HYPERSONIC AIRBREATHING VEHICLE CONCEPTUAL DESIGN
(FOCUS ON AERO-SPACE PLANE)

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FUNDAMENTALS OF AERO-SPACE PLANE DESIGN

INTRODUCTION

Hypersonic airbreathing vehicles are highly integrated systems involving strongly coupled, multi-discipline technologies. These vehicular visions and/or applications are now guiding a hypersonic technology maturation effort. In prioritizing the technology issues and focusing the research activity as well as setting goals for this endeavor, it is important to be able to examine the vehicle design options and performance envelope. This vehicle examination, the conceptual design process, requires unique evaluation procedures and analytical tools in all major technical disciplines.

The design procedure, to a considerable extent, depends on the nature of the vehicle to be examined and the design philosophy. In todays hypersonic scenarios, both cruise and accelerator type vehicles are of interest with visions such as the aero-space plane embodying both capabilities. The commonality shared by these hypersonic vehicles is that all operate within an airbreathing corridor; they will be powered by air breathing engines -- a subsidiary engine cycle for low-speed acceleration, ramjets to Mach 5, and scramjets to potentially Mach 20 plus -- and will take off and land horizontally on standard runways. Some will be designed to ascend to cruise at hypersonic speeds, Mach 6 to 12, twenty or more miles above the ground; others will continue to accelerate upward through an airbreathing corridor to Mach 25 and, with minimal rocket power, transition to a low earth orbit, one hundred miles up.

CRUISER VERSUS ACCELERATOR (ORBITER)

The technology itself represents the capability to cruise and maneuver into and out of the atmosphere, to provide rapid response for low-earth-orbit missions, or to attain very rapid transport service between remote Earth destinations. But, there are differences between configurations dedicated to cruise and those that accelerate to orbit. The accelerator must have a much bigger inlet area relative to body cross-section than the cruiser in order to facilitate sufficient thrust margin, and thus sufficient acceleration, to reach orbital speed. Acceleration time must be minimized so that the integrated drag loss in the air-breathing corridor is kept within manageable bounds. On the other hand, the cruiser requires no thrust margin at the design cruise speed. For the accelerator, the primary aerodynamic issue is minimizing configuration drag near zero angle of attack, while for the cruiser, the task is to maximize configuration lift-to-drag ratio at the design point; both are performed under specific volume-to-planform-area constraints. In structures, the differences are mainly in the design of the leading edges (materials and/or cooling) and the tank insulation -- the accelerator is heating rate impacted while the concern for the cruiser is heat load.

Basic Equation

Cruise:

$$\text{RANGE} = \frac{V \; I_{sp} \; \frac{L}{D} \; \ln \frac{w_i}{w_f}}{1 - \frac{V^2}{V_s^2}}$$

Where:
- V = Velocity
- I_{sp} = Specific impulse
- L/D = Lift to drag ratio
- w_i/w_f = Initial to final weight ratio
- V_s = Orbital velocity

Acceleration:

$$\Delta V = g \; I_{sp_{eff}}^{*} \; \ln \frac{w_i}{w_f}$$

where

$$I_{sp_{eff}}^{*} = \frac{\Delta V}{\int_{\Delta V} \frac{dV}{I_{sp_{eff}}}}$$

and

$$I_{sp_{eff}} = \frac{T-D}{\dot{m}} = \frac{\text{Thrust - Drag}}{\text{Fuel flow rate}}$$

Of the three distinguishing factors mentioned, the inlet area and in turn propulsion/airframe integration will be the dominant factor in shaping both the cruiser and accelerator configurations. This is because the propulsion system is sized at hypersonic speeds and must add minimum drag and weight to the vehicle while still processing as much air as possible. These stipulations are best met by considering the entire underside of the vehicle as part of the propulsion system such that the inlet is contiguous with the fuselage and captures nearly all the air processed by the bow shock. This concept, (ref. 1) referred to as the airframe-integrated design, is illustrated in figure 1. The vehicle forebody will have to provide inlet precompression without seriously compromising the aerodynamics, packaging, and Thermal Protection System (TPS) requirements. Not only must the precompression be efficient, but with modular engines closely stacked side-by-side (shown in figure), the flow must be relatively uniform in the lateral direction, across the speed range and during minor maneuvers, to avoid a complex engine oper-ating schedule. The vehicle afterbody must also serve as a nozzle expansion surface in order to provide net installed performance at the high speeds. Both the hypersonic accelerator and cruiser must rely on this engine/airframe integration scheme; the distinguishing factor(s) between the two become the extent of the engine modular stack within the shock layer (inlet size) and/or the amount of inlet over speed (inlet size) and the amount of LOX augmentation (increases thrust levels at the expense of engine specific impulse.)
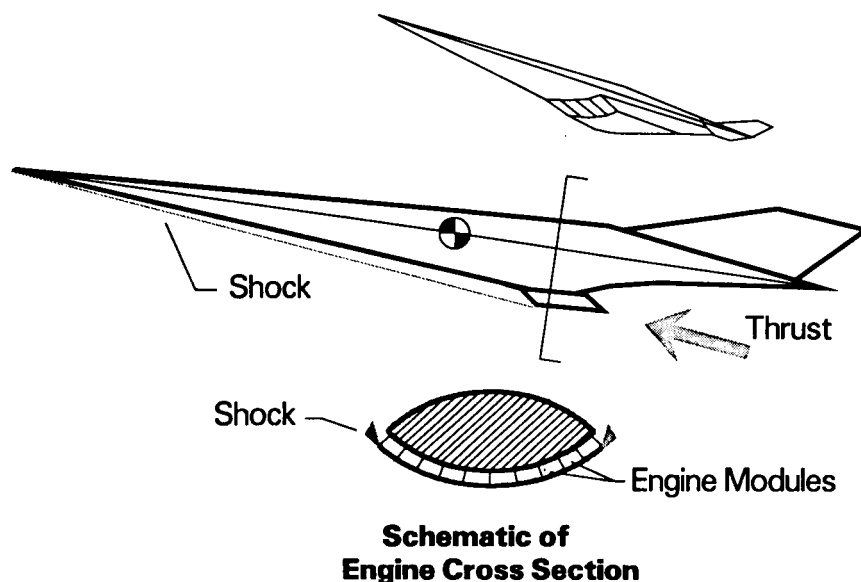
SCRAMJET - VEHICLE INTEGRATION

Schock

Shock

Thrust

Shock — Engine Modules

Schematic of Engine Cross Section

Figure 1

THE AERO-SPACE PLANE MATRIX

If the design is restricted to no inlet overspeed, and extremely light weight materials are available for engine nacelle construction, then the most optimum vehicle configuration could be a flying engine or cone -- cone derivative. But the use of inlet over speed and rocket thrust augmentation opens up the configuration matrix to underslung engine configuration -- especially when more conventional materials are considered for the engine structure and engine weight becomes a factor. See figure 2.
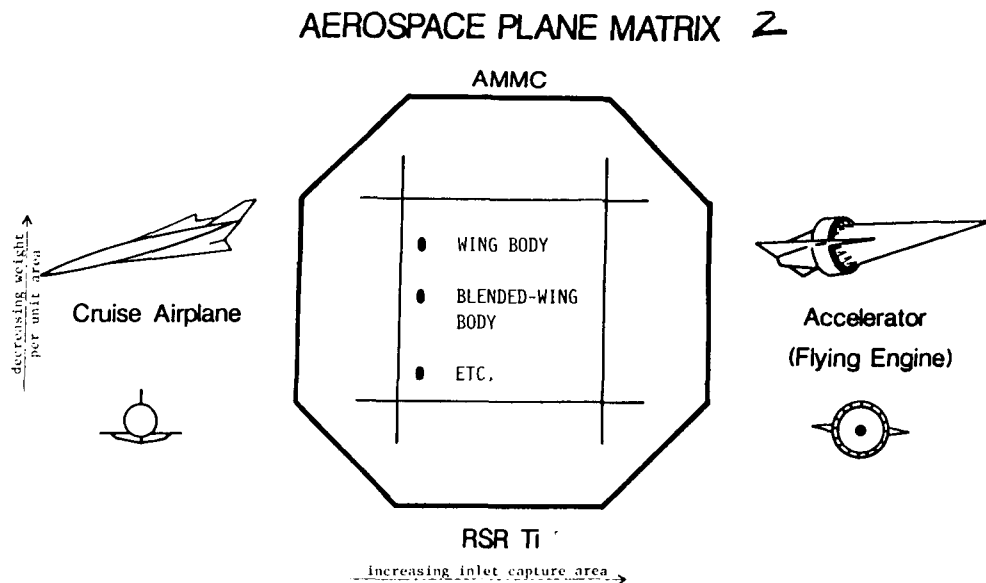
AEROSPACE PLANE MATRIX 2

AMMC

● WING BODY

● BLENDED-WING BODY

● ETC.

Cruise Airplane

Accelerator
(Flying Engine)

decreasing weight →
per unit area

RSR Ti

increasing inlet capture area →

Figure 2

VEHICLE SYNTHESIS

The purpose of this text is to present the conceptual design procedures and tools used for these hypersonic aircraft. Any aircraft design process is a compromise of all the engineering disciplines. An effective design is the integration of aerodynamics, propulsion, structures, and material, flight control, avionics and subsystems, blended in just the right manner to give a complementary effect. This is amplified in hypersonic aircraft design because of the additional acceleration requirements on the vehicle, the high degree of engine airframe integration, and the intrusion of aerothermal loads at the hypersonic speeds; the coupling between the technical disciplines are much stronger and the sensitivities much more intensified. The design process and analytical tool requirements for the hypersonic accelerator and cruiser are similar. Of course, for an orbiter, accomodations must be made for airbreathing acceleration to Mach 20 plus, rocket acceleration to Mach 24, orbital insertion and circulation, deorbit, and reentry; the discipline analytical tools must include the additional Mach delta, stronger viscous interactions, real gas effects in the vehicle flow field -- especially in the boundary layer, finite rate chemistry in the combustor/nozzle, frozen chemistry in the aftbody nozzle, and the transitional and rarefied flow regimes.

A vehicle design/synthesis flow chart is presented in figure 3. A vehicle concept, once conceived, is evaluated through this process. First, the airframe shape, engine flow path, and area distribution are defined and refined. Options on fuselage structural design (integral, non-integral tank, or aeroshell), and substructure (ring frames and bulkheads, ribs and spars, etc.) wing box and carry-thru, and materials are considered along with internal packaging arrangements. Engine structural design is usually selected between stiffened panel and/or honeycomb with or without ring frame or stringer supports.

Engine/airframe integration is the center of the design process. Here, load paths throughout the vehicle are optimized with particular emphasis on the synthergistic transfer of the thrust load from the engine to the airframe. Inlet and nozzle contours are laid-out; not only are these surfaces common to both the airframe and engine in the nested engine integration approach, they are absolutely crucial to the net performance of the propulsion system, and their importance increases with Mach number. Also, since the aftbody nozzle plays a key role in the trim of the vehicle at hypersonic speeds, control of the vehicle must now be considered.

At this point the design process becomes a true synthesis activity. Sizing and flight performance definition can be, and often is, performed in one synthesis operation with direct constraint coupling, but for the sake of explanation simplicity the discussion shall proceed along parallel fronts.
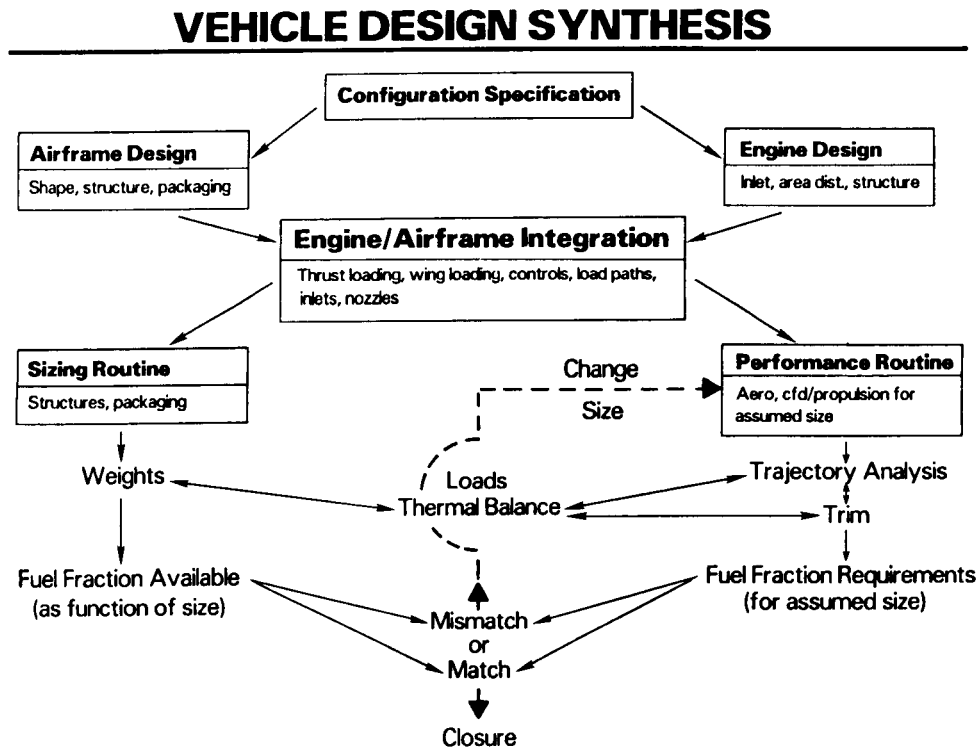
VEHICLE DESIGN SYNTHESIS

Figure 3

The design assessment process with the emphasis on the disciplines and their couplings is shown in figure 4 (ref. 2)
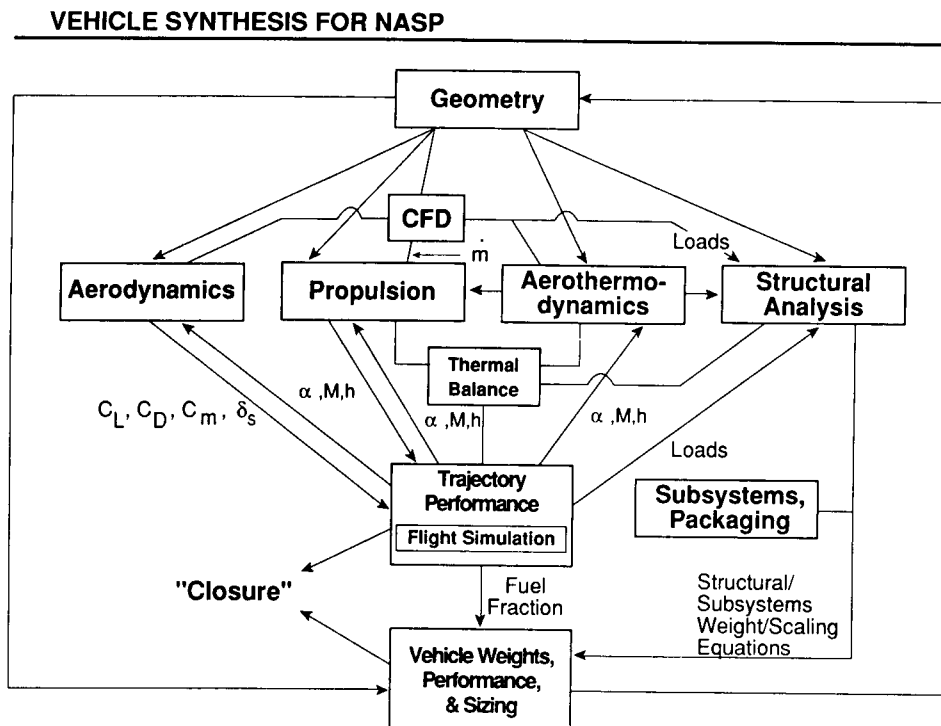
VEHICLE SYNTHESIS FOR NASP

Figure 4

SIZING

The sizing routine requires scaling relationships for the vehicle subsystem and structure. Subsystem weights are based on a technology enhancement extrapolation of historical algorithms; the scaling relationships are based on vehicle length, gross weight, and applicable areas such as inlet or control surfaces.

Structural weights/scaling are generally based on historical data bases; (ref. 3) such has been used in parametric first order sensitivity screening for the aero-space plane. But, because of the uniqueness of the aero-space plane design and performance sensitivity to weight, higher fidelity options are required. One method is to calculate structural weights based on vehicle loads and failure mode criteria and TPS weights based on a transient thermal analysis of the internal wall construction. Insulation requirements are determined by minimum weight to keep internal structure below material temperature limits -- minimum of combined boil-off and TPS[*] weight for tank region. Weights of segments of the structure are expressed in power law form as a function of component length or area. From this information set and the fuel density, the sizing routine calculates the fuel fraction available as a function of vehicle gross weight and/or length (figure 5).
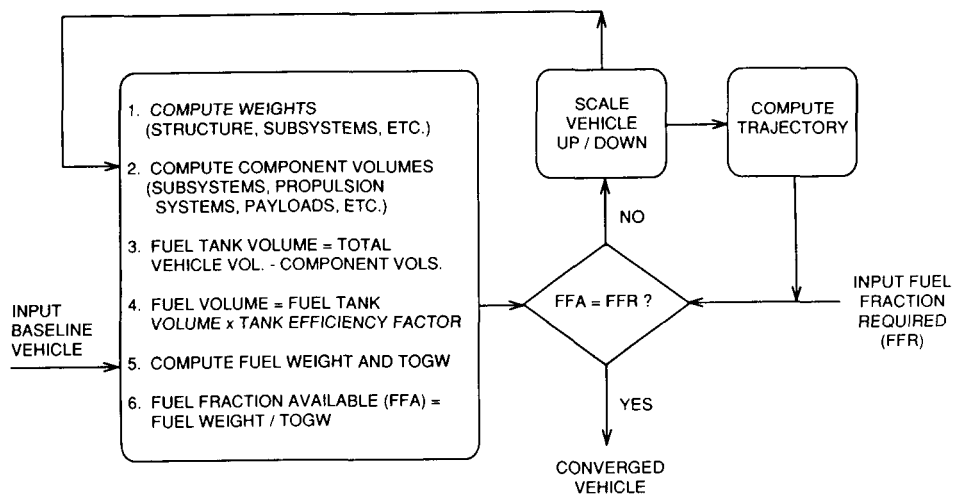
[*]Thermal Protection System

VEHICLE SIZING

Figure 5

PERFORMANCE/TRAJECTORY

The performance routine is a trajectory code, whether a simple energy-state integration approach or a three-degree of freedom dynamic version. Aerodynamic and propulsion performance are the required inputs. A force accounting scheme is selected -- free stream to free stream or inlet ramp to free stream. In the first, any surface that is washed by flow that goes through the engine is a propulsion surface; in the second, only the surfaces that are washed by engine flow from the beginning of the inlet ramping or cowl lip to the afterbody nozzle exit (free stream) is a propulsion surface. The latter is chosen here, again to expedite the discussion. Since the vehicle size and weight are not yet known, nominal values are selected to begin the iteration. The aerodynamic matrix (lift and drag coefficient, C_L and C_D, as a function of Mach number, angle of attack and altitude, M, α, and h) is calculated for an assumed trajectory bandwidth on dynamic pressure (500 psf $\leq$ q $\leq$ 2000 psf). Since the ramjet/scramjet cycle performance prediction codes require inlet flow conditions (mass flow, total pressure recovery, and enthalpy) that are contingent on the airframe forebody precompression, forebody flow field solutions over the range of hypersonic flight conditions are required; the boundary layer must be included in these calculations because of the substantial displacement thicknesses at the high speeds that rob the scramjet of air -- thrust is proportional to air mass flow. The cycle calculation provides the internal engine performance and cowl exit conditions for starting the aft body nozzle flow field calculations which are constrained by an external flow boundary. Integration of the pressures on the aftbody wall provides the nozzle forces.

The net engine performance matrix (thrust coefficient and specific impulse as a function of Mach number, angle of attack and fuel equivalence ratio) is then assembled, with the thrust coefficients vectored along the vehicle wind axis and referenced to free stream static in the same manner as the aero coefficients. With this aero/propulsion performance set, the fuel fraction required to perform the ascent (98 percent of fuel requirement), orbital insertion, circularization, and deorbit is determined from the trajectory analysis.

Iterations are now required in the synthesis process to adjust the structure and insulation for the optimum (off-nominal) ascent and descent trajectory and vice versa and to perform an iteration on size/weight in the performance routine. Trim also comes into play here since the afterbody nozzle must be shaped to minimize the trim penalties, especially at the high speeds (Mach 10 plus) and, of course, there is a trajectory and structure coupling in this nozzle tuning.

CLOSURE

The closure of the synthesis process is represented in figure 6 in terms of fuel weight-fraction required and fuel weight-fraction achievable as a function of gross weight for an airbreather ascent to orbital conditions and return with a fixed payload. The closure point is where the two curves cross. The fuel-fraction-required line is nearly independent of gross weight; however, as the vehicle is scaled up geometrically, the increase in wing loading and resultant drag due to lift induces a slight positive slope. The fuel fraction achievable curve increases significantly with gross weight; at least to a point; the bending of the curve to the right (knee) at the larger gross weights is due to the negative influence of size on the structural efficiency. The closure point provides the gross weight/size of the vehicle -- and more: the magnitude of the difference in the slope of the two curves at the closure point is indicative of the margins achievable or viability of the vehicle to performing the mission. If the closure point is near the knee on the fuel-fraction-achievable curve, then a small increase in the fuel fraction require- ment to achieve orbit could move the closure point far to the right and substantially increase the gross weight of the vehicle required to perform the mission. In this undesirable closure region, the validity of conceptual design methods are suspect because of the extreme sensitivity; very high fidelity number sets are required to resolve the design.
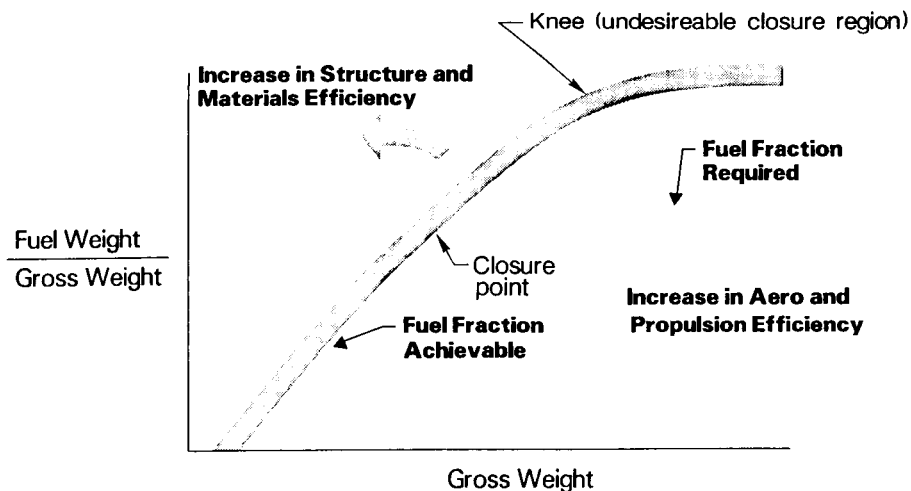

Figure 6

CONE CONFIGURATION SYNTHESIS

The conical configuration shown in figure 7 provides a good starting point/example with regard to configuration synthesis and the aero-space plane problem. In terms of desirable characteristics, its forebody, which provides an excellent precompression surface, also has a relatively thin boundary layer -- more mass flow and momentum to inlet. Also the circular cross-section is desirable from a structural perspective. More important, however, is the flexibility afforded by the conical configuration in such critical areas as engine inlet area which allows the necessary parametrics that provide understanding to the design problem. Also, the ability to make credible analytical predictions required for performance estimates because of the simplicity of the forebody shape is not a small advantage in starting with a conical configuration.
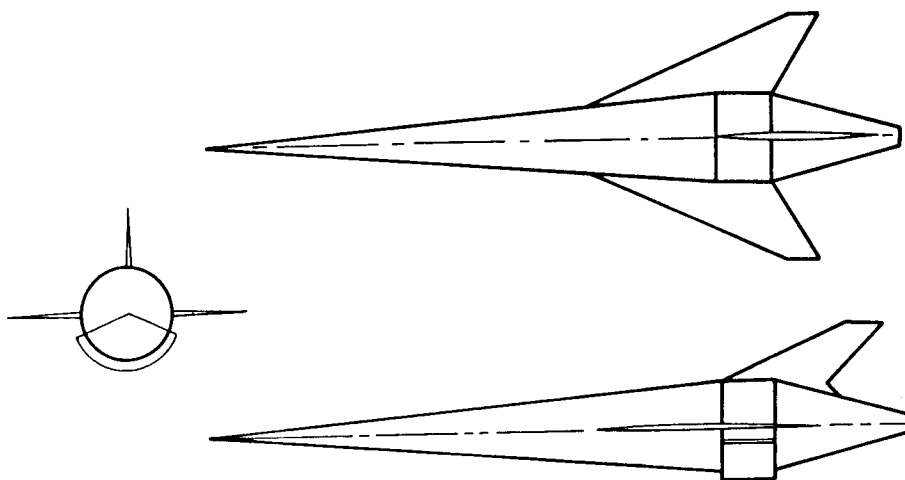
Figure 7

BOOKKEEPING

A force accounting scheme is selected -- free stream to free stream or inlet ramp to free stream. In the first, any surface that is worked by flow that goes through the engine is a propulsion surface; in the second, only the surfaces that are washed by engine flow from the beginning of the inlet ramping to the afterbody nozzle exit (free stream) is a propulsion surface. For this particular discussion, the classical route of free stream to free stream is used.

In the cycle analysis process, the increased pressure on the captured streamtube due to spillage at the cowl lip is not accounted for. This additive drag must be substracted from the thrust. Also, there is a spillage lift term which must be accounted for -- usually in the aerodynamic matrix.
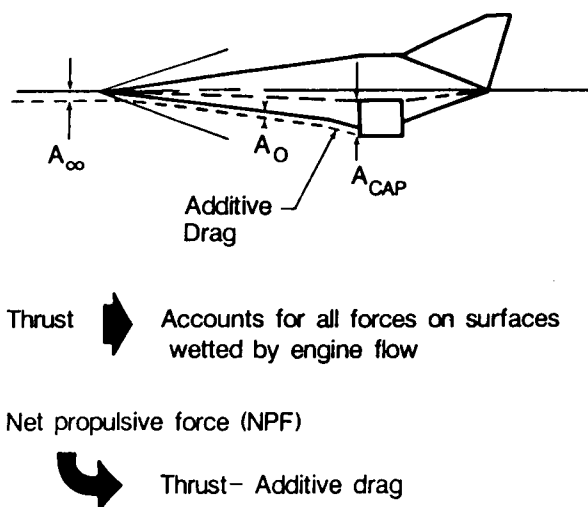

A_∞ A_O A_{CAP}

Additive Drag

Thrust ▶ Accounts for all forces on surfaces wetted by engine flow

Net propulsive force (NPF)

↳ Thrust- Additive drag

Figure 8

In the free-stream to free-stream accounting system, the larger the engine wrap angle, the more surface area that is accounted for in the propulsion matrix, as demonstrated in figure 9. Therefore, for a full engine wrap on a conical configuration, only the lifting and stabilizing/control surface appear in the aerodynamics.

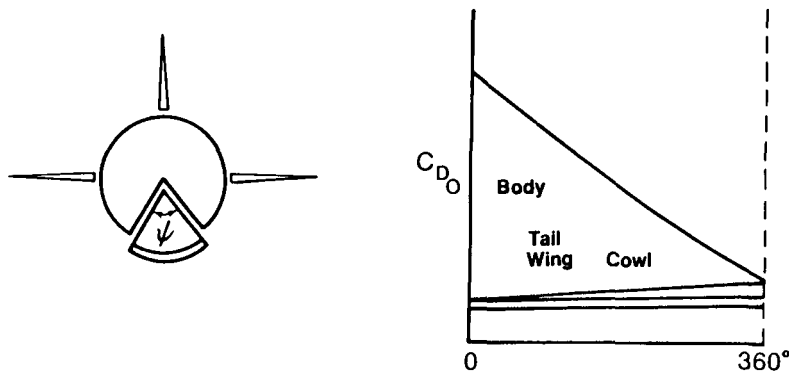**Vehicle Drag Includes All Surfaces Not Wetted By Engine Flow**

Figure 9

CONE CONFIGURATION PERFORMANCE

The cone configuration performance sensitivity can be ascertained from the w/P_s (fuel flow divided by specific excess power) distribution in figure 10 (the trend applies to much higher Mach numbers). Minimizing the area under the curve is minimizing the fuel consumed for the mission. Increasing the thrust to weight or the dynamic pressure (up to a point) for the cone reduces the fuel consumed. Increasing the throttle setting much above an equivalence ratio of 1 increases the fuel consumed (decreases engine I_{sp}).

$$T - D = (C_T - C_D) \; q \; A = \text{(thrust coefficient - drag coefficient)}$$
$$\times \; \text{(dynamic pressure)(reference area)}$$

where $C_D = f(M, \text{alt}, \alpha)$ and $C_T = f(M, \text{alt}, \alpha, \phi)$
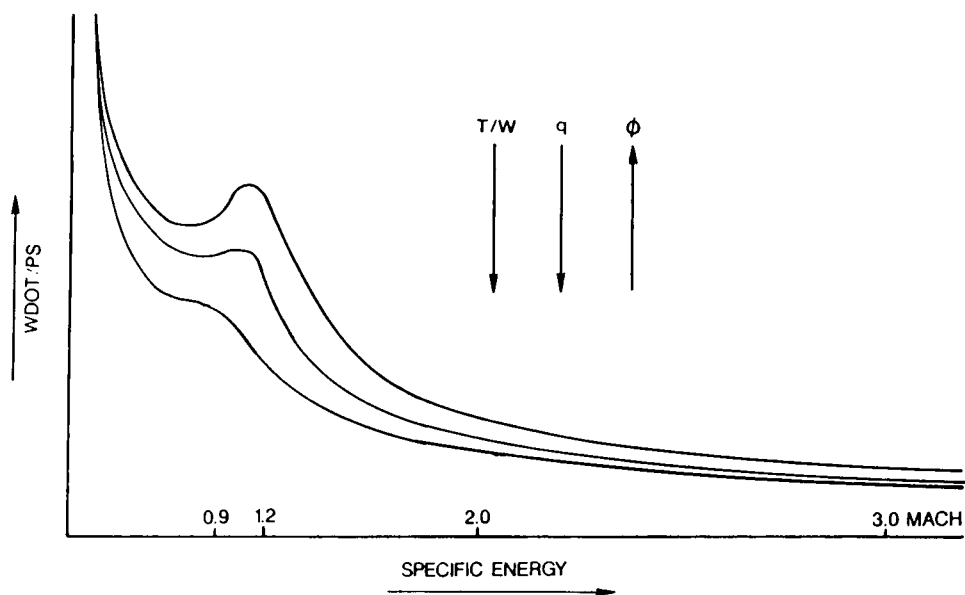
IMPACT OF T/W ON FUEL FLOW PARAMETER

Figure 10

DESIGN TRADES AND SENSITIVITIES

In the design process for SSTO's, the emphasis is on trades that will impact favorably on vehicle closure. Reduction in the fuel weight-fraction required can be realized with improvements in propulsion efficiency and reduction in vehicle drag. The influence of the drag on fuel fraction required is shown in figure 11 for a typical axisymmetric configuration. (The eight percent delta shown in fuel fraction required can be enormous in terms of closure capability.) The fuel weight-fraction achievable curve (fig. 6) moves to the left and rotates counter clockwise (increases) as the structural design and subsystems improve in efficiency and/or the materials advance in terms of strength-to-weight and stiffness-to-weight properties. The immediate discussion focuses on ways of reducing the fuel fraction required which will prove to have indirect and, in some cases, direct coupling to the fuel fraction achievable.

FUEL FRACTION REQUIREMENT SENSITIVITY
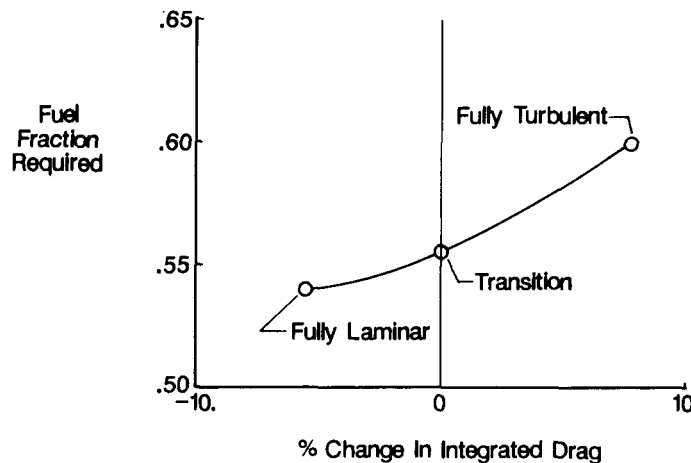## TO AIRFRAME DRAG LOSS
(CONSTANT q TRAJECTORY)

Figure 11

THRUST MARGIN

For an accelerating vehicle, the time derivative of its specific energy is equal to its specific excess power. That is:

velocity $\searrow$ $\swarrow$ thrust margin

$$\frac{d}{dt}(\frac{V^2}{2} + gh) = \frac{V(T-D)}{W} = P_s = \text{specific excess power}$$

weight

Increasing the thrust margin and/or decreasing the weight of the vehicle for a given velocity increases the instantaneous energy imparted to the vehicle and, as shown in the preceeding article, increasing the ratio of the propulsion energy imparted to the vehicle to that left in the atmosphere reduces the fuel fraction required. The thrust margin is the difference between two large numbers, thrust and drag, which makes it sensitive to small changes in either; both are functions of dynamic pressure:

reference area

$$T - D = (C_T - C_D)qA$$

dynamic pressure

Increasing the flight dynamic pressure by flying lower in the atmosphere increases the thrust margin assuming constant thrust and drag coefficients. But the vehicle drag coefficient decreases with increasing dynamic pressure because of the reduction in skin friction coefficient with increasing Reynolds number (the caveat here is boundary-layer transition) and lower drag due-to-lift with decreasing angle of attack since the angle of attack decreases with increasing dynamic pressure in order to maintain a given lift. Also, the thrust coefficient increases with increasing dynamic pressure because of a favorable trend in the ratio of inviscid to viscous forces inside the scramjet engine. Increasing dynamic pressure at the high Mach numbers also maintains a given pressure in the engine combustor at lower inlet contraction ratios so that less energy is lost to gas kinetics in the nozzle expansion process.

However, increasing flight dynamic pressure is advantageous only so long as the structure/weight of the vehicle is not unduly affected, which can easily happen because of increased heating rates, loads, and flutter tendencies. Also, the advantages and disadvantages of increasing dynamic pressure are configuration dependent. For example, the thrust margin of axisymmetric configurations should benefit from higher dynamic pressure because these vehicles are being driven toward zero angle of attack where they perform best. This is indicated in figure 12 where the nondimensional take-off gross weight for such a configuration is shown to decrease substantially with increasing dynamic pressure of the trajectory. (The caveats here are boundary layer transition and weight of engines, actively cooled airframe surface area, etc.) On the other hand, the thrust margin for vehicles with underslung engines, such as that shown in figure 1, peaks at modest angles of attack; the thrust increases faster than drag with properly shaped forebodies up to some small angle of attack because of the increase in the air flow and pressure recovery to the inlet system. Any increase in flight dynamic pressure that drives the angle of attack below that for which the thrust margin peaks is detrimental. (This type vehicle could be shaped for high dynamic pressure trajectories but the fineness ratio may be driven to a point of diminishing return.)

EFFECT OF DYNAMIC PRESSURE ON GROSS WEIGHT
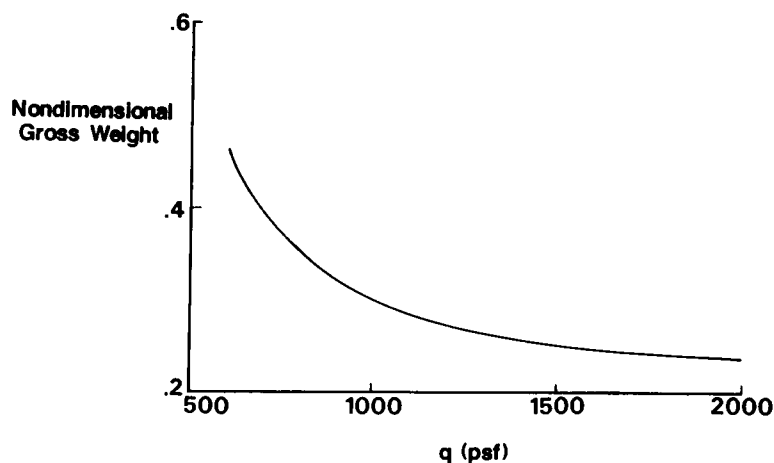## FOR AN AXISYMMETRIC VEHICLE

Figure 12

INLET AREA

Of course, there are ways of increasing the thrust margin of these vehicles other than just increasing dynamic pressure -- increasing inlet area and/or air capture area, increasing the fuel equivalence ratio beyond stoichiometric in the combustor, or rocket augmentation. For a given vehicle shape and size, increasing the inlet area decreases the fuel weight-fraction required, but it also decreases the fuel weight-fraction achievable because the engine weight, and thus vehicle dry weight, is increasing (not necessarily linearly) while the fuel weight remains constant. So, as illustrated in figure 13 for a given size vehicle, there is an optimum inlet area that maximizes the payload weight-fraction deliverable to orbit. This is also the case for a vehicle optimized to deliver a fixed payload to orbit as indicated in terms of TOGW (take-off gross weight) in figure 14.

EFFECT OF INLET AREA ON WEIGHT FRACTION
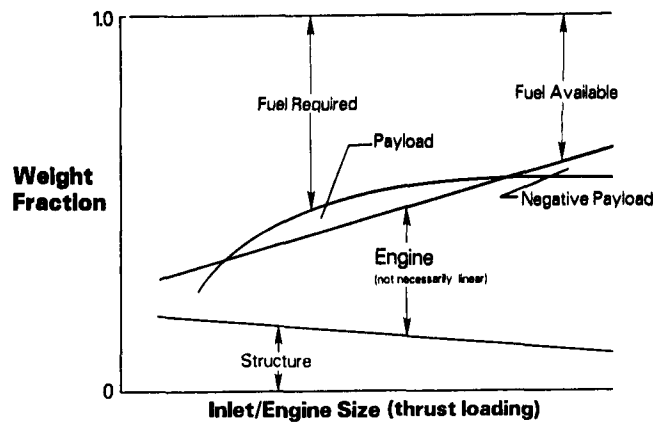
(Fixed size vehicle)

Figure 13

EFFECT OF INLET AREA ON TOGW
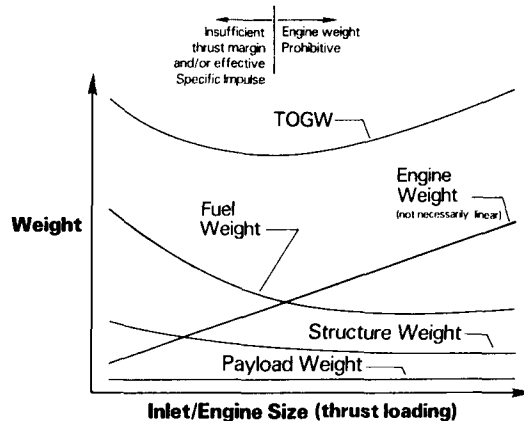
(Fixed payload size/weight)

Figure 14

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INLET AREA (Continued)

The optimum inlet area depends not only on the engine weight per unit inlet area, but on the engine performance per unit inlet area which is affected by where and how the inlet area is added. The inlet area can be added such that the cowl lip is kept within the shock layer throughout the airbreathing ascent or oversized such that the vehicle bow shock crosses the cowl lip at some designated top-end Mach number, and more of the inlet area protrudes into the free stream as the acceleration proceeds (inlet overspeed). In the first situation there are more limits on increases in inlet area since the shock layer has only a finite amount of thickness at the top end Mach numbers; also, the inlet air capture suffers at the lower Mach numbers. For the overspeed case, more inlet area is possible, and the air capture is greater at the lower Mach numbers; however, the mass flow per unit inlet area is less at the high Mach numbers and so is engine efficiency, but not thrust, since the inlet area is larger.

Rather than, or in addition to, increasing the physical inlet area to facilitate thrust margin, the inlet air capture area can be increased by the optimization of the forebody precompression contour and the trim attitude of the vehicle -- the result of effective engine/airframe integration. The objective is to maximize the capture area while minimizing the parasitic drag area (surfaces that compress air-flow that does not pass through the engines) and still provide the appropriate lift to sustain the vehicle in the airbreathing corridor.

THROTTLE

As for increasing the fuel equivalence ratio beyond stoichiometric, such may be required at the very high Mach numbers to cool the engine, but fortunately the decreases in specific impulse that nominally accompany fuel rich conditions are somewhat nulled; at the very high speeds the thrust benefits of mass injection of the hot, low-molecular-weight hydrogen can be very significant.

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ANALYTICAL TOOLS FOR HYPERSONIC VEHICLE DESIGN

AEROSPACE VEHICLE INTERACTIVE DESIGN

Vehicle design codes that consist of an executive with interfaces to geometry generation and to discipline data sets or data set generation capability are essential in conceptual design studies of hypersonic vehicles because of the many variables and couplings involved. The Aerospace Vehicle Interactive Design (AVID) (ref. 2) is a computer-aided design system based on designer participation. Its development began in the mid 1970s using interactive graphics on a minicomputer for geometry modeling of configurations and for interpreting a large volume of data generated on a mainframe computer. The current architecture of the AVID system is shown in figure 15. The core system consists of four separate modules. The key module is the engineering data management system that controls all data and programs. The user interface module aids in the utilization of the system by providing a standard set of commands for system operation. The program interface module utilizes a standard technique for integrating analysis programs into the system or short-circuiting to data sets generated externally. The final module is the geometry system for generating, displaying, modifying and sizing both externally and internally generated configuration data.
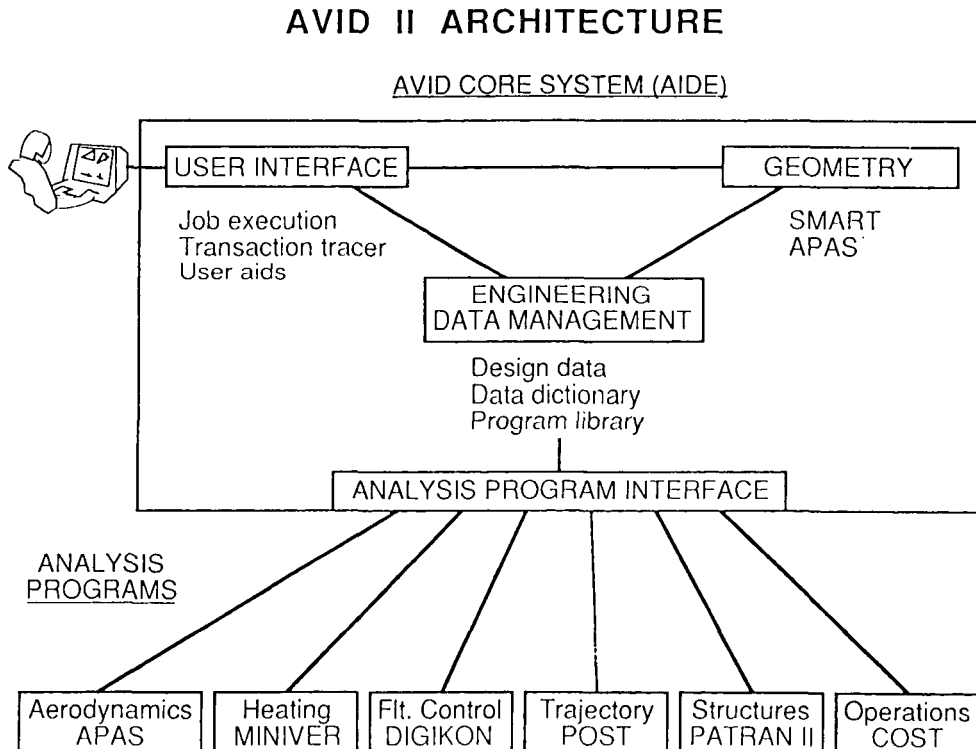
AVID II ARCHITECTURE

AVID CORE SYSTEM (AIDE)

Figure 15

AVID GEOMETRY

The geometry capabilities (figure 16) in AVID[*] include external lofting, internal arrangement, and geometry analysis. Present geometry programs in the AVID network are APAS, CDS, GEOMOD, and SMART.

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[*]Advanced Vehicle Interactive Design

- Capabilities
  - External lofting (creative and duplication modes)
  - Internal arrangement
  - Geometric analysis (areas, volume, cg's, I's)
- Present programs
  - Aerodynamic Preliminary Analysis System (APAS-Rockwell)
  - Configuration Development System (CDS-Rockwell Proprietary)
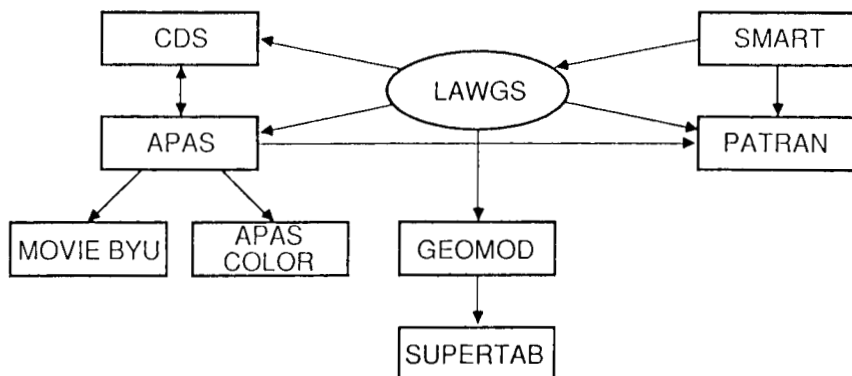 - GEOMOD (SDRC-vendor)
  - SMART - LaRC Real Time Solid Modeling

Figure 16

AVID's AERODYNAMIC PREDICTION CAPABILITY (APAS)
(As conveyed by Alan W. Wilhite, NASA Langley Research Center)

The Aerodynamic Preliminary Analysis (APAS) (ref. 4) is used to create a total aerodynamic profile for trajectory analysis. In the subsonic/supersonic region, slender-body theory is used to predict fuselage forces and vortex panels to predict wing/tail forces. Skin-friction, wave, and base-drag theories are combined with induced drag to predict total configuration drag. For high speeds, the Hypersonic Arbitrary Body (HAB) program has been integrated into APAS (figure 17). APAS program capabilities and related programs can be seen in figure 18.

AERODYNAMIC PRELIMINARY ANALYSIS SYSTEM
(APAS)

Rockwell developed and is using for NASP studies

Subsonic/supersonic analysis

    Distribute vortex panels with leading edge suction

    Slender body theory

    Laminar - Blasius with Echert's compressibility

    Turbulent - Van Driest

    Wave drag at angle of attack

    Hoerner corrections for thickness

    Base drag derived from Shuttle databook

Hypersonic arbitrary body program

Figure 17

INTERACTIVE APAS

- Geometry
  - Digitizing
  - Interactive
  - Editing
  - Component panelling
  - Display

- Analysis
  - Geometric parameters
  - Wave drag
  - Viscous drag

- Analysis setup
  - Mach, altitude, α sweep, β
  - Hypersonic method selection
  - Analysis model display

- Output display
  - Coefficients
  - C_p

BATCH APAS

- UDP
  - Vortex panel
  - Viscous drag
  - Wave drag
  - Base Drag

- HABP
  - Impact methods
  - Viscous drag/heating

RELATED PROGRAMS

- Movie BYU – Shaded image

- PATRAN – C_p, T_w display

Figure 18

The minimum drag coefficient on a five-degree half-angle cone configuration is given in figure 19 as a function of Mach number. Base drag, wavedrag, and viscous plus profile drag are shown. The flipper-door drag is that which resulted from the inward deflection of a flap at the trailing edge of the cowl in order to keep the afterbody nozzle plume attached (fill the nozzle) at transonic speeds.

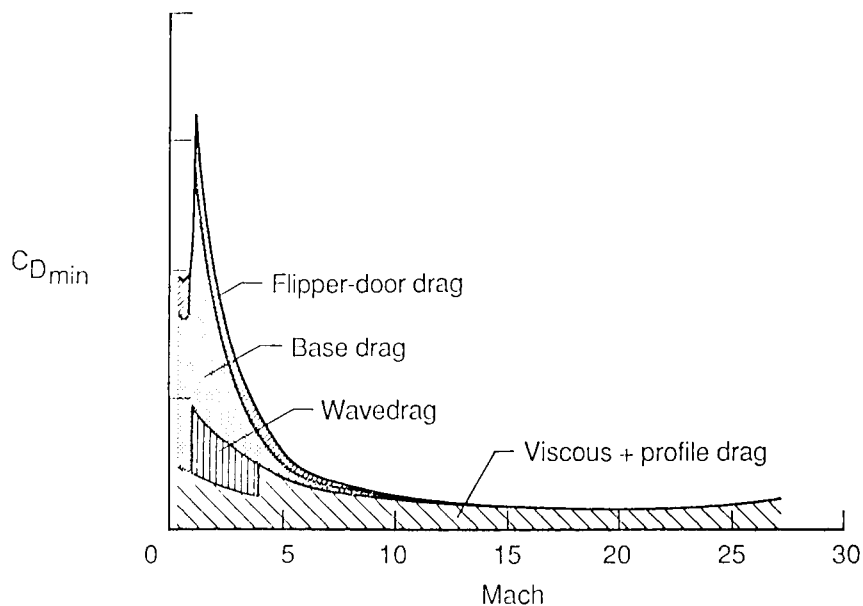## 5° CONE CONFIGURATION DRAG PREDICTIONS

Figure 19

APAS DRAG PREDICTION ON CONE

Minimum drag coefficient is presented as a function of Mach number as shown in figure 20. The APAS predictions (UDP, unified dispersive panel -- vortex panel, viscous drag, wavedrag, and base drag, and the Hypersonic Arbitrary Body Code) are compared with wind tunnel data.
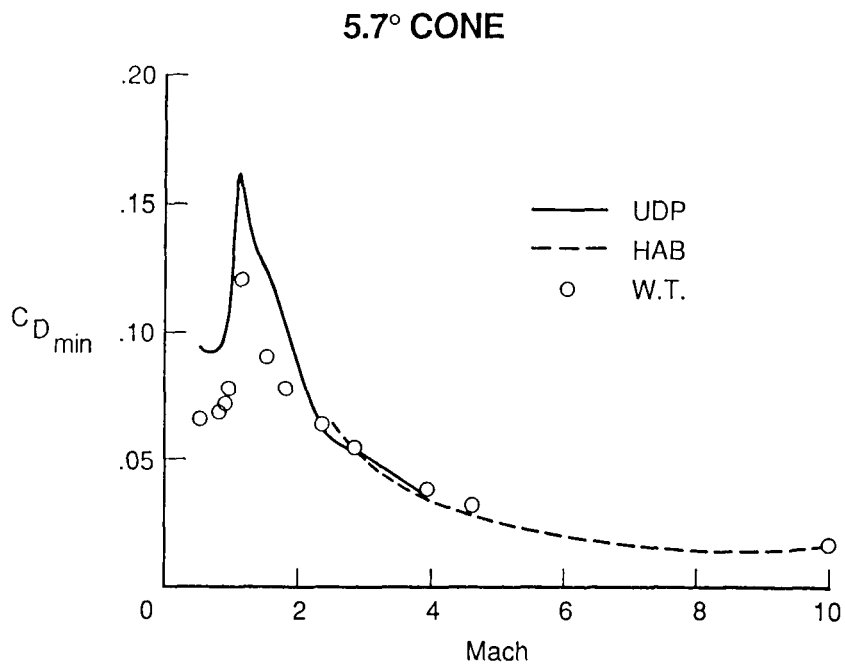
5.7° CONE

Figure 20

AERODYNAMIC COEFFICIENTS COMPARISON

A comparison of the aerodynamics generated on a cone with APAS and a PNS code are shown in figure 21 for Mach 20. A comparison of viscous and inviscid drag contributions, as calculated by APAS and PNS code for a cone at Mach 15 is shown in figure 22.

5° SPHERE-CONE

AOA	CODE	C_A	C_N	C_L	C_D	L/D	C_M
0	APAS	0.0232	0.0	0.0	0.0232	0.0	0.0
	PNS	0.0204	0.0	0.0	0.0204	0.0	0.0
2.5	APAS	0.0251	0.0920	0.0909	0.0291	3.12	-0.7068
	PNS	0.0218	0.0833	0.0823	0.0254	3.24	-0.6412
5.0	APAS	0.0308	0.1849	0.1815	0.0468	3.88	-1.4190
	PNS	0.0256	0.1681	0.1652	0.0405	4.08	-1.2920

Figure 21

5° CONE

$L = 140$ ft, $R_{NOSE} = 0.125$ ft

$MACH = 15$, $ALT = 150K$ ft
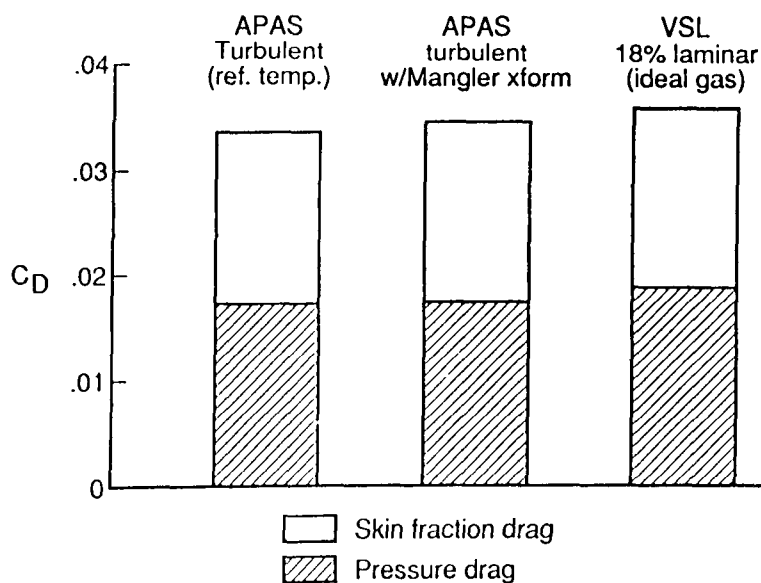
Figure 22

Heating predictions on a 5° half-angle cone assuming a laminar boundary layer is shown in figure 23. The difference in the Mark 3 Reference Enthalpy and the Mark 3B Reference Enthalpy levels is mainly that of the Mangler transformation (Mark 3B). The Mark 3B predictions agree with those given by the Viscous Shock Layer code. Heating predictions assuming a turbulent boundary layer are shown in figure 24.

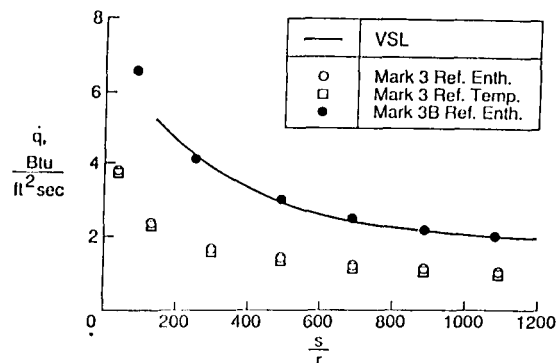**5° CONE LAMINAR BOUNDARY LAYER CONVECTIVE HEATING RATE**

Mach 15, α = 0°

Figure 23

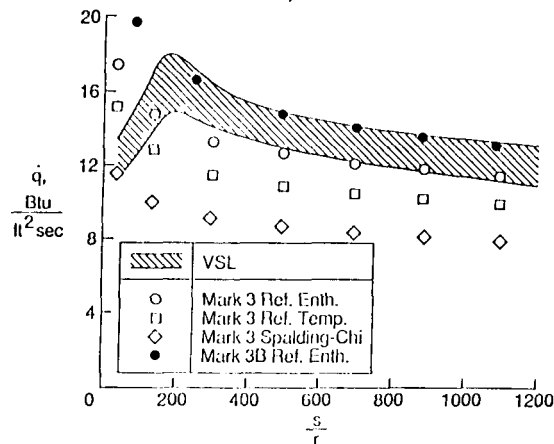**5° CONE TURBULENT BOUNDARY LAYER CONVECTIVE HEATING RATE**

Mach 15, α = 0°

Figure 24

TRANSITION CRITERIA

High Mach numbers tend to laminarize the flow on a 5° half-angle cone as indicated by the merger of the solid line drag prediction containing Beckwith's transition criteria (figure 25) with the dashed line representing predictions for laminar flow, as shown in figure 26. Thus above Mach 14, the flow on the cone appears to be all laminar. This is for a trajectory having a dynamic pressure of 1,000 psf.
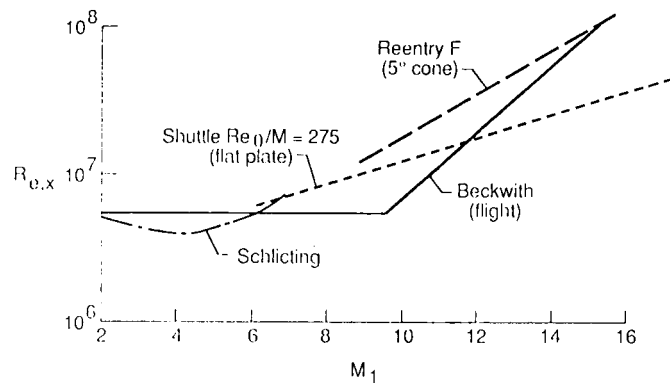
APPROXIMATE TRANSITION CRITERIA FOR APAS

Figure 25

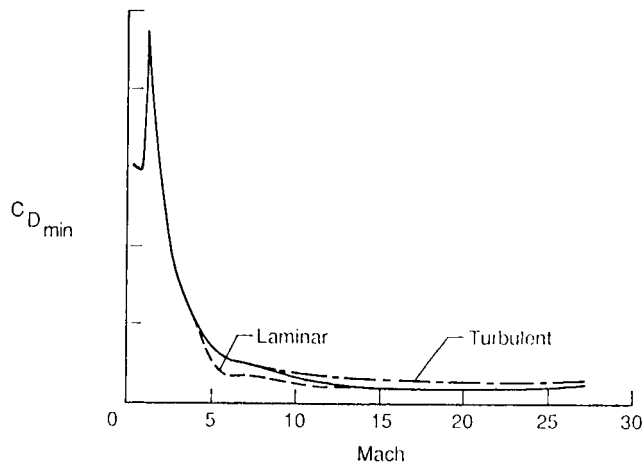## 5° CONE CONFIGURATION DRAG WITH TRANSITION

Figure 26

TRAJECTORY ANALYSIS (POST)
(As conveyed by Richard W. Powell, NASA Langley Research Center)

PROGRAM TO OPTIMIZE SIMULATED TRAJECTORIES (ref. 5)
(POST)

1. Three degree of freedom version

2. Suitable for ascent, entry, and orbital problems

3. Multiple guidance options and integration techniques

4. Powered (rocket and airbreathing) or unpowered vehicles

5. Option to calculate engine gimbal angles or flap deflections required to balance moments due to thrusting and aerodynamics

6. Simulate winds, horizontal take-off, hold down for vertical take-offs

7. Optimizes trajectory while meeting equality or inequality constraints

8. Optimization and constraint variables can be any calculated variable

PROGRAM TO OPTIMIZE SIMULATED TRAJECTORIES (POST)
APPLICATIONS TO NASP

A. CAPABILITIES

1. Both a 3 degree-of-freedom and a 6 degree-of-freedom version are available.

2. Flexible enough to apply to virtually any aerospace trajectory problem (ascent-orbital maneuvers, entry).

3. General targeting (including both equality and inequality constraints) and optimization capability.

4. Optimization criteria, constraints, and controls may be virtually any input or calculated parameter.

5. Modularity design allows for easy modification or addition of mathematical models.

6. Becoming an industry standard.

B. ENHANCEMENTS FOR NASP STUDIES

1. Propulsion module updated to simulate air-breathing propulsion used by candidate NASP vehicles.

2. Guidance system modified to allow for easy acquisition of desired dynamic pressure profile.

3. Additional output variables are calculated (ISP, effective ISP, propulsive efficiency, etc.)

PROPULSION FOR INPUT TO POST

The propulsion data set is generated external to AVID. The procedure and tools used in generation of this input performance data set for the scramjet is shown in figure 27. Flow field solutions (CFD) to check the forebody/inlet starting profiles for the scramjet analysis are also generated external using VSL, PNS, TLNS, and FNS codes.
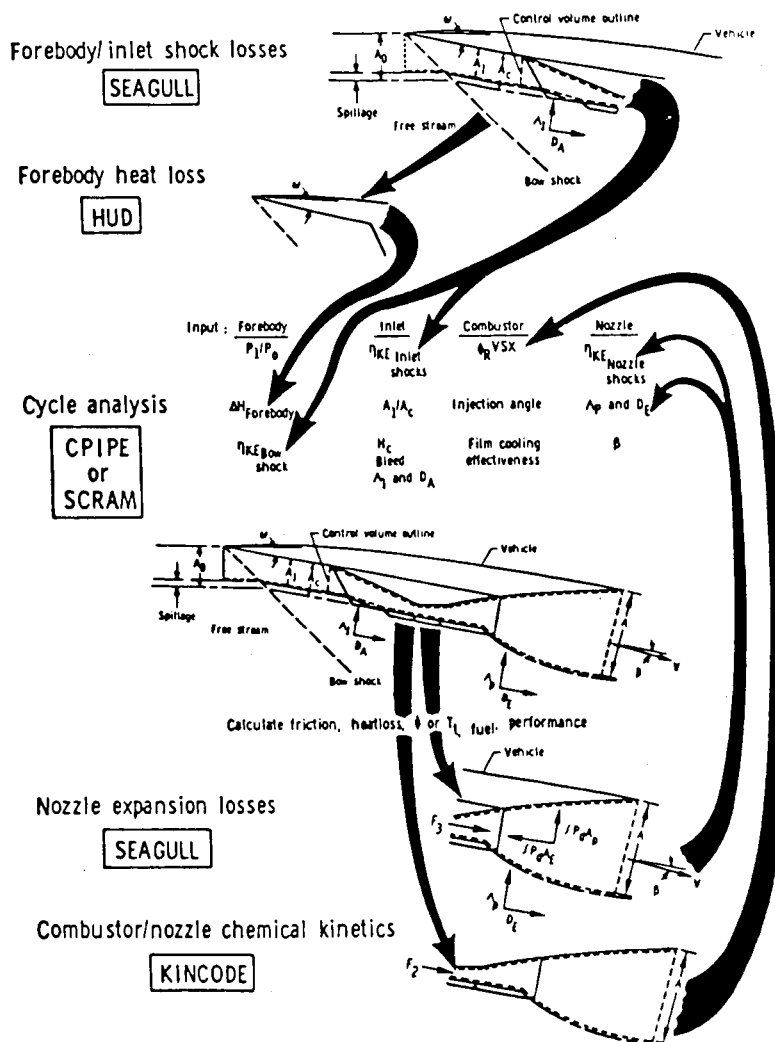
SCRAMJET PERFORMANCE ANALYSIS

Forebody/inlet shock losses
[SEAGULL]

Forebody heat loss
[HUD]

Cycle analysis
[CPIPE or SCRAM]

Nozzle expansion losses
[SEAGULL]

Combustor/nozzle chemical kinetics
[KINCODE]

Figure 27

COMPUTATIONAL FLUID DYNAMICS (CFD)

The flow field over the vehicle is calculated external to AVID with CFL3D (ref. 6). This is a thin layer Navier Stokes program that utilizes an upwind difference scheme; integration is performed in the physical plane. The primary purpose of calibrating the flow field is to provide a high fidelity solution of the air-flow properties at the inlet face and eventually at the inlet throat in order to adjust the scramjet performance analysis. These solutions are increasingly being sought to calibrate/adjust aerodynamic and heat transfer data sets generated from less sophisticated means.

AEROTHERMODYNAMICS

The aerothermodynamic slot in APAS makes use of both internal and external data generation sources. The MINIVER code is used to provide engineering heating predictions with such methods as Fay and Ridell (stagnation point), Cohen and Beckwith (leading edge), and Shultz and Grueno (fuselage). This analysis is agumented with the CFD solutions mentioned earlier.

THERMAL MANAGEMENT

The goal of the thermal management analysis effort is to analyze hypersonic vehicle concepts with realistic thermal loads applied and realistic thermal management system installed to obtain temperature distributions, cooling loads, hydrogen flow conditions, system weights, and system volumes. Once the thermal management system is designed and integrated, the challenge is a thermal balance of the vehicle that sets the fuel cooling equivalence ratio of the vehicle and the delta on engine performance due to the heat addition to the hydrogen before injection into the engine.

The tasks are to develop and/or obtain:

1. Surface heat loads for airframe and engine. (From MINIVER and SRGULL).

2. Thermal model of overall vehicle. (PATRAN generated condition models and translated into SINDA.)

3. Thermal model of coolant flow network. (Established in SINDA -- uses GASPLUS for fluid properties.)

4. Engine and airframe temperature. (From SINDA)

5. Hydrogen network flow rates, temperatures, and pressure. (From SINDA, ref. 7.)

STRUCTURAL ANALYSIS, WEIGHTS

The structural analysis is performed external to AVID. Weight of the structural architecture is estimated through a finite element/failure mode analysis (ref. 8). The procedure is as given below.

LOADS AND FAILURE MODE WEIGHT PREDICTION METHODOLOGY

A. Create a PATRAN finite element model of the desired component and include:

    1. nodes and connectivities
    2. rigid masses
    3. external loads
       a. distributed and point forces
       b. temperature loading
       c. inertial loading
    4. constraint cases
    5. element construction type, and material data
       a. bar
       b. beam
       c. honeycomb plate
       d. corrugated web
       e. hat stiffened skin

B. Translate PATRAN data to an EAL runstream.

C. Run the model through each applied loadset, or loadset combination. Use the element sizing code to calculate structural gages based on, minimum gage, buckling, yield, and ultimate strength design criteria.

D. Summarize calculated gages for each loadset and create a file of new element dimensions based on the heaviest of each element for each loadset.

E. Update the EAL runstream with new element stiffnesses reflecting dimensions from the worst case element dimension file. Repeat steps C through E until element dimensions remain unchanged between iterations.

F. Postprocess the converged element dimensions with appropriate non-optimum factors to permit the integration of calculated structural weights into a vehicle performance sizing program.

SIZING

Vehicle sizing is conducted external to AVID. The primary criteria for sizing a vehicle is propellant mass fraction (propellant weight/take-off gross weight). Vehicles are scaled to achieve a given (required) propellant mass fraction (PMF) as described below:

1. The vehicles airframe, wings, and tails are scaled photographically. Structural weight is based on unit weight scaling laws as determined by structural analysis conducted on various size vehicles; the weight per unit area of various components are fit to a quadratic equation form ($C_1 + C_2\ell + C_3\ell^2$) where ℓ is a nondimensional representative length or scale factor.

2. Engine weight and volume is scaled by engine inlet area which is scaled photographically, W_{Engine} = Const. × Inlet Area.

3. Subsystems weights and volumes are based on emperical/historical equations with advanced technology factors included.

4. Payload bay and crew compartment are fixed.

5. Available volume for propellant tanks is the total vehicle volume minus the volumes of items 1 through 4. Propellant volume available is the tank volume minus volume lost to tank packaging efficiency.

6. The vehicle is then scaled up or down in an iterative manner until a given PMF is achieved.

CLOSURE

The fuel fraction required as calculated from the POST trajectory code (for ascent, transition to orbit, orbit, deorbit, reentry and decent, and landing -- complete trajectory) and the fuel fraction achievable as calculated from the sizing code provide the closure point as indicated in figure 28.
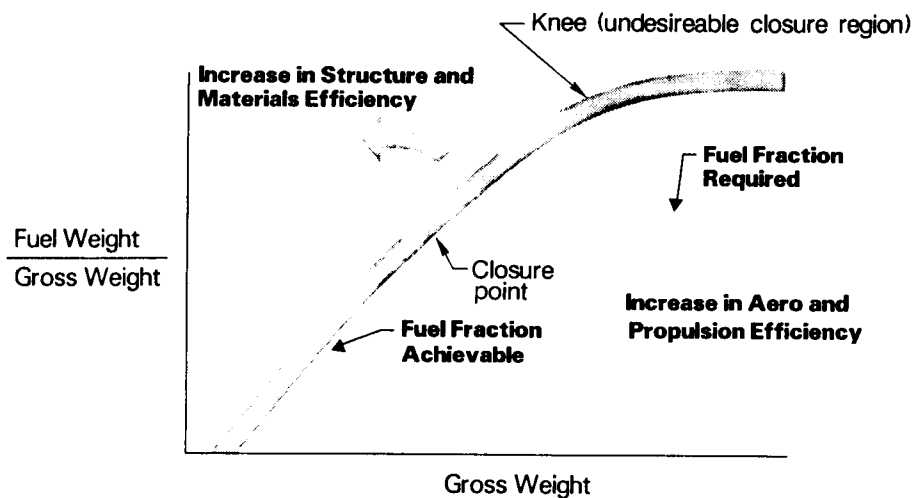

Figure 28

CONCLUDING REMARKS

The airbreathing SSTO vehicle design environment is variable-rich, intricately networked and sensitivity intensive. As such, it represents a tremendous technology challenge. Creating a viable design will require sophisticated configuration/synthesis and the synergistic integration of advanced technologies across the discipline spectrum. In design exercises, reductions in the fuel weight-fraction requirements projected for an orbital vehicle concept can result from improvements in aerodynamics/controls, propulsion efficiencies and trajectory optimization; also, gains in the fuel weight-fraction achievable for such a concept can result from improvements in structural design, heat management techniques, and material properties. As these technology advances take place, closure on a viable vehicle design will be realizable.

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