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AEROTHERMODYNAMIC FLIGHT SIMULATION CAPABILITIES FOR AEROSPACE VEHICLES

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Abstract

Aerothermodynamics, encompassing aerodynamics, aeroheating, and fluid dynamics and physical processes, is the genesis for the design and development of advanced space transportation vehicles and provides crucial information to other disciplines such as structures, materials, propulsion, avionics, and guidance, navigation and control. Sources of aerothermodynamic information are ground-based facilities, Computational Fluid Dynamic (CFD) and engineering computer codes, and flight experiments. Utilization of this aerothermodynamic triad provides the optimum aerothermodynamic design to safely satisfy mission requirements while reducing design conservatism, risk and cost. The iterative aerothermodynamic process for initial screening/assessment of aerospace vehicle concepts, optimization of aerolines to achieve/exceed mission requirements, and benchmark studies for final design and establishment of the flight data book are reviewed. Aerothermodynamic methodology centered on synergism between ground-based testing and CFD predictions is discussed for various flow regimes encountered by a vehicle entering the Earth's atmosphere from low Earth orbit. An overview of the resources/infrastructure required to provide accurate/creditable aerothermodynamic information in a timely manner is presented. Impacts on Langley's aerothermodynamic capabilities due to recent programmatic changes such as Center reorganization, downsizing, outsourcing, industry (as opposed to NASA) led programs, and so forth are discussed. Sample applications of these capabilities to high Agency priority, fast-paced programs such as Reusable Launch Vehicle (RLV)/X-33 Phases I and II, X-34, Hyper-X and X-38 are presented and lessons learned discussed. Lastly, enhancements in ground-based testing/CFD capabilities necessary to partially/fully satisfy future requirements are addressed.

Introduction

From a NASA perspective, funding and personnel levels for aerothermodynamics (defined herein as encompassing aerodynamics, aeroheating and fluid dynamics and physical processes) have been cyclic since the initial buildup for the Apollo program. The downturn in the post-Apollo period was reversed by the shuttle orbiter and, to a much lesser extent, the Viking programs. The low point in aerothermodynamics in this country is generally recognized to have occurred in the mid 1980s. For example, the number of researchers performing experimental aeroheating studies, either diagnostic development or application, at the NASA Langley Research Center (LaRC) fell to the equivalent of less than two people full-time. A tremendous loss of aerothermodynamic corporate knowledge occurred in the late 1970s and early 1980s as personnel moved to other, viable disciplines. Without funding advocacy, hypersonic wind tunnels fell into disrepair due to prolonged neglect, although upgrades were achieved in instrumentation, signal conditioning and data acquisition systems due to advances in solid-state technology and computer systems. Around 1986, an upward trend was ushered in by the Agency's Aeroassist Flight Experiment (AFE) program followed closely by the National Aero-Space Plane (NASP) program.

The Langley experimental aerothermodynamic capability benefited explicitly and implicitly from NASP. For example, a major Construction of Facilities (CoF) project to upgrade/enhance several hypersonic wind tunnels was approved and implemented in 1989 after 6 consecutive years of unsuccessful advocacy. The success was possible due to NASP requirements and corresponding advocacy. In this same time period,

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significant upgrades to instrumentation, signal conditioning and data acquisition systems were achieved for hypersonic wind tunnels via the Agency Aeronautical Wind Tunnel Revitalization Program and various other funding sources. The CFD community also benefited substantially from the NASP program, as this program placed strong emphasis on the application of CFD for aerothermodynamic, including propulsion/airframe integration, design and development.

The most significant benefit of NASP to Langley's aerothermodynamic competency was the associated buildup in civil servant personnel; i.e., the hiring of engineers/scientists. At the beginning of the NASP program, approximately 15 Langley engineers/scientists performed experimental hypersonic aerodynamic/aeroheating/fluid dynamic research, and only 3 were less than 45 years of age. The number of experimental aerothermodynamicists was increased substantially and the age distribution shifted significantly to the left (lowered) via hiring of highly educated, motivated engineers/scientists. These individuals, hired in late 1980-early 1990 time frame, now represent a major portion of the aerothermodynamic foundation at Langley. Even with this buildup, the aerothermodynamic community at Langley and within the country is but a small fraction of what it was in the 1960s and early 1970s.

Another downward turn in the aerothermodynamic cycle occurred with the demise of the NASP program and decline of the Personnel Launch Support (PLS) HL-20 activity. The downward trend was quickly reversed, however, by several Agency new initiations designed to build and fly advanced technology demonstrator vehicles for the Reusable Launch Vehicle (RLV) and for the technological advancement of hypersonic airbreathing vehicle concepts. RLV/X-33 Phase I and X-34 programs were initiated in the spring of 1995 for a 14-month period. During this period, aerothermodynamic screenings/assessments/optimizations were performed in parallel at Langley for four different industry concepts. Model fabrication shops were operated 24 hours per day and on weekends; hypersonic wind tunnels were operated extended/double shifts and weekends; and experimental and computational aerothermodynamicists worked diligently to provide huge volumes of aerothermo-dynamic information in this relatively short period of 14 months. This effort in 1995-1996 may well represent the most productive period for the generation of aerothermodynamic data/information at Langley. The high intensity level continued into X-33 Phase II, the second phase of X-34 and new additions including Hyper-X and X-38 in 1996.

This paper will review aerothermodynamic capabilities at the NASA Langley Research Center, present recent sample applications of the capabilities, and discuss plans to provide aerothermodynamic information faster and better with fewer resources. The review includes: (1) the highly iterative aerothermodynamic process for screening initial aerospace vehicle concepts, optimization of aerolines via parametric studies, and benchmark data for final design and establishment of the flight data book; (2) aerothermodynamic methodology which translates to the synergism between ground-based testing and CFD predictions throughout entry into the Earth's atmosphere; and (3) the resources/infrastructure required to provide accurate/creditable aerothermodynamic information in a timely manner. Impacts on Langley's aerothermodynamic capabilities due to recent (since 1993-1994) programmatic changes such as Center reorganization, downsizing, outsourcing, etc. are discussed. Sample applications of Langley's aerothermodynamic capabilities to the X-33, X-34, Hyper-X and X-38 programs are presented along with lessons learned.

Aerothermodynamic Background/Review

Definition

Aerothermodynamics is defined (Fig. 1) herein as a blending of three basic disciplines: (1) aerodynamics, involving forces, moments and pressure loads on the vehicle across the speed regime from take-off to orbit or beyond, and entry to landing (e.g., Mach 0.1 to 40); (2) aeroheating, which includes convective and radiative heat-transfer rates for a configuration at flight condition; and (3) fluid dynamics and physical processes which involve complex flow phenomena from the free molecular regime throughout the continuum regime (e.g., boundary layer/shear layer transition to turbulence, shock/shock interactions, shock impingement, flow separation and reattachment, plume-flowfield-surface interactions, etc.) and processes associated with high temperature gases (e.g., chemical reactions, transport processes, radiation, coupled relaxation and/or excitation processes, thermodynamic nonequilibrium, gas-surface interactions, etc.). From the perspective of aerospace vehicle designers, the emphasis is on aerodynamic performance, stability and control and vehicle surface pressure, shear and heating loads. Fluid dynamics and physical processes are closely coupled to aerodynamic forces and moments and aeroheating, and provide insight to phenomena observed at the surface by improved understanding of the flowfield about the vehicle. This three-part

definition of aerothermodynamics is applied across the subsonic-to-hypersonic speed regimes for the full spectrum of aerospace vehicle configurations.

Importance

Aerothermodynamics is the genesis for the design, development and flight of space transportation vehicles, planetary probes, and Earth return vehicles. It provides crucial information for the optimum flyability and survivability of aerospace vehicles (Fig. 2), and to other key disciplines such as structures, materials including thermal protection systems (TPS), avionics, guidance, navigation and control, propulsion/airframe integration, etc. The aerothermodynamic challenge is to provide the optimum design to safely satisfy mission requirements and reduce design conservatism, risk and cost; i.e., the optimum flying vehicle with minimum structural and heating loads and reduced operation costs.

Sources

The three sources of aerothermodynamic information (Fig. 1) are: (1) ground-based facilities, (2) computational fluid dynamics (CFD) and/or engineering computer codes, and (3) flight experiments.

Ground-based facilities have provided and continue to provide the majority of fundamental aerothermodynamic information for aerospace vehicle concepts (Fig. 3). As is well recognized, duplication of all flight conditions is not possible via ground-based testing. Experimental aerothermodynamicists are required to resort to the simulation of important flight parameters (e.g. Mach number, Reynolds number and ratio of specific heats (gamma)) for subsonic to high hypersonic conditions and to the duplication of certain aspects of flight (primarily velocity and the product of density and characteristic length). Although no one facility can provide all the aerothermodynamic information required for the design of a vehicle, the combination of several facilities collectively providing ranges of Mach number, Reynolds number and gamma can simulate a major portion of the flight trajectory. The success enjoyed by the Apollo, Shuttle Orbiter, and other hypersonic flight programs, for which the vast majority of the aerothermodynamic data used in the design and in the flight of the vehicle originated from ground-based facilities, is indicative of the applicability and importance of ground-based experimentation. Ground-based facilities such as subsonic, transonic, supersonic and hypersonic wind tunnels represent a tried and proven approach for providing accurate/creditable aerothermodynamic information.

CFD capabilities were not sufficiently mature in the early 1970s to contribute appreciably to the aerothermodynamic assessment/optimization of shuttle orbiter concepts nor benchmarking for flight. However, significant advances have been made in CFD since that period and particularly in the last decade. CFD has contributed significantly to the aerothermodynamic design and successful flight of planetary probes (e.g., Mars Pathfinder; Gnoffo, et al., 1998) and to aerospace vehicles having relatively simple shapes (Fig. 4). It is now in a position to contribute significantly to the design of advanced aerospace vehicles having complex geometries. CFD significantly complements informa-tion from ground-based facilities. Validated CFD may be used to predict surface and flow field conditions for the full-scale vehicle at atmospheric conditions (i.e., density, temperature and molecular weight) for points along the trajectory. The highest confidence in ground-based data and/or the pre-flight data book occurs when experimental and computational results are in full agreement. (Recent overviews of aerothermodynamic computational capabilities are provided by Kumar et al., 1997 and Gnoffo et al., 1997.)

Flight experiments represent the third source of aerothermodynamic information (Fig. 1). These experiments are generally performed with sufficient instrumentation to measure local phenomena (e.g., catalytic versus noncatalytic heating at hypervelocity conditions), but not sufficiently instrumented to accurately model spatially and temperally complex global phenomena such as laminar to turbulent boundary layer transition and shear layer reattachment. The advantages of flight experiments are well recognized; the primary disadvantages are they require considerable time and cost to perform.

Process

Initial concepts of advance aerospace vehicles are generally developed via systems analysis studies whereby the various components (e.g., propulsion system, tankage, payload, etc.) are sized, weighed and packaged. Mission requirements, as expected, have a major impact on the shape of the proposed vehicle. Aerodynamic performance characteristics for the vehicle concept are generally approximated using relatively simple engineering codes. Having generated

an initial concept that supposedly satisfies mission requirements, the aerolines (i.e., vehicle outer mold lines (OML)) are provided to the aerothermodynamic community for screening/assessment of aerodynamic performance and aeroheating characteristics (Fig. 5). Aerothermodynamic screening of proposed concepts begins with tests in wind tunnels using models having removable/replaceable components and control surfaces with different deflections. Since the purpose of the first series of tests is to provide a quick look at aerothermodynamic characteristics, model materials and fidelity are selected to expedite the fabrication process and minimize costs. The first aerothermodynamic tests performed are generally at subsonic conditions to assess approach and landing characteristics and hypersonic conditions to assess trim and lift-to-drag characteristics. Forces and moments are measured for various stages of configuration buildup (i.e. fuselage only; fuselage with wing/fin, canards, dorsal, vertical tail, engine module, etc.) over a range of attitude (i.e., angles of attack and sideslip) and control surface deflections. If subsonic aerodynamic characteristics are unacceptable, an iterative process is initiated whereby the experimentalists, often in concert with the systems analysts, modify the aerolines of the concept. The concept is modified and retested until acceptable subsonic aerodynamic characteristics are achieved. These tests are generally performed in relatively small, low cost tunnels.

Hypersonic testing is generally performed in parallel with the subsonic testing thereby necessitating close communication between the two tests. Forces and moments are measured over a range of Reynolds number, attitudes and control surfaces deflections. The effects of configuration buildup are usually examined at hypersonic conditions as are support interference effects. Thermal mappings to identify regions of high heating on the concept are measured using global thermography techniques. If required, and in concert with findings from subsonic testing, aerolines are modified to achieve acceptable hypersonic aerodynamic performance and aeroheating characteristics. If these changes are extensive, an iterative process is initiated via additional tests performed at subsonic conditions to determine if approach and landing characteristics have been compromised. These tests represent the first phase of the aerothermodynamic process, i.e., the screening phase. In this phase, wind tunnel results are generally complemented with predictions from relatively simple engineering codes.

Having refined the aerolines to provide acceptable aerodynamic performance at subsonic and hypersonic conditions (i.e., "bounding the problem"), the aerothermodynamic optimization phase of the process is initiated. In this phase, the subsonic and hypersonic aerodynamic data base is enriched via additional, more detailed testing in these speed regimes. Most importantly, aerodynamic tests at transonic and supersonic conditions are performed. Depending primarily on the shape and center of gravity (cg) location of the vehicle concept, the previously discussed approach (i.e., achieve acceptable aerodynamic performance at subsonic and hypersonic conditions and hope that nothing "bad" happens in-between) may not be successful. Often, "show stoppers" are revealed via testing at transonic conditions. These "show stoppers" require modifications, sometimes extensively, to the aerolines/cg location and retesting at subsonic and hypersonic conditions. Another difference between this and the screening phase is quantitative values of aeroheating are measured in hypersonic wind tunnels and predicted via CFD codes.

Measured aeroheating levels are extrapolated to flight and complemented with flight predictions via CFD to initiate TPS material selection (driven by peak heating), split line definition between different TPS materials (based on worst case heating distribution), and sizing or thickness determination (based on total heat load). CFD is also applied tip-to-tail to predict aerodynamic performance which complements the data base generated via wind tunnel testing.

Testing is continued in this optimization phase until closure is achieved on aerolines that provide: (1) the optimum aerodynamic performance across the subsonic to hypersonic speed regime; (2) acceptable pressure and heating loads during the entire flight trajectory; and (3) volumetric efficiency for effective packaging of the required components.

Once these criteria have been satisfied and, most importantly, the vehicle outer mold lines frozen, benchmark testing (third phase of process) is initiated in well-calibrated, high-flow quality wind tunnels. Depending on the risks associated with the program, testing is performed in the best available subsonic/transonic/supersonic/hypersonic tunnels. Emphasis is placed on the simulation of flight values of Reynolds number throughout the trajectory. This is especially important at transonic-low supersonic conditions for many classes of aerospace vehicles. Some redundancy is desirable via testing in different facilities but at same/similar flow conditions. Precision, quality assurance certified, highly instrumented models are used. Model attitude is accurately set/maintained and monitored during a tunnel run and support system effects are quantified. Steps are taken to provide highly accurate measurements of aerodynamic forces and

moments, detailed surface pressures to extract loads, and detailed heat transfer measurements in this final phase. Emphasis is on accuracy and creditability of the data, as these data will be use for final vehicle design, development of the aerodynamic flight data book and trajectory tailoring to ensure aeroheating levels in flight do not exceed TPS design limits.

Tests are performed during this phase to determine the performance of the reaction control system (RCS), to simulate multibody aerodynamic separation if applicable, to examine aerodynamic performance and aeroheating characteristics during abort maneuvers, etc. to complete the flight data book. Aerodynamicists work closely with system analysts exercising 6-degree-of-freedom codes to provide inputs and margins for guidance, navigation and control. It is during this benchmarking phase of ground-based testing that predictions of aerothermodynamic characteristics via CFD computer codes (i.e., high level, Navier-Stokes solvers) provide significant contributions. Having been calibrated against wind tunnel data, CFD codes are applied at flight conditions which often cannot be accurately simulated in wind tunnels.

The last phase of the aerothermodynamic process is the extraction of aerothermodynamic data from the vehicle in flight and the comparison of wind tunnel simulations and CFD predictions to flight data. This comparison of predictions via experimental and computational aerothermodynamic tools and flight results complete the aerothermodynamic triad (i.e., ground-based facilities, computer codes and flight experiments).

Methodology

From the perspective of the aerospace vehicle designer, the approximate balance for required experimental aerothermodynamic information is 75 to 80 percent aerodynamic performance and 20 to 25 percent aeroheating, with "trace" amounts of aerodynamic loads and fluid dynamic phenomena. For this reason, the proposed aerothermodynamic methodology shown in Fig. 6 is heavily weighed towards aerodynamic performance.

To illustrate the methodology, a hypothetical trajectory is shown in Fig. 6 for an aerospace vehicle descending into the Earth's atmosphere from low Earth orbit (LEO). The vehicle will encounter the classical hypersonic ($M>5$), supersonic ($1.2>M>5$), transonic ($0.8<M<1.2$) and subsonic ($M<0.8$) flow regimes. Within the hypersonic regime, three subcategories are shown in Fig. 6. At the highest altitudes, the "rarefied flow" regime extends from free-molecular behavior to near-continuum behavior as defined by the Knudsen number. At still relatively high altitudes, there is the "hypervelocity continuum" regime characterized by the flow everywhere within the shock layer surrounding the vehicle being laminar, and chemically reacting in portions or throughout the flowfield. At still lower altitudes, the "hypersonic continuum" regime is characterized by vibrational excitation within the shock layer, but no (or very low levels of) dissociation, and by laminar/transitional/turbulent boundary layers and shear layers. The principal source of aerothermodynamic information (with emphasis on aerodynamic performance) as required by the vehicle designer for each of these three subcategories is discussed next.

Aerothermodynamic information for aerospace vehicle concepts in the "rarefied flow" regime is provided by the Direct Simulation Monte Carlo (DSMC) method or by simple bridging relations between the free molecular and continuum regimes. DSMC capabilities have advanced significantly over the last decade (e.g., Kumar et al., 1997) and DSMC presently enjoys a high level of creditability and exposure due to numerous recent successes in predicting aerothermodynamic performance for a wide spectrum of configurations in flight about Earth or other planets, specifically Mars. There are no active rarefied flow facilities in this country capable of proving creditable aerodynamic performance characteristics for proposed aerospace vehicles. For this regime, vehicle designers rely on computational techniques (i.e., DSMC) exclusively.

There are several sources for aerothermodynamic information in the "hypervelocity continuum" regime. One is high-enthalpy ground-based facilities such as a piston-driven/combustion driven shock tunnel, expansion tube, arc-driven "Hotshot" tunnel and/or ballistic, free-flight range. The advantages and disadvantages of these "specialized" facilities are well recognized. A major disadvantage for the measurement of aerodynamic forces and moments on relevant aerospace vehicle shapes is very short test times. Typically, but not exclusively, test time for these facilities range from a few milliseconds to microseconds at high enthalpy levels. Although short run times may prove advantageous for aeroheating measurements (e.g., minimizes conduction effects in transient techniques), this is not the case for force and moment measurements.

A second source is simulation of certain aspects of a real-gas via testing with a heavy gas at relatively low enthalpy levels. The use of a heavy gas in a conventional-type, blowdown-to-vacuum hypersonic

wind tunnel provides all the advantages of a conventional, as opposed to impulse, facility while simulating the high normal shock density ratio and/or low values of specific heats ratio within the shock layer experienced in flight due to real gas effects. The only such facility in the country is the NASA LaRC 20-Inch Mach 6 CF_4 Tunnel (Micol, 1998) which simulates (approximates) Mach 20 flight in the Earth's atmosphere. Because of the relatively low values of unit Reynolds number for this facility (less than one million per foot for supply pressures up to 2000 psia) and small test articles, the boundary layer on the windward surface of models tested in this facility is laminar.

CFD predictions "duplicate" vehicle scale, velocity, attitude and atmospheric (free stream) conditions, thereby avoiding the uncertainties associated with extrapolation of ground-based data to flight. For this, and other reasons discussed previously, CFD calibrated against ground-based and flight data is the preferred source of aerothermodynamic information in the "hypervelocity continuum" regime.

Wind tunnels provide the majority of aerothermodynamic information in the "hypersonic continuum" regime. Flight simulation via ground-based facilities is tried and proven for this regime with over 40 years of experience. For a given interval of time, wind tunnels can provide several orders of magnitude more aerothermodynamic information than CFD codes. This discrepancy in productivity is made larger when aerodynamic performance and aeroheating characteristics are addressed for a range of angle of sideslip (i.e., CFD solutions involve the entire vehicle, not just half of it). Both ground-based testing techniques and CFD codes are challenged to accurately predict flight boundary layer/shear layer transition phenomena. The advantage in providing meaningful, relevant transition information resides with ground-based facilities even in the absence of hypersonic, low-disturbance/quiet testing capability. The "hypersonic continuum" regime affords the vehicle designer accurate/creditable aerothermodynamic information via both experimental and computational capabilities, unlike the "rarefied flow" and "hypervelocity continuum" regimes where only computational capabilities are available/viable.

Effect of Vehicle Shape

As the approach to generating aerodynamic and aeroheating information is developed, an important factor to be considered in ground-based testing is the basic shape of the aerospace vehicle concept(s), since the shape dictates which fluid dynamic phenomena are most important and which simulation parameters will dominate. From a hypersonic perspective, the test approach for a very slender configuration may be quite different from that for a blunt configuration. For example, a hypersonic airbreathing (i.e., scramjet) vehicle will most likely be slender, have sharp leading edges, and fly at low incidence and relatively high dynamic pressures. For a slender shape and low incidence, the flowfield about the vehicle will be principally supersonic/hypersonic. The effects of compressibility (Mach number), viscosity (Reynolds number), gas properties (e.g., ratio of specific heats), and thermal driver potential (ratio of wall-to-adiabatic wall temperatures) are all expected to be important; that is, to have a first order influence on aerodynamic/aeroheating characteristics. Of particular importance is the state of the boundary layer (i.e., whether the boundary layer is laminar, transitional, and/or turbulent) which may influence the level of control effectiveness and definitely influence the heating. Because of the small shock inclinations associated with sharp leading edged slender bodies at small incidence, finite-rate chemistry effects on aerodynamics (e.g., variation in center of pressure) and aeroheating (e.g., thermochemical nonequilibrium heating including surface catalytic effects) are usually small except in local regions of flow stagnation, shock/shock interactions and shock impingement. On the other end of the shape spectrum, the flow over the forebody of a very blunt configuration is principally subsonic becoming supersonic as it expands around the corners. For very blunt shapes, the most important hypersonic simulation parameter for aerodynamics is the density ratio across the normal portion of the bow shock or corresponding value of ratio of specific heats within the shock layer for continuum flow. In the continuum flow regime, Mach number effects for Mach numbers in excess of five are generally negligible (Mach number independence principal) as are viscous effects on the forebody where the boundary layer is quite thin. The detachment distance of the bow shock from the forebody surface and the location of the sonic line separating the subsonic flow and supersonic flow regions are a strong function of the density ratio/post-shock gamma and the influence of density ratio/gamma on these parameters may have a first order influence on the aerothermodynamic characteristics (e.g., Gnoffo et al., 1998). These effects for blunt and slender configurations are summarized, subjectively, in Fig. 7

(e.g., some vehicles may exhibit both slender body or blunt body hypersonic characteristics during entry, depending on the angle of attack).

Infrastructure

The "aerothermodynamic chain" is shown in Fig. 8. A container of aerodynamic performance and aeroheating information for the assessment/optimization/benchmarking of an aerospace vehicle concept is being retrieved from the bottom of a well using a chain. The weight (i.e., magnitude) of this information is established by the weakest link in the chain, as the chain can be no stronger than its weakest link. A broken link corresponds to no retrieval of information. The links of this chain are personnel, facilities, models or test articles, instrumentation, testing techniques and CFD. These links are universal to all aerothermodynamic activities in this country and abroad, and are reviewed in the following sections from a Langley perspective.

Personnel

The most important resource for the development of aerothermodynamic information for advanced aerospace vehicles is, naturally, personnel (Fig. 9(a)). Knowledge of aerodynamics across the subsonic-to-hypersonic speed regime, aeroheating, and complex fluid dynamic and high temperature flow phenomena is required. Whenever possible, the same engineers/scientists perform subsonic, transonic, supersonic, and hypersonic aerodynamic testing, as opposed to different personnel testing (i.e., specializing) in each speed regime. This approach provides continuity which is vitally important, particularly since iterations are generally required to optimize aerodynamic performance across the speed regime. Working closely with the aerodynamicists are engineers/scientists performing aeroheating studies for a range of hypersonic conditions. Regions of high heating are identified and, if deemed unacceptable, modifications are made to the vehicle aerolines, attitude and/or trajectory to reduce heating loads without jeopardizing the aerodynamic performance. Teams of experimentalists and computationalists assessing/optimizing concepts are, by design, generally small, consisting of 4 to 6 members. (Smaller teams have been observed to function more efficiently and effectively than larger teams.) Experience is a critical ingredient in the makeup of the teams. Senior engineers/scientists having experience with the Mercury/Gemini/Apollo, shuttle orbiter, Viking, etc. programs are teamed with highly educated and motivated junior engineers/scientists. This mix of junior and senior personnel encourages the transfer of corporate knowledge and unites the savvy of senior personnel with the "can-do-attitude" of junior personnel.

Facilities

Presently, there are 8 active, conventional-type (as opposed to impulse-type) hypersonic wind tunnels in the country used for aerothermodynamic (as defined herein) testing (Fig. 9(b)). The Langley Aerothermodynamic Facilities Complex (AFC; Micol, 1998) consists of 5, relatively small, blowdown-to-vacuum hypersonic wind tunnels (Fig. 9(c)) designed and constructed in the late 1950s to early 1960s. These 5 facilities represent NASA's entire aerothermo-dynamic ground-based testing capability. Three different test gases are used, namely dry air, helium and tetrafluoromethane (CF_4). These facilities complement one another to provide a range of free stream Mach number from 6 to 20, range of freestream unit Reynolds number, and most importantly for the simulation of real-gas effects, a range of normal shock density ratio or post shock ratio of specific heats (Figs. 9(d) and 9(e)). AFC members were designed primarily to perform basic hypersonic fluid dynamic studies and aerodynamic/aeroheating studies associated with screening and optimization of proposed hypersonic vehicles. Facilities using air as the test gas have typical run times of 2 up to 15 minutes at Mach 6 and 2 minutes at Mach 10. These test times, in conjunction with nominal run frequencies of 8 to 10 per shift, provide respectable quantities of aerothermodynamic information.

The country's premier aerothermodynamic testing capability resides in the Air Force's Arnold Engineering Development Center (AEDC) Tunnels B and C. These large (50-inch diameter nozzle exit/test section) air facilities provide free stream Mach numbers of 6, 8 (Tunnel B) and 10 (Tunnel C). Although the freestream unit Reynolds number range for these facilities is very similar to the air facilities of the Langley AFC, Reynolds number based on model length or diameter is typically 3 times greater due to the larger model size that may be accommodated. Tunnels B and C are continuous flow facilities, having run times of several hours, and thus are highly productive. The large size, high flow quality (uniformity) and continuous flow mode of operation makes Tunnels B and C ideally suited for aerothermodynamic benchmarking.

The Naval Surface Warfare Center (NSWC) Hypervelocity Wind Tunnel Number 9 (e.g., Marren and Lofferty, 1998), located in White Oak, Maryland

recently was transferred to the Air Force. This large (60-inch diameter nozzle exit) blowdown-to-vacuum facility operates at Mach numbers of 10, 14, and 16.5 with the large nozzle and Mach 7 full-flight duplication and Mach 8 with smaller nozzles. The test gas is nitrogen. Significantly higher values of Reynolds number are available from Tunnel 9 than from Langley AFC air facilities and AEDC Tunnels B and C. However, run time is about one second for normal operation and decreases to less than a second at the higher values of Reynolds number; thus Tunnel 9 is much closer to the impulse facilities portion of the time spectrum than the conventional-type blowdown facility portion.

Langley Aerothermodynamic Facilities Complex (AFC)

The five members of the AFC (Fig. 9(c)) are:
20-Inch Mach 6 Air Tunnel
15-Inch Mach 6 High Temperature Tunnel
31-Inch Mach 10 Air Tunnel (formerly, Langley Continuous Flow Hypersonic Tunnel)
22-Inch Mach 15/20 Helium Tunnel
20-Inch Mach 6 CF4 Tunnel
(Mach 20 Flight Simulation)

Facility designation uses the following format. First, the geometric nozzle exit diameter or height is given followed by the nominal Mach number and test gas. A pertinent feature of the facility may also be specified. Descriptions and capabilities are presented by Micol, 1998.

The 20-Inch Mach 6 Air, 31-Inch Mach 10 Air and 22-Inch Mach 15/20 Helium Tunnels were heavily utilized in the 1960s and early-to-mid 1970s, but relatively neglected thereafter until the late 1980s. Members of the AFC were upgraded in the late 1980s and early 1990s via a FY1989 Major Construction of Facilities (CoF) Project, several minor CoF projects, the Agency Aeronautical Wind Tunnel Revitalization Program and various other funding sources (Miller, 1990, 1992). The emphases of these upgrades were to improve flow quality, capability, productivity and reliability. Most facilities received new nozzles whereby the nozzle aerolines were predicted with CFD codes, a capability not available for the original design in the 1950s. Flow uniformity was significantly improved with the new nozzles. All facilities were equipped with in-line filters between the heater and settling chamber to reduce particulate in the flow. The same signal conditioning and data acquisition systems were installed in all 5 facilities to provide commonality/continuity. As one example of enhanced productivity, the number of runs per shift for the 31-Inch Mach 10 Air Tunnel was doubled via the addition of a steam ejector to the vacuum system.

Testing with gases lighter and heavier than air in conventional-type wind tunnels is unique to Langley in this country. The primary advantage to using helium is that this gas may be expanded from a high pressure and ambient temperature to Mach numbers around 25 without flow liquefaction. Also, for unheated helium flow, free stream values of unit Reynolds number are relatively high. Models made rapidly from resin, plastic, wood and other easily formed materials may thus be tested at high hypersonic Mach numbers without the concerns associated with high-temperature tunnels such as model degradation. The 22-Inch Mach 15/20 Helium Tunnel served as a workhorse for aerodynamic screening of RLV/X-33 concepts in Phase I and provides, for most configurations and attitudes, good agreement with aerodynamic coefficients measured in hypersonic air tunnels (Fig. 9(g)).

Testing with CF_4, which is 3 times heavier than air, provides higher values of density ratio than hypersonic air tunnels by approximately a factor of 2 and lower values of the ratio of specific heats (gamma) within the shock layer about the model. The values of density ratio and post shock gamma provided by the 20-Inch Mach 6 CF_4 Tunnel are representative of values encountered by moderately to very blunt vehicles in hypervelocity flight for which the gas within the shock layer of the vehicle is dissociated. The ability of this facility to simulate real-gas conditions for the shuttle orbiter in flight (e.g., Brauckmann et.al., 1995) is shown in Fig. 9(f). Values of gamma over the windward portion of the orbiter flowfield at Mach 20 in flight were simulated via CF_4 and this simulation revealed that the pitch-up anomaly experienced on the maiden flight (STS-1) of the orbiter was due to real-gas effects; i.e., real-gas effects resulted in a greater expansion of the flow over the aft portion of the orbiter. This more rapid expansion decreased the surface pressure, as compared to levels inferred from tests in "ideal-air" wind tunnels and used as inputs to the original flight aerodynamic data book, and thereby increased the pitching moment (more nose up). Ground-based simulation of real-gas effects on aerodynamic performance is important for all aerospace vehicles flying at Mach numbers above 12, or so, and for planetary probes. Testing in CF_4 provides a means to "certify" real-gas effects and complements CFD predictions as discussed previously.

Test Articles/Models

Often overlooked, but a major factor in ground-based testing, is the design, fabrication, and

instrumentation of models (Fig. 9(h)). Models must be fabricated accurately to avoid flawed aerothermodynamic data due to testing erroneous aerolines, incorrect attitude due to poor support alignment, etc.; and fabrication must be completed on schedule to avoid wind tunnel schedule perturbations that may possibly jeopardize the study. An on-going goal is to develop model design/fabrication procedures/techniques to provide models faster with no sacrifice in, and hopefully enhanced, accuracy. Aerolines developed during system analysis studies (e.g., with computer aided design (CAD)) or developed for incorporation into CFD computer codes are electronically transferred to stereolithography (SLA) and/or numerical cutting (NC) machines by compatible software. SLA/NC machines are then used to construct models, with close interaction between precision model builders and researchers (when fabrication performed in-house). This approach was used successfully in the NASP program, whereby aerolines describing the consortium developed vehicle (referred to as configuration 201) were transferred to Langley, immediately loaded onto numerical cutting machines, and models fabricated in a fast-paced manner without formal model design. High fidelity, configuration buildup, stainless steel models were fabricated for force and moment and for pressure testing and ceramic models made for thermal mapping studies. These models were constructed, tested and the data reduced, analyzed, and disseminated within a matter of months.

The fast pace of model fabrication and testing demonstrated for the NASP program can, in reality, be maintained in-house only for relatively short periods of time and requires high Center priority. However, the methods and procedures developed are applicable to normal priority experimental programs with significant savings in time and cost of producing quality test results.

Casting of metallic models for aerodynamic screening studies is used to reduce fabrication time and cost. With the development of stereolithography processes, high quality resin patterns may be made in a short time once the model surface geometry is known (i.e., CAD description available). These patterns are used to build molds which in turn are used to quickly cast the metallic model. If high precision is required, the model may be cast slightly oversized, and then final machining performed. This approach minimizes wasted material and time compared to traditional machining methods where blocks of a metal are machined down to final shape.

Aeroheating models are made of a ceramic via a patented process and using a mold generally made from SLA patterns or metallic force and moment models. These cermanic models, after being fired and coated, are tested to provide global qualitative thermal mappings and quantitative heating distributions via phosphor thermography (discussed subsequently). Because models can be made quickly and inexpensively, aeroheating tests for a given configuration may now be performed prior to force and moment tests. Thus, the designer of a proposed hypersonic flight vehicle is provided aerodynamic and aeroheating data at about the same time for trade studies, unlike previously when aeroheating data significantly lagged aerodynamic data.

Measurement Techniques

Measurement techniques routinely employed in AFC wind tunnels at Langley (Fig. 9(i)) are discussed briefly in this section. (Details are provided by Micol, 1998.) The most commonly performed studies involve the measurement of aerodynamic forces and moments on models followed by aeroheating studies. Measurements of model surface-pressure distributions may also be performed. In most studies, flow-visualization techniques are used to provide complementary information on shock locations and boundary-layer/shear-layer characteristics. Flowfield surveys within the shock layer/boundary layer of models may be performed with miniature probes in all facilities and nonintrusive flowfield measurements may be performed at Mach 6 in air using the Rayleigh scattering and/or planar laser induced fluorescence (PLIF) techniques.

Forces and Moments. An inventory of approximately 50 internal strain-gage balances that cover a wide range of maximum design loads and sensitivities for blunt, high-drag models to slender, high-lift models is maintained at Langley. Most balances are six component (normal, axial, and side forces and pitch, yaw, and roll moments) and are water cooled. These balances, generally having an outside diameter of 0.56 inch and length of about 4 inches, have uncertainties of less than 0.5 percent full scale.

Pressure. Pressure distributions on the relatively small-scale models tested in the AFC are measured with electronically scanned pressure (ESP) silicon sensors; limited measurements may be made with high-volume, multirange, variable-capacitance diaphragm-type transducers. ESP modules typically contain 16, 32, or 48 sensors, are relatively small, and combine internal multiplexing and amplification to provide scanning at high data rates. In some cases, the module may be mounted inside the model, in the support strut, or at its base to reduce the response time by minimizing tubing

length between model pressure orifice and the sensor. An integral, pneumatically controlled mechanism allows the sensors to be calibrated on-line. ESP sensors have been used in Langley hypersonic wind tunnels since the early 1980s to accurately measure pressure levels ranging from 50 to 0.05 psi.

Experimental work in a laboratory setting has shown promise for obtaining simultaneous luminescence barography and thermography results in hypersonic air wind tunnels (Buck, 1994). The original work used a two-color imaging system and adsorbed dye luminescence on silica ceramic test models. In trial applications, it was found that an adsorbed perylene dye on slip-cast silica was pressure (oxygen) sensitive and reusable to relatively high temperatures (~150C). Absorbed dye luminescence was excited by blue light (460 nm) or long-wave ultraviolet (365 nm). Visible emission was found to be green-red with color depending on absorbed film thickness and temperature. Surface pressures and temperatures were determined from emission brightness and green-to-red color-ratio measurements.

Qualitative Heat Transfer. Thermal-mapping studies gained increased use in the 1980s and early 1990s because they provide a rapid, relatively inexpensive determination of qualitative heating characteristics on models of various shapes and complexity. Four techniques have been used over the last decade: phase-change paint, thermographic phosphors, liquid crystals, and infrared emission. Developments in phosphor thermography have revolutionized aeroheating studies, and phase change paints and liquid crystals are no longer used at Langley. Although high quality infrared (IR) measurements can be made using charge coupled device (CCD) cameras without having to apply coatings to the models, this technique is seldom used due to the outstanding progress made with the phosphor technique.

Quantitative Heat Transfer. The relative intensity, two-color phosphor thermography technique developed at Langley (Buck, 1991) presently provides essentially all quantitative heat transfer measurements in the Aerothermodynamic Facilities Complex. The phosphor material is applied to the ceramic model and is illuminated by ultraviolet light that excites electrons; during their subsequent relaxation to lower energy levels, these electrons emit visible radiation that is temperature dependent. The amount of radiation measured with a 3-color CCD camera is used to determine surface temperatures everywhere viewed by the camera. This technique does not require models to be recoated after a run, is independent of the optical path, and models are relatively inexpensive and may be constructed rapidly. Several thermography systems have been assembled and may be moved from facility to facility without requiring lengthy setup and calibration. Data acquisition and reduction techniques (Merski, 1998) provide global quantitative aeroheating measurements to the researcher shortly after a test.

The phosphor thermography technique essentially replaced the thin-film resistance thermometer technique as the means to measure quantitative heat transfer rates in the Langley AFC in the mid 1990s. The thin-film technique, which was originally developed for impulse-type facilities with run times of a few milliseconds or less, was refined to a high technology level for use in the conventional-type hypersonic wind tunnels of the AFC. Thin-film models, typically fabricated from a machinable glass-ceramic, were extremely expensive to build, requiring considerable time (typically 9 to 12 months). In turn, the thin-film technique previously replaced the thin-skin transient calorimeter technique which served as the standard at Langley from the 1960s to mid 1980s. Signal conditioning and data acquisition/reduction are still available in AFC wind tunnels for the thin-film and calorimeter techniques.

Flow Visualization. Included in the category of flow-visualization techniques are shadowgraph/schlieren systems, interferometry systems, vapor screens, electron-beam flowfield visualization, and surface oil flow. For the latter technique, smooth, dark-color models are sprayed with a mixture of oils of various viscosities mixed with white artist pigment and injected rapidly into the flow. Movement of the oil is photographed while the model is in the flow and high quality single frame photographs taken afterwards, thereby qualitatively revealing surface streamline patterns—that is, the direction of the flow adjacent to the surface, including lines of separation and reattachment.

The schlieren method combines a relatively simple optical arrangement with a high degree of resolution. Parallel light is passed through the test section, and an image of the light source is focused in the plane of a knife edge. An example of schlieren photography being used to complement heat transfer measurements in a study of complex shock-shock interaction is provided by Berry and Nowak, 1997.

Computational Methods

Computational aerothermodynamic methods at Langley (Fig. 10) consist basically of engineering techniques, inviscid (Euler) solvers, 3-D boundary layer solvers that require inviscid solutions, viscous shock

layer (VSL) codes, full Navier-Stokes (NS) solvers and Direct Simulation Monte Carlo (DSMC) codes. For a comprehensive review of aerothermodynamic computational capabilities at Langley, the reader is referred to Kumar et al., 1997.

Engineering codes are used for initial screening of aerospace vehicle concepts, trade studies and data base construction. The primary engineering code used for aerothermodynamic studies at Langley is Langley Approximate 3-D Convective Heating (LATCH). This code was developed to provide rapid computations of approximate heating on complex, 3-D vehicles. LATCH is based on an axisymmetric analog for 3-D boundary layers and heating is calculated along inviscid surface streamlines using an approximate, integral boundary-layer method. Edge conditions are obtained from 3-D inviscid flowfield solutions; the code includes perfect gas, CF_4 and equilibrium air chemistry; and requires (typically) approximately 30 minutes on a desktop workstation for a solution over a complete vehicle (tip-to-tail, windward, and leeward surfaces). The LATCH code is complemented by the THIN shock layer—Boundary Layer (THINBL) code developed to rapidly compute windward heating on aerospace vehicles at incidence. THINBL requires approximately 5 minutes on a desktop workstation for solution over the windward surface. The Catalytic Heating Rate Analysis (CATHRAN) code was developed to rapidly compute changes in surface heating due to changes in surface catalysis and emissivity, and is based on integral boundary-layer analysis. This method requires a single baseline CFD solution with a gas kinetic model.

Inviscid techniques are used for aerodynamic screening/optimization, the prediction of aerodynamic pressure loads, and providing inputs to engineering codes. The primary inviscid solver for aerothermodynamic studies is FELISA, which was developed to compute flowfields over complex geometries. FELISA uses an unstructured volume mesh generation, solves the 3-D Euler equations, has grid adaptation capability, and is setup for perfect gas, CF_4 and equilibrium air. FELISA has been applied to subsonic, transonic, supersonic and hypersonic conditions. Also used to provide inviscid solutions is the Data-Parallel Lower-Upper Relaxation (DPLUR) method. This method was developed for complex 3-D vehicles and designed to run on massively parallel computers. This code is based on time-dependent solution of Euler equations and includes perfect gas and equilibrium and nonequilibrium chemistry.

The computational aerothermodynamic "standard" at Langley to which all other codes are compared is the Langley Aerothermodynamic Upwind Relaxation Algorithm (LAURA) code (Gnoffo et al., 1997). This is a viscous solver that provides benchmark solutions including separated flows/wakes. Predictions from LAURA have been compared extensively to aerodynamic and aeroheating measurements for the Shuttle Orbiter in flight, to Mars Pathfinder flight data (Gnoffo et al., 1998), and benchmark ground-based measurements over a wide range of supersonic-hypersonic flow conditions and spectrum of model geometry/attitude. LAURA was developed primarily for high-enthalpy, real-gas flows about blunt bodies (e.g., AFE vehicle) and solves the Euler, thin-layer Navier-Stokes and Navier-Stokes equations. This code includes chemical and thermal (2-temperatures model) nonequilibrium and equilibrium air models, finite-catalytic wall model, and laminar or turbulent flow (algebraic and 2-equation models).

The LAURA code continues to be enhanced and refined by the code developer P.A. Gnoffo of Langley. Lessons learned from a wide spectrum of applications for complex aerospace vehicles and for planetary probes have been incorporated into the code. Having the developer of the code readily available to computationalists at Langley who are applying the code has proven to be extremely valuable. It is for these and other reasons that the methodology discussed previously places the responsibility of obtaining aerothermodynamic (i.e., aerodynamic and aeroheating) information in the hypervelocity, continuum laminar flow regime on the computationalists as opposed to experimentalists using high-enthalpy ground-based testing.

Not to be overlooked is the General Aerodynamic Simulation Program (GASP) code, which traces its origin to the Langley developed CFL3D code. Like LAURA, GASP provides benchmark aerothermodynamic information and has benefited from a wide range of applications for external and internal flows. GASP is commercially available (AeroSoft, Inc.) and is exercised by computational aerothermodynamicists at Langley but to a much lesser extent than LAURA.

The Viscous Shock Layer (VSL) code was developed to rapidly compute viscous flowfield solutions over 2-D/axisymmetric bodies using detailed thermodynamic and chemistry models applicable to Earth and planetary entry. This code includes body and shock slip for application at rarefied conditions and surface ablation and radiation. The VSL code has been particularly useful for planetary studies involving simple shapes at zero angle of attack but having extremely complex flowfields from the viewpoint of high temperature chemical/physical processes. Runs

with the VSL code require minutes on a high end workstation.

The DSMC method is based on the tracking of large numbers (millions) of molecules (Fig. 10(c)) as they move through space and collide with one another and the surface of the vehicle. DSMC codes are used to simulate the highly rarefied condition associated with a vehicle in orbit or entering through the upper portion of the atmosphere where conventional continuum methods (e.g., Navier-Stokes equations) are not valid. DSMC codes model complex geometries and physical phenomena and have been calibrated/validated via flight data (e.g., high-temperature, chemically reacting flow about shuttle orbiter, Mars Viking probes, and Mars Pathfinder probe).

Impact of Programmatic Changes—1993 to Present

A few of the programmatic changes that have occurred within the past 5 years and impacted, positively or negatively, Langley's aerothermodynamic capabilities are discussed in this section. This section is not intended to be comprehensive, delineating most of the changes, but to simply provide a few examples that may help explain the present status of aerothermodynamic capabilities and activities at Langley.

Center Reorganization

In early 1994, a Center-wide reorganization was implemented at the Langley Research Center. Research engineers and scientists were placed in the Research and Technology Group (RTG) representing 25 to 30 percent of the civil servants at Langley. Independent offices to manage/advocate programs and funding and to provide customer interfaces were established. Unlike the former organization, the new organization separated research personnel from the management of funds, one of the rationale being that researchers would be free to pursue technology development without the burden of advocating for and managing funds.

In the new organization, computational aerothermodynamicists from the former Aerothermodynamics Branch, experimental aerothermodynamicists from the Experimental Hypersonics Branch (EHB), and experimentalists/computationalists from the Aerothermal Loads Branch were combined into a new Aerothermodynamics Branch (AB). The new AB consisted of 43 engineers/scientists and was very nearly balanced between the number of experimentalists and number of computationalists. Former AB members brought a rich heritage of hypervelocity flow phenomena associated with Earth/planetary entries and former EHB members a heritage of space transportation systems development. Also part of the reorganization was the transfer of the 20-Inch Mach 6 Air Tunnel into the new AB, bringing the total number of hypersonic wind tunnels managed by the branch to 6. (The 5 members of the AFC discussed previously plus the 20-Inch Mach 17 Nitrogen Tunnel.)

The former Space Systems Division, to which the former AB and EHB belonged, was split apart with systems analysis personnel belonging to the Vehicle Analysis Branch (VAB) placed in the Space and Atmospheric Sciences Program Group as opposed to the RTG. Thus, the integration/synergism of experimental and computational aerothermodynamics and systems analysis which had evolved over an extended period of time was, on paper, dissolved. (Fortunately, excellent working relationships continued.)

A period of adjustment was required upon implementation of the new organization, as expected. AB members and associated equipment were required to be moved to a different location (building). Computationalists and experimentalists who had worked in relatively "homogeneous" environments (i.e., computationalists worked with and mentored computationalists; experimentalists worked with and mentored experimentalists) and formed respective cultures were suddenly brought together into a single organization. Each discipline was somewhat uncertain of the other and teaming between computationalists and experimentalists was initially slow to evolve.

Shortly following the reorganization, the new AB incurred a substantial reduction in funding. When the former AB, ALB and EHB were combined, the respective funding for these branches was also combined. This allowed the new branch to continue aerothermodynamic research at the pre-organization level. However, several months after the new AB was formed, it was learned that funding in the next fiscal year would be reduced by over one third. This necessitated a significant reduction in grants to universities and in contractors supporting the branch. Roughly two thirds of AB contractors, all computationalists with years of experience, were released. The university grants eliminated/reduced also supported AB computational activities. Thus, the branch experienced a significant loss in the capability to develop advanced computational techniques including Direct Simulation Monte Carlo (DSMC), and in hypervelocity aerothermodynamic expertise.

Agency Zero Base Review (ZBR)

The ZBR was initiated in the mid 1990s principally to restructure NASA program management for enhanced efficiency/effectiveness. One objective of

ZBR was to reduce/eliminate redundancy between NASA Centers. As an outcome of the ZBR, Langley was established as the lead Center for Aerothermodynamics. All of the Agency's aerothermodynamic ground-based testing capability now resides at Langley, as the Ames Research Center (ARC) shutdown the 3.5 Foot Hypersonic Wind Tunnel and other facilities. As part of the downsizing of facilities associated with ZBR, the Langley 20-Inch Mach 17 Nitrogen Tunnel was deemed a non-core facility and was mothballed.

The ZBR required considerable self examination and the term "sandbox" was reserved for studies deemed irrelevant and targeted for deletion. Such an exercise is particularly challenging in a basic/fundamental research environment. The accountability for allocation of funds to research organizations was increased substantially and stringent guidelines established. Again, research organizations were required to make adjustments and strictly adhere to funding schedules, often making the chase of exciting, unscheduled technical spin-off activities associated with research difficult, if not impossible, to perform.

Focused, Industry-Led Programs

The initiation of the RLV/X-33 Phase I and X-34 programs (Fig. 11) in April, 1995, followed a year later by X-33 Phase II, X-34 (second phase), Hyper-X, X-38, Missions from Planet Earth (i.e., planetary exploration and Earth sample return missions) and other studies requiring aerothermodynamic information (Fig. 12), ushered in a time of great excitement and expectations; these studies, collectively, also challenged Langley's aerothermodynamic capabilities/resources. As to be expected, the AB work environment changed significantly due to the rather abrupt increase in generation of aerothermodynamic information for external customers. Demands on AB experimentalists were especially large since the Langley AFC represents the Agency's sole source of experimental, ground-based aerothermodynamic data. Downsizing of personnel also occurred in this period, corresponding to AB members and facility technicians who retired, transferred, etc. generally not being replaced.

Another change to the work environment was due to programs being industry led as opposed to NASA led. Because some customers did not have working knowledge/experience with Langley's aerothermodynamic capabilities, schedules/milestones established by customers failed to take advantage of the full-range of capabilities or, at the other extreme, were unrealistic in expectations. To satisfy customer milestones, AFC wind tunnels were operated in a "production-like" manner, running extended/double shifts and weekends. Demands were particularly high on the 20-Inch Mach 6 Air and 31-Inch Mach 10 Air Tunnels.

Most, but not all, customers "teamed" with the AB, providing a sense of program ownership to AB members (team pride) and actively seeking their technical inputs/recommendations. Those not teaming were primarily interested in AB members "turning the crank" for AFC wind tunnels to fill a matrix. Regardless of the teaming aspect, focused programs required adjustments by AB members whose expectations when joining the branch prior to the mid 1990s were to advance aerothermodynamic experimental/computational capabilities via a blend of basic/fundamental and applied research; i.e., to develop and apply. The departure from this blend to a primarily focused research program represents a significant change. One concern is that a 100 percent focused program is equivalent to eating the seed corn. Technology breakthroughs such as the Langley developed two-color phosphor thermography technique, which has revolutionized aeroheating measurements in ground-based facilities, may become difficult to achieve. Another perception, particularly for experimentalists closely associated with the AFC wind tunnels, is that total dedication to focused programs is analogous to a job versus a career. These and other concerns may represent, collectively, the major influence/impact to Langley's aerothermodynamic capabilities since 1993.

Construction of Facilities (CoF) Funding

Between FY 1994 and FY 1995, total NASA CoF funding was reduced by a factor of approximately two and one-half (Fig. 13) and Major CoF funding for NASA Code R Centers (i.e., Ames, Dryden, Langley and Lewis) vanished. Although funding for Minor CoF projects has remained relatively constant since 1993, the nominal level of $70 M is for all NASA Centers; thus competition for these funds is quite keen. With the exceptionally low probability for successful advocacy of a Major CoF project, upgrades/enhancements to AFC facilities reside with Minor CoF projects which presently are restricted to under $1.5 M. The AB has an aggressive Minor CoF plan, but the probability for successful advocacy appears to be decreasing with time.

Infrastructure Changes—1993 to Present

This section is intended to complement a previous section entitled "Infrastructure" by presenting recent changes to links in the "Aerothermodynamic Chain" or components of the infrastructure. The change in the type of testing performed in the AFC is illustrated in

Fig. 14(a). This figure shows the shift from a blended basic/fundamental and applied research program. Casualties due to this transition include diagnostics development, facility enhancements/calibrations, studies of complex fluid dynamic phenomena, relevant to aerospace vehicles, such as boundary layer/shock layer transition, shock/shock interaction, RCS flowfield-surface interactions, etc., and CFD code calibration studies. Decreases in engineers/scientists, facility technicians and contractors since 1993 are shown in Fig. 14(b). The total decrease for engineers/scientists is approximately 30 percent, and approximately 35 percent for facility technicians. The decrease for experimentalists is somewhat larger than for computationalists. In 1993, there were 7 active hypersonic wind tunnels at Langley used for aerothermodynamic studies and two addiitonal facilities were undergoing modifications/upgrades. Three of these nine facilities have been mothballed/terminated, leaving the 5 wind tunnels operated by AB (i.e., AFC) and the Mach 8 Quiet Tunnel presently in shakedown.

Several changes have occurred in the area of model fabrication (Fig. 14(c)). SLA resin models are now used extensively for testing in unheated flows and for making molds to cast metallic and ceramic models. Ceramic models (Fig. 15) used for aeroheating studies are now being used for force and moment testing and this trend is expected to increase significantly in the future. The major change in model fabrication, and corresponding impact, is a result of the outsourcing of most precision metallic models beginning in the fall of 1996. It is ironic that this outsourcing occured just after completion of the RLV/X-33 Phase I and initial X-34 programs. All models, metallic and ceramic, tested in these fast-paced programs involving four different concepts were constructed in-house at Langley, and all models were delivered on/ahead of schedule and within cost. With increased outsourcing of metallic model fabrication came the downsizing of the corresponding in-house capability just months after setting new standards for metal model fabrication.

The major change in the "instrumentation link" of the chain is due to advancements in relative intensity, two-color phosphor thermography for global quantitative heat transfer measurements (Fig. 14(d)). Data acquisition is performed with patented state-of-the-art video acquisition systems and color solid-state video cameras (Fig. 16). The systems digitize phosphor fluorescence intensity images to resolution of 512×481 pixels and these images are transferred to workstations for processing. A workstation-based image processing package developed at Langley and called IHEAT (Merski, 1998) consists of six programs to handle systems calibrations along with data reduction, editing, and viewing. Using IHEAT, which is written in a user-friendly windowing format, data can be reduced to heat transfer images immediately after a run. An automated routine also provides plots of heating along the centerline and axial cuts. Global quantitative data are obtained using a unique nonlinear relative-intensity method. Because of the relatively small models for the AFC, only qualitative data is available on small leading edges as the 1-D conduction assumptions within the code are not valid and temperatures often exceed the limits of the current system (70° to 340°F). This technique has an uncertainty of approximately 10 to 12 percent in heat transfer rate, as compared to 5 to 8 percent for conventional thin-film resistance thermometry methods, and has exhibited excellent run-to-run repeatability.

The phosphor thermography process is illustrated in Fig. 17. Fabrication of ceramic models using a Langley developed, patented ceramic investment casting technique usually requires about 3 weeks for 5 to 10 models having different control surface deflections. The models are tested in the AFC, data acquired, reduced via IHEAT, analyzed and disseminated within about 5 weeks after start of model construction. Advantages of phosphor thermography compared to thin-film resistance gages are shown in Fig. 18. Phosphor thermography is "better" because of the large data density provided by video as compared to discrete gages; is "faster" because of the rapid model fabrication; and thus is "cheaper." (The contributions of Langley's phosphor thermography technique to Agency aerospace programs was recently (May, 1998) recognized via an award at the Thirteenth Annual NASA Continual Improvement and Reinvention Conference.)

It should be noted that significant advances in thin-film resistance thermometry techniques were made in the mid 1990s to construct thin-film models faster and with more instrumentation (e.g. Berry and Nowak, 1997). Thin-film gages were etched onto a flat polyimide film and the film wrapped and bonded to the cylindrical test article. With this technique, resolution was improved to 0.015 inch spacing between gages compared to 0.025 inch with the standard technique. Detailed heating distributions along the stagnation line of swept fins subjected to an incident shock were measured with this technique. Also shown in the study by Berry and Nowak is the benefit of high resolution schlieren photography to complement the heating measurements by illustrating the complex interactions.

The last change to the infrastructure discussed in this section relates to teaming of experimentalists and

computationalists. Langley experimental and computational aerothermodynamicists had worked together previous to the Center reorganization on studies such as AFE, HL-20 and basic CFD calibration, and continued to do so following formation of the "new" Aerothermodynamics Branch. A dramatic increase in teaming between experimentalists and computationalists occurred due to the common causes associated with fast-paced, industry-led, focused programs such as X-33 and X-34. The need for synergism between these two disciplines was apparent as "anomalies" occurred in databases and as boundary layer transition criteria/information were required to address this major aeroheating issue for these suborbital vehicles. Experimentalists and computationalists pooled resources to address anomalies observed in force and moment and in aeroheating measurements. Extensive CFD predictions have been compared systematically to measurements and numerous lessons have been learned by both disciplines. The close working relationships required to establish boundary layer transition criteria will be demonstrated in a subsequent section. This increased teaming represents an implicit change to the Langley aerothermodynamic infrastructure.

Recent Applications

In recent years, large volumes of aerothermodynamic information have been generated at Langley, both experimentally (e.g. Fig. 19) and computationally, in support of high-priority, fast-paced programs. This section provides examples of results generated in some of these studies. As noted previously, essentially all testing in the Langley AFC since 1995 has been in support of focused programs. The first example in this section, however, illustrates a basic/fundamental study performed prior to the beginning of X-33 Phase I and X-34 in mid 1995.

Engineering codes were used to develop the flight aerodynamic database for the winged first stage of the Orbital Sciences Corporation (OSC) Pegasus vehicle. The Pegasus vehicle was designed, built and flown by OSC to place small payloads into low Earth orbit (LEO). First stage separation of the air-launched winged vehicle occurs at Mach 8. No wind tunnel tests were performed to simulate aerodynamic performance of this vehicle prior to its maiden flight, thus no Pegasus wind tunnel models existed. To take advantage of the Pegasus flight data, Langley, working in concert with the Dryden Flight Research Center, designed, fabricated and tested a Pegasus force and moment model in AFC tunnels and at supersonic conditions in the Langley Unitary Plan Wind Tunnel (UPWT). The stainless steel model was of high fidelity since the objective of this basic/fundamental study was to compare wind tunnel flight simulation results to flight. After 6 successful flights of the Pegasus, the vehicle was stretched for added performance (fuel) and the tail modified for the L-1011 carrier; this modified version was designated the Pegasus XL. On June 27, 1994, the Pegasus XL was lost on its maiden flight and Langley was requested to assist in determining the reason(s) for this loss. The Langley Pegasus force and moment model was quickly modified to the Pegasus XL outer mold lines (OML) and tested in Langley facilities. The resulting aerodynamic data were made available to the Langley accident study team within weeks of the start of the study and revealed the probable cause of the accident was associated with uncertainties/unknowns of the vehicle aerodynamic characteristics (Fig. 20). This example illustrated several points, most notably the need for flight simulation via tried and proven wind tunnel testing regardless of the geometric simplicity of the vehicle; and the benefits of basic/fundamental research studies.

The screening and optimization iterative process for the 3 industry RLV/X-33 Phase I concepts is "summarized" in Fig. 21. In all cases, the initial tests in AFC wind tunnels revealed the need to modify outer mold lines (OML) to improve aerodynamic performance and/or decrease aeroheating. This was the first aerothermodynamic screening/optimization study at Langley for which experimental qualitative and quantitative aeroheating information was available to the vehicle designer ahead of or about the same time as aerodynamic information. Thus, vehicle designers were able to perform parametric studies in which decisions on OML changes, center of gravity movements, trajectory tailoring, etc., were made via aerodynamic and aeroheating considerations. In previous aerospace vehicle design studies in the 1970s, 1980s and even early 1990s, experimental aeroheating information significantly lagged aerodynamic information; heating models required long fabrication times and were relatively expensive, thus often not constructed until near closure or closure on OMLs from an aerodynamic perspective. Although examples of global aeroheating distributions are illustrated in Fig. 21, the majority of AFC wind tunnel occupancy time was devoted to aerodynamic testing. Not revealed in Fig. 21 is the large amount of complementary CFD predictions performed for X-33 and for the corresponding Reusable Launch Vehicle (RLV) to demonstrate aerothermodynamic scaleability and traceablility. From an aerothermodynamic perspective, the success enjoyed in RLV/X-33 Phase I is the direct result of having a complete infrastructure, as defined by the "aerothermodynamic

chain," in place at Langley and the excellent planning, teaming and cooperation between industry and Langley.

A critical aeroheating issue for all aerospace vehicles is boundary layer transition; that is, in what region of the descent portion of the flight trajectory does transition from a fully laminar boundary layer over the vehicle to a fully turbulent boundary layer occur and what is the corresponding increase in heat transfer/ surface temperature. (Typically, the windward heat transfer rate increases by a factor of 3 to 5 as the boundary layer transitions from laminar to turbulent.) To avoid the worst case from an aeroheating perspective, boundary layer transition needs to occur well past the region of peak heating on the trajectory (i.e., the region where the product of freestream density and velocity to the third power is a maximum). Studies to deduce boundary layer transition characteristics for the Lockheed Martin Skunk Works (LMSW) X-33 Phase II concept are being performed at Langley. These comprehensive studies clearly illustrate the benefits of a synergistic experimental and computational capability. Complex boundary layer transition patterns measured on X-33 models in AFC wind tunnels with the phosphor thermography technique are complemented with predictions from inviscid/boundary layer and CFD codes, and results extrapolated to flight.

The effects of angle of attack (α) and Reynolds number on windward surface heating for the forebody of an early version of the LMSW X-33 is shown in Fig. 22 for Mach 6 air. At the lower values of α (Fig. 22(a)), two separate regions of transition, symmetrically opposed about the centerline, develop on the aft portion of the forebody. As α is increased, these regions of boundary layer transition move upstream and merge together; however, as α is increased from 40° to 45°; the transition region moves aft (i.e., shrinks). A large number of aeroheating (i.e., phosphor thermography) tests with the LMSW X-33 forebody were performed in the Langley 20-Inch Mach 6 Air Tunnel for a range of α , Reynolds number, and discrete boundary layer trips of various heights (K) and locations on the model surface (e.g., Thompson et al., 1998). For each tunnel run, the boundary layer thickness(δ), momentum Reynolds number (Re_θ) and boundary layer edge Mach number (M_e) were predicted via the LATCH code and an inviscid solution provided by the LAURA or DPLUR codes (Hamilton et. al., 1998). This large data set, involving several hundred tunnel runs and corresponding predictions, was correlated in terms of Re_θ/M_e vs. K/δ (Fig. 23(a)), which has been demonstrated to predict boundary layer transition onset and completion to full turbulence reasonably well for the shuttle orbiter (Berry et al., 1997). The findings from this correlation of ground-based measurements on the X-33 forebody at Mach 6 were applied to flight conditions (Fig. 23(b)). The primary drivers for this study were to: first, determine if laminar flow would indeed be achieved during ascent and the first portion of descent; and secondly, determine where along the descent trajectory, relative to the region of peak heating, would transition to turbulence most likely occur. These results were used to establish a "transition plain in the sky," (Fig. 23(c)) whereby the X-33 vehicle would experience laminar flow above the plain and turbulent flow below it. Such information is extremely valuable in the design of the thermal protection system (TPS) and/or assessing whether TPS design limits will be exceeded in flight. Details of this study are presented by Thompson et al., 1998.

The effect of metallic panel bowing due to thermal expansion (i.e., temperature gradients across panels) in flight on boundary layer transition is a concern for the X-33 vehicle. Ceramic models have been constructed at Langley that scale panel bowing for an array of panels. Over 20 high fidelity ceramic models, having bowed panel heights of 0.002, 0.004 and 0.006 inch, have been constructed. To build ceramic models with such high fidelity detail is in itself a significant accomplishment. Global heating distributions for a smooth surface model and a bowed-panel surface model at Mach 6 are shown in Fig. 24. It is interesting to note the influence of the bowed panels on heating to the fins and the effect that a very small surface discontinuity near the leading edge of the fin can have on heating. This is another example of capabilities not available to the aerothermodynamic community until recently.

A complex flowfield exists over the aft portion of the X-33 lifting body that represents a technical challenge to both aerothermodynamic experimentalists and computationalists (Gnoffo et al., 1997). Considerable computer resources are required to perform a true tip-to-tail solution for this vehicle as required to predict aerodynamic performance. The aerospike engine module resides within the wake of the fuselage as do the body flaps. Computer resources required for tip-to-tail solution are illustrated in Fig. 25 where the wake flow was predicted in order to determine aeroheating to the engine module at flight conditions. Using the LAURA code, a converged solution amenable for heat transfer predictions (i.e., relatively fine grid) required 3 hours on a Cray C-90 for the nose region and another 37 hours were required to solve the flowfield downstream to the beginning of the fin. 60 additional hours were required for the remaining 30 percent of the fuselage for a total of 100 hours for a nose tip-to-tail

(without engine module) solution. However, to achieve the true tip-to-tail solution, another 150 to 200 hours of C-90 time was required for the wake region. Langley stepped up to this computational challenge, has run several cases, including both laminar and turbulent wakes, and has provided the only such aerothermodynamic information within the program.

Like the X-33 vehicle, the X-34 is a hypersonic flight, autonomous landing vehicle that requires a comprehensive aerothermodynamic database for successful flight. Following a fast-paced aerothermodynamic assessment/optimization for X-34, the OMLs were frozen in December 1996 (Fig. 26) and benchmarking studies initiated. High fidelity models were designed, fabricated and tested at Langley over a range of flow conditions (subsonic through hypersonic) and complementary CFD predictions performed. The corresponding aerodynamic (Brauckmann, 1998) and aeroheating (e.g., Berry et al., 1998; Riley et al., 1998; Kleb et al., 1998) databases were developed in a timely manner, in large part due to planning and to the teaming of Orbital Sciences Corporation (OSC) and Langley aerodynamicists and individuals working aeroheating issues.

In striving to obtain the most accurate aerothermodynamic information possible for extrapolation to flight, the X-33 and X-34 programs provided opportunities to perform extensive comparisons of CFD predictions to wind tunnel measurements for complex geometries and to make code to code comparisons. An example of such comparisons is shown in Fig. 27 for the baseline X-34 configuration at Mach 6 and for laminar and turbulent boundary layers (Merski, 1998). Comparisons between fully laminar and between fully turbulent predictions with the LAURA and GASP codes are quite good. (Collectively, the LAURA and GASP codes have provided essentially all the CFD aeroheating predictions for the X-33 and X-34 programs. This does not include the large number of inviscid/boundary layer solutions via the LATCH/ LAURA or DPLUR codes. GASP is exercised by the Ames Research Center to determine the aeroheating environment for TPS design/sizing.). Predictions via LAURA/GASP of the X-34 windward heating corresponding to a laminar boundary layer and to a turbulent boundary layer are in good agreement with measurement. These comparisons illustrate a well recognized deficiency in present computational capabilities; i.e., the inability to accurately predict the onset of transition and the aeroheating within regions of transitional flow. Vehicle designers must rely on wind tunnel measurements and previous experience in extrapolations to flight for this information.

The first application of a newly developed technique for extrapolation of global aeroheating measurements in a hypersonic wind tunnel directly to vehicle surface temperature values in flight was for the X-34 program (Merski, 1998). Examples of this capability are shown in Fig. 28 for laminar and turbulent boundary layers. Vehicle surface temperatures determined from extrapolation of wind tunnel measurements are observed to, in general, be in good agreement with LAURA predictions for flight conditions. Extra-polations to flight are performed immediately after a tunnel run; thus, vehicle designers have access to surface temperature distributions for the vehicle in flight within 3 to 4 weeks after initiation of the study (i.e., time required to build and test ceramic models with the phosphor thermography technique).

The final examples presented in this section are for the Hyper-X vehicle (Fig. 29) and the X-38 (Fig. 30). Unlike blunt configurations such as the X-33 and X-38 at high incidence where flow within the shock layer is low supersonic and perhaps subsonic, the flowfield for the slender Hyper-X configuration corresponds to high supersonic-hypersonic Mach numbers for hypersonic free- stream conditions and thus exhibits different fluid dynamic characteristics. The hypersonic aerodynamic data base for the Hyper-X was also generated in a fast-paced manner (Fig. 29), being performed following the completion of X-33 Phase I. Aerodynamic screening/optimization was performed for the Hyper-X vehicle mated to the Pegasus booster on ascent, separation interference, and the Hyper-X as a free flyer on descent. This extensive aerodynamic data was complemented by forebody boundary layer transition studies for smooth surfaces and for discrete trips. Hypersonic aerodynamic and aeroheating measurements for the X-38 (e.g., Berry et al., 1997) have been performed in the AFC (Fig. 30). These tests support JSC's development of a crew return vehicle or lifeboat for the International Space Station. The opportunity will exist in the near future to compare aerodynamic/aeroheating measurements for the X-38 in European hypersonic wind tunnels, most notably ONERA S4 and F4, to those obtained in the Langley AFC.

Several "lessons learned" or observations from recent applications of Langley's aerothermodynamic infrastructure have been discussed previously, such as: the need for detailed planning and clear definition of customer and supplier expectations; the advantages of industry and NASA aerothermodynamicists teaming in spite of very different cultures; the advantages of a combined, synergistic experimental and computational approach, as compared to an all experimental or an all

computational approach; the flexibility and efficiency provided when computationalists, experimentalists and wind tunnel scheduling for the AFC reside within a single organization; the critical role model design/fabrication plays in the overall success of the program; the advantages provided by phosphor thermography; the challenges associated with increasing demands on resources during a period of downsizing and rapid change; and so forth. Many, if not most, of these observations are intuitive and possibly could have been listed prior to the recent applications. However, buried within the details of most observations were surprises or lessons learned; some were minor and others were significant. Collectively, these lessons learned, both large and small, provide the corporate knowledge to be leveraged against in future aerothermodynamic studies of advanced space transportation system concepts and planetary exploration including sample returns to Earth.

Another lesson learned concerns the premature freezing of OMLs. Due to the extremely fast pace of programs such as X-33 and X-34, the OMLs may be frozen prior to optimizing the aerodynamic performance/aeroheating characteristics in order to satisfy milestones; e.g., prior to performing/completing configuration buildup tests to determine the contribution of each component (e.g., body flaps, wings/fins, tails, etc.) to aerodynamic performance. Knowledge and understanding of each component's contribution and of possible interactions between components is particularly important if "anomalies" occur while establishing the flight aerodynamic data book via benchmark testing. If component buildup was not performed/completed during aerothermodynamic optimization, then the high-fidelity benchmark models should be designed and fabricated to allow configuration buildup testing just in case.

The last sample lesson learned discussed in this section is also related to the fast pace of present programs relative to present capabilities. Insufficient time for a benchmarking task may lead to basic rules for aerodynamic benchmarking being abbreviated/ ignored, thereby possibly compromising the quality of the data. Some of the basic rules for benchmarking are summarized as follows:

1. Know what you are testing—models must be very high fidelity and fully certified via extensive quality assurance prior to testing.
2. Know what you are testing in—facility must have been extensively calibrated and flow quality deemed adequate for benchmarking. Ideally, facility should have contributed creditably to Shuttle Orbiter or other flight vehicle aerodynamic data book. Effect of model in test section on facility performance should be determined (e.g., nozzle wall pressures measured).
3. Know model attitude accurately—model angles of attack and sideslip and roll should be measured with wind on. If this is not feasible, care must be exercised to correct for sting bending and thermal expansion of support components. Naturally, when varying angle of attack over a large range, the model shock layer must be maintained well within the facility test core.
4. Match the strain-gage balance to the task—select balance(s) that provide the highest possible accuracy for critical components (e.g., pitching moment) and test balance for compensation to thermal gradients. Perform accurate weight tares and monitor balance stability (i.e., drift).
5. Perform no-flow run—determine performance of setup without flow on (i.e., background).
6. Perform uncertainty analysis for each run.
7. Augment principal measurement with other diagnostics—e.g., complement force and moment measurements with schlieren/ shadowgraph photography; thermography techniques to examine regions of boundary layer transition, shock impingement, etc.; surface oil flow to examine regions of flow separation-reattachment, etc.
8. Exercise redundancy—test in different facilities at same/similar flow conditions; last run in test series should be repeat of first run; test different strain-gage balances for identical test conditions; test different scale models in same facility; determine hysteresis effects by varying angle of attack upward and downward; "soak" model at given angle of attack to determine drift (e.g., from balance thermal gradients) during a run; etc.
9. Measure base pressure and model cavity pressure—determine if base pressure is steady during "soak" run; follows expected trends during change in angle of attack; is reasonable (i.e., expected) level.
10. Quantify effect of support system—determine effect of sting diameter and length; determine effect of sting vs. blade support (e.g., test sting, blade and combination sting and blade); determine effect of model position relative to strut; etc.

11. Complement wind tunnel measurements with CFD predictions.
12. Bound the problem—at each Mach number, test over maximum Reynolds number range possible within constraints of balance/hardware; test in heavy gas whether or not significant real-gas effects expected; exceed expected ranges of angles of attack and sideslip and control surface deflections.
13. Question everything—take nothing for granted.
14. Communicate—communicate—communicate—clearly communicate requirements to facility engineers and to technicians via pretest meetings; customers should be present during setup and testing, etc.

These rules are not detailed nor all-inclusive; they are intended to serve as basic guidelines in future studies and provide impetus for improvement (i.e., perform specific rules faster and better).

As the time allowed for aerothermodynamic assessment/optimization/benchmarking decreases for future flight experiment programs involving new vehicles, the aerothermodynamic community must perform faster with no sacrifice in quantity/quality of information generated. Doing aerothermodynamic studies faster, better, cheaper is addressed in the next section.

Future Plans

As is well recognized, plans to advance aerothermodynamic capabilities over the next 3 to 5 years will not become reality without adequate resources in terms of people (i.e., expertise/competency), equipment (i.e., proper tools) and, naturally, a source of steady, long-term funding. Future projections are generally associated with various levels of uncertainty. This level of uncertainty is particularly high in the present environment of rapid change via external forces. Some future plans for the Langley aerothermodynamic program are presented in this section. In attempting to be reasonably realistic, these plans are based on working within the framework of present resources. The plans presented herein are much less ambitious than those offered in previous years, particularly those previous plans projecting the construction of new facilities.

In formulating these plans, a basic assumption is made that future utilization of Langley aerothermodynamic capabilities will be primarily for focused programs with relatively little basic/fundamental research performed. As the pace quickens for future space transportation and planetary programs, customers will require significant reductions in the time required to obtain aerothermodynamic data for screening; and will require more data faster so vehicle designers can optimally match vehicle performance to mission requirements. To enhance the vehicle optimiza-tion process, integrated aerothermodynamic computa-tional and experimental data will be needed early in the design process.

The primary goal of the subject future plans is to provide aerothermodynamic capabilities that are faster, better, cheaper than presently available. Several of the planned technologies, if successfully achieved, will provide all three objectives; i.e., provide a faster, better, cheaper capability. The primary emphasis in the plan is on obtaining aerothermodynamic information faster. The overall goal is to significantly reduce the time for, and enhance the fidelity to, the vehicle (e.g., RLV) design cycle process, and collectively provide a leap forward in aerodynamic and aeroheating design, assessment, and optimization for future aerospace vehicle concepts.

From a facility perspective, future emphasis will be placed on maintaining present capabilities without backsliding. No new facilities will be planned/ advocated. None of the hypersonic wind tunnels at Langley previously placed on standby or mothballed will be reactivated. Facility upgrades/enhancements will continue to be performed via minor CoF projects, but not with the level of success enjoyed in previous years. Maintaining the 20-Inch Mach 6 Air and 31-Inch Mach 10 Air Tunnels in good working order will receive priority.

The major contributor to the total time required to perform a force and moment study in a hypersonic wind tunnel (i.e., time from study initiation to completion of testing and data analysis) is generally the time required to design and fabricate the model. Techniques to construct complex, high fidelity metal-matrix/ composite matrix force and moment models, amenable for heated flows, in terms of days as opposed to months will be explored. Techniques/processes to construct ceramic models will continue to be refined and simplified to decrease time of construction and increase fidelity (e.g., decrease shrinkage/slumping), strength and repeatability of physical properties. Coating techniques for phosphors will also be refined for more smoothness and uniformity. The use of ceramic models for force and moment testing will increase for aerodynamic assessment/optimization studies and perhaps for benchmarking. Assuming the successful development of advanced diagnostic techniques (discussed next), the vast majority of models tested in the Langley AFC in the future will be ceramic.

Construction of model sting/blade support systems from a material(s) having low thermal conductivity so to minimize heat conduction into the strain-gage balance will be revisited. Also on the subject of model support, systems will be designed and built for multiple-body separation studies, both from the perspective of aerodynamic performance and aeroheating studies. Models having remotely controlled control surfaces will be explored. Although this technique is commonly used in Langley subsonic to supersonic tunnels, it has not been used in the AFC primarily because of small model size and cost (trade between increased model cost and cost to operate facility). Technological advancements in recent years may make remotely controlled force and moment models feasible for the AFC.

Advancements in diagnostic techniques are expected to provide major enhancements in aerothermodynamic testing capabilities. A quantum leap in capability will be provided by the development and application of a non-intrusive 3-color surface fluorescence technique for simultaneous global measurements of model surface pressure and temperature in heated hypersonic wind tunnels (including the CF_4 tunnel). This technique will extend the temperature range of the presently used phosphor thermography system and provide a smoother model surface via a different coating technique. Simultaneous global pressure and temperature measurements represent a powerful capability for aerothermodynamicists analyzing complex surface patterns corresponding to complex flowfield phenomena, and for vehicle designers seeking information for structural and TPS design. The subject technique will represent a critical step towards the ultimate goal of simultaneous force and moment, pressure, and temperature (heat transfer) measurements.

This 3-color technique will eventually replace the highly successful 2-color phosphor thermography technique presented used for essentially all aeroheating studies in the Langley AFC. Until that time, the 2-color technique will continue to be improved. Two or three systems will be applied simultaneously (Fig. 31) to provide multiple views of the model, thereby providing more information faster and reducing the number of tunnel runs required (i.e., cost). Fluorescence techniques will be used during force and moment testing to identify the state of the boundary layer (i.e., laminar, transitional or turbulent) and locations of flow separation and reattachment often important in the interpretation of force and moment data (e.g., for deflected control surfaces and for wings/fins). Theories/procedures for the extrapolation of phosphor thermography aeroheating measurements to vehicle flight surface temperatures will continue to be developed, calibrated against the rich flight data base for the Shuttle orbiter, and applied to future aerospace vehicle concepts. This capability provides the vehicle designer with a powerful tool for TPS material selection, split line definition and a sizing. Information is made available to designers immediately following a tunnel run.

It has been stated that for conventional-type hypersonic wind tunnels, experimentalists can accurately measure aeroheating everywhere except where it is needed most—thin model surfaces such as fins, tails, wings and surfaces with small radii of curvature such as leading edges. Strides will be taken to more accurately infer heating to such surfaces via advances in fluorescence techniques including time response, routine use of 2D/3D conduction codes and/or use of model materials having extremely low thermal conductivity to minimize conduction.

Techniques will be developed and applied to accurately measure model angle of attack and angle of sideslip throughout the tunnel run. Development of stings containing accelerometers will continue. These stings must remain small in diameter to avoid significant support interference effects yet be capable of withstanding maximum loads including those associated with flow breakdown in the tunnel. Installation of accelerometers in models will be explored. Optical techniques will be examined for monitoring model attitude during the run when under pressure and thermal loads. Although the pitch-pause method for force and moment measurements is a tried and proven standard in the Langley AFC, continuous pitch capability will be implemented to substantially increase productivity and reduce thermal effects on the balance. Continuous pitch will require sufficient strain-gage balance and data acquisition response and real-time model attitude measurements.

Future advancements in computational aerothermodynamics will include the development of a process for coupling an unstructured inviscid code to a boundary layer solver; again with the emphasis on reduction of vehicle design cycle time. The capability to create an unstructured grid directly from the geometric surfaces generated via systems analysis studies will be implemented. This will reduce the time required to generate a grid, enabling rapid aerodynamic force and moment predictions as well as provide inviscid solutions to the boundary layer solver. This solver will be used to generate a database of heating solutions over the expected design envelope and the process automated, as will be the interrogation of this database.

Thus, rapid assessments will be made to determine impacts due to changes in vehicle geometry, center of gravity, trajectory, etc.

Optimization of the LAURA code for massive parallel processing on nonvector machines will be completed to provide viscous solutions more rapidly for the vehicle design phase. An unstructured hypersonic viscous solver will be developed by modifying the FELISA code. Such a code will reduce the total time to obtain a full viscous, tip-to-tail solution by approximately an order of magnitude when compared with a structured solver. This significance reduction in total time will allow viscous solutions to be used to calibrate fast-running engineering codes and complement ground-based testing in the early phase of vehicle design. The plan is to evolve the unstructured hypersonic viscous solver to the same capabilities as the LAURA code; and to eventually replace the LAURA code as the primary source of benchmark, computational aerothermodynamic information at Langley. (The relationship of the subject unstructured hypersonic viscous solver to the newly developed General Unstructured Software Toolkit (GUST) by AeroSoft, Inc. is expected to be analogous to the present relationship at Langley between LAURA and GASP.)

Additional computational plans include the continued development and validation of advance turbulence models for hypersonic flows; development of jet plume-flowfield-surface interaction (i.e., reaction control system) capabilities; possible revival of equilibrium radiation codes to address aerothermodynamic issues associated with very high velocity return to Earth missions; continued advancements in DSMC via coupling to continuum Navier-Stokes solvers to predict RCS phenomena in the rarefied flow regime and extension of DSMC capabilities lower into the atmosphere; the exploiting of boundary-layer theory and triple-deck theory to compute the effects of global and local changes to surface catalysis on computed heating rates; and development of laminar wall function approximation to help reduce grid requirements and accelerate convergence of CFD solutions for hypersonic, viscous flow.

Lastly, Langley aerothermodynamicists will work with, and in support of, Dryden engineers in the extraction of aerodynamic and aeroheating data from flights of X-33 (Fig. 32), X-34 and Hyper-X. Ground-based measurements and CFD predictions used in the development of the respective flight aerodynamic data books will be compared to flight data. Additional wind tunnel tests and/or CFD predictions will be made as appropriate. These comparisons will complete the aerothermodynamic triad (Fig. 1) and close the cycle (i.e., establish the missing link) in the aerothermodynamic process illustrated in Fig. 5.

Resume

Aerothermodynamics, encompassing aerodynamics, aeroheating, and fluid dynamics and physical processes, is the genesis for the design and development of advanced space transportation vehicles and provides crucial information to other disciplines such as structures, materials, propulsion, avionics, and guidance, navigation and control. Sources of aerothermodynamic information are ground-based facilities, Computational Fluid Dynamic (CFD) and engineering computer codes, and flight experiments. Utilization of this aerothermodynamic triad provides the optimum aerothermodynamic design to safely satisfy mission requirements while reducing design conservatism, risk and cost. The iterative aerothermodynamic process for initial screening/assessment of aerospace vehicle concepts, optimization of aerolines to achieve/exceed mission requirements, and benchmark studies for final design and establishment of the flight data book are reviewed. Aerothermodynamic methodology centered on synergism between ground-based testing and CFD predictions is discussed for various flow regimes encountered by a vehicle entering the Earth's atmosphere from low Earth orbit. An overview of the resources/infrastructure required to provide accurate/creditable aerothermodynamic information in a timely manner is presented. Impacts on Langley's aerothermodynamic capabilities due to recent programmatic changes such as Center reorganization, downsizing, outsourcing, industry (as opposed to NASA) led programs, and so forth are discussed. Sample applications of these capabilities to high Agency priority, fast-paced programs such as Reusable Launch Vehicle (RLV)/X-33 Phases I and II, X-34, Hyper-X and X-38 are presented and lessons learned discussed. Lastly, enhancements in ground-based testing/CFD capabilities necessary to partially/fully satisfy future requirements are addressed.

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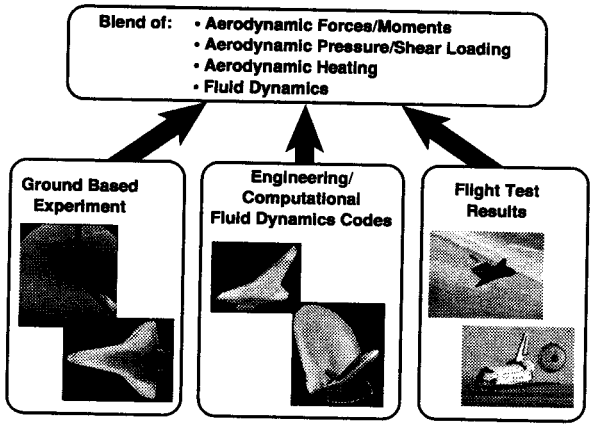


Fig. 1 Aerothermodynamic information sources.

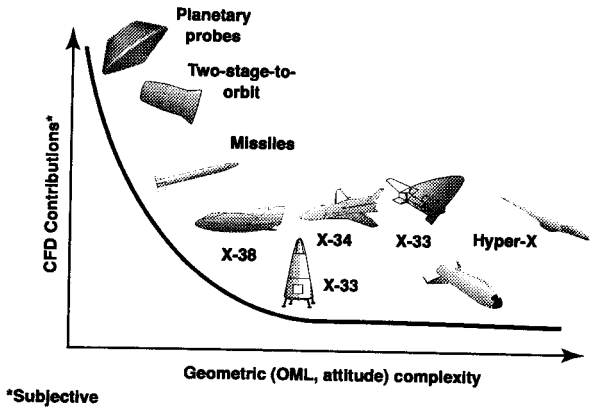


Fig. 4 CFD contribution.

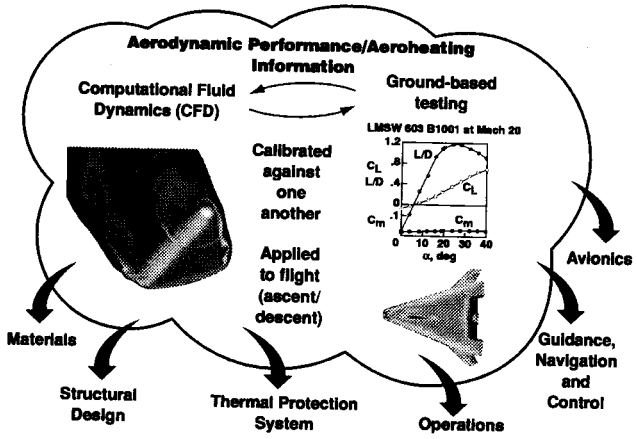


Fig. 2 Provider of critical information to other disciplines.

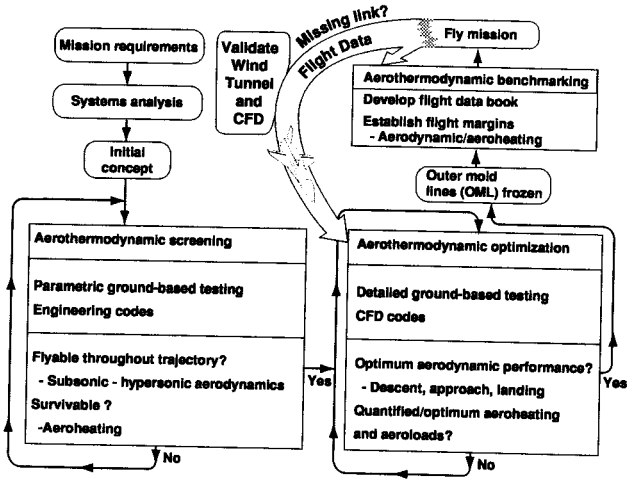


Fig. 5 Aerothermodynamic process.

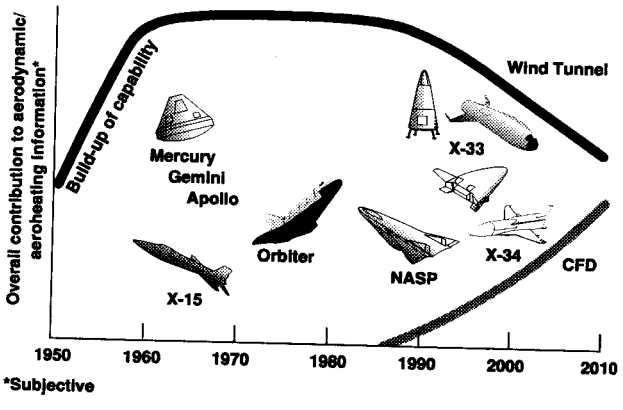


Fig. 3 Relative contribution of ground-based testing and CFD.

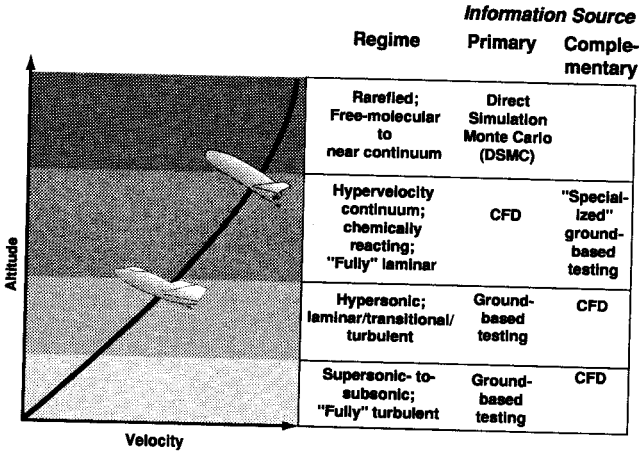


Fig. 6 Proposed aerothermodynamic methodology.

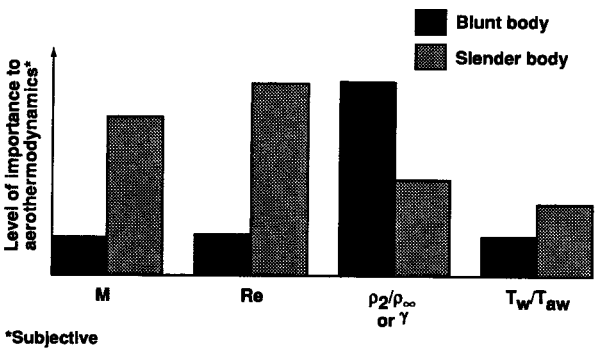


Fig. 7 Importance of basic flight simulation parameters.

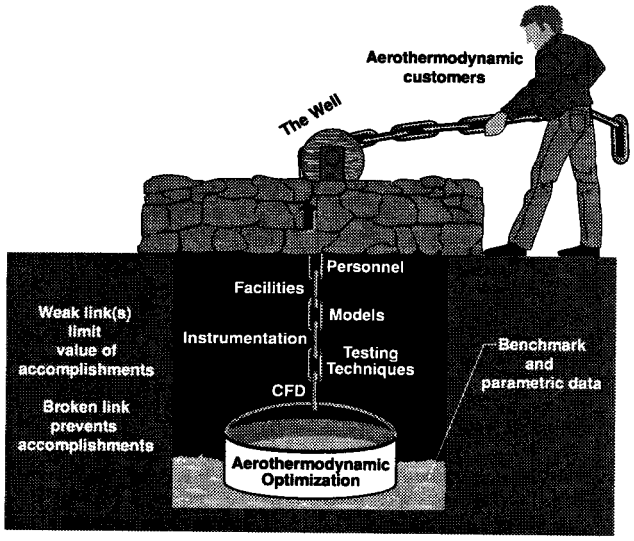
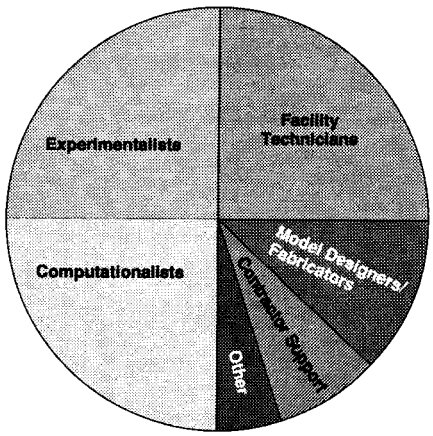
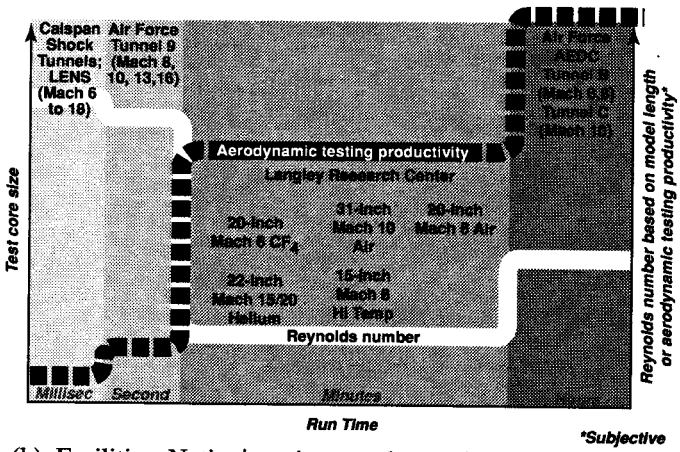


Fig. 8 "Aerothermodynamic chain" representing infrastructure.



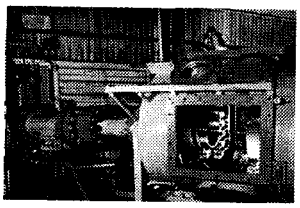
(a) Personnel.

Fig. 9 "Aerothermodynamic chain" links (i.e. components of infrastructure).

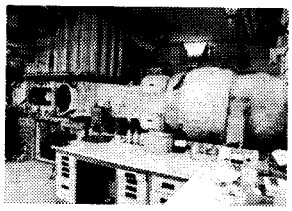


(b) Facilities: Nation's active aerothermodynamic facilities.

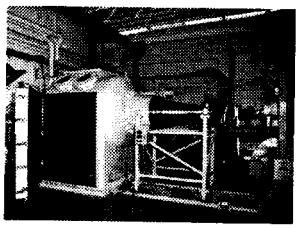
Fig. 9 Continued.



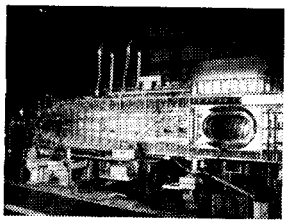
15-Inch Mach 6 Hi Temp. Air



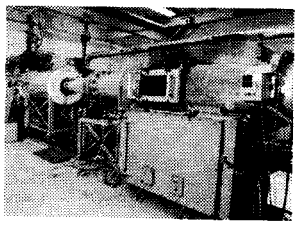
20-Inch Mach 6 Air



20-Inch Mach 6 CF_4



31-Inch Mach 10 Air



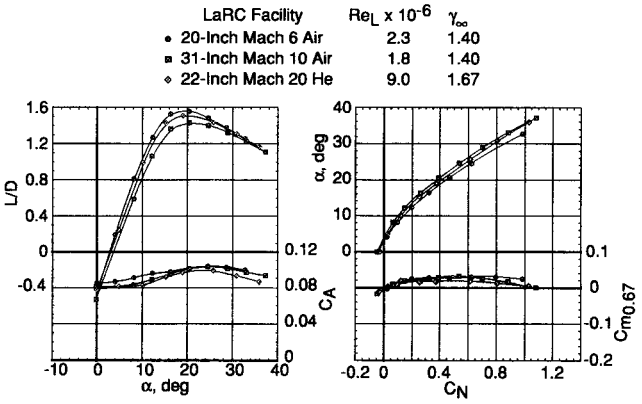
22-Inch Mach 15/20 He

(c) Facilities: Langley Aerothermodynamic Facilities Complex (AFC).

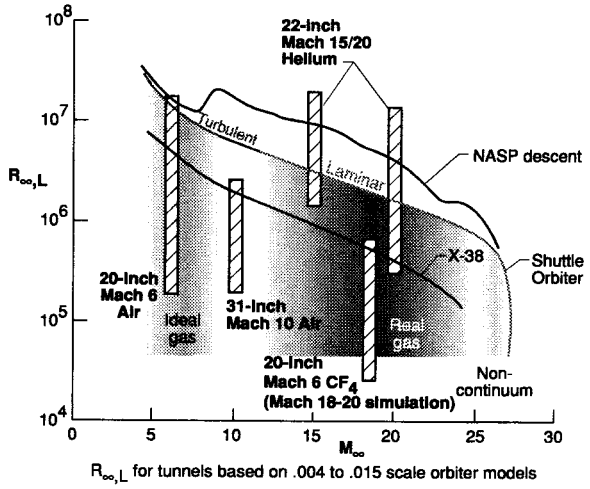
Fig. 9 Continued.

Facility	P_o, psia	T_o, °R	M_∞	R_∞ x10^{-6} ft^{-1}	$\frac{\rho_2}{\rho_\infty}$	Run time, sec	Core size, in
15" M-6 Hi T. Air B 1251A	45-550	940-1260	6	0.5-10.0	5.3	180	10-11 Diam
20" M-6 Air B1247D	30-500	760-940	6	0.5-9.0	5.3	120-500	12x12 14x14
31" M-10 Air B 1251A	150-1450	1850	10	0.2-2.2	6.0	120	12x12 14x14
20" M-6 CF_4 B 1275	100-2000	1100-1480	6	0.05-0.75	11.8	20	12-14 Diam
22" M-15/20 He B 1247B	300-3500	500-1060	14-20	2.0-30	4.0	45	10-12 Diam

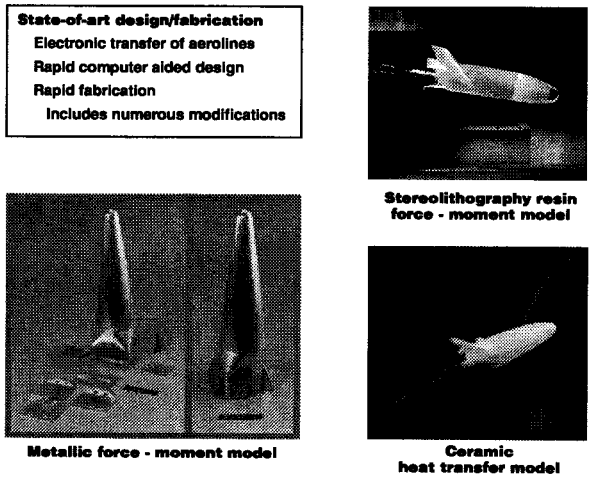
(d) Facilities: AFC flow conditions.
Fig. 9 Continued.



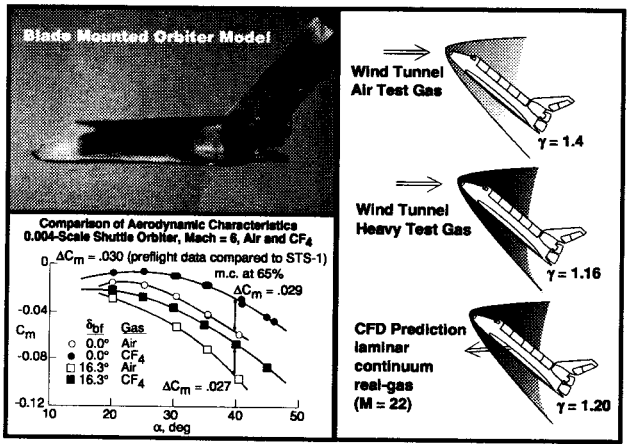
(g) Facilities: Air-helium simulation for SSTO winged body.
Fig. 9 Continued.



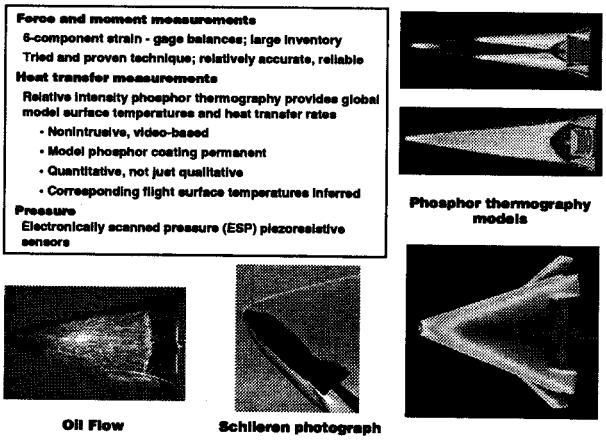
(e) Facilities: AFC Mach number - Reynolds number simulation capability.
Fig. 9 Continued.



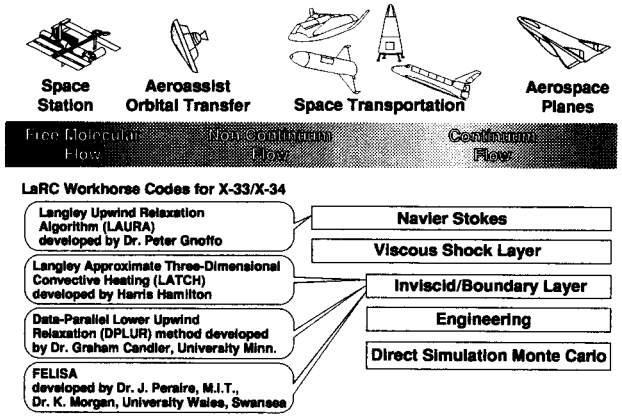
(h) Test articles/models.
Fig. 9 Continued.



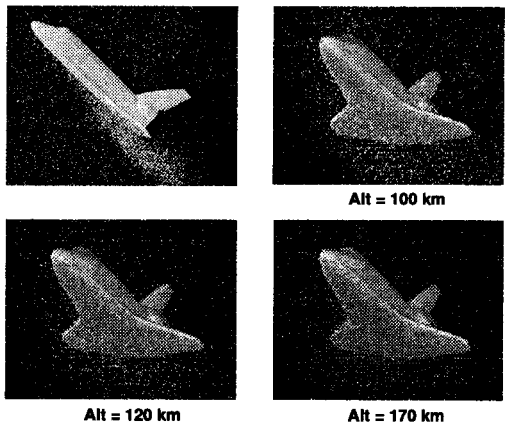
(f) Facilities: Simulation of real-gas effects on aerodynamic characteristics via heavy gas (CF_4).
Fig. 9 Continued.



(i) Instrumentation.
Fig. 9 Concluded.



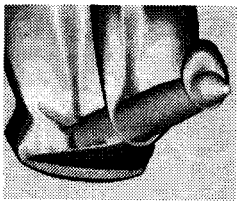
(a) Rarefied/continuum flow capabilities.
Fig. 10 Computational aerothermodynamics.

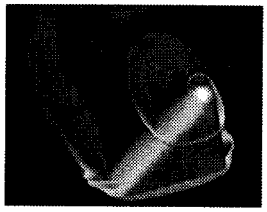


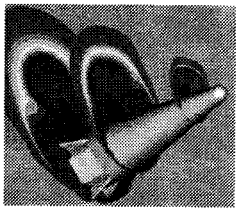
(c) Direct Simulation Monte Carlo (DSMC).
Fig. 10 Concluded.

Provides aerodynamic performance, pressure loads and heating loads for take-off, ascent to suborbit/orbit, descent and landing (subsonic to hypersonic)

- Extremely complex, three dimensional configurations
- Rarefied (very high altitude) to continuum
- Ideal (M < 6) and real gas - dissociated/ionized shock layer
- Laminar and turbulent conditions







(b) Sample applications.
Fig. 10 Continued.

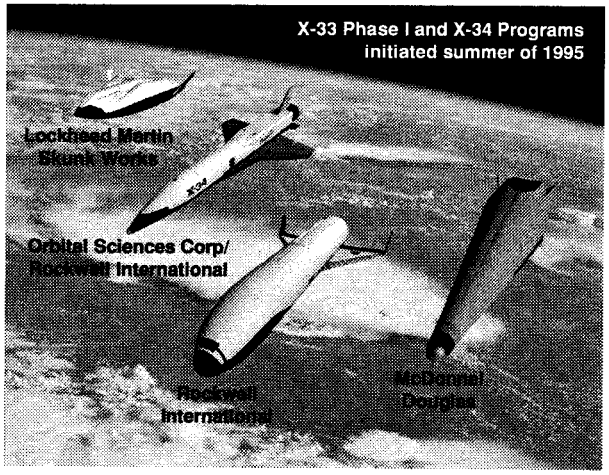
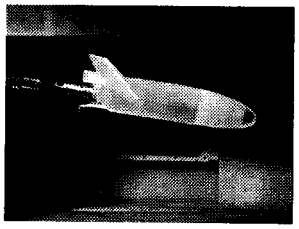
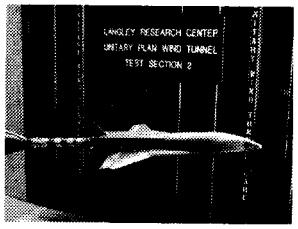
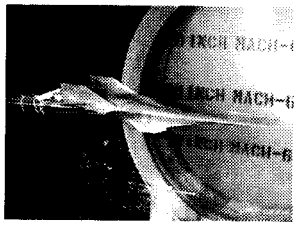


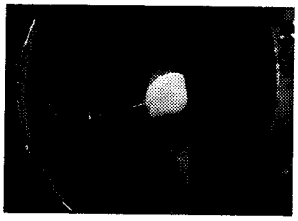
Fig. 11 Aerothermodynamic studies initiated 1995.

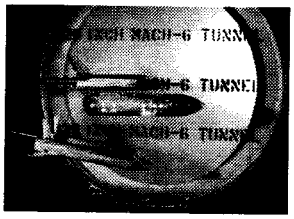

X-33



X-34


Hyper-X


Planetary


Liquid Flyback Booster

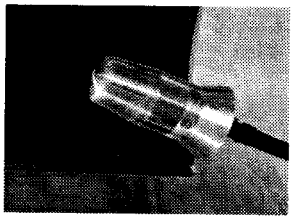

Kistler

Fig. 12 Sample of aerothermodynamic studies; 1996 to present.

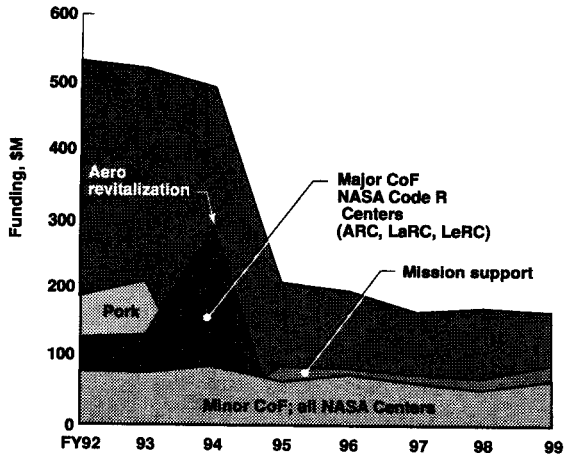
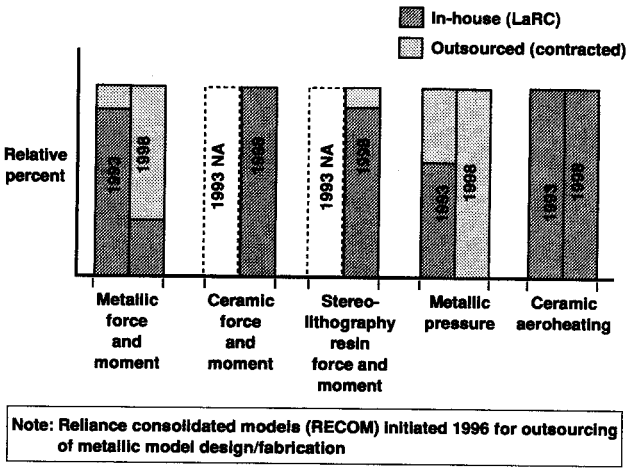
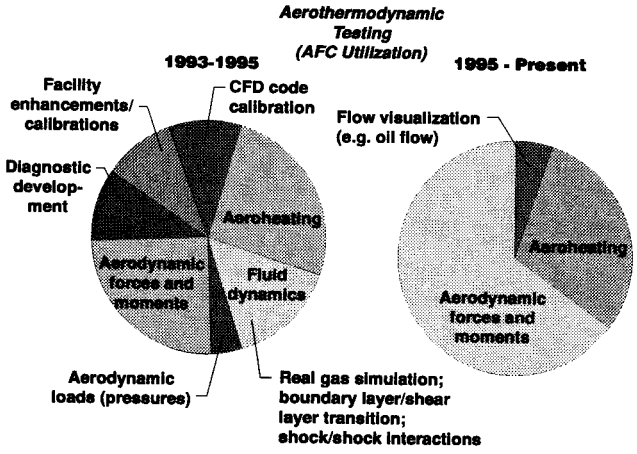


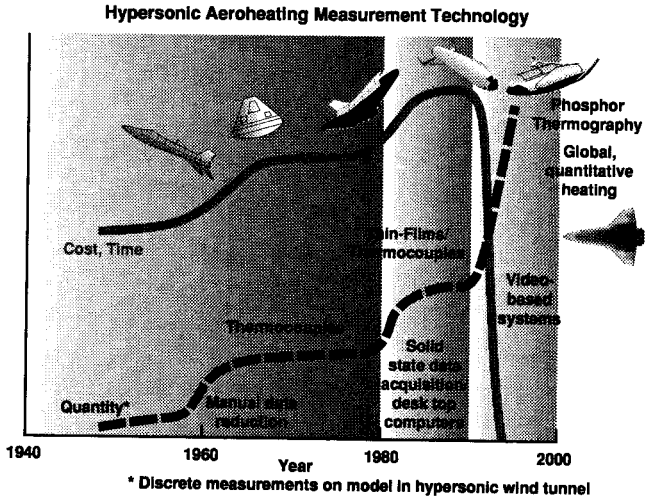
Fig. 13 NASA Construction of Facilities (CoF) funding.



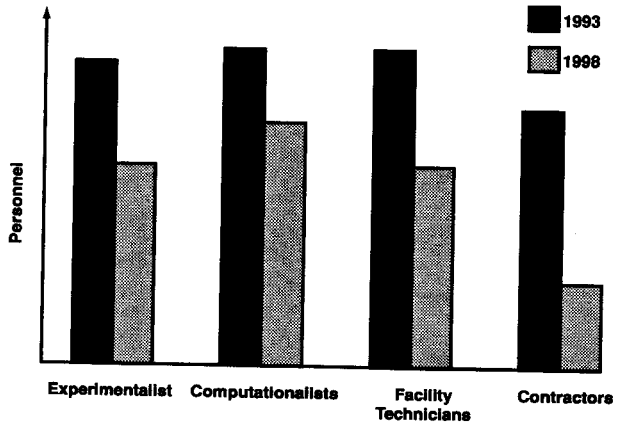
(c) Model construction.
Fig. 14 Continued.



(a) Technical disciplines.
Fig. 14 Changes in Langley aerothermodynamic capabilities; 1994 to present.



(d) Instrumentation.
Fig. 14 Concluded.



(b) Personnel.
Fig. 14 Continued.

Fig. 15 Ceramic model fabrication.

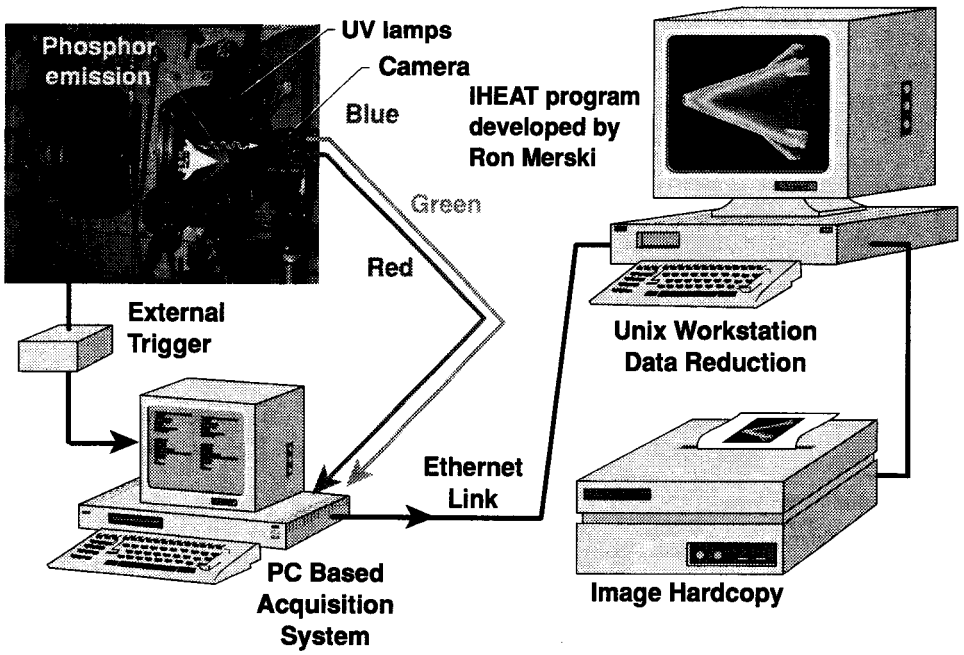


Fig. 16 Phosphor thermography system.

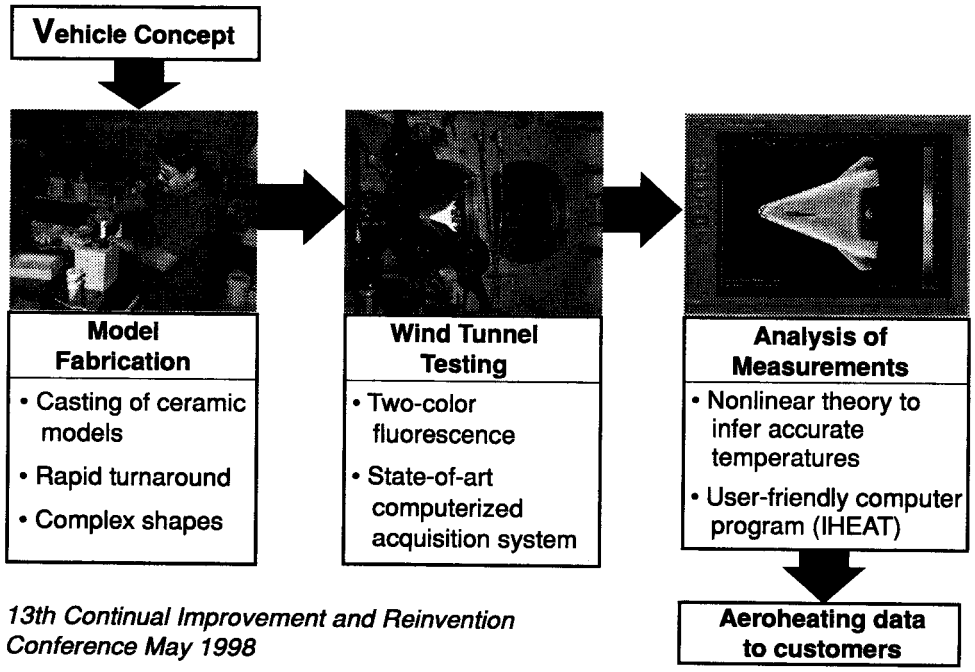


Fig. 17 Phosphor thermography quantitative aeroheating process.

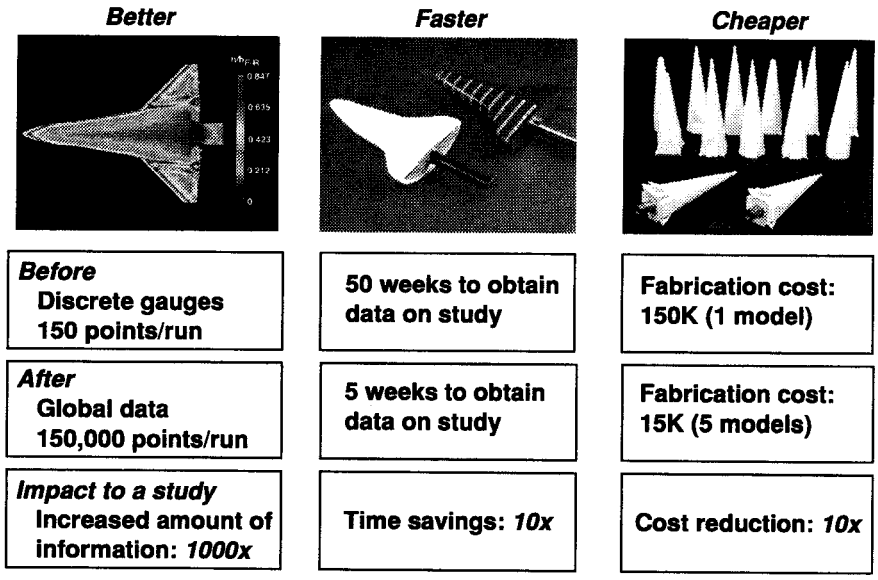


Fig. 18 Advantages of phosphor thermography technique.

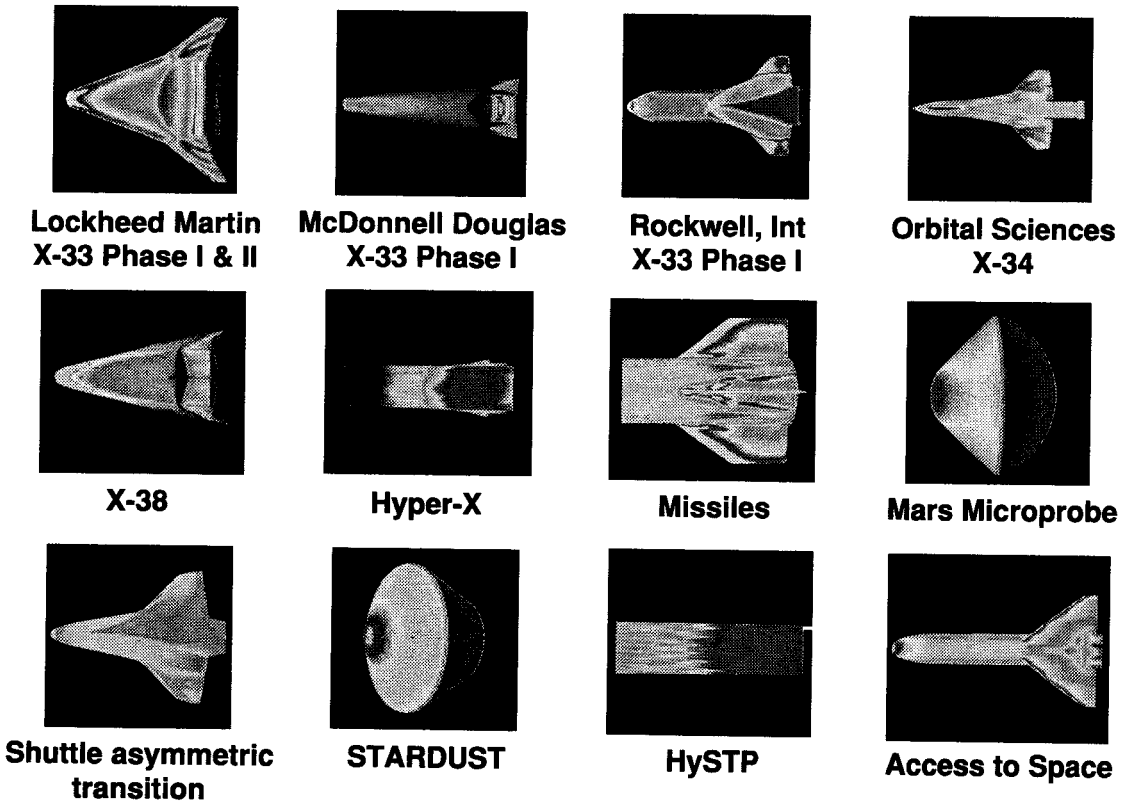


Fig. 19 Sample applications of aerothermodynamic capabilities; 1996 to present.

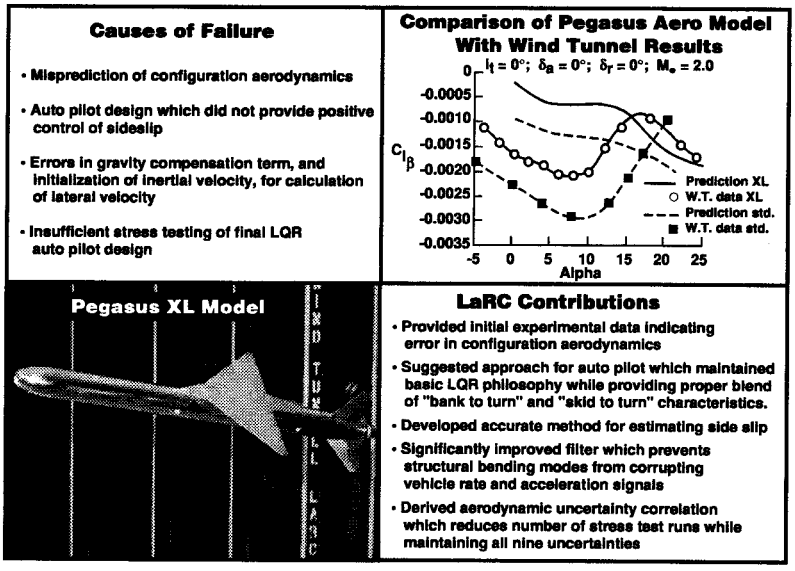
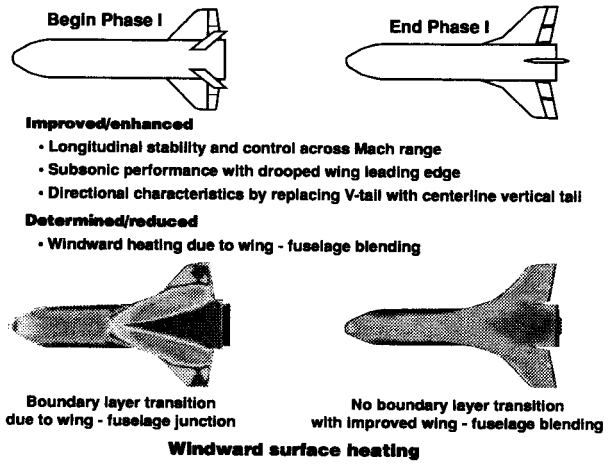
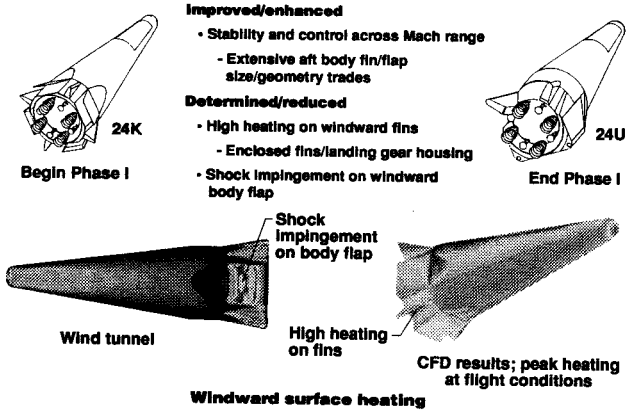


Fig. 20 Pegasus XL study.



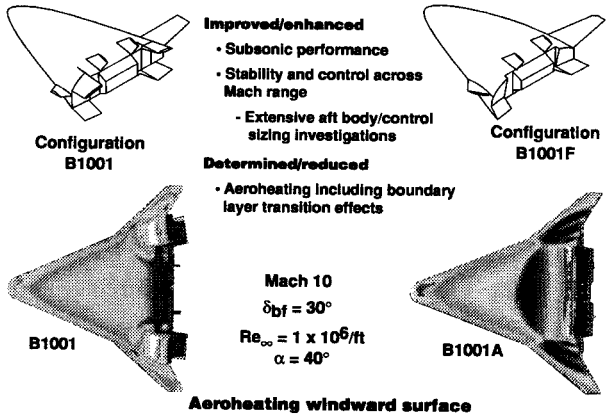
(a) Rockwell International concept

Fig. 21 RLV/X-33 Phase I configuration development.



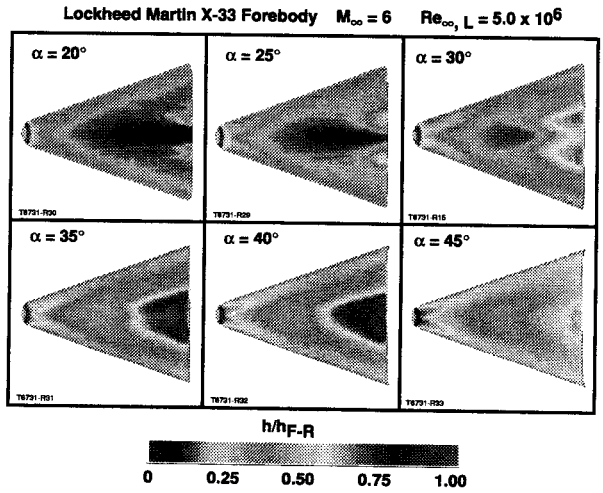
(b) McDonnell Douglas/Boeing concept.

Fig. 21 Continued.



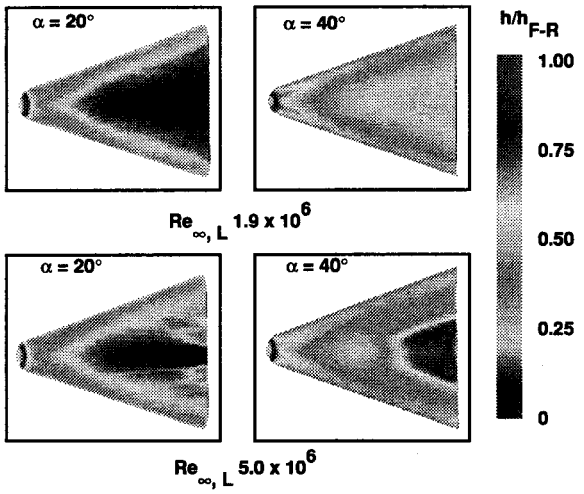
(c) Lockheed Martin concept.

Fig. 21 Concluded.

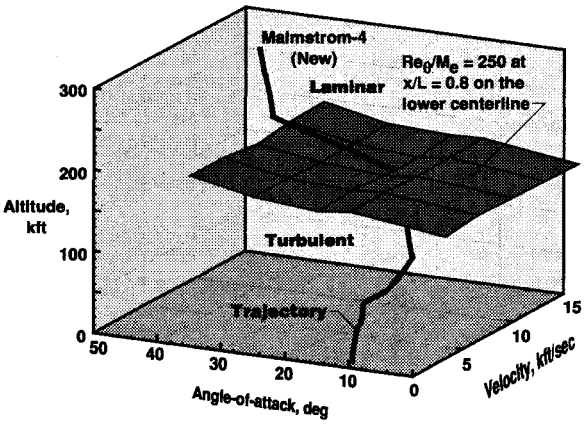


(a) Effect of angle of attack.

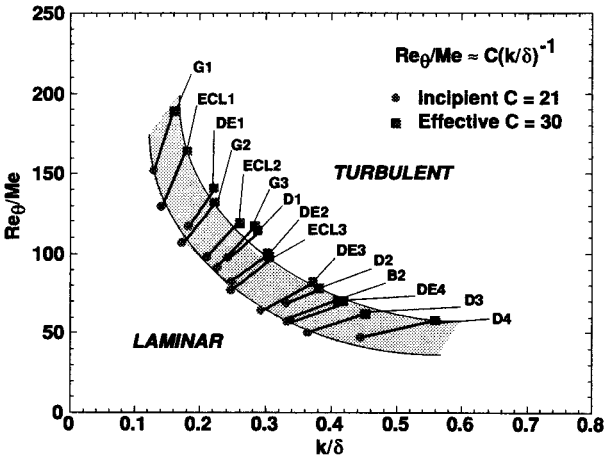
Fig. 22 X-33 forebody boundary layer transition at Mach 6.



(b) Effect of Reynolds number.
Fig. 22 Concluded.



(b) Altitude - angle of attack - velocity.
Fig. 23 Concluded.



(a) Roughness transition correlation.
Fig. 23 X-33 boundary layer transition criteria.

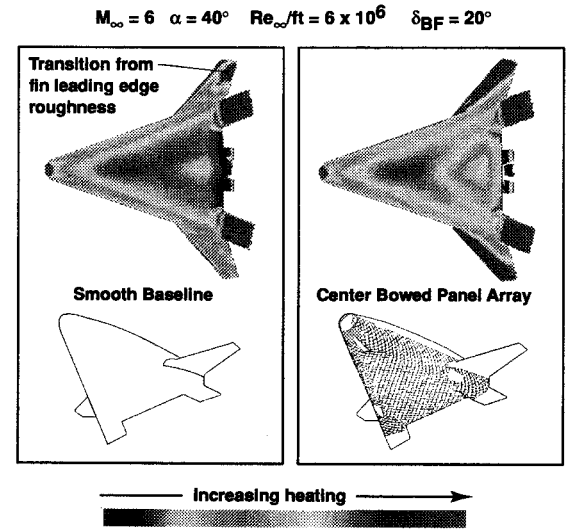
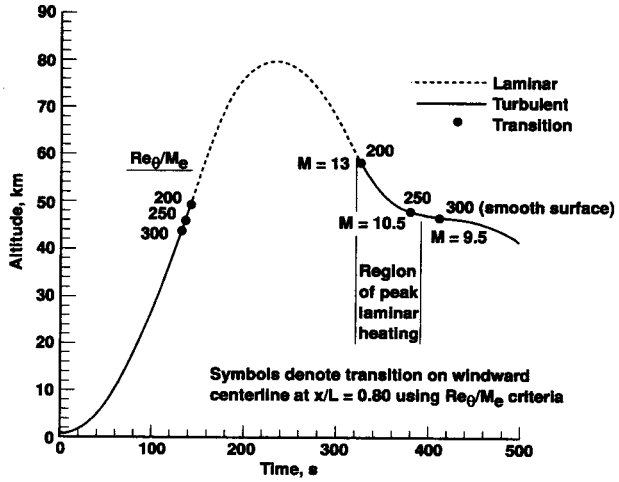


Fig. 24 Effect of X-33 panel bowing on aeroheating.



(b) Altitude vs. time
Fig. 23 Continued.

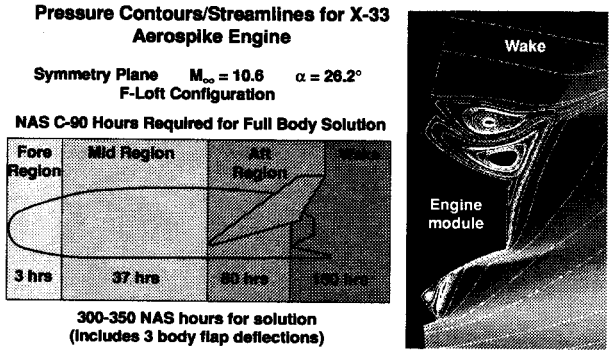


Fig. 25 Computer requirements for X-33 tip-to-tail CFD solution.

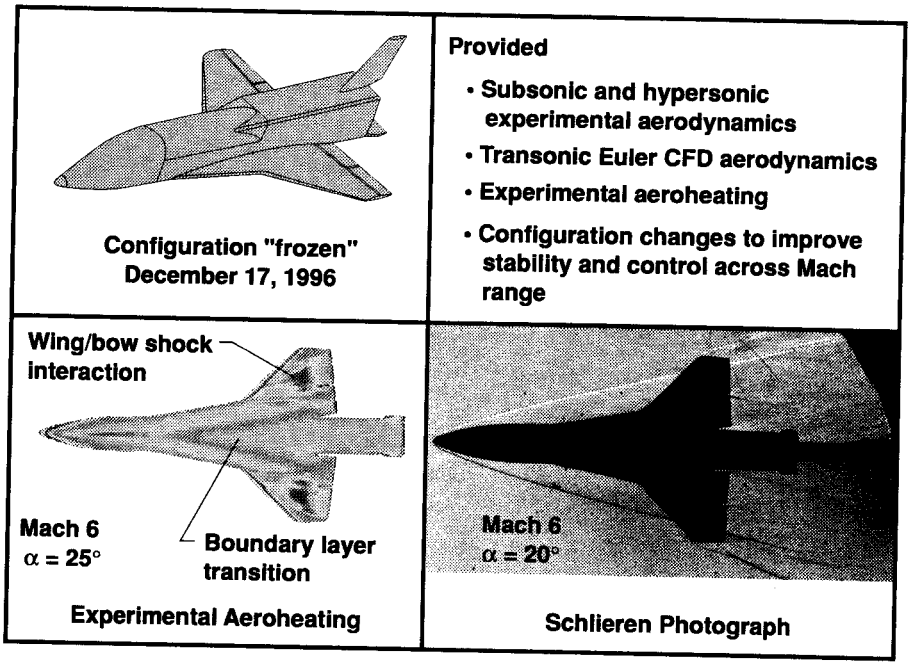


Fig. 26 X-34 configuration development.

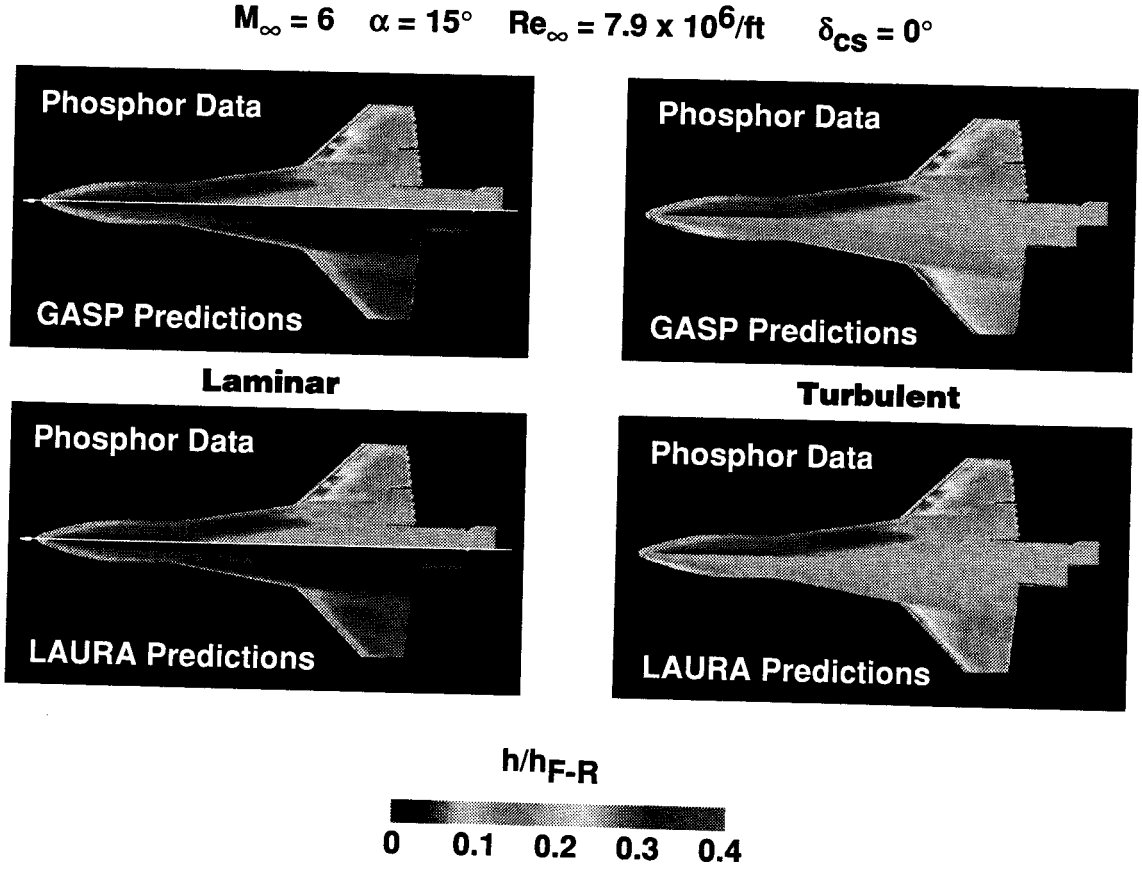
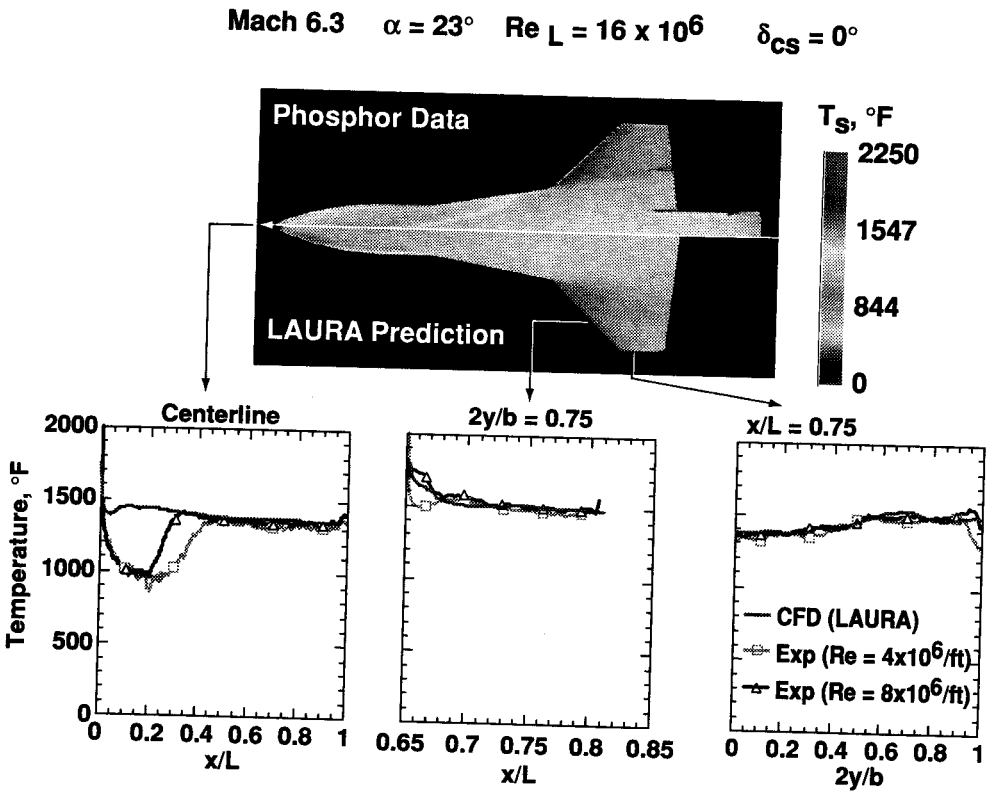
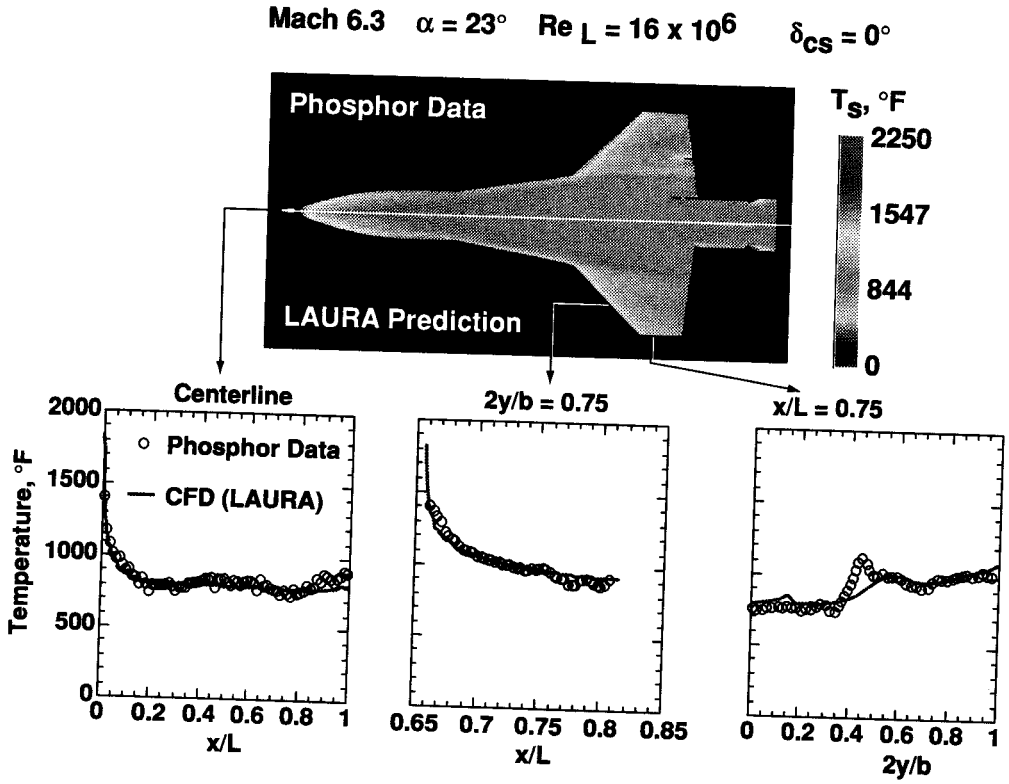


Fig. 27 Comparison of predicted to measured X-34 aeroheating.



(a) Laminar boundary layer.

Fig. 28 Extrapolation of X-34 wind tunnel aeroheating measurements to flight.



(b) Turbulent boundary layer.

Fig. 28 Concluded.

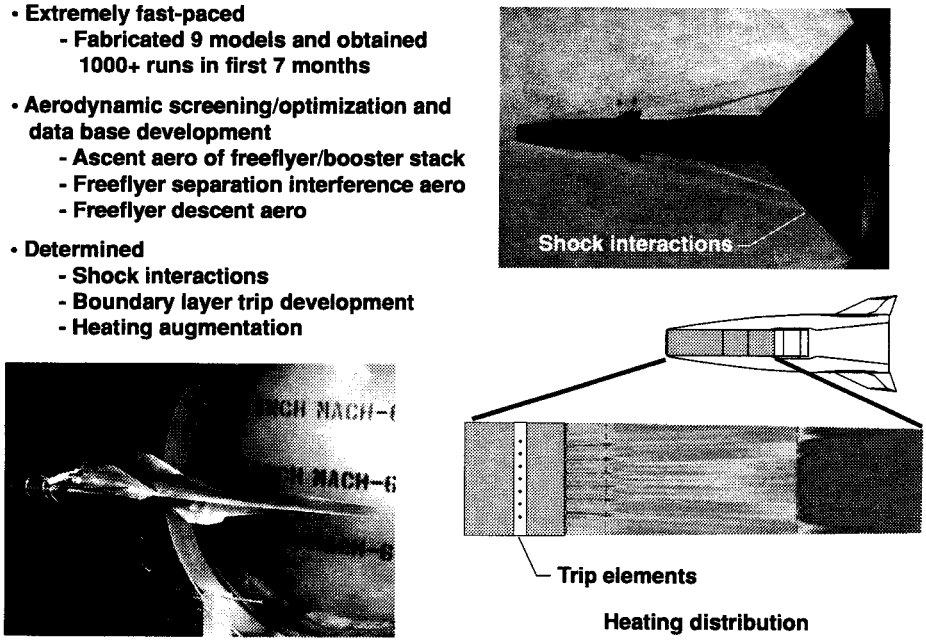


Fig. 29 Hyper-X aerothermodynamic assessment.

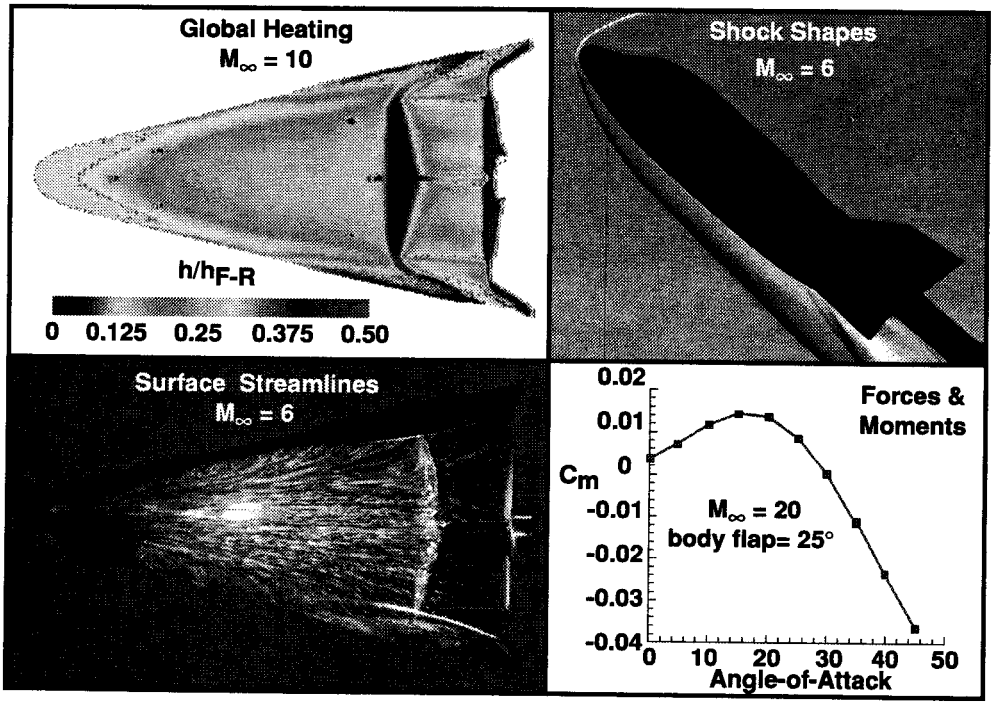


Fig. 30 X-38 aerothermodynamic assessment.

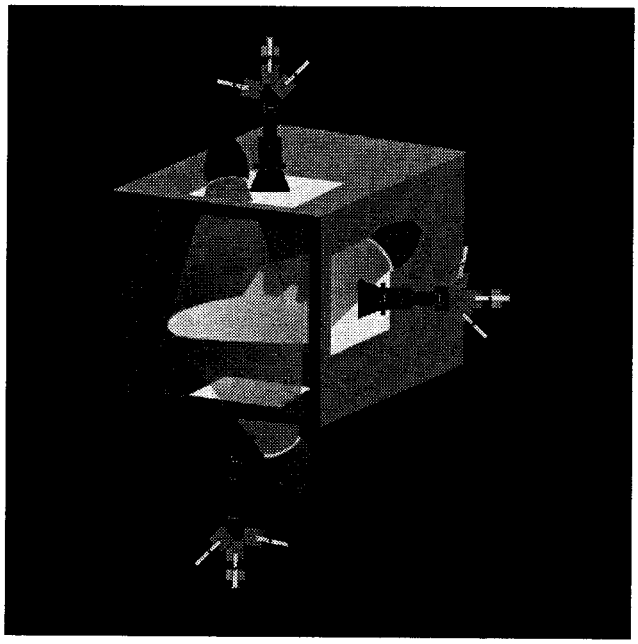


Fig. 31 Multiview thermography capability.

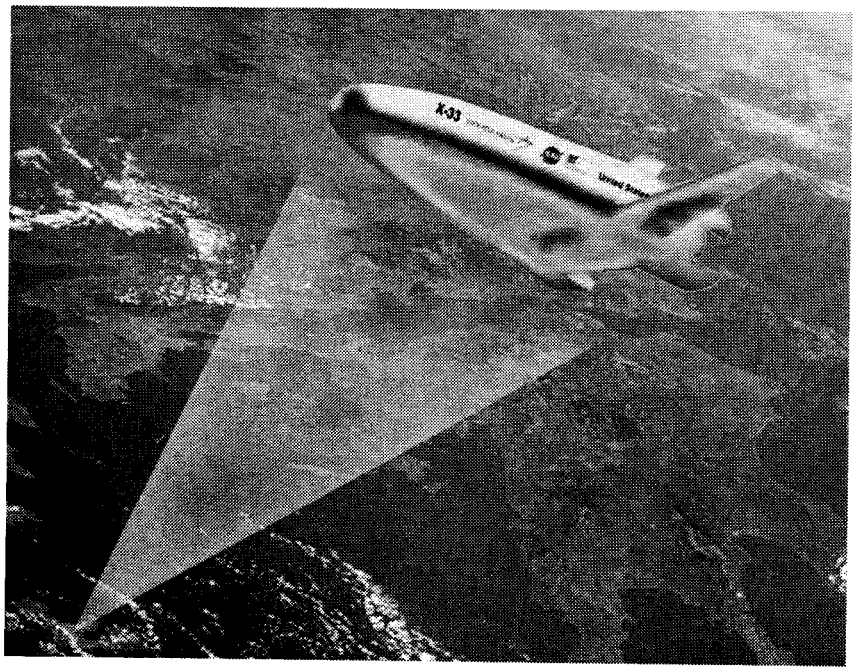


Fig. 32 Flight data extraction and comparison to ground-based data and CFD predictions.