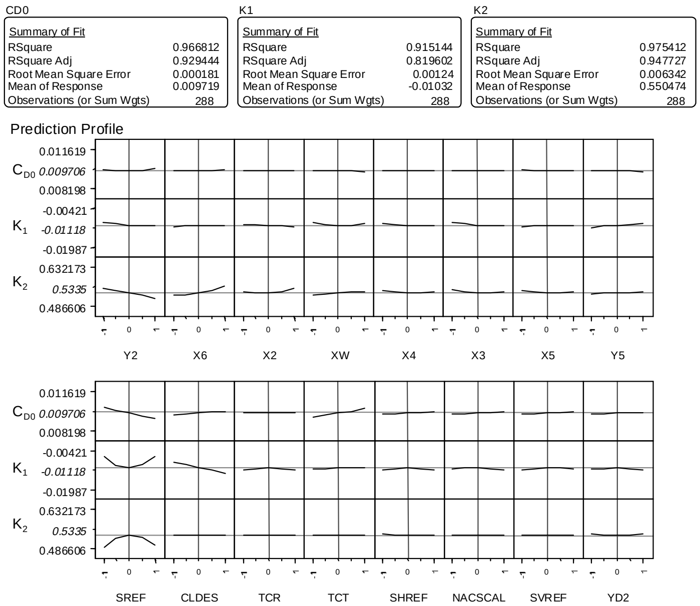


Figure 15. Environment for the Prediction and Sensitivity Analysis of Drag Contributors (M = 2.4)

$$R^2 = 1 - \frac{\sum_{i=1}^{n}(Y_i - \hat{Y}_i)^2}{\sum_{i=1}^{n}(Y_i - \bar{Y})^2} \tag{3}$$

where Y_i is the given value of response Y, $\hat{Y}_i$ is the prediction of the response and $\bar{Y}$ is the mean of response Y observations.

All three responses were found to have excellent fits in the subsonic regime with K_1, K_2, and C_{D0} having R^2 values of 99%. Furthermore, K_1 was found to be a very small contributor to the drag and for all practical purposes, it can be ignored in the subsonic regime. However, for the supersonic flight conditions, the R^2 for K_2 and C_{D0} drop to about 96~97%, and K_1's significance increases. The discrepancies were traced back to the way VORLAX handles and accounts for the interactions between planform discretization and Mach number regions of influence.

RSE VALIDATION

The RSEs have to be tested before they can be applied for practical use. For basic testing, the accuracy of the RSE is validated against *experimental* points, i.e. the data points used in RSE generation. This can be viewed using JMP (Figure 16). The solid line is the ideal fit. The lower and upper dashed lines about the solid line are the lower and upper 95% confidence limit of the predicted value. The chart indicates a good fit of the RSE for the tested points.

A residual plot shown in Figure 17 is used to observe the error distribution of the RSE regression for K_2. The residuals in this case are the differences between the actual responses and the responses accounted in the RSEs. The residual plot bears patterns that cannot be predicted by the model. Therefore, there should be no strong patterns of correlation between the residuals and the responses for a successful regression. For this study, the points in the residual plot are expected to demonstrate a scattering pattern of normal random distribution with zero mean.

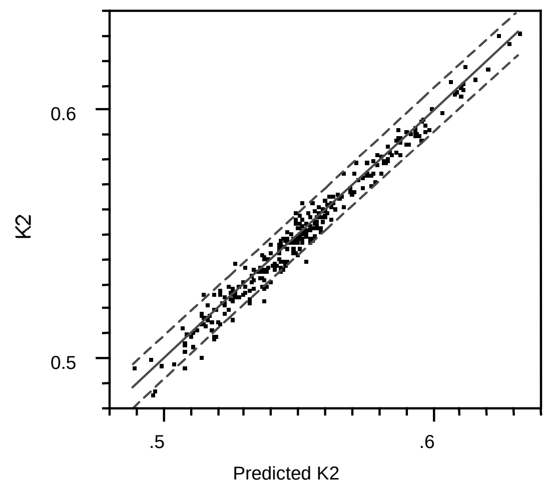


Figure 16. Representation RSE validation for K_2 at M=2.4

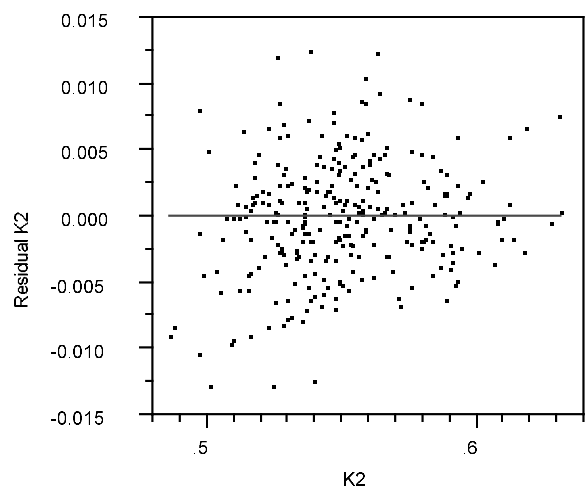


Figure 17. Residual Plot for K_2 at M=2.4

The R^2 test as was mentioned above is a good measure of model fidelity for the points examined. In most occurrences when computations are expensive, the R^2 fit is the only indicator we have. On the other hand, when the option of re-computing additional cases is not prohibitive, it is recommended that a number of random cases, other than the DOE points, are executed to assess the models fidelity throughout the entire examined space. Since our intention was to create these equations for design purposes and use them again and again, the added computational effort was well worth the effort and time spent.

The RSE has been shown to have a good correlation with all the *experimented* points, i.e. lower, higher limits and mid-values. However, this does not guarantee good fits with the RSE for points which arbitrarily fall between the lower and higher limits. For this reason, a random test was selected to verify the RSEs in a strict way. Therefore, 289 random cases were picked between lower and upper limits of the values of all 16 variables. The responses K_2, K_1 and C_{D0} were calculated by both the RSEs and the aerodynamic analysis package. Overall L/D values from two approaches were compared based on the design C_L at 0.1.

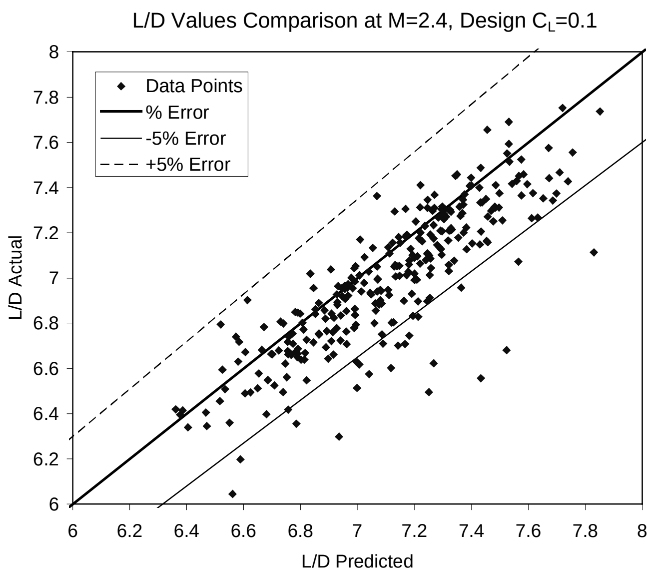


Figure 18. L/D Values Comparison for M=2.4

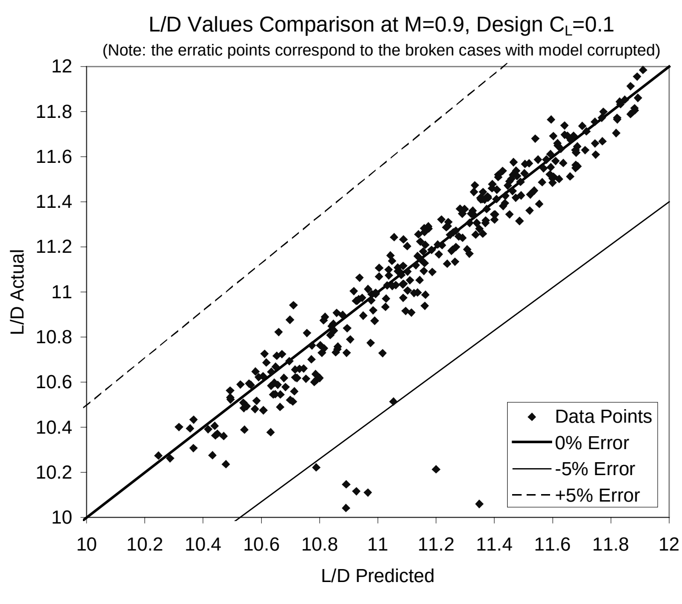


Figure 19. L/D Values Comparison for M=0.9

Using the results from the set of random samples examined, it was observed that the quadratic form assumed for these RSEs seems to be missing some higher order effects that appear to be present. Generally, this random testing technique does provide more clear information about the RSE accuracies and provides a measure of confidence in terms of the deviations of RSEs from the actual responses. For the L/D values comparison at M=2.4 in Figure 18, the majority of the tested points fall in between the lower and higher 95% confidence limit, which is a proof of appropriate accuracy for engineering usage.

Comparison of the data obtained from the lift to drag measurements in the subsonic regime yielded that the RSE predicted values are very close to the values generated from running the actual codes (Figure 19). It was evident that the linear codes used for the creation of the metamodels were better suited for prediction of the subsonic conditions. The reason for this is that the presence and movement of Mach line in linearized supersonic flow analyses are strongly influenced by the discretization of the geometric model. In VORLAX, the influence area for a panel (L.E. vortex line at ¼ chord and control point at the of ¾ chord[16]) is within the two Mach lines starting at the tip of two trailing vortices. Although the Mach lines are represented as straight lines, the region of influence captured by these Mach lines is represented by a set of discrete panels. These panels along the each side of the Mach lines have to be classified by whether they are inside or outside the region of influence. Therefore, actual region of influence is enclosed by jagged lines.

The region of influence will be varied if discretization parameters are changed even though the same planform and Mach number are maintained at the same time since *different* panels are likely to be intersected with the same Mach lines. The variation of region of influence is also present when the planform or Mach number is varied due to the same reason. Generally, the actual region of influence deviates from the theoretical one formed by straight Mach lines, depending on the factors, such as Mach line sweep, planform geometry, discretization parameters, etc.

In this way, pressure variations are produced in the supersonic regime, which cause the variations of induced drag for the RSE fits. For example, nearly 5% variation in the induced drag responses was observed when VORLAX was used to analyze one configuration at M=2.4 according to different combinations of valid discretization parameters. On the other hand, region of influence is not varied for different Mach numbers in the subsonic regime, since the region of influence always includes the entire panels. Refined discretization parameters could alleviate the problem for the supersonic condition, but it is constrained by the code's capability (maximum number of panels), and it may cause other problems, like the increase of running time, and more singular points.

The drag polar in the form of L/D values for certain C_L will be provided to system level studies. The error of the L/D prediction can also be provided after the random testing has been performed. The errors define as the difference between actual and predicted values of the

289 random cases were fed into Crystal Ball® in order to obtain a probability distribution function (Figure 21). This provides the variability of the RSEs generated for system RDS, which treats this kind of variability as noise variables. By this means, probabilistic analysis with variance of meta-model accuracy provided becomes more meaningful for robust design than deterministic approach with just simple L/D.

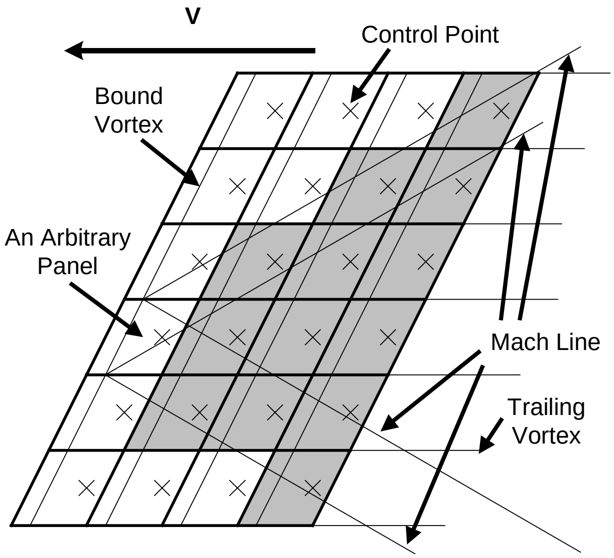


Figure 20. Illustration of Potential Modeling Problem in Supersonic Flight due to the Misrepresentation of the Region of Influence

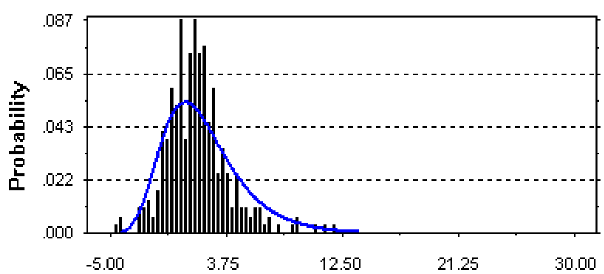


Figure 21. Probability Density Function (PDF) for RSE Error of L/D at M=2.4, Design C_L=0.1

APPLICATIONS OF RSE

RSEs have a variety of applications in design. Disciplinary RSEs can be incorporated into system level design studies. The RSEs can also be used solely at the disciplinary level for detailed technological development.

As an example of a deterministic application, RSEs can be combined with optimization routines to find an optimal configuration corresponding to maximum L/D at design CL. Especially, the RSEs were input to Excel® and objective function defined as a combined supersonic and subsonic L/D with weighting factors. The supersonic L/D that is at M=2.4 was weighted by 85%, while the subsonic at M=0.9 was weighted by 15%. This selection is based on the proposed mission profile of 85% supersonic cruise and 15% subsonic cruise. Once the RSEs were implemented in the spreadsheets and objective defined, the Solver in Excel® was used to seek the variable values which correspond to a maximum combined L/D. The aerodynamic optimum configuration for a notional HSCT is listed in Table 5, with a rendered picture shown in Figure 22.

Table 5. Variable Values For Maximum Combined L/D

No.	Name	Group	Optimized	Unit
1	Y2	WING	0.5216	
2	X6	WING	2.18	
3	X2	WING	1.6298	
4	XW	RELLOC	0.28	
5	X4	WING	2.58	
6	X3	WING	2.36	
7	X5	WING	2.37	
8	Y5	WING	0.6	
9	SREF	WING	9000	sq.ft
10	CLDES	MISS	0.12	
11	TCR	WING	3.8832	% t/c
12	TCT	WING	2	% t/c
13	SHREF	HT	400	sq.ft
14	NACSCAL	FUSNAC	1.1	
15	SVREF	VT	350	sq.ft
16	YD2	FUSNAC	0.49	

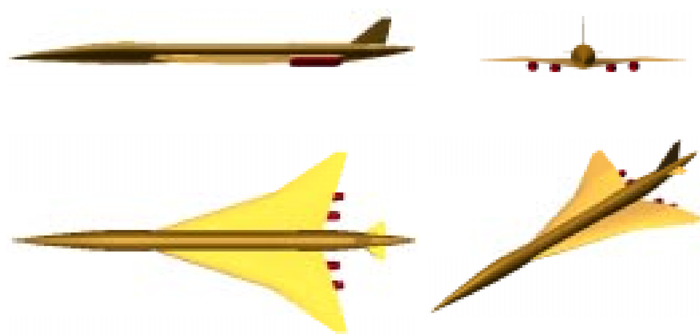


Figure 22. Different Views of a Notional HSCT Optimized for Combined L/D

The RSEs can also be used to analyze the impact on other disciplines resulting from aerodynamic influences. For example, the size and location of the empennage depend on the stability and control requirements. If the variations of size and location of empennage are treated as uncertainties, the RSEs can be used in a Monte-Carlo simulation to simulate the influence of the empenage, and a "robust" solution less susceptible to the variability of the empennage size and location can be found.

FUTURE WORK

The RSEs developed here are based on the hypothesis that their underlying behavior can be captured by a quadratic polynomial representation. If higher order terms prevail in any of the computer codes used, the second order model may not be an appropriate meta-model to replace those codes. A series of independent or dependent variable transformations may be needed with or preferably without running any more cases. If all fails, other metamodels, such as Neural Networks, may be more appropriate. However, if the variable ranges can be subdivided, variable grids can be generated, and an RSE can be generated for each grid. Therefore, several RSEs can be used to approximate high order terms[17]. In this case, physical understanding of the tendency of response (sensitivity to variable ranges) is required in order to make appropriate subdivisions. Since the RSE approach is generally inexpensive to generate with appropriate codes used, highly sophisticated problems can also be modeled with multiple RSEs within different variable grids.

The current approach to generate the RSEs also needs improvements in the modeling process. At present, the modeling for each of the aerodynamic codes is separated. Format transformations between models are used to connect one code to another, which causes complicated model exchanges. In order to simplify the modeling, a single geometric model, like the RAM model, can be parametrically generated as a reference at the beginning of the process. The computational models needed by different codes will be directly obtained from the RAM model while the modifications to the configuration by optimization codes, such as WINGDES, will be reflected in the RAM model after the optimum configuration is found. In this way, the entire process can be much tightly integrated with better modularity.

CONCLUDING REMARKS

In this investigation, our intention was to create a set of HSCT specific relationships to replace or to enhance the historical or simplified analysis modules present within the sizing and synthesis code, FLOPS. The result of the research was an environment for the rapid and accurate evaluation of aerodynamic characteristics for a family of HSCT vehicles. These characteristics are given in the form of polynomial representations as a function of the most influential variables. Because of their explicit nature, metamodels are easy/fast to use in parametric studies. Our goal was to create these equations and incorporate them into any Multidisciplinary Analysis/ Design/ Optimization environment.

A general, practical approach to generate these polynomial representations using Design of Experiments has been established and is referred to as a Response Surface Method (RSM). This method includes problem identification, computational modeling, computational architecture, screening test, RSE generation and validation. The computational methods for generating RSEs are based on linearized aerodynamics and boundary layer theory. Several appropriate computational aerodynamic codes, including wing design, area-rule optimization, parasite and induced drag analyses are selected based on the compromise of accuracy and efficiency concerned. The RSEs for a notional HSCT with assumed mission were generated with satisfactory accuracy, which is used as an example to show the implementation process as well as the techniques related to the generation of RSEs. Although the process shown in this paper is primarily concerned with the aerodynamic issues, this general procedure can be applied to the generation of RSEs to approximate analysis codes of other disciplines. In this way, a variety of disciplinary knowledge has been acquired as a series of compact quadratic polynomial equations for system level studies, resulting in a highly efficient, inexpensive platform to support the design space exploration. The RSEs application to studies inside their discipline is possible but not so productive as compared to their contributions to complex system studies in a Multi-disciplinary Analysis/Optimization environment.

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