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WORKING DRAFT

ASSESSMENT OF NASP: FUTURE OPTIONS

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PREFACE

This Working Draft consists of an outline presenting an alternative structure for the National Aerospace Plane (NASP) program. This examination of an alternative structure was initiated in response to the recent re-examination of the NASP program within the Department of Defense.

The work draws on the results of a continuing project, The National Aerospace Plane (NASP): Development Issues for the Follow-on Systems, RPN 4523, which was preformed within the Technology Applications Program of Project AIR FORCE.

This work may be of interest to those structuring the NASP program.

ASSESSMENT OF NASP-FUTURE OPTIONS

I. Background and Summary

We are winding up a PAF study (RCN 4523) on NASP this year. From this study we are tending to certain observations which bear on recent DoD decisions on NASP, and on the NASP deliberations which the Space Council and Congress are initiating.

Briefly, we believe that of the array of NASP program options which will likely be presented to the Council (ranging from full restoration of the NASP X-30 program, to effectively a NASA R&D program), a specific option which preserves joint DoD/NASA management of a significantly redirected and reduced annual cost program warrants close attention.

Such a program is described in the following discussion. It can actually add critical research efforts, even though the overall program pace is reduced.

II. RAND NASP Study Observations

1. Our NASP study to date suggests typical findings:

a. NASP Utility and Applications

- No compelling "golden mission" exists for NASP
- Competitors/counterparts exist for important NASP missions:
 - ALS & NASA - NASP space launch
 - ASAT plans - NASP space control
 - DARPA Pegasus - NASP quick reaction, all azimuth capability
 - NTM stable of assets - NASP surveillance capabilities
 - Others

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- NASP could do many missions (but none is singularly persuasive); a very small fleet might serve NCA, SecDef in opportunistic roles
 - NASP mission utility determination depends on currently unfunded subsystem work.
- b. NASP Program
- Grave doubts exist that NASP could come anywhere near its stated/advertised cost, schedule, payload fees to orbit, etc.
 - Schedule driven technology plan is extremely fragile, and contains unrealistic expectations.
 - A very ambitious acquisition plan for coupling X-30 with operational vehicles is proposed.
- c. NASP technology
- Primary NASP X-30 objective - manned single stage to orbit (MSSTO) - is exceedingly sensitive to full success in technology maturation; the hypersonic cruiser mode is more forgiving.
 - Outgrowths of NASP technology drivers - high performance aeropropulsion, high temperature light structures, new design/analysis tools, integration experience with many demanding systems - will have relevance in areas such as many future military aircraft, possible commercial aircraft, industry fabrication and manufacturing advances, and general bolstering of U.S. technology base.
 - The NASP program gives the U.S. opportunities to focus on many specific technologies of industrial and economic importance which might otherwise lie fallow.

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2. From such findings two conclusions emerge requiring resolution:

- On the basis of current knowledge, it is hard to defend previous DoD plans for NASP on the basis of any singular mission utility sufficiently attractive to operators.
- But: a technology driven program could be constructed which matures many relevant technologies - albeit at a reduced pace - of intrinsic interest to the U.S., and which would also prevent "surprises" from German, French, Japanese, British and Soviet counterpart programs.

It seems to us that a Space Council/Congressional/Departmental decision on the future of NASP should accommodate these two conclusions, to best serve national interests.

In this connection, one can note that if a DoD NASP program disappeared entirely, a NASA hypersonic technology program currently exists which could continue to advance the hypersonic art (assuming NASA in turn continues its funding). The NASA program suffers from several apparent defects, however:

- It is primarily a fairly basic R&D program, the residue of earlier hypersonic vehicle work.
- It does not focus adequately on meeting major milestones and subsystem/system tests crucial for both military and commercial development decisions.
- Doubt exists in some minds that NASA management would or could step up to accomplishing such milestones and tests. Therefore the current NASP decision - devolve the residual program entirely to NASA - does not seem a preferred strategy.

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III. What might a preferred NASP strategy look like? A NASP strategy worth serious consideration would be:

1. Maintain joint DoD/NASA program direction, in which, for example, NASA might monitor relevant R&D, and DoD would develop and monitor major milestone/experiment/test plans for prospective applications.
2. Redirect the current NASP program, and formulate a coherent master plan to continue fundamental NASP efforts. Such a redirection might include, for example:

- a. Reduced pace technology maturation (late 90s instead of 19947)

- b. Initial reduced scale, unmanned flight testing; for example:

- Module tests of aeropropulsion in Pegasus, other vehicles?
- Component tests where possible of sensors, instruments, comm links, payload separation, controls, observables, etc. - i.e., realistic exploration of the hypersonic environment impact on operations.
- Reevaluation of hypersonic cruiser, SSTO objectives in a phased program.
- Compilation of data adequate to support fully a sub-scale or full scale manned vehicle construction and flight test series prior to commitment to such a series.

4. The content of such a redirected program would be rather different from the initial natural reactions of NASP devotees to a reduced funding level. The role of reduced scale, unmanned testing is complex to explore; the relevant arguments and responses would warrant separate treatment. The formulation of such a redirected

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program into a sensibly paced, event driven, technology maturation program accomplishing critical experiments culminating in the late 1990s is a hard job warranting close attention - but the job appears doable to us.

4. Without the availability of the redirected plan, the cost estimates to attain the goals of 2. and 3. above are imprecise. However, a reasonable initial guess is that perhaps \$200-300 million/year would mature the requisite technology by the late 1990s if initial emphasis includes subscale unmanned flight experiments. The numbers currently suggested - \$100 M from DoD and estimated \$120 M from NASA - would then coincidentally be roughly adequate if maintained annually and jointly to meet late 1990s technology maturation dates, with information then being adequate for operators' decisions on NASP continuance, and the later evolution of discrete fleet sizes.

IV. What are the potential merits of such a NASP redirection strategy?

- a. Maintain vigorous technology base - research - experiment - demonstration program; the latter two aspects are needed to sustain interest.
- b. Defer major funding decisions until data base is adequate.
- c. Reduce the evident risks of recent/current NASP program:
 - Excessive schedule driven technology risks
 - Excessive USAF funding share, both absolute and as percentage of USAF RDT&E funds
 - Susceptibility to political targeting of very high visibility program
 - Scant/no attention to military subsystem testing

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- Excessive coupling of concurrency, short schedule, and technical risks in a single program
 - d. Increase the data base for follow-on and operational decisions:
 - Experiment with applications systems
 - Develop insights into utility as well as technology
 - Let missions, fleet sizes evolve, if at all, on basis of hard experiments and better defined cost data
 - e. Program phasing allows many decision points
 - f. More compatible with current budget environment
- V. The suggested NASP redirection plan is premised on these understandings:
- Prudent to maintain mainstream U.S. aerospaceplane role
 - Joint DoD/NASA program management combines comparative advantages
 - A great deal learnable from subscale/unmanned tests
 - Attention to formulation of coherent flexible master test plan

All in all, the initial DRB zeroing of the NASP budget (and the proposed reinsertion of \$100 M of DoD money) can be regarded as an opportunity to proceed with a more realistic NASP program which incidentally adds efforts to plumb issues of utility during an extended experiment phase.

EXCERPTS FROM TONY DUPONT'S LECTURE
AT THE UNIVERSITY OF CALIFORNIA IRVINE
ON JUNE 20, 1990

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INTRODUCTION AND ACKNOWLEDGEMENTS

The following pages are lecture notes prepared by Dr. Tom Heppenheimer based on a paper by Bob Jones from NASA Langley. The idea of using propulsion energy efficiency to simplify the analysis was suggested by Dr. Coleman Donaldson. Amendments have been made to Heppenheimer's original notes by Tony duPont to account for a higher level of propulsion efficiency than was originally used. Propulsive efficiency is a very powerful tool for gaining a fundamental insight into the aerospaceplane propulsion system and its impact on the aircraft design.

The propulsive efficiency is used as a constant in the analysis, but it is only constant at the higher speeds where SCRAMJETS are used. At zero speed it is zero, and increases as speed increases leveling off at a constant value in the SCRAMJET regime. SCRAMJET engines can potentially have very high propulsion efficiency because the exhaust is very close to flight speed and the engine has a very high static pressure ratio. SCRAMJETS with inlet and nozzle contours that minimize shock losses can have propulsive efficiencies approaching 70 per cent.

The impact of required fuel fraction on aircraft size is quite dramatic and points to strong emphasis on attaining high propulsion efficiency. The write up on cycle analysis following Heppenheimer's notes contains an example of a high efficiency SCRAMJET inlet designed using principles originally suggested by Oswatitsch.

CONSIDER THE AEROSPACE PLANE

- ENGINE THRUST DEPENDS ON ATMOSPHERIC DENSITY
- VEHICLE DRAG DEPENDS ON ATMOSPHERIC DENSITY
- HUGE CHANGE IN KINETIC ENERGY IS REQUIRED
- IT MUST ACCELERATE WELL WITHIN THE ATMOSPHERE TO VERY HIGH SPEED
- IT MUST ALSO MINIMIZE THE AMOUNT OF ENERGY LEFT BEHIND IN THE ATMOSPHERE AS HEAT

What is required to do this?

What are the governing parameters?

REQUIRED ENERGY IS DIRECTLY PROPORTIONAL TO M^2

$$\text{At } M=8: \left(\frac{8}{25}\right)^2 = 0.10$$

- 90% of the energy must be acquired above MACH 8

$$\text{At } M=15: \left(\frac{15}{25}\right)^2 = 0.36$$

- 2/3 of the energy must be acquired above MACH 15
- In addition to the energy the vehicle must have, there is energy left behind in the atmosphere in the form of heat
- All energy must come from the fuel

IMPLICATIONS ON TECHNOLOGY DEVELOPMENT

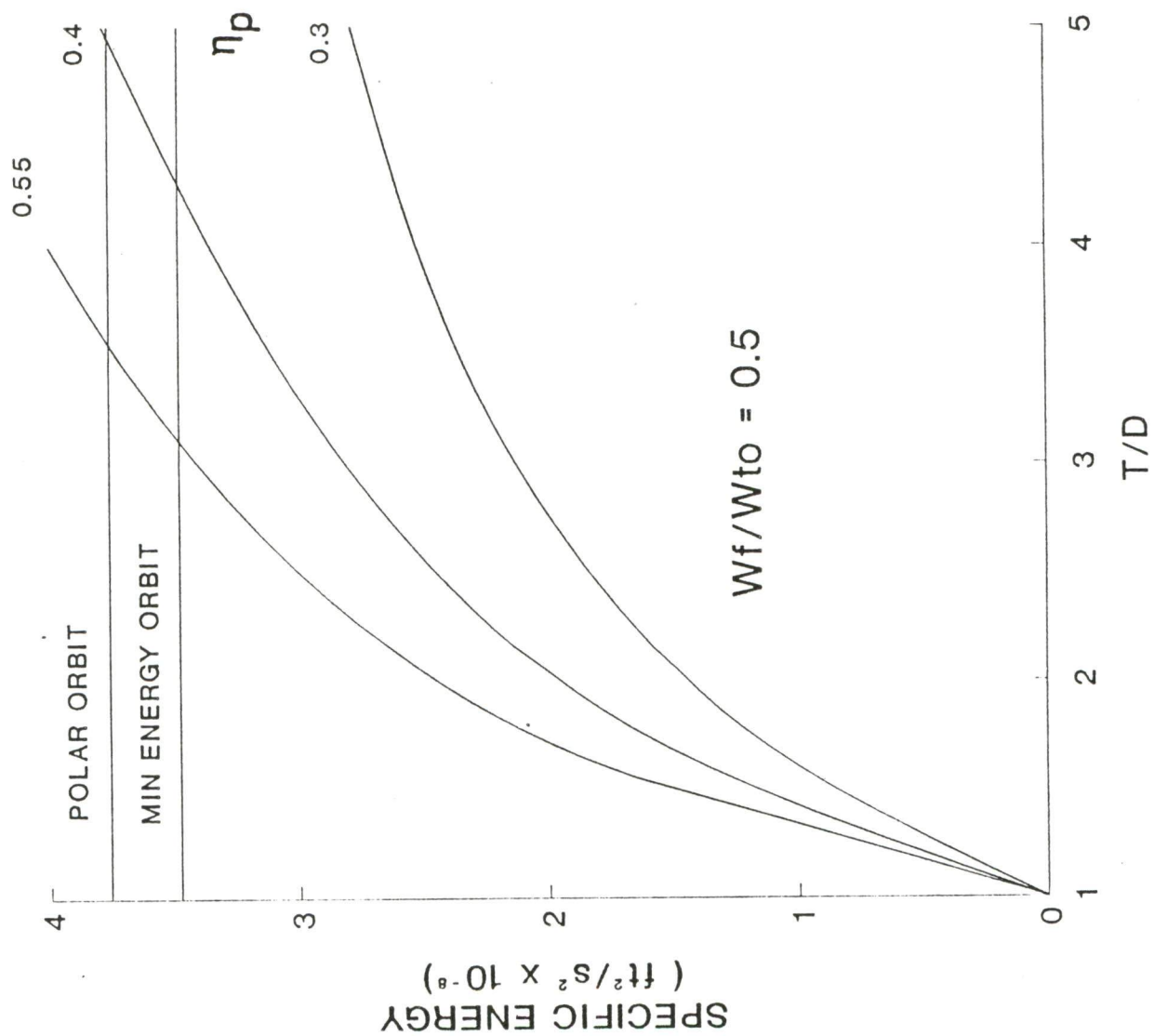
- GETTING TO ORBIT WITH A SINGLE-STAGE AIRBREATHING VEHICLE
REQUIRES APPROPRIATE VALUES FOR EACH OF THE THREE KEY
GOVERNING VALUES:

$$(T/D)$$

$$\eta_p$$

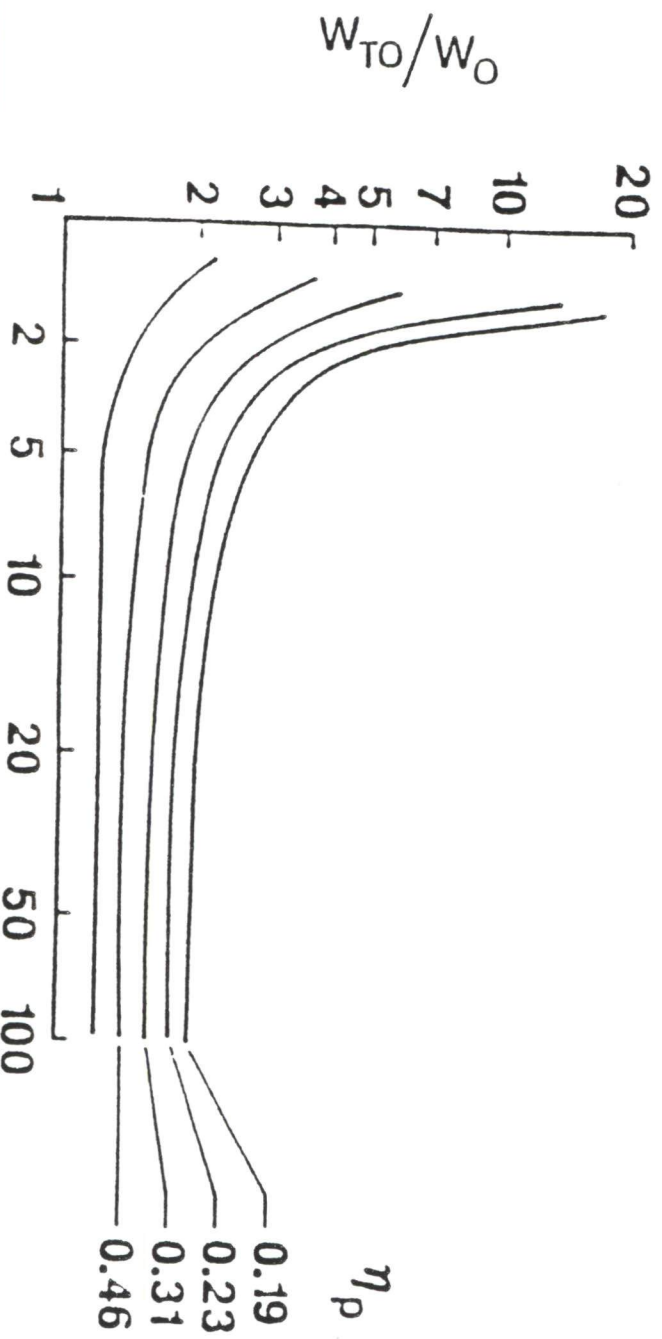
$$W_{TO}/W_O$$

AVAILABLE SPECIFIC ENERGIES

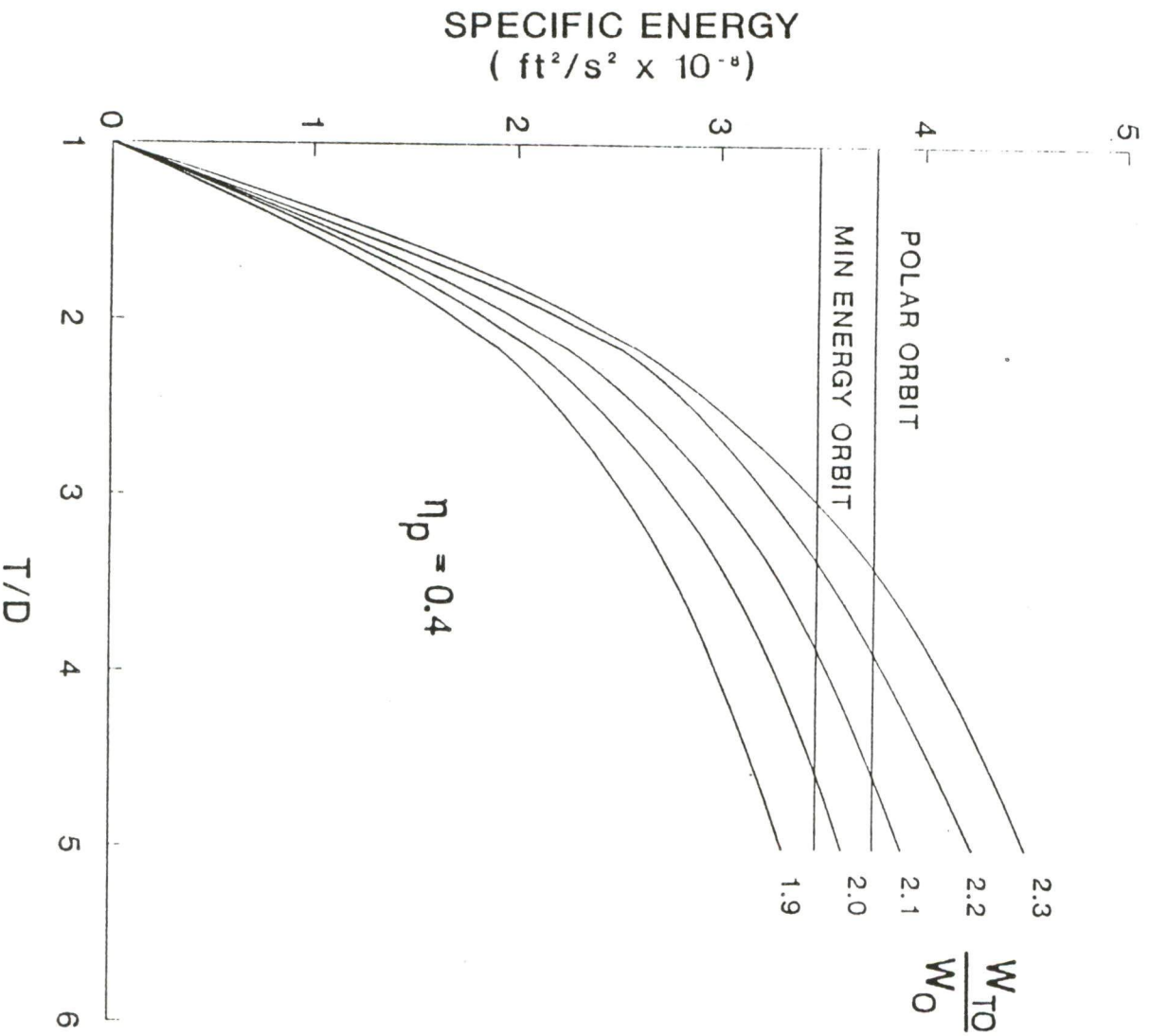


IMPLICATIONS OF T/D

- T/D OVER 3 IS REQUIRED
- PROBABLY THE MOST EASILY ACHIEVED OF THE 3 KEY GOVERNING PARAMETERS
- T/D > 5 IS OF LITTLE ADDITIONAL VALUE



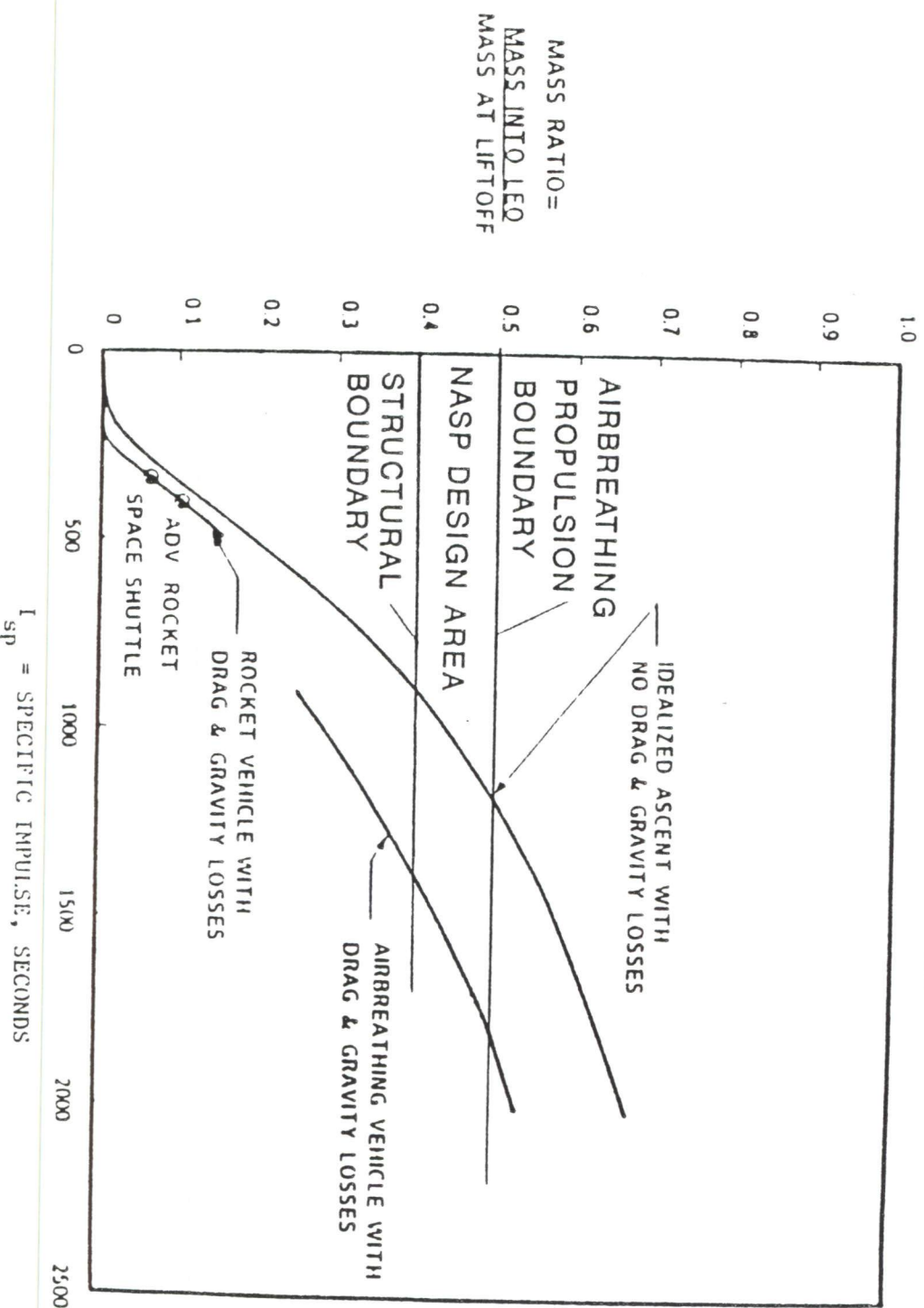
AVAILABLE SPECIFIC ENERGIES





INCREASED I_{sp} RESULTS IN: REDUCED FUEL AND INCREASED WEIGHT TO ORBIT

FINAL STAGE INTO LEO

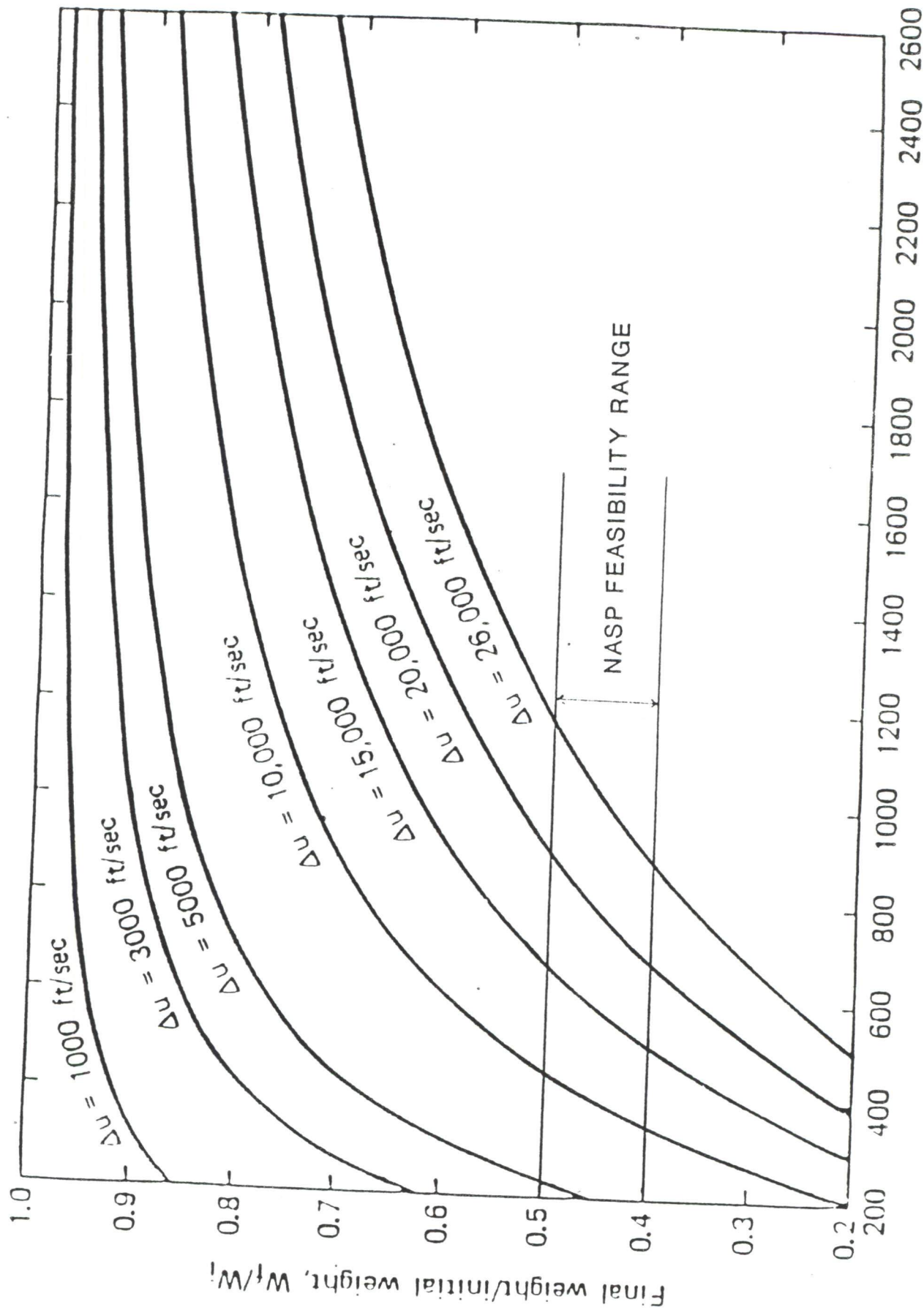


IMPLICATIONS OF η_p

- η_p ABOVE 0.35 IS REQUIRED
- 90 % OF ENERGY IS ACQUIRED ABOVE MACH 8
- 2/3 OF ENERGY IS ACQUIRED ABOVE MACH 15
- HIGH η_p AT HIGH SPEED IS ESSENTIAL
- η_p AS HIGH AS 0.69 IS PREDICTED AT HIGH SPEED

BUT

- WE HAVE NO EXPERIENCE WITH SCRAMJETS ABOVE MACH 7
- THE PHYSICS IS NOT WELL UNDERSTOOD ABOVE MACH 10
 - TURBULENCE MODELING
 - FUEL/AIR MIXING
 - SKIN FRICTION
 - FINITE RATE CHEMISTRY
 - MODELING OF MIXING, CHEMISTRY, ETC
- CAN BOUND EFFICIENCY, FLIGHT TEST WILL DEFINE IT



Net specific impulse = (thrust-drag)/fuel flow rate = $(T-D)/\dot{w}_0$

CHAPTER 4. PROPULSION

It is easy for the National Aerospace Plane to give engineers a combination of future shock and what in young students would be diagnosed as math anxiety. The idea of airbreathing ascent to orbit lies well beyond the realm of general experience, and carries with it a heavy load of difficulties. In particular, scramjet engines are not only unfamiliar; they also have the daunting reputation of demanding supercomputers and advanced flow codes for their design. What is needed, then, are some elementary analyses that could be taught in an introductory course. These can present the basic physics of the problem.

Ascent to orbit: introductory dynamics

Define the following quantities:

D	drag	γ	flight path angle
g	gravity	η	propulsive efficiency
h	altitude	Subscripts:	
m	mass of vehicle	-	exhaust
Q	heat of combustion of fuel	-o	orbit
t	time	-r	takeoff
T	thrust		
V	velocity		

Newton's second law, $F = m a$, takes the form

$$m \frac{dV}{dt} = T - D - m g \sin \gamma \quad (1)$$

when we consider only forces along the flight path. Here $m g \sin \gamma$ gives the gravity loss while D gives the drag loss. We now will integrate eq. (1) for three cases.

Case 1: Ascent of a rocket. This case is presented for comparison. In an elementary analysis we may neglect drag and gravity. The thrust is given by the product, (mass flow rate) $\times$ (exhaust velocity):

$$T = -\dot{m} V_e \quad (2)$$

where the minus sign appears because the mass of the rocket is decreasing. We take $m = \text{constant}$ and eq. (1) becomes

$$\frac{dv}{dt} = -v_e \frac{\dot{m}}{m} \quad (3)$$

To integrate with respect to t , we need m as a function of t :

$$m = m_T + \dot{m} t \quad (4)$$

and eq. (3) now becomes

$$dv = \frac{-\dot{m} dt}{m_T + \dot{m} t} \quad (5)$$

Both sides of eq. (5) are exact differentials and hence are immediately integrable. The integration has the following limits:

$$t = 0: m = m_T \text{ and } V = 0 \text{ at takeoff}$$

$$t = t_0: m = m_0 \text{ and } V = V_0 \text{ in orbit}$$

Also the right-hand side of eq. (5) has the form, du/u for which the integral is $\ln u$, the natural logarithm. Hence there is the solution,

$$V_0 = v_e \ln \left(\frac{m_T}{m_0} \right) \quad (6)$$

which is the well-known rocket equation. We usually see it written

$$V_0 = g_0 I_{sp} \ln \left(\frac{m_T}{m_0} \right) \quad (7)$$

where $g_0 = 32.174 \text{ ft/sec}^2$ is the acceleration due to gravity and I_{sp} is the specific impulse in seconds. The thrust T then is in pounds and the mass-flow rate $\dot{m}$ is in pounds per second. However, this is purely a convenience for the sake of the units we prefer to use.

We now can consider an airbreathing vehicle. Coleman duP. Donaldson recommends that we introduce the assumption

$$T/D = \text{constant} \quad (3)$$

which amounts to taking the thrust-to-drag ratio as an appropriately averaged value

over the entire flight path. This assumption is reasonable in that both T and D scale as proportionate to the dynamic pressure, with both the drag and thrust coefficient declining slightly with increasing Mach number during the critical high Mach portion of the flight. Increased fuel flow above stoichiometric at these high speeds offsets to an appreciable extent the decline in thrust due to less engine airflow.

Equation (1) then is written

$$m \frac{dv}{dt} = T \left(1 - \frac{D}{T} \right) - m g \sin \gamma \quad (9)$$

Multiplying both sides by V and dividing through by m , and rearranging,

$$V \frac{dv}{dt} + g V \sin \gamma = \frac{TV}{m} \left(1 - \frac{D}{T} \right) \quad (10)$$

Also, $V \sin \gamma = dh/dt$, rate of ascent. Hence we have the basic equation for airbreathing ascent:

$$\frac{d}{dt} \left(\frac{V^2}{2} + gh \right) = \frac{TV}{m} \left(1 - \frac{D}{T} \right) \quad (11)$$

since the left-hand side of eq. (11) is the differential given by the left of eq. (10). We recognize this differential as the specific energy of the ascending vehicle; that is, its energy per unit mass, kinetic plus potential.

Now consider the thrust term $T V$. We recognize this as the power transferred from the exhaust to the vehicle; that is, the rate at which the exhaust does work to accelerate the vehicle. In an airbreathing engine, unlike a rocket, we do not accelerate by taking mass from a fuel tank and ejecting it at high speed. Instead, our fuel is a source of heat. It adds energy to the airstream within the engine. The rate at which this energy is made available for acceleration is

$$Q \eta \, dm/dt$$

which is the product: (heating value of the fuel) $\times$ (fraction of this heating value which does useful propulsive work) $\times$ (fuel flow rate). Hence we have

$$T V = - Q \eta \, dm/dt \quad (12)$$

where again the minus sign appears because the aircraft is losing mass. Equation (12) states that the rate of energy transfer from fuel to vehicle, $Q \eta \, dm/dt$,

is equal to the rate at which the exhaust does work to accelerate the vehicle, $T V$. This is simply the work-energy theorem of elementary mechanics. It amounts to a definition of the propulsive efficiency, η :

$$\eta = T \frac{V}{\left(Q \frac{dm}{dt}\right)} \quad (13)$$

In practice, η will be the Carnot efficiency of the aircraft propulsion as a heat engine, reduced in consideration of other losses. In hypersonic flight, these include incomplete burn of the fuel as well as radiative losses. In radiative losses, the energy from the fuel goes into thermal radiation which is lost to the surroundings.

Equation (11) now takes the form

$$\frac{d}{dt} \left(\frac{V^2}{2} + g h \right) = - \frac{dm}{dt} \frac{Q\eta}{m} \left(1 - \frac{D}{T} \right) \quad (14)$$

which is integrated in two cases.

Case 1. $\dot{m} = \text{constant}$. This provides a direct comparison with the rocket case. Equation (4) applies, giving $m = m(t)$, and eq. (14) now becomes, when we multiply through by dt ,

$$d \left(\frac{V^2}{2} + g h \right) = \left(1 - \frac{D}{T} \right) Q \eta \frac{-\dot{m} dt}{m_T + \dot{m} t} \quad (15)$$

where the exact differential on the right-hand side is the same logarithmic form as in the rocket equation, eq. (5). The integration limits are

$$t = 0: V = h = 0; m = m_T \quad \text{at takeoff}$$

$$t = t_0: V = V_0, h = h_0, m = m_0 \quad \text{in orbit}$$

Hence the integration of eq. (15) gives the solution,

$$\frac{V_0^2}{2} + g h_0 = \left(1 - \frac{D}{T} \right) Q \eta \ln \left(\frac{m_T}{m_0} \right) \quad (16)$$

which is the counterpart of the rocket equation, eq. (6) or (7), for

airbreathing flight to orbit with constant m .

Note that the exhaust velocity, or the specific impulse, does not appear explicitly in eq. (16). However, we can introduce the I_{sp} by using its definition

$$T = g_0 I_{sp} \dot{m} \quad (17)$$

along with eq. (12). This gives the result,

$$I_{sp} = \frac{Q \eta}{g_0 V} \quad (18)$$

Hence I_{sp} is related to the engine design, which affects η . The same is true in a rocket. But in an airbreather, I_{sp} is not constant but falls off as $1/V$. That is why curves of I_{sp} for such engines, which show large peak values at moderate Mach numbers, fall off markedly as the Mach number rises.

Case 2. $T = \text{constant}$. To fly at constant thrust, under the assumption of eq. (8), is to fly at constant drag D and hence at constant dynamic pressure as well. This case then is realistic in that it is anticipated that the X-30 will indeed fly a trajectory of approximately constant dynamic pressure. Again we are to integrate eq. (14). We have dm/dt from eq. (12), which can itself be integrated:

$$\begin{aligned} \frac{dm}{dt} &= -T \frac{V}{Q} \eta \\ &= -\left(\frac{T}{Q} \eta\right) \frac{ds}{dt} \end{aligned} \quad (19)$$

where s = distance along trajectory and $V = ds/dt$. Integrating,

$$m = m_T - \left(\frac{T}{Q\eta}\right) s \quad (20)$$

and eq. (14) takes the form when multiplied through by dt :

$$d\left(\frac{V^2}{2} + g h\right) = \left(1 - \frac{D}{T}\right) Q \eta \frac{\left(\frac{T}{Q\eta}\right) ds}{m_T - \left(\frac{T}{Q\eta}\right) s} \quad (21)$$

which is the same as eq. (15). The integration limits are

$$t = 0: V = h = s = 0; m = m_T \text{ at takeoff}$$

$$t = t_0: V = V_0, h = h_0, m = m_0 \text{ in orbit}$$

where we could use eq. (20) to write the fraction on the right-hand side of eq. (21) explicitly as dm/m . Integration of eq. (21) then gives the solution,

$$\frac{v_0^2}{2} + g h_0 = (1 - D/T) Q \eta \ln (m_T/m_0) \quad (22)$$

which is precisely the same as eq. (16)! Hence the two cases constant $\dot{m}$ and constant T , give identical solutions.

This is understandable if we think of the airbreathing ascent as a process that transfers a fraction η of the specific energy in the fuel into kinetic Plus potential energy of the vehicle. If D/T is constant, then the choice of flight path is irrelevant; any trajectory that conserves D/T will yield the same result, eq. (16) or (22).

Now we put in values. Cole Donaldson recommends the following:

$$T/D = 3.5$$

$$\eta = 0.4$$

$$Q = 51,600 \text{ BTU/lbm for hydrogen}$$

$$\frac{v_0^2}{2} + g h_0 = 11.0 \times 10^6 \text{ ft-lbf/lbm}$$

and Q is to be multiplied by 778 ft-lbf/BTU to give consistent units. Then, from eq. (16) or (22),

$$\begin{aligned} m_T/m_0 &= \exp [11 \times 10^6 / ((1 - 1/3.5) \times 51,600 \times 778 \times 0.4)] \\ &= 2.609 \end{aligned}$$

The fraction of takeoff weight m_T that represents fuel is $1 - m_0/m_T$, or 61.7 percent. Taking $T/D = 5$ would change this to 57.5 percent. Thus, under this elementary analysis, about 60 percent of the takeoff weight must be fuel. An arithmetic average of E_{taP} taken from actual engine cycle calculations over the engine speed range is $E_{taP} = 0.477$. using this value the fuel fraction is 55.3 per cent. A more appropriate, fuel mass flow weighted average is $E_{taP} = 0.56$. Using this more accurate mass weighted value the fuel fraction is just under 50 per cent. This puts the aerospaceplane fuel fraction in the same level as a long range jet transport. For example the Douglas DC-8 had a fuel fraction of 50 per cent a passenger payload fraction of 10 per cent and an empty weight fraction of 40 per cent.

This can be compared with corresponding figures for several famous rockets and aircraft:

1944: German V-2 (structural material: steel): 68 percent

1952: U. S. Viking rocket (aluminum): 80 percent

1962: B-52 (aluminum), record flight of 12,532 miles: 68 percent

1986: Voyager (composites), round-the-world flight: 79 percent

Hence the fuel fractions for aircraft can be comparable to those for rockets. Moreover, the X-30 is to be built of materials, which are likely to compete well with aircraft aluminums for strength-to-weight ratio. But the cited rockets and aircraft all used fuels of high density: alcohol and liquid oxygen for the V-2 and Viking, jet fuel or aviation gasoline for the B-52 and Voyager. By contrast, the X-30 will use liquid hydrogen, which is far lower in density. Cole Donaldson points out that if the Voyager had been filled with liquid hydrogen, the fuel fraction would have been, not 79 percent, but 28 percent. Even a Boeing 747, which is quite voluminous, would have a fuel fraction of only 33 percent using liquid hydrogen if all available space were filled, including the lavatories and the overhead racks.

With this analysis, then, one may assess the merits of airbreathing aircraft versus rockets in ascent to orbit. It is easy to be beguiled by the following facts:

- The specific energy of a payload in orbit is 11×10^6 ft-lbf/lbm
- The specific energy of hydrogen-oxygen propellant at the mixture ratio of the Space Shuttle Main Engine, 6.0 pounds LO, per pound of LH₂, is 4.3×10^6 ft-lbf/lbm.
- The specific energy of hydrogen burned in air is 40.1×10^6 ft-lbf/lbm.
- The specific impulse of a scramjet can be as great as 3,000 seconds at Mach 6. That of the SSME is 455.2 seconds in vacuum, at any speed.

All this tends to suggest that scramjet-powered craft may resemble the legendary nuclear-powered rockets of advanced-design studies, whose payload bays are larger than their fuel tanks.

A more insightful assessment would begin with the following:

- A rocket is a "momentum machine," which accelerates by producing a momentum flux in the exhaust. An airbreather is an "energy machine" which accelerates by transferring energy into the exhaust.
- On its way to orbit, a single-stage rocket would typically burn half its fuel in getting to Mach 6 or less. An airbreather gains half its energy above Mach 17. This is to be used as a reference velocity.
- A scramjet's I_{sp} falls off as $1/V$. If it is 3,000 seconds at Mach 6, it is more like 1,000 seconds at Mach 17. This represents an average value along the entire trajectory which describes the scramjet's overall performance.
- An airbreather is more subject to drag losses than a rocket.
- An airbreather is less structurally efficient than a rocket.

A rocket can be built as a uniaxially-loaded cylinder, which leads to a high strength-to-weight ratio. An aircraft sustains large bending moments, particularly near wing roots, and must be designed as an array of cantilevers. These are intrinsically heavier than a columnlike cylinder, for equal loads.

- The bottom line is that if we take 50 to 60 per cent as a typical mass fraction for an airbreather, the corresponding average specific impulse for ascent to orbit, found from the rocket equation (eq. (7)), is 900 to 1,200 seconds. An airbreathing ascent may more than double the specific impulse available from our most advanced rocket motor, the SSME. Furthermore this value, 900 to 1,200 seconds, takes account of the drag losses as well as propulsive inefficiencies.

Hence the development of suitable airbreathers appears well worth pursuing.

2. Structure, fuel and thermal management

The two missions, hypersonic cruise and ascent to orbit, impose different heat loads. Cruise gives a prolonged thermal soaking; a two-hour flight in the atmosphere will expose a hypersonic aircraft to a thousand times more heat than the Shuttle receives on its quick trip down from space. Ascent to orbit, by contrast, imposes a more sharply-peaked but shorter surge of heating.

However, cruise offers a designer a particularly wide range of options for the thermal management. There are several reasons for this. Unlike the ascent-to-orbit mission, cruise does not demand that the skin receive active cooling. Nor does cruise require that heat be recovered and dumped into the fuel to raise its energy. Further, cruise imposes no demand that at high speed, much more fuel is to be injected into the engines than can be burned within the available airflow. To sum things up in a single phrase, what hypersonic cruise offers is design margins. In contrast with ascent to orbit, which demands that all technologies must be pushed to the limit, hypersonic cruise can be accommodated through extensions of current aircraft design practice.

The key to these margins lies in the fuel weight fraction. For an aircraft, this is found from the Breguet range equation:

$$R = (L/D) V I_{sp} \ln (m_T/m_l) \quad (3)$$

where

R = range, statute miles

L/D = lift-to-drag ratio

V = velocity, miles per second

I_{sp} = specific impulse, seconds

m_T = takeoff weight

m_l = landing weight

m_f = fuel weight = $m_T - m_l$

The term $V I_{sp}$ can be given in terms of the fuel heating value:

$$V I_{sp} = 778 Q \eta \quad (4)$$

where V is now in feet per second and

Q = heating value of fuel, BTU/lbm

η = propulsive efficiency (fraction of Q that produces thrust)

Take $\eta = 0.4$, as in the analysis in Chapter 4. This a good representative value at Mach 6. The fuel weight fraction $m_f/m_T = 1 - m_e/m_T$ since the fuel weight is simply the difference,

$m_T - m_e$. We then can consider two fuels, hydrogen and methane. We assume a waverider configuration flying at Mach 6; then from Eq. (2), $L/D = 8.0$. The range R is to be 7500 miles. We then have:

Hydrogen: $Q = 51,600$ BTU/lbm

$$m_f/m_T = 26.5 \text{ percent}$$

Methane: $Q = 21,500$ BTU/lbm

$$m_f/m_T = 52.2 \text{ percent}$$

However, we are not only interested in the weight of the fuel, but in its bulk. The densities of the two fuels, as liquefied gases, are:

Hydrogen: 4.6 lbm/ft^3

Methane: 28.8 lbm/ft^3

Thus, with over six times the density, methane offers the advantage of greater compactness while still preserving aircraft-type fuel weight fractions.

Now consider ascent to orbit. The pertinent equation was derived in Chapter 4:

$$\text{Energy in orbit} = Q \eta (1 - D/T) \ln (m_T/m_O) \quad (5)$$

where the energy in orbit = 11.0×10^6 ft-lbf/lbm and

T/D = mean thrust-to-drag ratio along the trajectory

m_O = mass in orbit

with the fuel mass fraction being $m_f/m_T = 1 - m_O/m_T$ as before. Equations (5) and (4) are very similar. Indeed, we can make them identi-

cal by considering that the orbital energy is equivalent to a range of 11×10^6 feet or 2083 miles, which is to be flown by a "brick" with $L/D = 1$. Physically, that range at that L/D is equivalent to rising through that distance, 2083.3 miles, within a constant gravity field. Taking $T/D = 3.5$, the recommended value, we have:

Hydrogen: $m_f/m_T = 61.7$ percent

Methane: $m_f/m_T = 90.0$ percent

Thus, despite its greater density, methane imposes upon the designer the need to accommodate thermal management, active cooling, and much else within an airframe with half the structural mass fraction of the Voyager; that is, 10.0 percent versus the 21 percent of that round-the-world aircraft. No one has built an airplane of any type with ninety percent of its weight as fuel. Even sixty percent pushes the technology very hard, in using hydrogen. Unlike the case of hypersonic cruise, in ascent to orbit the weight margins just aren't there.

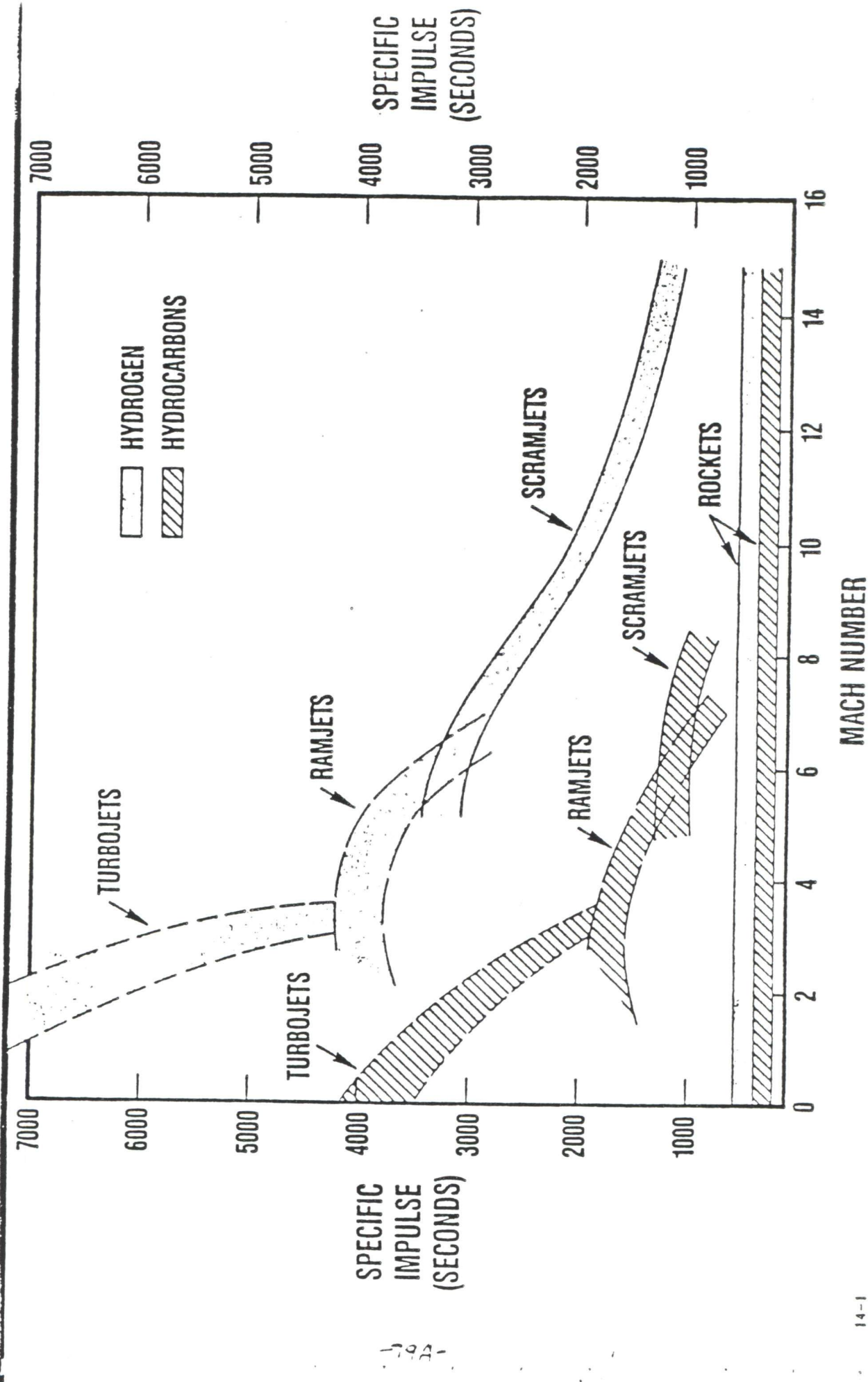
This analysis also carries within it, though in hidden form, the requirement that the thermal management system transfer recovered heat to the fuel. Suppose we are to inject hydrogen into an airflow passing through the scramjets at velocity V . Just to get up to the speed of this flow, the hydrogen demands an energy of

$$\frac{1}{2} V^2 / (778 \times 32.174) \text{ BTU/lbm}$$

In Chapter 4 we noted that an appropriate reference velocity for ascent to orbit is Mach 17, or 18,500 feet per second. The energy for the hydrogen then is 6836 BTU/lbm. This is a sizeable portion of the 51,600 BTU/lbm the fuel carries in its molecules. If we allow it to be lost, then Q in eq. (5) must be taken as $51,600 - 6836$ or 44,764 BTU/lbm. The associated fuel weight fraction at takeoff then is $m_f/m_T = 66.9$ percent; the fuel fraction, which is already quite demanding, must be increased further from the original 61.7 percent.

PROPULSION OPTIONS

After Escher



If scramjet efficiency is low the increase in rocket propellant to complete the flight to orbit is modest--even if the scramjet poops out at speeds as low as Mach 18. This result is astonishing and paradoxical. It amounts to asserting that the X-30 may gain half its orbital energy through rocket burn, at minimal cost. Yet it follows from the fact that at very high Mach, the specific impulse for an airbreather is only about 1.5 to 2.0 times

that of a rocket. Hence one may switch from one to the other with an acceptable penalty. The actual penalty will now be calculated.

5. Use of a rocket for final ascent

From eq. (4) on p. 146, and with the cited parameters,

$$I_{sp} = 617.6 (V_0/V) \text{ seconds} \quad (7)$$

where V_0 = orbital velocity = 26,000 ft/sec. That is why I_{sp} , which exceeds 3200 seconds at $V = 5000$ ft/sec (Mach 5), falls to values only moderately higher than that of the Space Shuttle Main Engine, 455.2 seconds. Indeed, if the scramjet could give no better efficiency than 30 percent, with $\eta = 0.3$ in eq. (4), then the SSME would actually give better I_{sp} above $V = 26,450$ ft/sec, if such velocities were needed.

We now introduce the concept of a changeover Mach number, M_c . From $M = 0$ to M_c , the vehicle is to fly by airbreathing; from M_c to M_0 , orbital Mach number, the craft is to rocket-burn. Correspondingly, there are three vehicle masses: m_T at takeoff, m_c at changeover, m_0 in orbit. From eq. (16) on p. 80, writing m_T/m_c in place of m_T/m_0 , there is the expression,

$$\frac{m_T}{m_c} = \exp \left[\frac{(M_c/M_0)^2}{(1 - D/T) Q\eta} (\frac{1}{2} V_0^2 + gh_0) \right] \quad (8)$$

where the orbital energy, $\frac{1}{2} V_0^2 + gh_0$, is scaled for different values of M_c by being multiplied by the factor $(M_c/M_0)^2$. Now put in values:

$$\frac{1}{2} V_0^2 + gh_0 = 11.0 \times 10^6 \text{ ft-lbf/lbm}$$

$$T/D = 3.5$$

$$Q\eta = 16.06 \times 10^6 \text{ ft-lbf/lbm}$$

and eq. (8) becomes

$$m_T/m_c = \exp [0.96 (M_c/M_0)^2] \quad (9)$$

On the rocket-propelled leg, from M_c to M_0 , there is the rocket equation, eq. (7) on p. 78. It gives the mass-ratio m_c/m_0 :

$$\frac{m_c}{m_0} = \exp [(1 - M_c/M_0) V_0/g_0 I_{sp}] \quad (10)$$

We take $I_{sp} = 440$ seconds, the value appropriate to Pratt & Whitney's RL-10 of 15,000 pounds thrust, which has been discussed as a choice for the X-30. Then with $V_0 = 26,000$ ft/sec and $g_0 = 32.174$ ft/sec²,

$$m_c/m_0 = \exp [1.88 (1 - M_c/M_0)] \quad (11)$$

Finally, the propellant mass is m_f , equal to $m_T - m_0$. The propellant mass fraction is m_f/m_T :

$$\frac{m_f}{m_T} = 1 - \frac{m_0}{m_T} = 1 - \frac{m_0}{m_c} \frac{m_c}{m_T} \quad (12)$$

and m_0/m_c , m_c/m_T are found respectively from eqs. (11) and (9) above.

The results are plotted on the adjoining graph.

We take $M_0 = 26.6$. There are at least four cases of interest:

1. $M_c = 0$. This is the case of a rocket-propelled single-stage-to-orbit (SSTO) with no airbreathing. It gives $m_f/m_T = 0.847$; that is, nearly 85 percent of the liftoff weight is propellant.

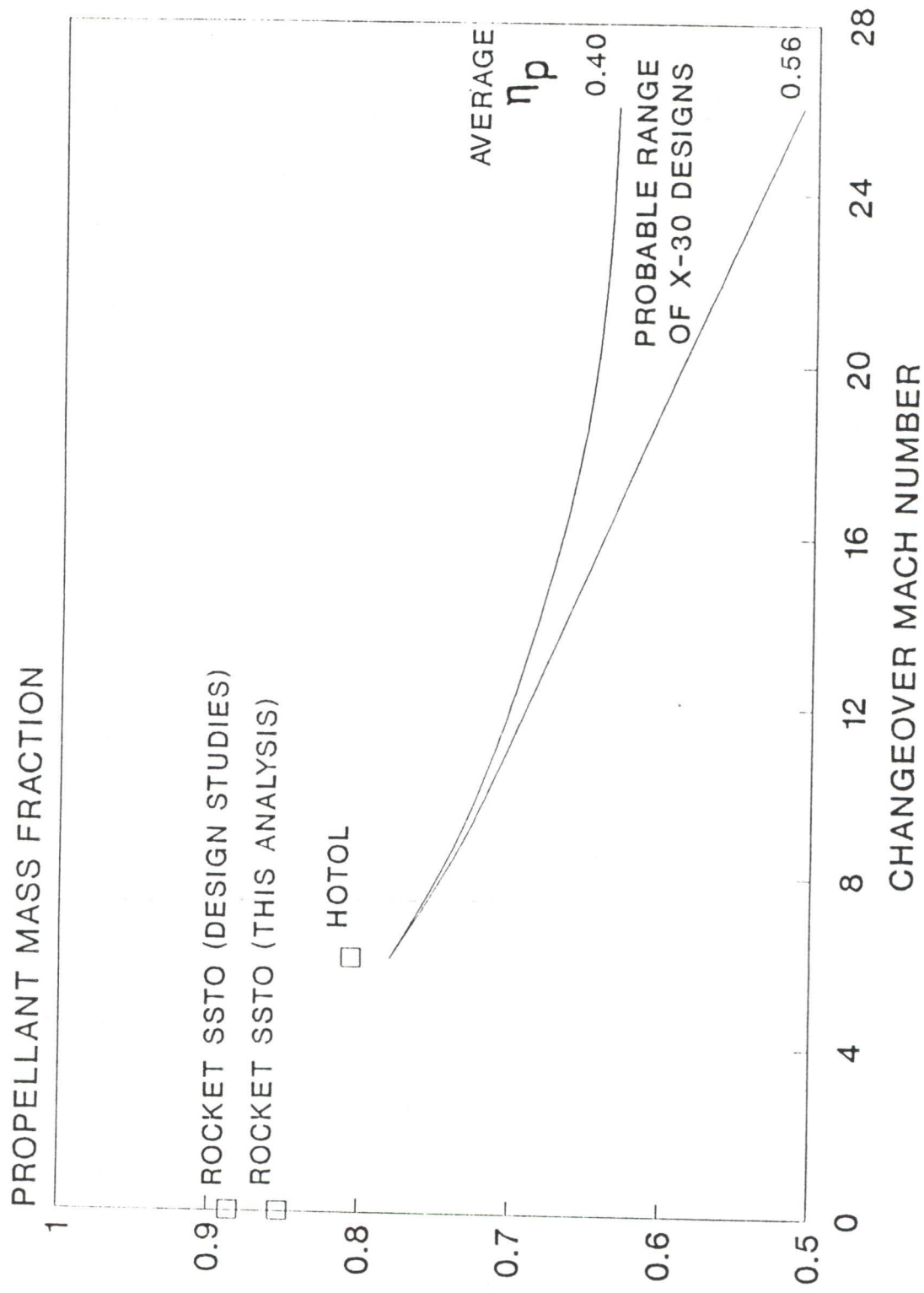
2. $M_c = 6$. Changeover at Mach 6 holds for airbreather-assisted single-stage craft studied to date, such as the British HOTOL. Here $m_f/m_T = 77.8$ percent. There is an improvement of seven percent, which is modest but welcome.

3. $M_c = 18$. This might hold for the X-30, if its scramjets poop out well short of orbit. We have $m_f/m_T = 0.649$, nearly 13 percent better than HOTOL and 20 percent better than a rocket-powered SSTO.

4. $M_c = 26$. This is a craft that airbreathes all the way to orbit. If it can do that, then m_f/m_T ranges from 50 to 61.7 percent depending on the design of the SCRAMJET.

The significant conclusion is that a falloff in M_c from Mach 26

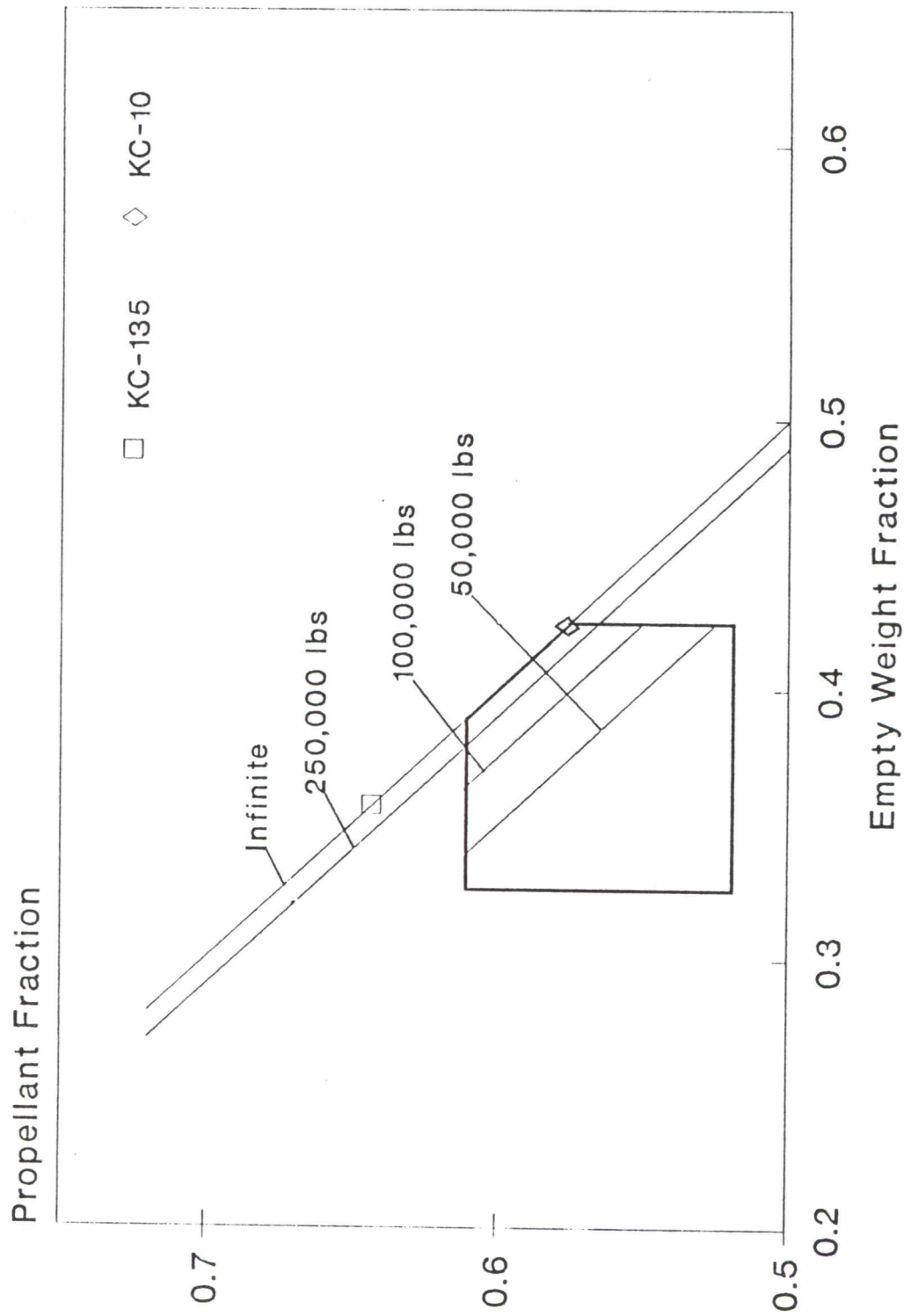
AIRBREATHING TO ROCKET CHANGEOVER



The impact of fuel fraction on gross weight can be obtained from the sizing charts. Two cases are shown. The first one is the NASP, X-30 problem where the payload is limited to flight test instrumentation only, and the second one is an operational airplane designed to carry 25 tons into a polar orbit. Fuel fraction is the vertical coordinate on these charts and empty weight to gross weight ratio is the horizontal coordinate. Lines of constant gross weight are plotted on the chart. Each point on a gross weight line represents a combination of empty weight fraction and fuel fraction that will meet the mission requirements at that gross weight. The box in the center of the chart is the region where the final designs will most probably lie. Two operational airplanes with a high design fuel fraction are shown for reference. These aircraft also have relatively high pressurized volume since they are designed to carry passengers. The reduction in gross weight with reduced fuel fraction is evident from the chart. For example the X-30 could weigh 250,000 pounds with a 62% fuel fraction and a roughly 37% empty weight fraction but only 50,000 pounds with a 53% fuel fraction and around 42% empty weight fraction. The empty weight fraction will increase as the gross weight becomes smaller. Pilots, avionics and some other items are fixed weights that are an increasing fraction of gross weight as gross weight is reduced.

NASP SIZING CHART

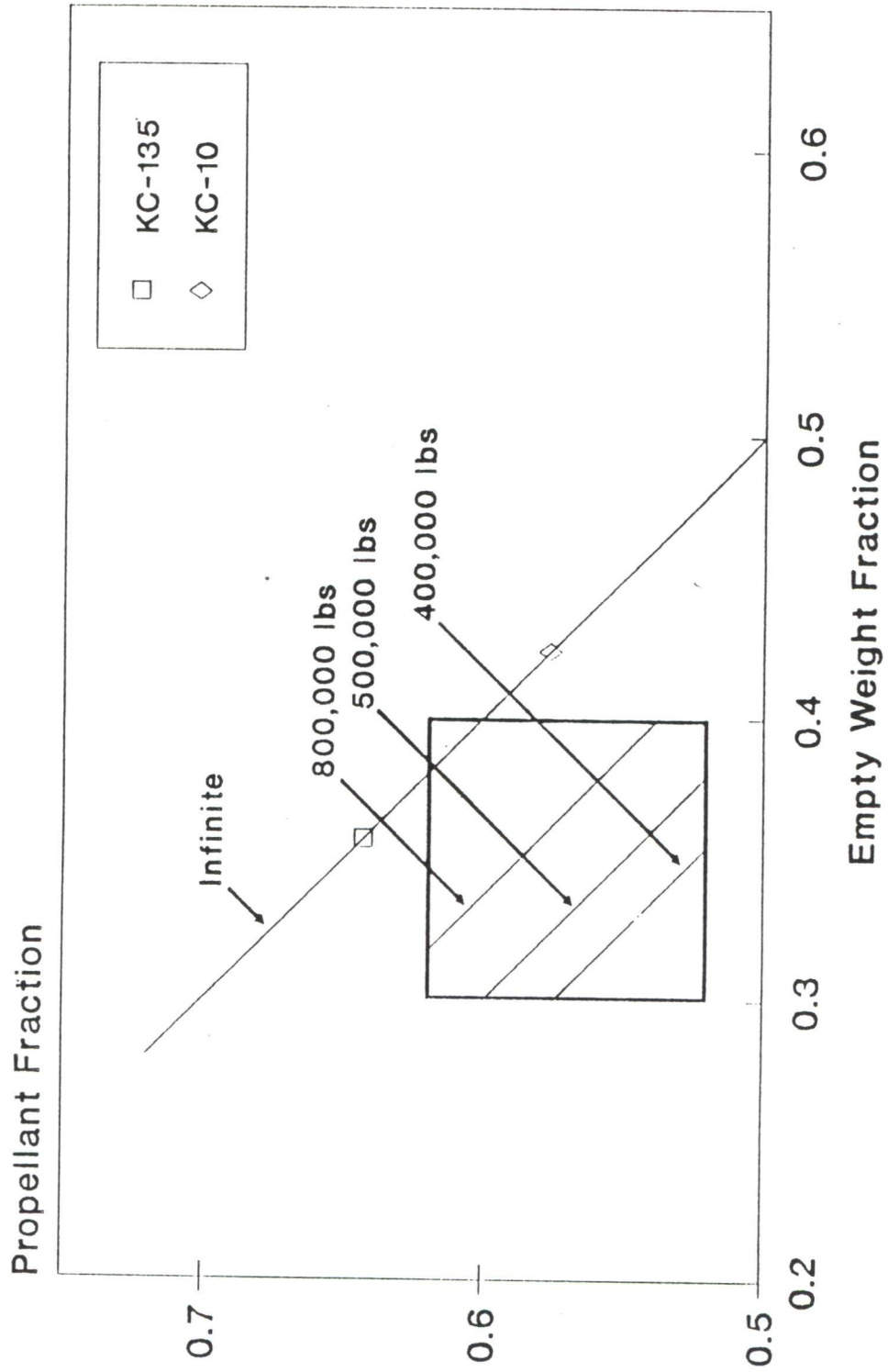
FLIGHT TEST INSTRUMENTATION ONLY



SIZING CHART

50,000 Pound Payload

Polar Orbit



SCRAMJET CYCLE ANALYSIS

(SCA)

1.0 INTRODUCTION

The SCRAMJET, or Supersonic Combustion RAMJET, is the engine cycle which permits efficient flight at hypersonic speeds. Hypersonic speed is defined as a speed at which the shock wave closely approaches the shape of the body generating the shock. For the slender, low drag shapes typical of hypersonic aircraft the lower end of the hypersonic regime can be taken as approximately Mach 6.0. At this speed the efficiency of the conventional subsonic combustion ramjet and the SCRAMJET are nearly equal. Above this speed SCRAMJETS are more efficient.

An appropriate static pressure rise for the cycle can be obtained at flight speeds above Mach 6.0 without slowing the air all the way down to subsonic speeds in the engine combustor. At the higher speeds compression to subsonic velocity will create large increases in entropy and dissociation of the air and combustion products which can only be partially recovered in the engine nozzle expansion. It is to avoid these losses and limit static pressures in the combustor that the SCRAMJET cycle is preferred at hypersonic flight speeds.

The SCRAMJET cycle is a Brayton cycle similar to a subsonic combustion ramjet or turbojet. The cycle is shown schematically on an enthalpy, entropy diagram in Figure 1. The free stream air velocity, point 1, is reduced in the inlet with an associated increase in static pressure to conditions at point 2, the inlet

ENTHALPY, ENTROPY DIAGRAM

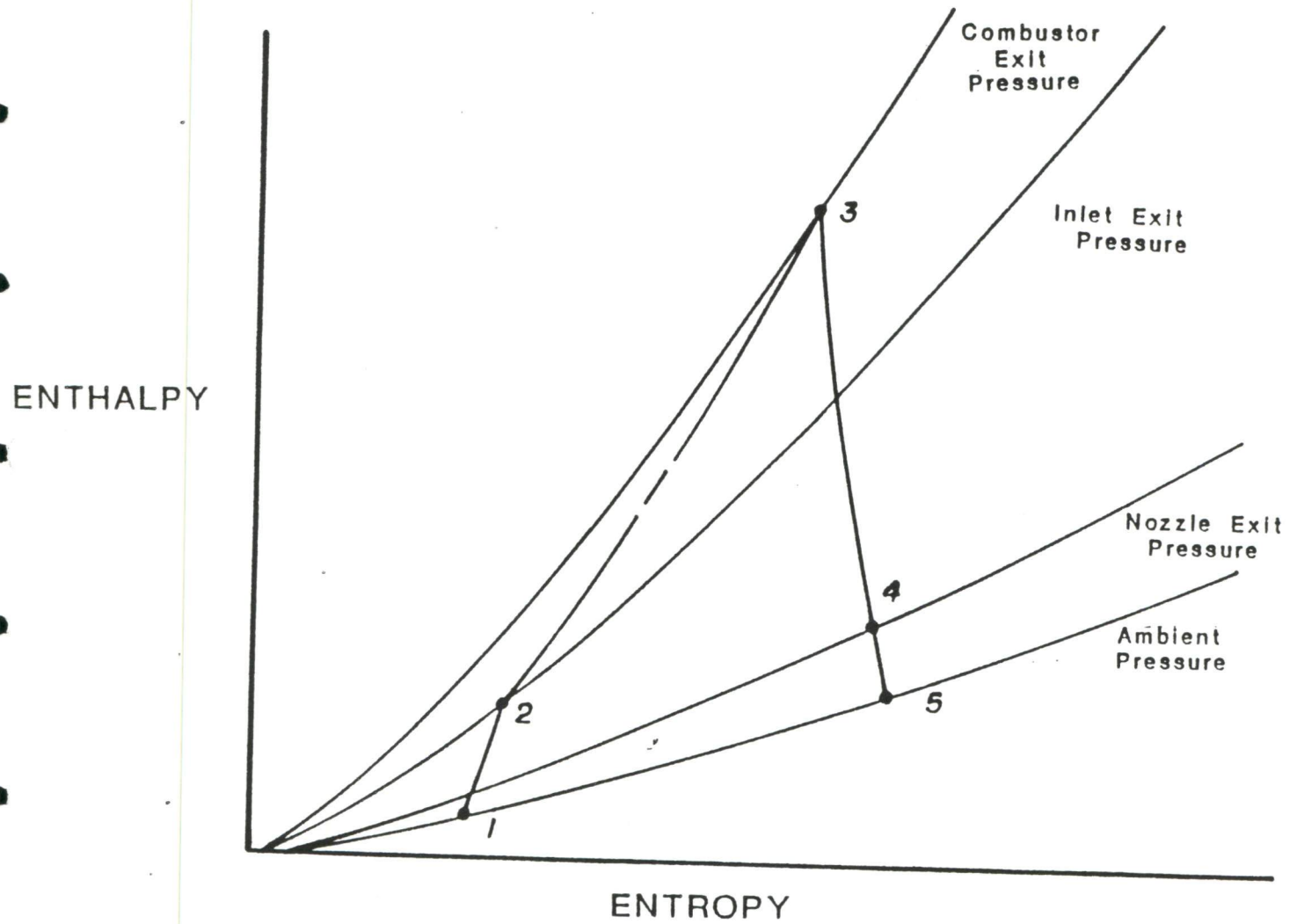


Figure 1

exit. In the combustor fuel is burned increasing the enthalpy, or energy, of the flow. In most SCRAMJET designs the static pressure in the combustor will increase due to heat addition as shown in point 3 in the diagram, the combustor exit. The compressed and heated flow is expanded in the engine nozzle to point 4, the nozzle exit. Expansion all the way to ambient pressure, point 5, is not possible with the nozzle exit area available on most SCRAMJET designs. The difference in momentum between the air at point 1 and the combustion products at point 4 is the net thrust of the engine. Although the enthalpy, entropy diagram is useful in understanding cycles, a single diagram like Figure 1 is only conceptual. Multiple diagrams are necessary for a rigorous presentation because the relationship between enthalpy and entropy is different for air and combustion products.

A schematic of a SCRAMJET engine corresponding to the diagram is shown in Figure 2. Note the large reduction from the free stream area at the inlet exit. The ratio of free stream area, at 1, to inlet exit area, at 2, is called the inlet contraction ratio, or contraction. Similarly the ratio of the nozzle exit area, at 4, to the combustor exit area, at 3, is called the nozzle area ratio, and the combustor exit area, at 3, to the combustor inlet area, at 2, is called the combustor area ratio. Performance of SCRAMJETs is almost entirely determined by these three ratios and by the entropy loss in the inlet compression process. The lower the entropy increase, more vertical on Figure 1, during the inlet compression process the

SCRAMJET ENGINE SCHEMATIC

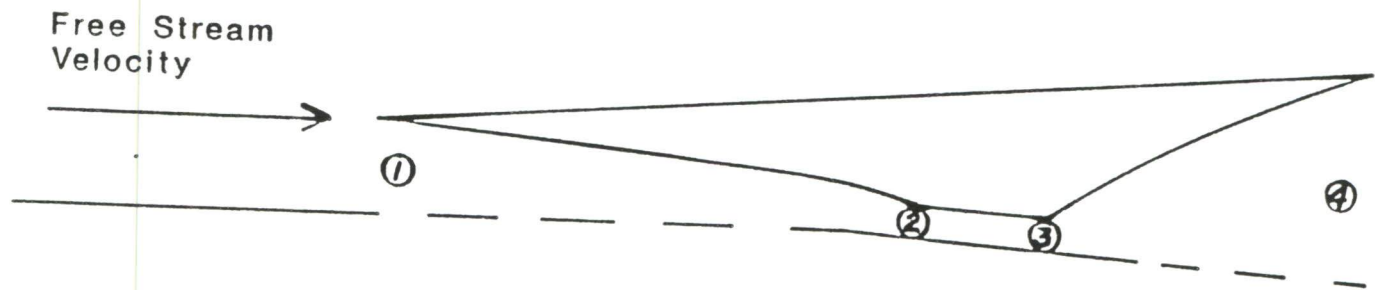


Figure 2

more efficient the engine. As speed increases, the efficiency of the inlet becomes progressively more important. It has a negligible effect at Mach 8.0 and can cause large changes in performance by Mach 16. This trend is shown graphically in Figures 3 through 5. In each of these figures engine thrust per pound of fuel flow per second, specific impulse, is plotted against the inlet kinetic energy efficiency divided by the ratio of total enthalpy at the throat to free stream total enthalpy. This parameter is a measure of the entropy increase in the inlet compression process. A value of unity is isentropic, and the lower the value the larger the entropy increase. The definition of kinetic energy efficiency and its relationship to pressure recovery at constant gamma is derived in Figure 6. The nozzle expansion process, on the other hand, is nearly isentropic in a well designed nozzle; that is nearly vertical on Figure 1. Nozzle losses are due primarily to three causes: (1) Nonalignment of the flow called divergence losses, (2) Skin friction drag and (3) inability of dissociated combustion products to reach their equilibrium composition level before they exit the expansion nozzle called kinetic losses.

Inability to completely mix the fuel and air in the combustor is usually called chemical efficiency or simply combustion efficiency and is defined as the fraction of fuel that is mixed and burned compared to that which is available to be burned. In most hydrogen fueled SCRAMJET cases it is safe to assume that if the fuel and air are mixed, the fuel is burned. The ignition delay

EFFECTS OF INLET SHOCK LOSSES

MACH 8, $Q=2000$ PSF, $\text{PHI}=1.0$
CONSTANT AREA COMBUSTION

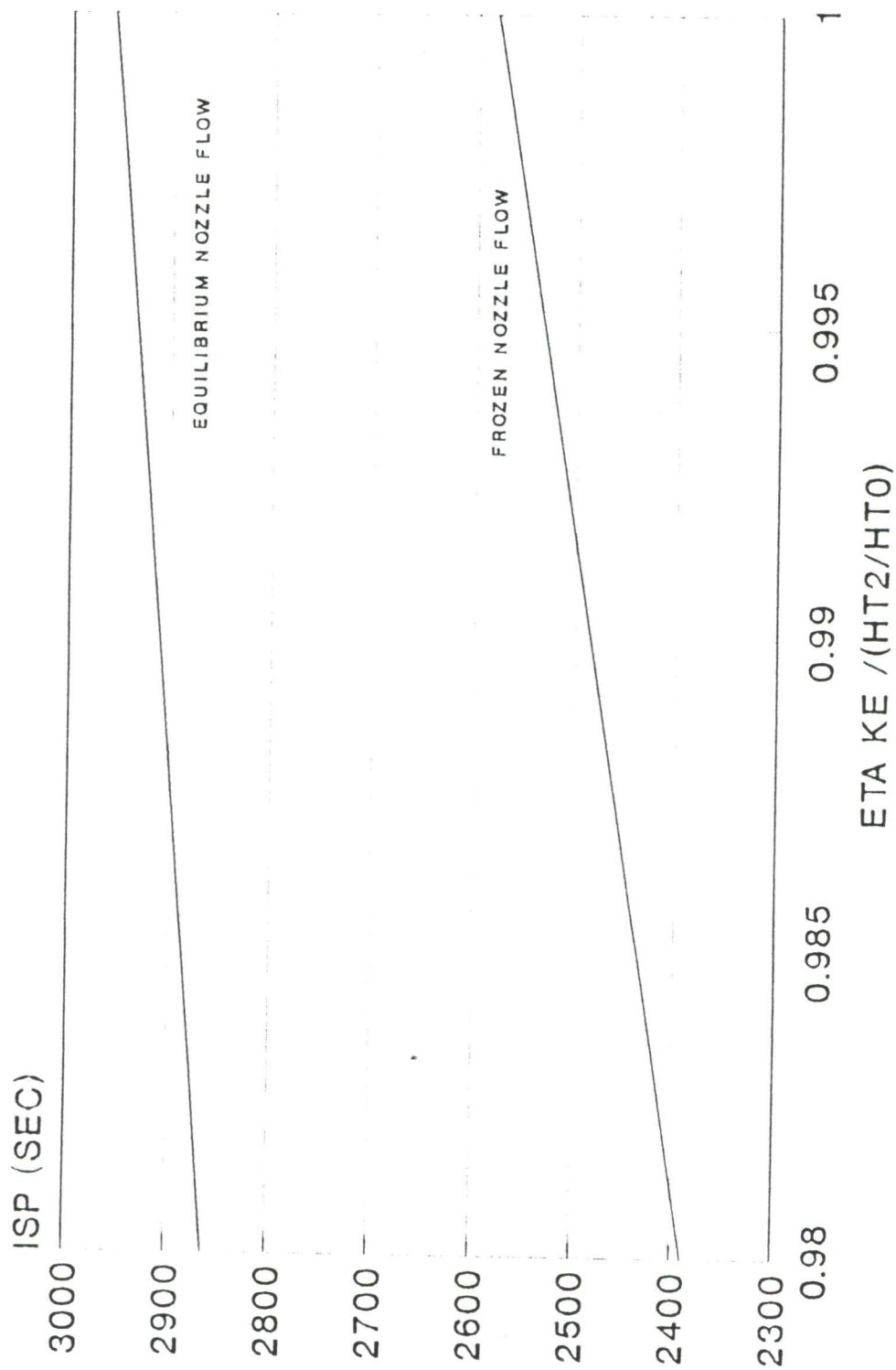


Figure 3

EFFECTS OF INLET SHOCK LOSSES

MACH 16, $Q=2000$ PSF, $\text{PHI}=1.0$
CONSTANT AREA COMBUSTION

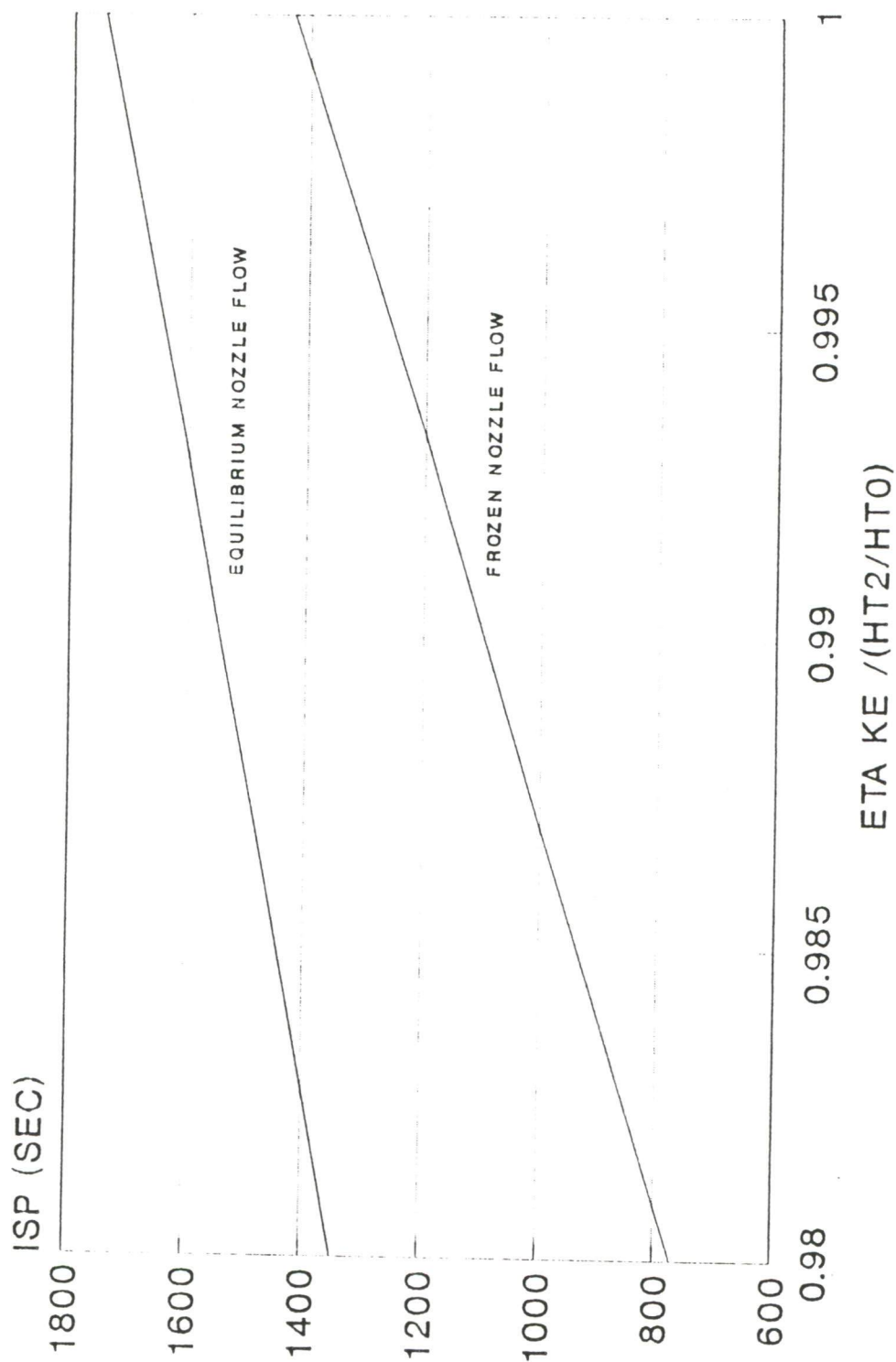


Figure 4

EFFECTS OF INLET SHOCK LOSSES

MACH 24, $Q=1000$ PSF, $\text{PHI}=2.0$
CONSTANT AREA COMBUSTION

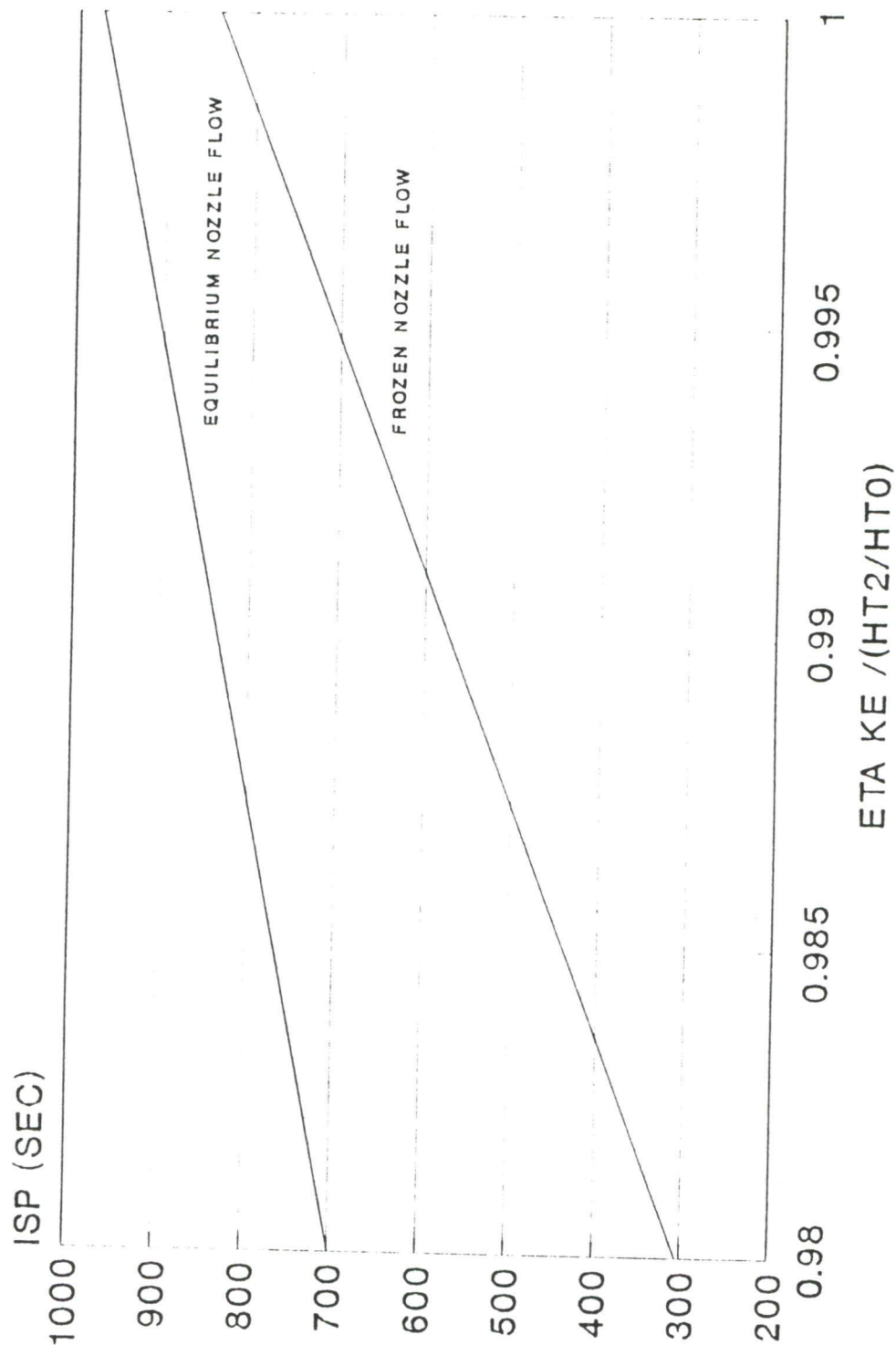


Figure 5

KINETIC ENERGY EFFICIENCY

UNCLASSIFIED

V_E = VELOCITY AFTER ISENTROPIC EXPANSION FROM INLET THROAT

V_∞ = FLIGHT VELOCITY

$$\eta_{KE} \equiv \frac{V_E^2}{V_\infty^2} = \frac{h_{TE}}{h_{T\infty}} \left(\frac{1 - h_E/h_{TE}}{1 - h_\infty/h_{T\infty}} \right)$$

CONSTANT γ

$$\eta_{KE} = \frac{h_{T_2}}{h_{T\infty}} \left(1 - \frac{\left(\frac{P_{T\infty}}{P_{T_2}} \right)^{\frac{\gamma-1}{\gamma}}}{\frac{\gamma-1}{2} M_\infty^2} \right)$$

Figure 6

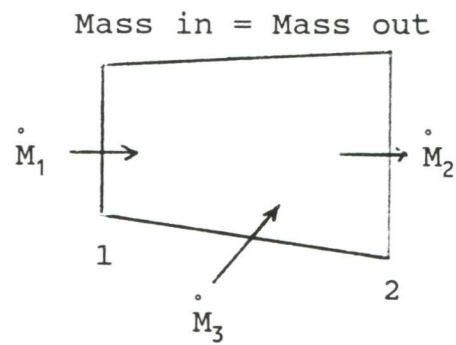
time, or time to complete the reaction, is so short in most SCRAMJET combustors that the length required to complete the reaction is negligible compared to the length required to complete mixing down to the molecular level required for combustion.

Cycle analysis of SCRAMJET engines consists of solving the equations of mass, momentum and energy conservation and the equation of state between adjacent stations in the engine applying the known boundary conditions. For example, the conditions at the inlet exit, point 2 in Figure 1, can be determined from the free stream conditions if the contraction ratio, compression efficiency and total heat transferred to the inlet walls is known. The combustor exit conditions can similarly be solved for if the inlet exit conditions are known, and the nozzle exit conditions obtained with knowledge of these combustor exit conditions which are the nozzle entrance properties.

2.0 FOUR BASIC EQUATIONS

The four basic equations are the conservation of mass, momentum and energy and the equation of state which relates static pressure, density and static temperature. With these four equations and known or calculable conditions on the engine walls between two stations, downstream conditions can be calculated from the upstream conditions. The derivation of these equations in a form suitable for engine cycle analysis follows for the general case of mass and heat addition between stations.

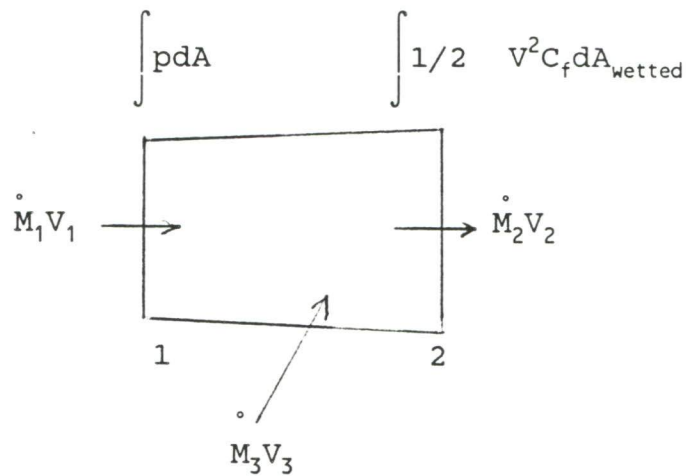
Conservation of Mass or Continuity Equation



$$\dot{M}_1 + \dot{M}_3 = \dot{M}_2$$

$$\rho V_1 A_1 + \rho_3 V_3 A_3 = \rho_2 V_2 A_2 \quad (1)$$

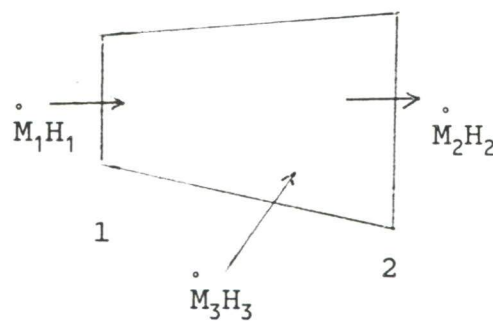
Conservation of Momentum or Momentum Equation



$$\dot{M}_1 V_1 + \dot{M}_3 V_3 = \dot{M}_2 V_2$$

$$\begin{aligned}
 & p_1 A + \rho_1 \dot{V}_1^2 A_1 + p_3 A_3 + \rho_3 \dot{V}_3^2 A_3 \\
 & + \int_1^2 p dA - \int_1^2 \frac{1}{2} \rho V^2 C_f dA_w = p_2 A_2 + \rho_2 \dot{V}_2^2 A_2 \quad (2)
 \end{aligned}$$

Conservation of Energy or Energy Equation



H is defined as total enthalpy or $h + \frac{V^2}{2}$ where h is the static enthalpy or static energy level above some reference state. Total

enthalpy will, of course, depend on the location of the observer, or reference point, since velocities are measured relative to this reference point.

$$\dot{M}_1 H_1 + \dot{M}_3 H_3 + \dot{q} = \dot{M}_2 H_2$$

$$\rho_1 V_1 A_1 \left(h_1 + \frac{V_1^2}{2} \right) + \rho_3 V_3 A_3 \left(h_3 + \frac{V_3^2}{2} \right) + \dot{q} = \rho_2 V_2 A_2 \left(h_2 + \frac{V_2^2}{2} \right) \quad (3)$$

The term $\dot{q}$ is introduced to account for heat added or removed through the walls. Chemical combination which releases or absorbs sensible heat is not included in the energy equation. It will be accounted for in the equation of state.

Equation of State or State Equation

A general equation of state is:

$$p = f(\rho, T) \quad (4)$$

For gases of constant composition this relation is simple:

$$p = \rho RT \quad (5)$$

R is a constant which depends on the molecular weight of the gas:

$$R = \frac{\text{R universal}}{\text{Molecular Weight}}$$

For SCRAMJET cycle analysis a thermochemistry computer program is required which will iteratively solve for the chemical equilibrium composition of the gas at the appropriate conditions and relate pressure, temperature and density at these conditions. An example is the program used at duPont Aerospace, "The Thermochemistry of Arbitrary Chemical Reactants," developed by Bonnie McBride and Sanford Gordon at NASA Lewis Research Center.¹

Although a Thermochemistry analysis is necessary to do accurate SCRAMJET cycle analysis, a solid conceptual understanding of SCRAMJET cycles and reasonable accuracy can be achieved using the simple equation $p = \rho RT$ and selecting a value for R that is appropriate for the conditions at each station in the engine. This approach amounts to perturbing a known solution and is useful in doing design analysis of the effect of geometric variations and checking computer program results. It is the approach that will be used throughout this course. Students will remember that a computer program will use the exact equation of state, otherwise the methods are exactly the same.

The interesting, and hard, part of SCRAMJET cycle analysis is in obtaining the boundary conditions that enable a solution to the four basic equations. The balance of the lecture will be devoted to approaches to obtaining the boundary conditions and some examples of solving the four simultaneous basic equations.

3.0 THE INLET

The SCRAMJET inlet compresses the flow from free stream to combustor entrance conditions by a series of oblique shocks which successively increase the static pressure, temperature and density of the inlet flow. Since velocity changes only slightly in the SCRAMJET inlet, a good approximation to the final ratio of inlet density to free stream density, from equation 1, is the contraction ratio. The higher the flight speed the more accurate is this approximation because the velocity change is proportionately less. In determining the effective or aerodynamic contraction ratio the

concept of boundary layer blockage or the discharge coefficient is used. The physical inlet throat area is reduced by the amount of the boundary layer displacement thickness, which is mathematically equivalent to reducing the throat area by multiplying the physical throat area by a discharge coefficient. The discharge coefficient is equal to the physical area minus the area occupied by the boundary layer displacement thickness divided by the physical area.

A well designed SCRAMJET inlet will have a nearly uniform profile outside the boundary layer as the flow enters the combustor. To accomplish this coalescence of the shocks must be avoided which makes it possible to calculate the inlet performance by adding up the results of several oblique shock calculations. The pressure ratio, density ratio and total pressure recovery across each of the inlet shocks can be multiplied together to get the static pressure ratio, density ratio and pressure recovery at any point in the inlet and at the inlet exit. At high flight speeds shock tables can no longer be used because the high temperature air properties are no longer the same as ambient air with a ratio of specific heats, γ , of 1.4. Shock formulas using an appropriate average γ , averaged before and behind the shock, can be used instead with good accuracy.

An example of a simple inlet operating at Mach 16 and a contraction ratio of 30 illustrates a typical inlet analysis. Figure 7 shows the inlet and shock geometry with an exaggerated vertical scale so the geometry is visible. Figure 8 shows the detail of the throat area. The concept is to focus the ramp shocks

duPont Inlet

Contraction Ratio 30

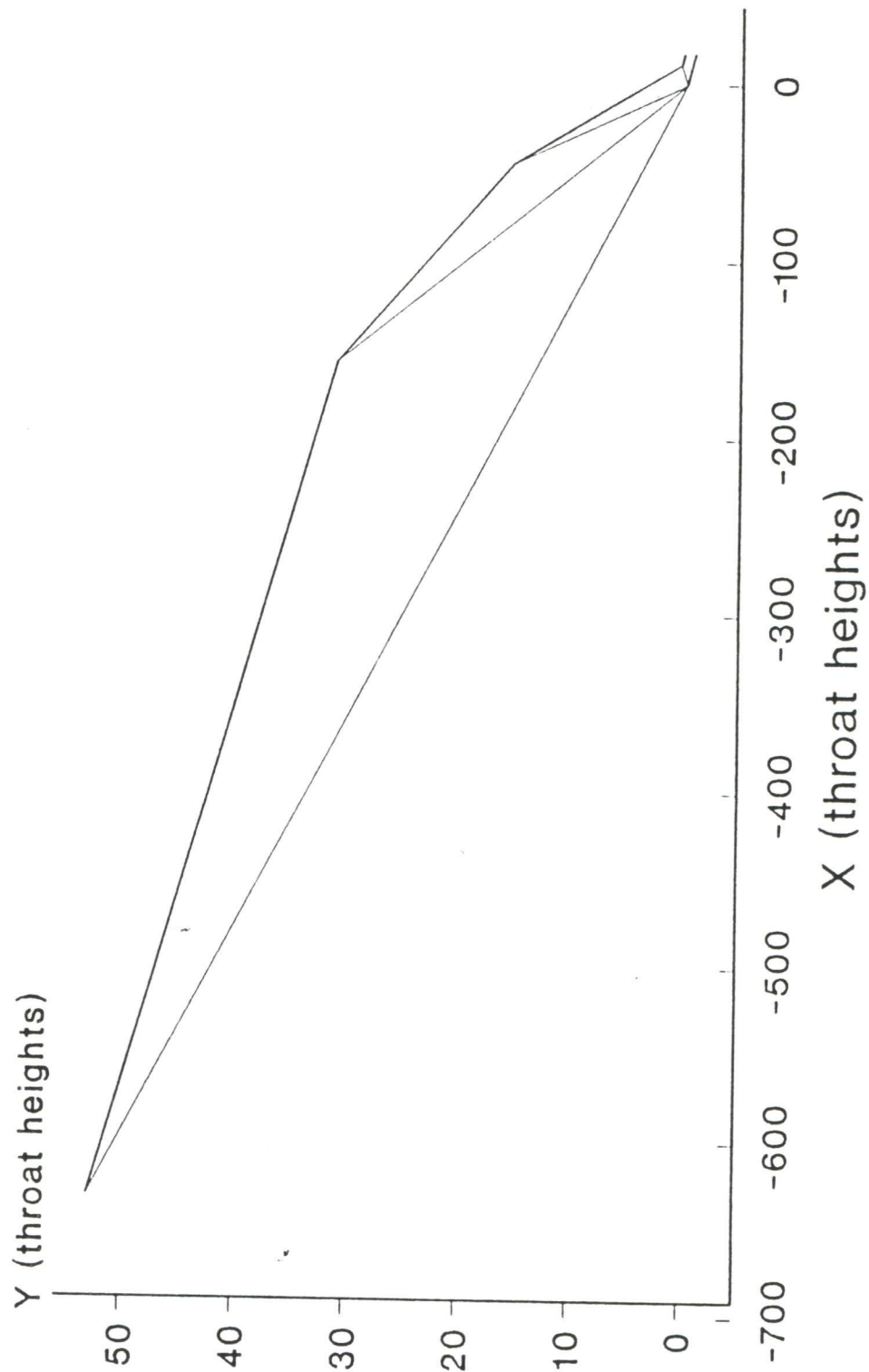


Figure 7

on the cowl lip and then turn the flow back to the combustor direction with a single shock that hits a shoulder at the beginning of the combustor. The corner of the shoulder prevents any reflection of the shock wave and "cancels" the shock. Note that the combustor is inclined a few degrees to the flight path, which is typical of an efficient SCRAMJET installation. Inclining the combustor maintains flow direction roughly parallel to the aircraft wing or forebody and preserves any downwash or lift that was generated by the wing or forebody. Inclining the combustor also reduces the pressure ratio or strength of the cowl generated shock and increases the inlet efficiency or pressure recovery.

A higher performance inlet designed on the same principles is shown in Figures 9 and 10. This inlet has a contraction ratio of 55 at Mach 16 and uses three cowl shocks to turn the flow back to the combustor direction. Using three shocks instead of one greatly increases the pressure recovery of the inlet. The low pressure rise associated with each shock also eliminates the possibility of separating the boundary layer due to an excessive pressure ratio across the shock.

Now that we can see what a typical inlet looks like, it is appropriate to introduce the concept of inlet efficiency. At low speeds inlet efficiency is measured by pressure recovery which is the total pressure after the inlet compression process divided by the free stream total pressure. Total pressure is the static pressure that would be obtained if the flow was brought to rest, or stagnation conditions, isentropically. For a constant gamma

duPont Inlet

Contraction Ratio 30

Cowl Detail

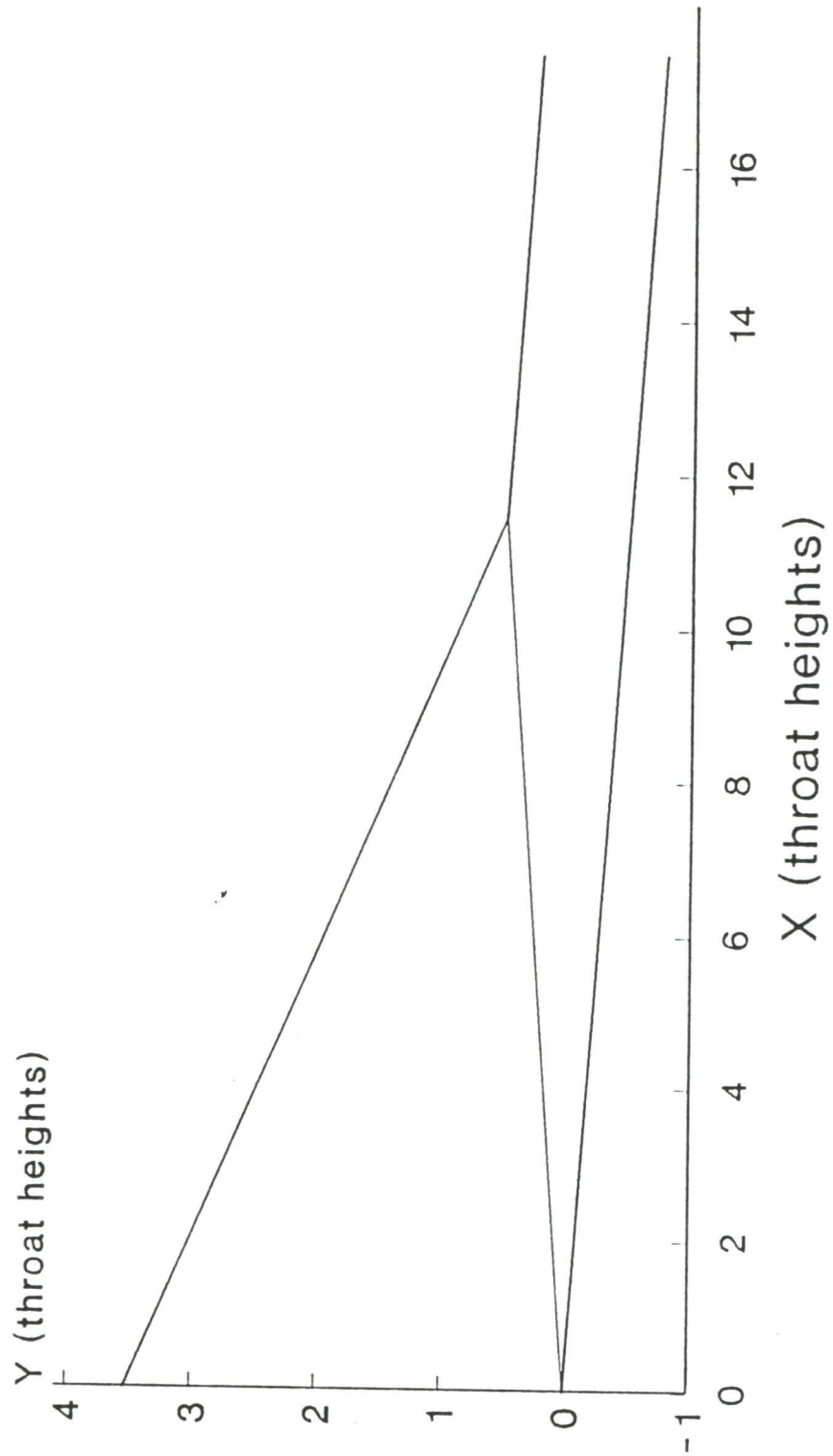


Figure 8

duPont Inlet

Contraction Ratio 55

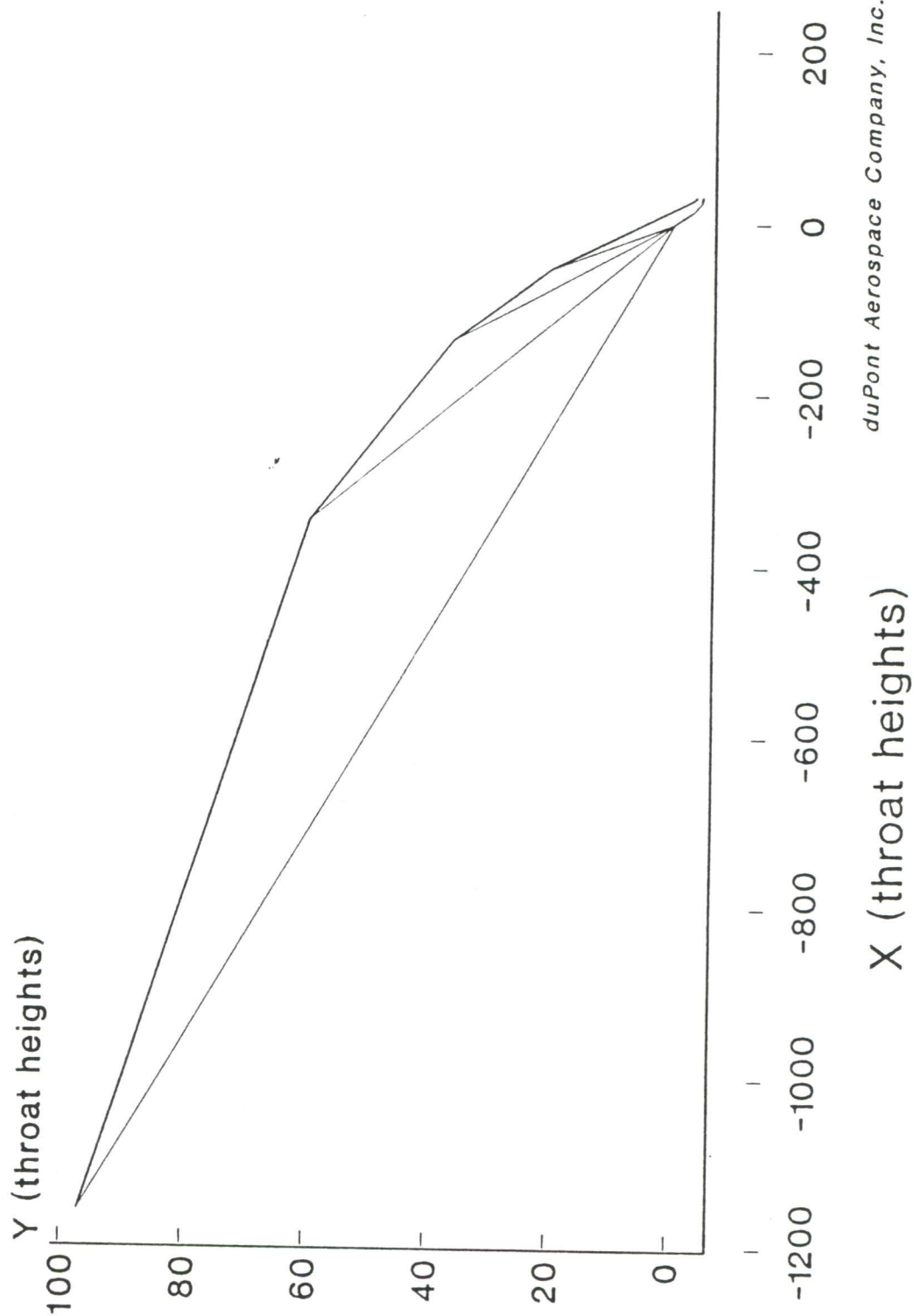


Figure 9

duPont Inlet

Contraction Ratio 55

Cowl Detail

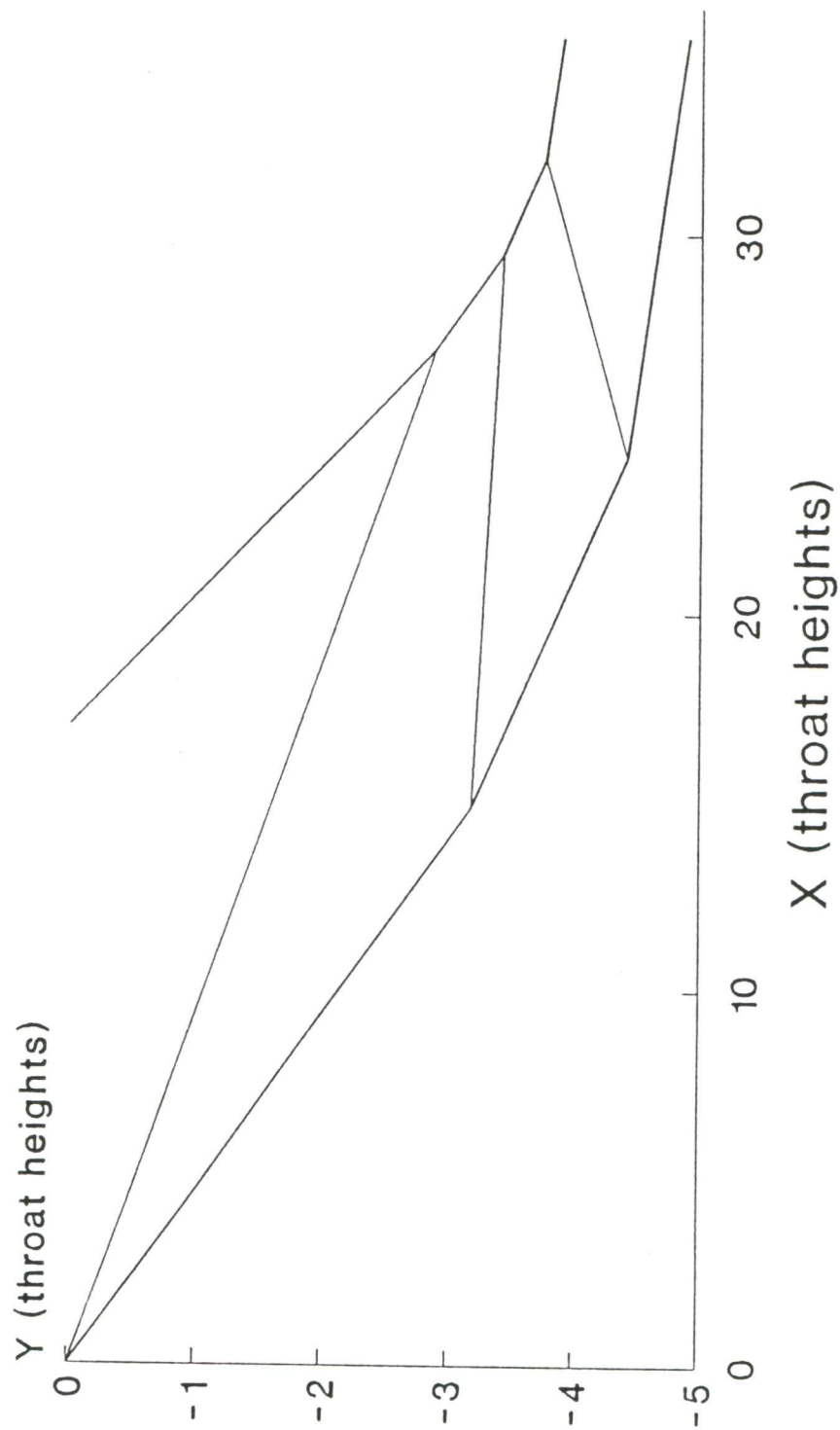


Figure 10

the total pressure is related to the flow Mach number by

$$P_T/P_S = 1 + \frac{\gamma - 1}{2} M^2$$

SCRAMJETS use the concept of kinetic energy efficiency to define inlet performance. Kinetic energy efficiency is defined as the ratio of the square of the velocity attainable by isentropically expanding the inlet exit flow conditions to ambient pressure divided by free stream velocity squared. If the inlet process is isentropic, the kinetic energy efficiency is 100%. At constant gamma and adiabatic flow the kinetic energy efficiency, η_{KE} , is related to the pressure recovery by

$$\eta_{KE} = \left[\frac{1 - [(P_T^\infty/P_T)^{\gamma-1/\gamma}] - 1}{\frac{\gamma-1}{2} M_\infty^2} \right]$$

For nonadiabatic flow, typical of an actual inlet,

$$\eta_{KE} = H_2/H^\infty \left[\frac{1 - [(P_T^\infty/P_T)^{\gamma-1/\gamma}] - 1}{\frac{\gamma-1}{2} M_\infty^2} \right]$$

where H_2 is the total enthalpy at the inlet throat and $H^\infty - H_2$ is the heat, per unit mass of flow, transferred to the inlet walls. The three ratios: contraction, kinetic energy efficiency and H_2/H^∞ uniquely define the conditions at the inlet throat. These are the three boundary conditions necessary to solve the four basic equations for the inlet. The wall integral and the skin friction integral in the momentum equation (2) are accounted for by η_{KE} and H_2/H^∞ , and the heat transfer, q' , in the energy

equation (3) is the difference between H^∞ and H_2 .

Skin friction affects the inlet compression process primarily with the energy difference due to heat transfer in the energy equation and the discharge coefficient due to the boundary layer displacement effect. The effect of inlet skin friction on momentum is small because the pressure integral in the momentum equation is large relative to the skin friction integral. Another interpretation is that the hypersonic boundary carries relatively little mass due to its relatively high average temperature and correspondingly low average density. Therefore, momentum changes in the boundary layer caused by wall shear have a relatively small effect. A working approach is to reduce η_{KE}

by a fraction of $\frac{H^\infty - H_2}{H^\infty}$ that has been obtained by a detailed

study of the boundary layer under similar conditions. The fraction is much less than unity, typically 25 percent. A value of 25% was used in the analysis of the two examples of Mach 16 inlets previously shown.

Results of the analysis of the two inlets shown in Figures 7 and 8 and Figures 9 and 10 are presented in Table 1 and Table 2. The procedure is to locate the shocks by appropriately orienting the displacement surface to obtain the desired inlet shock pattern. Conditions behind the shocks are then used in a boundary layer analysis to determine heat transfer per unit area,

Table 1
INLET DESIGN NUMBERS
MACH 16 CONTRACTION 30

$$\eta_{KE} = 0.9739$$

$$H_{T_2} / H_{T_\infty} = 0.9823$$

$$\eta_{KE/H_{T_2}/H_{T_\infty}} = 0.9915$$

$$\eta_{KE} \text{ (INVISCID CORE)} = 0.9959$$

Table 2
INLET DESIGN NUMBERS
MACH 16 CONTRACTION 55

$$\eta_{KE} = 0.9745$$

$$H_{T_2} / H_{T_\infty} = 0.9819$$

$$\eta_{KE/H_{T_2}/H_{T_\infty}} = 0.9925$$

$$\eta_{KE} \text{ (INVISCID CORE)} = 0.9970$$

total heat transfer and the boundary layer displacement thicknesses. The displacement thicknesses are then used to locate the physical inlet surfaces. At duPont Aerospace the inlet cooling system is usually designed as part of a final inlet analysis, so that actual wall temperatures can be used for the boundary layer computations. Cooling has to be at least roughly allowed for since the temperature of many of the panels is quite low because the fuel has to be introduced at pump exit conditions.

Contrary to the intuitive perception that a higher contraction causes increased shock losses, the contraction 55 inlet at an inviscid η_{KE} of 0.9970 is more efficient than the contraction 30 inlet at 0.9959. This is true because the higher contraction geometrically permits more cowl shocks and therefore has lower shock losses in spite of a higher compression ratio.

4.0 THE COMBUSTOR

When the combustor entrance conditions are known either from inlet test data or from the inlet portion of the cycle analysis, the four basic equations can be solved to find the combustor exit conditions, provided the boundary conditions are known or calculable. Fortunately, the optimum SCRAMJET combustor geometry for best performance is a constant area duct for which the wall pressure integral vanishes, eliminating the necessity of measuring or calculating the wall integral. Calculation of this integral for available duct area is a very difficult task involving assumptions about the rate of fuel mixing and therefore combustion in the duct, or scaling of experimental data from a similar duct. Since there

is virtually no experimental data on variable area combustors above Mach 7 flight speeds to support either approach, calculated wall integrals are not very reliable. The other boundary condition, skin friction shear force, is more amenable to estimation.

With a reasonable estimate for the average skin friction coefficient on the combustor walls, all necessary boundary conditions for a constant area, or constant pressure combustor are known and the four basic equations can be readily solved for the combustor exit conditions. One of the demonstrations that constant area is the optimum configuration is to compare it to the constant pressure case, which is a special case of a diverging area combustor. An example of Mach 16 shows that thrust is reduced 1.5 percent in the constant pressure case compared to the constant area case.

The boundary conditions for the combustor that have to be obtained are the skin friction, the fuel temperature, the heat transferred out through the combustor walls and the fuel velocity. Since film cooling of the combustor walls is often employed in SCRAMJETS, all four of these conditions can be interrelated. A brief description of how each one is calculated follows.

Skin friction can be obtained by estimating an average skin friction coefficient using standard calculation procedures for turbulent boundary layers such as the method of Spalding and Chi.¹ Such a skin friction coefficient can be corrected, that is to say reduced, for the effects of film cooling by applying empirical data. An example of such a correction is shown in Figure 11. The

FILM COOLING EFFECTIVENESS

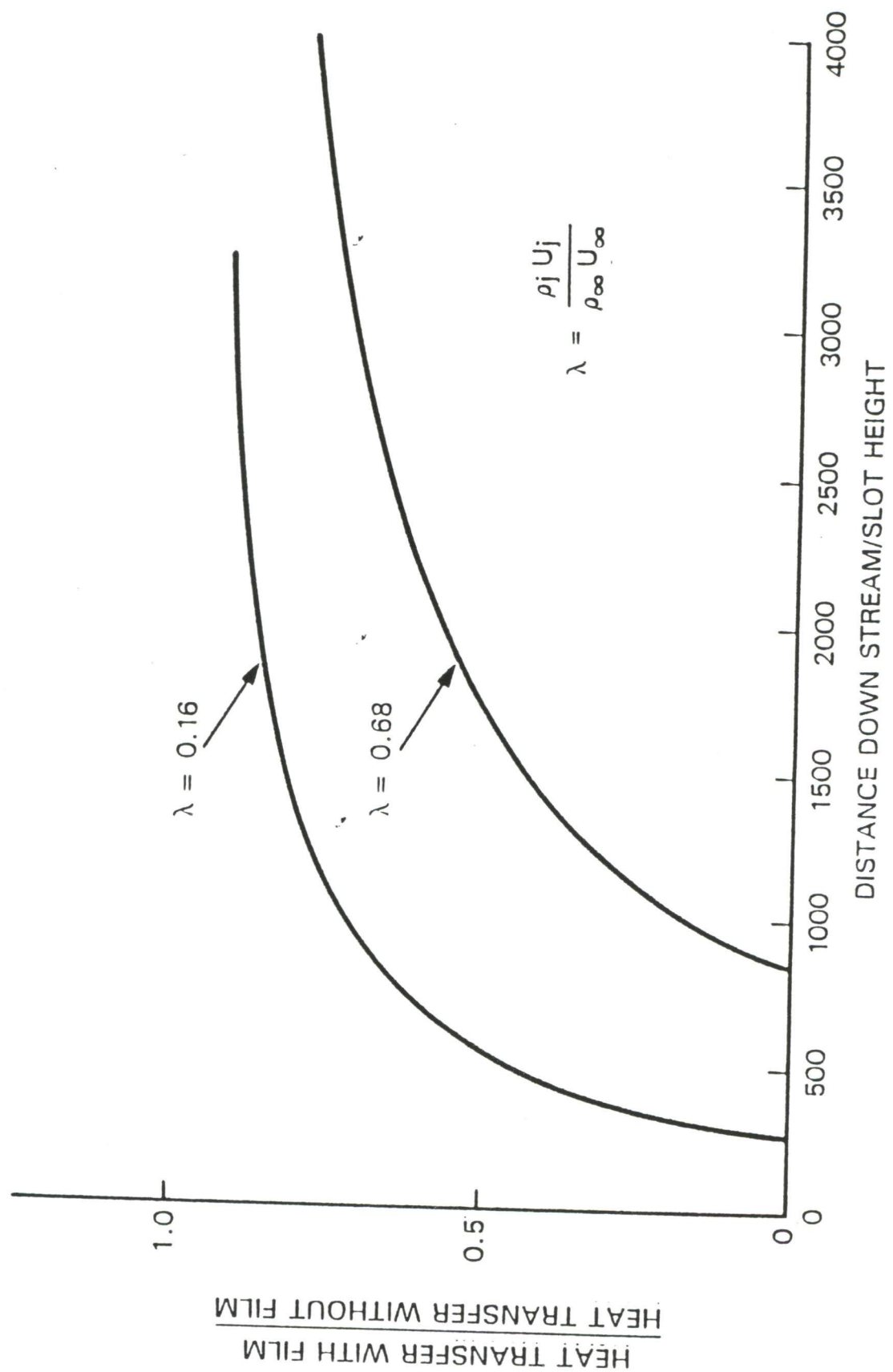


Figure 11

average skin friction coefficient is multiplied by the average of the combustor entrance and combustor exit dynamic pressure and the combustor wetted area to obtain the total shear force. Usually a cycle code will only require the skin friction coefficient and the combustor wetted area as the inputs.

The fuel temperatures is obtained by heat balancing the cycle. All the heat transferred to the fuel in the regeneratively cooled walls of the engine heats the fuel above its storage temperature in the tank. For refined calculations the work input to the fuel to pump it to pressure and any work later extracted in an expander turbine used to run the pump must be accounted for. It is usually a good approximation to use the maximum allowable fuel temperature set by structural limits. This is usually in the vicinity of 1,700 degrees Rankine. Since the back side of a cooled passage operates at close to fuel temperature, the thermal gradient will produce a maximum skin temperature of about 1700° Fahrenheit with a 1700° Rankine fuel temperature. Higher surface temperatures will correspond to higher maximum fuel temperatures.

Heat transferred to the combustor walls must be specified. If there are no severe pressure gradients along the combustor, the heat transfer is related to the Skin friction by Reynolds analogy which says the heat transfer coefficient is one-half the skin friction coefficient. Using Reynolds analogy the heat transfer per unit area is:

$$\dot{q} = \frac{C_f}{2} \times \frac{\dot{W}}{A} \times (h_r - h_w)$$

Where there is a large pressure gradient Reynolds analogy does not apply and there exists high heat transfer and relatively low friction. An approach to this situation, which occurs in most experimental combustor data, is to assume zero skin friction in the vicinity of the pressure rise and use a high skin friction coefficient, perhaps twice as high, in this region for purposes of heat transfer computations using Reynolds analogy. There has been some theoretical work by Zane Pinkney² that support this type of approach, but its primary justification today rests on agreement with data.

The fuel velocity is obtained by a nozzle calculation. Sonic velocity is used if the nozzle is an orifice. Expansion to the available area is used if the nozzle is a convergent divergent nozzle similar to a rocket nozzle. In this case an isentropic expansion is used with an appropriate velocity coefficient to reduce the velocity for nozzle losses; a typical number is 98%. The velocity is multiplied by the cosine of the injection angle relative to the combustor walls in order to obtain the component fuel momentum parallel to the air flow. The four basic equations, rewritten for a constant area combustor, are:

$$\dot{W}_{air} + \dot{W}_{Fuel} = \dot{W}_{Combustor\ Exit} = \rho_2 V_2 A_{Combustor} \times g$$

$$(P_1 - P_2) A_{Combustor} + \frac{\dot{W}_{air}}{g} V_{Entrance} + \frac{\dot{W}_{Fuel}}{g} V_{Fuel} \cos \beta$$

$$\frac{(\dot{q}_{Entrance} + \dot{q}_{Exit})}{2} \times A_{Wetted} \times C_f = \frac{\dot{W}_{Combustor\ Exit}}{g} V_{Exit}$$

$$\dot{W}_{Air} \times H_{Entrance} + \dot{W}_{Fuel} \times H_{Fuel} - \dot{q} \times A_{Wetted} = \dot{W}_{Exit} \times H_{Exit}$$

$$P_1 = \rho_1 R_1 T_1 \text{ and } P_2 = \rho_2 R_2 T_2$$

Where R is appropriate to the entrance (1) and exit (2) conditions.

The unknowns are the exit conditions: V_2 , P_2 , ρ_2 and T_2 . With four equations and four unknowns the exit conditions can be obtained.

5.0 THE NOZZLE

In cycle analysis the nozzle is not generally solved by using the four basic equations, but an isentropic expansion is used with appropriate loss factors called thrust coefficients. This treatment is convenient because the thermochemistry program only provides an equilibrium gas composition. The nozzle expansion is typically done with an equilibrium composition throughout the expansion process or with the composition frozen at the combustor exit composition called a frozen expansion. The typical cycle analysis is limited to these two extremes, but the actual expansion is somewhere in between.

Methods of solving the nozzle expansion including the efforts of kinetic recombustion rates have been developed. They include Computerized Flow Development, CFD, codes such as the SAIC SCHNOZ code¹ and method of characteristics methods developed by SEA² and GASL³. An approach to nozzle analysis is to develop thrust coefficients from two dimensional analysis that will define the kinetic losses, skin friction and divergence losses by a thrust coefficient which is the measured thrust or the analytical thrust divided by the isentropic equilibrium expansion thrust. The result is a number such as .985 which can be multiplied by the gross nozzle thrust from an equilibrium expansion to the available area to obtain the actual thrust. Once a value for the thrust coefficient has been carefully determined, it can be applied to other similar cycle cases with very little error. If the impact on thrust coefficient of the important nozzle variables such as area ratio, Reynolds number, Mach number and combustor exit temperature are known, the thrust coefficient approach can be used accurately over a wide range of cycle possibilities without further detailed nozzle calculations.

A preliminary design approach, originally suggested by Marino of GASL in 1960, of using 10% of the static enthalpy drop across the nozzle as the measure of nozzle losses has shown surprisingly good agreement with experimental and analytical data throughout the SCRAMJET operating range. This approach gives much better results than using a single number for the thrust coefficient. It, for example, avoids the obviously absurd answer that thrust would be

penalized even when there is no nozzle expansion. In SCRAMJETS, where in some cases the combustor exit flow is 90% or more of the gross thrust, an approach which relates nozzle losses to the part of the thrust created by the nozzle makes more sense than a straight fraction of gross thrust.

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1. Gordon, Sanford; and McBride, Bonnie J. : One-dimensional equilibrium. (a preliminary report, NASA Lewis)
2. Spalding, D. B.; and Chi, S. W. : The drag of a compressible turbulent boundary layer on a smooth flat plate with and without heat transfer. Journal Fluid Mechanics Vol. 18. October 1966. pp. 117-143.
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4. SAIC : SCHNOZ, a parabolized Navier Stokes nozzle code.
5. Nickerson, Gary : TDK, a two dimensional method of characteristics nozzle code with finite rate chemistry.
6. Chinitz, W. : NOZ, a two dimensional method of characteristics nozzle optimization code with finite rate chemistry.

EXAMPLE OF HIGH PERFORMANCE SCRAMJETS

PREPARED FOR
X-30 PLANNING

by

duPont Aerospace Company, Inc.

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La Jolla, California 92037
Telephone: (619) 456-7833

July 11, 1990

NASP HIGH SPEED PERFORMANCE

The size of the NASP X-30 aircraft and the potential utility of follow on production aircraft depends on the level of fuel specific impulse attained by the SCRAMJET engine between Mach 16 and Mach 24. At low levels approaching the 455 seconds attainable with a rocket engine, a rocket is preferred over a SCRAMJET because of the roughly five times higher density of a LOX hydrogen mixture as compared with pure hydrogen and the ability to operate outside the atmosphere eliminating drag. Any heavy use of rockets above Mach 16 results in a heavy X-30 with limited to no follow on potential. A specific impulse well above the rocket level results in a small X-30 with potential for follow on aircraft to reduce the cost of space transportation by a factor of between ten and a hundred. The critical issue for the X-30 is the one that it was conceived to resolve, namely the engine performance attainable between Mach 16 and 24.

This paper presents an example of a SCRAMJET operating at Mach 16 and at Mach 24. The data is presented in a manner that can be verified by standard aerodynamic methods. The data was generated using the most accurate methods available. It was prepared for the use of persons involved in NASP planning to show conclusively that excellent airbreathing engine performance is attainable between Mach 16 and 24. When this performance level is used, the X-30 is a small aircraft in the 50,000 pound class, and larger airplanes have payload capacity equal to or greater than the Space Shuttle.

ENGINE PERFORMANCE SUMMARY

	MACH 16	MACH 24
ISP (frozen nozzle)	1,202 sec	716 sec
ISP (equilibrium nozzle)	1,408 sec	818 sec
ETAKE	0.98108	0.98643
ETAKE/ $H_T/H_{T\infty}$	0.99205	0.99479
ETAKE inviscid	0.99546	0.99702
CFG (frozen nozzle)	0.9687	0.9824
CFG (equilibrium nozzle)	0.9843	0.9912

FIXED COWL INLET COORDINATES

50,000 POUND AIRPLANE

FIXED COWL

X (in)	Y (in)
841.50	83.21
870.10	84.87
878.26	85.52

MACH 24 BODY

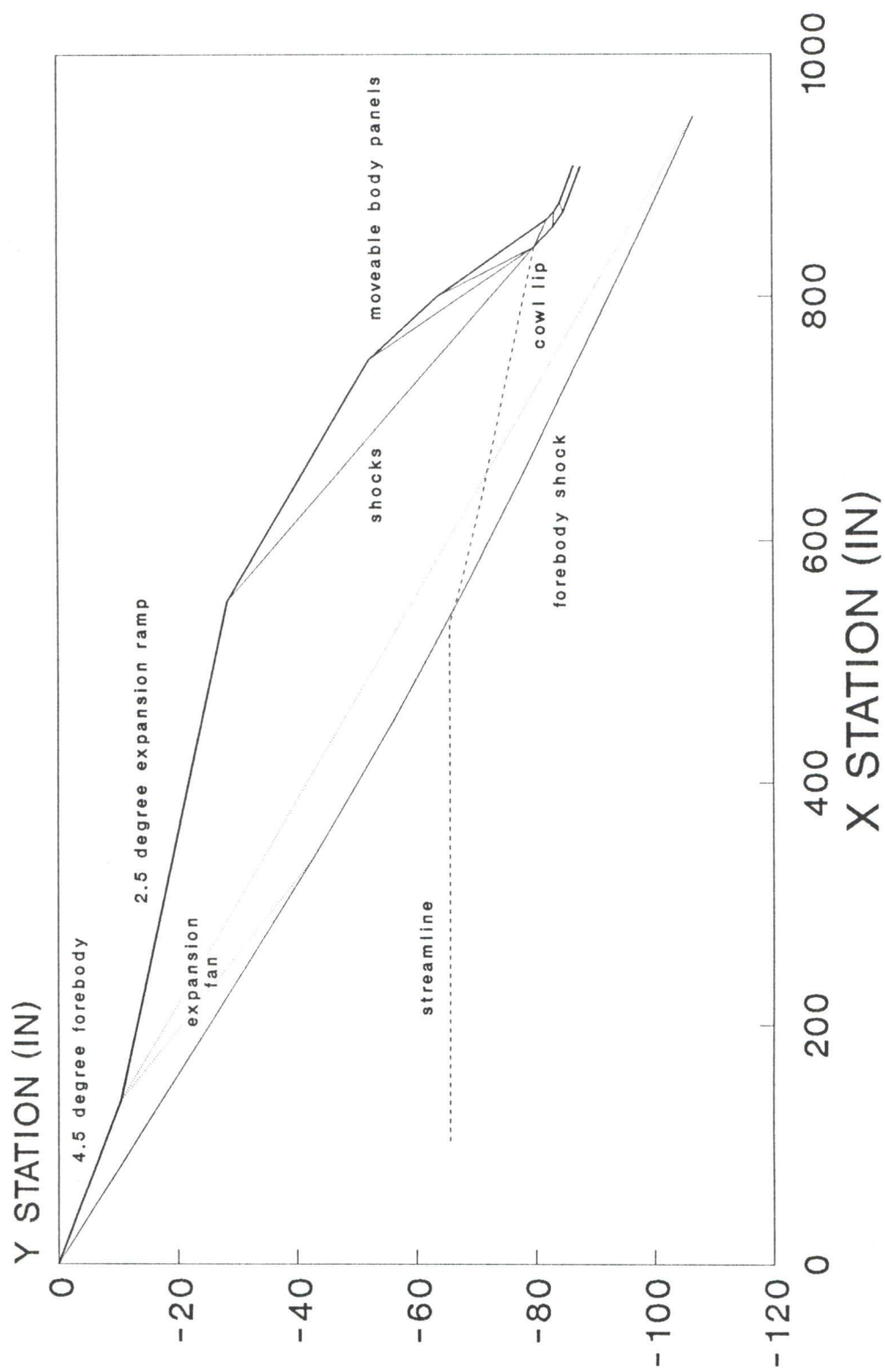
X (in)	Y (in)
0.00	0.00
136.96	10.39
420.55	22.32
698.55	49.31
790.97	64.38
867.38	83.05
871.76	83.90
877.50	84.69

MACH 16 BODY

X (in)	Y (in)
0.00	0.00
136.96	10.39
550.42	28.34
747.95	52.25
800.00	63.69
864.64	82.24
870.65	83.38
878.26	84.32

INLET SHOCK GEOMETRY

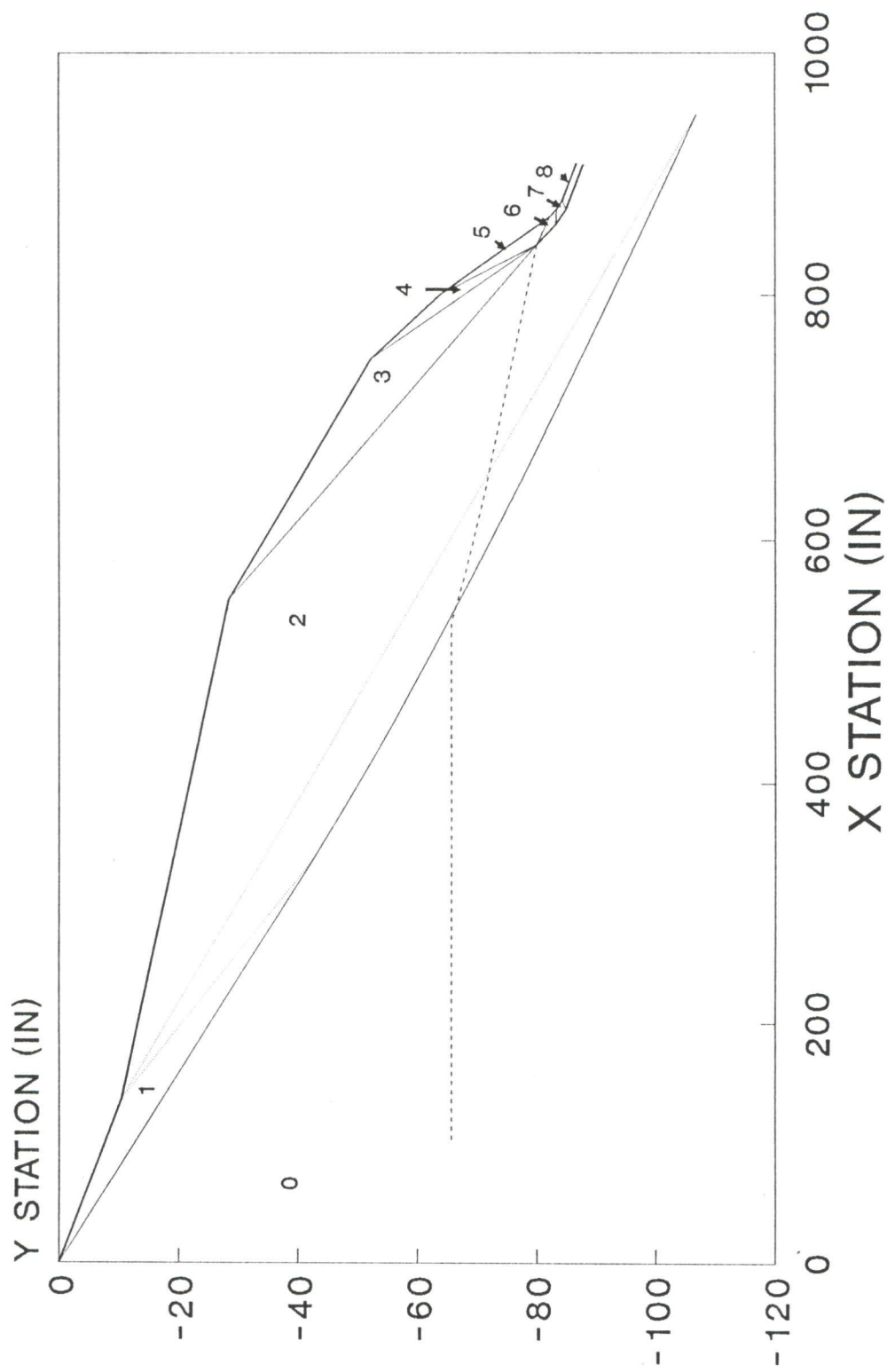
MACH 16



duPont Aerospace Company, Inc.

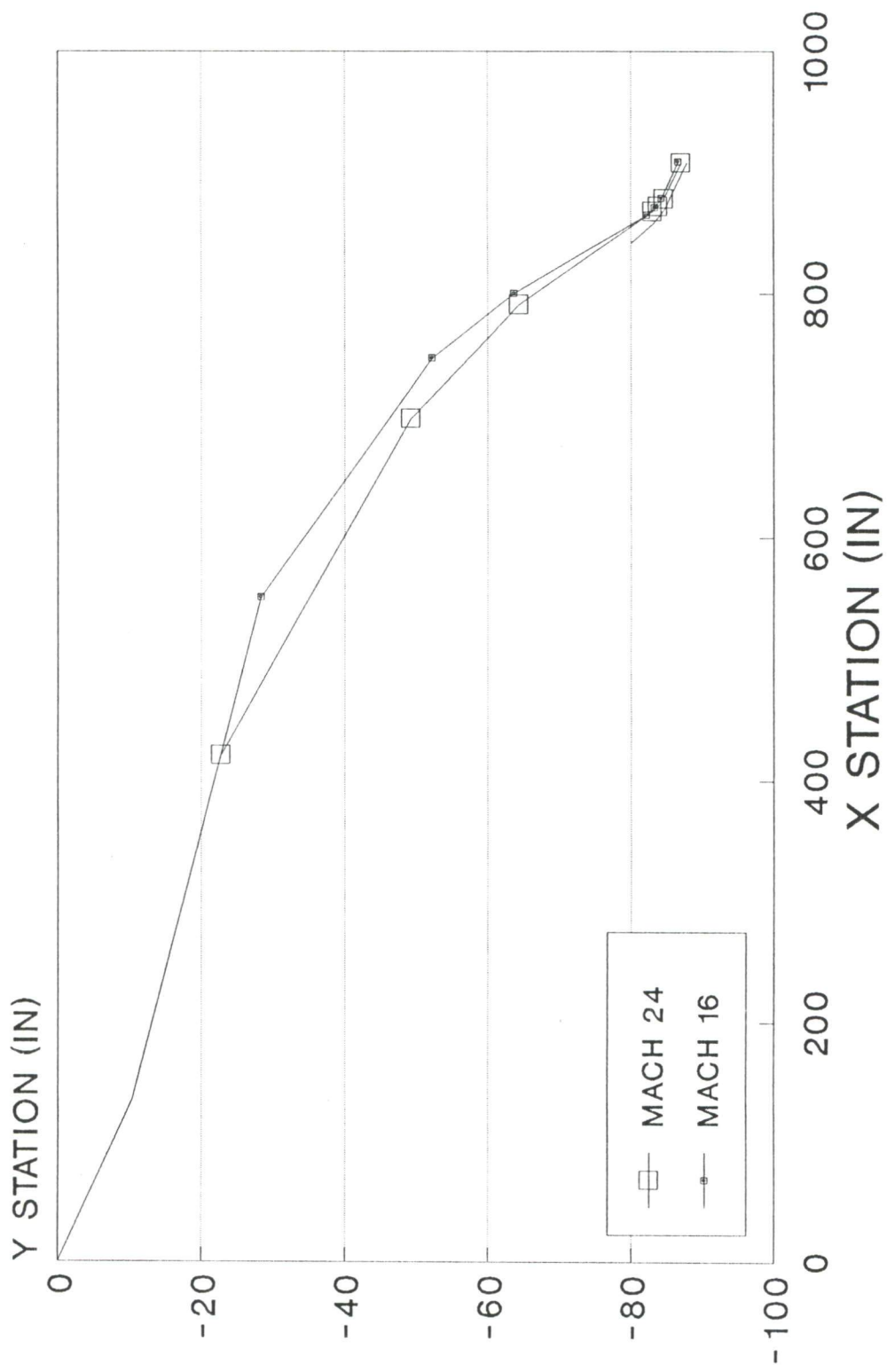
INLET SHOCK GEOMETRY

MACH 16



duPont Aerospace Company, Inc.

INLET RAMPS AND COMBUSTOR GEOMETRY FIXED COWL



duPont Aerospace Company, Inc.

MACH 16 FLOWPATH

INLET INVISCID FLOWFIELD QUANTITIES

	INLET STATION				
	0	1	2	3	4
P (PSI)	0.0775	0.3501	0.1859	0.6665	2.376
T (R)	428	720	596	901	1334
V (FT/S)	16229	16119	16167	16049	15870
M	16.0	12.3	13.5	11.0	8.99
Mn1		2.01	NA	1.80	1.80
ANGLE (DEG)	0.00	4.50	2.50	6.91	12.39
GAMMA	1.400	1.395	1.399	1.386	1.360
	5	6	7	8	
P (PSI)	4.909	12.35	19.66	32.50	
T (R)	1624	2080	2333	2638	
V (FT/S)	15746	15539	15422	15278	
M	8.13	7.14	6.71	6.27	
Mn1	1.39	1.53	1.23	1.26	
ANGLE (DEG)	16.02	10.76	7.86	4.50	
GAMMA	1.344	1.325	1.319	1.313	

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MACH 16 FLOWPATH

ENGINE CYCLE QUANTITIES

	FREESTREAM	INLET EXIT	COMBUSTOR EXIT	NOZZLE EXIT
A (SQFT)	65.6	1.158	1.158	130.4
MASS (LBM/S)	520.2	520.2	543.0	543.0
P (PSI)	0.0775	32.50	93.63	0.1669
T (R)	428	2,928	5,543	1,339
V (FT/S)	16,229	15,077	14,402	17,327
M	16.0	5.88	3.66	8.32
GAMMA	1.400	1.307	1.160	1.347
MOL. WT.	28.96	28.96	20.69	20.69
MOMENTUM (LBS)	263,271	249,184	258,686	295,555
HEAT (BTU/S)	-30,109	-63,713	-62,587	
FRICTION (LBS)	3,235	1,223	7,367	

CONSTANT AREA COMBUSTION WITH DISCHARGE COEFFICIENT
APPLIED AT INLET

EQUIVALENCE RATIO OF 1.5

FUEL HEATED TO 2000 DEGREES R AND EXPANDED FROM 2500 PSIA
DURING INJECTION

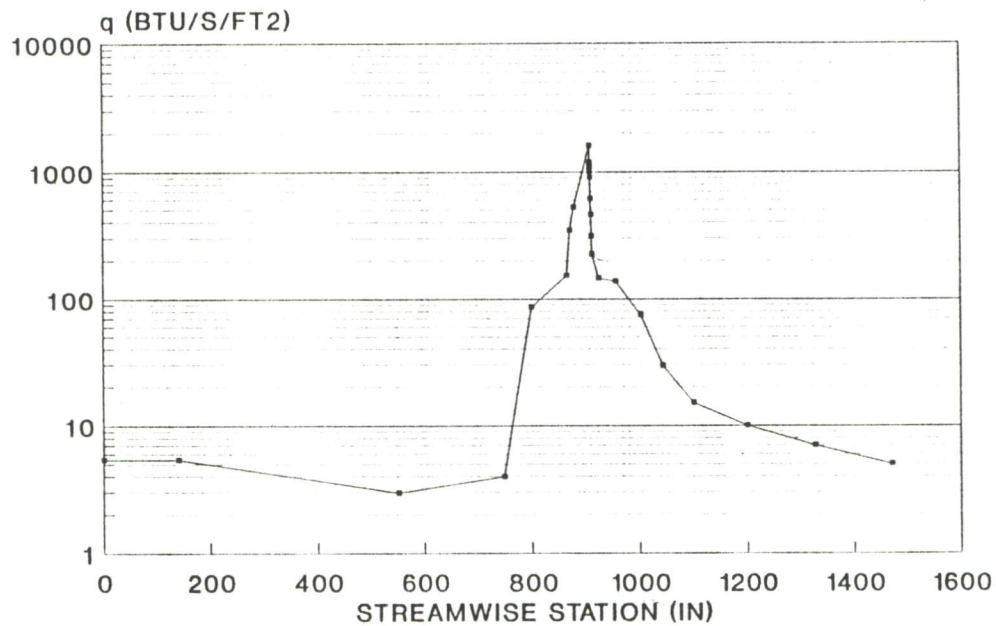
COMBUSTION EFFICIENCY OF 100 PERCENT

SPILLAGE DRAG OF 426 LBS (ABSOLUTE PRESSURE DATUM)

duPont Aerospace Company, Inc.

BODY HEAT LOADS

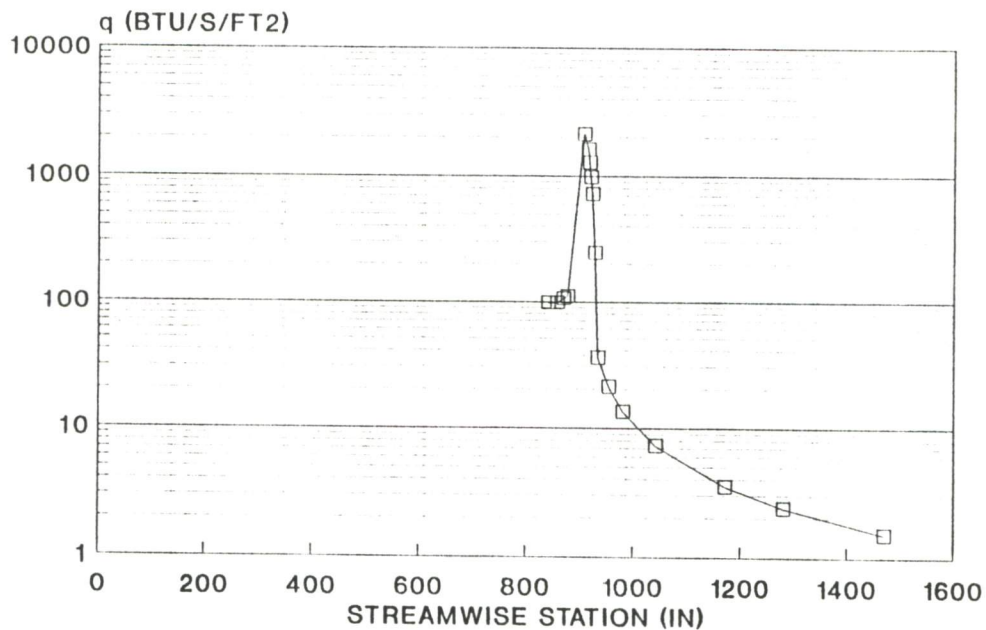
MACH 16, Q=2000 PSF



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COWL HEAT LOADS

MACH 16, Q=2000 PSF



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MACH 24 FLOWPATH

ENGINE CYCLE QUANTITIES

	FREESTREAM	INLET EXIT	COMBUSTOR EXIT	NOZZLE EXIT
A (SQFT)	80.0	0.755	0.755	130.4
MASS (LBM/S)	198.7	198.7	213.8	213.8
P (PSI)	0.0172	17.16	42.80	0.0438
T (R)	484	4,322	5,320	1,035
V (FT/S)	25,887	24,721	23,458	25,805
M	24.0	8.07	5.36	12.31
GAMMA	1.400	1.262	1.167	1.374
MOL. WT.	28.96	28.91	16.10	16.10
MOMENTUM (LBS)	160,174	154,526	160,513	172,274
HEAT (BTU/S)		-22,336	-70,356	-12,886
FRICTION (LBS)		1,501	1,228	872

CONSTANT AREA COMBUSTION WITH DISCHARGE COEFFICIENT
APPLIED AT INLET EXIT (CD = 103.9/106.0 = 0.980)

EQUIVALENCE RATIO OF 2.6

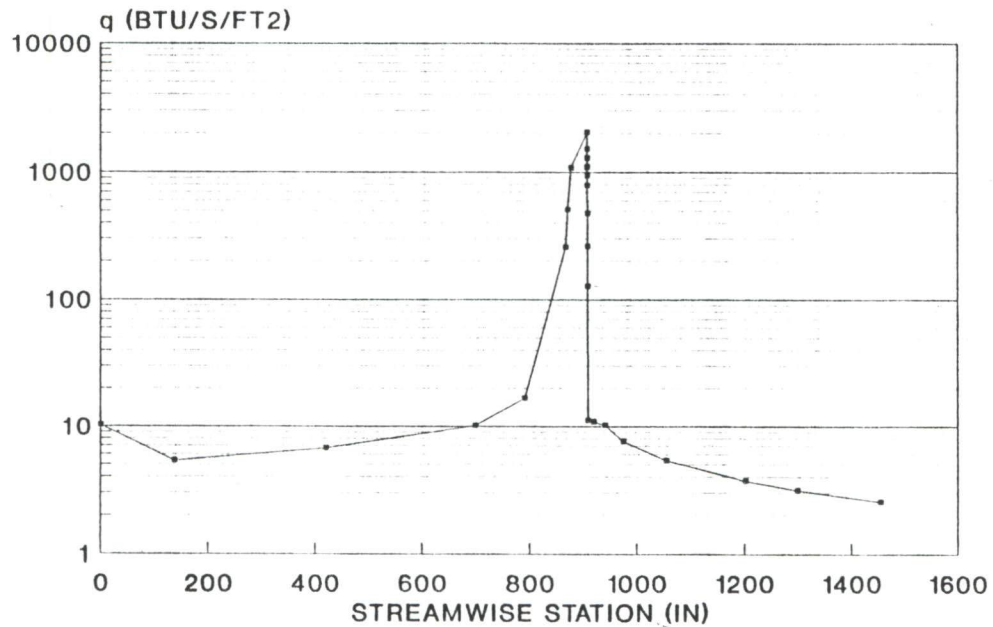
FUEL HEATED TO 2000 DEGREES R AND EXPANDED FROM 2500 PSIA
DURING INJECTION

COMBUSTION EFFICIENCY OF 100 PERCENT

duPont Aerospace Company, Inc.

BODY HEAT LOADS

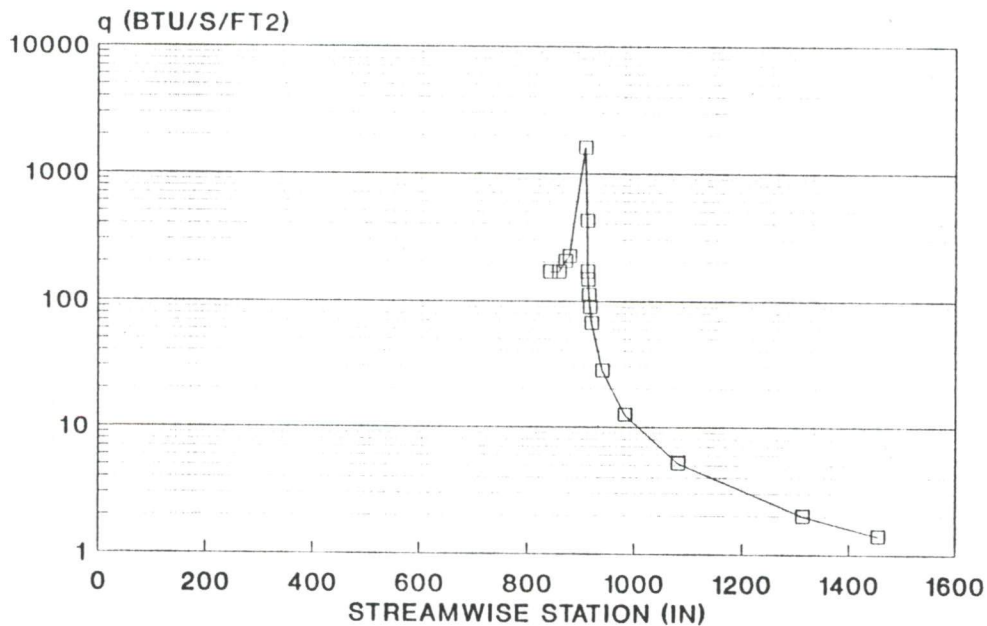
MACH 24 , Q=1000 PSF



duPont Aerospace Company, Inc.

COWL HEAT LOADS

MACH 24 , Q=1000 PSF



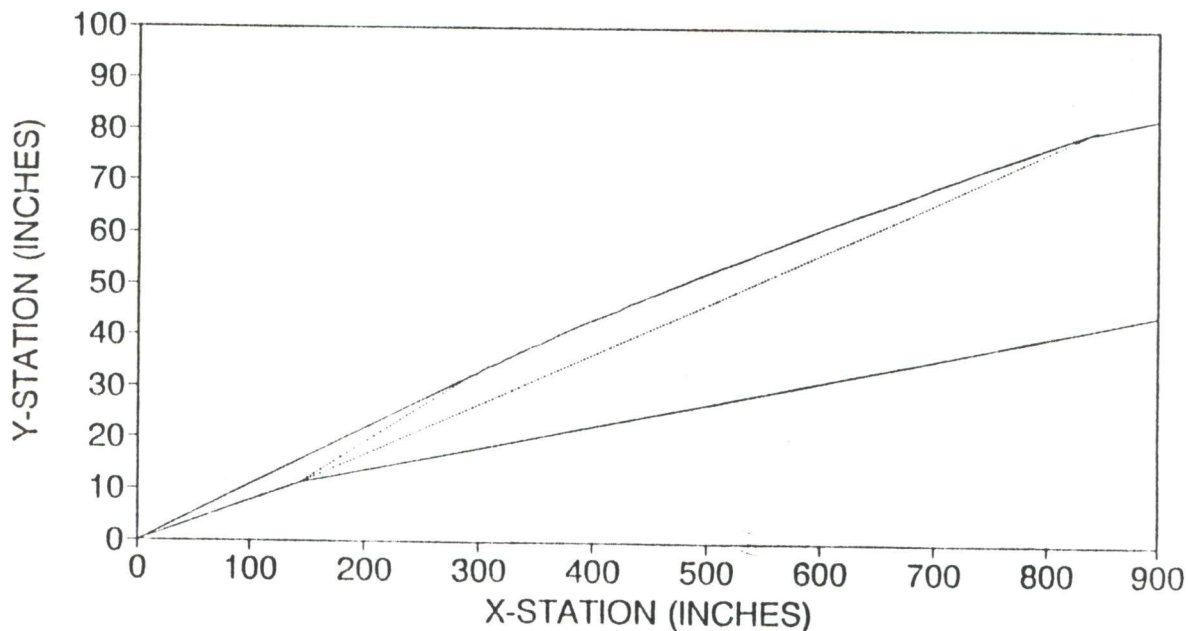
duPont Aerospace Company, Inc.

MACH 24 NOZZLE PERFORMANCE

	ISP	CFG
FROZEN	716	0.9824
KINETIC	746	0.9846
EQUILIBRIUM	818	0.9912

MACH 24 SHOCK GEOMETRY

WEDGE WITH EXPANSION RAMP

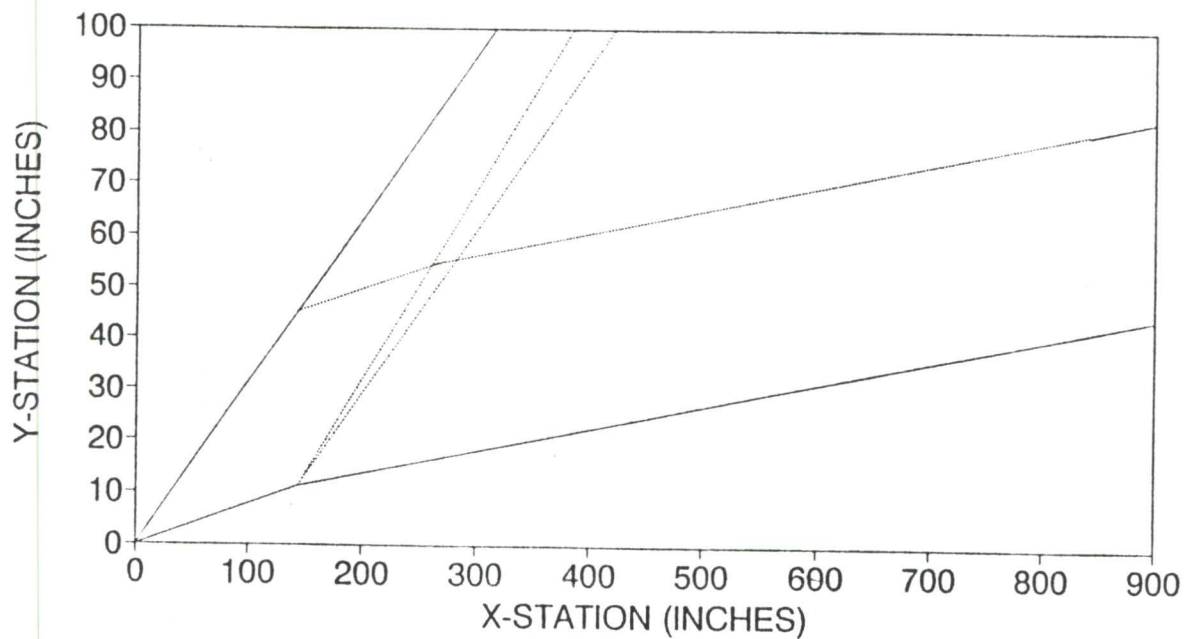


The forebody geometry used to generate the engine performance is shown above. It is an easily analyzable configuration that represents the latest ogive forebody of the Government Baseline aircraft. The Government Baseline is a Configuration developed by duPont in 1983 under DARPA contract to show the industry what it was expected to do and, if possible, surpass. It is still the lowest drag and highest performing airplane design in the program.

The latest version of the Government Baseline has an axisymmetric ogive forebody which starts with a five degree half angle cone. To have a simple, easily audited propulsion analysis the ogive is replaced by two ramps. The initial one is inclined at 4.5 degrees to give approximately the same compression as a five degree half angle cone. The second one is inclined at 2.5 degrees, the same angle as the Government Baseline wing lower surface. At Mach 24 the curved shock resulting from the two degrees expansion at the junction of the two wedges just touches the inlet cowl lip just behind the last expansion wave from the junction. The result is a uniform flow at the inlet plane with the same compression history as the actual three dimensional flow beneath the Government baseline Aircraft.

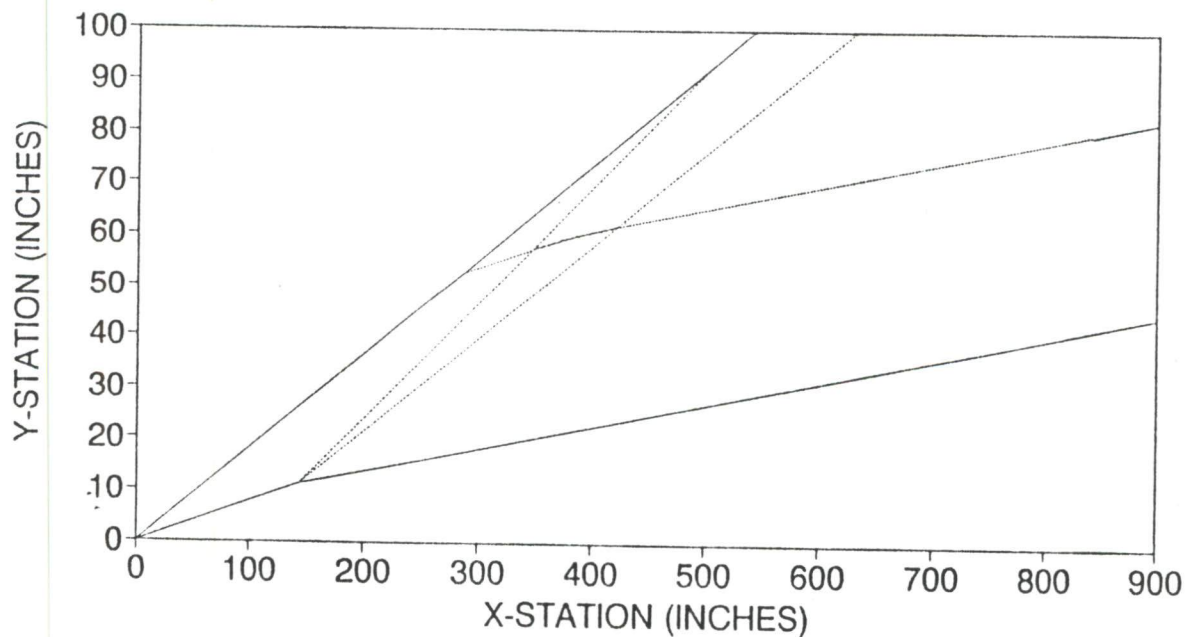
MACH 4 SHOCK GEOMETRY

WEDGE WITH EXPANSION RAMP



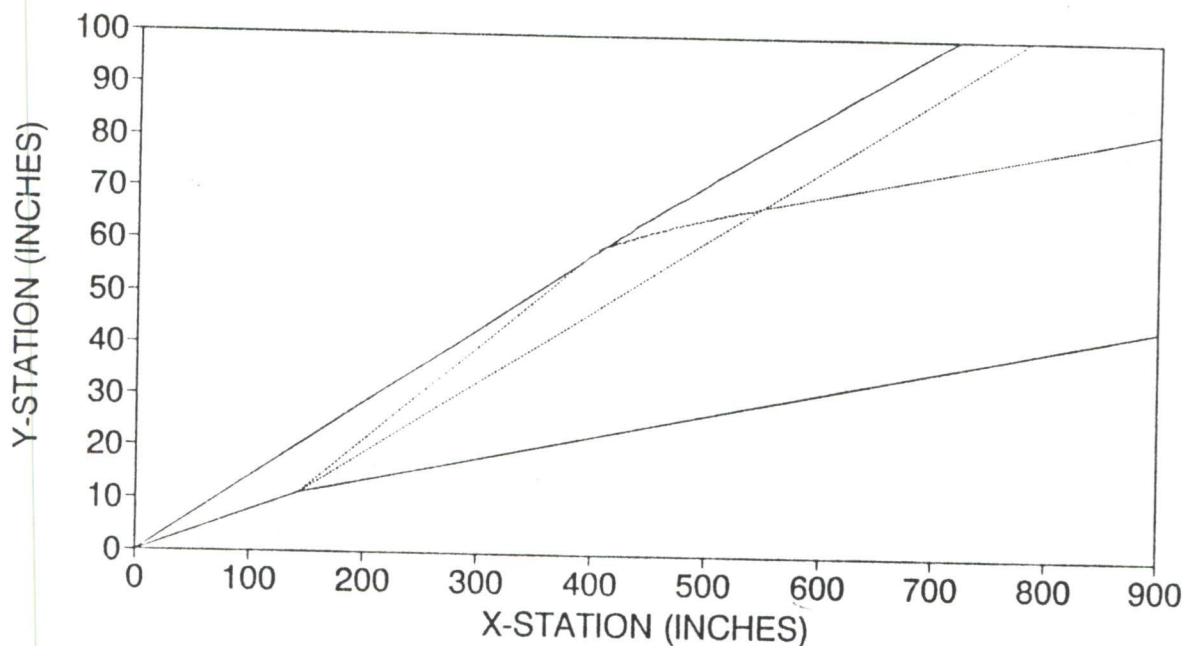
MACH 8 SHOCK GEOMETRY

WEDGE WITH EXPANSION RAMP



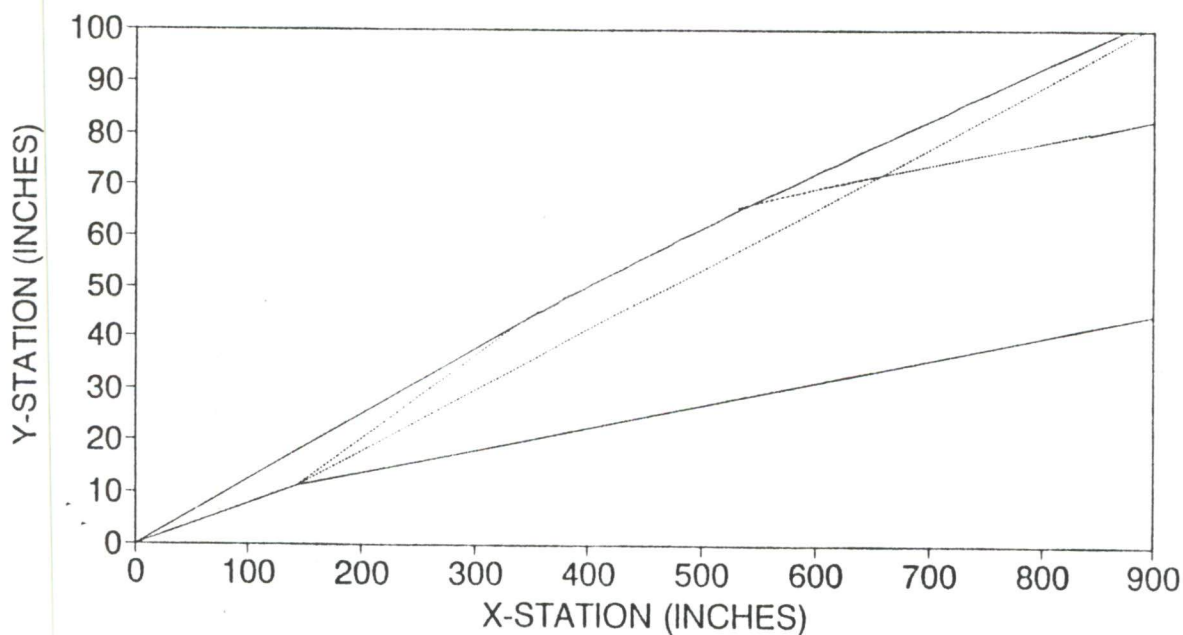
MACH 12 SHOCK GEOMETRY

WEDGE WITH EXPANSION RAMP



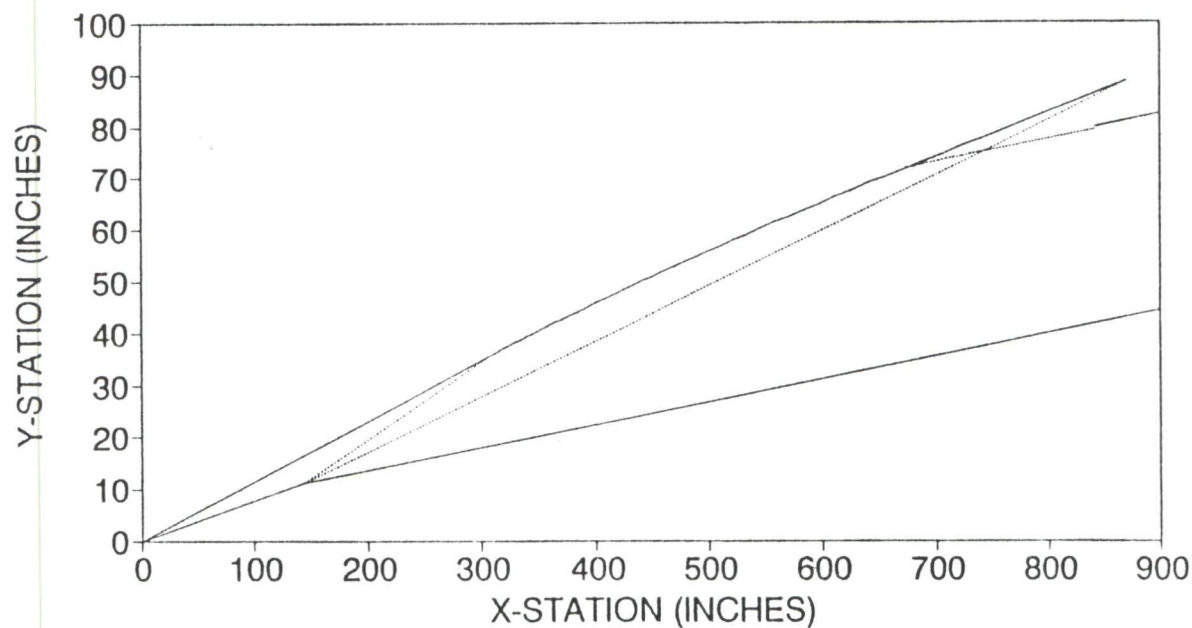
MACH 16 SHOCK GEOMETRY

WEDGE WITH EXPANSION RAMP



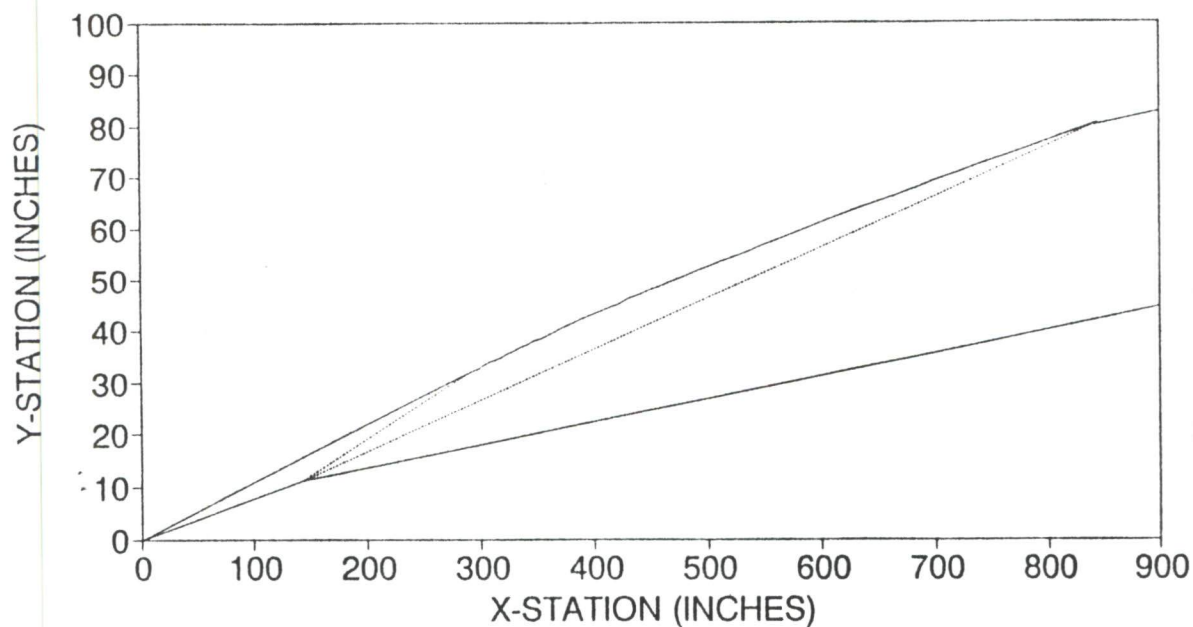
MACH 20 SHOCK GEOMETRY

WEDGE WITH EXPANSION RAMP

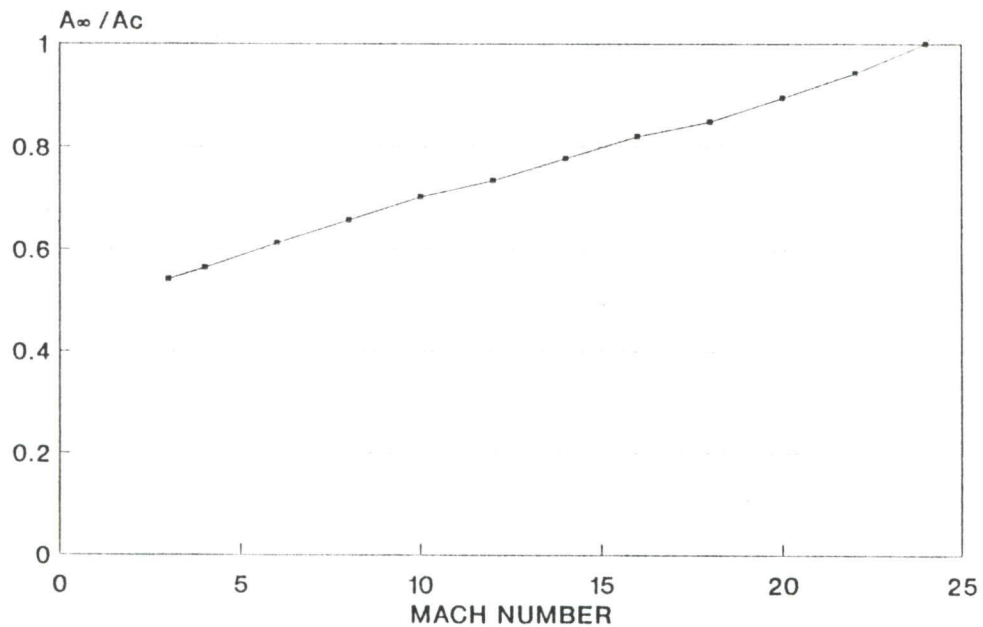


MACH 24 SHOCK GEOMETRY

WEDGE WITH EXPANSION RAMP

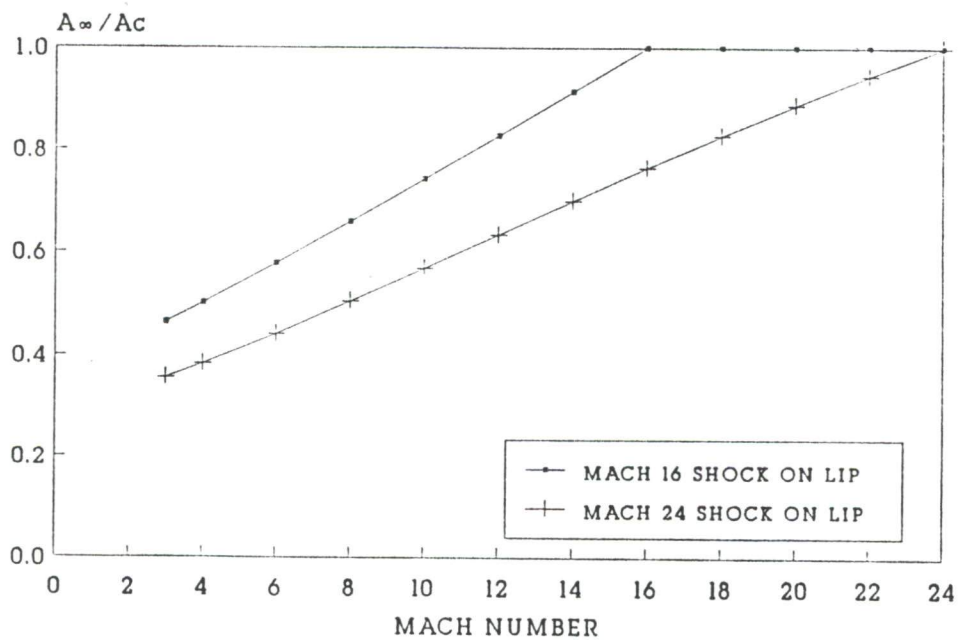


FREESTREAM MASS CAPTURE SHOCK WITH EXPANSION



duPont Aerospace Company, Inc.

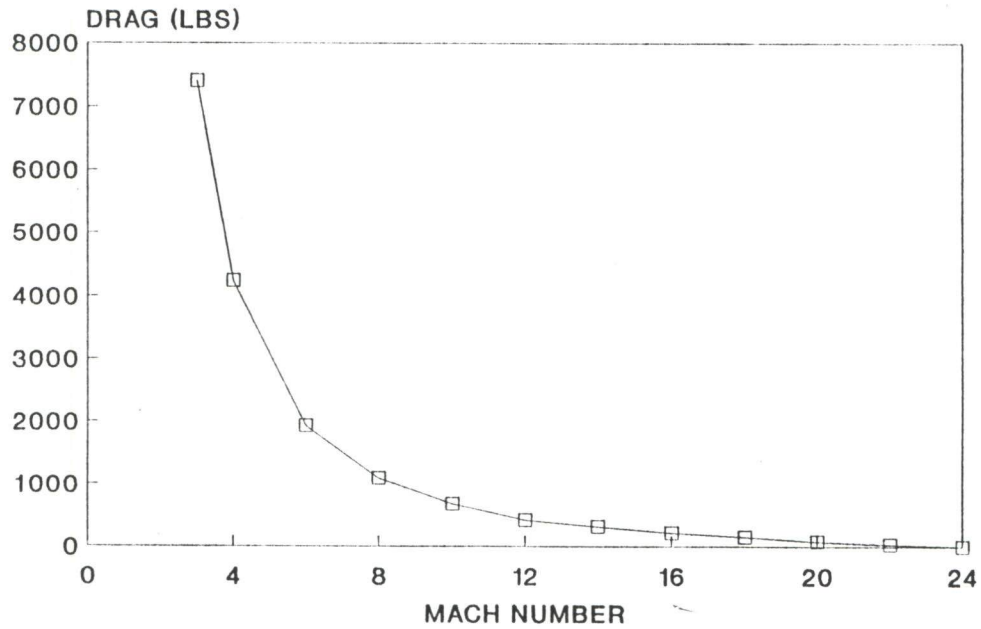
FREESTREAM MASS CAPTURE FIXED COWL GEOMETRIES



duPont Aerospace Company, Inc.

ABSOLUTE SPILLAGE DRAG

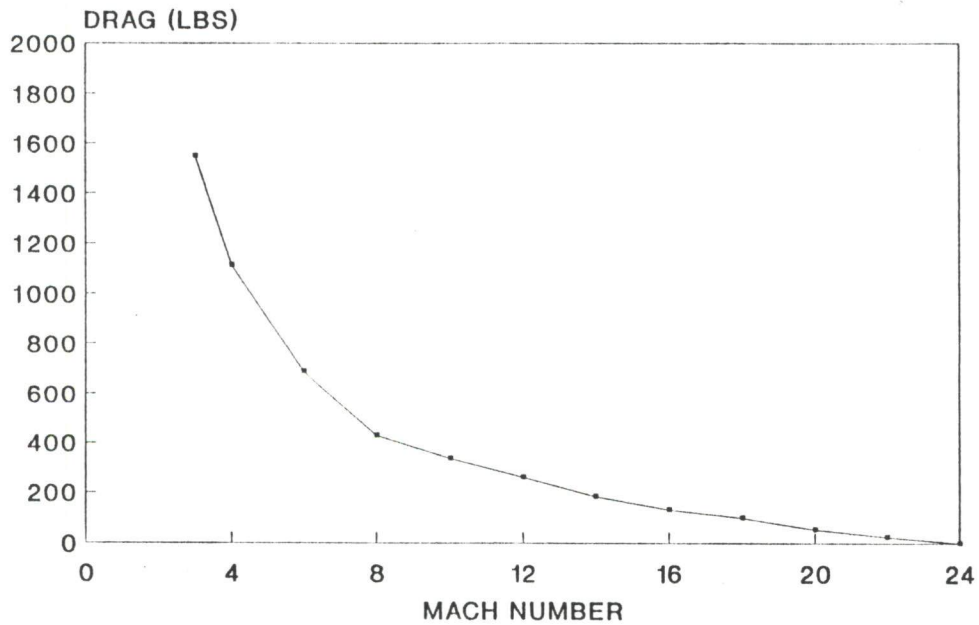
SHOCK WITH EXPANSION



duPont Aerospace Company, Inc.

GAGE SPILLAGE DRAG

SHOCK WITH EXPANSION



duPont Aerospace Company, Inc.

A HIGH PERFORMANCE X-30

PREPARED FOR
X-30 PLANNING

by

duPont Aerospace Company, Inc.

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July 11, 1990

HIGH PERFORMANCE X-30

With good SCRAMJET and low speed performance the X-30 can be a small aircraft in the 50,000 pound class with excellent performance potentially available in follow on aircraft.

The NASP SIZING CHART shows the relation between propellant fraction, empty weight fraction and takeoff gross weight for all possible X-30 aircraft. A nominal payload of 2500 pounds of flight test instrumentation was used to construct the chart. Two examples illustrate the use of the chart: with a propellant fraction of 62% and an empty weight fraction of 37% a 250,000 pound X-30 results; with a 52.5% propellant fraction and a 42.5% empty weight fraction the gross weight is reduced to 50,000 pounds. The logarithmic increase in gross weight as propellant fraction increases is evident from the chart.

The effect of using rocket propulsion before reaching orbital speed is shown on the AIRBREATHING TO ROCKET CHANGEOVER chart. The effect of using airbreathing propulsion is to decrease the propellant fraction. At relatively low SCRAMJET propulsive efficiency the effect of the changeover speed on mass fraction is small above Mach 12 to 16. This picture has led to designs that are heavily reliant on rocket propulsion above Mach 16 and to high gross weight X-30 designs. A higher propulsive efficiency shows a continual reduction in propellant fraction as the rocket takeover speed increases. This picture suggests airbreathing all the way to orbit and low X-30 gross weight.

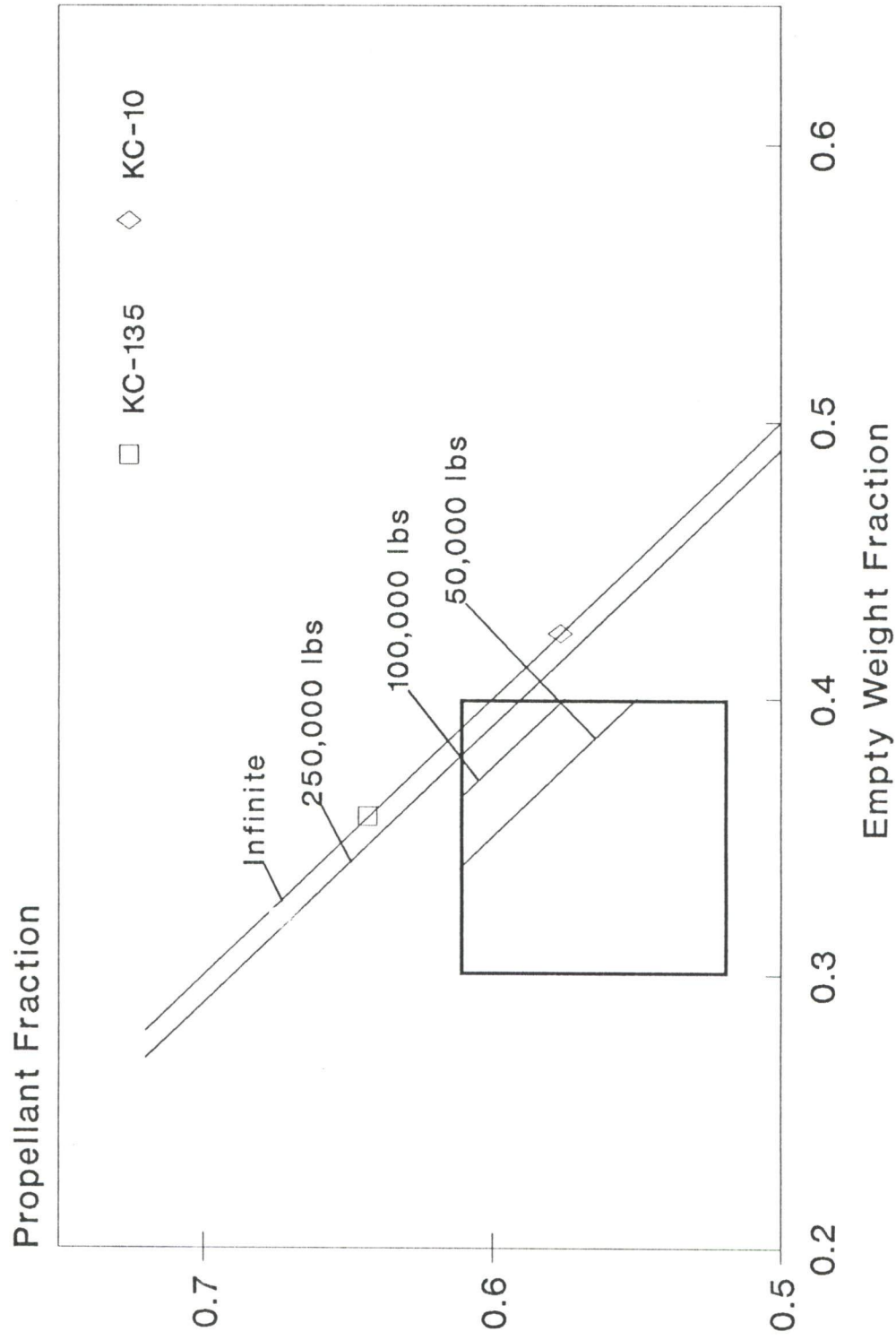
The PROPULSIVE EFFICIENCY attainable with a good SCRAMJET design is shown. With either an equilibrium or frozen nozzle expansion the propulsive efficiency in the critical high speed region is well above 40% suggesting the use of airbreathing propulsion all the way to orbit. The THRUST TO DRAG RATIO consistent with the propulsive efficiency is shown. A thrust to drag of 3.5 was used to prepare the rocket changeover chart, and this example shows somewhat higher values over most of the speed range making the propellant fractions a little bit less than shown in the chart.

Tables showing the performance of the engine on the updated Government Baseline airplane complete the presentation. Two high speed charts are shown one for frozen nozzle chemistry and one for equilibrium chemical composition during the nozzle expansion. The effective specific impulse includes the drag of the aircraft and is the value to be compared to the value attainable with rocket propulsion in a vacuum, 455 seconds. The original Government Baseline effective specific impulse is shown for reference.

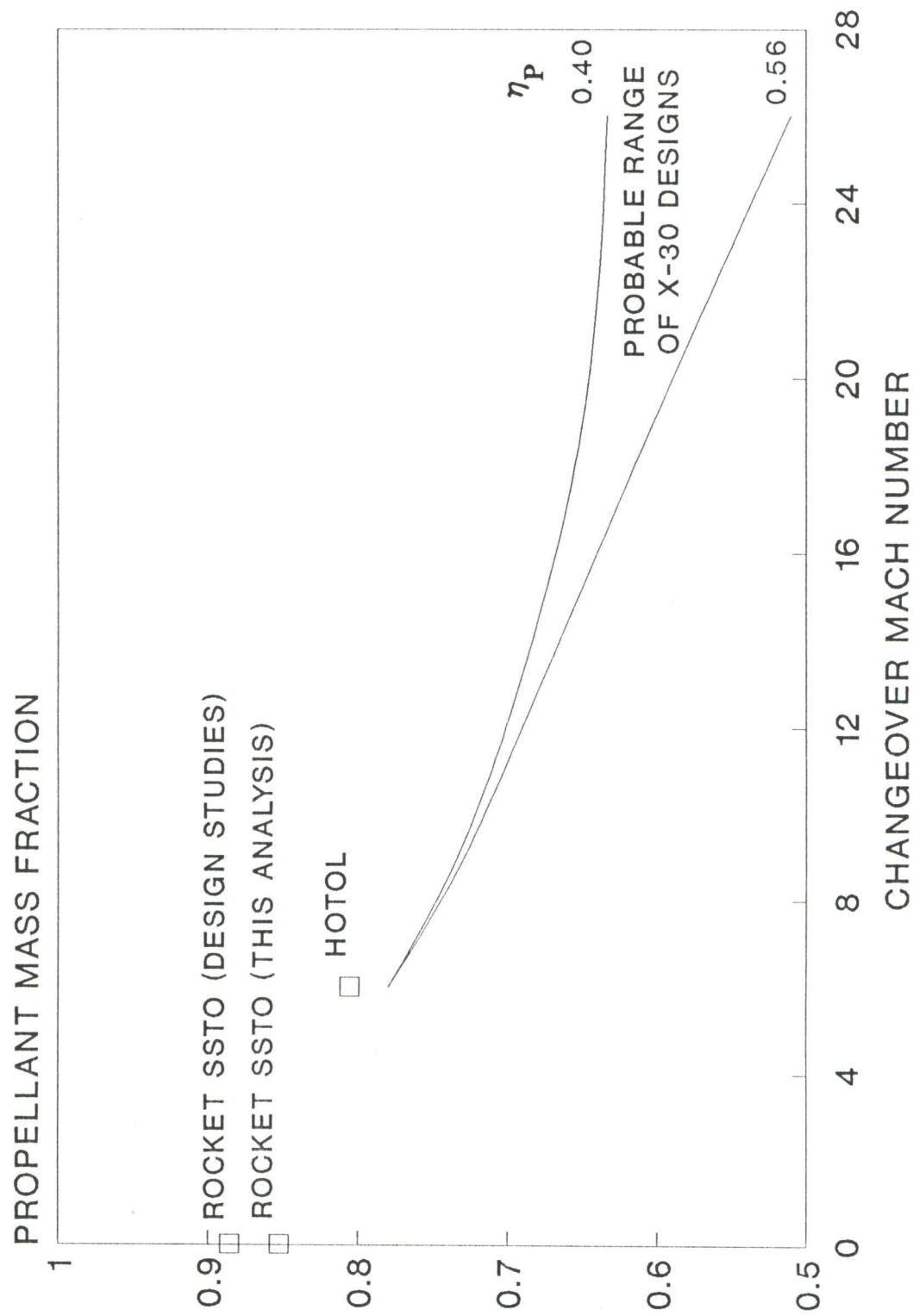
The Government Baseline is an airplane and engine design prepared by the duPont Aerospace Company under DARPA contract in 1983. It was given to the NASP contractors as a reference of what they were expected to achieve and, if possible, surpass. It is still the most efficient airplane in the NASP program.

NASP SIZING CHART

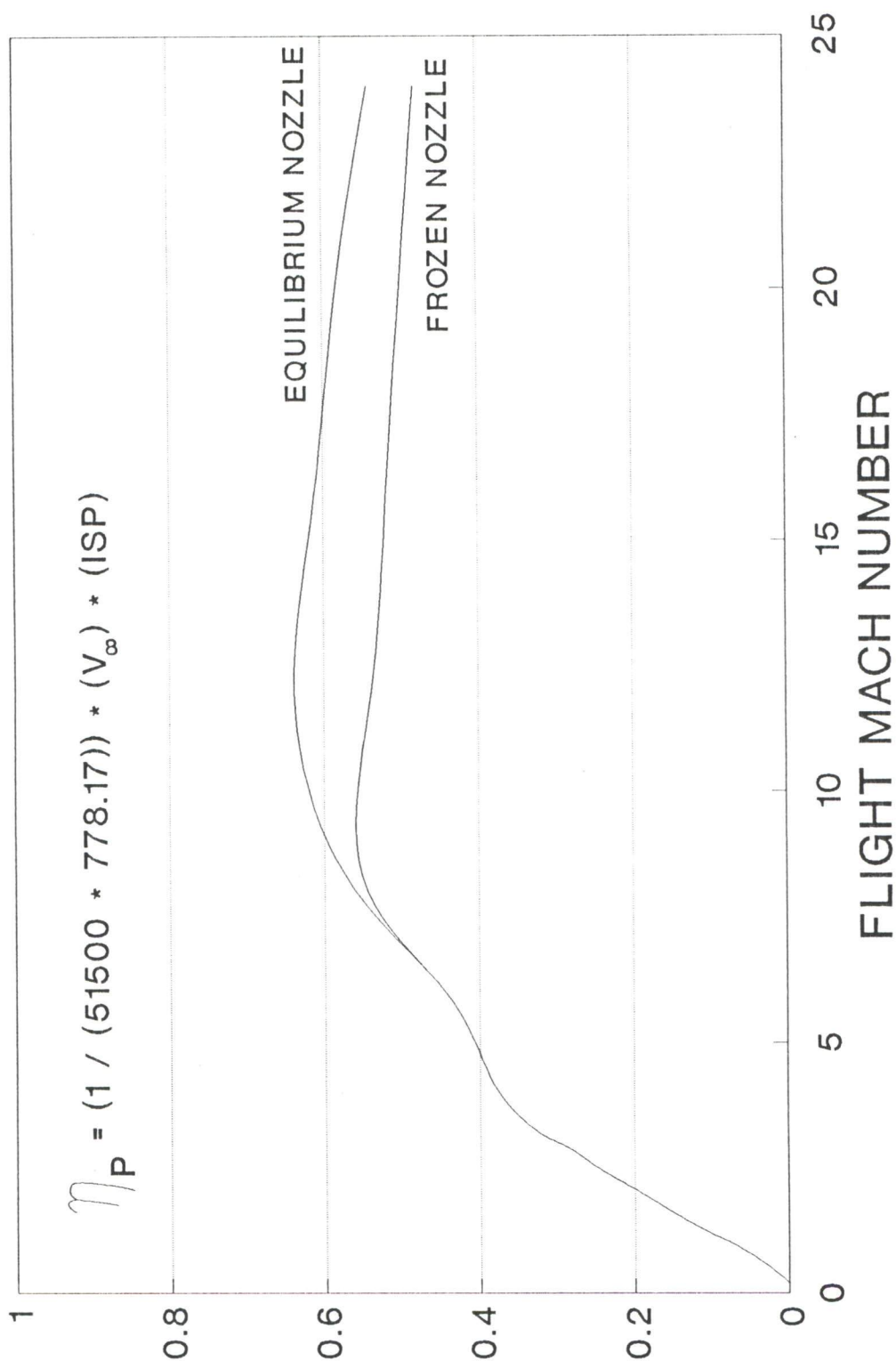
FLIGHT TEST INSTRUMENTATION ONLY



AIRBREATHING TO ROCKET CHANGEOVER

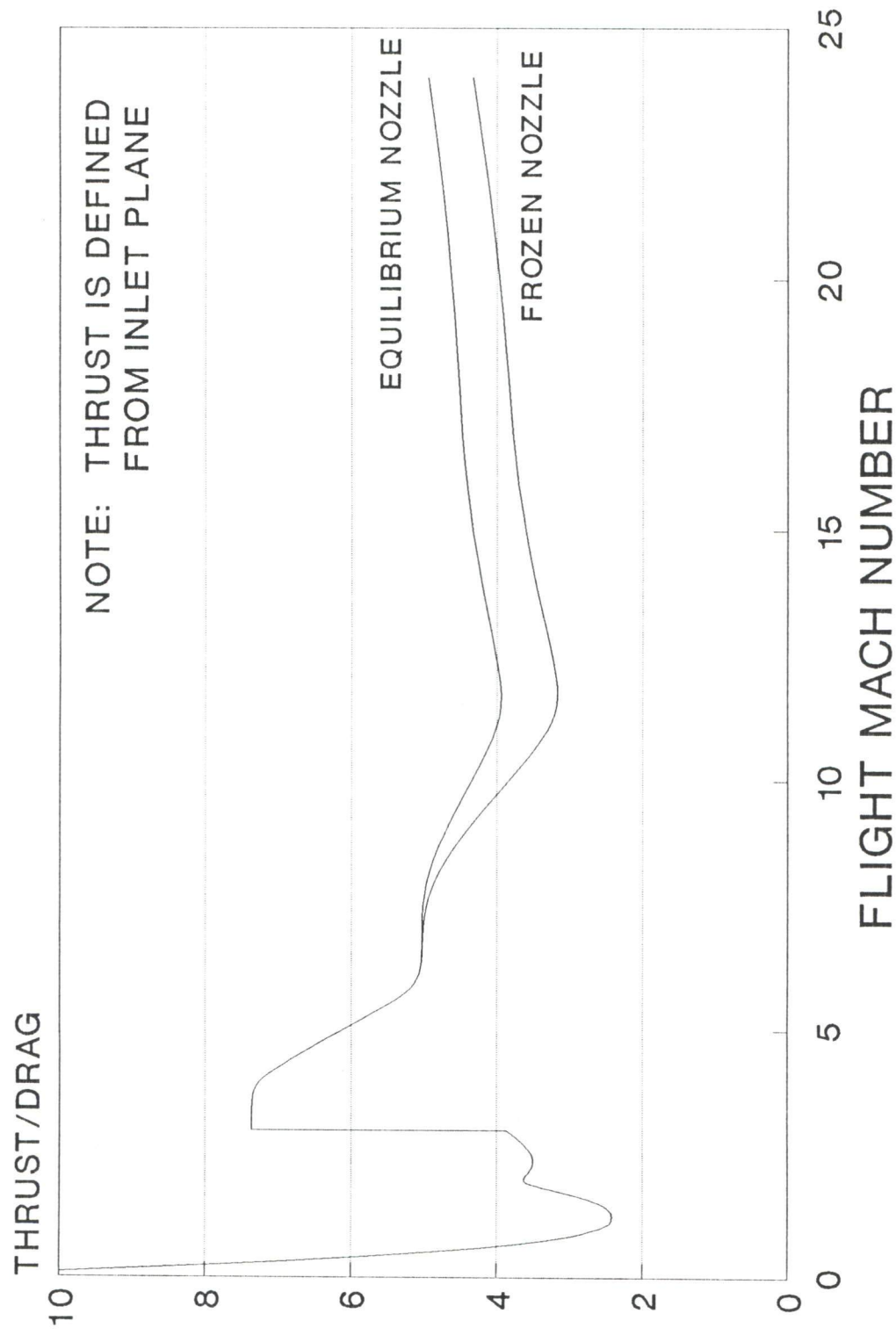


PROPULSIVE EFFICIENCY



duPont Aerospace Company, Inc.

THRUST TO DRAG RATIO



duPont Aerospace Company, Inc.

LOW SPEED ENGINE PERFORMANCE

FIXED COWL GEOMETRY (4 MODULES)

FLIGHT MACH NUMBER	0	0.5	1.0	1.1
CAPTURE AREA (SQFT)	INFINITE	15.35	11.58	11.65
DYNAMIC PRESSURE (PSF)	0	370	1,074	1,190
TOTAL MASS FLOW (LBM/SEC)	518	656	746	768
PARTIAL MASS FLOW (LBM/SEC)	126.5	127.1	114.6	124.4
THRUST (LBS)	43,392	44,664	59,546	65,398
SPECIFIC IMPULSE (SEC)	1,913	1,944	2,739	2,920
FUEL FLOW (LBM/SEC)	22.68	22.95	21.75	22.40
THRUST/AIR FLOW	83.72	68.03	79.87	85.16
EQUIVALENCE RATIO	1.5	1.2	1.0	1.0
DRAG (LBS)	0	8,618	23,919	23,397
DRAG/FUEL FLOW	0	375	1,100	1,045
EFFECTIVE ISP	1,913	1,569	1,639	1,875
EFFECTIVE ISP ORIGINAL GOVERNMENT BASELINE	1,800	1,839	2,159	1,248

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LOW SPEED ENGINE PERFORMANCE

FIXED COWL GEOMETRY (4 MODULES)

FLIGHT MACH NUMBER	1.5	2.0	2.5	3.0
CAPTURE AREA (SQFT)	13.54	18.74	19.33	20.60
DYNAMIC PRESSURE (PSF)	1,500	1,330	1,000	1,000
TOTAL MASS FLOW (LBM/SEC)	840	775	516	456
PARTIAL MASS FLOW (LBM/SEC)	129.7	117.9	75.6	65.5
THRUST (LBS)	84,921	86,594	59,779	52,200
SPECIFIC IMPULSE (SEC)	3,466	3,831	3,972	3,927
FUEL FLOW (LBM/SEC)	24.50	22.61	15.05	13.29
THRUST/AIR FLOW	101.13	111.75	115.87	114.56
EQUIVALENCE RATIO	1.0	1.0	1.0	1.0
DRAG (LBS)	33,166	21,049	12,248	9,509
DRAG/FUEL FLOW	1,354	931	814	715
EFFECTIVE ISP	2,112	2,900	3,158	3,212
EFFECTIVE ISP ORIGINAL GOVERNMENT BASELINE	1,689	2,557	3,097	3,249

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HIGH SPEED ENGINE PERFORMANCE FIXED COWL GEOMETRY (4 MODULES)

FLIGHT MACH NUMBER	3	4	6	8
CAPTURE AREA (SQFT)	43.3	45.1	49.0	52.6
DYNAMIC PRESSURE (PSF)	1,000	1,000	1,000	1,000
MASS FLOW (LBM/SEC)	960	750	543	434
THRUST (LBS)	117,558	87,480	44,355	37,279
SPECIFIC IMPULSE (SEC)	4,200	4,000	2,800	2,943
FUEL FLOW (LBM/SEC)	27.99	21.87	15.84	12.67
THRUST/AIR FLOW	122.47	116.64	81.65	85.81
EQUIVALENCE RATIO	1.0	1.0	1.0	1.0
DRAG (LBS)	12,248	9,509	7,498	6,064
DRAG/FUEL FLOW	438	435	473	479
EFFECTIVE ISP	3,762	3,565	2,327	2,464
EFFECTIVE ISP ORIGINAL GOVERNMENT BASELINE	3,498	3,597	2,306	2,480

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W. A. Fraser
10/20/80

HIGH SPEED ENGINE PERFORMANCE FIXED COWL GEOMETRY (4 MODULES)

FLIGHT MACH NUMBER	12	16	20	24
CAPTURE AREA (SQFT)	58.7	65.6	72.0	80.0
DYNAMIC PRESSURE (PSF)	1,000	1,000	1,000	500
MASS FLOW (LBM/SEC)	317	258	224	100
THRUST (LBS)	15,585	13,604	10,787	5,427
SPECIFIC IMPULSE (SEC)	1,686	1,206	918	716
FUEL FLOW (LBM/SEC)	9.25	11.28	11.75	7.58
THRUST/AIR FLOW	49.16	52.73	48.15	54.27
EQUIVALENCE RATIO	1.0	1.5	1.8	2.6
DRAG (LBS)	4,687	3,266	2,549	1,150
DRAG/FUEL FLOW	507	290	217	152
EFFECTIVE ISP	1,179	916	701	564
EFFECTIVE ISP ORIGINAL	1,455	1,034	898	663
GOVERNMENT BASELINE				

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EQUILIBRIUM HIGH SPEED ENGINE PERFORMANCE

FIXED COWL GEOMETRY (4 MODULES)

FLIGHT MACH NUMBER	12	16	20	24
CAPTURE AREA (SQFT)	58.7	65.6	72.0	80.0
DYNAMIC PRESSURE (PSF)	1,000	1,000	1,000	500
MASS FLOW (LBM/SEC)	317	258	224	100
THRUST (LBS)	19,619	15,894	12,655	6,200
SPECIFIC IMPULSE (SEC)	2,121	1,408	1,077	818
FUEL FLOW (LBM/SEC)	9.25	11.28	11.75	7.58
THRUST/AIR FLOW	61.89	61.60	56.49	62.00
EQUIVALENCE RATIO	1.0	1.5	1.8	2.6
DRAG (LBS)	4,687	3,266	2,549	1,150
DRAG/FUEL FLOW	507	290	217	152
EFFECTIVE ISP	1,614	1,118	860	666
EFFECTIVE ISP ORIGINAL GOVERNMENT BASELINE	1,455	1,034	898	663

duPont Aerospace Company, Inc.