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AIR VEHICLES INTEGRATION AND TECHNOLOGY RESEARCH (AVIATR)
Task Order 0015: Predictive Capability for Hypersonic Structural Response and Life Prediction, Phase 1 - Identification of Knowledge Gaps

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Final Report

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REPORT DOCUMENTATION PAGE

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14. ABSTRACT

Structural design of hypersonic vehicles requires additional considerations and effort, relative to conventional subsonic and low-supersonic aircraft, because of the wide Mach number range and the associated heating effects at high Mach numbers. At high temperatures, conventional structural materials, such as metals, suffer from reduced strength, reduced stiffness, increased creep, increased oxidation, increased thermal stresses, and other detrimental effects that impact structural design; many of these effects are of limited concern for aircraft with more conventional Mach number range and so hypersonic vehicles require a different approach to structural design. Hypersonic vehicle project analysts can no longer use independent methodologies coupled in a linear superposition approach for analysis and inputs to the vehicle design. This necessitates the use of coupled aerothermal and structural design methodologies for detailed hypersonic vehicle design, eventual reusable flight capability, and service life predictability.

15. SUBJECT TERMS

hypersonic performance, predictive capability, hypersonic service life, hypersonic simulation, coupled analyses, hypersonic environment

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1.0 EXECUTIVE SUMMARY

This document describes the results of the Phase I Hypersonic Predictive Capability – Identification of Knowledge Gaps effort as administered by the Air Force Research Lab (AFRL) as part of their efforts with the Structural Sciences Center (SSC).

The AFRL is interested in developing a technology base for the future design of reusable long duration cruise hypersonic aircraft. Such aircraft will likely possess gross takeoff weight magnitudes over 300,000 pounds, fly unfueled for over 2000 nautical miles, and cruise at speeds between Mach 5.0-7.0. These speeds will subject skin surface structures to temperatures over 1000F for the majority of the anticipated multi-hundred to thousands of hours of service life of the aircraft. To meet these requirements and be viable, the vehicle structure must be able to have accurate service life prediction capability methods in place in ensuring mission reliability, maintainability, and viability, along with an overall guide to reduced structural mass fraction. These issues must be fully addressed before a M5.0-7 Hypersonic Cruise Vehicle (HCV) becomes a reality.

Lockheed Martin Aeronautics (LM Aero) has actively pursued the development of new technologies that promote superior high speed aircraft design. One of the more visible examples is the SR-71, the first and only Mach 3.0+ aircraft in the United States Air Force. This aircraft required the development of entirely new structural concepts and material systems to meet performance and service life requirements. Within the past five years LM Aero has studied air breathing vehicle technology for the M5.0-7.0 hypersonic flight regime within the DARPA Falcon and HTV-3X programs. Today, new standards and criteria are being established in the areas of service life predictability to meet the flight performance requirements of future AFRL programs.

The overall objective of this Phase I program is to identify gaps in structural analysis and life prediction methods as applied to reusable, integrated structures for sustained operations in a hypersonic environment. The objectives of this report are to:
1. Report on the LM Aero team findings of Phase I
2. Provide an overview of the LM hypersonic vehicle design and methodology

2.0 INTRODUCTION
2.1 HYPERSONIC VEHICLE STRUCTURAL ANALYSES OVERVIEW

Structural design of hypersonic vehicles requires additional considerations and effort, relative to conventional subsonic and low-supersonic aircraft, because of the wide Mach number range and the associated heating effects at high Mach numbers. At high temperatures, conventional structural materials, such as metals, suffer from reduced strength, reduced stiffness, increased creep, increased oxidation, increased thermal stresses, and other detrimental effects that impact structural design; many of these effects are of limited concern for aircraft with more conventional Mach number range and so hypersonic vehicles require a considerably different approach to structural design.

Improvements in airframe structural efficiency through the utilization of unique and minimum weight concepts is a primary applied-engineering objective for the hypersonic environment. The essential elements for hypersonic structures criteria are temperature environment, low load

intensities, room temperature load condition capability after sustained high heat flux, structural stability, minimum gage material requirements, material manufacturability, reusability, non-optimum factors, and vehicle configuration compatibility.

Aerothermal loads exerted on the external surfaces of the flight vehicle primarily consist of pressure, skin friction (shear stress), and aerodynamic heating (heat flux). Pressure and skin friction have critical roles in aerodynamic lift and drag, however, aerodynamic heating is the predominant structural load. Aerodynamic heating is extremely important as induced elevated temperatures can affect the structural behavior in several detrimental ways. First of all, elevated temperatures degrade a material's ability to withstand loads due to the fact that elastic properties such as Young's modulus are significantly reduced. Moreover, allowable stresses are reduced and time-dependent material behavior such as creep come into play. In addition, thermal stresses are introduced because of restrained local or global thermal expansions or contractions. Such stresses increase deformation, change buckling loads, and alter flutter behavior.

Strength sizing of hypersonic vehicles requires the inclusion of temperature as that additional critical variable. At high Mach numbers, aerodynamic and propulsion limitations restrict the maneuverability of hypersonic aircraft and increasing Mach number radically reduces the aerodynamic efficiency of wings. In turn, the external loads on the airframe are reduced but because of the higher temperatures, which also reduces the strength of conventional structural materials, it is unclear as to which of these loading conditions actually size the structure. A further complication is the fact that for many materials the yield and ultimate strength vary differently with temperature, so both limit and ultimate loading conditions, which correspond respectively to yield and ultimate material strengths, require consideration.

Figure 1 illustrates limit loads with material yield strength shown on the left and ultimate loads, with material ultimate strength shown on the right as a function of Mach number. Note that this requires knowledge of the structural temperature at the point under consideration as a function of Mach number. In these cases, there are two potentially critical Mach numbers: one at the transonic phase of flight and another within the hypersonic Mach regime. For subsonic Mach numbers and for those in-between transonic and hypersonic Mach numbers, there is a noticeable strength margin. Modifications to Figure 1 would allow including thermal stresses in a straightforward manner by converting the bending moments to stresses (i.e. $\sigma_{MAX} = Mc/I$). The thermal stresses could then be added directly.
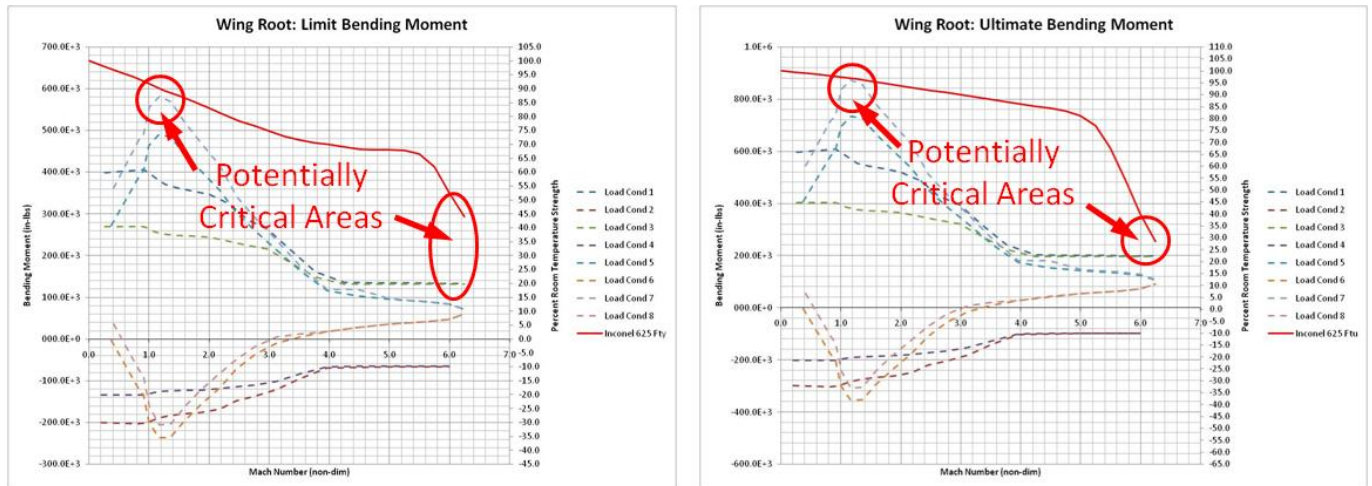
Figure 1. Limit and Ultimate Loads With Superimposed Material Limits

In addition, the most important heat transfer problem in flight at high supersonic and hypersonic speeds is that of determining skin temperature magnitudes since the skin temperatures will normally be the boundary condition for internal heat transfer problems. Not only is the skin temperature influenced by such external conditions as radiation from other surfaces and solar heat load but by such internal factors as conduction through structures, convection to fuel and gases, radiation to fuel and structure, and transient effects.

As the vehicle design Mach number increases to the hypersonic range, that above Mach 5.0, new physical phenomena become progressively of greater importance to the hypersonic airframe structural analysts, making hypersonic flow much more complex than supersonic flow. These phenomena include: 1) fluid dynamics effects that limit the validity of boundary-layer approximations, and 2) high-temperature effects that introduce chemical reactions with the structure.

The prediction capability of transition from laminar to turbulent boundary layer flow is highly critical to the design of hypersonic vehicle airframes. There are, however, severe limitations in analytical and experimental methods that are used to make these predictions. Overall improvement in the ability to predict boundary layer transition for the numerous factors that cause the transition affects is highly needed.

With all of the above as mentioned, hypersonic vehicle project analysts can no longer use independent methodologies coupled in a linear superposition approach for analysis and inputs to the vehicle design. This necessitates the use of coupled aerothermal and structural design methodologies for detailed hypersonic vehicle design and eventual reusable flight capability.

3.0 PAST EXPERIENCE ASSESSMENT (LM HIGH SPEED VEHICLE PROGRAMS)
Upon the research into the identification of knowledge gaps in hypersonic predictive capability LM Aero investigated previous key programs that contributed to the overall knowledge base. Four programs, the Mach 3.0+ SR-71/YF-12, the Mach 25.0 National Aerospace Plane (NASP), the Mach 15.0 X-33, and the Mach 6.0 HTV-3X are highlight programs that serve as "points of

departure" for the future standards that need to be put in place for truly reliable, reusable hypersonic flight predictive capability. While not illustrating a vehicle that falls into the "classical" definition of hypersonic platforms, the SR-71/YF-12 air vehicle program still serves as a precedent for the highest sustained speed reusable air vehicle and therefore provided a wealth of lessons and insight that can be adapted for future thrusts in predictive capability.

3.1 SR-71/YF-12

Even though the SR-71/YF-12 platform does not meet the classical description of a hypersonic vehicle, that of being capable of M5.0+ flight, it still represents the fastest sustained flight vehicle ever to go into service flight operations. With this premise, numerous high speed structural lessons and experience were a result of the program. These lessons detail the very origins of numerous gaps in analysis capability that are still prevalent for hypersonic vehicle studies of today and for the future.

3.1.1 SR-71/YF-12 Airframe

The SR-71 thermal environment (Figure 2) was one of the most severe, if not the most severe, in regards to operational aircraft flying today and in the past. The heat generated by the Mach 3.0+ environment resulted in skin friction heating and in stagnation temperatures up to 1200°F, depending upon high altitude ambient temperatures. Aerodynamic heating led to leading edge temperatures of up to 800°F, external surface temperatures on the fuselage and wings of up to 640°F, and the acreage surface and structural temperatures in the engine nacelle area of over 1100°F. In addition, all of the materials had to sustain strength and structural life capability during steady-state operation at the temperatures expected in the various areas of the vehicle. Finding solutions to the design challenges was a thermodynamics department dream or
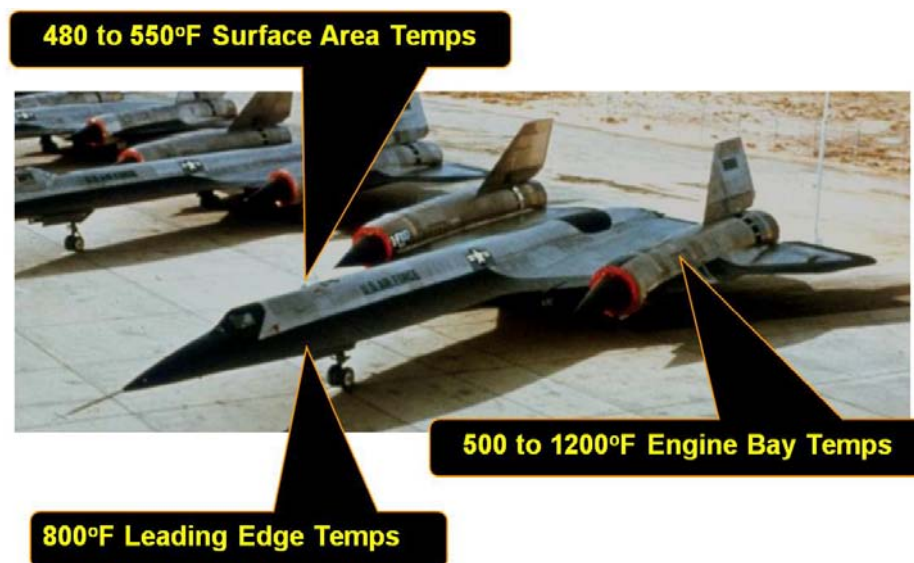
Figure 2. SR-71/YF-12 Up to Mach 3.0+ Thermal Soak Environment

nightmare, depending upon how an engineer looked at the concern to be addressed. According to SR-71 structural analysts, the environment in each area of the vehicle was relatively easy to determine, but designing methods to allow structure and equipment to live within those zones was extremely difficult. Yet, in the case of the SR-71/YF-12 design phase, the challenges were met and

successfully overcome through analysis methods considered even somewhat rudimentary by today's standards.

Over 93% of the aircraft's structure and skin surface was constructed of Titanium alloy. Other high heat resistant alloys such as Hastelloy X and Rene 41 were employed in the design of the ejector flaps and ejector rings, respectively. The SR-71 was essentially a flying fuel tank that carried over 40 tons of fuel principally in the five fuselage tanks and the wing tanks. The wing tanks were used for the climb phase of flight and were depleted first due to their high ratio of surface to volume for thermodynamic reasons. The wing of the SR-71 consisted of pre-formed longitudinal corrugations to ensure the expansion of the wing in the streamwise direction. Most notably, the fuselage of the SR-71 increased in length by about 10 inches per cruise portion of flight and the cruise phase temperature delta between the windward and leeward surfaces of the fuselage caused a significant structural droop of the nose section of the aircraft. Yet, the structure was most critically loaded in the relatively low temperature transonic phase of its trajectory which enabled room temperature testing on many structural components to be adequate for its design. In some cases a "room temperature overload" artificial factor of safety was utilized to simulate hot conditions while other components were tested at elevated temperatures.

3.1.2 SR-71/YF-12 Thermal Environment
In general, the primary areas of concern in the external thermal environment were divided into the following rather broad airframe groups: 1. Fuselage, 2. Wing, 3. Nacelle, 4. Control Surfaces. Further subdivision of the structural thermal environments for detailed analyses were then established: 1. Beams, 2. Longerons, 3. Rings, 4. Mounts, 5. Brackets, 6. Skins, and the 7. Nacelle structural elements. To aid in the determination of the thermal environment baseline assumptions were made as governing criteria early on:
- a 25°F increase for each 0.1 increase in the Mach number
- a 16°F increase for each 10°F increase in ambient temperature (at constant altitude and speed)
- a 15°F increase for each 5000 foot decrease in altitude (at constant speed and ambient temperature)
- a 17°F increase for each 0.1 decrease in external surface emmisivity

Through use of these guideline assumptions flight test data showed that actual surface temperatures were within +/-20°F of the predicted skin surface values except in the following areas:
- where ram air leaked into an area through gaps or from a nacelle
- where radiation from a nacelle surface became an influence
- where shocks from the nacelle crossed the surface with a larger than expected influence on local heat transfer coefficients and local adiabatic wall temperatures
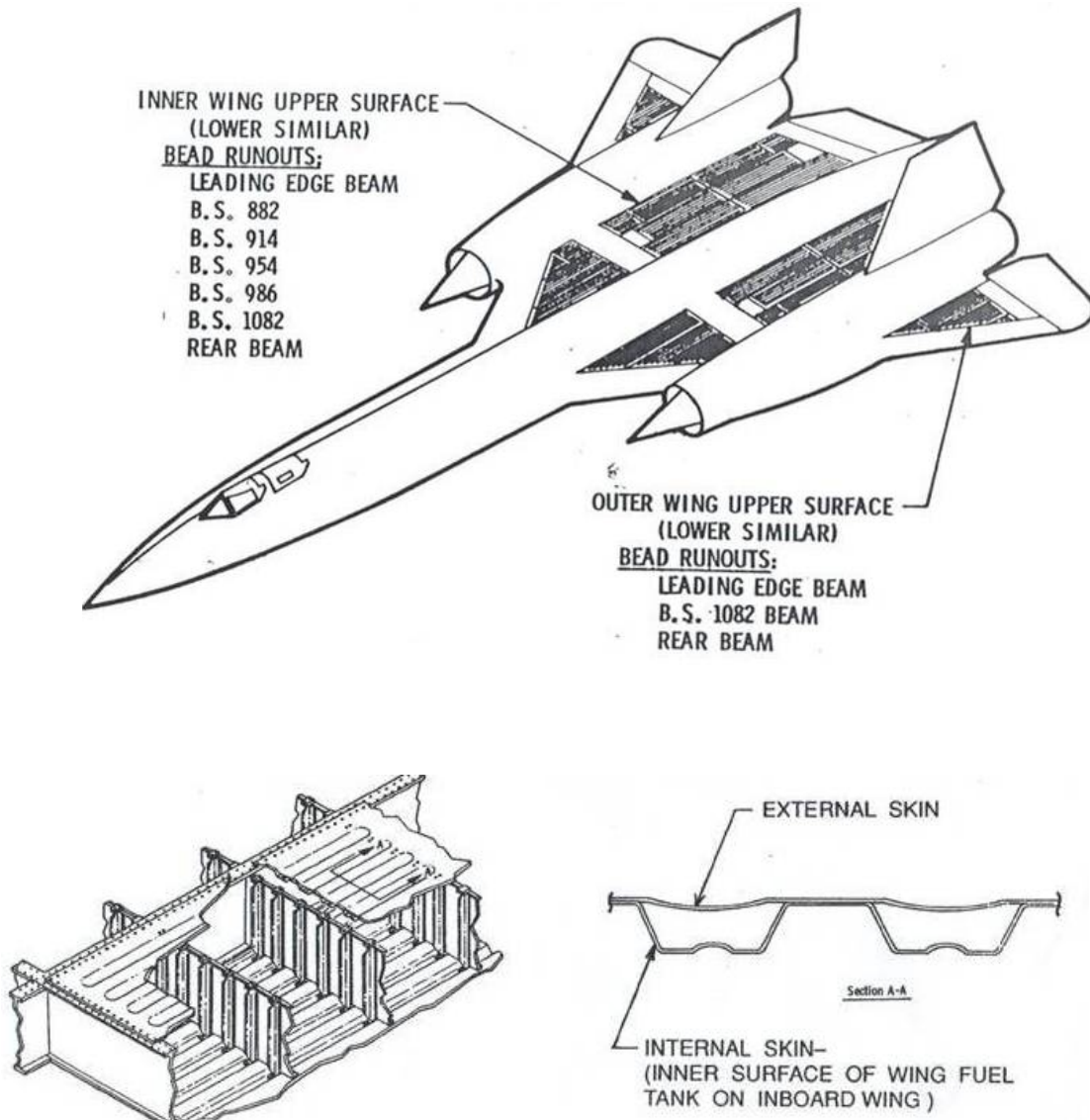
Figure 3. SR-71 "Double Panel" Wing Skin Surfaces

Along these lines of external thermal environment prediction was the special concern that arose in the design of the wing surfaces. The problem was how to address the distortion of the wing surface panels due to differences in thermal expansion of fast heating skin panels and slower heating internal support structure. The solution was to build double panels with fore-to-aft corrugations. Therefore, distortions would be in a known direction and would not adversely affect the airflow pattern over the wings so the loads would be distributed through known paths. These "double panels" (Figure 3) were designed for this lateral growth in the outboard and inboard wing areas.

The SR-71 structural elements were designed to sustain loads during steady-state and transient application of external heating. Coupled structure/thermal "lag" effects with respect to time depended upon heat paths and heating modes which comprised the effects of convection, conduction, and insulation in some areas and the effects of direct ram air convection and engine radiation in others. In addition, the wings and fuselage structure heated slowly while engine

components such as the inlet system, nacelle, and exhaust system heated quickly. This phenomena is one that is still inherent on air-breathing vehicle studies of today.

Overall, with environments being established for all areas of the vehicle, tests being run on most systems and structures at temperature, and flight tests being undertaken to verify environments and equipment operation, the vehicle was very successful at operating in the severe thermal environment that it was designed for.

Flight test data indicated what increments had to be added to the computed values in order to obtain temperatures in other flight regimes at the various locations where leakage and shock interference occurred. Examples of these areas are along the inboard edge of the nacelle, in the vertical stub fin area, and at the inboard wing connection to the nacelle.

In reviewing numerous Lockheed Aircraft Corporation A-12 stress reports it is a true testimony to the "well done" of the engineers and most notably the analysts of the time that the airframe had survived the service requirements that it did. Today's computer capability did not exist in the late 1950s/early 1960s when the SR-71 was designed and first built. This in itself dictated the design and analysis engineers to reduce problems to their lowest common denominator before doing the calculations. What follows is a listing of analysis assumptions that were made for the vehicle' design maturity well before the advent of structural sizing analytical tools for the high speed environment and most notably, without the coupling of environmental effects into the inherent complex loading schemes prevalent to such high speed vehicles:

- All analyses were performed using the estimates on steady state temperature predictions with no internal heat transfer effects included with the exception of the engine nacelle area. It was stated that conservatism had to be well employed to mitigate the foreseen risks and many structural components were most likely "oversized" as a result.

- The skin was always considered as "perfectly insulated" from internal effects. Hence, there was no inclusion of internal heat transfer such as fuel or engine bay heating and cooling effects.

- Radiation effects to/from aircraft external surfaces were not included in structure sizing operations.

- Radiation Equilibrium Temperature (RET) data was used at only 59 points over the surface of the aircraft for all detailed structural analyses.

- The design team looked at control surface deflections as functions of Mach number, dynamic pressure, vehicle center of gravity, and vehicle acceleration at just a few discrete flight points: M0.68, M0.95, M1.25, and M3.2 for the envelope. The relatively big "gap" between M1.25 and M3.2 was noteworthy. The aero inputs were derived from wind tunnel forces and the pressure distribution measurements.

- The stiffness magnitude of the A-12/SR-71 characteristic wing root location chordwise corrugation stiffeners was not modeled and therefore represented a rather large uncertainty

in the prediction of the wing root's thermal growth. To complete all analyses of this a great amount of test results were needed before proceeding onto the final sizing stage. The data that was highly needed were the wing's root's tension and compression resistance to warping.

- The external skin emissivity was assumed to remain at 0.9 throughout the vehicle's service life with the radiation heat sink temperature constant at 60°F. Turbulent boundary layer conditions were always assumed as is still customary today.

- On the A-12 outer wing stress analysis, all the redistribution of the wing torsional shear flow into the vertical load on the nacelle frames was carried out at room temperature conditions. No thermal effects upon the Titanium alloy of the design were included due to test data not being available.

- In determining the wing surface loads and stress levels due to chordwise bending no consideration was granted upon thermal deflections. The surface and cap areas were lumped together to arrive at an average surface stress level.

- On the sizing of the joints uniform temperatures were assumed due to detailed thermal analysis data and heat transfer data not being available. On the W.S.213 joint sizing, for instance, the entire joint was assumed to be at a uniform 500°F without any transient effects included.

- Joint efficiency factors used for structural joint sizing were not adjusted per service temperatures.

- The A-12 chine airloads analysis was based on a "rather rough" approximation of the airload distribution. It constituted a uniform load applied to cover all positive load factor conditions with no thermal loading included.

3.2 NATIONAL AEROSPACE PLANE (NASP)
3.2.1 NASP Program Overview: Predictive Capability Was a Part of the Plan
The National Aerospace Plane (NASP) program completed two phases of a planned three phase RDT&E effort to develop and fly two X-30 manned reusable air-breathing SSTO vehicles. Competing contractors developed vehicle and system concepts and identified critical technologies in Phase I. Technology maturation, risk reduction plans, and initial technology development were also accomplished in Phase I. During that phase the experimental vehicle requirements included limited operational utility with a payload bay sized for a significant weight and volume. At the end of Phase II the requirements were reduced to envelope expansion and orbital capability demonstration only. At that point the X-30 vehicles could be retired and focus would shift to operational system development.

A key initial NASP program assumption was that revolutionary improvements in computational analysis would reduce the time and cost required to produce two man-rated SSTO vehicles. The planned total program cost of $3.3 Billion (1988 dollars) was significantly lower than previous manned spacecraft and launch vehicle programs. Program management made it clear that an

aggressive stepping stone approach to validate analysis tools with increasingly complex ground tests was required to achieve success within the planned cost. Ground test facilities could not (and generally still cannot) duplicate some critical flight environments beyond ~Mach 10.0 for large propulsion, system, or structural tests. The NASP plan was to validate tools to the maximum extent possible with testing. Those tools would then allow the vehicles to be certified for high Mach flight envelope expansion and eventual orbital insertion and re-entry based on analysis. In addition, the development of the tools held the promise of integrated vehicle synthesis and design optimization. Hypersonic predictive capability was the key to achieving acceptable risk, and therefore, cost.

The airframe contractors competing in Phase I quickly identified critical structural technologies:
- High temperature passive leading edges
- High temperature actively cooled leading edges (heat pipe and/or fuel cooled)
- High temperature acreage Thermal Protection System (TPS) materials, design, & integration
- Cryogenic fuel tank materials, structures, & integration
- Cryogenic fuel management systems
- High temperature airframe/propulsion integrated structure
  - Passive acreage including external inlet & engine external TPS
  - Fuel cooled acreage including internal inlet & nozzle
  - Propulsion companies addressed internal engine structure & systems

Critical technology risk reduction articles were built in Phase I. These included fuselage tank & TPS systems. Some of those tests stretched into Phase II.

Government evaluators continually asked contractors about their vehicle synthesis tools in Phase I, pushing for development of the elusive integrated hypersonic design/analysis/optimization tool. The contractor teams attempted to assemble that tool, but the complexity of the NASP problem and the aggressive schedule resulted in long "man-in-the-loop" design and analysis cycles. Several highly respected team members developed synthesis and optimization tools. The complexity of the problem and the difficulty of validating results limited their use to prediction and interpretation of trade studies that were completed with detailed analysis. This was an early disappointment in hypersonic computational tool capability on this program.

The NASP Phase II goal was to mature a preliminary vehicle/propulsion design and the key technologies to enable vehicle fabrication and flight test in Phase III. Due to the varied strengths of the contractors a National Team was formed instead of down-selecting. The ambitious airbreathing SSTO goal made nearly every technology critical. Top-level categories included:
- Airbreathing propulsion Mach 0-16.0
- Hypersonic lifting body aerodynamics & handling
- Hi-temperature & cryogenic materials, structures, & integration
- Cryogenic fuel systems with hi temp active cooling
- Avionics

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The detailed list of critical structural technologies was essentially unchanged from the Phase I list already discussed in this section.

The Phase II configuration, systems, and propulsion design trades converged on a National Team vehicle. Critical technology maturation tasks were focused on selected approaches. Risk reduction plans were developed and executed. The USAF, DARPA, and NASA monitored the risk reduction plans, progress, and risk outlook. The Defense Science Board, Air Force Studies Board, & RAND monitored and assessed NASP knowledge deficiencies during Phase II. These reviews identified predictive uncertainties at the maximum ground test Mach numbers, the "lower hypersonic range" of 5 to 7%, with increasing uncertainty at higher speeds.

Although X-plane programs often handle such uncertainty by adding significant margin to the vehicle, the SSTO requirement precluded that risk mitigation approach because weight fraction was so critical. In order to achieve the SSTO goal and to meet manned spacecraft safety requirements it was clear that Phase III would require more extensive testing than the original program allocated. Hypersonic predictive capability was not sufficiently accurate to meet program safety, vehicle performance, and cost goals.

3.2.2 Effect of New Materials and Structural Technologies on Hypersonic Predictive Requirements

The challenge posed by the NASP mission required new material systems for both high temperature and cryogenic environments. Significant areas, especially in and around the propulsion system, experienced both high temperature and cryogenic environments during a flight. Hypersonic predictive capability was limited by the lack of material data. The NASP Program Office established a Materials and Structures Augmentation Program (MASAP) funded at over $100 Million to develop the materials and databases required.

An example of the early research that was needed for hydrogen containment is seen in Figure 4, which shows micrographs of toughened BMI and Graphite/Epoxy (Gr/Ep) that was cycled at cryogenic temperatures. The BMI micro-cracked faster than the epoxy matrix. By the end of the program the team recognized that the tanks would leak Hydrogen (H2). A leak rate specification was set at a value that allowed reasonable quantities of purge gas to avoid flammable concentrations inside the vehicle structure.

LH₂ CONTAINMENT ACHIEVABLE WITH GRAPHITE/EPOXY TANK (u)

TOUGHENED GRAPHITE/BMI
(T300/5250-2, [±45] 2S)
UTS = 31.7 ksi

BASELINE GRAPHITE/EPOXY
(IM7/8551-7, [±45] 2S)
UTS = 41.9 ksi

• EXCESSIVE BMI MICROCRACKING (at −423°F) PREVENTS CONTAINMENT

• GRAPHITE/EPOXY SHOWS INCREASED SHEAR STRENGTH CAPABILITY OVER BMI
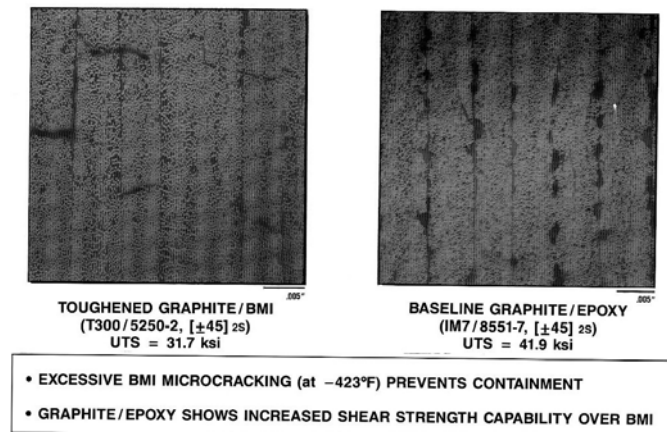
Figure 4. Material Research Was Required Before Structures Could Be Analyzed Accurately

The ability to model the environment and resulting structural loads at hypersonic conditions progressed quickly. Available test data and modeling approaches were researched and applied. The ability of AFRL facilities to test at realistic combined load conditions was vital to progress in this area. Figure 5 illustrates early action by one of the NASP contractors to develop aero-acoustic load prediction capability. Similar progress was made in predicting propulsion related acoustic & pressure loads and vibration loads inside the vehicle.

BOUNDARY LAYER NOISE PREDICTION (u)

ATTACHED TURBULENT FLOW

B.L. EDGE

• NORMALIZED PRESSURE PREDICTION: (p-rms/q) = 0.006/F where
q=local edge dynamic pressure, and F=0.5+(h-wall/h-adiabatic)(0.5 (1+r(Gamma-1)M**2)+0.11r(Gamma-1)M**2

• NORMALIZED POWER SPECTRAL DENSITY FUNCTION:

$$\frac{PSD(w)(v)}{(q**2)(l)} = \frac{2.293E-5(F**-0.5733)}{(1+F**2.867(l/v)(w**2))}$$ where l=displacement thickness, and v=local edge velocity

• REFERENCE: A. LAGANELLI AND H. WOLFE
PREDICTION OF FLUCTUATING PRESSURE IN ATTACHED AND SEPARATED TURBULENT BOUNDARY LAYER FLOW
AIAA-89-1064

• INPUT: NOZZLE REGION: "SCNHOZ", a finite rate chemistry, 2-D parabolized Navier-Stokes CFD code
FOREBODY/REAR UPPER SURFACE: "GDPNS3D", a forward marching 3-D parabolized Navier-Stokes CFD code

SHOCK/BOUNDARY LAYER INTERACTION

• EXTREME ACOUSTIC ENVIROMENT

• RMS PRESSURE LEVEL CHARACTERIZED BY SPIKE AHEAD OF SEPARATION REGION WHICH DROPS RAPIDLY TO A PLATEAU.

• NORMALIZED PRESSURE (PLATEAU):

$$\frac{P\text{-}rms}{P\text{-}static} = \frac{0.006(Gamma/2)(M**2)}{1.+0.13(M**2)}$$

B.L. EDGE   SHOCK   SEPARATION

BOUNDARY LAYER PSD - UPPER FUSELAGE @ 18 FT.

PSD (PSI**2/Hz)

OVERALL SPL (dB):
MACH 6   139
MACH 10   135
MACH 14   136
MACH 19   136

□ MACH 6
◆ MACH 10
■ MACH 14
● MACH 19

FREQUENCY (Hz)

UNCLASSIFIED

Figure 5. Early NASP Contractor Predictive Capability for Hypersonic Environment

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The final Phase II baseline structural approach was a complex system that included a "warm" Titanium Matrix Composite (TMC) body structure in which cold Gr/Ep cryogenic tanks were suspended. Areas that were too hot for the TMC had a parasitic heat shield of Space Shuttle type blanket insulation. Where blanket TPS was insufficient, Carbon-Carbon (C-C) plates were fastened over Shuttle-type tile standoffs and quartz fiber insulation. Actively cooled structure was needed for the nose, the engine cowl leading edges, internal engine, and nozzle acreage. Figure 6 illustrates the vehicle structural material usage. In addition, Figure 7 shows the stack of structure and insulation that were specified for high temperature areas such as the lower forebody.

The cost, complexity, and weight of a 3-layer structure was an issue within the NASP team because a cold-structure approach using load-bearing tanks with parasitic TPS was considered to be a lower weight design when all details were accounted for. The cold structure approach was considered to be an uncomfortable risk for a "hot" hypersonic vehicle by many decision makers. The CAD model development time, changes in TPS, tank, and structure details, and time required for structural analysis of details such as TPS fasteners and tank structural interfaces precluded high fidelity weight calculations. The coupling of these issues required managers to make decisions based on their assessment of the relative performance, cost, and safety risks of warm or cold structures using partial models and analysis of critical features.
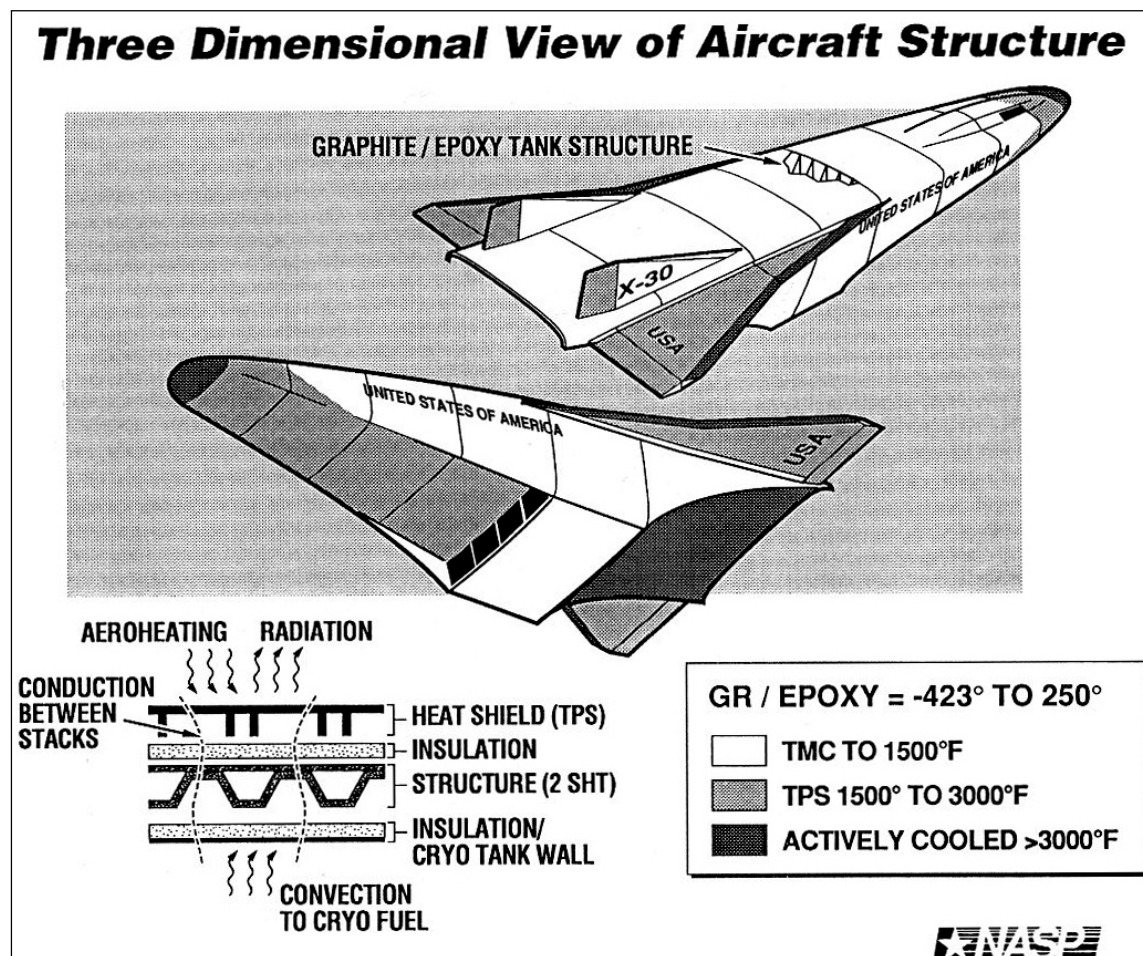
Figure 6. NASP Structural Material Usage

12
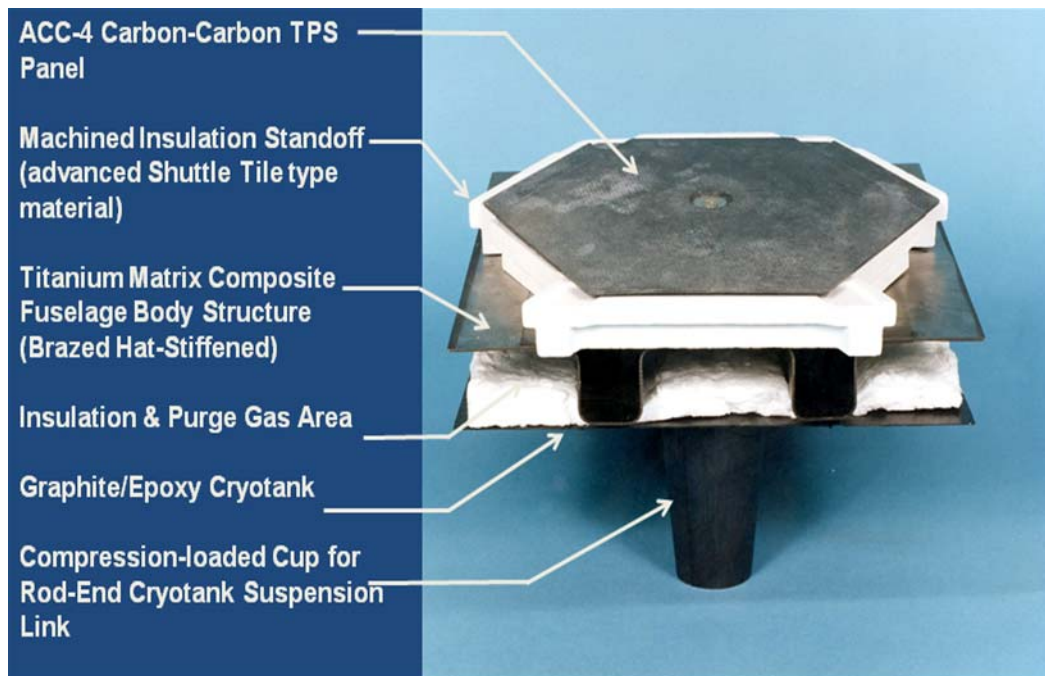
Figure 7. NASP Suspended Cryotanks in TMC Body With Parasitic TPS Where Necessary

Cryotank technology maturation tasks demonstrated materials, complex tooling approaches, innovative load transfer structure, and fuel system integration. Figure 8 shows several steps in the development of the conformal cryotank approach. This is the first of two tanks that were tested through fill and drain cycles with liquid H2. Tests uncovered complex issues such as peel loads in some of the bondlines. Repair techniques were developed and tested. The conformal tank approach and a multi-lobe approach were carefully matured in Phase II. The multi-lobe approach offered lighter weight but with an offsetting fuel volume penalty. The multi-lobe approach was selected. Example analysis results for that design are presented in the next section of this report.

High temperature structure technology maturation included all C-C structure as well as TMC structure. An example of progress in this area was the large C-C wing box that was tested at the AFRL. Manufacturing capability was matured, demonstrating that the structure could be built, coated, and still retain the fastener holes and features align for assembly. Processing was matured when the lower skin delaminated during densification cycles. Process adjustments and repair procedures were exercised. Figure 9 shows the scale and some details of this significant technology accomplishment. The diverse material and structural approaches resulted in an extremely complex, highly integrated design. This also increased the extremely critical need for predictive capability.
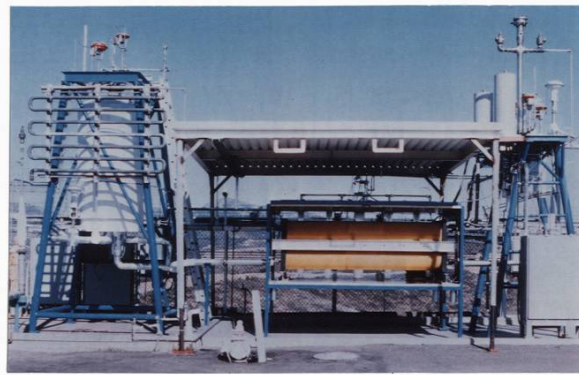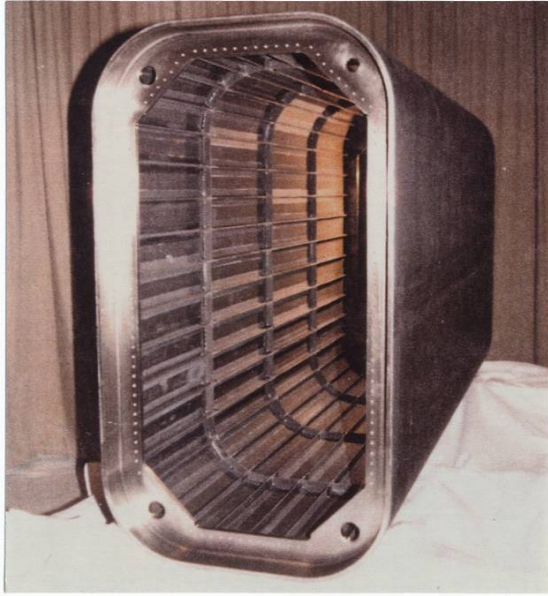
Figure 8. Conformal Cryotank Was Fabricated and Tested at Convair's Rye Canyon Facility
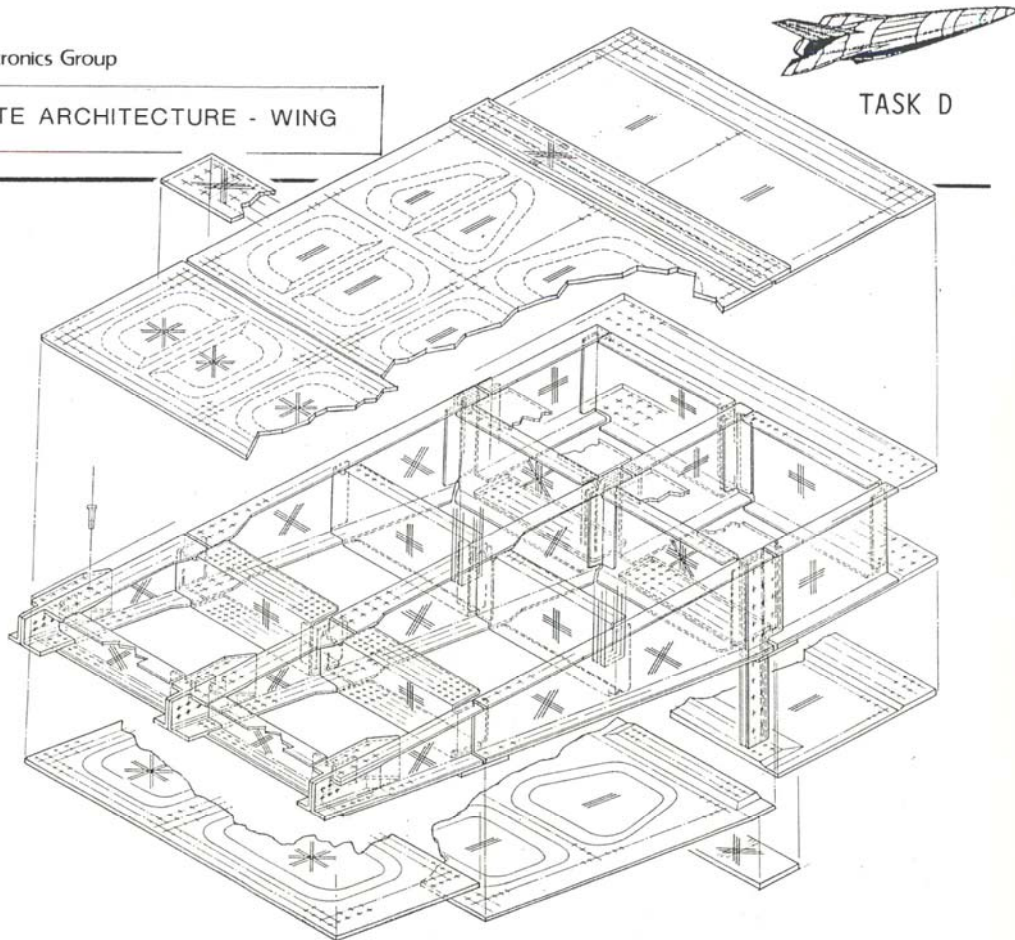
Figure 9. Carbon-Carbon Wing Box Tested at AFRL

1500F: Titanium Matrix Composite (TMC) "warm" primary structure with brazed hat stiffeners.

1500F to 3000F: Parasitic Carbon-Carbon or CMC TPS mechanically fastened over insu...

Over 3000F: Actively cooled Carbon-Carbon, CMC, or brazed metallic panels.

- 423F to +250F: GR/Epoxy tanks suspended by swiveling links from TMC body.

Parts

Carbon/Carbon Structure Manufacturing Flow

Titanium Metal Matrix Structure Manufacturing Flow

Tank Manufacturing Flow

WELD RETORT · VACUUM SEAL · HIP · REMOVE RETORT · ASSEMBLE PARTS · BRAZE · COAT · CLEANER · FINAL TRIM

AUTOCLAVE CURE

CLEAN-UP

ACTIVELY COOLED PANELS

FINAL BRAZE

CHEMICAL ETCH

All Movable Wing

Aft Fuselage

VERTICAL FINS

Center Fuselage

Forward Fuselage

To Final Assembly Site

Figure 10. NASP Program Combined Complex New Materials & Structural Technologies

3.2.3 NASP Structural Areas of Concern

At the end of Phase II there were areas of concern in all of the critical structures in the NASP vehicle. Cryotank concerns included micro-cracking during thermal cycling. Tank and fuel system details such as penetrations and load transfer points required very detailed modeling and testing to validate predictions. Fuselage exterior durability and performance under repeated aeroheating and acoustic loading cycles were a concern. Exterior surface smoothness requirements were critical concerns because of the danger of TPS failure during flight. Shock interactions and local high heating concerns were difficult to analyze and test. Internal volume required purge not only to control hydrogen content and flammability, but also to eliminate condensation and ice formation. Thermal cycling, oxidation, hydrogen embrittlement, creep, and thermal buckling were concerns that required test data. Areas of concern are summarized in Figure 11.
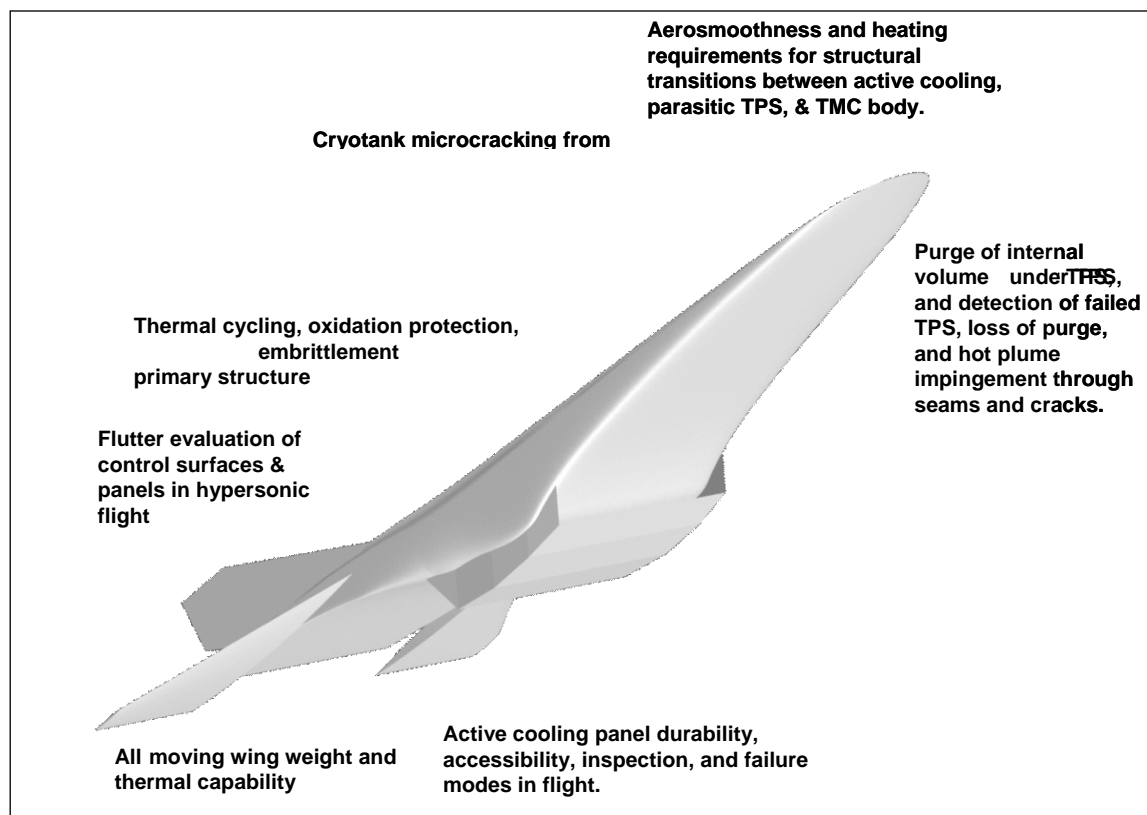
Figure 11. NASP Structural Areas of Concern

These concerns were addressed with the structural modeling and analysis tools available at the time. The primary analysis tool was MSC/NASTRAN, and the application top-level test approach is summarized in Figure 12. Thermal mapping was done by developing specialty codes. These tools could not predict the material behavior after repeated exposures, requiring significant investment in material characterization testing, followed by element and subcomponent tests in increasingly rigorous environments.

All NASP team members contributed to unprecedented computational capability improvements. Model fidelity increased rapidly, but the size and complexity of the NASP vehicle required careful budgeting of highly detailed analysis. This was accomplished for representative critical features at maximum load cases. An example of the budgeted analysis approach is shown in Figure 13, where a relatively coarse grid was developed for a thin-walled multi-lobe tank, and the critical bonded joint on top of the tank was analyzed in more detail at the critical load case. The results were impressive in the 1990 time frame, but are sparse by 2010 Standards.

Computational throughput was also a limiting factor. One engineer recalled an Aviation Week & Space Technology article that reported the NASP program using 1/3 of the nation's supercomputing capability in the late 1980's. Even at this pace, a complete vehicle design cycle needed 6 months of work, although the program schedule did not always allow a complete analysis cycle before a new configuration was drawn.

NASP TPS & Structural Analysis & Test Summary Table: Phase I, II, & III.
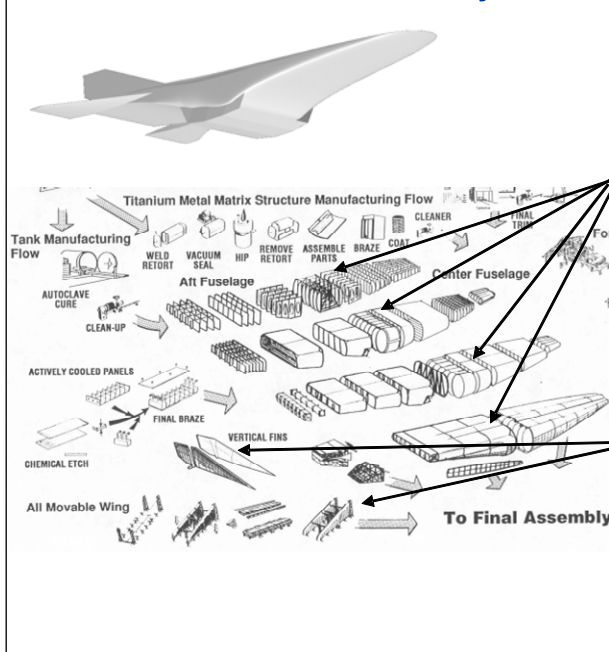
Vehicle Section	Phase I (Three Airframe Concepts)	Phase II National Team Single Design	Phase III Build & Fly X-Planes (Not Funded)
Fuselage & Tank Structure	NASTRAN + subscale (Task D Tank) static tests to calibrate predictions	NASTRAN + subcomponent tests	Extensive analysis & full scale integrated tests, incl. dynamic WT tests before design freeze. Integrated flight vehicle dynamic tests before flight approval
Flight Controls Structure, Thermo, & Aeroelastic	NASTRAN + full scale wing box & other static tests to calibrate predictions	NASTRAN + all-moving tail struct component static tests	
Vehicle Structure + Propulsion Dynamics	NASTRAN analysis	NASTRAN analysis	Extensive wind tunnel and integrated ground tests planned before flight approval

Figure 12. NASP Structural Analysis & Test Overview by Program Phases
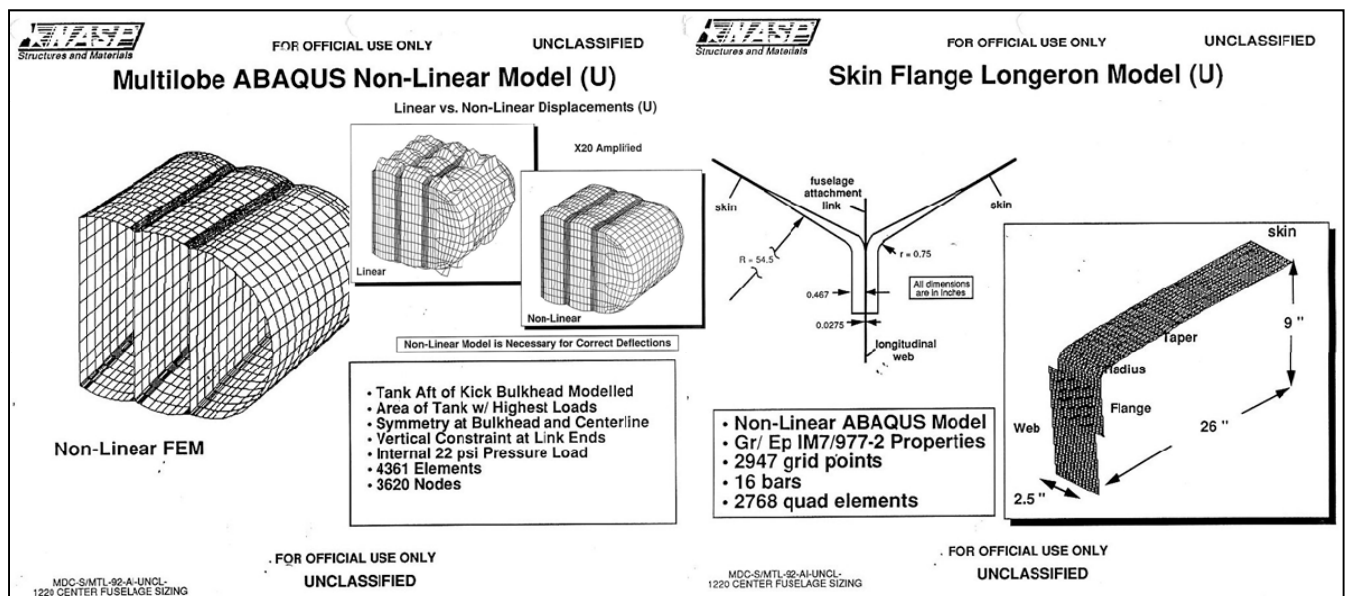
Figure 13. Nonlinear Modeling of Cryotanks Sized the Structure for the Highest Load Cases

Modal analysis as described in NASP National Program Office (NPO) design notice DN-92-0368 was accomplished using a dynamic NASTRAN Finite Element Model (FEM) that was developed

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directly from the strength FEM. According to the design notice, the mesh was coarsened to reduce the overall model size. The model included fuel mass changes throughout the trajectory as well as reduction in material stiffness due to thermal conditions. For the NASP configuration 201 analysis, primary structural stiffness had been updated after two iterations of optimization. The first iteration had used "best guess" structural sizing. Actuator stiffnesses were reduced to reflect a single stage failure. The skins on the rudder, all-moving horizontal tails, and body flap were "beefed up" based on preliminary flutter assessment. The all-moving tail/wing carry-through structure was modified to improve the loadpaths.

Figure 14 illustrates the first 15 symmetric vehicle bending modes at various points in the trajectory. First body bending modes were low, appearing below 4.0 Hertz. The body modes were closely packed and couple readily with both the tank and control surface modes. NASP team members recognized that these predictions were not sufficient to build a vehicle that would be cleared for manned flight to orbital speeds without significant testing to verify the tools and validate the results.

Configuration 201 Modal Analysis Summary(U)

	TAXI	M 0.9	M 1.2	M 3	M 4	M 6	M 8	M 10	M 13	M 16	M 17
BODY FLAP	1.98	1.99	1.98	1.98	1.98	1.98	1.98	1.98	1.98	1.98	1.98
WING PITCH/BODY BENDING	3.23	3.26	3.27	3.27	3.24	3.18	3.13	3.12	3.10	3.07	3.06
WING PITCH/BODY BENDING	3.66	3.80	3.84	3.92	3.89	3.83	3.79	3.77	3.81	3.88	3.92
WING YAW	4.79	4.79	4.79	4.75	4.71	4.59	4.48	4.45	4.36	4.26	4.23
WING BEND./BODY BEND.	6.62	6.83	7.00	7.27	7.38	7.50	7.62	7.76	7.97	8.17	8.30
WING BENDING	6.86	6.90	6.89	6.87	6.82	6.68	6.57	6.53	6.45	6.33	6.28
TAIL/RUDDER BEND	8.70	8.21	8.70	8.65	8.59	8.48	8.38	8.23	8.02	7.61	7.59
TANK/BODY BEND.	9.30	9.56	9.68	9.88	9.91	9.92	9.93	9.97	10.26	10.65	10.89
TANK/FWD FUSELAGE	10.29	10.46	10.59	10.90	11.04	11.20	11.30	11.41			
TANK/WING BEND.	11.28	11.58	11.73	11.89	11.94	11.95	12.02	12.21	13.01		
WING BEND	12.24	12.68	12.78	12.95	12.90	12.79	12.69	12.71		11.65	11.58
WING TORSION	14.10	14.22	14.24	14.27	14.18	13.93	13.72	13.64	13.52	13.27	13.19
WING BEND	14.22	14.32	14.38	14.50	14.55	14.61	14.67	14.70			
FWD LOX/FWD FUSELAGE	14.79	14.97	14.98	14.99	14.94	14.85	14.81		14.58	14.36	14.26
TANK/WING/RUDDER	15.34	15.62	15.62	15.64	15.57	15.40	15.26	15.80	15.13		14.31
WING/RUDDER									14.95	14.62	14.52
WING/BODY TORSION									11.68	14.00	
WING BEND								14.80			
TANK/WING BEND.									13.25		15.13
WING/BODY TORSION										14.91	14.78
WING/BODY TORSION										14.93	

Figure 14. NASP Modal Analysis Was Done With a Simplified NASTRAN Strength FEM

3.2.4 Test Requirements Were Affected by Predictive Capabilities

The NASP program management was under pressure to control the cost and schedule of Phase III, especially since the program had originally been planned for $3.3 Billion. During the development of the Phase III plan, the engineering team and government team subject matter experts identified knowledge gaps and predictive uncertainties that required extensive testing. Program management planned to control the cost and schedule by presenting a "success oriented" approach where vehicle design & technology development were performed in parallel. Rapid design updates would

be accomplished as test results became available. This approach involved schedule and cost risk. Design changes to accommodate test results would be made during the hardware fabrication process, with hopes of minimal delays for design modification.

An example of the parallel vehicle and technology development was the plan for fabrication of a large scale tank component that would not be tested until the initial design of the flight vehicle tank. Another example was the intent to design the vehicle and flight engines in parallel. Only propulsion test rig data was to be available, and no flight-weight engine would be tested until three years into the plan. This resulted in the Airframe Structural Integrity Program (ASIP) and Engine Structural Integrity Program (ENSIP) running in parallel with flight vehicle design and optimistically incorporating technology maturation from the Phase II test articles, which were not completely tested until early in Phase III. Likewise, subsystem technology was to be integrated into the vehicle design as the subsystem designs quickly incorporated test results. System updates were planned to be incorporated incrementally throughout the design process. Propulsion improvements from the continuing ground test and wind tunnel engine test programs would be incorporated incrementally into the propulsion system design. Figure 15 illustrates the overall timing of the testing, vehicle and engine fabrication, and flight test plans.
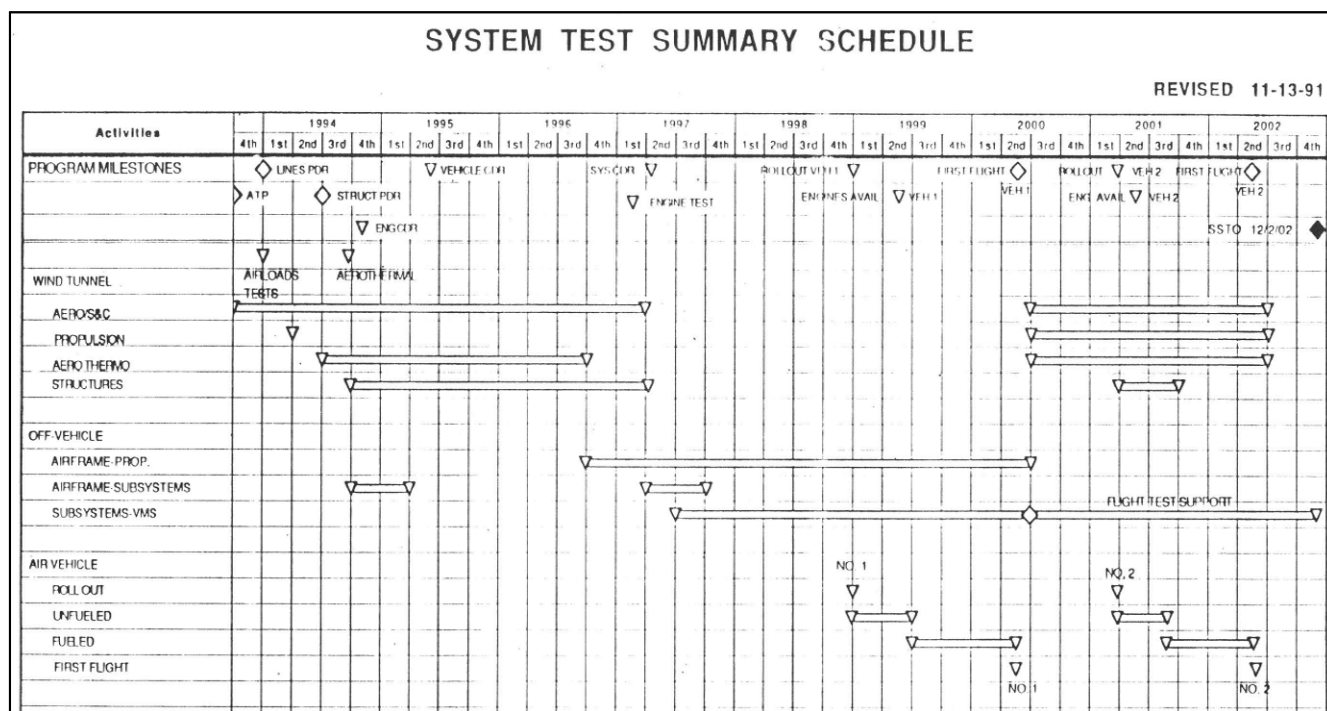
Figure 15. The NASP ASIP & ENSIP Overall Schedule Was Aggressive - 5 Years to Roll-Out

Airframe structural integrity could only be assured by rigorous design development tests, followed by full scale integrated component tests. Figure 16 lists the test types, sizes, heat, and cryogenic requirements.

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Phase 3 Planning Estimated ASIP Tests

	DESIGN DEVELOPMENT TESTS				FULL-SCALE COMPONENT TESTS			
	Total Tests	No. of Tests over 10x10x10 ft	No. of Tests Req Heating	No. of Cryo Tests	Total Tests	No. of Tests over 20x20x20 ft	No. of Tests Req Heating	No. of Cryo Tests
FOREBODY	31	3	28	5	9		9	2
Center Fuselage ref: MacGuire	20		15	4	10		10	2
Aft Fuselage	11		11	2	1	1	1	
Wing	9		9	1	3		3	1
Empennage	6		6		4	1	4	
Engine Integration	6		5	3	9	1	4	1
H2 Cyro Tanks	43	4		43	17	6		17
LOX Tanks	31	4		31	13	6		13
Passive TPS	10		10		5		5	
Active TPS	10		10	8	5		5	5
Landing Gear	10		1		5		1	
Full Vehicle Sections	0				4	4	4	1+
TOTALS	187	11	95	97	85	19	46	42+

NPO-JCM-91-MR05

Figure 16. The NASP ASIP Included 187 Tests, With 19 Larger Than 20x20x20 ft.

The Phase III plan called for structural tests, integration tests, environmental tests, system tests, and 16,000 hours of wind tunnel testing. Wind tunnel tests were planned to close knowledge gaps across Mach/altitude and configuration conditions. Round one included 53 entries to obtain vehicle design data and verify tools. Round two included 21 entries to complete the database & resolve uncertainties. Moreover, the total hours for rounds 1 & 2 were 16,340 which was a very significant test cost. Round 3 included 14 entries to support flight test, bringing the total hours to 17,620. The wind tunnel test plan included a full set of dynamic models and entries at all appropriate Mach numbers. Some improvements to existing test facilities and some new capabilities were also required. These tests are detailed in Figure 17.

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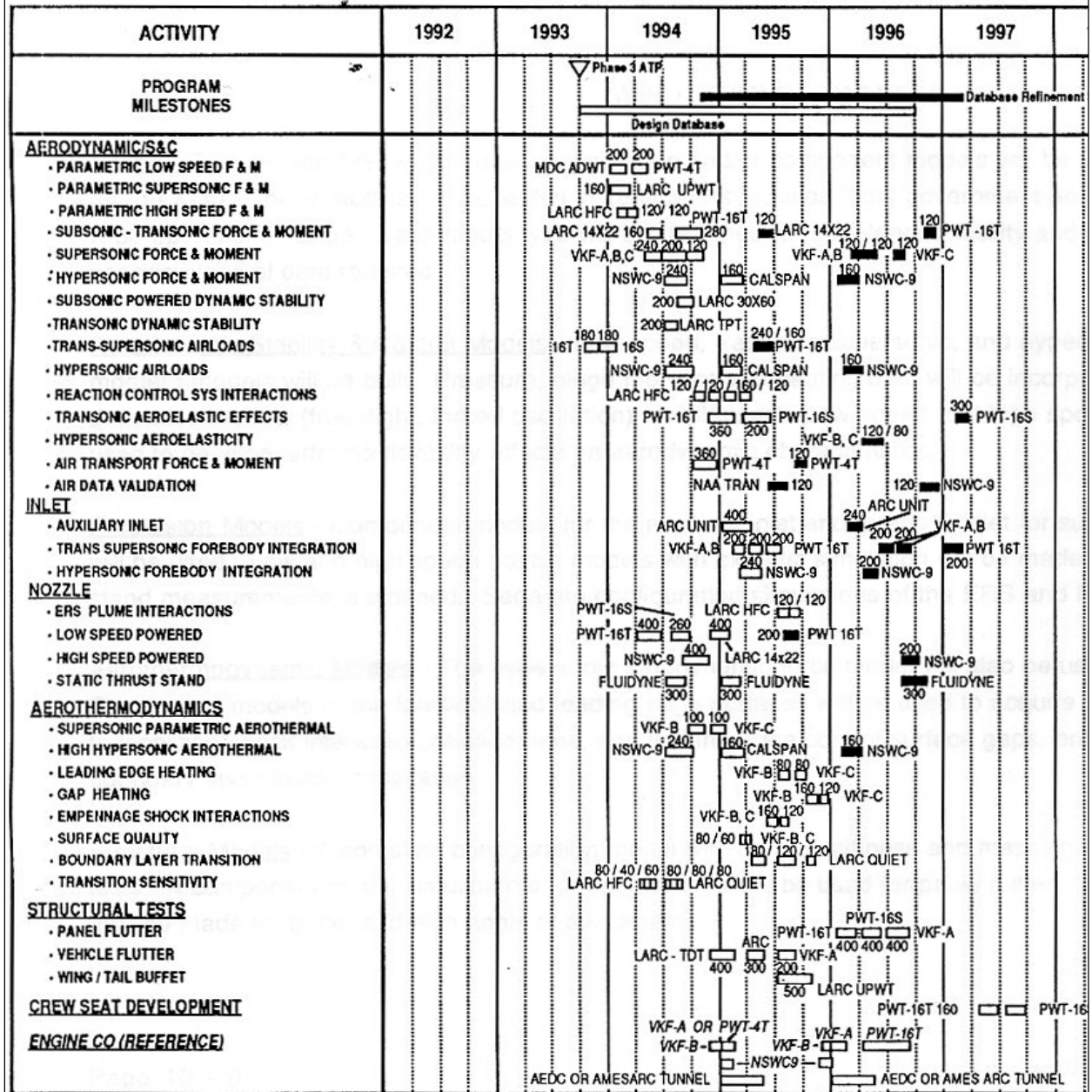
Figure 17. NASP Phase III Wind Tunnel Models Include Transonic & Hypersonic Aeroelasticity, Panel & Vehicle Flutter, & Vehicle Buffet

Since the ability to computationally predict hypersonic conditions, the resulting loads, and structural response was not sufficiently accurate, testing was required. The models, sizes, and test types are summarized in Figure 18. Testing was similar to a complex supersonic aircraft but with the addition of high Mach conditions. No significant tests were eliminated by hypersonic predictive capability in this case.

MODEL	DATA TYPE	SIZE	DESCRIPTION	MODEL	DATA TYPE	SIZE	DESCRIPTION
AERO, S&C, LOADS, MODELS				**PROPULSION MODELS**			
LOW SPEED	Force, Pressure	3 Ft	Complete Airplane Model With Some Parametric Configuration Variables and Analysis Pressures	ISOLATED AUX INLET	Pressure, Mass Flow	5%	Two Dimensional Inlet/Duct Model
TRANS/SUPERSONIC	Force, Pressure Loads	5 Ft	Similar Configuration Variables With Structures Quantity Pressure, Hinge Moment , and Heat Transfer Data	SUPERSONIC FOREBODY/INLET	Pressure, Mass Flow	4.5 Ft	Forebody With Inlets Flow Path
HYPERSONIC[1]	Force, Pressure Loads, Heat Transfer	4 Ft	Similar Configuration Variables With Structures Quantity Pressure, Hinge Moment , and Heat Transfer Data	TRANS/HYPERSONIC FOREBODY/INLET	Pressure, Mass Flow, Heat Transfer	6 Ft	Forebody With Dual Inlet 5 and Flow Paths Adjustable Transonic Hypersonic Flight
DYNAMIC STABILITY SUBSONIC, POWERED	Force, Deflection	6 Ft	Free Flight Model	ERS PLUME INTERACTIONS	Flow Vis. Heat Transfer	1.4 Ft	Complete Airplane With Rocket Simulation
DYNAMIC STABILITY TRANSONIC	Force, Deflection	3 Ft	Forced Oscillation Model	LOW SPEED POWERED	Force, Pressure, Heat Transfer	9 Ft	Complete Airplane With Engine Exhaust and External Burning
RCS INTERACTION	Pressure, Heat Transfer	1.2 Ft	Complete Model for Control Rocket Effectiveness	HYPERSONIC POWERED	Force, Pressure, Heat Transfer	5 Ft	Complete Airplane With Engine Exhaust Simulation
TS AEROELASTIC	Strain, Deflection, Force	6 Ft	Complete Model With Structural Flexibility Simulation	STATIC THRUST	Force, Pressure,	1%	Nozzle Model for High NPR Nozzle Performance, Heat Transfer
HS AEROELASTIC	Strain, Deflection, Force	4 Ft	Complete Model With Structural Flexibility Simulation	**AEROTHERMAL** [1]			
AIR TRANSPORT	Force, Pressure	1 %	Transport/X30 Loads and Separation Model	LE HEATING, GAP HEATING, SURFACE QUALITY	Heat Transfer	4 Ft	Component Model
AIR DATA	Pressure, Optical	10 %	Pressure and Optical Air Data System Calibration MOdel	SHOCK/EMPENNAGE INTERACTION	Heat Transfer	4 Ft	Complete Model for Global Heat Transfer Measurements
STRUCTURES				BOUNDARY LAYER TRANSITION	Temperature	0.8 Ft	Forebody Model
PANEL FLUTTER	Strain, Deflection	2x2 Ft	Component Model	TRANSITION SENSITIVITY	Heat Transfer, Surface Flow	1.4 Ft	Forebody Model
VEHICLE FLUTTER	Deflection, Acceleration	9 Ft	Complete Airplane With Stiffness and Mass Simulation	[1] 4FT HYPERSONIC MODEL IS DUAL AEROTHERMAL MODEL			
WING/TAIL BUFFET	Force, Pressure	3.5 Ft	Complete Airplane for High AOA Testing				

Figure 18. A Comprehensive Test Plan Was Developed in Response to Knowledge Gaps

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Figure 19. NASP Test Flow for Vehicle & System Integration Knowledge Gaps Included Full Scale Test Stands

The 11 systems based integrated test articles shown in Figure 19 included the following:

Six integrated propulsion tests-

1. A low-speed engine integrated with a 10 ft high by 20 ft long portion of actively cooled exhaust nozzle. Thermal and acoustic loads and dynamic structural loads would be obtained. Liquid hydrogen would be required. The test would be incorporated into planned engine tests at NASA/Stennis.

2. A low speed engine with the propellant system components and controls. The test article would be palletized and tested in conjunction with tests of a flight engine at NASA/Stennis.

3. The auxiliary inlet and the low speed engine with integrated Vehicle Management System controls.   This would be tested in the Aero-propulsion Systems Test Facility (ASTF) to verify integrated performance at altitudes up to 100,000 ft and up to Mach 3.8.

4. An auxiliary inlet mated with a low speed engine pre-cooler. The structural integrity of the seal would be validated by subjecting the test article to maximum pressure loads and temperatures.

5. The Reaction Control System (RCS) integrated with Vehicle Management Systems (VMS) controls to determine integrated RCS-VMS controls performance during live development tests firings of the RCS.
6. An engine-airframe full scale "soft" mockup. This would verify fit and form compatibility and develop and verify installation, maintenance, and inspection techniques. A dummy engine would be used.

Three full-scale integrated subsystems-VMS test stands would verify propellant, electrical, and hydraulic systems. Each stand would include simulated user loads, fluid quantities, and VMS controls over the flight envelope. The propellant and hydraulic test stands would duplicate X-30 3-D vehicle arrangement. Both liquid and slush hydrogen plant with safety barriers and proper distance are required. The electrical test stand would be a rack-based system with wire coils to minimize space.

Two airframe-subsystems integration tests-
1. Variable stability aircraft that will have the ability to simulate X-30 control laws and handling qualities would have the remote pilot vision system installed. Piloted flights would evaluate and verify the pilot-vision-airframe interfaces and operations with the flight control system.
2. Live ignition and suppression tests would be conducted using actual sensors and systems installed in a simulated X-30 to verify operational capability of the fire detection & suppression system.

The overall schedule for the integrated tests spanned 6 years. Figure 20 summarizes the planned schedule for design, fabrication, and testing.

OFF VEHICLE INTEGRATED TEST PLAN

TEST NO.	TITLE–INTEGRATED TEST	1993	1994	1995	1996	1997	1998	1999	2000	2001
AIRFRAME – PROPULSION										
2.2.1	ENGINE/AIRFRAME/DYNAMICS			DSN		FAB	TEST			
2.2.2	ENGINE/AIRFRAME FIT CHECK			DSN	FAB	T	AVAILABLE FOR USE			
2.2.3	AUX. INLET/LSOS/VHS				DSN	FAB	TEST			
2.2.4	AUX. INLET SEAL–STRUCTURE			DSN	FAB	T				
2.3.2	RCS/FCS			DSN		FAB		TEST		
2.4.1	PROPELLANT/PROPULSION			DSN		FAB	TST			
AIRFRAME – SUBSYSTEMS										
2.1.1	FIRE DETECTION/AIRFRAME			DSN	FAB	TEST				
2.3.1	VISION SYSTEM/FCS	DSN	FAB	TST						
SUBSYSTEMS – VMS										
2.3.3	FUEL SYS/OX/VPPD/VMS (UMS) (Battleship)			DSN		FAB		TEST	TST	FLT TST SUPP
2.3.4	APU/ELEC/HYD/VMS (UMS) (Electrical Test Stand)			DSN		FAB		TEST	TEST	FLT TST SUPP
2.5.1	FCS/HYDRAULIC/VMS/ PROPULSION (Iron Bird)			DSN		FAB		TEST		FLT TST SUPP

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Figure 20. Integrated Structure & Systems Tests Planned Over Six Years

3.2.5   Rand, Defense Science Board, and Air Force Studies Board Assessments

The Defense Science Board (DSB), Air Force Studies Board, & RAND assessed NASP knowledge deficiencies and identified predictive uncertainties at higher Mach as a key factor in program risk. The 1988 DSB report pointed out the progress and the uncertainties: *"Having looked in some depth into the technologies of importance to the NASP, we are impressed with the progress being made.   But we are even more impressed by what has yet to be done to reduce the remaining uncertainties to a reasonably manageable level.   Until these uncertainties are reduced, the NASP should not be a schedule driven program.   Rather, it should be paced by events.   In particular, we recommend that a set of technical milestones be established which must be demonstrated before a configuration is baselined and Phase 3 detailed design, fabrication, and flight test initiated."*

The DSB further explained the materials and structural challenges: *"The lack of scaled up production processes also affects the quality of the material characterization data available to the structural designer. Small quantity lots will not provide the range of material properties required to establish design allowables, damage tolerance and fatigue characteristics for production materials.*

The NASP structure will be exposed to high temperature, high enthalpy, and disassociated gas. Reusable coatings will be essential to protect the materials.   In areas where the structure is exposed to hydrogen at high temperature and pressure (such as active cooling channels), the hydrogen molecules can penetrate the material and cause embrittlement. The problem is not well understood. The program is raising contractor awareness of the problem, but no funded effort was

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underway at the time of our review. It is the opinion of the Task Force that availability of suitable materials in production quantities will be the pacing element in the NASP schedule, and that resources must be identified to fund the necessary scale up and characterization effort.

Structure
The structural designer has the fundamental task of designing an optimum structure to acceptable minimum margins of safety commensurate with man rating the NASP. To do that requires that:
1) The materials to be used must be fully characterized from material reasonably close to or in production, not from small laboratory samples.
2) The complete operating environment must be reasonably known.
3) The analysis methodology to determine external loads and derive them from internal loads must be available, verifiable, accurate and reasonably efficient.
4) The design can be verified through adequate ground and flight test.
Because of the uncertainties noted in earlier sections in aerodynamic loads and heating, materials availability, precision of computation and lack of ground test facilities to replicate thermal and structural flight loads, the current ability to meet the structural designers requirement are marginal to non-existent.

To achieve the NASP performance goals, the vehicle structural weight fraction will have to be twenty five to thirty per cent less than the Shuttle. In most conventional aircraft the prime loads are aeroelastic. Environmental loads (thermal, acoustic, dynamic response) may be critical locally, but are not usually coincident with the critical aero loads and are normally analyzed as separate design conditions. But for the NASP the loading is aero thermal elastic acoustic and is coincident at the critical design conditions. Achieving the required structural mass fraction in the face of existing computational capability and uncertainties in the load and material data bases is problematic."

RAND Report R-3878/1-AF, "The National Aerospace Plane (NASP): Development Issues for the Follow-on Vehicle, 1993, focused on Computational Fluid Dynamics (CFD) uncertainties, but engineers working on the program agree that this also applied to integrated structural predictive capability. The following is from the RAND report:

Our examination of the **technology** issues surrounding the program leads us to conclude that substantial progress has occurred in overcoming formidable challenges but that many problems remain unresolved. The present CFD state of the art will only allow CFD to serve as a design tool for NASP with a substantial program of testing, experimentation, and analysis to narrow the major hypersonic aerodynamic and combustion uncertainties. Also, the integrated propulsion system contains some unprecedented hurdles, and the current approach to solving these problems allows little margin for error. In our view, more conservative, less precarious approaches exist. Specifically, we would cite subscale experimentation to complement the basic X-30 program.

- CFD simulations are gradually replacing or supplementing wind tunnel experiments for a class of well-understood aerodynamic configurations and applications.

- In terms of current modeling, the accuracy of CFD compared to experimental results in the lower hypersonic range is about six to seven percent. This level of accuracy falls below what would be required for high confidence in the results, particularly for certain flow situations that are likely to be important for NASP, and when limited data exist to guide computational analyses.

- Given the absence of substantial validating data for the CFD codes at higher hypersonic speeds, the errors will likely be much larger. This area needs a great deal of research.

NASP engineers recognized that the CFD uncertainties mentioned above were similar to the structural prediction uncertainties about material properties after repeated exposure to NASP environments. As the DSB noted, the predictability of complex integrated structure and systems in challenging thermal & chemical environments with critical loads essentially occurring all at the same time was not adequate to reduce the need for an expensive 6 year test program.

Despite tremendous progress, computational capability reduced but did not eliminate enough of the testing needed to keep the program sold. Knowledge gaps and the estimated cost to build & test NASP brought the end of this ambitious program. In the Dec, 1993 Government Accounting Office report to Congress the NASP original 1986 estimated cost of $3.3 Billion (1992 dollars, or $4.8 Billion 2008 dollars) was then estimated at up to $14.6 Billion ($21.3 Billion 2008 dollars).

3.3   NASA X-33
3.3.1 Background and Structural Layout
The X-33 vehicle, while primarily considered a rocket with the majority of its structure sizing heavily dictated by the thrust portion of flight deserves special mention in the area of LM Aero hypersonic vehicle experience and specifically in the study areas of coupling environments in analyses. In all respects, the vehicle's return from orbit and horizontal landing design deems its consideration, at least in part, as part aircraft. In addition, the X-33 program represented the most ambitious effort in the development of TPS type hardware since the space shuttle design by Rockwell International in the early 1970s. In doing so, numerous deficiencies in the state of predictive capability for this type of vehicle, along with a number of lessons in the design of thin sheet and thin gauge metallic structure, were gained. Hence, it is referenced in this report.

The windward portion of the X-33 aeroshell (Figure 21) was chosen to demonstrate key metallic thermal protection concepts and was designed to be traceable to future development of the RLV/Venturestar reusable type operational vehicle. The majority of the windward surface acreage of the X-33 and the canted fins were constructed of a diamond-shaped honeycomb sandwich structure bolted to a stand-off bracket that was attached to the tank/Thermal Protection System Substructure (TPSS). The stand-off brackets were designed to maintain the shape of the aeroshell Outer Mold Line (OML), transmit aerodynamic loads to the vehicle internal structure, minimize heat flow to the substructure, and manage metallic TPS panel thermal growth. The corners of each panel were attached to the stand-off brackets which were attached to the TPSS. The panel size and construction were designed to be traceable to the operational RLV/Venturestar panels. Even

though the X-33 was a 53% scaled model of the RLV/Venturestar the metallic panels used were 100% scale. Panel contours ranged from completely flat to complex contoured. Inconel 617 alloy was selected for the windward honeycomb panels due to its performance at temperature, oxidation life, and creep resistance as it was available as a thin foil (.016" thick). The thin facsesheets ranging between .006" - .010" were brazed to .0015" - .0035" foil honeycomb.

While the TPS employed in the design of the X-33 vehicle does not fall under the true definition of hot structure it did share one very key element with the design of load bearing hot structure – the design drive toward thin gauge metallic skin under complex and coupled loading environments. Below the honeycomb panels were soft insulation "packs" that constituted the term of "Thermal Protection System". Yet, the panels were designed to work both as a temperature shield as well as the aeropressure load sharing, or "load transfer mechanism", to the vehicle substructure. Changes in the rate of vehicle heating or cooling causing severe thermal gradients in the metallic TPS were seen as large obstacles in the design. In addition, the phasing of the pressure loads acting on the TPS with the hardware temperature had a critical effect on the overall structural capability of the metallic sandwich panels. And, this rate of change of the heating and phased pressure loads and temperatures became the TPS constraints for trajectory design. As a result, the peak temperature limits for a set of locations on the X-33 and therefore the TPS panel integrity, in turn, were critical driving factors on the overall vehicle trajectory design.
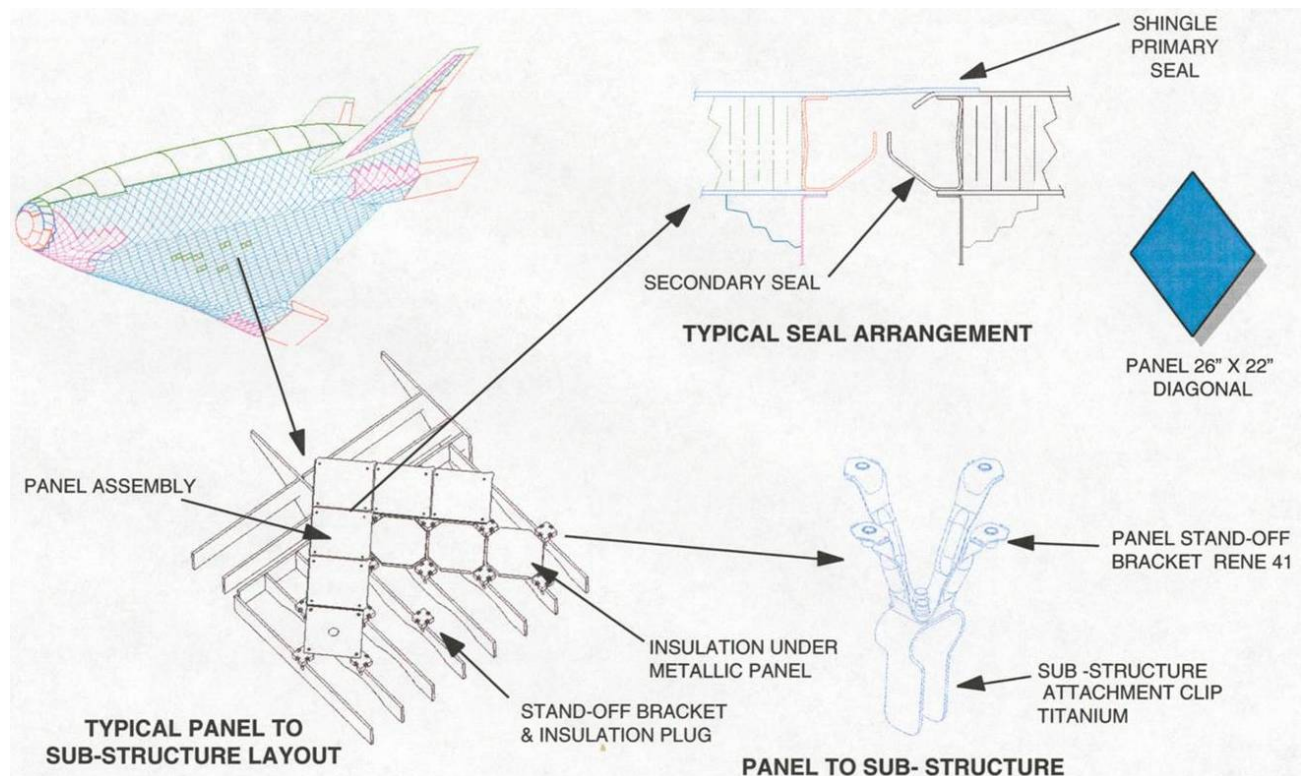

Figure 21. X-33 Metallic Skin Thermal Panel Integration

Limiting factors of temperature were, at first, considered to be too conservative and very non-realistic in the realm of the complex loading environment. Later on in the design phase and instead of setting temperature limits the metallic panel constraints were defined by monitoring the

estimated maximum stresses and the resultant creep strain. For the monitoring of the panel stresses simple relations for the panel stresses were developed in the form of the following equation:

$Ts = (C_1*deltaP + C_2*deltaT)/Fu$

With: C_1 and C_2 = constants dependent on the size, material, and thickness of the panel
    deltaP = pressure load acting on the panel
    deltaT = temperature difference between the inner and outer facesheets of the metallic honeycomb panel
    Fu = ultimate tensile strength dependent upon temperature
    Ts = the pre-set limiting value

With the above is a relation that takes into account, together, the thermal induced stress levels combined with the pressure induced stress levels. The delta P level was computed via the difference between the local pressure acting on the OML and the internal pressure being assumed equal to the atmospheric pressure through regulating hardware means. Where the uncertainty factors and factors of safety were added were in the delta P and delta T applied loads as well as in a factor for dynamic loading due to vibro-acoustics environments being added to the delta P term.

Metallic hardware that is designed for exposure to multiple thermal cycles will experience creep and therefore an allowable creep deformation had to be defined for the service life of the part with margin. As a result, an allocated amount of creep deformation was defined for each flight of the X-33 during its 15 flight test demonstration cycles such that the TPS panels would approach their limits of creep deformation as the vehicle had reached the end of its design life. In this manner the cumulative creep strain for each flight would therefore be compared against its allocation. In order to assess the projected cumulative creep strain for each flight certain coupled environment relations had to be established.

$$Cs = \text{function of } f(deltaP,T) \, dt$$

With the function, f, as a function of the outer facesheet temperature and the pressure load the combined loading was integrated over time in order to obtain a cumulative creep deformation for the X-33 flight program. Once it was attained it was compared to the allocation or "budget" that was assigned to it. The constant in the function, Cs, is a function of the panel material and size. These function values were then called "trajectory indicators" with the following data required as the input:

1. The local pressure acting on the surface and the atmospheric pressure
2. The freestream total enthalpy and the local heat transfer coefficient.
3. Estimates of the temperatures of the inner and outer factsheets of the metallic panels with the (above) heat transfer coefficient, the local pressure, and the total freestream enthalpy through 1-D analysis.

As updated material properties, panel geometries, and better defined structural models were developed as the program progressed the constraints were modified to reflect the revised boundary conditions.

As it would similarly pertain to the edges of a hot skin structure panel, whether stiffened sheet or of

honeycomb, one of the biggest design and analysis challenges of a metallic panel system was to seal the gaps between the panels and non-metallic panels where large deltas in coefficients of thermal expansion were both known to, and assumed to, exist. Together with this was the criticality and challenge to accommodate the substructure motion with its inherent deflections due to thermal expansion and aero loading. When gaps between panels developed so did the additional heating effects which drove the huge necessity of minimizing the steps and gaps between the panels.

3.3.2 X-33 Metallic Panel Structural Analysis

Overall, the flight qualification of the X-33 metallic panels (Figure 22) was based on analyses closely connected to correlation in tests. The primary reason for this was due to the fact of not having any one-test facility that could simulate all of the X-33 environments through the full trajectory of the vehicle and the amount of unique structural (varying material types) configurations of the vehicle. The way around this was to combine as many major loading combination tests as possible with model correlation performed afterwards.

The major static loads on the metallic panels were:
- Aerodynamic heating
- Aerodynamic normal pressures
- Local aero heating effects due to panel panel deformation and movement
- Substructure motion
- Engine plume heating effects

Minor static loads that affected overall sizing were:
- Inertial loads
- Aerodynamic shear loads

Major dynamic loads were:
- Acoustic pressures from primarily sound sources

Minor dynamic loading effects were:
- Random vibration levels
- Liftoff transient and the transient ignition over pressure condition with a very low frequency (more static than dynamic)

31
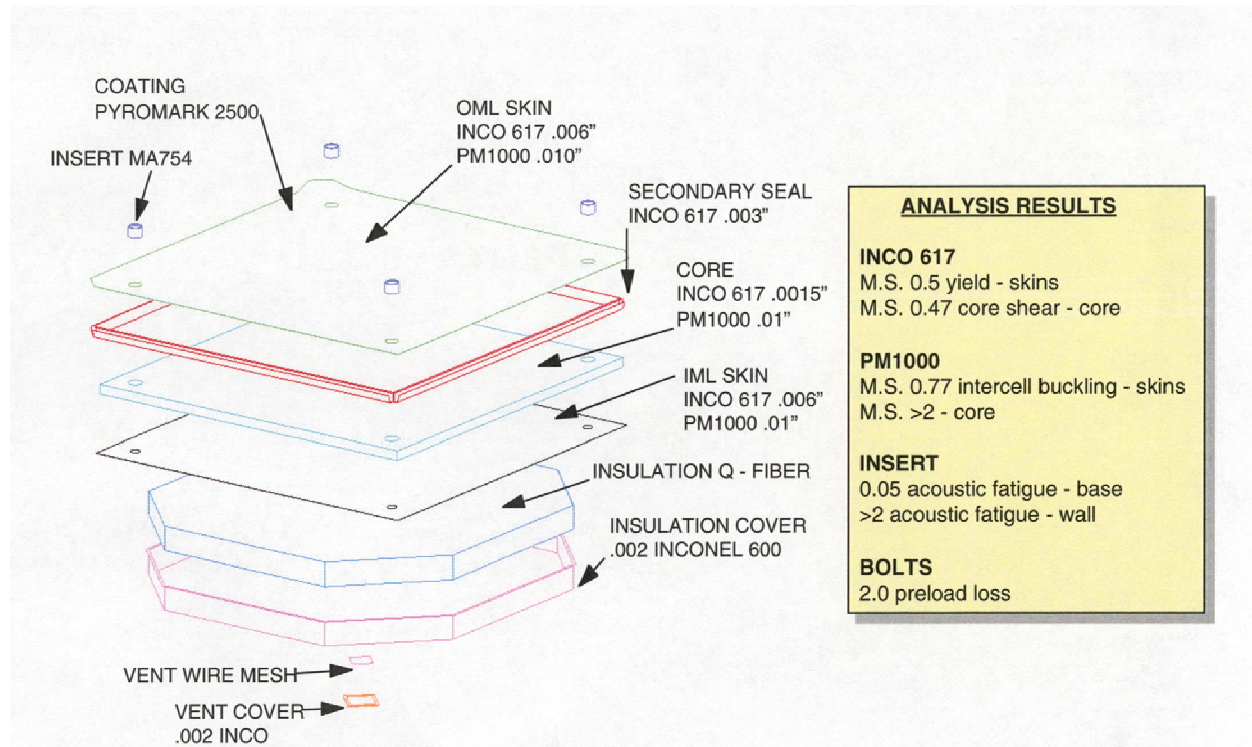
Figure 22. X-33 Metallic Inconel 617 Windward Side – Typical Panel

The challenge to the analysis team was to retain the appropriate coupling of all the static and dynamic loads with the proper time phasing during the flights. Another critical aspect of the analysis approach was to address all of the varied configurations of the panels (various shapes). Each panel location on the vehicle had its own unique loading condition which was trajectory dependent. Attachment hardware and its varied configurations also had to be addressed in a like manner.

When it got right down to it on the panels a critical design factor and therefore service life determining factor were the inserts and the analyses thereof. A mechanical insert would share its criticality with that of the mechanical attachment of a hot structure panel to a structural member. As it turned out the most critical stress levels in the insert were generated by the insert acoustic and thermal loads. Delta temperatures between the inner and outer facesheets of the panels caused the panels to bow which resulted in significant punch loads on the inserts. In turn, the thermal loads induced by temperature gradients, in both the in-plane and out of plane directions, caused highly critical stresses in the inserts. In this manner, fine mesh 3-D panel models were created for detailed 3-D analyses.

Attachment bolts were analyzed in combinations of punch loads as induced by panel thermal bowing, vibration loads, and time at temperature under load. It was also later found out that the material strengths of the PM-1000 fasteners were extremely sensitive to how they were processed in that the rolling of the threads were thought to most likely change the strength of the material. Component tests of these were heavily employed.

The down selection of the metallic panels, and their locations, for detailed and finer mesh analyses,

was performed through analyses using linear elastic small deflection shell FEM using the MSC/NASTRAN finite element code. However, the constructed NASTRAN models did not account for:

- Plan form temperature distributions due to flow coupling with panel deflections
- Large deflection analysis for large radius panels that yield twice the stress of small deflection analyses
- Local temperature loads around inserts
- Panel internal loads due to plastic and creep strains
- Edge cooling from the lower insulation bag conduction
- Thermo mechanical panel seal analysis

Linear type analyses methods were used for acoustic fatigue, flutter and vibration analyses for panels and their seals.

Once structural zones for higher fidelity meshes and highly detailed analyses were down-selected the MARC nonlinear structural and thermal finite element code was used for detailed nonlinear finite element analyses. Detailed thermal/structural analyses were performed as a time history simulation of a trajectory to account for:

- Coupling between the aero-heat flux and the panel deflections
- Thermal transients in the entire panel stack-up assembly (honeycomb panel, inserts, fasteners, etc.)
- Visco-plastic material response
- Large deflections of the panel and attachment hardware

Highly critical plastic and creep strains were developed in the metallic panel skins and core that affected the subsequent stress levels later in the trajectory with the combined effects of varying pressure and temperature. Temperatures around panel inserts were derived through an account of the insert edge distance and the acuteness of the panel corner. In addition, the insert thermal loads were combined with the punch loads in a linear manner. By scaling the enthalpy heat transfer coefficient by a NASA/Ames supplied function of the panel deflection and panel location at each time point during the coupled thermal/structural time history trajectory simulation the aerodynamic heating coupling with panel normal deflections was accounted for. Large deflection effects were included to account for significant changes in panel curvature as well as any stress stiffening the metallic panel might undergo during the trajectory. The metallic panel field stresses were a strong function of the radius of curvature, edge length, panel skewness, and diagonal ratio.

The coupling effects of random vibration and combined acoustic loading were reviewed to ensure compliance with vehicle life requirements. For each component's acoustic analysis, a normal modes analysis (NASTRAN SOL 103) was first run on the panel FEM. Afterwards, the modal output along with the vehicle acoustic environment data for different stages of flight, lift-off, ascent, and reentry, was co-processed using a Rohr-proprietary Fortran code. This code computed the overall element strains and stresses for each of the flight segments.

Power Spectral Density (PSD) curves were provided by LM Aero and based on the normal modes

analysis previously conducted, a mass participation method was utilized to compute composite random vibration load factors relative to an orthogonal coordinate system. Then, static analyses using NASTRAN analysis on the FEM was then executed for unit (1g) inertial loads along each of the orthogonal axes. Element stresses from the three axes load cases were then scaled by the corresponding random vibration load factors in order to obtain effective stresses due to vibration. To follow on, the element vibration and acoustic stresses were then combined and applied in a Miner's rule calculation to characterize the total fatigue damage in each element and to assess the fatigue life.

In general, aerodynamic heating was input as a film-boundary condition that was a function of panel bowing and location on the panel itself and this heat load was very dominant as compared to other loads. Heating amplification connected with the panel deformation was provided by NASA/Ames as a function of the panel center displacement and location over the OML surface of the panel. As an overall uncertainty in the analyses, variable heating across the metallic panel widths had to be accounted for by the analysts subjectively.

During the course of the analyses it was found that the aeroheating amplification factors occurring when the flow skips over panel to panel gaps and steps between gaps were becoming more and more critical to the design. And, heating amplification factors varied as to whether the step was aft facing or forward facing. The steps in the X-33 panels would be quite similar to those in a hot structure design on account of panel edge bowing, sub-structure motion, and tolerance stack-up. Steps between panels could cause the thin gauge metallic edge to rotate up into the air-stream inducing a heating amplification on the panel edge as well as on the adjacent panels. In the X-33 program the areas that were considered most likely to experience these amplifications were the metallic TPS behind the leading edges and around the soft seals surrounding gear doors, in addition to transition areas to different structural materials – such as a carbon fiber reinforced leading edge.

3.3.3   Panel Edge Effects

By the cancellation of the program in early 2001 the effect of aeroheating amplification had not yet been modeled for the structural analyses. Since many of the possible steps encountered would be a function of the magnitude of the panel bowing, this coupling had been planned for incorporation in the coupled thermal-structural analysis in the same way as the panel bowing effects were addressed.

Inherent to any curved panel, with the higher radius dictating the higher edge loading, it was shown that the panel curvature and edge loads were directly interrelated. Edge compression loads were developed from aero panel loads, thermal panel loads, and thermal gradients normal to the panel edge. And, if a panel edge buckled, both the wavelength and height of the edge would change with the load level. No panel edge loads or steps between panels or local seal (between panels) were considered during the analysis phase. Once panel edge loads were considered there was the potential to double the seal compressive strain level. At the time, this was considered to possibly increase the deflections by 50%. Deflection levels such as these were around 20% of the in-flight boundary layer thickness during the high heating portions of the trajectory. In addition, steps between the panels had the potential to substantially increase these deflections by rotating the seal further out into the boundary layer. The deflections in coupling with the reversed flow over the

edges and seals could have lead to some substantial additional heating that would have been extremely challenging to predict or even test for. This all points to the fact that local edge heating and its inherent effects, was required to complete the panel analyses before final flight qualification. As it stands, this area remains a critical gap in the full understanding of local heating effects on metallic panels in hypersonic flight.

3.3.4 Detailed Panel Stress and Stability Checks

Aside from the aeroheating and local heating effects within the metallic panels were the effects of the residual stress within the panels as a result of the brazing of differing material skins. To illustrate, the panels were constructed of Inconel 617 and the inserts were of MA-754 alloy, with a differing coefficient of thermal expansion between the two of approximately 7%. As the materials were brazed together at 2000°F, they were considered to be essentially stress free. After the braze solidified at temperature, the panel was cooled which resulted in an accumulated strain difference between the materials at room temperature. To account for this residual stress in and around the inserts the strain difference was modeled and therefore used as a starting point in the detailed stress analyses. A MARC finite element subroutine had to be developed in order to write and read in the stresses that were incurred from the brazing of dissimilar metals at 2000°F. This subroutine read in results from a mechanical load case where initial temperatures were applied to elements to provide the accumulated strain difference between the MA-754 rod inserts and the Inconel 617 skins from 2000°F down to 70°F. However, the relaxation of the residual stress at any intermediate temperature within the range, based on mechanical loading at the temperature, were an element for consideration in future work.

Metallic panels were considered to be most critical in stability analyses. For the structural stability analyses, completed through the NASTRAN buckling Eigenvalue analysis of the structure, the Eigenvaulue was the ultimate load factor (1.25) times the applied load where buckling was considered to occur. However, certain limitation were set and contributed to a level of uncertainty in the analyses and resultant sizing. These limitations included that fact that the material was considered linear elastic, the structure was free of imperfections, and the analysis did not indicate whether the buckling mode was stable or unstable. And, unstable buckling modes could have resulted in buckling loads that were factors of two or more lower than what the Eigenvalue analysis prediction resulted in, causing additional uncertainty.

During the program it was considered to perform FEM based buckling analyses of thin curved shell structures with high membrane loads, to check eigenvectors (buckling modes) for unstable bifurcation (buckling) points. What was considered highly critical and therefore difficult to predict was the fact that imperfections in shells with high membrane loads had to be assessed and analyzed for strength whether or not post buckling behavior was deemed a structural concern. As a result, post buckling analyses for each mode under Eigenvalues of 2.0 were run.

Flutter margins on the panels were to be reported on the basis of vehicle velocity versus dynamic pressure. The factor of safety for flutter was provided by Space Shuttle criteria and industry standard factors. However, as it was not available, the subsonic and supersonic factor of safety was derived from prior Shuttle analysis as a 11.5 factor on the ratio of critical/vehicle dynamic pressure. The dynamic pressure criteria were converted to velocity through Bernoulli's equation which resulted in a 1.23 factor on the vehicle. Yet, due to uncertainty, an additional factor of two

was applied for hypersonic cases. Flutter was only considered a potential design issue for the metallic panels but the analyses had not been fully completed at the time of the program's cancellation.

For analysis, the X-33 metallic panels were divided into seven groups based on similarity in vehicle environments, panel geometry, and induced stress levels. Thermal stresses were primarily related to the metallic panel radius of curvature and diagonal ratio whereas the pressure induced stresses were mainly related to the panel's longest edge length. The metallic panels that were eventually down selected for detailed structural analysis were based on a thermal delta temperature loading case coupled with a unit pressure case applied to a linear elastic small deflection shell FEM built in the NASTRAN code. The thermal delta temperature case corresponded to the maximum temperature difference between the outer and inner skin but only at the center of the panel, and with the temperature in each skin applied only uniformly across the entire plan form of the panel. This effect opened the gap on the effects of fluid/structural coupling and the large local temperature gradients around the inserts. In reality, the local temperature gradients around the inserts might be even higher than the through thickness gradient through the panel. Per each group of panels three criteria were used to select the panels for refined analyses for both thermal and pressure loading conditions:

> Thermal Loading Conditions:
> > 1. One panel with the highest core shear loads (at the insert)
> > 2. One panel with the highest skin stresses at an insert
> > 3. One panel with the highest field area (in-plane) skin stress level

3.3.5 Metallic Panel Structural Indicators

Each group of metallic panels had many different environments associated with it. This dictated the need for a methodology formalized in order to justify the most critical environments for analysis. From this arose the structural indicator program that estimated the environments and the resulting load level in the panel. Then, the resulting linear stress levels were normalized by the material strength and the stress/strength ratio was used as the load indicator value. The environments that yielded the highest indicator for thermal stress alone, creep due to pressure loads, and thermal plus pressure stress were applied to the panel FEM. However, what was not included was the internal loading due to the panels' creep, plasticity, or local heating effects during the flight. At the time of program cancellation these effects were not yet incorporated and clear plans as to how to do it were not yet formalized.

Six locations across a panel were used to create a stress envelope as illustrated:

1. First location: at the maximum stress location from thermal loading
2. Second location: at the maximum stress point from pressure loading
3. Third location: at the highest stress location resulting when maximum pressure and thermal stress fields were added together
4. Fourth location: by determination of the highest stress location resulting when maximum pressure and thermal stress fields were subtracted from each other
5. Fifth location: at the maximum stress location due to the uniform thermal expansion of the panel

6. Sixth location: at the point of highest insert stress; the highest of the OML or IML points were used

In addition to this, honeycomb panel indicators were employed to track its loading:

1. A static strength indicator
2. A fatigue indicator that was based on the tracking of the strain range induced from thermal and pressure loads
3. A creep indicator which calculated the amount of creep strain that the panels would accumulate from the aerodynamic load case only. In a way this was a stiffness indicator and it estimated the amount of deformation that would accumulate over the 15 flight test program and then be normalized by the allowable deformation of the panel. In this manner, this indicator set the maximum temperature capability of the part.
4. A thermal-plastic indicator was proposed, which would estimate the amount of creep and plastic deformation that the panel would incur from thermal loads alone. Inherent in this was the assumption that the total strain field did not influence plastic or creep strains. However, this remained an uncertainty and essentially a gap in the overall analysis path.

In review of the stress indicator program it was evaluated that a key missing ingredient that was adding risk was the fact that material nonlinear response was not included which thereby made the stresses and strains a function of the sequence of loading and not just of the load level.

3.3.6 Varying Material Property Estimations

In estimating the inelastic behavior of the metallic materials a classical plasticity model using a Von Mises, or J2, invariant flow vector with a linear kinematic hardening rule was used. Limited cyclic test data for the brazed Inconel 617 honeycomb panels and MA-754 insert material suggested that the kinematic hardening model was a reasonable representation of the metallic material's behavior. However, cyclic data on the Inconel 617 alloy in strain hardened condition was not available. The creep model employed the same flow vector as the plasticity model along with a creep strain hardening rule but the analysis did not account for the following effects: creep recovery, creep ratcheting, creep loading rate affect, compressive creep response, compressive plastic response, thermal cycling effects on plasticity, plasticity-creep interaction, and non-proportional loading. And, as mentioned previously, the effects of the residual stress from previous flight and their results upon the material's state was not taken into account.

Lastly, even though the X-33 flight test program consisted of 15 flights random fatigue analysis was performed with shell element models using the NASTRAN code and an in-house FORTRAN code that calculated stress levels using the zoned third octave acoustic sound pressure levels, panel damping and normal eigenvectors and corresponding stress/strain fields. The average and mean stress level fields were obtained from the 3-D non-linear static strength analyses. Fatigue damage was generated on an element-by-element basis using the stress invariant from the average and alternating stress fields. Thereafter, the number of cycles was determined from the zero pass frequency for the element. This approach may have been considered too conservative for some panels which therefore dictated the need for stress fields to be taken into account.

LM Aero's integration of metallic panels into the X-33 lifting body RLV concept was the

stimulant that initiated the rapid evaluation of metallic panels on a hypersonic vehicle on a large scale. The lifting body design was used to take advantage of the inherent high specific strength mechanical properties of metallic structures by shaping trajectories that separated the metallic panel maximum temperature and maximum pressure points. In doing so, lower temperature capable metallic materials (as opposed to the silica based tile design of the Space Shuttle) were then allowed to be used on the majority of the windward side of the RLV type vehicle rather than higher temperature ceramic tile panels. The development of higher fidelity design environments as the analysis and testing matured was heavily critical in the process.

Even though the design of the metallic panels used on the X-33 were much more parasitic in nature, the gaps in analyses capability uncovered formed strong lessons that need to be studied in hot skin structure design in future hypersonic vehicles.

3.4 HTV-3X
3.4.1 Background
Between 2006-2008 the DARPA HTV-3X program comprised the conceptual design effort of an air-breathing Mach 6.0 reusable aircraft (Figure 23). This project in itself has provided numerous design and analyses lessons learned in relation to this class of sustained hypersonic flight vehicles as well as serving as the primary study base for the execution of the Phase I program tasks.

In summary, the program consisted of a 2006 Feasibility Study Phase followed by the 2007-2008 Conceptual Design Study Phase that produced an aircraft design concept for further evaluation in the later to be executed Preliminary and Critical Design Phases. Even though the program was cancelled a tremendous amount of work was performed by LM Aero in outlining the design needs, risks, and technology gaps that needed to be surmounted in order to successfully demonstrate such a vehicle concept.
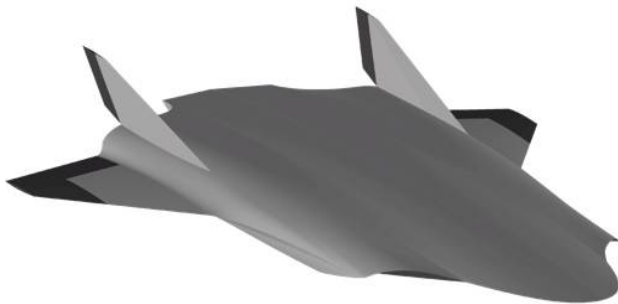
Figure 23. DARPA HTV-3X Early Concept Design Vehicle Outline

Numerous critical questions arose during the HTV-3X airframe structure sizing iterations. The effects of hot skin panel flutter and whether it was most critical at the transonic or high Mach flight phase at the period of max heating, the dissipation of the internal cooler fuel heat sink and its effects on the stability of the hot outer structure in skin thermal buckling, hot structure thin gauge

skin panel joints and their susceptibility to high acoustic loading coupled with transient heating, and hot structure skin deflections and their effects on high Mach air flow are just some of the challenges that were, and still are, in critical need of being addressed during preliminary design of a reusable air-breathing vehicle. In addition, inspectability and repairability of both fuselage and wing skin hot structure and hot structure joints, coupled with the practical aspects of manufacturability, are additional design factors that the team felt had to be answered early in the preliminary design phase. One extremely valuable lesson learned was a great majority of structure sizing and load path layout was determined by the vehicle's ascent to its cruise level altitude and speed. This particular flight phase posed the maximum thermal gradients between the hot structure outer skin and internal cooler structure and subsequently generated the most critical thermo-structural loading cases upon the airframe.

3.4.2 Employment of Thermal Stress Metrics

One issue that arose during the conceptual design phase was the increasing amount of vehicle "dry" weight amidst the efforts of closing the conceptual level design. The HTV-3X structural baseline consisted primarily of nickel based Inconel 718 and 625 structure. While the material database of these alloys is well established and the alloys are readily available, refined and detailed thermal stress and stability analyses of these alloys as applied to the airframe design were to be continued in the upcoming preliminary design phase.

With the potential of overall vehicle weight savings and the drive to reduce the anticipated predominance of thermal stress in structure sizing LM Aero conducted trade studies on the substitution of high temperature Titanium alloys, primarily alloys Ti 6-2-4-2S, Beta 21S, and Ti Beta C, for the mid-body fuselage re-design. The results of these trade studies are as follows (the C310 label on the chart refers to the study version of the HTV-3X) in Figure 24:
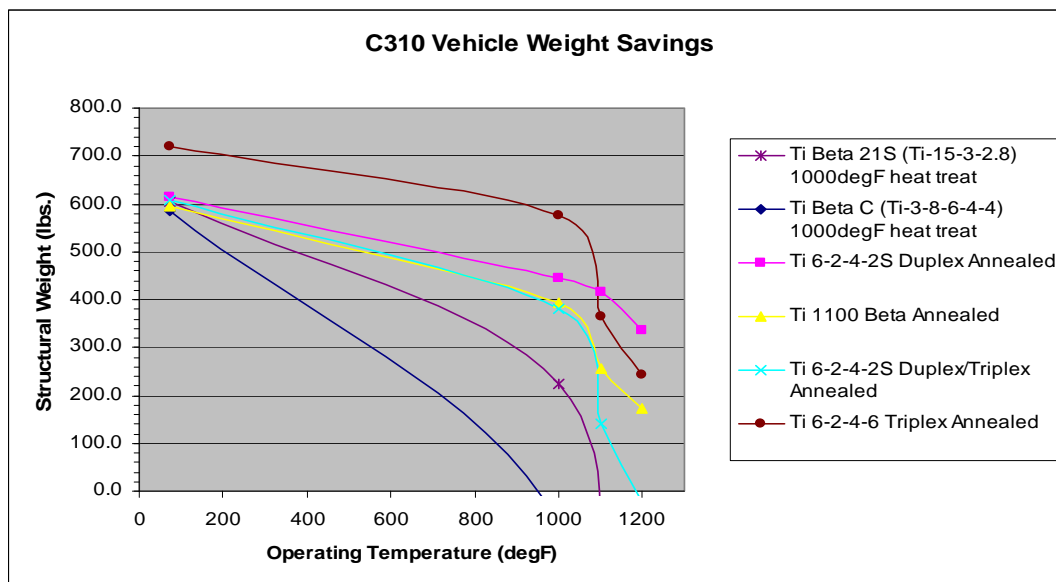

Figure 24. DARPA HTV-3X Airframe Weight Trade Study Results

The results reflect the weight savings as a reflection of pounds of airframe structure as compared to

the baseline nickel. However, the interesting aspect of the discussion is the method in how the weight savings were derived. This in turn points out the need for continued evaluation.

Category	Failure Mode	Weight Ratio (W_2/W_1)
1	tensile strength	$(\delta_2/\delta_1) * (Ftu_1/Ftu_2)$
2	compressive strength	$(\delta_2/\delta_1) * (Fcy_1/Fcy_2)$
3	crippling	$(\delta_2/\delta_1) * ((Es_1/Es_2)*(Fcy_1/Fcy_2))^{**.25}$
4	compression surface column and crippling	$(\delta_2/\delta_1) * ((Es_1*Et_1*Fcy_1)/(Es_2*Et_2*Fcy_2))^{**0.2}$
5	buckling compression or shear	$(\delta_2/\delta_1) * (E_1/E_2)^{**.33}$
6	aeroelastic stiffness	$(\delta_2/\delta_1) * (E_1/E_2)$
7	durability and damage tolerance allowable (DADTA)	$(\delta_2/\delta_1) * (F_1/F_2)$
8	thermal stress	$(\delta_2/\delta_1) * ((k_1Ftu_1/E_1a_1)/(k_2Ftu_2/E_2a_2))$

Figure 25. Criteria for Conceptual Design Structural Weight Trade Comparison

The method incorporated a spreadsheet comparing % contributions of typical aircraft failure modes with weight factors. Mechanical property values for Inconel alloys and up to 1000°F for Ti Beta 21S, Ti Beta C, and Ti 6-2-4-2S alloys were derived from MIL-HDBK-5. Tested and collected data were used for Ti alloys that were under study for service between 1000-1200°F.

The method employed was one that has been utilized by LM Aero mass properties engineers in performing rapid structural weight comparison and comprises a spreadsheet incorporating various weight ratios made via equations (Figure 25) that summed to an overall value. The study included comparisons between the materials' mechanical properties as such: tensile strength, compression strength, crippling, compression surface column and coupling, buckling compression or shear, aeroelastic stiffness, and Durability and Damage Tolerance Allowable (DADTA). Yet, a more accurate comparison between structural materials in the hypersonic regime, and especially those of widely varying coefficients of thermal expansion, would need to include considerations of thermal stress for a more realistic and encompassing comparison. Therefore, LM Aero felt it necessary to augment the study in order to capture the factor of thermal stress performance within the study.

As thermal stress is proportional to $E\alpha$, (with E = Young's modulus and α = coefficient of thermal expansion) materials with lower $E\alpha$ would therefore constitute lower thermal stress fields. However, what is also pertinent to consider is be the magnitude of thermal stress in relation to material strength. In using Ftu (or Fty or Fcy could have been used as desired) as a measure of

material strength, the thermal stress metric became Ftu/Eα, with the grading criteria as "higher is better". In addition, for specified heat flow it can be shown that the metric becomes kFtu/Eα (k = thermal conductivity value of the material).

After the factor was derived the selection of how governing that percentage should be came to the forefront. Should the factor constitute 10% or even perhaps 50% of the evaluation criteria? This was a critical question during the study with no clear answers arrived at as the detailed thermal analyses had not been performed. As a result, a window range of 25%-50% was used to envelope the potential weight savings within the material substitution study. In the field of early conceptual design of hypersonic vehicles where material comparisons are performed, a thermal stress metric, substantiated through material testing under thermal load tests, would be highly valuable to the design and analyses community.

3.4.3 Conceptual Level Thermal Loads Development

The conceptual level design study of the DARPA HTV-3X vehicle called for early assessment of thermal loading and its effects, and contributions thereof, toward the overall sizing of the airframe. Program cost and budget factors precluded thermal transient analyses and so thermal loading consisted of a derivation of displacement boundary condition effects, from thermal loading of the primary heat sources during flight: 1. internal high speed flight aerothermal factors and 2. the internal heat sources generated from scramjet operation over an extended period of the trajectory. To encompass these effects and the complex coupling thereof a careful and methodical integration of various separately performed thermal analyses was required. This is illustrated in Figure 26.

In order to couple thermal loading with aeropressure and inertia effects, and within the charter of conceptual level design, linear static NASTRAN analyses runs were performed with an inclusion of the displacement boundary condition results separately arrived at through thermal analysis runs. It was highly desireable to use the same fineness ratio mesh of the airframe stiffness model for all thermal modeling and thermal analysis runs, and this was performed. However, as aeropressure and inertia varying loads were developed across the flight trajectory, thermal loading input was kept only at the pre-defined Mach threshold points (Mach 1.0, Mach 2.0, Mach 3.0, etc.). The unavailability of more closely matching thermal load data to integrate into a critical aeropressure loading case at an intermediate Mach point, Mach 4.34 for example, dictated the subjectivity and need for assumptions to be made.

During the course of conceptual design it was found that aeropressure load cases within the transonic descent phase of flight, roughly between Mach 1.2 and Mach 0.9, were truly governing the structural sizing of certain airframe zones. Upon descent the majority of internal load carrying structure would remain at a relatively high temperature. To capture these effects, the displacement boundary condition results of the Mach 2.0 thermal descent case, as opposed to those results of the Mach 1.0 descent case, would be integrated into the static model runs for aeropressure and inertia loading for Mach 1.2. The degree of conservatism which perhaps led to an "oversizing" (or "undersizing") of the structure with extra weight committed to a design that didn't need it was never fully detailed. A more closely matching thermal load set to the mechanical load set is what would have been required for the follow-on phases of the program.

Other aspects of thermal loading input that were considered highly critical and therefore necessary for refined thermal analyses using commercial software packages, and would have aided

tremendously in the conceptual level design phase, were

- A feature that would allow temperature to be passed at each increment for combined structure analysis. This could then mitigate the need to map the temperature field by using a secondary input file. To illustrate, In NASTRAN usage, this is called a .pch file.
- For 2-D element type analyses, a feature that would calculate and retain the thermal gradients between the outer skin (or top surface), mid-plane, and inner skin (or bottom surface) of a shell element.
- Finer detailed thermal contact features for true contact and near contact for non-matching meshes between coarser and finer grid models.
- A feature that would allow temperature to be interpolated between different mesh models. Temperature translation capability between fine grid and coarse mesh models would become extremely critical in the preliminary design phase of the DARPA HTV-3X program.

The benefits of the above features are well illustrated in the following analysis detail experienced in the DARPA HTV-3X program.

3.4.4 Conceptual Level Thermal Analysis Detail

The HTV-3X vehicle conceptual design consisted of welded fuel tank load bearing structure comprising the mid-fuselage zone of the vehicle. At the early design stage the thought process was that there needed to be effective and accurate methods for thermal stress calculations through welded structure. One such example is discussed. Even though this constitutes a discussion of a design pertinent and exclusive to a particular vehicle design the phenomena discovered still points to the need for thermal transfer around small joggles and details of joints, etc. as highly critical areas for further research and invaluable to the vehicle level structural analyst.

Figure 26 illustrates a component of the nickel base alloy fuel containment system that was conceptualized to line the mid-fuselage in the vehicle longitudinal direction. Each of these small components relied upon the structural integrity and rigidity in flight of perimeter seam welds undergoing combined effects of varying thermal loading due to the fuel drain per flight, subsequent differing inertia loading, aeropressure loading, and, in locations near the propulsion system, high levels of acoustic pressure levels. In these areas especially were concerns of possible sonic fatigue effects upon the thin gauge structure welds.
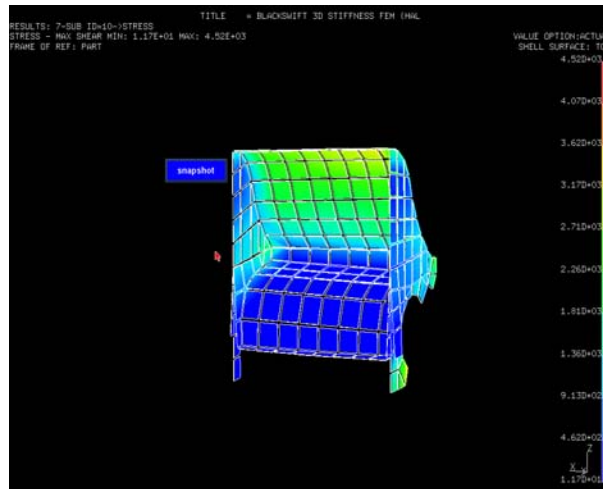
Figure 26. HTV-3X Conceptual Level Fuel Containment System Component

During the conceptual level design phase linear static analyses were conducted along the perimeter weld seams in order to establish requirements in the program's weld processing development phase. Results from static analyses indicated that the combined loading effects experienced at the Mach 4.3 ascent phase of flight produced the highest combined Von Mises stress levels for the first level structural sizing phase. The load cases employed at this point constituted aeropressure loading on the outer skin, thermal loading from both internal (propulsion system associated) and external heat sources (aero induced), and inertia loading of the variant fuel load upon the thin gauge skin or "floor" of the fuel containing component.

While investigating the contributions of the loading the following was determined. Figure 27 displays the results of linear static analysis based upon the application of aero induced and thermal loading only. No results from thermal loading are displayed in these results as the region along the weld zone perimeter indicates a magnitude of around 1ksi.

The results of Figure 27 are then compared to those displayed in Figure 28 which constitute the Mach 4.3 thermal loading results only. Lastly, Figure 30 illustrates the combined, albeit linearly, effects of the coupling of the sources of the loading with some amount of stress relaxation being afforded by mechanical relief of the thermal expansion. Nonetheless, it is shown that in the comparison of the results for the same weld zone near the highest combined loading point under a varying of the input, the thermal stress results constitute ~99% of the contributing shear stress upon the weld. The follow-on questions that arise from this are whether or not the weld material coefficient of thermal expansion would aid or worsen the transfer of the heat, along with the displacement condition, of the weld. In addition, how does the thermal transfer really negotiate the irregularities of the weld surface. And, what amount of heat would be radiatively transferred to the gaseous vapors still within the component even after the fuel load is depleted. All these constituted a huge need for a heavily refined and detailed mesh non-linear analyses of the weld joint to be undertaken in the preliminary design phase of the HTV-3X program. Due to the number of welds a stress indicator program similar to that undertaken in the sizing operations of the X-33 metallic panels, as discussed in Section 3.3 of this report, would most likely have to be undertaken if welded metallic structure remained as the design baseline for the fuel containment system.
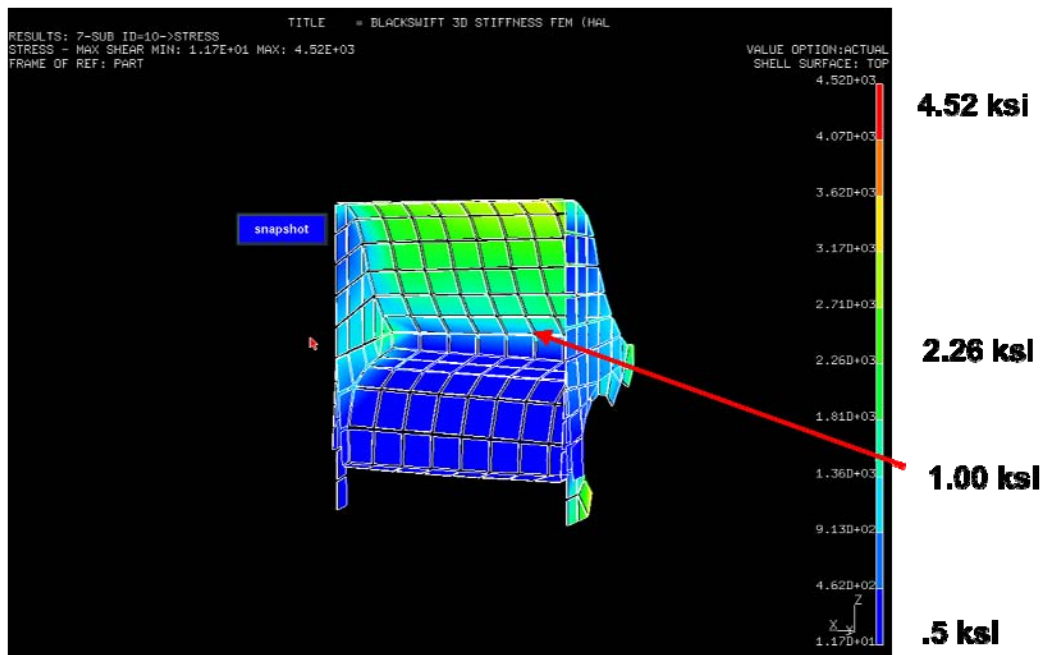
Figure 27. Fuel Containment System Component Resultant Von Mises Shear Stress (Aero and Inertia Loading Factors Only)
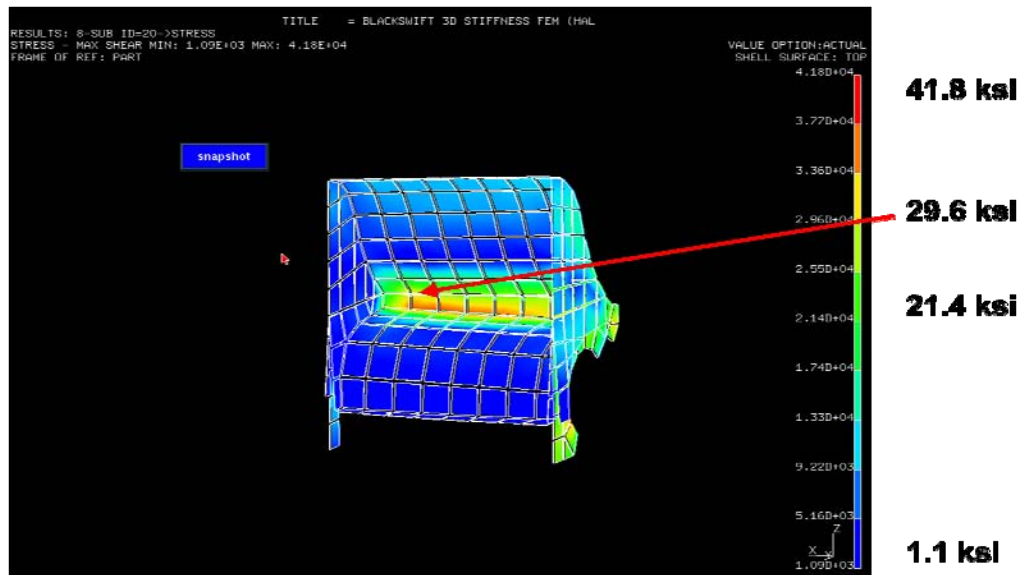
Figure 28. Fuel Containment System Component Resultant Von Mises Shear Stress (Thermal Loading Factors Only)
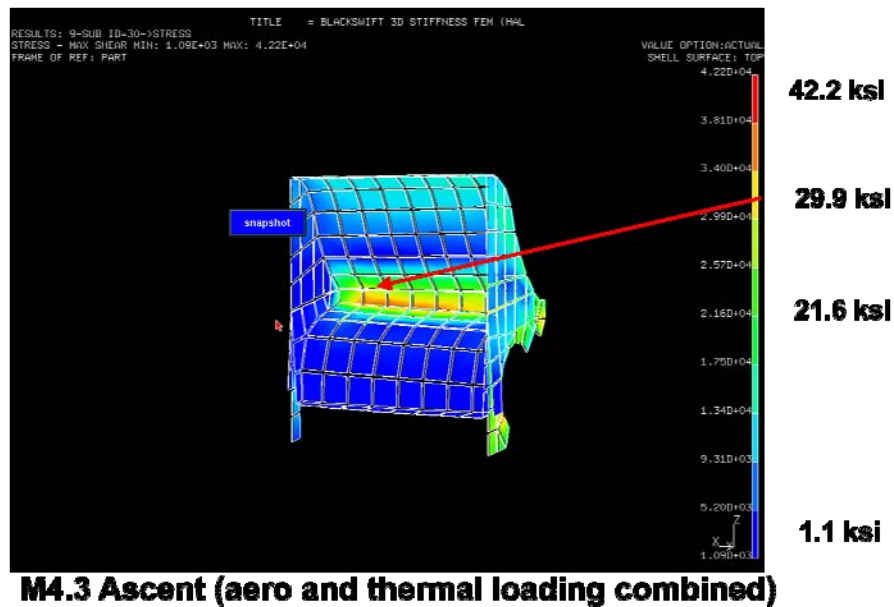
Figure 29. Fuel Containment System Component Resultant Von Mises Shear Stress (Combined Loading (Coupling))

As a review of just some of these instances experienced during the relatively short tenure of the DARPA HTV-3X program, given enough funding and time to analyze a single design, it is theoretically possible today to account for these effects. Unfortunately, that is not always practically possible. The usual and most likely result is an overly heavy structure caused by conservative design in an attempt to make up for ignorance of the real stresses in the structure. During final design when it is necessary to make accurate predictions of life for further sizing refinement, the degree of difficulty increases further. This comes from two factors. Not only are there many competing failure mechanisms due to the many types of loading, but the material itself is changing as it oxidizes, embrittles, loses strength, creeps and redistributes load, etc. Consequently, both ends of the design and analysis spectrum need improvement.

4.0   ASSESSMENT OF HYPERSONIC STRUCTURAL PREDICTION AND SIZING TOOLS

4.1   LOADS DERIVATION

Service life prediction requires generating load spectra, but for hypersonic aircraft, the temperature at each load application point is important. For conventional aircraft, load spectra include all significant ground, maneuver, and gust loads that occur within a specified block of time, such as a 1000-hour block, but hypersonic aircraft will have to include thermal loads as well. These blocks model the various missions and the mix of missions the aircraft flies with each mission further divided into mission segments. Each mission segment, where applicable, includes loads for ground, maneuver, gust, and thermal; the number of occurrences of each load determined by criteria for ground, maneuver, and gust and the time spent in each mission segment. With hypersonic vehicles, each load will require a corresponding temperature because temperature governs the structure's material properties used to determine fatigue damage or crack growth.

The process for generating load spectra generally involves the creation of databases for ground, maneuver, and gust conditions. These databases span a grid of applicable variables defining the vehicle loading state, such as fuel, cargo, and/or stores weight, altitude, Mach number, sink rate, taxi speed, runway type, control system mode, load factor, roll rate, flap and/or speed-brake setting, gear up or down, etc. For each loading state, the database contains a set of loads for the entire vehicle. Some of these loading conditions will represent a single event and others will represent several statically analyzed points during a time-history, such as a roll maneuver, which has loads peaking at different times in the maneuver for different loading components (i.e. inertia or aerodynamic) and at different locations (i.e. wing root, aileron or flaperon attachment, or wingtip). For hypersonic vehicles, temperature will also be a key variable which will require an increase to the size of the already sizeable database.

An additional problem, especially for hypersonic cruise aircraft, may be structural creep and its effect on loading conditions after a number of hypersonic flights have been performed on the airframe. The structure of a hypersonic cruise vehicle will spend much of its operational life at elevated temperatures and, depending upon the structural material, it may be challenging to maintain the airframe within dimensional tolerances due to thermally accelerated material creep. While most service life prediction programs determine inspection intervals for crack propagation, for hypersonic aircraft structures subject to creep, inspection intervals may be required for checking dimensional tolerances for aerodynamic and propulsion efficiency and for control surface and door clearance/fit. In addition, the coupling of material oxidation into a function that dictates the mechanical property degradation within the high temperature phase of flight is a very imposing challenge. Naturally, the calculation of the revised loading due to these stiffness changes, from mechanical property reductions due to oxidation effects, would represent a key element of the service life capability of the airframe.

Acoustic pressure loads, which can rapidly result in sonic fatigue of panels subjected to this high frequency, high-pressure loading, are more of a problem on hypersonic vehicles than on conventional ones because of the increased integration of propulsion and aerodynamics. The need to incorporate inlets and nozzles in the overall aerodynamic design exposes more of the vehicle to engine generated acoustic loading. Some vehicle designs are more susceptible to noise transmission through the airframe, with buried engine designs being the most critical.

Hypersonic aircraft are flexible vehicles subject to deflections caused by both external loads and temperature gradients that act to change the aerodynamic loading distribution. This redistribution of static load is included in aerothermoelastic effects. It is possible, through aerothermoelastic effects for a vehicle's trim and stability characteristics to change substantially, possibly even to the point the vehicle would become uncontrollable, so this is an important effect to include during structural design.

Including the effects of thermal deformations are relatively simple, since the thermal deformations are an incremental change on the unloaded vehicle geometry similar to wing twist or camber or to control surface deflections, but the aeroelastic effects are more difficult to incorporate. These require generating a transformation to convert rigid loading distributions to elastically flexible loading distributions. The transformation requires a linear Aerodynamic Influence Coefficient (AIC) matrix and a linear Structural Influence Coefficient (SIC) matrix or stiffness matrix.

Several aerodynamic theories allow generating AICs for the Mach number range required for hypersonic vehicle load cases, which unfortunately for the design of skin panels and other details, are all flat-panel models suitable only for calculating changes in pressure differential across major structural components, such as wings, empennage surfaces, or lifting fuselage bodies:

- Vortex (steady) or Doublet (unsteady) Lattice – subsonic flow
- Commercially available software – supersonic flow
- Piston Theory – hypersonic flow

Certain commercially available codes provide modules for generating AICs using these theoretical approaches and the means to generate a SIC using FEMs. However, with hypersonic aircraft, it will be necessary to generate multiple SICs to represent accurately the reduced stiffness over the entire airframe at elevated temperatures. Accurate temperature distributions over the entire vehicle structure will be necessary to account for the local thermally induced stiffness variations. With these tools, including structural flexibility effects, calculating steady trim and loading conditions are possible. Unsteady AIC calculations are also possible for flutter analyses, although these require many unsteady AICs for a range of reduced frequencies at each Mach number.

LM Aero uses commercially available Finite Element Analysis (FEA) codes that provide for panel geometry input for AIC generation consistent for the Doublet Lattice and other code models, but requires different geometry input for Piston Theory models. This poses a substantial difficulty if the aerodynamic box meshes are to remain consistent for aerodynamic box based inertia distributions, unless the user is prepared to generate another set of inertia distributions for hypersonic conditions. Making the geometry input in FEA codes for Piston Theory aerodynamic models consistent with that for Doublet Lattice and commercially available code models would reduce the labor and the likelihood of errors when analysts would like consistent panel geometries for all aerodynamic models.

One commercially available code provides a three-dimensional aerodynamic modeling tool that will generate steady AICs through hypersonic Mach numbers; however, it will currently not model all vehicle configurations. Surface panels must lie within the local Mach cone, which fixes a maximum analysis Mach number, angle of attack, and angle of sideslip combination for any given

aircraft configuration, since all of combine to determine whether the individual surface panel lies within the local Mach cone. The generation of AICs for three-dimensional panel geometry can be a time-consuming process because some schemes require a separate solution for each infinitesimal variation of each panel's local angle of attack to generate the AIC element by element.

In addition, current aerodynamic load prediction tools have shown to provide poor results for low aspect ratio aero surfaces. Specifically, the pressure distribution (ΔCp) in the chordwise direction shows a very unrealistic character when subject to a small angle of attack. In general it has been observed that the ΔCp is highest at the leading edge as expected for a small angle of attack. However as you move aft along the chord ΔCp moves from positive to negative and back again, continuing to "oscillate " between positive and negative with decaying amplitude until you reach the trailing edge. This leads to unrealistic pressure distributions and consequently poor $C_{L\alpha}$, $C_{M\alpha}$, and aerodynamic center estimates. This also gives low confidence in the unsteady aerodynamic loads used to predict flutter, divergence, and ASE margins. This is a significant issue in design of hypersonic vehicles which can lead to incorrect stress/strain distributions and flutter/divergence margins and ultimately to incorrect structural sizing. One could expect anything from poor vehicle performance as aero surfaces deflect more that predicted to outright structural failure from static overload, divergence, or flutter within the flight envelope.

4.2   AEROELASTIC EFFECTS
Numerous obstacles are seen in the prediction of aeroelastic effects in hypersonic airframes. Current aerodynamic load prediction tools have been shown to provide less than adequate results for low aspect ratio aero surfaces. Specifically, the pressure distribution (ΔCp) in the chordwise direction shows a very unrealistic character when subject to a small angle of attack. In general, it has been observed that the ΔCp is highest at the leading edge as expected for a small angle of attack. However, as you move aft along the chord ΔCp moves from positive to negative and back again, continuing to "oscillate " between positive and negative, with decaying amplitude until you reach the trailing edge. This leads to unrealistic pressure distributions and consequently poor $C_{L\alpha}$, $C_{M\alpha}$, and aerodynamic center estimates. This also gives low confidence in the unsteady aerodynamic loads used to predict flutter, divergence, and ASE margins. This is a significant issue in the design of hypersonic vehicles which can lead to incorrect stress/strain distributions and flutter/divergence margins and ultimately to incorrect structural sizing. One could expect anything from poor vehicle performance as aero surfaces deflect more than predicted to outright structural failure from static overload, divergence, or flutter within the flight envelope. Effort should be given to understanding why these tools are predicting this type of response and possibly "correlating" these results with test data and/or higher order prediction tools.

From the LM Aero perspective there are two commercial packages capable of hypersonic aeroelastic predictions with an accurate aero model formulation and in using Piston Theory. Piston theory is a simplified aerodynamic method that dates back to the 1950s but apparently has been validated in the Mach range of 2.5 to 7. Piston theory is similar to strip theory and does not consider spanwise interactions. One of the commercially available tools used by LM Aero is a linear frequency based method and handles airfoil thickness and bodies.

LM Aero experience gained from the Fast Access to Space Technology program, or FAST, using its suite of commercially available tools was that flutter dynamic pressure went up steadily as

Mach number went up (the transonic region was critical for the LM Aero case) (Figure 30). However, a cold structure protected by TPS was assumed, so stiffness reductions due to elevated temperatures were not an issue. For metallic structures at elevated temperatures, however, stiffness properties are reduced considerably at elevated temperatures. These stiffness reductions need to be factored into the aeroelastic analysis. With this premise it is anticipated that the same trend would hold for a metallic hot skin structure. This is not a limitation of any of the packages but strengthens the position that a fairly decent representation of the temperatures on the structure needs to be estimated and then temperature dependent properties of the materials are to be considered in the mode shapes. For time-domain, CFD-based aeroelastic predictions, the required time-step must be very small due to the high-velocity, and thus run time becomes a concern, as well as solver stability.
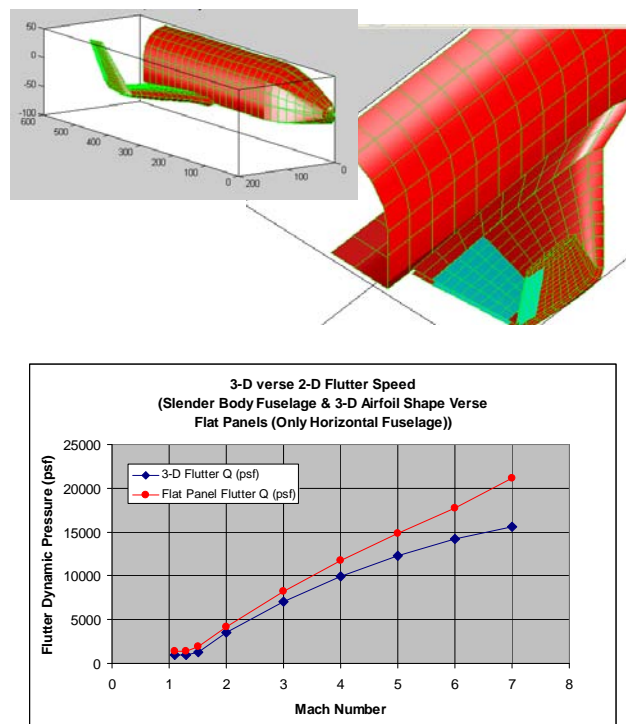
Figure 30. Illustration of Hypersonic Aeroelastic Analysis in the FAST Aeroelasticity Program

From discussion with NASP engineers, correlation of analytical flutter results with test data at hypersonic speeds is a big need. In addition, in other interviews with NASP engineers, flutter was not predicted to be an issue at hypersonic speeds, however, divergence was hard to predict (NASA Langley Research Center (LaRC) was known to be predicting a body freedom flutter mode at high supersonic speeds, not hypersonic speeds). And, these methods did rely on piston theory aerodynamics.

It seems that a major obstacle is how much do we trust our analytical tools at hypersonic speeds since test data is so limited? Along these lines, NASP engineers related a story that a high supersonic clamped, flat plate aeroelastic wind tunnel model was very difficult to correlate

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analytically. It turned out that the only way test results could be matched was by tuning boundary conditions. We really need to understand what are the limitations of our aeroelastic analysis codes at hypersonic speeds. To add to this, in further discussions with NASP engineers, the modeling of fuel slosh was a major issue and one that the analysts considered a big program challenge.

All of these frequency domain aeroelastic tools rely on a modal set that remains fixed during the flutter analysis (i.e. linear structures). Any non-linear structural interaction due to thermal heating will not be captured.The impact of not only thermal stiffness reductions but also the impact of thermal gradients on both dynamic aeroelasticity and static aeroelastic deformation needs to be taken into account. The interaction of aeroelasticity and thermal behavior (aerothermoelasticity) needs to be better incorporated into our production analysis tools and validated against test data.

As hypersonic vehicles were designed, from the LM Aero experience, for static aeroelastic effects in the transonic range where the lift curve slope was the highest, then at very high Mach numbers where the lift curve slope was much lower, the aeroelastic effects would be anticipated to be minimal. This in part tends to explain why hypersonic aeroelastic effects have been relatively ignored in the past. Another main reason in support of this design precept is that at hypersonic Mach numbers, dynamic pressure is relatively low, compared to the maximum dynamic pressure of the flight envelope. Dynamic pressure is a key driver of aeroelastic (static or dynamic) effects since it is somewhat of a measure of the available energy in the flowfield.

If the hypersonic vehicle were primarily structurally designed for static aeroelastic effects in the transonic range where the lift curve slope was the highest, then at very high Mach numbers where the lift curve slope was much lower, the aeroelastic effects would be considered to be minimal. The effects may be different and possibly less pronounced - partly because of this. However at higher Mach numbers even though aerodynamic effects may be reduced the heat generated during flight causes reduction of structural modulus and stiffness. This can lead to reduced flutter margin, divergence margin, and worsening static aeroelastic effects, etc. However, hypersonic aeroelastic effects - static or dynamic – should not be ignored even though the transonic region effects drive the overall vehicle design.

Aeroelasticity analysis relies on aerodynamic influence coefficients which define how the pressure at one location on the structure changes with deflection on another part of the structure. As the Mach number increases the Mach cone angle gets "steeper". Since air on one side of the cone isn't influenced by air on the other side of the cone there becomes less coupling between different locations of the structure (i.e. when one part of the structure deflects and disturbs the air this disturbance cannot create a pressure disturbance at structural locations on the other side of the Mach cone).

However, this was not the case with NASP where at hypersonic speeds, dynamic pressure was high. This is not to say that hypersonic aeroelasticity was ignored during NASP (in fact, a very large flutter effort was planned). In talking with NASP engineers the case was that flutter was not analytically predicted to be critical at hypersonic Mach numbers. The transonic region has traditionally been where flutter is most critical and this was also the case with NASP.

The general question as to if there is some upper Mach number aeroelastic effects are to be ignored was entertained. In short, aeroelastic effects are a combination of dynamic pressure and Mach

number.   Generally, at really high Mach numbers a vehicle structure will be at low dynamic pressure. It is apparent that the determination of just what is the dynamic pressure at high Machs and how does that compare to max envelope dynamic pressure is the underlying question. Again, this was not the case for NASP which flew at high dynamic pressure at hypersonic speeds.

In addition, interaction of fuselage modes (which are generally low frequency due to LM Aero past experience in long slender fuselage design) with flight controls is an issue. In fact, fuselage stiffness may need to be increased to achieve required flying qualities, resulting in the customary weight penalty. Generally, flutter and static aeroelastic phenomenon are not life issues. These are "ultimate conditions". What are life issues is buffet (e.g., Mach buffet), acoustics, and potentially limit cycle oscillation. These aeroelastic issues pose high criticality thereby causing degradation in the cyclic life of the material due to being at high temperatures for sustained periods of time (the effect on material S-N diagram of elevated temperatures).

Unsteady aerodynamic predictions at high Angle of Attack (AOA) and high Mach due to highly non-linear flowfield features pose the potential for buffet, and particularly on vertical tails. If higher order methods such as Navier Stokes analyses are being used, then Solver stability and Grid Deformation capabilities can be an issue.

Another area of concern is how would a hypersonic vehicle be Ground Vibration Tested, or GVT'ed. During heating the frequency and damping characteristics may change considerably as TPS panels, for instance, expand and slide over one another.   The conduct of a GVT and the subsequent capture of these changes to stiffness and damping under elevated temperatures pose significant challenges.

Effort should be given to understanding whether industry's current aerodynamic load prediction tools are predicting accurate response for low aspect ratio surfaces and possibly "correlating" these predictions with test data and/or higher order prediction tools. As far as the concerns in the area of incorporating aeroelastic effects the primary areas of concern are seen as, and especially as seen as limitations in currently available codes are:

1. The ability to predict unsteady aerodynamic loads of low aspect ratio surfaces
2. The ability to predict heating effects on structure, and the ability to predicted structural material behavior under the influence of thermal effects.

4.3   FLUTTER PREDICTION

Flutter analysis of hypersonic aircraft is more difficult than for lower speed aircraft because of the wider Mach number range that requires more AICs and also due to the temperature variations that require multiple SICs. Note that a localized high temperature on the airframe that would not cause a static structural problem could reduce the effective stiffness of a structural vibration mode sufficiently to cause a flutter or stability problem. The lack of experience with hypersonic vehicles causes these issues. Theoretically, any vehicle will flutter, if, at constant altitude, it accelerates to a high enough Mach number. Generally, hypersonic vehicles have their lowest flutter margins in the transonic flight regime. Yet, analysis at higher Mach numbers is still important. The heat generated during flight at higher Mach numbers causes reduction of structural modulus and structural stiffness. This can lead to reduced flutter and/or divergence margin. Practical experience with

supersonic and not hypersonic vehicles indicates that the critical design regime is within the transonic zone where the aerodynamic stiffness is highest. Because of this, it can be difficult to motivate experienced flutter engineers to explore the entire hypersonic flight envelope when it seems unlikely flutter would be a problem beyond transonic Mach numbers.

Flutter mechanisms generally tend to be most critical in the transonic regime. Body Freedom Flutter (BFF) and panel flutter are no exception. However, without further analysis, they can't be ruled out as issues in the hypersonic regime. It may be possible for a set of unique conditions to arise in any given vehicle design flight regime to result in significant aeroelastic issues in the hypersonic regime. Hypersonic and high supersonic vehicles were and will be susceptible to BFF. This was true for the SR71 & YF-12 vehicles. Typically, this involved rigid body 'modes' of the vehicle such as short period mode coupling with one or more vibration modes of the vehicle to create the flutter mechanism. For hypersonic vehicle design this is generally a body or fuselage bending vibration mode.

Having a BFF mechanism within the flight envelope or with insufficient margin is a very real possiblity. Resolving this issue may require an active control system separate from the typical flight control system. While this type of solution has been accomplished in the past it is by no means commonplace and the Technology Readiness Level (TRL) level for this type of technology is very low at this point in time. In addition, the effort required to accomplish this is not a thoroughly understood item.

Typically, BFF involves rigid body 'modes' of the vehicle such as short period mode coupling with one or more vibration modes of the vehicle to create a flutter mechanism. For the hypersonic vehicle this is generally a body or fuselage bending vibration mode and this is expected to be an issue for a similarly flexible metallic based Mach 5.0-6.0 vehicle. If the flutter mechanism has insufficient margin the resolution may become difficult. Additional stiffness required to obtain sufficient margin may come with too big of a weight penalty and active control techniques may have to be considered which come with their own set of issues.

Body Freedom Flutter could be a problem for hypersonic vehicles depending upon the individual vehicle geometry and it affects aircraft with predominately a single beam-like structure, either high-aspect ratio flying wing configurations or long, thin low-aspect ratio wing-body shapes when the natural frequencies of the rigid-body and elastic degrees of freedom are sufficiently close together. Some hypersonic wing-body designs fall into the latter category and could have body freedom flutter issues.

Panel flutter of skin panels will be a design consideration for hypersonic aircraft that will likely determine the allowable combinations of skin thickness and spacing for skin stiffeners. The experience gained through supersonic vehicles, as opposed to hypersonic vehicle experience, dictates, again, that the most critical speeds will probably not be hypersonic unless temperatures at hypersonic speeds substantially reduce the panel bending stiffness. First experienced on the German V-2 rockets of World War II, panel flutter was seen to occur in supersonic flows and tension in the panel. This was attributed to either a pressure differential across the thickness of the panel or from a mechanical load carried across the length of the panel which increases the flutter speed by effectively increasing the bending stiffness. Compressive loads, however, reduce the

flutter speed by effectively decreasing the bending stiffness.

In hypersonic vehicle airframe analyses panel flutter needs to be checked and particularly for thin metallic skins. There is a NASA report (Section 8.0 References) that has some hand methods for panel flutter. During the NASP program, panel flutter of the TPS panels was very hard to predict because of the complexity of the TPS boundary conditions while the panel flutter methods of the time assumed very simple boundary conditions (such as a panel simply supported on all four sides). Panel flutter predictions need to include generality of boundary conditions. Moreover, panel flutter was reported on the X-15 flight test program which led to continued NASA studies of panel flutter at elevated temperatures.

4.4 ACOUSTICS AND STRUCTURAL DYNAMICS
4.4.1 Tool Limitations
Acoustic sources within the hypersonic flight regime include turbulent boundary layer, Shock Boundary Layer Interaction (SBLI) and propulsion system noise. SBLI will drastically increase the OASPL because of the step-increase in pressure, oscillating near panel resonance.

In general, there is a gap in the current state of the art ability to predict an acoustic spectrum at hypersonic speeds. LM Aero reviewed the list of obstacles foreseen to be limiting the prediction of accurate acoustic sound pressure effects in hypersonic airframes from both external and internal sources. Questions entertained were:

- Are certain factors not formulated in codes in use today?
- Are there other limitations in existing analysis packages for high level acoustic behavior generation?

CFD++ and CAA++ offer capabilities to predict aero-acoustic noise sources and are well documented. Structure borne acoustics is a highly critical consideration as additional true acoustic sources (propulsion system noise) are thought to provide high impact to hypersonic vehicle design. Overall, more wind tunnel and flight measurements are definitely needed to validate hypersonic vehicle predictions. Much more heavily required is additional validation of currently available prediction tools – not just hypersonically, but in the supersonic flight regime as well.

One high priority area required in the acoustics prediction field is the development of sensors that can measure combined loads and response at elevated temperatures in the lab, wind tunnel, and flight. Actual wind tunnel and flight test data are limited and estimates and predictions of the in-flight environment are not as precise as desired; especially when combined propulsion, aero (shed vorticies, separated boundary layers, oscillating shocks etc), thermal, acoustic and vibration loads simultaneously stress the vehicle structure. The acoustic engineer needs to be able to better validate their numerical tools with test data. Thereby, the need to acquire validation data with advanced sensors is paramount.

From the LM Aero perspective acoustic loads combining with other flight loads during different flight regimes is considered the acoustic engineer's worst nightmare in accuracy of high acoustic sound level pressure prediction. The identification of failure modes and damage accumulation theories for the more recently developed ceramic fiber composites, metal matrix, carbon-carbon composites, and nanotube type materials will be especially critical on the list of information which

is not currently available.

Propulsion system acoustic noise, the "pure noise", is not considered to be such a critical factor in hypersonic flight. Rather, it will be the mechanical excitation of the airframe which will transmit acoustic energy to the vehicle interior. Engine placement and installation details will therefore be critically important. Turbulent boundary layer and oscillating shocks will excite the vehicle structure which will transmit acoustic energy to the vehicle interior. This is not to say that propulsion system noise will not be a factor, as the propulsion system will excite vibrations which will propagate, and/or create, additional noise sources that must be dampened internally through sound structural design.

CFD++ and CAA++ can predict the desired aeroacoustics loads but validation of the predictions with wind tunnel and/or flight measurements is needed. There is also an interdisciplinary component that involves combined loads due to high temperature, oscillating shocks, flutter/vibration and acoustics. Acoustic loads are generated during three flight segments: 1. at max power at takeoff 2. at max dynamic pressure during climb and 3. at sustained hypersonic cruise flight in the atmosphere. It is agreed that the prediction tools do not necessarily need to be CFD++/CAA++, but any coupled CFD/Acoustics suite should be adequate for the structural analyses phase.

Another huge need is the understanding of the material sonic fatigue life as a function of temperature. Much testing is required in this regard. Additional hypersonic vehicle phenomena that needs to be exhaustively studied are propulsion effects on acoustic environment, acoustic energy introduced by shear layer between the engine exhaust and freestream as well as the interaction between the engine exhaust and the boundary layer near structurally integrated portions of the propulsion system. NASA LaRC has developed some methods to analyze acoustic fatigue using NASTRAN for NASP. (ref. NASA/TM-2001-210838). In the course of the NASP program, panels were designed such that response to acoustic load would be kept below the "infinite life" stress levels. Significant testing was for correlation to pre-test predictions was also planned.

For boundary layer noise, a spreadsheet tool is available which is based on Wyle Research Reports WR-69-3 and WR-70-10, both of which were written by J.E. Robertson. Thus, the tool is called Robertson.xls. The spreadsheet based tool basically calculates fluctuating pressure levels (or sound levels) for transonic and supersonic boundary layers and is subsequently limited in the hypersonic range.

For the high-speed, high-temperature jet, a spreadsheet tool based on NASA SP-8072 is available. This is also available as a tool (SPJet) written by Peter Whitney in the Python programming language and it is believed that this is based on Saturn V measurements. However, it is uncertain if it's fully applicable up to hypersonic speeds. However, at those speeds the plume effects should be well aft of the vehicle structure except for the case of "scrubbing" exhaust. In LM Aero usage this tool was used in the early phases of the DARPA Falcon and HTV-3X programs. In general, validation of analytical tools against an aerodynamic database at hypersonic speeds is a huge need.

In summary, in the area of hypersonic acoustic environment prediction the leading deficiency, or deficiencies, in being able to accurately predict sonic fatigue issues analytically on a hypersonic airframe are listed as:

- to ability predict acoustic levels
- the capture of the large acoustic response requiring nonlinear analysis
- Ability to accurately model boundary conditions including preloading

4.4.2 LM Aero Acoustic Sound Pressure Generation

To further highlight foreseen gaps in the coupling of acoustic sound pressure induced effects with other hypersonic environment factors the manner in which LM Aero generates the acoustic sound pressure levels as required for structure detailed sizing is illustrated as follows. For the analyses the information required is as such:

1. The mission profile, with a preference of dynamic pressure (Q) versus time data.
2. The performance envelope plot with Mach Speed versus Altitude.
3. Several regions of interest: max level speed, transonic Q (Between M = 0.8 to 1.1), and lower right hand corner of the performance plot (translated to Q)
4. Engine information, starting with the location and number of engines, fan diameter(s), turbine diameter(s), fan and exhaust velocities.
5. Weight of the component of interest.

After acquiring the above information preliminary Power Spectral Density (PSD information can be developed from the equation listed in Mil Std 810 F or later, Method 514.5C2, Table 514.5CIII

$$W_0 = W_a + \sum_1^n W_j$$

$$W_a = a \times b \times c \times Q^2 = 10^{(0.60 - 0.0075 \times W)} \times 1.40 * 10^{-7} \times 1.0 \times Q^2$$

$$W_j = \left\{ \left[0.48 \times a \times d \times \cos^2(\theta)/R \right] \times \left[D_c \times \left(\frac{V_c}{V_r} \right)^3 + D_f \times \left(\frac{V_f}{V_r} \right)^3 \right] \right\}$$

$$= \left\{ \left[0.48 \times 10^{(0.60 - 0.0075 \times W)} \times 1.0 \times \cos^2(\theta)/R \right] \times \left[D_c \times \left(\frac{V_c}{V_r} \right)^3 + D_f \times \left(\frac{V_f}{V_r} \right)^3 \right] \right\}$$

$$V_r = 1850 \, ft/Sec$$

Where W_0 is:
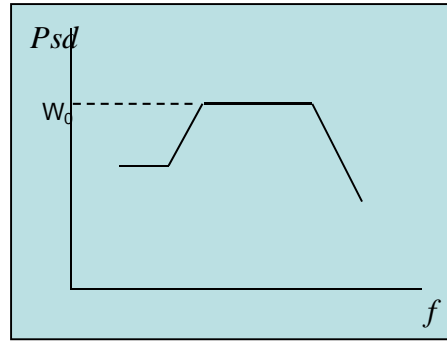
With:

a   is the platform / material interaction factor
b   is the proportionality factor between vibration and dynamic pressure
c   is the Mach number correction
d   is the afterburner factor
Q   is the dynamic pressure in psf
      and the following view illustrates the engine parameters:
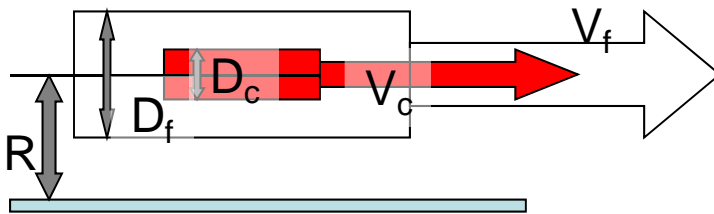
W_a is a function of the vehicle dynamic pressure, and W_j is a function of the jet engine(s).

A 3 dB factor for conservatism is usually added to the predictions on locations where supersonic shocks are likely to happen, which are loosely based on other supersonic fighters and bombers like B-58, B1, F-15 and F-16.

As for acoustic predictions, for the preliminary surrounding environment the published papers such as

1. Fluctuating Surface Pressure Characteristics on Slender Cones in Subsonic, Supersonic, and Hypersonic Mach-Number Flow, by H. Heller and A. Clemente Oct. 1974 NASA Contractor Report CR-2449
2. Study to Define Unsteady Flow Fields and Their Statistical Characteristics, Wyle Lab Report WR 75-1, May 1975
3. Boundary Layer Noise: Wyle Reports WR-69-3, WR-70-10, WR-75-1 (all by J. Robertson)
4. Propulsion Noise (high speed jet): NASA SP-8072, "Acoustic Loads Generated by the Propulsion System", NASA Space Vehicle Design Criteria (Structures), 1971

Normally, at this point, the dynamics analyst would take the acoustic prediction and verify the preliminary analysis with historical aircraft based data like Mahaffey-Smith acoustic – vibration, B-58 correlation charts, or other historical data base of similar mission aircraft. However, with hypersonic vehicles, due to the lack of actual historical flight test data, this step is normally not performed.

Traditionally, most of the dynamics focus is on the transonic phase, since it is generally assumed that the vibration magnitude during supersonic cruise is lower. Yet, it was known from the space shuttle program and other re-entry vehicles that high Mach number entry can be a very rough dynamic environment to deal with.

The secondary analysis is normally performed after the vehicle Preliminary Design Review (PDR) when the OML is locked, the internal primary structure is developed, and the analysis was based on the input using the PDR acoustic predictions. The primary dynamics tools used are Statistical Energy Analysis (SEA), and FEA. In both analyses, the basic primary structures and internal arrangements are defined (not locked), and both the external acoustic and internal acoustic (inlet) are coupled in as both correlated and un-correlated input. The two responses are then sorted to develop the maximum component vibration level and the internal structural loads.

The maximum component vibration levels are inputs to the vehicle environmental criteria document and are used to design the components and sub-system components. The internal structural load, which echo the structure's "aset", shall be Root Sum Squared (RSS) to the inertia relief loads for sizing the structure and the fittings. In most cases, for primary and most secondary structure, the dynamic environment loads are small as compared to the maneuver or inertia relief loads, but in cases where the light weight, large surface area components (aka acoustic receivers) or heavy equipment are on flexible structure, the dynamics loads can be a major part of the total loading.

The SEA/FEA based analysis is then compared to the early pre-PDR predictions, and if necessary, an envelope will be generated based on the post PDR predictions.

However, one limitation in this step of the SEA/FEA analyses is the coupling of the structural stiffness variation based on thermal cycling. Other commercial codes are seen as "stronger" in this area than other codes. Yet, only thermal "snapshots" with dynamic "snapshots" are made as opposed to a continuous coupling effect.

4.4.3   Final Dynamics Analysis
After the internal primary and secondary structures are "locked", a final updated SEA / FEA analysis that reflects the final OML, IML, and structure is then performed using the earlier acoustic predictions, and this will now incorporate any acoustic / wind tunnel data updates. The results are then used to update the latest structural criteria document and the environmental criteria document, and the information will be used to qualify the components.

4.4.4   Flight Level, Qualification Level, and Test Tolerance
The max predicted acoustic sound level is the flight level. To demonstrate design margin, all components are to be qualified to the qualification level, which is +3dB above flight level. Likewise, the test tolerance is usually set to +/- 3 dB above the mean. The +3dB margin normally

covers component acceptance test repeats (up to 3 times due to anticipated manufacturing problems and uncertainty) and one lifetime of flight operation. This margin also ensures that the component will not be tested below flight level during qualification testing.

Conversely, these margins are quite low when compared to those employed in space and missile programs, when the qualification to flight margin is normally set to +6dB.

Based on one recent supersonic program experience there was a 75% first time qualification failure rate due to the requirements of combined environment testing. In learning this from the supersonic regime and therefore extrapolating it to hypersonic cruise flight, components, sub-systems, and systems all need to function while being exposed to the indigenous elevated operation temperature and vibration environment.

4.5 AEROTHERMAL PREDICTION
Thermo-structural analysis of hypersonic vehicles requires an accurate characterization of the shock layer surrounding the entire airframe. Specifically, a high-fidelity simulation of the airframe flowfield (fuselage, wings, and control surfaces) provides the shear stress distributions and surface pressure required for structural analysis and the convective heat transfer rate to the hot structure skin. The evolution of advanced CFD models in the past 20-30 years now allows the solution of high speed vehicle flows to be obtained with relatively high levels of confidence for more realistic airframe geometries and flight conditions along the trajectory of interest. The enhanced solution accuracy and higher robustness of the algorithms that have come about over these years since the NASP program attribute their enhancement to the use of upwind differencing methods. The Aero analysts can now more accurately calculate the flow properties where strong gradients appear, as in with the phenomena of shock impingement around vehicle structure sharp corners.

Yet, the factors that can limit the applicability and/or confidence of numerical results can be attributed to the lack of adequate validated physical submodels in the codes, and especially for chemistry and turbulent flow mixing. CFD based modeling of high supersonic and hypersonic flows is usually accomplished using one of two general classes of codes: either Euler/boundary layer models or parabolized Navier-Stokes methodology.

In general, the Euler type code generates a body-aligned grid network around conceptual vehicle configurations using conformal transformations. A character based differencing algorithm is used to accurately model the flow around regions of high curvature as well as curvature discontinuities. The pressure distribution for structural loading is thereby solved but the solution for the viscous layer directly adjacent to the vehicle surface is required to accurately model the heat transfer rate and surface shear. This flow near the vehicle surface can be analyzed and determined through available boundary layer codes. However, the key challenging aspect in this part of the simulation for accurate surface shear and heat transfer rate determination is in providing accurate and appropriate inviscid flow properties at the boundary layer edge. Usually, the monitoring of the mass within the boundary layer is involved in order to determine the local edge entropy. Again, the accuracy or inaccuracy of the inviscid flow properties at the boundary layer edge is paramount and can lead to levels of either over or under conservatism in the eventual design. Parabolized Navier-Stokes equations represent the other principal approach for determining vehicle surface conditions for subsequent thermal analyses and prediction of reliable thermal field magnitudes.

One asset that Navier-Stokes equation methodology has is that it can be more reliably applied at higher altitude design concept vehicle ranges that Euler boundary layer models since viscous interaction effects that are associated with lower Reynolds numbers and thick viscous layers are computed automatically. Previously, many numerical difficulties were experienced in the regions of strong shocks. The overall robustness and accuracy of the codes enjoyed several enhancements over the years with "Roe's upwind algorithm" being implemented.

Of high concern is the estimation of heat transfer rate into a vehicle's skin from the external environment.   This is primarily composed of 1) the convective heat transfer from the air into the skin, and 2) the radiative heat transfer from the skin to its colder surroundings. The convective heat transfer rate is a function of the geometry, flight conditions, boundary layer state, and surface temperature. The radiative heat transfer rate is a function of the material emissivity and surface temperature. The difference between these determines the heat flux into the structure. The heat flux time history is combined with the vehicle thermal capacity to generate a time history of the surface temperatures.

4.5.1   Physical Factors Affecting the Heat Transfer Rate

There are several complex physical phenomena in hypersonic flows that affect the estimation of heat transfer rates. These are listed below to serve as an introduction to the rest of the discussion and they represent some of the inherent problems of hypersonic flight.

1) The boundary layer state (laminar, turbulent, or transitional) has a dominating effect on heat transfer rate over most of a typical trajectory. Turbulent flow gives higher heat transfer rates, but can be an overly conservative assumption at high Mach, high altitude conditions. It can also be overly conservative near the nose of the vehicle at lower Machs and altitudes. The best methods for estimating the boundary layer transition point are empirical correlations based on previous flight test data or analysis based on Linear Stability Theory (LST) or higher order techniques, calibrated to flight test data. Wind tunnel results from conventional (noisy) tunnels are generally considered too conservative. The surface roughness and presence of gaps and steps can also have a large effect on the transition location. As an example, Figure 31 displays the results of a study to evaluate some of the boundary layer transition criteria options in the LM Aero utilized BART thermal analysis code. All of the transition criteria are from the NASP era and are based on monitoring the momentum thickness Reynolds number divided by the local Mach number. All of them include some data from the reentry F transition test. The last one applies a correction to convert from conical to planar flow. For each criteria, the percentage drag reduction from all turbulent flow is shown. One of the strong conclusions from the results of Figure 31 is that there will be a lot of uncertainty about the effects of transition until better methods are applied or flight data is obtained. The good news is that there is a potential drag reduction due to delayed transition at high Mach numbers gaps, surface roughness, and their combined overall coupling effects will factor as well.
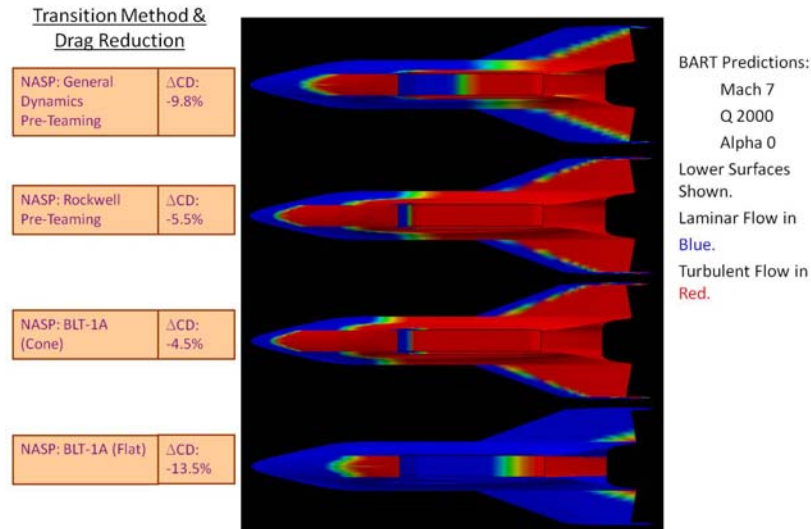
Figure 31. Boundary Layer State Predicted by Various Methods Shows Large Variation

2) The chemical state of the environment has a strong influence on heat transfer rate. As air gets hotter, it activates vibrational modes that absorb energy. At higher temperatures, oxygen molecules begin to dissociate, absorbing more energy and lowering the surface temperatures. Mach 7.0 flow is generally at the low end of the oxygen dissociation region. The degree of catalycity of the surface can also influence the degree of dissociation and the resulting heat transfer.

3) The intersection of shocks on boundary layers or on other shocks can generate regions of very high local heating. This is a common problem on cowls and wings, leading to local hot spots.

4.5.2 Vehicle Conceptual Design Phase Methods

In reference to an LM Aero hypersonic vehicle conceptual design phase, low order, approximate methods are used to estimate heat flux. These are often based on impact theory methods to get boundary layer edge conditions, such as pressure and Mach number. This information is then used by a flat plate boundary layer model to estimate skin friction and heat transfer rate (through a Reynolds analogy). There are many approximations inherent in such an analysis. As such, they are not considered the "final" heat transfer predictions, but they provide average acreage conditions. Nevertheless, since many vehicle characteristics are determined in this phase, it is desirable to avoid gross errors. Below is a list of some of the potential areas for improvement of these methods.

1) The adequacy of the basic pressure and heat transfer models needs to be verified by more comparisons with higher level methods, such as CFD. Typically, impact methods provide many model choices; narrowing this field down to the "best" ones would be desirable.

2) The estimation of boundary layer state is difficult for these methods. A limited number of empirical models are the only tools available. Additional standardized empirical models would be desirable. These need to be calibrated with flight test data to establish their adequacy and avoid an overly conservative, all turbulent, analysis.

3) Incorporation of chemical effects is also difficult for these methods. Impact methods typically provide a number of choices for pressure estimation, but only a limited number of these include equilibrium air assumptions. Both the impact and boundary layer methods need to be reviewed for proper chemistry effects, and wall catalycity models need to be incorporated.

4) A good estimate of the actual emissivity over the life of the vehicle is needed in order to avoid an overly pessimistic (degraded) emissivity value.

4.5.3 Vehicle Preliminary Design Phase Methods

During the preliminary design phase, higher order methods are brought to bear. On the analysis side this is primarily CFD and on the testing side are heat transfer tests in conventional hypersonic wind tunnels. Below is a list of some of the issues that arise from these methods.

1) The estimation of boundary layer state continues to be a problem for all analysis methods. CFD methods can provide input data for more sophisticated LST analysis methods, but experimental calibration is still needed. It would be desirable to invest in an effort to incorporate LST into a CFD code, rather than act as a post-processing step requiring iteration.

2) CFD methods can incorporate a variety of chemical models of the air. Perfect gas models will over-predict the heat transfer rates at high Mach numbers. Variable gamma models are better, but may become deficient at the stagnation point by Mach 7. Equilibrium air models can handle higher temperatures, but the chemistry may not be in equilibrium. For those cases, a reacting chemical model is called for; these are typically much more expensive to run. Investment is needed to quantify when the various models are needed. This could be accomplished by comparing them on a representative model at various conditions.

3) Reacting chemical models will also need to be employed to study heat transfer effects when flowing from a non-catalytic wall (ceramic) to a catalytic wall (metallic) to quantify the increase in heat transfer rate due to recombination. This may be small at Mach 7.0 but needs to be quantified to increase confidence.

4) CFD methods have turbulence models, wall function models, and grid resolution practices calibrated by low speed problems. These models should be tested against high Mach experimental data to verify their adequacy. One particular question is whether wall function will provide good heat transfer data at hypersonic speeds.

5) Hypersonic wind tunnel models can generally incorporate only a few heat flux gauges. This makes it possible to miss local hot spots. Testing with heat sensitive surfaces or IR photography can help in this regard, but more comparison and verification is needed.

4.5.4 Vehicle Detailed Design Phase Methods

During the detailed design phase, individual parts are designed and tested. More detailed CFD and FEM based heat transfer models are constructed, and arc jet testing is done on components. The importance of modeling local conditions and hot spots becomes more important. Below is a list of some of the issues that arise in this phase.

1) The local heating modifications in the vicinity of steps and gaps are frequently needed in order to design the parts to resist damage. Old empirical methods are frequently relied upon. Parametric tests using modern photographic instrumentation would be useful to update and verify this database.

2) Arc jet testing can often simulate the pressure and heat transfer rates expected in flight, but the chemical effects are expected to be different. This is due to the non-equilibrium flow in the arc jet and the evaporation of materials from the electrodes themselves. More investment is needed to understand the comparative conditions in arc jets and flight.

3) Arc jets typically allow only a small model size and have a non-uniform flowfield. Larger facilities would be desirable for more realistic testing.

4.6 MATERIAL INSTABILITIES

One of the big questions in going forward in hyperonic vehicle design is: Do we know the effects of the flight operating environment on the mechanical properties of the alloy or alloys selected for airframe design? In other words, do we know effects of temperature, sustained and cyclic loading, loading rates, moisture environments (incl. NaCl), hot air (partial pressures of nitrogen and oxygen), etc., on the material properties such that we can proceed with the design effort? At this point we cannot quantitatively predict the effects on the proposed material properties. The obvious answer is no and that is the basis of this section. For "low" hypersonic flight, that of within the Mach range of M5.0-7.0, Titanium alloys offer high potential for airframe design due to their high specific strength ratios as well as lower coefficients of thermal expansion rates as compared to nickel and cobalt based alloys. Based on this, the discussion of material instabilities will therefore be restricted to Titanium alloys.

4.6.1 Thermal Stability

Thermal stability is the ability of the material to withstand micro-structural, and thus mechanical property, changes due to its extreme flight environment. Thermal stability is often referred to as *bulk stability,* referring to those changes within the material. Bulk stability pertains to micro-structural changes that occur due to the combined loads/stresses, elevated temperatures, and exposure times. Cyclic temperatures and loads also come into play within this regime. Micro-structural changes can induce property changes - usually to the negative. This is why it is important to establish *bulk stability*.

The micro-structural instability can be initiated by precipitation of the Ti_3Al (α_2) and silicide phases. These are the phases that provide the good elevated temperature strengths. Normally, these stay in solution under normal heat treatment and operating conditions, but can precipitate during welding, improper processing, and excessive operating temperatures, which can lead to loss of properties. Subsequent use in a hot environment should refrain from exposing the material to temperatures in excess of 1100^oF for extended periods of time. Close examination of the bulk stability at these temperatures will be required to determine when the alloy becomes unstable. In other words "How high a temperature can we run it to, under what stress level, and for how long?" For answers to these questions additional testing will be required.

4.6.2 Oxidation

Near-alpha alloys are the best of all titanium alloys in terms of oxidation resistance. The high aluminum content increases the alpha phase and increases the beta transus. Oxygen is another alpha stabilizer. It can be absorbed into the alpha phase hcp unit cell thus increasing the alpha phase present. Near alpha alloys can not only withstand higher operating temperatures for longer times than other $\alpha-\beta$ and β titanium alloys, but they can do so without incurring high levels (thickness) of scale and alpha case, and subsequent/detrimental property loss. Oxidation and its effects on material properties pertains to the subject of *surface stability,* a term used to describe what happens to the immediate surface, but also to the adjacent subsurface – both of which can deleteriously effect mechanical properties.

Typically, when exposed to increasing temperatures for any length of time in air, the material will first exhibit a characteristic light straw to yellowish-gold color (approx. 850-950°F), then a violet to dark blue (1000-1100°F), and then a dark brown to dull gray (>1100-1200°F). Sustained use above 1200°F will cause the alloy to turn gray in color and very likely impact mechanical properties to the negative. With that said, the alloy should suffice in the design environment of 950-1000°F for 50 to 100 hours and short time exposures to 1100°F. The latter would need to be defined in subsequent testing.

4.6.3 Oxidation Resistant Coatings

A plethora of oxidation resistant coatings have been developed for retarding the formation of scale (sub-scale) and alpha case on titanium alloys and titanium intermetallics (a.k.a. titanium aluminides). The engine companies have led the effort and there is a significant body of work that shows coatings do work (Figure 32). Some of the more salient coating systems include noble metal coatings, aluminized coatings, ion implanted titanium or titanium aluminide coatings, and sol-gel coatings to name just a few.

The NASP program showed it possible to protect the substrate from harmful oxidation using a variety of materials (Si, Al, Cr, Ni), lay-up schemes, and methods of application (plating, Chemical Vapor Deposition (CVD), etc.). However, to date, the author knows of very few coatings that can successfully demonstrate maximum substrate protection (from oxidation/weight gain) without a concomitant knockdown in fatigue strength. This is a critical point should a coating be required for hypersonic flight in the near future. If a Materials and Processes (M&P) group were forced to pick a coating today the best choice would be a platinum or platinum-aluminide coating. The latter has demonstrated itself in laboratory testing to mitigate oxide growth (weight increase), and actually increase static, fatigue and creep strengths over baseline material conditions up to 1100°F.

Should it be determined that oxidation protection be needed, there are basically two approaches to invoke: 1.) modify the base alloy chemistry, or 2.) develop a durable, compatible oxidation resistant coating. Depending upon what has been done before it would be prudent to explore all options before committing to one or the other. One would want to minimize the need to re-qualify an alloy based on simple changes to its melt chemistry. If a coating is used, application will have to be after all parts are made to assembly and then coated. Alternate would be to employ getter material in mechanical fastened joints. All primary processing (i.e, machining, forming, cleaning, welding, drilling, etc.) would have to be performed prior to coating. In other words, the coating would go on as last step.

Ion Plating Materials	Exposure Temperature		Weight Gain Rate $(mg/cm^2/hr)$
	^{o}C	^{o}F	
No Coating	590	1100	6.9×10^{-2}
Gold	430	800	2.2×10^{-4}
Gold	480 (a)	900	2.6×10^{-3}
Platinum	590 (a)	1100	1.2×10^{-3}
Tungsten/Platinum	650	1200	3.3×10^{-4}
Tungsten/Platinum	700 (a)	1300	1.7×10^{-3}

(a) Highest temperatures under which no spalling or loss of the coating
was detected after 500 hours. Source S. Fujishiro, and D. Eylon.

Figure 32. 500-hour Weight Gain of Ti-6242 Alloy in Air

<u>Corrosion Resistance</u> - General corrosion in reducing environments is not well documented, but is believed to be similar to $\alpha - \beta$ titanium alloys (Ti-6-4). Crevice corrosion is likely less than that of Grade 2 titanium.

<u>Hot Salt Stress Corrosion</u> – Near-alpha alloys tend to be more susceptible to hot salt stress corrosion cracking than say other titanium alloys such as alpha-beta and beta titanium alloys. However, Ti-6242 is a bit more resistant to hot-salt cracking than Ti-8Al-1Mo-1V (an alternate near-alpha alloy) and Ti-6Al-4V (an alpha-beta alloy). The alloy is only slightly susceptible to aqueous chloride stress corrosion cracking. The triplex annealed condition appears to offer more resistance than the duplex annealed condition.

Ti-6242 can be employed in high temperature applications provided precautions are taken to prevent chloride contamination and contamination due to soft metals. Precautions would be on the same level as that for the beta titanium alloy, B120VCA, used on the SR-71.

<u>Soft Metal Compatibility</u> – Titanium alloys in contact with soft metals (cadmium, silver, lead, mercury, or compounds of the same) may contribute to embrittlement of the base alloy, and lead to pre-mature failure of the part or assembly. Like hot-salt stress corrosion, the conditions for such are dependent on the presence of a tensile stress state (applied or residual), presence of a soft metal (intimate contact), and an elevated temperature. Many of these issues were addressed and resolved during the SR-71 program and are in place today to prevent such occurrence.

As with the SR-71 Blackbird it will be important to address compatibility of titanium alloys with other non-metallic materials with halogen content sufficient to embrittle the base metal. For this reason all materials not already tested for titanium compatibility at high temperatures will need to become qualified to mitigate such phenomena.

4.6.4 In-House Use Experience
The U-2 program introduced the Ti-6242 alloy (with Silicon) at LM Aero for use in the tailpipe back in the 1989-1990 timeframe (Figure 33). This was the approximate time when the program moved to the new General Electric engine and the application has been very successful. The design is a multi-piece sheet metal assembly and detailed parts are comprised of roll formed, hot formed

and sized, subsequently assembled together using spot seam and fusion weld methods. Operational stresses are fairly low, and aside from the elevated temperature requirements (910-920°F), the material is not really being pushed. The assembly is designed to last 10,000 hours and it has PDM inspections every 3600 hours. Repairs generally consist of grind-out and re-weld of cracked spot welds which typically occur on major attachment boss/fitting (about mid-length). A photo of the assembly is shown below. There one can see the areas cleaned using scotch-brite pads. This was done to permit inspection of the welded joints. Larger repairs are needed when the cracked welded approach 30-40 cracks. At that point the center section (about 12 inches to both sides of the center attachment fitting) is cut away and replaced with a newer section. The new section is welded in-place and the tailpipe is placed back into service.

The above well illustrates the fact that where modeling capability of oxidation effects on candidate hypersonic vehicle airframe materials are understood to be of low TRL, routine inspection, and hardware replacement are still substituted but at great program cost. The coupling of material instabilities within the vehicle level modeling and analyses practices of LM Aero are not there at this time, representing a great void in the full understanding of coupling and complex loading upon hypersonic airframe structure.

Figure 33. Side View of U-2 Tail-Pipe

4.7 HYPERSONIC AIRFRAME MODELING ISSUES

In the modeling of hypersonic vehicle airframe structure there are issues in the treatment of thermal stress considerations. And, these issues are worthy of careful treatment in both the analysis and test points of view. It's easy to get wrapped up in analysis and worry about high stresses from a model, but without doing some tests to see actual failure modes the analyst doesn't know if the effects are real. To illustrate, the X-15 program performed a vehicle wing box test in which the mechanical loads were applied at room temperature and at elevated temperature. The failure load of the box in both cases was about the same!   Perhaps there was something unique about the design causing this occurrence, but engineering intuition would say that the analysis would predict a much lower strength at elevated temperature. So, the question is, is the analysis correct?

Practical airframe design is typically based on a global finite element model representing the entire vehicle. This model is used to obtain internal loads on the major structural members, as well as for overall dynamic and aeroelastic analyses. Structural details are sized using conventional stress

analysis methods (either automated software or hand analysis) using the member loads from the FEM. A large number of load conditions (100s or 1000s) are usually considered, which makes anything more sophisticated than linear static analysis impractical in realistic design situations with tight schedules. Local areas of concern can be addressed using more detailed stand-alone models and more complex analyses (e.g. nonlinear) where the extra time and effort is warranted.

It has already been discussed how hypersonic vehicles are subject to extreme thermal and acoustic environments which pose additional challenges to the structural analysis task. High temperatures and temperature gradients can cause severe thermal stresses, material strength degradation, and time dependent deformation (i.e. creep). While these phenomena can be represented in isolation with sufficiently detailed special purpose models, there are difficulties in addressing multiple phenomena simultaneously at the global design level with confidence. What is needed are techniques and methods that can be applied in an efficient manner during the design process without excessive analytical burden.

In the structural design process, the material assignment and element sizing (areas and thicknesses) are assumed based on past experience or limited hand analysis. The resulting global structural FEM is used to generate internal loads in the primary members, which are less sensitive to sizing changes than member stresses. For example, given a fixed external wing bending moment and wing dimensions, the cover running loads (Nx, lb/in) are approximately independent of cover thickness, whereas cover stresses vary drastically with thickness (approximately inversely proportional). Therefore, approximate initial sizing may be adequate for structural analysis, but not for the thermal analysis. The thermal analysis model may have to wait until an initial structure is sized using only mechanical loads for meaningful results. The design process could be improved dramatically if the thermal and structural tools and models were integrated such that these iterations were easy and quick.

4.7.1   Temperature Definition and Integration

When a structural model uses temperatures obtained from a separate thermal model, the issue of data transfer between tools and models becomes important. Ideally, the mesh densities of the structural and thermal analysis models should be similar, but there are legitimate reasons for differences. Features important for the structural analysis may not be important for the thermal analysis and vice versa. There may be different assumptions or requirements for lumping various areas for each discipline. When the mesh densities are significantly different, judgment must be used to apply temperatures to the structural model. The simplest approach would be to assign the temperature of the nearest thermal data point to each structural point. However, the simplest approach is not necessarily the best. More accurate behavior may result if an "equivalent" temperature is smeared over the structural points. This may require an analytic interpolation function to obtain reliable and repeatable results.

As a related example, consider the problem of applying aerodynamic pressures from a CFD model to the FEM of a wing. Even if the local pressure on every FEM point matches the pressure at the corresponding CFD point, the total external load (shear and bending moment) at the wing root may be under estimated.   The structural model may have gaps in the surface that were considered non-structural. The load on these omitted areas is then lost. Also, if there are significant differences in mesh density between the CFD and structural models, the orientation of the element

surface normals may be different enough to cause different load resultants even if the local pressures are the same. This could happen on a leading edge, where the pressures are high and vary rapidly, whereas the FEM may be relatively coarse. Therefore, in load interpolation the objective would be to maintain total resultant load rather than only match pressures in a point-wise manner.

In thermal structural analysis, a similar situation may arise. Assume the temperature distribution in a wing spar web is approximately parabolic (high at the upper and lower surfaces, low in the middle of the web).   If the structural model was very coarse, say only one element through the thickness, what value of temperature should be assigned to that element to produce the same overall effect (e.g. interface stress between hot skins and cool web) as the actual temperature gradient? If the actual temperature at the element centroid and corner points are used, that would not give the same effect as the parabolic temperature distribution. Something closer to the average between the outer surface and web middle temperatures would probably give better results. Perhaps detailed stand-alone FEM studies could be performed to determine generalizations such as these that could be used in practical modeling applications. The mechanical loading conditions typically represent critical points in the vehicle's flight envelope. However. the thermal conditions may not correspond to these specific points. Often, a few thermal conditions are available to superimpose on many flight load conditions to determine the critical loads. Ideally, the thermal conditions should correspond with the mechanical conditions. Even if this is possible, it may not be practical if the number of combinations becomes excessive. Investigation into combined thermal-mechanical loading could help to define rational criteria for combining effects without being overly conservative.

4.7.2 Mesh Fidelity
The structural Finite Element mesh must be sufficiently detailed to capture both the temperature gradients as well as the resulting thermal stress gradients (Figure 34). Thermal stresses typically have steep gradients, so a mesh that is sufficient for internal loads analysis will probably be too coarse to model the thermal stress distribution. Having a more detailed model allows the temperatures to transition more smoothly, which in turn may reduce the stress predictions (i.e. minimize artificial stress peaks due to modeling).

4.7.3   Joint Modeling With Flexibility
In a linear elastic finite element analysis, very high thermal stresses are often predicted at structural interfaces, such as joints, which may not be realistic and lead to over-conservatism in structural sizing practices. Real-life effects that alleviate these stresses may not be captured in a linear elastic model, such as local yielding, joint slop, or out-of-plane displacements. One approach would be to incorporate additional detail in the model to represent these effects, and possibly perform nonlinear analysis as well. At the other extreme, the linear analysis results could simply be adjusted in some manner (based on analysis, experiment, or judgment) to account for these effects. An intermediate approach would be to modify the model in an approximate manner to capture the gross effects, but maintain the same mesh density and continue with linear analysis.
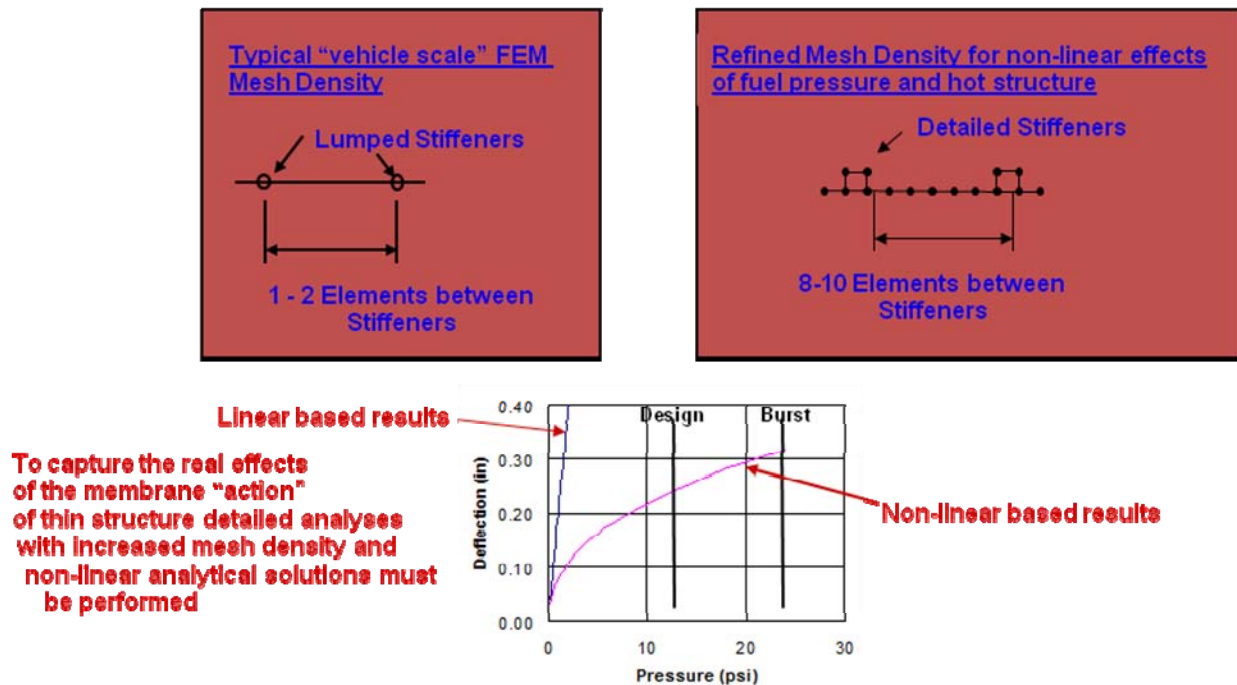
Figure 34. The Necessity of Refined Meshes for Effective Thermal Transfer Analyses

For modeling structural joints, one approach is to insert spring elements between the joined components (Figure 35). This practice "softens" the joint and distributes the load more gradually and presumably more realistically. Drawbacks of this approach include the time consuming and error prone modeling effort required, as well as the proper definition of the appropriate spring stiffness. An automated method of constructing these joints, which should be oriented along potentially curved joints, would speed model generation and reduce errors. Detailed stand-alone models, validated with relevant testing, could be used to develop criteria for generating the appropriate stiffness values to use in such models. Additional testing is also required to validate these modeling techniques and assumptions.

Small local deformations, perhaps due to local plasticity, local yielding, joint slipping, etc, can relieve the high stress concentrations that can occur in a linear elastic FEM. Since these nonlinear effects are not captured in the linear elastic FEM, some other approach is needed to account for them. One approach that has been used is to separate skins and substructure using coincident nodes on each part, and connecting them with springs. These springs provide some flexibility between parts and help relieve the artificially high thermal stresses.
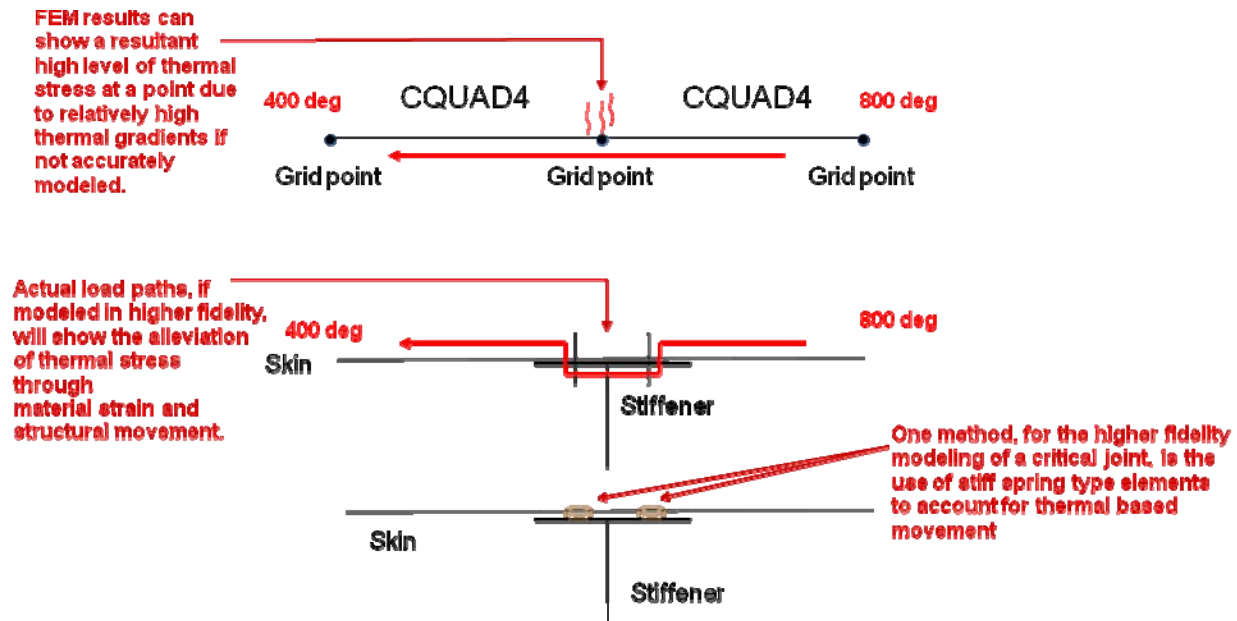
Figure 35. Modeling Joints More Accurately for Thermal Heat and Thermal Stress Transfer

4.7.4   Creep Considerations

Prediction of creep life of a hypersonic airframe can be problematic. Most creep test data is for simple stress states (e.g. uni-axial tension) and at constant load and temperature. Real applications experience multi-axial stresses and continuously varying load levels and temperatures. While theoretical models exist to extrapolate simple test data to such complex situations, uncertainty remains for the designer. This results in conservative assumptions where worst case loads and temperatures are combined simultaneously (with some judgment) to ensure structural integrity.

While time dependent nonlinear analyses are possible, they are prohibitive to use during the iterative design process. An ideal approach would be one in which the results of a linear static vehicle FEM are fed into a post-processing tool that performs rapid creep life calculations. More experimental data under realistic conditions is also required to validate the available theoretical models. To handle uncertainties in creep life prediction, a scatter factor, similar to what is used in fatigue analysis, can be used. However, additional data would be useful to help define what this factor should be for specific materials under specific load-temperature time histories.

4.7.5   Allowance for Uncertainties

In order to provide for uncertainties in loads, analysis methods, material properties, and fabrication quality, a Factor of Safety (FS) is used. The largest loads expected during the planned mission are defined as "limit" loads. "Ultimate" loads are defined as limit loads multiplied by the factor of safety. The design criteria typically require that there is no excessive deformation (or yielding) at limit load and no failure (rupture or collapse) at ultimate load. The FS, along with the airframe static test, ensures that structural failures at limit load are rare in service.

In stress analysis, limit loads are multiplied by a factor of safety to obtain ultimate loads for

strength analysis. The FS accounts for a variety of uncertainties, including uncertainties in the applied loads, structural analysis methods, failure mode predictions, fabrication quality, etc. Also, statistically based strength allowables (e.g. B-basis allowables) are used to account for material variability. Since thermal stresses result from temperatures that are the result of heat transfer analyses with various built-in assumptions, one might ask how uncertainties in the thermal analysis are handled. Are the inputs to the heat transfer analysis typical values or worst case values? What should the factor of safety on thermal stresses be? A common methodology seems to be to use FS=1.25 for thermal stresses that add to mechanical stresses, but to not consider thermal stresses when they relieve mechanical stresses.

However, when thermal stresses are present, this appropriate factor of safety must be considered more closely. The uncertainty in the thermal stresses may or may not be of the same level as the mechanical stresses, and so may require different factor to be consistent. Also, in cases where the thermal stresses relieve the mechanical stresses, the design criteria usually require that the beneficial effect be neglected, or at least reduced. Factors are typically applied to thermal stresses in the structural analysis phase, but it may be worthwhile investigating uncertainties in the thermal analysis as well. What is the effect of realistic variations in heat transfer coefficients and other parameters on the resulting temperatures and thermal stresses?

In summary, a practical design tool should fit into the existing airframe design process, which uses relatively coarse vehicle finite element models for internal loads analysis. Closer coupling with thermal analysis to ensure consistent thermal and mechanical load combinations and consistent mesh densities would be of benefit. Validation of modeling approaches using detailed nonlinear analysis would provide a more rational basis for preferred modeling recommendations. Additional experimental validation of complex creep analysis is required to gain confidence in theoretical models. The necessity of large complex analysis models should be avoided in favor of rapid post-processing utilities which utilize linear static internal loads model results.

4.8   DAMAGE ASSESSMENT AND DAMAGE TOLERANCE ANALYSIS
The practice of damage assessment and damage tolerance, fatigue, and fracture analyses is generally not conducted within rapid design concept demonstrator programs due to the relatively low hour count called upon for the demonstrator service life. However, the requirements of certain programs have called for such an assessment to be done. Two such relatively recent LM Aero programs are the JSF X-35 and an unmanned vehicle demonstrator.

Under the X-35 Concept Demonstrator program DaDT analyses was performed using ADAMsys and a Linear Elastic Fracture Mechanics (LEFM) code developed by the LM Aero Fort Worth group. In general, the work comprised a select group of critical parts analysed for two lifetimes of the expected service life of approximately 600 hours. F-16 load spectra was used for the various aircraft components (i.e., wing, fuselage, empennage, etc.) and the spectra was adjusted by aircraft loading to match the X-35 loads that were arrived at by program loads derivation tools and then correlated and checked by airframe proof test results. One example of the exercise is that the F-16 wing bending spectra was adjusted by the X-35 wing root bending moment results. As a standard practice on the program an initial flaw size of 0.050" for corner flaws was used along with 0.100" for initial surface flaws. The materials that were analyzed were fatigue and fracture resistant (i.e. high fracture toughness, critical crack lengths greater than 0.25 inches, etc.). All crack growth

predictions used the Forman and Newman crack growth equation (da/dN curves) based on the selected material forms.

On one unmanned vehicle demonstrator program fatigue life investigations were carried out on the vehicle's control surfaces. DaDT analysis was performed using AFGROW and the LEFM code that was developed by the Air Force from the pre-existing code NASGRO. The metallic portions of the control surface assemblies were analyzed for a single lifetime of the expected service life which was approximately 4000 hours total. For this program a modified B-2 control surface load spectra was used and this resulted in a conservative spectrum and therefore analyses had to be run to a single lifetime (typically, damage tolerance analysis is conducted at two lifetimes). The load spectra was then adjusted to match the control surface hinge moments. Again, the same initial flaw sizes of .050" for the corner flaws and 0.100" for the surface flaws was used along with the same Forman and Newman crack growth equations (da/dN curves) being employed.

The above examples comprise only non-hypersonic platforms in adding to the fact that DaDt analysis code needs to be developed to handle the material non-linearity indigenous to high supersonic and hypersonic aircraft. For this class of vehicles the general trend needs to move away from LEFM based fatigue work in that material data (i.e., da/dN rates, fracture toughness, yield strengths, degradation effects, material knockdowns, etc.) all need to be varied by temperature. A huge gap in this field is that there is not only a need in hypersonic regime change in complex loading over time, but the development of the thermal spectra needs to be well incorporated with changes in thermal and mechanical properties also being varied with time.

4.9   TESTING AND INSTRUMENTATION
LM Aero investigated the issues that are seen in high temperature instrumentation capability as pertaining to hypersonic and other high temperature testing work. The factors of whether these issues will just take more time and/or investment to eventually solve or whether they pose scientific or technical challenges associated with their sustainment in high temperature and/or vibratory environments are still open for debate. The following illustrates some of the key issues in this field from the LM Aero perspective.

In review of previous LM Aero hot structure test efforts it was seen that a "true" distributed loading of heated structures is not possible due to interference of loading devices and heating apparatus. The lack of this capability severely limits the level of complexity and accuracy of representative loading a structure for the purpose of simulated hypersonic flight ground testing. As a result structural tests are limited to simple cantilever loading or moment induced loading rather than distributed surface loading.

In addition, high Q simulated loading of surfaces presents a challenge and can only be created in a wind Tunnel or jet impingement environment over limited areas of a structure. Together with this, testing of surface coatings or erosion of a structure are generally limited to coupon level characterizations. Another factor to add to this is the fact that temperature measurement and control represents a significant challenge in acoustic environments. Thermocouple instrumentation techniques are highly susceptible to fatigue induced failure while non-contacting dual color IR thermal control is limited to application temperatures of +500°F to 3000°F. The lack of low range (ambient to 700°F) non-contacting IR capability creates significant challenges for

development of thermal durability testing and flight spectrum application.

Another question looked into is whether health management instrumentation for sustained M5.0-6.0 flight vehicles (30 minute-1.5 hour cruise) is "just around the corner" or whether there is significant investment that needs to be made. The question was posed as to whether these are issues of primarily a material nature and/or are there other such issues?

Strain measurement techniques are severally limited due to the acoustic and thermal environments encountered and are limited to non-contacting methods due to fatigue, bonding, and material breakdown. Standard "wired" instrumentation including fiber optic methods suffer from fatigue, corrosion, and significant calibration issues when used in a combined thermal and acoustic environments. Laser extensometers offer some potential for addressing this issue but again are subject to the same harsh environmental challenges.

Also explored were previous testing experiences where test boundary conditions and the inaccuracy/inadequacy thereof posed problems and therefore hindered the accuracy of test results. Whether or not these instances posed issues within post test correlation with pre-test predictions were investigated. Were these boundary condition or other testing factors that could not have been anticipated beforehand? (the first flight of the X-43A comes to mind).

As an illustration, combined structural and thermal testing of X-37 body flaps required development of cold-plates for simulation of thermal boundary conditions. While complex and cumbersome, they did prove effective. Development of actively cooled restraint and loads application hardware and monitoring of the thermal conditions at the load interfaces is necessary when developing post test correlations. Thermal isolation of load measuring devices is also a challenge as drifting temperatures introduce additional inaccuracies and complexity to post test analysis.

Aside from such systematic errors as instrument calibration, there are real sources of discrepancy between flight test and theoretical skin temperatures. Some of these are:
1. All thermocouples are not located at the same stations as the pressure taps on the model. Therefore, some pressure coefficient values have to be interpolated.
2. The amount of solar radiation varies with flight conditions. Since no attempt is usually made to determine the zenith angle, this will be a source of discrepancy.
3. Aside from calibration errors, there are inherent errors in the thermocouples, themselves. The temperature at the thermocouple location is rarely the same as if the thermocouple absent. Therefore, the mere presence of a thermocouple is a source of error.

In the areas of high temperature instrumentation and high speed combined environment testing (in general) the two primary needs for future improved accuracy for hypersonic vehicle service life prediction are considered to be:

- Embedded fiber optic strain and temperature measurement systems
- Application of simulated ground air ground mechanical, thermal, and acoustic loading to complex aircraft structures

In summary, testing and instrumentation needs areas that are needed to be heavily investigated for more accurate service life prediction in hypersonic flight hot structures include the following:

1) Thermal control and measurement methods
2) Strain measurement techniques
3) Loads introduction and reaction
4) Durability of above in an acoustic environment

5.0 EVALUATION OF MDO (ANALYSIS BASED) TOOLS

Many commercially available finite element programs for structural analysis also have optimization capabilities. In the thermostructural analysis context, these codes can apply temperatures as input conditions in a structural analysis. However, when the temperature distribution is sensitive to the structural design variables (such as members thicknesses/areas), these tools do not directly account for this coupling. Instead, the analysis cycle must be interrupted for thermal model updates and re-application of the resulting temperatures to the structural model. A tool that considers thermal and structural analyses simultaneously would be ideal. This may or may not require identical models, but at a minimum the data transfer between the models would have to be automatic and seamless. Data transfer is needed in both directions: the thermal model needs member sizing data from the structural model, and the structural model needs temperatures from the thermal model.

For an analysis tool to be most useful to industry, it must be easy to use and not require a high level of special expertise compared to existing tools. While there may be existing academic and research oriented codes currently available, they are not easy to transition to use in industry for a variety of reasons. Often large aerospace organizations has a preference to limit the number of tools supported internally, and standardize on well known software packages, rather than using multiple tools that are less well known and possibly considered "freeware" or "shareware". For that point of view, incorporating new capabilities into existing software packages has some advantages. Also, it is much more likely to gain acceptance of a new tool if it is compatible with existing industry "standard" tools. Since it is likely that finite element models will be generated using standard software, any new tools should leverage that existing capability.

The above illustrates the limitations, in general terms, of structural analysis type Multi-Disciplinary Optimization (MDO) techniques. In the FAST program as previously mentioned, thermal loads were based on hand-sized material gauges, and then those thermal loads were kept constant through the structural sizing. Also, aerodynamic loads were based on a rigid vehicle. There needs to be iteration between the thermal loads, aerodynamic loads, inertial loads, and structural sizing using high fidelity methods. Currently the thermal loads analysis task at the vehicle level is an expensive task and updating is costly, as is calculating CFD-based aerodynamic loads. Structural MDO methods that allow for the efficient coupling (and associated transfer of data) of these disciplines could enable these iterations.

High-fidelity aerodynamics that can be integrated with an MDO tool suite and allow for rapid trades is of dire need. Reduced order modeling techniques are out there that could possibly enable this.

Coupled Fluid-Thermal (CFT) analysis (conjugate-heat transfer), involving direct iteration

between vehicle thermal analysis and temperature boundary condition assumed in CFD analysis may be less conservative than adiabatic wall assumption. In a conjugate heat transfer analysis, the heat flux derived by the CFD solution is applied to the thermal analysis and the resulting wall temperature predicted by the thermal analysis is applied as a boundary condition in the CFD analysis. This necessitates iteration between thermal and CFD analysis to arrive at a converged wall temperature. And, this would also allow for an update of the thermal loads previously mentioned.

High-fidelity modeling and optimization is needed earlier in the design process for hypersonic concept vehicles as opposed to conventional type aircraft. The rapid generation of models to allow for trade studies to identify critical design drivers early in the design process is paramount.

For the area of high speed/hypersonic MDO study needs for improvement in accuracy for service life prediction, for concept design levels such as M5.0-7.0 hypersonic cruise type platforms, a list of the biggest gaps and needs were collected. These are:

1. Predicting aeroelastic characteristics of very thin metallic structure at high temperatures for sustained periods of time.
2. Actuator stiffness predictions (Stiffness of NASP's all moveable tail was hard to nail down).
3. Sonic fatigue under elevated temperature. Predicting the acoustic environment.
4. Damage tolerance under elevated temperatures.
5. Interaction of fuselage dynamics in flutter analysis. How to model the fuselage, as a flat plate or body of revolution, and do current methods represent fuselage aerodynamics well?
6. Adequately characterizing stiffness of vehicle at hypersonic temperatures, under complexities of stiffened panels and TPS.
7. Analysis needs to be validated with testing. A building block test approach to validate analytical tools is essential.
8. Accurately characterizing the mass, stiffness, and damping of a hot structure and/or thermal protection system.

6.0 LISTING OF KNOWLEDGE GAPS AND RECOMMENDATIONS:

As a summary of the findings of the complete investigation and as per LM's perspective, the compilation of the uncertainties and gaps in the investigation of hypersonic vehicle analyses capability and predictability concerns are illustrated in the following section.
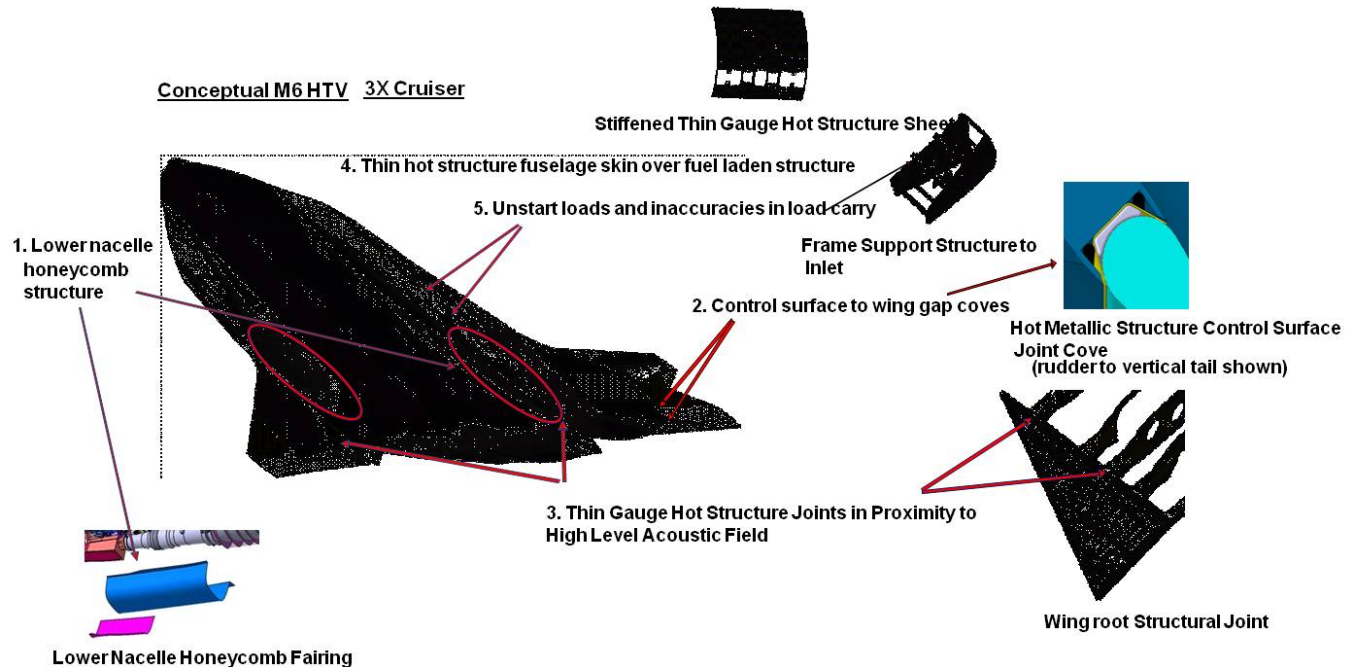
Figure 36. Conceptual M6/HTV-3X Cruiser Vehicle (Nos. 1-5)

Program: HTV-3X:

1. **Structural Component or General Discipline Area:**
   Hot structure lower nacelle honeycomb structure

   Gap Description:
   Complete understanding of the combined thermal loading from "warm" environment of M2.5 at transition coupled with thermal "shock" of scramjet turn - internal heat with acoustic loading.

   Sizing Approach:
   Static loading at concept design phase considered only steady state temp considerations without thermal shock parameters; acoustic loading sourced only from turbine engine testing to date.

   Loads & Boundary Conditions:
   Uncertainty in thermal prediction accuracy along with uncertainty of dissipation of acoustic source from engine (highly conservative in conceptual design)

   Current Approach to Address This Gap
   Size for steady state temperature condition loading with acoustic sound effects added to combined loading - not in sequence.

2. **Structural Component or General Discipline Area:**
   Control surface to wing gap coves: aileron and rudder to vertical tail - heating gaps.

Gap Description:
Radiation effects resulting in high thermal loading concentration leading to unpredictability in true combined loading state

Sizing Approach:
Selection of points (2) along the control surface movement arc used for characterization

Loads & Boundary Conditions:
simple assumptions on travel movements at max and min travel points

Current Approach to Address This Gap
Two points along the travel for determination of steady state heating conditions - possible coupling of max pressure and max heat flux condition - over conservative

3. **Structural Component or General Discipline Area:**
Structural integrity of **t**hin hot structure skin joints near intense acoustic environment

Gap Description:
Combined environment affects with material degradation during flight transition phase.

Sizing Approach:
Flight trajectory based sizing point that may or may not capture the worst acoustic condition

Loads & Boundary Conditions:
"Lumping" of SINDA based thermal result with worst case acoustic loading assumed; material knockdown at the prescribed

Current Approach to Address This Gap
Uncertainty in the thermal loading condition variance and the mechanical property breakdown vs. time exposure, conservatism

4. **Structural Component or General Discipline Area:**
Thin hot structure fuselage skin over fuel laden zones

Gap Description:
Stability performance of the stiffener strengthened skin (.025" IN baseline) under the combined environment loading in a (possible) high fatigue induced environment from acoustic sound pressure

Sizing Approach:
Uncertainty in the property breakdown value vs. thermal exposure time per the trajectory; uncertainty in the thermal environment; tools prediction capability limits to a possibly conservative stiffness matrix

Loads & Boundary Conditions:
Uncertainty in thermal prediction accuracy along with uncertainty of dissipation of acoustic source from engine (highly conservative in conceptual design)

Current Approach to Address This Gap
Thermal exposure testing of the thin skin in a vibratory environment would have to be performed before detailed analyses ($$$); this could lead to hot structure testing boundary condition inaccuracies

5. **Structural Component or General Discipline Area:**
Inlet unstart loads coupled with the effects of nonlinear material property degradation per temperature exposure and duration.

Gap Description:
Cumulative exposure effects: H2, thermal, & fatigue. Modeling hat stiffened structure was not the major challenge.

Sizing Approach:
Static loading at concept design phase considered only SS temp considerations without thermal shock parameters; acoustic loading sourced only from turbine engine testing to date

Loads & Boundary Conditions:
Uncertainty in thermal prediction accuracy along with uncertainty of dissipation of acoustic source from engine (highly conservative in conceptual design)

Current Approach to Address This Gap
Size for steady state temperature condition loading with acoustic sound effects added to combined loading - not in sequence

6. **Structural Component or General Discipline Area:**
Hot thin gauge facesheet panel flutter (refer to Figure 38)

Gap Description:
Panel flutter of cantilevered edges (seals) or unsupported edges between fasteners (hot structure) can lift under unsteady aero effects (also reported in X-15 reports as a high speed phenomena). NASA report of 2004 analytically revisited the X-33 panel flutter issue.

Program: X-33
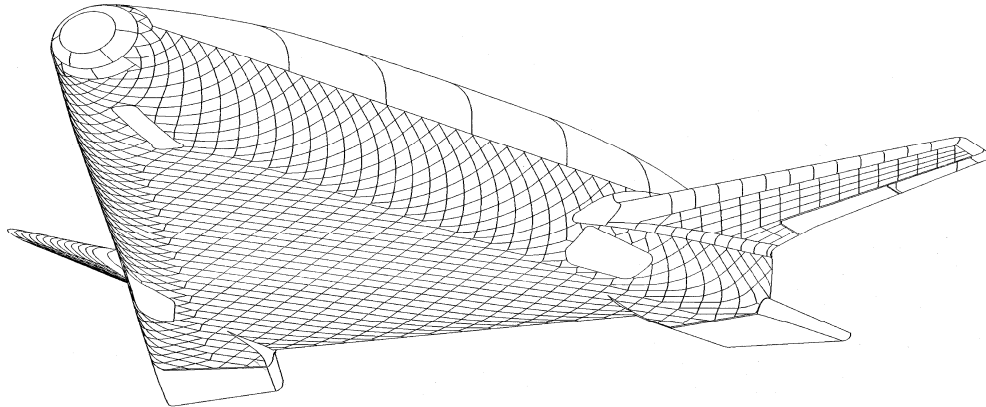
Figure 37. X-33 Windward TPS Metallic Panel Arrangement (Nos 6-8)

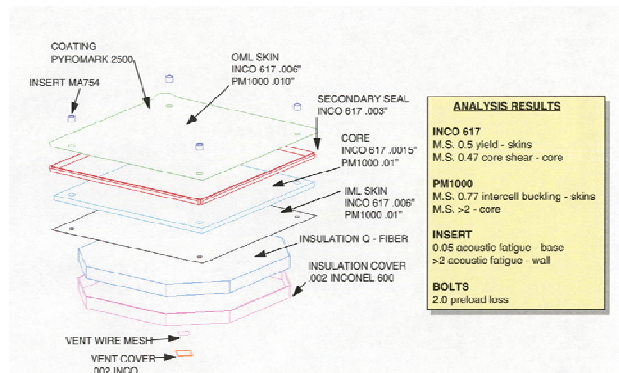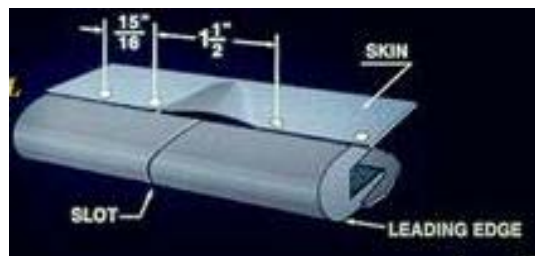**Figure 38. Typical X-33 Metallic TPS Thin Gauge (.006'') Facesheet Panel**

Figure 39. X-15 M5.28 skin buckle (similar behavior to item 6.)

Sizing Approach:

Gauge of facesheet is "bumped" up to compensate to mitigate potential flutter effects leading to sealing issues between hot structure skins. This area is primarily sized for stability considerations.

Loads & Boundary Conditions:
Second order piston theory is employed to calculate the unsteady aerodynamic loads for hypersonic speed - this theory neglects the effects of 3-dimensional flow; commercial code Marc does not incorporate the material effects upon temp exposure and material props are independent of the calculations.

Current Approach to Address This Gap:
Use of second order theory in Marc and lip seal thickness (or entire skin gauge) increase to compensate inter-fastener span skin lifting

7. **Structural Component or General Discipline Area:**
Metallic Panel Inserts Analysis (refer to Figure 38)

Gap Description:
Delta temperatures between the inner and outer facesheets of the sandwich panels caused the panels to bow which resulted in significant punch loads on the fastener inserts.

Sizing Approach:
Extremely fine mesh 3-D panels models were created for detailed 3-D analyses. These would not be practical for "macro-size" type skin section analyses due to computer memory limitations strength at temperature was critical.

Loads & Boundary Conditions:
The most critical stress levels in the inserts were generated by the insert acoustic and thermal loads. In-plane movement translations were determined by the motion of the vehicle sub-structure.

Current Approach to Address This Gap:
Fine mesh 3-D insert models would need to be created for certain zones and worst case assumptions, per vehicle zones, would have to be applied.

8. **Structural Component or General Discipline Area:**
Metallic Panel Edge Effects (refer to Figure 39)

Gap Description:
Steps between panels had the potential to substantially increase edge deflections by rotating the seals further out into the boundary layer. The deflections in coupling with the reversed flow over the edges and seals could have led to some substantial additional edge heating.

Sizing Approach:
No panel edge loads or steps between panels or local seal (between panels) were considered during the analysis phase. Once panel edge loads were considered there was the potential to double the seal compressive strain level.

Loads & Boundary Conditions:
High magnitude edge compression loads developed from aero panel loads, thermal panel loads, and thermal gradients normal to the panel edge. In-plane movement was dictated by adjoining panels and sub-structure motion.

Current Approach to Address This Gap:
Local edge heating was required to complete the panel analyses. Fine mesh 3-D insert models would need to be created for certain zones and worst case assumptions, per vehicle zones, would have to be applied.

9. "Warm" TMC fuselage primary structure - cumulative exposure effects

10. Composite cryogenic tank suspended in fuselage - cumulative exposure effects

14. High temp sharp passive leading edges for 2000q ascent, re-entry.

12. Parasitic TPS - Acoustic and thermal cycling fatigue.

15. Management of actively cooled leading edges.

13. Hot flight control structure - Flutter evaluation control surfaces & panels in hypersonic flight.

11. Vehicle structure and propulsion dynamics - coupling effects

Figure 40.   NASP X-30 Configuration and Areas of Structural Concern (Nos. 9-15)

Program: NASP

9. **Structural Component or General Discipline Area:**
"Warm" TMC fuselage primary structure

Gap Description:
Cumulative exposure effects: H2, thermal, & fatigue.

Sizing Approach:
TMC Min gage ~ 0.055 was lower boundary on acreage weight.   Final plan assumed life was adequate for 30 suborbital flights (aircraft #1) or 12 suborbital +1 orbital flight (aircraft #2).

Loads & Boundary Conditions:
High uncertainty in prediction of worst thermal stress conditions. Conservative assumptions were needed.

Current Approach to Address This Gap
Size for worst case with margin for uncertainty including magnitude of loads,

combined conditions, & locations on the vehicle.

10. **Structural Component or General Discipline Area:**
Composite cryogenic fuel tanks suspended in fuselage.

Gap Description:
Cumulative exposure effects, micro-cracking, slosh & dynamic loads, thermal cycling, and properties of large fabricated tanks (out of autoclave).

Sizing Approach:
Very aggressive thin tank membranes (5 to 7 plies) sized by static analysis (linear NASTRAN and nonlinear ABAQUS).   Shear webs react pressure & inertial loads.

Loads & Boundary Conditions:
8 load conditions for tank & entire vehicle: taxi, M0.9 pullup, M1.2 pullup, M1.2 pulldown, M8.0 pullup, M17.0 pullup, M17.0 pulldown, & 22 psi only.

Current Approach to Address This Gap:
Use results of post NASP large out of autoclave composite cryotank development to update predictions & validate with another large scale test tank & flight demo.

11. **Structural Component or General Discipline Area:**
Vehicle Structure + Propulsion Dynamics

Gap Description:
Engine dynamics in flight and internal engine bay environment effect on structural interfaces.

Sizing Approach:
Use turbojet & rocket experience, NASP analysis, & planned Phase III flight hardware ground tests.

Loads & Boundary Conditions:
Integrated vehicle + propuls loads & deflections based on numerous assumptions.

Current Approach to Address This Gap:
Build flight propulsion structure and ground test to calibrate prediction tools.

12. **Structural Component or General Discipline Area:**
Parasitic TPS

Gap Description:
Combined environment effects, integrated system modeling, durability, failure modes and predictions.

Sizing Approach:
Static & dynamic analysis calibrated to combined environment tests

Loads & Boundary Conditions:
Rocket & turbojet derived load environment. Assumptions made about boundary conditions.

Current Approach to Address This Gap:
Use latest Falcon, FAST, & AFRL TPS test & prediction results.

13. **Structural Component or General Discipline Area:**
Hot flight control structure

Gap Description:
Aeroelastic & flutter predictions for high Q trajectory & high control power needs for engine-out or c.g. excursions & emergency maneuvers.

Sizing Approach:
Hot ascent condition drove stiffness. Body flap was added for additional pitch control due to aeroelastic effects

Loads & Boundary Conditions:
Cruise demo requirement drove load condition to fully heat soaked.

Current Approach to Address This Gap:
Same basic approach.

14. **Structural Component or General Discipline Area:**
High temperature sharp passive leading edges

Gap Description:
Integrated structure deflections, thermal stresses, & local loads for multiple edge segments. Edge gap control throughout the flight. Reusability max 25 cycles.

Sizing Approach:
Segments sized by allowable overlap similar to Space Shuttle. Min gage for coated Carbon-Carbon and CMCs provides adequate strength & stiffness.

Loads & Boundary Conditions:
Max heating at initial atmospheric interface during re-entry. Pressure loads not a driver. Material conductivity kept thermal stresses acceptable (ground test).

Current Approach to Address This Gap:
Size for steady state temperature condition loading with acoustic sound effects added to combined loading - not in sequence

15. **Structural Component or General Discipline Area:**
High temperature sharp actively cooled leading edges.

Gap Description:
Integrated structure deflections, thermal stresses, & local loads for multiple edge segments. Edge gap control throughout the flight. Reusability is a big question.

Sizing Approach:
Sized by thermal stresses & pressure loads. Ground tests sized the details. Primarily sized for stability considerations.

Loads & Boundary Conditions:
Heat flux sized the cooling approach. Pressure & thermal stresses sized the details. Boundary conditions not well understood, rely on major assumptions.

Current Approach to Address This Gap:
Build integrated flight structure with edges & ground test to calibrate prediction tools.
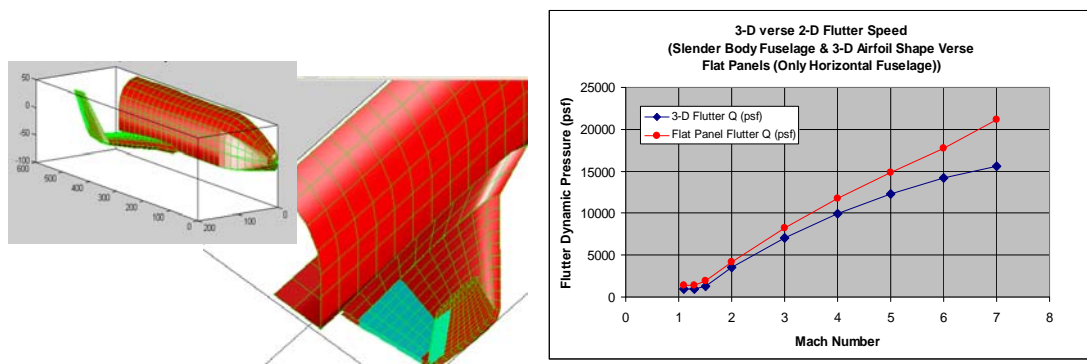
Figure 41.   Fully Reusable Access to Space Technology (FAST) Configuration (No. 16)

Program: FAST

16. **Structural Component or General Discipline Area:**
Aeroelastic effects - vehicle body flutter

Gap Description:
Two highly used commercial hypersonic aeroelastic prediction tools, do not take into account material stiffness variations upon elevated temperature and duration.

Sizing Approach:
Since the vehicle utilized TPS the structure was assumed cold and stiffness reductions due to elevated temperatures were not an issue.

Loads & Boundary Conditions:
The tool of choice uses a linear frequency based method and handles airfoil thickness and bodies. Flutter dynamic pressure went up steadily as Mach number went up as the transonic region was considered worst case (cold structure assumption). For metallic structures at elevated temperatures stiffness properties reduce considerably and need to be factored into the aeroelastic analysis.

Current Approach to Address This Gap:
Material property knockdown assumptions need to be made based on available empirical data are to then be coupled into the code material input. In addition, two other vehicle body flutter issues are of concern. 1. For time-domain, CFD-based aeroelastic predictions, the required time-step must be very small due to the high-velocity and thus run time becomes a concern as well as solver stability. 2. In discussions with FAST as well as NASP engineers, correlation of analytical flutter results with test data at hypersonic speeds is a BIG need.
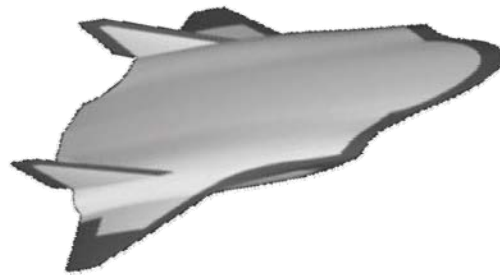
Figure 42.   Horizontal Takeoff/Horizontal Landing Airbreather Concept Vehicle (FALCON) Configuration (Nos. 17-20)

Program: Various, General Hypersonic Program Needs

17. **Structural Component or General Discipline Area:**
    Hot Structure Testing Boundary Conditions #1

    Gap Description:
    Distributed loading of heated structures is not possible due to interference of loading devices and heating apparatus.

    Sizing Approach:
    Conservative factors employed for both test and eventual flight hardware drawing release.

    Loads & Boundary Conditions:
    As a result hot structure component and strength tests are limited to simple cantilever loading or moment induced loading rather than distributed surface loading.

Current Approach to Address This Gap:
Data resultant from "point" or cantilevered loading are distributed to the structure for "best estimate" approximations of distributed loading and eventual model correlation.

18. ### Structural Component or General Discipline Area:
Hot Structure Testing Boundary Conditions #2

Gap Description:
High Q simulated loading of surfaces presents a challenge and can only be created in a wind tunnel or jet impingement environment over limited areas of a structure.

Sizing Approach:
Either conservative (uniform coating level but lower performance (emissivity = .5 or .6) approach) or unconservative assumptions (uniform e=0.9) are assumed which could lead to serious thermal performance inaccuracies.

Loads & Boundary Conditions:
Testing of surface coatings or erosion of a structure are limited to coupon level characterizations. Panel level assessments and sub-component to component level testing is primarily performed only at the validation (or certification) level.

Current Approach to Address This Gap:
Estimates based on FEA analysis are used to supplant the cooldown temperature data of structure below 500F which are inherent to inaccuracies in estimated radiative heat loss and convection effects.

19. ### Structural Component or General Discipline Area:
Hot Structure Testing Boundary Conditions

Gap Description:
Temperature measurement and control represents a significant challenge in acoustic environments.

Sizing Approach:
Simplistic 1-D thermal analysis methodology are usually used to estimate heating and cooling of structure below 500F. This has lead to conservative assumptions to have to be made on descent types of conditions on sizing structure.

Loads & Boundary Conditions:
Thermocouple instrumentation techniques are highly susceptible to fatigue induced failure while non-contacting dual color IR thermal control is limited to application temperatures of +500F to 3000F. The lack of low range (ambient to 700F) non-contacting IR capability creates significant challenges for development of thermal durability testing and flight spectrum application.

Current Approach to Address This Gap:
Estimates based on FEA analysis are used to supplant the cool-down temperature data of structure below 500F which are inherent to inaccuracies in estimated radiative heat loss and convection effects.

20. **Structural Component or General Discipline Area:**
Hot Structure Testing Boundary Conditions #4

Gap Description:
Strain measurement techniques are severely limited due to the acoustic and thermal environments encountered and are limited to non-contacting methods due to fatigue, bonding, and material breakdown.

Sizing Approach:
Assumed recorded data have inaccuracies upon extended duration and exposure to combined loading environments. Analytical based data are used to augment potential "extended range" dataset.

Loads & Boundary Conditions:
Standard "wired" instrumentation including fiber optic methods suffer from fatigue, corrosion, and significant calibration issues when used in a combined thermal and acoustic environments.

Current Approach to Address This Gap:
Laser extensometers offer some potential for addressing issue but again are subject to the same harsh environmental challenges.

7.0   SUMMARY AND CONCLUSIONS

Important gaps in structural analysis and life prediction methods have been identified. Major reuseable hypersonic vehicle programs were researched. Areas where state-of-the-art methods are incapable of predicting the response and life of structure are identified. Uncertainties due to predictive capabilities have been major issues that affected the success or failure of hypersonic programs. The review of A-12/SR-71 predictive approaches and history reveals the skill of the engineers as they made conservative simplifying assumptions. Program documents reveal careful ground tests of center fuselage and tank structure, propulsion testing through the available Mach range of ground facilities, and a careful envelope expansion approach to flight testing. Even with that approach there were significant aircraft loss rates. The program was a national priority and was managed by extremely skilled leaders, producing an amazing operational capability.

The NASP program was reviewed, and the major predictive uncertainties are reported. Approaches to address the identified gaps are summarized. The monitoring of the accumulated uncertainties by national review boards is documented, and the effect on program cost and schedule is included in this report. The NASP program history clearly illustrates that structural analysis and life prediction gaps are recognized and monitored by decision makers. These can play

a part in decisions to proceed or cancel a program.

The X-33 history reveals a critical relationship between program goals and predictive uncertainty. An allocated amount of TPS panel creep deformation was defined for each flight of the X-33 during its 15 flight test demonstration cycles such that the TPS panels would approach their limits as the vehicle reached the end of its design life. Accumulated uncertainties not only for creep but also for acoustic and pressure loads and issues such as panel flutter could quickly lead decision makers to conclude that an X-33 vehicle might have a useful life of only a few flights, or many more than 15, depending on unknowns. This is another example of the significant effect that predictive gaps have on perceptions of overall program risk.

The HTV-3X Mach 6.0+ combined cycle propulsion demonstrator program cost environment was understood to be severely limited. As a result, the design approach could not include significant weight margin for structural predictive uncertainty. A metallic structure without significant parasitic TPS was the reasonable goal. Mach 6.0 conditions are at the practical limit of metallic structural thermal capability, however. Lack of margin required reduction of thermal, acoustic, and mechanical loading uncertainty.   The approach was to characterize the environment at pre-defined Mach threshold points (Mach 1.0, Mach 2.0, Mach 3.0, etc.) along the trajectory. Critical aeropressure loading cases did not always occur at the predefined thermal analysis points, leading to approximations and increasing uncertainty.

Thin gage welded fuel tanks formed much of the center body structural backbone. Welded joint analysis identified uncertainties related to weld material coefficient of thermal expansion effect on heat transfer and the displacement. Also, weld surface irregularity effects on thermal transfer were unknown. Radiative heat transfer to fuel vapors and purge gas in the tanks was difficult to model. A heavily refined and detailed mesh non-linear analysis of the weld joints in the preliminary design phase would reduce these uncertainties. Due to the number of welds a stress indicator program similar to that used for sizing X-33 metallic panels would reduce uncertainty for welded metallic fuel-tank structure.

The fast pace of the HTV-3X preliminary design program and the challenging program cost environment led to a design with thin gauge welded metallic tank as the backbone of the vehicle. Significant gaps in the ability to generate the high fidelity highly refined analysis that captured critical weld characteristics in preliminary design led to general discomfort on the part of many program reviewers. Although funding decisions were made at a Congressional level and are not attributed to monitoring of uncertainties and potential cost impacts as happened on the NASP program, subject matter experts were eager to see their concerns addressed by planned design cycles. Once again gaps in structural analysis and life prediction caused concern at a program level, not just within the engineering community.

In conclusion, a historical review of significant hypersonic programs and the associated critical life prediction and structural analysis gaps have been identified. Current methods and tools have been assessed. Processes that would address the identified gaps have been documented in this report.

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9.0 LIST OF ACRONYMS

Acronym	Definition
AFRL	Air Force Research Laboratory
AFGROW	Air Force Growth life prediction program
AIC	Aerodynamic Influence Coefficient
AOA	Angle of Attack
ASE	AeroServoElasticity (The interaction between aerodynamics, Controls, and Flexible Vehicle)
ASIP	Airframe Structural Integrity Program
ASTF	Aero-Propulsion Systems Test Facility
BART	Basic Aerodynamic heating for Rapid Turnaround
BFF	Body Freedom Flutter
BMI	Bismaleimide Matrix composites
c	distance to the extreme fiber
c.g.	center of gravity
C-C	Carbon-Carbon
CAA++	Computational Aero Acoustics ++
CAD	Computer Aided Design
CFD	Computational Fluid Dynamics
CFD++	Computational Fluid Dynamics++
CFT	Coupled Fluid Thermal
$C_{L\alpha}$	Lift Coefficient per unit alpha (alpha = Angle of Attack)
$C_{M\alpha}$	Moment Coefficient per unit alpha (alpha = Angle of Attack)
CMC	Ceramic Matrix Composite
CVD	Chemical Vapor Deposition
dB	decibel
DaDt	Durability and Damage Tolerance
DaDtA	Durability and Damage Tolerance Allowable
DARPA	Defense Advanced Research Projects Agency
DSB	Defense Science Board
E	Young's Modulus
ENSIP	Engine Structural Integrity Program
FAST	Fast Access to Space Technology Program
Fcy	allowable compressive yield stress at which permanent strain equals .002

FEA	Finite Element Analysis
FEM	Finite Element Model
FS	Factor of Safety
Ftu	allowable tensile stress
Fty	allowable tensile yield stress at which permanent strain equals .002
GrEp	Graphite Epoxy
GVT	Ground Vibration Test
H2	Hydrogen
HCV	Hypersonic Cruise Vehicle
HTV	Hypersonic Technology Vehicle
I	Moment of Inertia
IML	Inner Mold Line
IR	Infrared
J2	second deviatoric stress invariant
JSF	Joint Strike Fighter
k	thermal conductivity
LaRC	Langley Research Center
LEFM	Linear Elastic Fracture Mechanics
LST	Linear Stability Theory
LM	Lockheed Martin
M	Moment
M&P	Materials and Processes
MIL-HDBK-5	Military Handbook – 5 (Materials)
MDO	Multi-Disciplinary Optimization
MSC	MacNeal-Schwendler Corporation
NASA	National Aeronautics and Space Administration
NASP	National Aerospace Plane
NASGRO	NASA Growth Analysis code
NASTRAN	NASA Structural Analysis
OASPL	Overall Sound Pressure Level
OML	Outer Mold Line
P	Pressure
PDM	Periodic Depot Maintenance
PDR	Preliminary Design Review
PSD	Power Spectral Density
Q	dynamic pressure
RCS	Reaction Control System
RDT&E	Research Development Technology and Evaluation
RET	Radiation Equilibrium Temperature
RLV	Reusable Launch Vehicle
RSS	Root Sum Squared
SBLI	Shock Boundary Layer Interaction
SEA	Statistical Energy Analysis
SIC	Structural Influence Coefficient

	Matrix
SOL	Solution (NASTRAN)
SSC	Structural Sciences Center
SSTO	Single Stage To Orbit
T	Temperature
TMC	Titanium Matrix Composite
TPS	Thermal Protection System
TPSS	Thermal Protection System Substructure
TRL	Technology Readiness Level
USAF	United States Air Force
VMS	Vehicle Management System
W.S.213	Wing Station 213
σ_{MAX}	Stress (Maximum)
ΔCp	pressure distribution
1-D	One Dimensional
2-D	Two Dimensional
3-D	Three-Dimensional

APPENDIX A: NASP AIRFRAME DEVELOPMENT MASTER SCHEDULE:

The overall program plan includes extensive testing for design data and verification. Essentially all of the relevant U.S. test facilities would have been dedicated to NASP in 1993 if the program had continued as planned. Concurrent design, test, and technology maturation was planned to control program cost and schedule, but at relatively high risk. The highly concurrent ASIP tests are also evident in this figure. The engine development schedule was equally aggressive.
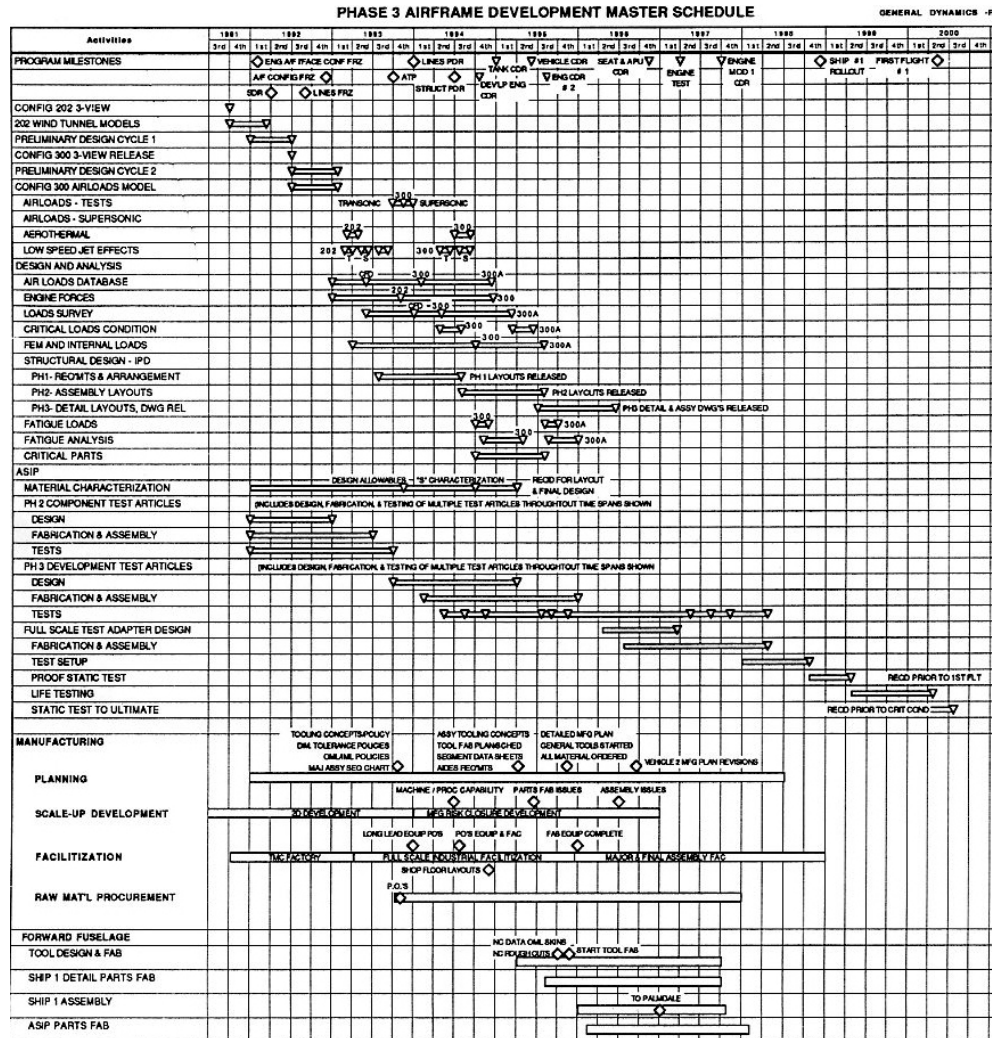

Figure A1. NASP Airframe Development Master Schedule